CONVAIR F-102 MANUALS

CONVAIR	F-102A	Flight Manual, TO 1F-102A-1	11/29/57 revised 5/30/58
CONVAIR	F-102A	Flight Manual, TO 1F-102A-1 (no title page)	revised 7/26/57
CONVAIR	F-102A	Maintenance Training Supplement (Experimental) Airplane General (no title page)	7/1/56
CONVAIR	F-102A	Maintenance Training Supplement (Experimental) Electrical System (no title page)	7/1/56
CONVAIR	F-102A	Maintenance Training Supplement (Experimental) Flight Control System (no title page)	7/1/56
CONVAIR	F-102A	Maintenance Training Supplement (Experimental) High-Pressure Pneumatic System (no title page)	7/1/56
CONVAIR	F-102A	Maintenance Training Supplement (Experimental) Hydraulic System (no title page)	7/1/56
CONVAIR	F-102A	Maintenance Training Supplement (Experimental) Instruments (no title page)	7/1/56
CONVAIR	F-102A	Maintenance Training Supplement (Experimental) Low-Pressure Pneumatic System (no title page)	7/1/56
CONVAIR	F-102A	Maintenance Training Supplement (Experimental) Power Plant Installation (no title page)	7/1/56
CONVAIR	F-102A, TF-102A; F-106A/B	Flight Manual, TO 1F-102A-1 (no title page); Production Flight Test Procedures, Section 1; Inspection Testing for Production Model F-106A/B Aircraft without MA-1 System, Rept. No. ZM-8-518; Pratt & Whitney Aircraft Service School Notebook, J-75	revised 7/26/57; 8/1/57; 9/3/57; undated

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FLIGHT MANUAL

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~~PERFORMANCE APPENDIX~~

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Add 6-8

see A1-10

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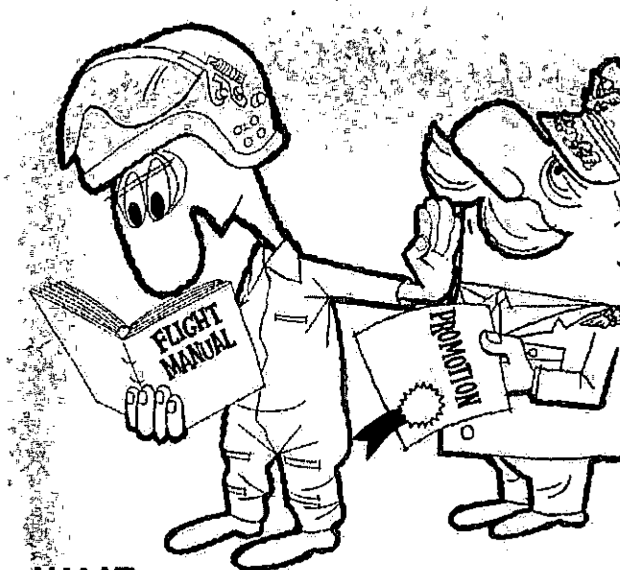

21802-1

WAIT...

THIS INFORMATION IS **IMPORTANT!!**

SCOPE

This manual contains all the information necessary for safe and efficient operation of the F-102A. These instructions do not teach basic flight principles, but are designed to provide you with a general knowledge of the aircraft, its flight characteristics, and specific normal and emergency operating procedures. Your flying experience is recognized, and elementary instructions have been avoided.

SOUND JUDGMENT

The instructions in this manual are designed to provide for the needs of a pilot inexperienced in the operation of this aircraft. This book provides the best possible operating instructions under most circumstances, but it is a poor substitute for sound judgment. Multiple emergencies, adverse weather, terrain, etc., may require modification of the procedures contained herein.

PERMISSIBLE OPERATIONS

The Flight Manual takes a "positive approach" and normally tells you only what you can do. Any unusual operation or configuration (such as asymmetrical loading) is prohibited unless specifically covered in the Flight Manual. Clearance must be obtained from SAAMA before any questionable operation is attempted which is not specifically covered in the Flight Manual.

STANDARDIZATION

Once you have learned to use one Flight Manual, you will know how to use them all — closely guarded standardization assures that the scope and arrangement of all Flight Manuals are identical.

ARRANGEMENT

The manual has been divided into ten fairly independent sections, each with its own table of contents. The objective of this subdivision is to make it easy both to read the book straight through when it is first received and thereafter to use it as a reference manual. The independence of these sections also makes it possible for the user to rearrange the book to satisfy his personal taste and requirements. The first three sections cover the minimum information required to safely get the aircraft into the air and back down again. Before flying any new aircraft these three sections must be read thoroughly and fully understood. Section IV covers all equipment not essential to flight but which permits the aircraft to perform special functions. Sections V and VI are obvious. Section VII covers lengthy discussions on any technique or theory of operation which may be applicable to the particular aircraft in question. The experienced pilot will probably be aware of the information in this section but he should check it for any possible new information. The contents of the remaining sections are fairly obvious.

YOUR RESPONSIBILITY

These Flight Manuals are constantly maintained current through an extremely active revision program. Frequent conferences with operating personnel and constant review of UR's, accident reports, flight test reports, etc., assure inclusion of the latest data in these manuals. In this regard, it is essential that you do your part! If you find anything you don't like about the book, let us know right away. We cannot correct an error whose existence is unknown to us.

PERSONAL COPIES, TABS AND BINDERS

In accordance with the provisions of AFR5-13, each pilot is entitled to have a personal copy of the Flight Manual. Flexible, loose leaf tabs and binders have been provided to hold your personal copy of the Flight Manual. These good-looking, simulated leather binders will make it much easier for you to revise your manual as well as to keep it in good shape. These tabs and binders are secured through your local materiel staff and contracting officers.

HOW TO GET COPIES

If you want to be sure of getting your manuals on time, order them before you need them. Early ordering will assure that enough copies are printed to cover your requirements. Technical Order 00-5-2 explains how to order Flight Manuals, classified supplements thereto, and Safety of Flight Supplements so that you automatically will get all original issues, changes, and revisions. Basically, all you have to do is order the required quantities in the Publication Requirements Table (T.O. 0-3-1). Talk to your Senior Materiel Staff Officer—it is his job to fulfill your Technical Order requests. Make sure to establish some system that will rapidly get the books and Safety of Flight Supplements to the pilots once they are received on the base.

SAFETY OF FLIGHT SUPPLEMENTS

Safety of Flight Supplements are used to get information to you in a hurry. Safety of Flight Supplements use the same number as your Flight Manual, except for the addition of a suffix letter. Supplements covering loss of life will get to you in 48 hours; those concerning serious damage to equipment will make it in 10 days. You can determine the status of Safety of Flight Supplements by referring to the Weekly Supplemental Index (T.O. 0-1-1A). This is the only way you can determine whether a supplement has been rescinded. The title page of the Flight Manual and title block of each Safety of Flight Supplement should also be checked to determine the effect that these publications may have on existing Safety of Flight Supplements. It is critically important that you remain constantly aware of the status of all supplements—you must comply with all existing supplements but there is no point in restricting the operation of your aircraft by complying with a supplement that has been replaced or rescinded. Technical Order 00-5-1 covers some additional information regarding these supplements.

CHECK LIST

The Flight Manual now contains only amplified check lists. The abbreviated check lists have been issued as separate cardboard technical orders. For the T. O. number and date of the latest check list applicable to this manual, see the back of the title page. Order your check list as you would any technical order. Line items in the Flight Manual and applicable cardboard check lists are identical as pertains to arrangement and item number. The cardboard check list is designed for use with binders having plastic envelopes into which the individual cards are placed. You will be advised via your command headquarters when the binders are available for distribution through normal Air Force supply channels.

WARNINGS, CAUTIONS, AND NOTES

For your information, the following definitions will apply to the "Warnings," "Cautions," and "Notes" found throughout the manual:

WARNING

Operating procedures, practices, etc., which will result in personal injury or loss of life if not carefully followed.

CAUTION

Operating procedures, practices, etc., which if not strictly observed, will result in damage to equipment.

Note

An operating procedure, condition, etc., which it is essential to emphasize.

MB-8 FLIGHT COMPUTER

The MB-8 Flight Computer for this aircraft is presently available. This computer is designed to provide pilots of single and twin jet engine aircraft with compact cruise control data which will aid in preparation of flight plans, inflight operation, and emergency inflight planning and operation. The computer is a five-disc, metal and plastic circular computer with a canvas carrying case. Three of the discs can be used with any aircraft and are referred to as "standard discs." The remaining discs contain data only for this aircraft and are described as "data discs." The standard discs and carrying cases are carried in Class 05-A and are available through normal supply channels. The data discs are distributed automatically to all bases having this aircraft. New or revised discs are issued each time the performance data in the Flight Manual is revised. The performance data in the computer and the manual is always kept current and consistent. If you have not yet received your computer, see your Base Operations Officer or T.O. 5F5-1-1. Reference should also be made to T.O. 5F5-1-1 for information on the operation of the computer.

COMMENTS AND QUESTIONS

Comments and questions regarding any phase of the Flight Manual program are invited and should be forwarded through your Command Headquarters to Commander SAAMA, Kelly Air Force Base, Texas. Attn: SANEF.

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THE AIRPLANE

The F-102A is a single-place, supersonic, all-weather interceptor built by CONVAIR, A Division of General Dynamics Corporation. The airplane is equipped with a radar fire control system and is powered by a J57-P-23 axial-flow turbojet engine with afterburner. The airplane is best characterized by the large area, 60-degree delta wing and the absence of a conventional empennage. Later airplanes* are equipped with a modified wing (Case XX wing) which produces greater lift and increases performance. The Case XX wing may be distinguished from the wing on earlier airplanes (Case X wing) by the droop at the wing-tip. The delta wing is equipped with "elevons" which provide combination aileron and elevator action from conventional cockpit controls. All control surfaces are hydraulically actuated and incorporate an artificial feel system. The airplane is equipped with a pressurized cockpit which contains an ejection seat. Conventional tricycle landing gear is utilized for takeoff and landing. The aft fuselage mounted speed brakes also serve as compartment doors for a drag chute. The six integral wing tanks are serviced by a single-point pressure refueling system and fuel usage is sequenced automatically to maintain desirable center of gravity.

DIMENSIONS

The overall dimensions of the airplane under normal conditions of gross weight, tire and gear strut inflation are as follows:

Wing span .... 38 feet, 1.6 inches

Length (including pitot boom) ... 68 feet, 1.8 inches

Height (to tip of vertical stabilizer—
AF 54-1390, -1398, -1401, 55-3357
and subsequent, and airplanes
modified by TCTO 1F-102A-538) 21 feet, 2.5 inches

**AF 55-3385, 56-1317 & on.*

Height (to tip of vertical stabilizer—
all other airplanes) .... 18 feet, 2.0 inches

Tread .... 14 feet, 2.25 inches

Refer to Section II for minimum turning radius and ground clearance.

GROSS WEIGHTS

Gross weights range from approximately 28,150 pounds to approximately 31,276 pounds, according to various mission loading conditions. The figures below are averages, and are provided to show average gross weights at different configurations:

	Clean	With External Tanks
Empty Weight*	19,903	20,234
Usable Fuel	7,053	9,848
Armament	1,194	1,194
Totals	28,150	31,276

Note

The figures above are averages and are not to be used in flight planning. For further information refer to the Handbook of Weight and Balance Data, T. O. 1-1B-40, and to Weight Limitations, Section V of this manual.

ENGINE

Thrust is supplied by a Pratt and Whitney J57-P-23 or J57-P-23A engine with afterburner (figure 1-5). These two engines are identical except for the type of afterburner igniter. Approximate standard sea level static thrust ratings for either engine is as follows:

MAXIMUM (with afterburning) ....16,000 pounds

MILITARY (without afterburning) ....10,200 pounds

The J57 engine is an axial-flow gas turbine, commonly known as a "two spool" engine because it has two rotors revolving on concentric shafts. Each rotor assembly includes a compressor and turbine. One distinctive characteristic of this type of engine is that the ratio of the turbine discharge pressure to compressor inlet pressure is a better indication of thrust than is rpm. One percent variation in rpm results in approximately five percent variation in thrust at the higher thrust settings; but one percent variation in turbine discharge pressure results in only approximately one and one-half percent variation in thrust. The nine-stage, low-pressure compressor is mounted on a solid drive shaft with a two-stage turbine. The seven-stage, high-pressure compressor is mounted on a hollow drive shaft, rotating around the first with a single-stage turbine. Each rotor set revolves independently of the other. A compressor air bleed system is used to direct part of the low-pressure, compressor air overboard at low engine rpm to aid in fast engine acceleration. The air bleed system is actuated automatically and is controlled by a governor driven by the low-pressure rotor. The engine combustion section includes eight interconnected combustion chambers (burners) arranged in an annular configuration. The main engine accessory section, driven by the high-pressure rotor, provides reduction gearing and mounting pads for all the engine-driven accessories.

Includes 713 pounds clean or 752 pounds with tanks for the weight of the pilot, survival equipment, trapped fuel and oil, etc.

ENGINE FUEL CONTROL SYSTEM

Fuel flow requirements are established by the pilot's throttle movement, and fuel flow to the engine is delivered and regulated by the engine fuel control system (figure 1-6). The system includes the engine-driven fuel pump unit, the fuel control unit, the fuel pressurizing and dump valve, and the afterburner fuel system. For details of the afterburner system refer to ENGINE AFTERBURNER SYSTEM, this Section.

Engine-Driven Fuel Pump Unit

The engine-driven fuel pump unit (figure 1-6) supplies the fuel pressure required by the engine and afterburner systems. The unit consists of three pumps. A centrifugal pump draws in fuel and forces it to two gear-type pumps. One gear-type pump is the engine stage fuel pump; the other is the afterburner stage fuel pump. The engine fuel pump furnishes fuel to the fuel control unit which regulates fuel flow to the engine combustion chambers. The afterburner fuel pump furnishes fuel to the afterburner metering valve which regulates fuel flow to the afterburner when afterburner operation is selected. When the afterburner is not operating the afterburner shutoff valve in the engine-driven fuel pump is closed and the output of the afterburner fuel pump is routed back to the discharge stream from the centrifugal pump. A warning light in the cockpit illuminates if the engine fuel pump fails, and a transfer valve in the engine-driven fuel pump unit automatically opens to allow fuel from the output side of the afterburner pump to flow to the fuel control unit. During this condition, fuel is supplied to both the engine and afterburner systems, if the throttle is in AFTERBURNER range. If the engine stage fuel pump fails during conditions of afterburner operation (i.e., takeoff), some additional thrust will be obtained if afterburner operation is continued. However, if the engine is in a non-afterburning condition at the time of engine stage fuel pump failure, afterburner operation should not be attempted. If the afterburner pump fails, the engine fuel pump cannot supply fuel to the afterburner.

Fuel Control Unit

The fuel control unit (figure 1-6) regulates fuel flow to the engine combustion chambers and incorporates normal and emergency fuel control systems. The normal fuel control system contains a mechanical computer, a governor, and temperature and pressure sensing elements which control the main metering valve. The computer, in addition to throttle position requirements, senses changes in flight conditions and regulates fuel flow to insure optimum engine operation. During rapid engine accelerations the normal fuel control system regulates fuel flow to prevent overspeed, overtemperature, compressor stalls and flameouts. The normal fuel control system also maintains a minimum fuel flow at high altitudes and during rapid decelerations to prevent engine flameout. Excess fuel not required by the engine is routed back to the engine-driven fuel pump unit by the main bypass valve. The emergency fuel control system provides an alternate system of regulating fuel flow to the combustion chambers in event of failure of components within the normal system. There are no provisions for automatic transfer to the emergency system in event of failure of the normal system. When the emergency fuel control system is energized, the normal system is rendered inoperative and fuel flow is controlled by the emergency throttle valve. This valve is connected directly to the throttle; therefore, emergency fuel flow is manually controlled. The emergency fuel control system will, however, compensate for altitude variations up to approximately 30,000 feet. (At higher altitudes the throttle must be successively retarded to maintain a constant rpm.)

CAUTION

> When operating on the emergency fuel control system, rapid throttle movements must be avoided to prevent overspeed, overtemperature, compressor stalls and flameouts since only the normal system contains these compensating features.

Excess fuel not required by the engine when operating on the emergency system is routed back to the engine fuel pump unit by the emergency bypass valve. When the throttle is placed in OFF, a mechanically controlled cutoff valve in the fuel control unit shuts off all fuel flow to the combustion chambers.

Fuel Control Takeoff Locks

As fuel flow control is not transferred automatically to the emergency system if the normal system fails, mechanical locks are included, in early airplanes, in the fuel control unit to maintain adequate fuel flow for takeoff and initial climb. These locks are controlled by placing the throttle to TAKEOFF position. To avoid feeding too much fuel at high altitude, it is necessary to retard the throttle lever below TAKEOFF position before reaching 7000 feet altitude.

Note

> Since normal thrust variations are not obtainable when the takeoff locks are engaged, TAKEOFF position of the throttle is not normally suitable for formation takeoff operations.

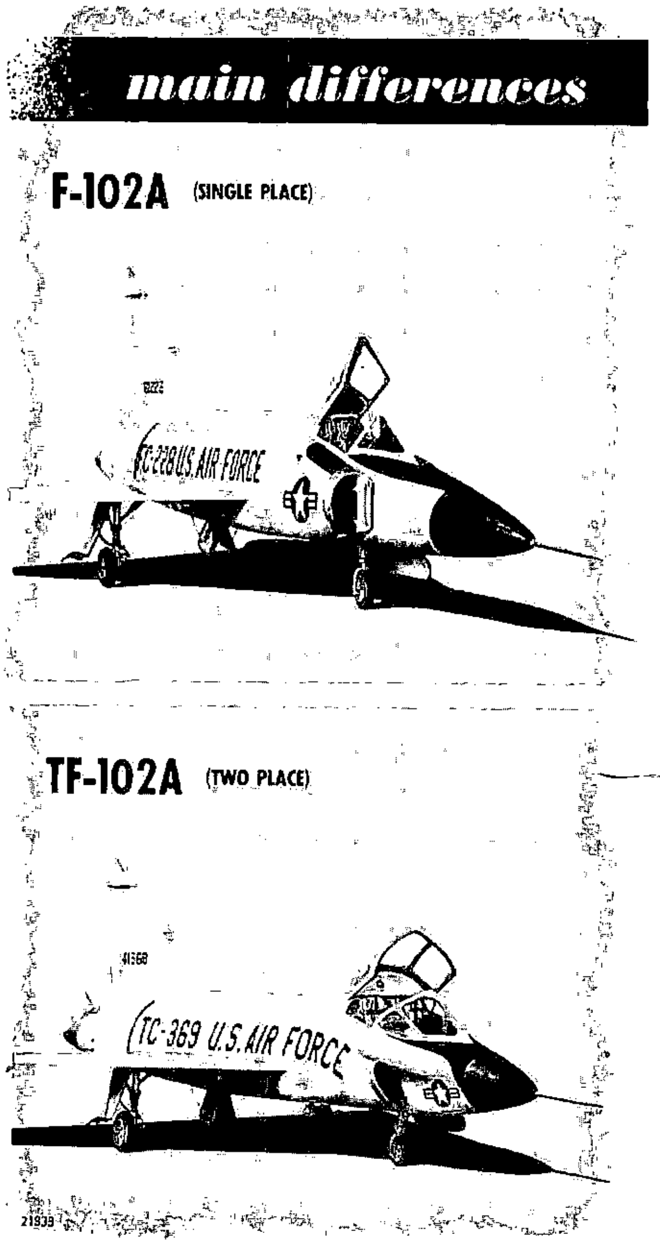


Figure 1-1

On later airplanes, the fuel control takeoff locks are removed to preclude the possibility of critical engine overspeed or temperature conditions when the throttle is inadvertently left in TAKEOFF while climbing above 7000 feet.

Note

> A placard is installed on the throttle quadrant on all airplanes which do not incorporate the takeoff locks.

Fuel Pressurizing and Dump Valve

The fuel pressurization and dump valve (figure 1-6) is located in the fuel control system between the fuel valve and the combustion chambers. The unit controls flow to the primary and secondary dual injector nozzles in the

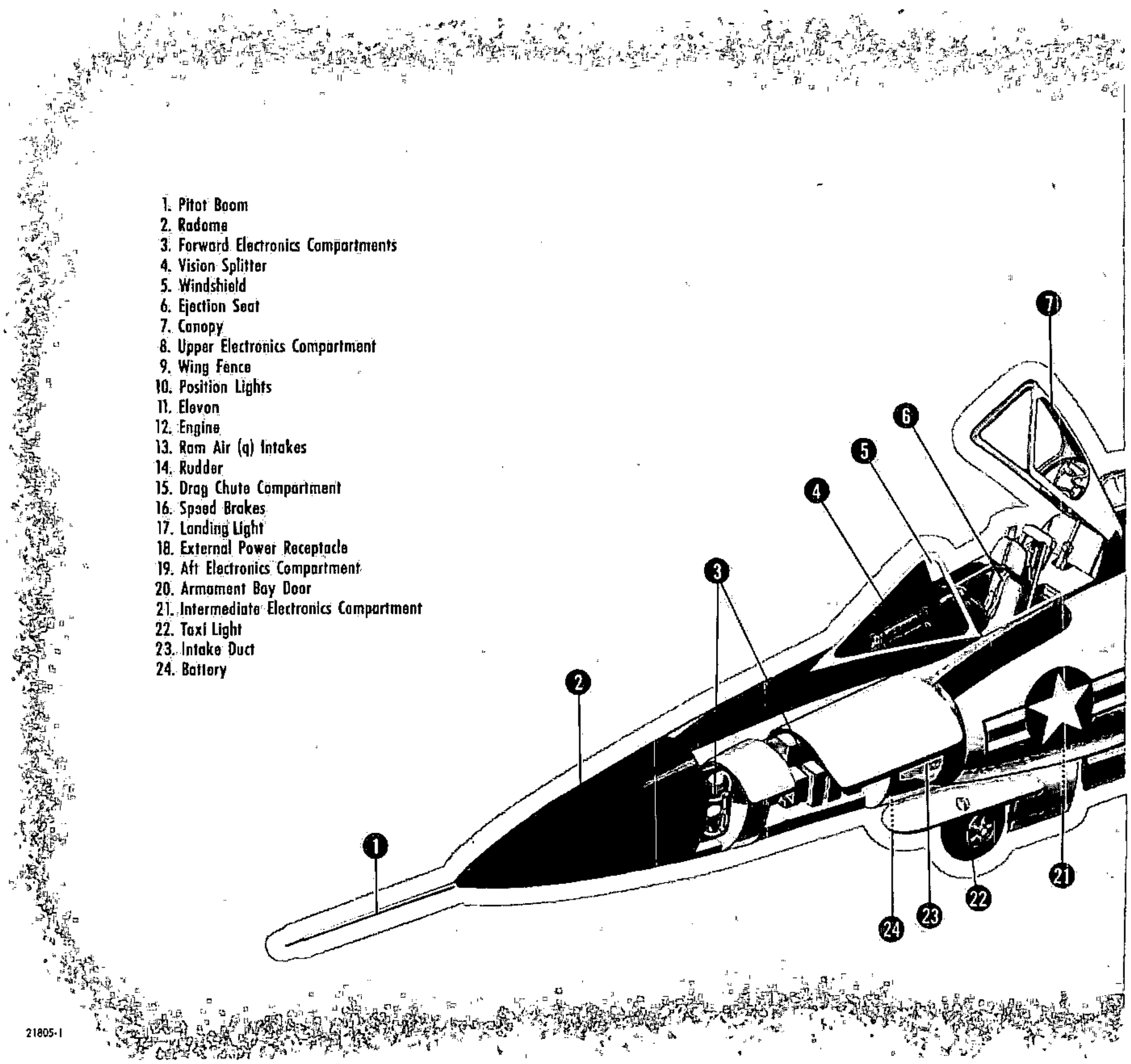


Figure 1-2

engine combustion chambers. To facilitate starting, fuel at relatively low pressure is directed through the primary manifold. Spring tension on the pressurizing valve keeps the port to the secondary manifold closed until increasing engine speed builds up fuel pressure high enough to overcome the spring tension and open the valve. When this happens, fuel flows through both primary and secondary manifolds. When the engine is stopped, the cutoff valve in the fuel control unit is closed by throttle movement, and the dump valve in the pressurizing and dump valve unit opens to permit residual fuel to drain overboard.

THROTTLE

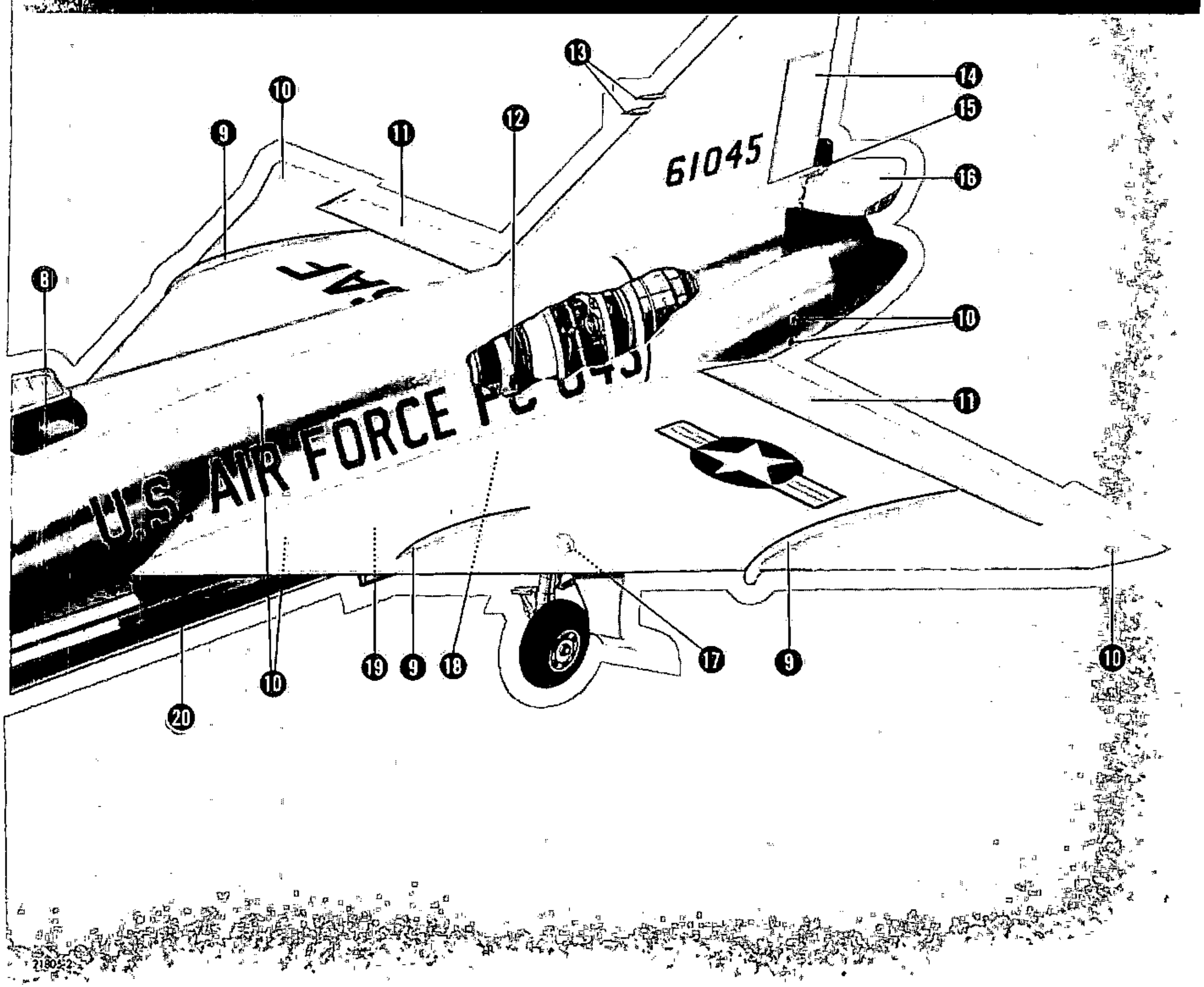
Engine thrust is controlled by the throttle (figure 1-7) which is located on the left-hand console. The throttle is mechanically linked to the fuel control unit and controls the normal and emergency fuel system and afterburner operation.

Note

The throttle is actually a thrust control lever as its function is to set up a thrust condition for which the fuel control meters fuel to the engine with automatic compensation (under normal fuel control) for engine rpm, ambient air conditions and compressor discharge pressure.

The throttle grip contains the speed brakes switch, ignition button, microphone button and takeoff lock trigger (some airplanes). An automatic anti-creep spring assembly is installed in the throttle quadrant to prevent throttle

airplane general arrangement



creep in fore and aft directions, thus eliminating the need for a manual friction lock. Positioning the throttle outboard from OFF to START arms the starter. When the ignition button is depressed (with throttle in START position) the combustion starter is actuated by air and airmotors the engine to approximately six percent rpm. With the ignition button depressed, moving the throttle inboard to OFF energizes the starter and ignition circuit. At the forward IDLE stop, the throttle must be moved inboard to clear the stop. This places the throttle in the IDLE position of the normal range. The throttle may be advanced to the FULL MIL POWER position and after depressing the takeoff lock trigger, to the TAKEOFF position which engages the takeoff locks in the fuel control unit. On airplanes not equipped with fuel control takeoff locks, the throttle should also be advanced to TAKEOFF position after depressing the takeoff lock trigger. Placing the throttle outboard of the mechanical lock prevents inadvertently moving the throttle out of AFTERBURNER when retracting the landing gear. To disengage the mechanical locks, the throttle is moved aft of TAKEOFF position without depressing the trigger as the FULL MIL POWER stop offers no resistance when retarding the throttle. On other airplanes which are not equipped with a takeoff lock trigger, there is no TAKEOFF position on the quadrant. On these airplanes, placing the throttle in FULL MIL POWER and moving it outboard will lock the throttle outboard in AFTERBURNER range until the throttle is retarded 5° aft of the stop. At any point along the normal thrust range forward of the afterburner rear stop, the throttle may be moved outboard to the AFTERBURNER range. A microswitch in the throttle quadrant fires the afterburner when the throttle is moved into this range. A detent mechanism in the quadrant separates the two arcs of travel, so that the throttle must be forced from one side of its slot to the other, and cannot accidentally slip out of the AFTERBURNER range. At maximum power setting, the afterburner may be shut off by retarding the throttle slightly and moving it inboard, without disengaging the fuel control mechanical locks. Throttle position can also cause the landing gear warning light to illuminate. Refer to LANDING GEAR WARNING LIGHT AND TEST BUTTON, this Section. Retarding the throttle to IDLE on early airplanes also activates a jet pump to provide for electronics compartment ground cooling. On later airplanes,* retarding the throttle to 72% rpm or less activates the jet pump. (Refer to Section IV for electronic cooling.)

TAKEOFF LOCK TRIGGER

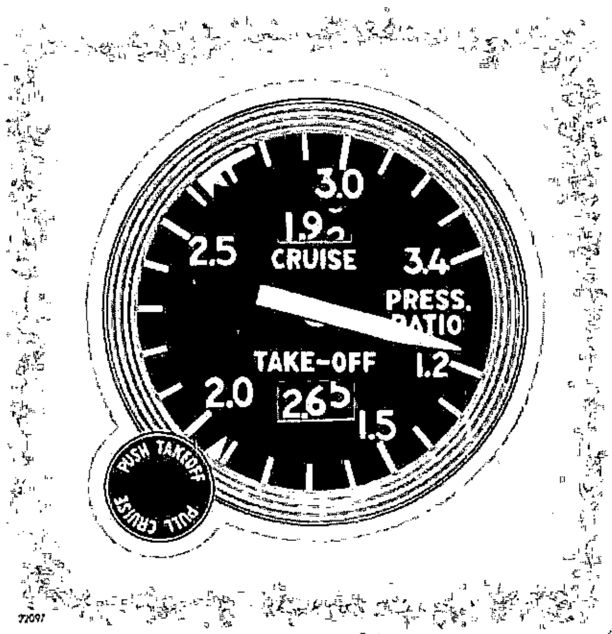
A two-position takeoff lock trigger (some airplanes) is located on the throttle quadrant (figure 1-7). The trigger is spring-loaded in the up position and prevents inadvertently moving the throttle into the TAKEOFF thrust range. When depressed, the trigger mechanically lowers a latch that allows the throttle to be advanced to the TAKEOFF thrust range. This position, besides engaging the takeoff locks in the fuel control unit on some airplanes, places the throttle outboard of a mechanical lock that prevents inadvertently moving the throttle out of AFTERBURNER when retracting the landing gear. In retarding the throttle, it is not necessary to depress the takeoff lock trigger, as the FULL MIL POWER stop offers no resistance. The two positions of the takeoff lock trigger are not placarded.

FUEL CONTROL SWITCH

The guarded fuel control switch (figure 1-7) is located on the throttle quadrant, to the left of the throttle, and is used to select either normal or emergency fuel control system. The switch is placarded "Fuel Controls" and has two positions, NORMAL and EMERGENCY. When in NORMAL position the emergency shuttle valve in the fuel control unit is electrically positioned to permit normal fuel control system operation. When in EMERGENCY position, the emergency shuttle valve is electrically positioned to permit emergency fuel control operation. A warning light illuminates when the emergency fuel control switch is in EMERGENCY position. Power is taken from the 28-volt dc essential bus.

ENGINE PRESSURE RATIO GAGE

The engine pressure ratio gage (28, figure 1-4; figure 1-8) is used during preflight engine checks and for establishing inflight cruise thrust settings. During the pre-



flight engine checks the gage is used to determine whether the engine thrust on the ground at full throttle is acceptable for takeoff. This gage is located on the instrument panel and compares pitot pressure (from pitot-static system) and engine turbine discharge pressure. This differential pressure is indicated in pressure ratio ranging in increments from 1.2 to 3.4. Maximum and minimum takeoff thrust limits are obtained from the Takeoff Check Chart in the Appendix. These limits vary with ambient temperature. The pressure ratio gage incorporates two adjustable reference marks which are set by a knob on the lower left side of the gage. The small triangular reference mark references the setting which appears in the cruise window, located in the upper portion of the instrument. Cruise thrust settings are obtained from the cruise charts in the Appendix. The extreme ends of the elongated reference mark reference the minimum and maximum takeoff limits and the notched portion of the reference mark represents the trim limits which are utilized for stabilized engine operation. When using the pressure ratio gage to check thrust prior to takeoff, a thrust overshoot may be noted when the throttle is advanced to FULL MIL POWER from IDLE on a cold engine. This thrust overshoot will gradually diminish to the specified value within approximately five minutes. This condition is considered to be normal. Power to the engine pressure ratio gage is supplied from the 200/115-volt, 400-cycle, ac essential bus.

TACHOMETER

The tachometer (15, figure 1-4; figure 1-8) indicates percentages of high-pressure rotor rpm, based on the figure 9976 rpm as 100%. The rpm at which full military thrust is obtained varies for each engine. Therefore, the rpm percentage indicated by the tachometer is not necessarily a direct indication of percentage of military thrust

*AF 55-3371 & on.

cockpit general arrangement (typical)

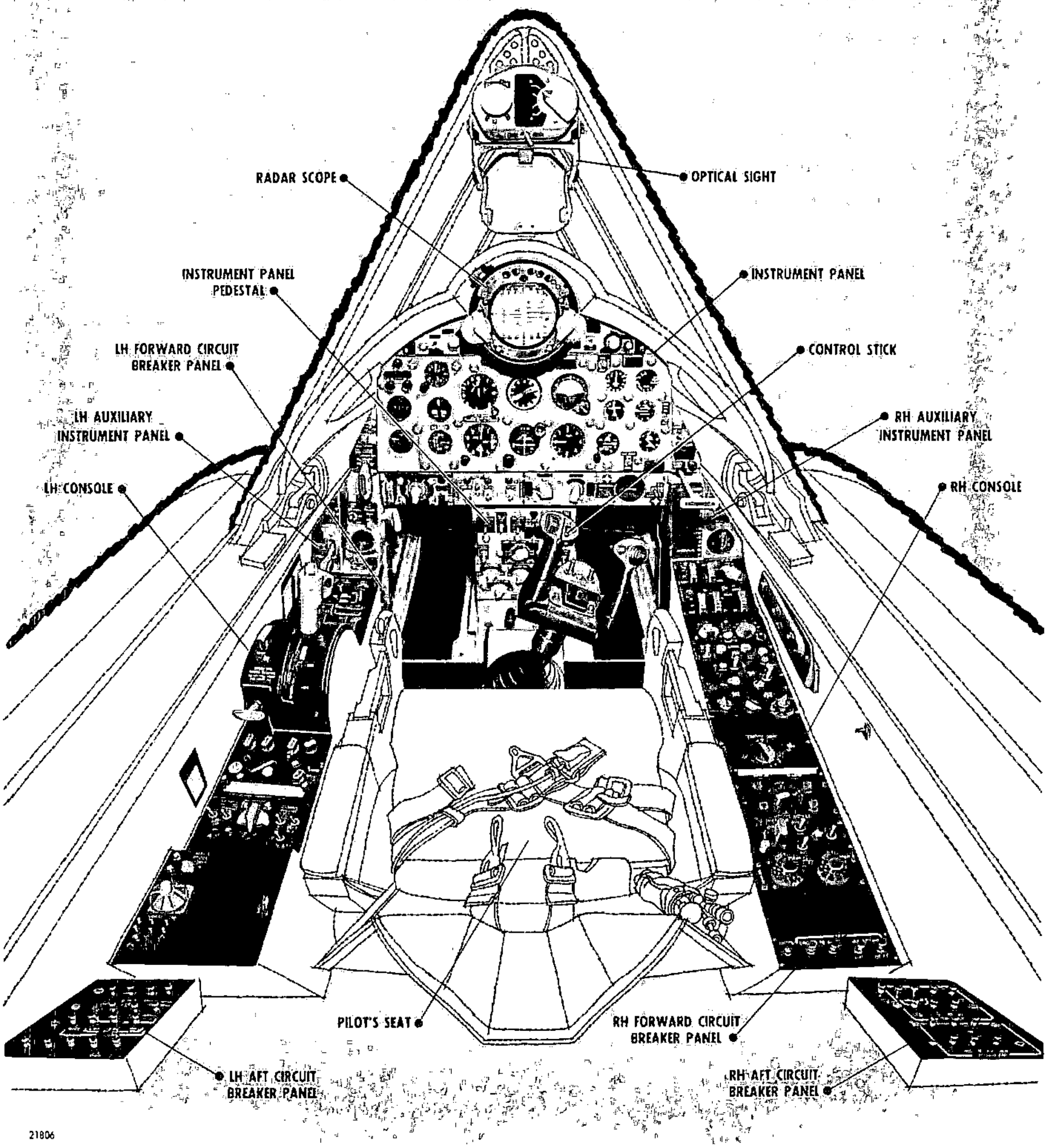


Figure 1-3

rpm. The military power rpm for each engine is placarded on the engine data plate. The tachometer is a synchronous motor which responds to the speed of an engine-driven generator and therefore operates independently of the airplane electrical system. The main pointer is calibrated up to 100% rpm and the sub-pointer makes one complete revolution for each 10% change in engine rpm. By using the sub-pointer, up to 110% rpm can be read.

EXHAUST GAS TEMPERATURE GAGE

The exhaust gas temperature gage (17, figure 1-4; figure 1-8), located on the instrument panel, is a remote indicating instrument which registers in degrees centigrade the turbine discharge temperature measured by thermocouples located in the tailpipe aft of the last stage of the turbine. On some airplanes the face of the indicator is calibrated to indicate from 0° to 1000°C (32° to 1832°F) in increments of 100°C up to 500°C, in 20°C from 500° to 800°C, and in 100°C from 800° to 1000°C. On other airplanes, the indicator is calibrated in increments of 20°C from 0° to 1000°C. The indicator is actuated by voltages produced by the thermocouples, and therefore the system is independent of the airplane electrical system.

FUEL FLOW INDICATOR

The fuel flow indicator (18, figure 1-4; figure 1-8) registers the rate of flow (consumption) of fuel to the engine in pounds per hour. Indications shown are measured by a transmitter located in the fuel line downstream from the fuel control unit (figure 1-6). All fuel entering the engine passes through the transmitter. The indicator is calibrated to show the rate of flow from 0 to 12,000 pounds per hour. The indicator dial is graduated in 100-pound increments up to 3000 pounds and in 1000-pound increments from 3000 to 12,000 pounds. The fuel flow indicator does not indicate fuel flow to the afterburner system. The indicator power is supplied from the 26-volt, 400-cycle ac essential bus.

OIL PRESSURE-LOW WARNING LIGHT

The oil pressure-low warning light (5, figure 1-26), located on the warning light panel, illuminates and displays "OIL PRESS" when the engine oil pressure falls below 36 (±2) psi and goes out when the pressure rises above 40 psi. The light is controlled by a pressure switch in the oil line. The oil pressure-low warning system receives power from the 28-volt dc essential bus.

ENGINE FUEL PUMP FAILURE WARNING LIGHT

The engine fuel pump failure warning light (8, figure 1-26), located on the warning light panel, illuminates and displays "ENG FUEL PUMP" if the engine stage of the engine-driven fuel pump unit fails. Power is supplied from the 28-volt dc essential bus.

EMERGENCY FUEL CONTROL WARNING LIGHT

The emergency fuel control warning light (4, figure 1-26), located in the warning light panel, illuminates and displays "EMER FUEL ON" when the fuel control switch is moved from NORMAL to EMERGENCY position. The light receives power from the 28-volt dc essential bus.

ENGINE STARTER AND IGNITION SYSTEM

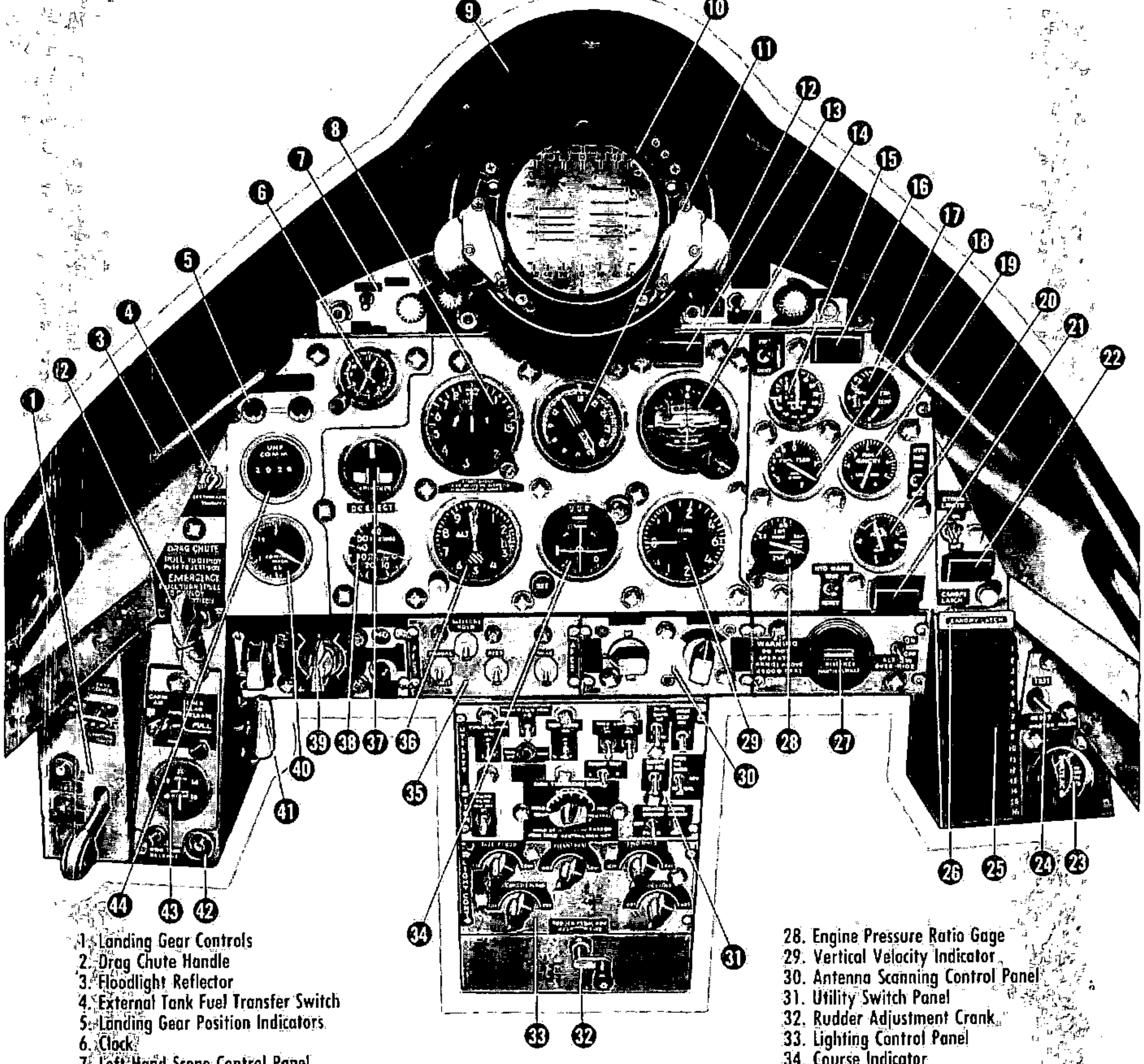
The combustion starter is a self contained unit consisting of a small gas turbine, a drive assembly, and electrical control valves. Air under 3000 psi pressure is supplied from either a ground source or the airplane's high-pressure pneumatic system. When the starter is supplied with air pressure from a ground source, a high-pressure air hose is attached to the pneumatic system filler valve on the forward side of the left main wheel well. The pressure is reduced to 300 psi by an air control valve, through which air flows to the starter combustion chamber and the starter fuel flask. Fuel supply is tapped from the engine fuel supply line, at a point downstream from the left-hand fuel boost pumps. The electrically operated fuel and air valves and the starter ignition system are controlled by switches in the throttle quadrant, and by the engine ignition button on the throttle lever grip. Automatic controls within the starter unit prevent overspeeding of the unit and shut off starter fuel and air valves if combustion stops, or when the speed of the engine increases to approximately 35% engine rpm. Power for all electrical controls is taken from the 28-volt dc essential bus.

Note

> The air supply, under normal conditions, for the combustion starter will be from a ground cart. In the event that a ground cart is not available, starting air for one start may be taken from the airplane high-pressure pneumatic system supply flasks by placing the manual shutoff valve in the left main landing gear wheel well to the open position placarded AIRCRAFT. The number of starts available will depend upon the mission requirements. For a tactical armament firing mission, air will normally be available for one start. For a mission not requiring armament firing, air will be available for two starts with each start using about 900 psi air pressure.

Ignition is accomplished by the use of high tension transformers, two igniter plugs located in combustion chambers No. 4 and 5, and the ignition button which is electrically connected with the starter control circuit. The engine ignition circuit is energized only during engine starting as combustion is continuous once the engine starts. The afterburner is ignited by "hot streak" ignition and requires no electrical ignition for operation. Since the ignition and starter circuits are electrically

instrument panel (typical)



1. Landing Gear Controls
2. Drag Chute Handle
3. Floodlight Reflector
4. External Tank Fuel Transfer Switch
5. Landing Gear Position Indicators
6. Clock
7. Left-Hand Scope Control Panel
8. Machmeter—Airspeed Indicator
9. Glare Shield
10. Radar Scope
11. Radio Magnetic Indicator
12. Right-Hand Scope Control Panel
13. Engine Fire and Overheat Warning Light
14. Attitude Indicator
15. Tachometer
16. Master Warning Light
17. Exhaust Gas Temperature Gage
18. Fuel Flow Indicator
19. Fuel Quantity Gage
20. Hydraulic Pressure Gage
21. Hydraulic Pressure Low Warning Light
22. Canopy Unlocked Warning Light
23. AC Voltmeter
24. Warning Light Panel Test and Reset Switch
25. Warning Light Panel
26. Canopy Latch Handle
27. Tacan Range Indicator Panel
28. Engine Pressure Ratio Gage
29. Vertical Velocity Indicator
30. Antenna Scanning Control Panel
31. Utility Switch Panel
32. Rudder Adjustment Crank
33. Lighting Control Panel
34. Course Indicator
35. AFCS Control Panel
36. Altimeter
37. Turn-and-Slip Indicator
38. Target Altitude Indicator (1)
39. Armament Control Panel
40. Command Mach Indicator (1)
41. Landing Gear Emergency Extension Handle
42. Wing Tank Release Button
43. Cabin Pressure Altitude Gage
44. Remote Indicator (UHF)

(1) SOME AIRPLANES—SEE APPLICABLE TEXT

22128

Figure 1-4

interconnected, the throttle must be moved outboard to START in order to arm the ignition circuit. The ignition button must be depressed to the START position and held depressed while the throttle is moved inboard to energize the igniters. On some airplanes, the same action is required for an air start, although the starter circuits are bypassed. Other airplanes* incorporate an all-points ignition system that permits energizing the igniters from any throttle position when the airplane is airborne.

Ignition Button

The ignition button (figure 1-7) is used to energize the igniters during starting and is located on the throttle. During ground starts, the throttle must be used in conjunction with the ignition button to arm the ignition circuit and control the starter (refer to THROTTLE, this Section). The ignition button serves the same basic function when making an air start; however, on airplanes with all-points ignition*, it is not necessary to arm the ignition circuit by placing the throttle to START. With the all-points system, the ignition circuit is armed and the button is "hot" whenever the weight of the airplane is off the left main landing gear. The ignition button receives power from the 28-volt dc essential bus.

Engine Ignition Disconnect Switch

The guarded engine ignition disconnect switch is located in the left main wheel well. The switch is used to interrupt power between the ignition power circuit breaker and the engine igniters. The engine can then be "motored" without ignition in the combustion chambers. The two-position switch is placarded "Eng Ign Arm" with ON and OFF positions. The switch is ON during flight. The circuit receives power from the 28-volt dc essential bus.

ENGINE COOLING

In flight, cooling air is taken from the engine air intake ducts to cool the engine and accessories and the electronic bays. On the ground, that portion of the engine within the engine shroud is cooled by bleed air ducted from the low-pressure compressor section of the engine. The bleed air valve is controlled by the down-and-locked switch on the main landing gear. When the main landing gear is down and locked, the cooling air valve opens. When the landing gear is unlocked for retraction, the switch recloses the valve. During ground operation, and at speeds up to 150 knots, the flow in the outer annulus between the shroud and fuselage skin is reversed due to the existence of a vacuum, or low pressure area within the engine air inlet duct. Air enters the fuselage at the tail cone and flows forward over the engine shroud, through the engine accessory section, then flows into the engine air inlet duct via the air-oil cooler duct and the intake duct scroll. Reverse airflow also occurs at the aft electronic bay cooling air duct, whereby air is drawn in from the landing gear wheel well and exits into the engine air intake duct. The cooling air for reverse flow of the ac and dc generator ducts is drawn from the outside of the fuselage. This air is controlled by a differential pressure flapper valve. Additional cooling, especially of the engine bearings and adjacent parts, is accomplished by forced circulation of the engine oil, which absorbs heat from engine bearings and conducts it to heat exchangers (fuel and air-oil coolers).

ENGINE AFTERBURNER SYSTEM

The engine is equipped with an afterburner which augments engine thrust to obtain maximum thrust (approximately 50% additional thrust at sea level). Engine operation with maximum thrust results in a high fuel consumption rate; therefore, afterburner operations should be used only when this additional thrust is required. Operation of the afterburner is controlled by the throttle (see figure 1-6). When the throttle is placed outboard to AFTERBURNER, the afterburner shutoff valve (in the engine-driven fuel pump unit) is opened by power from the 28-volt dc essential bus. Fuel is routed to the afterburner metering valve which regulates afterburner fuel in relation to compressor discharge pressure (which is governed by flight conditions and engine speed). Metered fuel is then supplied to the afterburner igniter and afterburner fuel spray bars. At the same time, unmetered afterburner fuel actuates the exhaust nozzle control to pneumatically open the nozzle during afterburner operation. When engine thrust is varied, afterburner thrust will also vary. Thrust variation will range from maximum thrust available to approximately 50% afterburning thrust when the throttle is retarded in the AFTERBURNER range.

Note

The afterburner can be used while the engine is operating on the emergency fuel control system if the main system fails. However, rapid throttle movement should be avoided to prevent overspeed, overtemperature, compressor stalls and flameouts.

AFTERBURNER EXHAUST NOZZLE

The two-position exhaust nozzle at the end of the tail pipe is opened or closed to provide the proper exhaust area for either normal or afterburner engine operation. The nozzle flaps are moved to the full open position during afterburner operation and returned to the minimum nozzle opening area when the afterburner is not in use. Positioning of the nozzle flaps is accomplished automatically by means of the exhaust nozzle control unit. No emergency override control is included.

AFTERBURNER IGNITER

When the afterburner system is actuated, fuel from the afterburner metering valve is directed to the igniter unit. (See figure 1-6.) This metered fuel flow actuates the

*Airplanes modified by TCTO 1F-102-746.

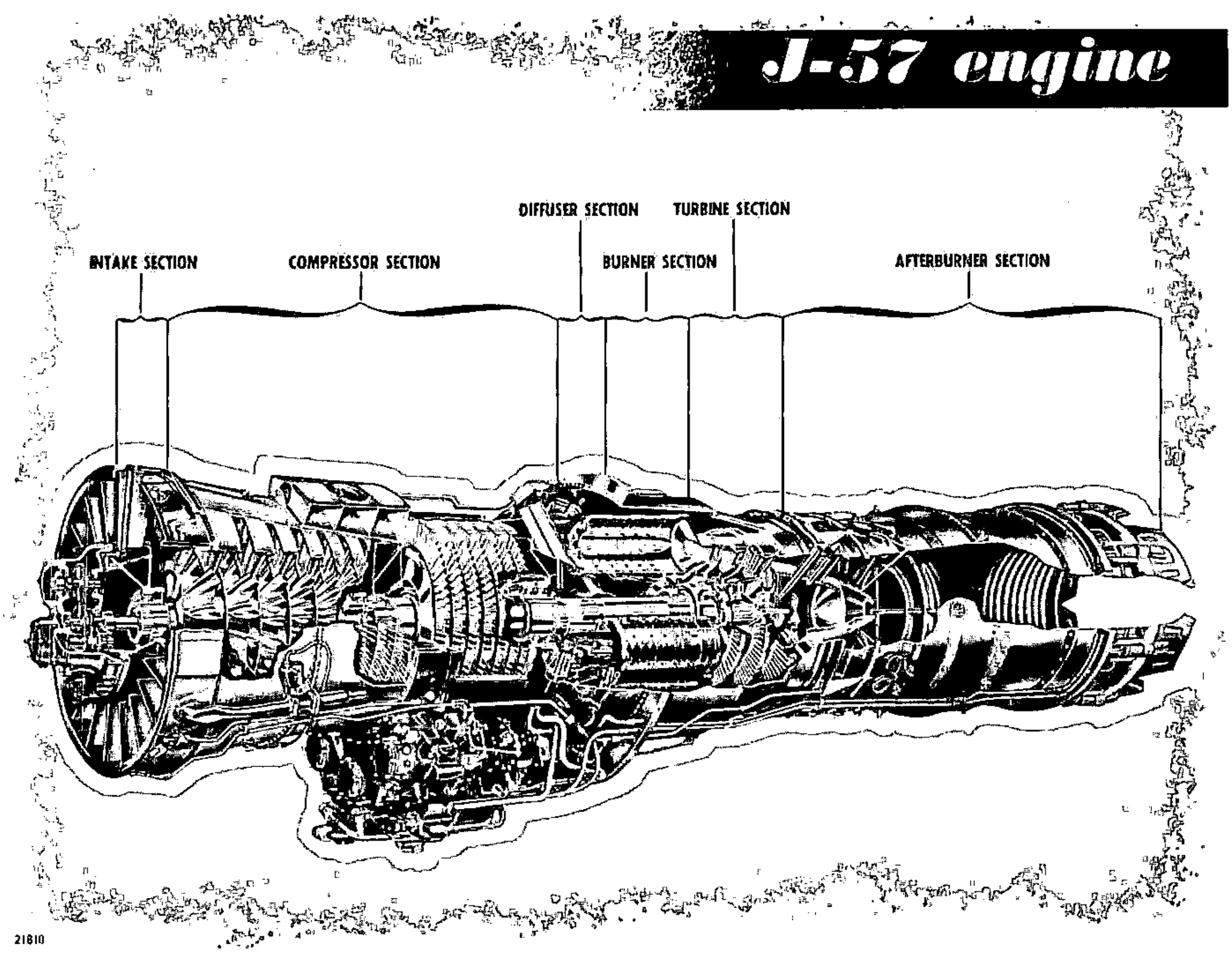


Figure 1-5

igniter unit, which momentarily injects a small amount of fuel into one of the engine combustion chambers, thereby creating a local excessively rich fuel-air mixture. The excess fuel forms a longer flame front that continues to burn past the turbines. The extended flame provides "hot streak" ignition to ignite the fuel being discharged from the afterburner fuel spray bars. The igniter is actuated only when full pressure is built up within the afterburner manifold, so that fuel is available at the spray bars when the igniter introduces fuel to the burner for ignition. If afterburner light-up is not accomplished within two seconds at sea level (five seconds at altitude) after the throttle is moved outboard to the AFTERBURNER range, the throttle should again be moved inboard, and then after two to five seconds it should be moved to AFTERBURNER range to recycle the igniter. An afterburner recirculating igniter is installed that improves internal cooling, thereby preventing "coking" and increasing reliability of the system.

AFTERBURNER EXHAUST NOZZLE CONTROL UNIT

Opening and closing the afterburner exhaust nozzle is controlled by afterburner fuel pressure through the exhaust nozzle control unit (figure 1-6). When the throttle is moved outboard to AFTERBURNER, the electrically operated shutoff valve in the fuel pump unit is opened (by power from the 28-volt dc essential bus), permitting unmetered fuel from the afterburner fuel pump to enter the afterburner exhaust nozzle control unit. This fuel pressure moves a valve within the control unit, which directs engine bleed air to the nozzle actuating cylinders to open the nozzle flaps. Moving the throttle inboard from AFTERBURNER range closes the shutoff valve, so that fuel pressure is no longer supplied to the nozzle control unit. The valve in the control unit is then positioned to route engine bleed air to close the nozzle flaps for normal engine operation.

engine fuel control system

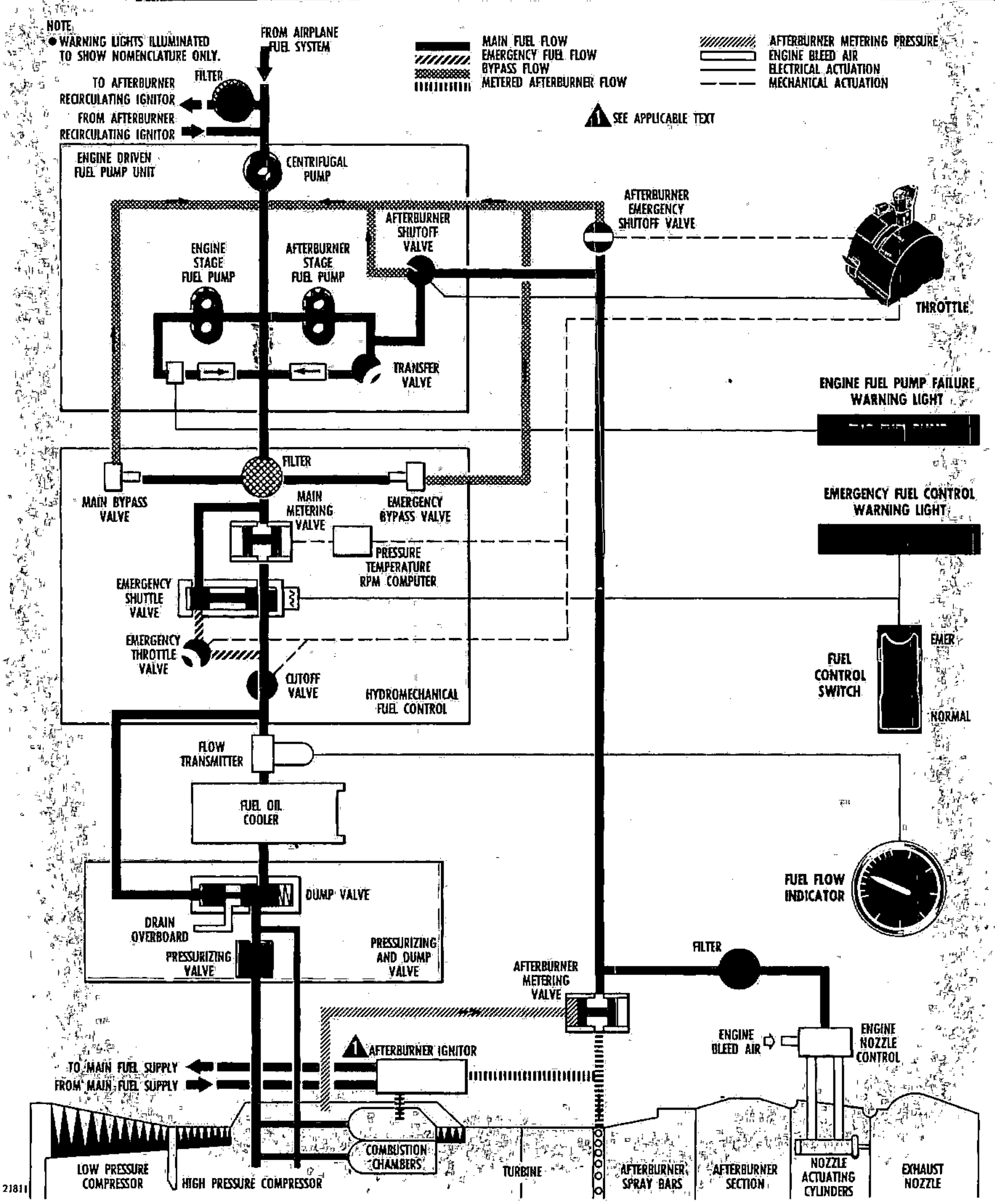


Figure 1-6

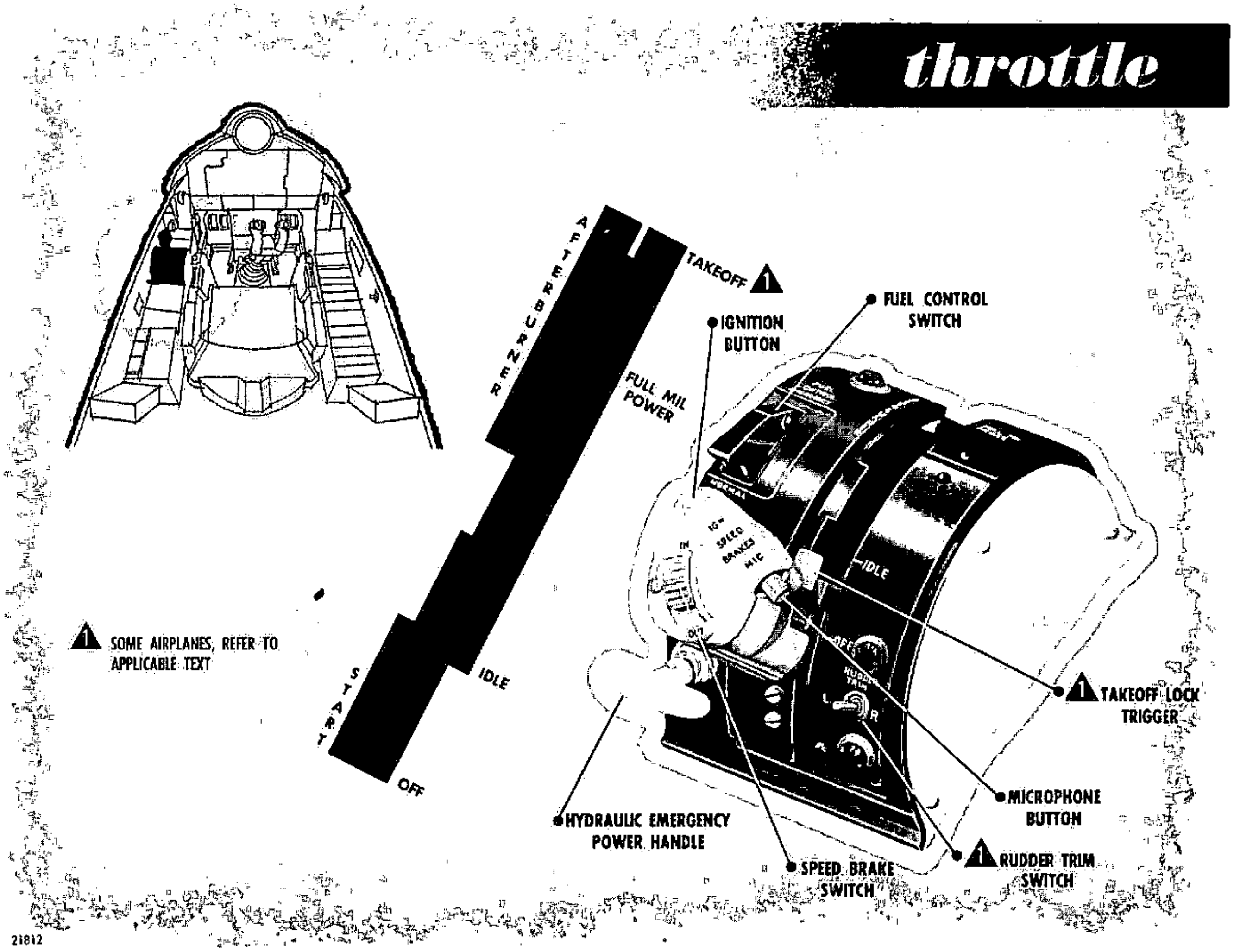


Figure 1-7

AFTERBURNER EMERGENCY SHUTOFF

The afterburner system is shut off mechanically in the event afterburner normal electrical control fails. When the throttle is moved out of the AFTERBURNER range and then retarded below approximately 90% rpm (throttle should be retarded smartly through this range) mechanical linkage will open the afterburner emergency bypass valve (see figure 1-6). All fuel being supplied to the afterburner system will then be returned to the engine fuel pump unit. This terminates afterburning operation and closes the exhaust nozzle. Whenever the throttle is advanced through the AFTERBURNER cut-off range, the bypass valve will close and again permit afterburning operation. If the afterburner shuts off normally, the operation of the emergency shutoff bypass valve will not affect normal operation.

OIL SUPPLY SYSTEM

The engine oil system supplies the engine with oil for lubrication and cooling and also supplies oil to serve as both a lubricating and an actuating fluid in the ac generator constant-speed drive unit. The engine oil tank has a capacity of 5.5 U.S. gallons plus an expansion space of 1.6 U.S. gallons. Maximum engine oil consumption is four pints per hour. Oil from the engine oil tank is taken into a gear-type pump, from which it is discharged under pressure to the engine gears, bearings, and also to the constant-speed drive, engine-mounted gear box. Scavenged oil is picked up by six gear-type pumps and routed through an air-oil cooler and to a fuel-oil cooler thermostatic valve. If the oil has been cooled sufficiently it is then routed to the oil tank. If it is not cooled sufficiently the oil is routed through a fuel-oil cooler and then returned to the oil tank. The oil tank contains a de-aerator which separates entrained air from the oil. The fuselage mounted constant-speed drive transmission oil supply is taken from the bottom of the engine oil tank. On some airplanes the oil is gravity-fed to the transmission. On other airplanes the oil reaches the transmission through a combination boost pump and negative g recirculating valve. During negative g conditions, the

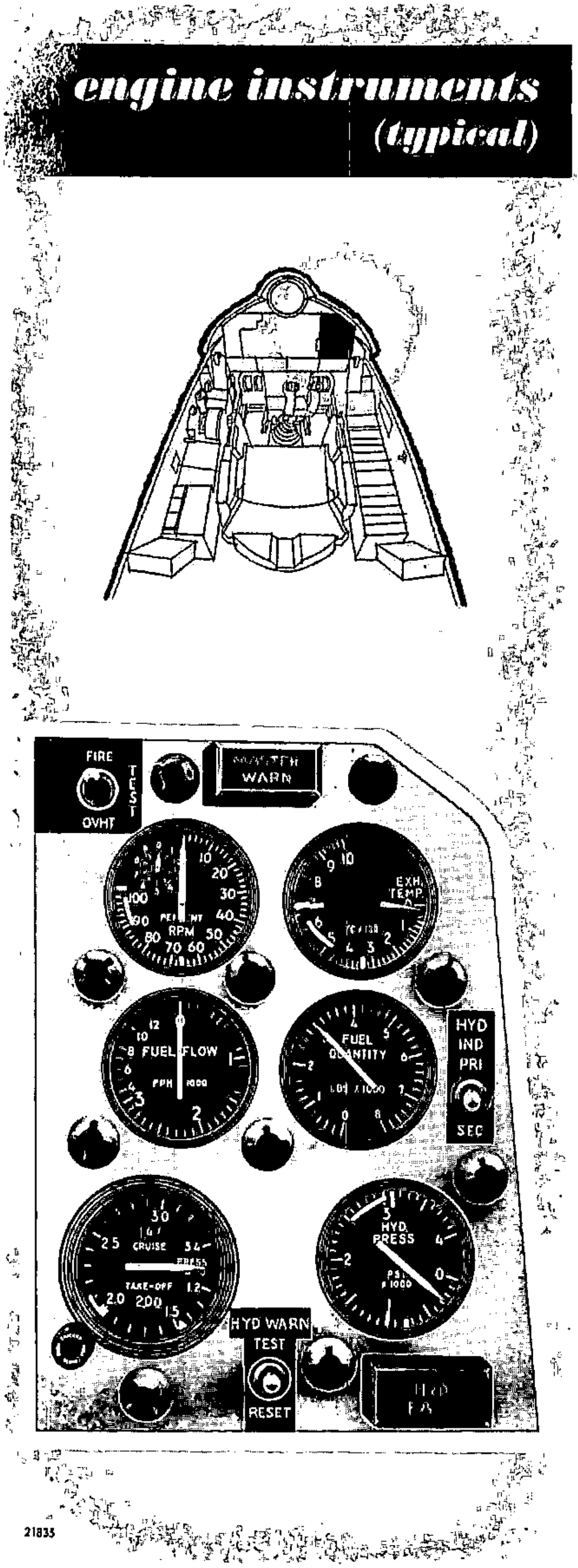


Figure 1-8

boost pump, recirculating valve, and an accumulator maintain a steady supply of oil to the constant-speed drive transmission by diverting the scavenge oil to the inlet of the transmission and recirculating it until g conditions return to normal. On some airplanes* a standpipe oil system has been installed. The standpipe reduces the possibility of complete loss of engine oil if a constant-speed drive failure results in loss of oil. The standpipe system provides a seven-quart emergency reserve of oil for the engine. See figure 1-33, for oil specifications.

CAUTION

Refer to PROHIBITED MANEUVERS, Section V, for negative g flight limits for airplanes with and without negative g recirculating valve installed.

FUEL SUPPLY SYSTEM

Fuel is supplied to the engine fuel control system by the airplane fuel system which consists of six integral tanks located in the wings (figure 1-9). The fuel supply can be augmented by installing droppable external tanks. The three fuel tanks in each wing are numbered in the order in which they are emptied. The No. 3 tanks are the last to empty and the only tanks which feed to the engine. No. 1, No. 2, and No. 3 tanks are located respectively in the aft-outboard, forward, and aft-inboard sections of each wing. The wing spars, ribs, and skin embody the individual tanks, and joints within the tanks are sealed by a bonding agent. Access to the individual tanks is obtained through access doors on the wing lower surface. Fuel is transferred from tank to tank within each wing by pressurization. This pressurization is provided by engine compressor bleed air and also serves to prevent excessive vaporization at high altitudes. Air pressure enters each No. 1 tank through a pressure regulator. As fuel is drawn from the No. 3 tank, the pressure differential between tanks forces fuel from the No. 1 tank to the No. 2 tank which, in turn, forces fuel into the No. 3 tank. Relief valves are provided to automatically protect the tanks against excessive pressure and vacuum. During a dive, the fuel transfer lines to the No. 3 tanks might be uncovered, disrupting fuel transfer and allowing air to enter these tanks while fuel remains in the No. 2 tanks. As level flight is resumed, float valves automatically vent the No. 3 tanks to atmosphere, restoring the pressure differential and normal fuel transfer. Fuel is supplied from the No. 3 tanks through fuel shutoff valves and a fuel flow equalizer to the engine fuel control system (figure 1-6). Two electrically driven boost pumps in each No. 3 tank provide additional system pressure. However, if the boost pumps are inoperative, the engine-driven fuel pump, aided by tank pressurization, will supply fuel through a dual check valve inlet bellmouth located on the output side of the boost pumps. When operating without boost

**Airplanes modified by TCTO 1F-102-694 and 2J-J57-591.*

pumps, care is required to prevent unporting the inlet bellmouths which will let air enter the engine fuel control system and could cause flameout.

SINGLE-POINT REFUELING

The internal fuel tanks can be serviced only by a single-point refueling adapter located in the aft side of the left main wheel well. During refueling operations fuel is routed to each No. 3 tank through a flow limiter-pressure regulator and refueling shutoff valve. When each No. 3 tank has filled, fuel passes into and fills each No. 2 tank and then the No. 1 tanks. As soon as each No. 1 tank fills, a high-level float valve will automatically close the refueling shutoff valve by hydraulic action and stop fuel flow input. External electrical power is not required during refueling. Inflight refueling provisions are not provided on this airplane.

Note

Fuel pressure of 55 to 60 psi should be provided from the fuel truck during refueling to obtain satisfactory operation of the refueling system.

Fuel tank capacities are given in figure 1-11 and fuel specifications in figure 1-33.

EXTERNAL TANKS

Droppable external fuel tanks can be installed on some airplanes* to augment the internal fuel supply. Installation provisions are made on the lower surface of each wing for two standard 230-gallon tanks which can be jettisoned by a ballistic charge. The fuel in these tanks is transferred into the No. 1 tanks by engine bleed air pressure which is regulated at a higher pressure than that of the normal fuel system pressurization. On some airplanes, once the engine has started, fuel transfer from the external tanks commences automatically. On later airplanes**, fuel from the external tanks may be controlled by an external fuel transfer switch (4, figure 1-4; figure 1-10). High-level floats within the No. 1 tanks will close shutoff valves in the fuel transfer lines between the external tanks and the No. 1 tanks by hydraulic action. This prevents the higher fuel pressure within the external tanks from entering the internal fuel system. A low-level float switch (power from the dc nonessential bus) in each external tank will close a shutoff valve in the pneumatic pressure line, preventing excessive pneumatic pressure from entering the internal fuel system, when fuel is exhausted from the external tanks. Vacuum relief valves in the pneumatic pressure lines will relieve negative pressures in the external tank during high rates of descent. Since external fuel is not indicated on the fuel quantity gage, fuel transfer from the external tanks can be noted by the fact that fuel quantity indication will not decrease until the external tanks have emptied.

**AF 53-1791, -1794, 56-1045 & on, & airplanes modified by TCTO 1F-102-534.*

***AF 53-1791, 1794, 56-1045 & on, & airplanes modified by TCTO 1F-102-677.*

Note

In the event of dc generator failure, loss of power to the low-level float switches in the external tanks would cause fuel transfer from these tanks to stop.

The external tanks are jettisoned by depressing the external tank jettison button. A ballistic charge is used to unlock and separate the tanks and pylons from the wings. Refer to Section V for operating limitations with external tanks installed.

FUEL BOOST PUMPS

Two electrically driven, submerged dual impeller boost pumps are installed in each No. 3 fuel tank (figure 1-9) to supply fuel under pressure to the engine fuel control system. Operation of these pumps is controlled by switches located on the fuel control panel. Each pump contains two suction-feed lines for fuel pickup at both top and bottom of the tank. Also, the pumps are diagonally located in the fuel tank to insure fuel supply regardless of flight attitude when there is usable fuel remaining in each No. 3 wing tank. If the boost pumps are inoperative, a dual-check valve inlet, located in the output fuel line, will permit the engine-driven pump to supply fuel to the engine by suction feed. These impeller boost pumps will not pump air. As long as any of the four pumps is operating in fuel, an uninterrupted flow of fuel is assured and entry of air into the engine fuel system is precluded. The pumps are powered by the 200/115-volt, 400-cycle nonessential ac bus and are protected by fuses. Control relays receive power from the 28-volt dc essential bus.

Note

Failure of boost pumps within one wing fuel system will establish a bypass condition of the fuel flow equalizer. Asymmetrical fuel usage should be corrected by shutting off boost pumps within the opposite wing.

FUEL SHUTOFF VALVES

Two electric, motor-driven, sliding-gate shutoff valves are used to cut off fuel supply to the engine from the fuel tanks. These valves, located one in each No. 3 tank outlet fuel line, can be controlled from the fuel control panel on the left console. On some airplanes, the valves can be controlled individually or simultaneously by use of the fuel selector switch. On other airplanes* a separate fuel shutoff valve switch is provided for each valve. The fuel shutoff valves are normally open throughout a flight and are powered by the 28-volt dc essential bus.

Note

The valves require three to five seconds to rotate fully closed. If the master switch is turned OFF during rotation, the sliding gate shutoff valves remain partially open.

**Airplanes modified by TCTO 1F-102-690.*

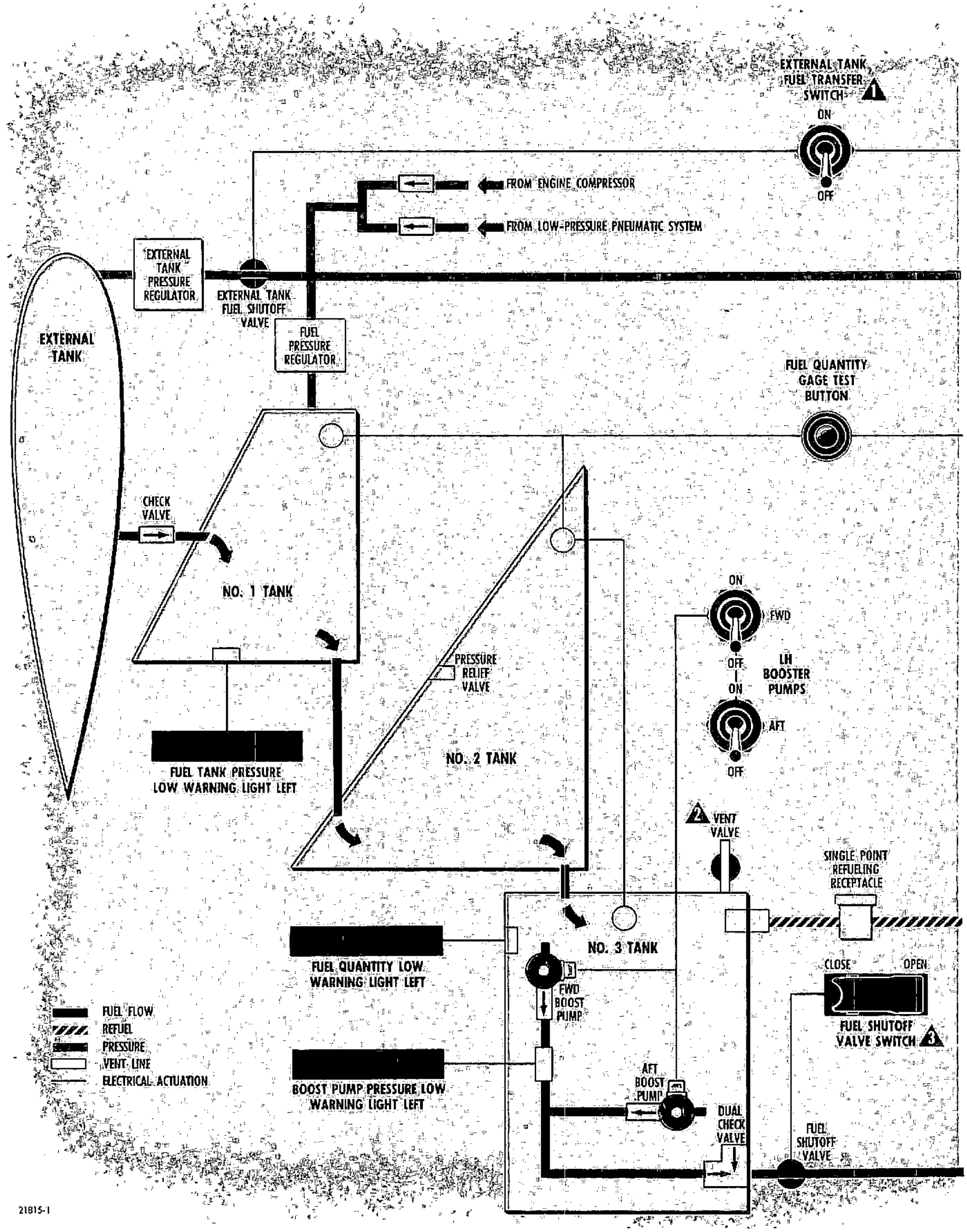
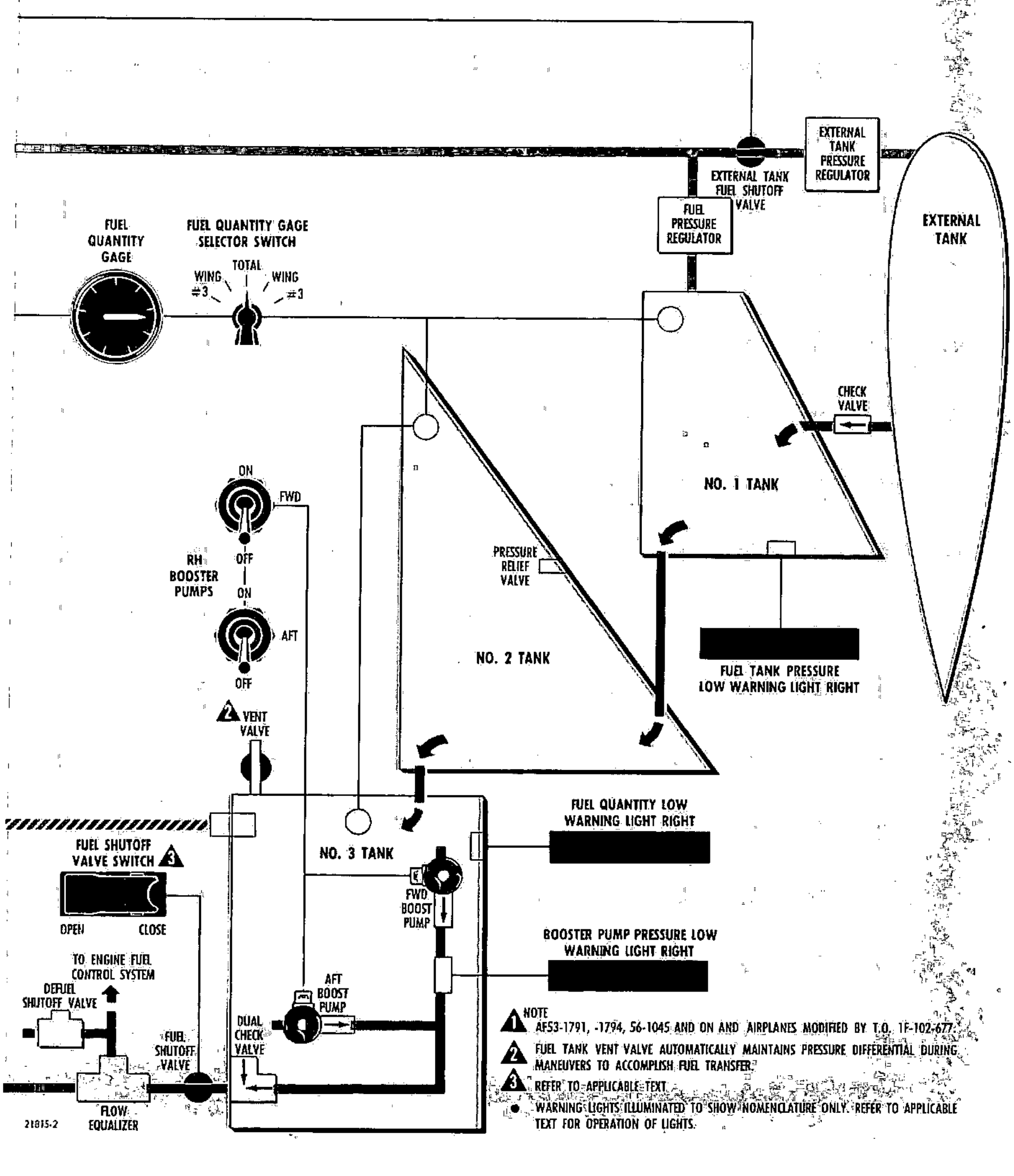

EXTERNAL TANK FUEL TRANSFER SWITCH 1
ON
OFF
FROM ENGINE COMPRESSOR
FROM LOW-PRESSURE PNEUMATIC SYSTEM
EXTERNAL TANK PRESSURE REGULATOR
EXTERNAL TANK FUEL SHUTOFF VALVE
FUEL PRESSURE REGULATOR
EXTERNAL TANK
FUEL QUANTITY GAGE TEST BUTTON
CHECK VALVE
NO. 1 TANK
ON
FWD
OFF
ON
LH BOOSTER PUMPS
AFT
OFF
PRESSURE RELIEF VALVE
NO. 2 TANK
FUEL TANK PRESSURE LOW WARNING LIGHT LEFT
2 VENT VALVE
SINGLE POINT REFUELING RECEPTACLE
NO. 3 TANK
FUEL QUANTITY LOW WARNING LIGHT LEFT
FWD BOOST PUMP
CLOSE
OPEN
FUEL SHUTOFF VALVE SWITCH 3
FUEL FLOW
REFUEL
PRESSURE
VENT LINE
ELECTRICAL ACTUATION
BOOST PUMP PRESSURE LOW WARNING LIGHT LEFT
AFT BOOST PUMP
DUAL CHECK VALVE
FUEL SHUTOFF VALVE
21815-1

Figure 1-9

fuel supply system



NOTE

1. AF53-1791, -1794, 56-1045 AND ON AND AIRPLANES MODIFIED BY T.O. 1F-102-677.
2. FUEL TANK VENT VALVE AUTOMATICALLY MAINTAINS PRESSURE DIFFERENTIAL DURING MANEUVERS TO ACCOMPLISH FUEL TRANSFER.
3. REFER TO APPLICABLE TEXT.

● WARNING LIGHTS ILLUMINATED TO SHOW NOMENCLATURE ONLY. REFER TO APPLICABLE TEXT FOR OPERATION OF LIGHTS.

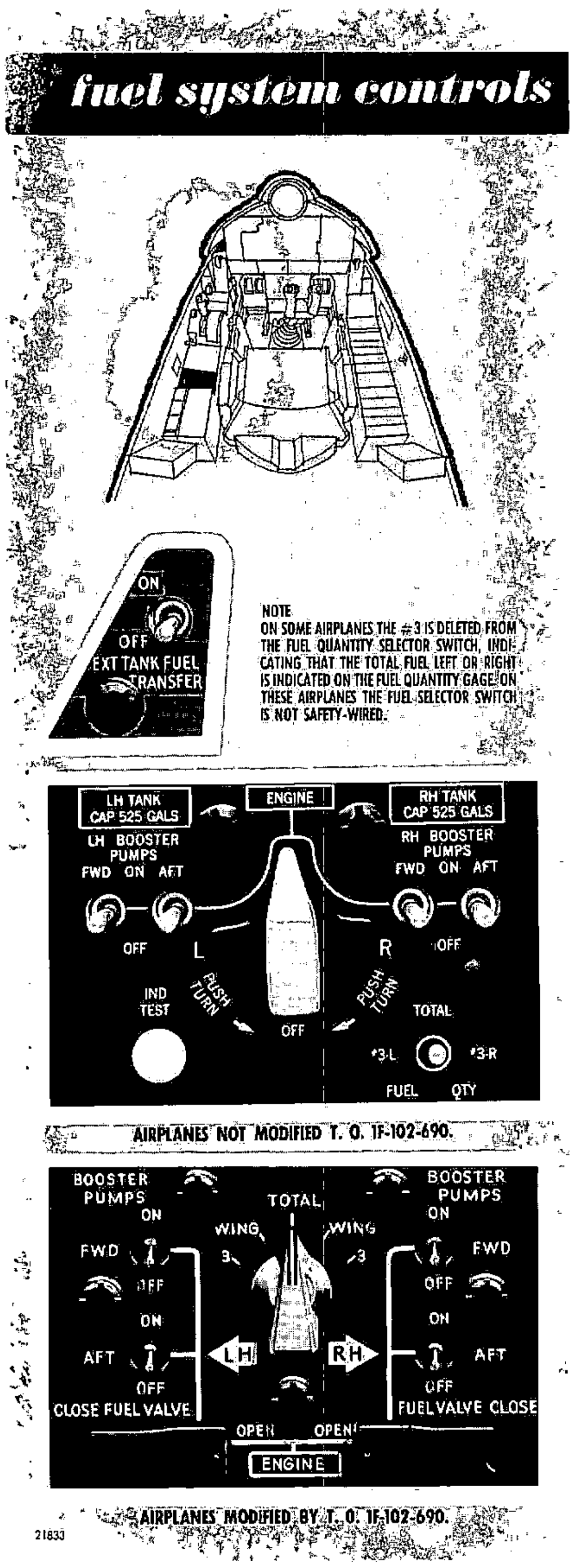


Figure 1-10

FUEL FLOW EQUALIZER

A fuel flow equalizer is used to regulate symmetrical fuel usage from each wing tank system. The flow equalizer is located in the fuel line between the No. 3 tank and the engine fuel control unit. A bypass condition is automatically established to insure fuel supply in the event of boost pump failure within one wing or malfunction of the flow equalizer.

FUEL SELECTOR SWITCH

On some airplanes* a four-position selector switch (figure 1-10) controls the fuel shutoff valves and is located on the fuel control panel. Switch positions are placarded L, ENGINE, R, and OFF. When positioned to L, fuel is supplied from the left-hand fuel system only. ENGINE position allows fuel to be supplied from both left- and right-hand fuel systems. When positioned to R, fuel is supplied from the right-hand fuel system only. OFF position closes both fuel shutoff valves. On some airplanes** the fuel selector switch is safety-wired to the ENGINE position and is placarded with operating instructions. Power is supplied from the 28-volt dc essential bus.

Note

- To preclude the possibility of a shutoff valve failing to open, the fuel selector switch should remain on ENGINE position during normal flight. The symmetrical fuel usage should be controlled by use of boost pump switches.
- Due to the proximity of the fuel selector switch to the ARC-34 command radio selector control, be certain that the desired control is selected prior to movement.

FUEL SHUTOFF VALVE SWITCHES

On some airplanes†, the left- and right-hand fuel shutoff valves are controlled by individual switches located at the bottom of the fuel control panel (figure 1-10). These switches, placarded "Fuel Valve," have OPEN and CLOSE positions and are guarded in the OPEN position. Each switch is individually operated to control the corresponding fuel shutoff valve. All fuel flow from a wing may be shut off by placing the appropriate switch in the CLOSE position. The fuel shutoff switches receive power from the 28-volt dc essential bus.

Note

To preclude the possibility of a shutoff valve failing to open, the fuel shutoff valve switches should remain at OPEN during normal flight. Symmetrical fuel usage should be controlled by use of boost pumps.

**Airplanes not modified by TCTO 1F-102-690.*
***In accordance with TCTO 1F-102-686.*
†Airplanes modified by TCTO 1F-102-690.

FUEL BOOST PUMP SWITCHES

Four two-position fuel boost pump switches (figure 1-10) are located on the fuel control panel to provide individual control of the fuel boost pumps. The pairs of switches are placarded "LH Boost Pumps" and "RH Boost Pumps" with ON and OFF positions which control the boost pumps accordingly. Individual switches are placarded "Fwd" and "Aft" to identify the specific pump within each fuel tank. On some airplanes*, fuel boost pumps switches are wired in series with the fuel shutoff valves, and the valves must be open for the pumps to operate. The control circuit receives power from the 28-volt dc essential bus.

EXTERNAL TANKS JETTISON BUTTON

The ring-guarded external tanks jettison button (figure 1-24) is provided to jettison the external tanks. The button is located on the landing gear control panel and is placarded "Wing Tank Release." When the button is pressed electrically fired ballistic charges unlock and separate the pylons and tanks from the wings. The jettison circuits receives power from the 28-volt dc essential bus.

CAUTION

The external tank jettison button should not be depressed while the landing gear is extended except during an emergency. The landing gear fairing doors will be damaged as the external tanks are released.

EXTERNAL TANK FUEL TRANSFER SWITCH

An external tank fuel transfer switch** (4, figure 1-4; figure 1-10) is installed on some airplanes to control fuel transfer from the external tanks. This switch is located on the left-hand side of the instrument panel and is placarded "Ext Tank Fuel Transfer" with positions ON and OFF. Placing the switch in the ON position opens a solenoid-operated valve in each external tank pressurization line. When fuel in the external tanks is expended pressurization to these tanks is shut off automatically. The switch is powered by the 28-volt dc nonessential bus.

Note

Refer to Section V for limitations on external tank use without the external tank fuel transfer switch installed.

EXTERNAL TANK JETTISON WARNING LIGHTS AND SAFETY PIN

Some airplanes** are equipped with an external tank ground safety switch and a ground safety switch warning light panel, both of which are located in the left side of the main wheel well. The ground safety switch is mounted on the side of the panel and consists of a receptacle into which a safety pin is inserted. The safety pin has a red streamer and should be installed during all ground operations. Two red, press-to-test lights are mounted on the panel, one placarded LH and the other RH. If either or both lights are illuminated, it is an indication that either or both tank jettison circuits are energized.

WARNING

Do not remove safety lock pin if red light is on.

Prior to removing the safety pin the lights should be press-tested for satisfactory operation. The warning lights are powered by the 28-volt dc essential bus.

FUEL QUANTITY GAGE SELECTOR SWITCH

The fuel quantity gage selector switch (figure 1-10), located on the fuel control panel, connects the fuel quantity gage to various tank quantity indicating circuits. On some airplanes, the switch is a three-position toggle switch with L, R, and TOTAL positions, permitting the indicator to read total internal fuel in both wings or total internal fuel in either wing. On some airplanes the L and R positions are replaced by #3L and #3R. Selecting these positions provides an indication of fuel quantity in the No. 3 fuel tanks. On still other airplanes†, the fuel quantity gage selector switch has a TOTAL position as well as a WING and #3 position for both left and right wings. Thus, it is possible to determine total internal fuel, total fuel in each wing, and total fuel in each No. 3 tank. There are no provisions for indicating fuel quantity in the external tanks. Power to the fuel quantity gage selector switch is supplied by the 28-volt dc essential bus.

FUEL QUANTITY GAGE AND GAGE TEST BUTTON

The fuel quantity gage (19, figure 1-4; figure 1-8) is located on the instrument panel and indicates internal fuel quantity in pounds. Each fuel tank is equipped with two tank probes to measure the quantity of fuel. The fuel quantity indicating system is a capacitance-type and compensates for changes in fuel density. The indicator dial reads from 0 to 8000 pounds and is marked in 200-pound increments. The fuel quantity gage indication depends upon the position of the fuel quantity gage selector switch.

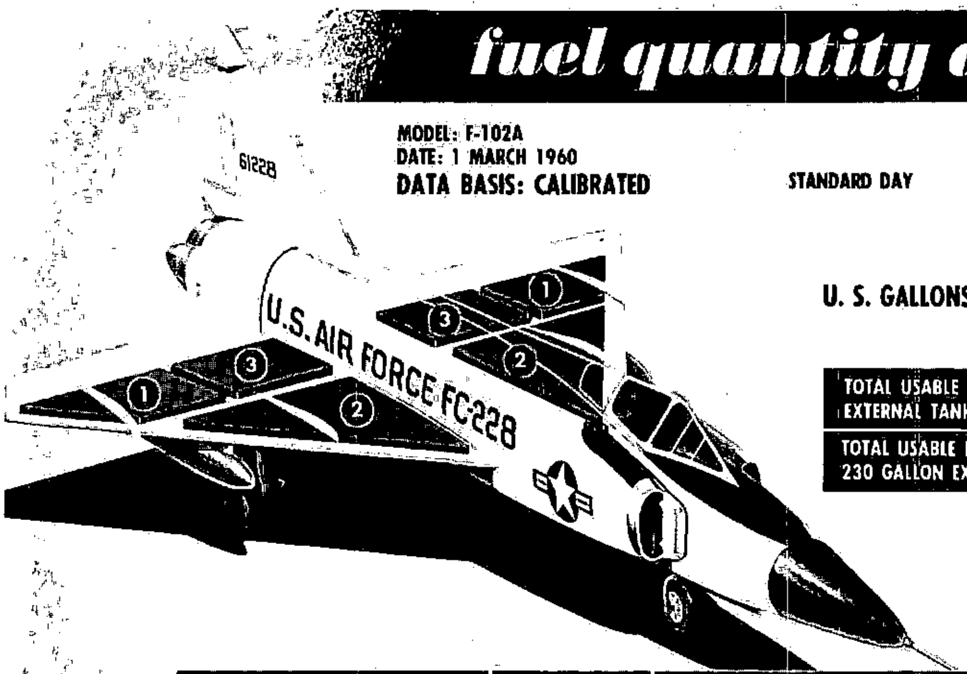
Note

External tank fuel quantity is not indicated on the fuel quantity gage; therefore when external tanks are used, a decrease is not indicated until external fuel is exhausted.

**AF 56-1206 & on, & airplanes modified by TCTO 1F-102-651.*
***AF 53-1791, -1794, 56-1045 & on, & airplanes modified by TCTO 1F-102-677.*

†Airplanes modified by TCTO 1F-102-690.

fuel quantity data table



MODEL: F-102A
DATE: 1 MARCH 1960
DATA BASIS: CALIBRATED

STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

U. S. GALLONS AND POUNDS

	POUNDS	GALLONS
TOTAL USABLE FUEL WITHOUT EXTERNAL TANKS	7053	1085
TOTAL USABLE FUEL WITH TWO 230 GALLON EXTERNAL TANKS	9848	1515

TANKS	NUMBER	USABLE FUEL		FULLY SERVICED	
		POUNDS	GALLONS	POUNDS	GALLONS
RIGHT WING	TANK NO. 1	920.0	141.5	3595.5	563
	TANK NO. 2	1631.5	251.0		
	TANK NO. 3	975.0	150.0		
LEFT WING	TANK NO. 1	920.0	141.5	3595.5	563
	TANK NO. 2	1631.5	251.0		
	TANK NO. 3	975.0	150.0		
EXTERNAL TANKS	LEFT HAND	1397.5	215.0	1417.0	218
	RIGHT HAND	1397.5	215.0	1417.0	218

21816

Figure 1-11

On some airplanes, gage operation can be checked by a test button (figure 1-10) placarded "Ind Test" on the fuel control panel. When the test button is held depressed, the gage pointer should move toward zero, and when the button is released, the pointer should return to its original position. Failure of the pointer to move indicates a faulty system. On other airplanes*, the only test of the indicator is to select the various positions of the selector switch and observe the indications. The indicating system receives power from the 115-volt essential bus and the 28-volt dc essential bus.

**Airplanes modified by TCTO 1F-102-690.*

FUEL QUANTITY-LOW WARNING LIGHTS

Two fuel quantity-low warning lights (9 and 10, figure 1-26), located on the warning light panel, illuminate and display "FUEL LOW—L" and "FUEL LOW—R" when the usable quantity of fuel in each No. 3 tank reaches approximately 88 gallons (570 pounds). The fuel quantity-low warning circuits receive power from the 28-volt dc essential bus.

FUEL BOOST PRESSURE-LOW WARNING LIGHTS

Two fuel boost pressure-low warning lights (13 & 14, figure 1-26), located on the warning light panel, illuminate and display "FUEL BOOST PRESS—L" or

"FUEL BOOST PRESS—R" if the left or right tank outlet pressure drops below 10.5 psi or on some airplanes* if a fuel shutoff valve is not in fully open position. The appropriate light will remain illuminated until the boost pump pressure exceeds 12 psi if the illumination was due to boost pump failure. The warning light will not illuminate if only one boost pump fails. The fuel boost pressure-low warning circuits receive power from the 28-volt dc essential bus.

FUEL TANK PRESSURE-LOW WARNING LIGHTS

Two fuel tank pressure-low warning lights (11 & 12 figure 1-26), located on the warning light panel, illuminate and display "FUEL TANK PRESS—L" or "FUEL TANK PRESS—R" if the left or right No. 1 tank pressurization falls below 0.5 psi. The appropriate light will remain illuminated until tank pressurization exceeds 1.5 psi. The fuel tank pressure-low warning circuits receive power from the 28-volt dc essential bus.

Note

Fuel tank pressure-low warning lights may illuminate temporarily during rapid descent from high altitude or negative g maneuvers. The lights should extinguish immediately after resuming level flight.

ELECTRICAL POWER SUPPLY SYSTEM

The airplane is equipped with direct-current and alternating current electrical power systems. The dc system is powered by an engine-driven generator and a battery. The ac system is powered by a three-phase main ac generator driven by an engine-driven, constant-speed drive unit. Two ac transformers are used to reduce the ac voltages. A hydraulically driven emergency ac generator serves as a standby power source. All electrical power can be shut off by use of the master switch, located on the electrical control panel. Both dc and ac power systems can be connected to an external power source for ground operations through external power receptacles.

DC ELECTRICAL POWER DISTRIBUTION

The 28-volt dc power supply system (figure 1-12) is powered by an engine-driven, 200-ampere generator and a 24 ampere-hour battery. Direct current is distributed through an essential and nonessential bus and, on some airplanes**, an emergency bus. The dc generator and battery are connected to the essential bus which in turn is connected to the nonessential bus by the nonessential bus tie relay. In the event of dc generator failure the nonessential bus is disconnected from the battery by the bus tie relay. The battery then becomes the source of power for the essential bus. The emergency dc bus normally receives power from the dc essential bus through an emergency dc bus changeover relay. This relay is energized by power from the dc nonessential bus. In the event of dc generator failure, the changeover relay automatically connects the emergency bus to a transformer-rectifier bus powered by the ac essential bus (see figure 1-13), thereby reducing the load on the battery. If ac power also fails, the emergency dc bus can be reconnected to the dc essential bus by operation of the battery switch to the TR FAIL position. The only power indication for the dc system is the dc power failure warning light.

AC ELECTRICAL POWER DISTRIBUTION

The ac power supply system (figure 1-13) is powered by a constant-speed, 30-kva generator. Alternating current is distributed through four bus networks, and by the use of two transformers, supplies three separate voltage systems. These systems consist of 200/115, 3-phase, 400-cycle essential and nonessential busses; a 115-volt, 3-phase 400-cycle essential bus; and a 26-volt, single phase, 400-cycle essential bus. In addition, some airplanes** have a transformer-rectifier connected to the 115-volt, 3-phase essential bus to provide emergency 28-volt dc power to the ac control circuits and for certain other systems (see figure 1-12). In the event of main ac generator failure selecting the emergency generator disconnects the nonessential bus and energizes and connects the hydraulically driven emergency ac generator to the essential buses. There is no provision for automatic switchover from the normal to the emergency ac generator.

ELECTRICALLY OPERATED EQUIPMENT

See figures 1-12 and 1-13 for complete reference to electrically operated equipment.

EXTERNAL POWER RECEPTACLES

The dc and ac power systems can be connected to an external power source for ground operations through main external power receptacles located on the forward side of the left main wheel well. These receptacles are protected by a spring-loaded dust cover.

Note

The dc essential bus must be energized to actuate the ac external power relay when external ac power is connected to the airplane unless the power cart contains a dc energizing circuit for the relay.

A dc generator test receptacle is located above the main external power receptacles for maintenance purposes. An auxiliary dc power receptacle is located on the aft side of the aft electronics bay to facilitate landing gear maintenance retraction tests. An outlet test receptacle is located on the nose wheel well switch panel, for electronics test purposes.

AF 56-1206 & on, & airplanes modified by TCTO 1F-102-651.

***Airplanes modified by TCTO 1F-102-727.*

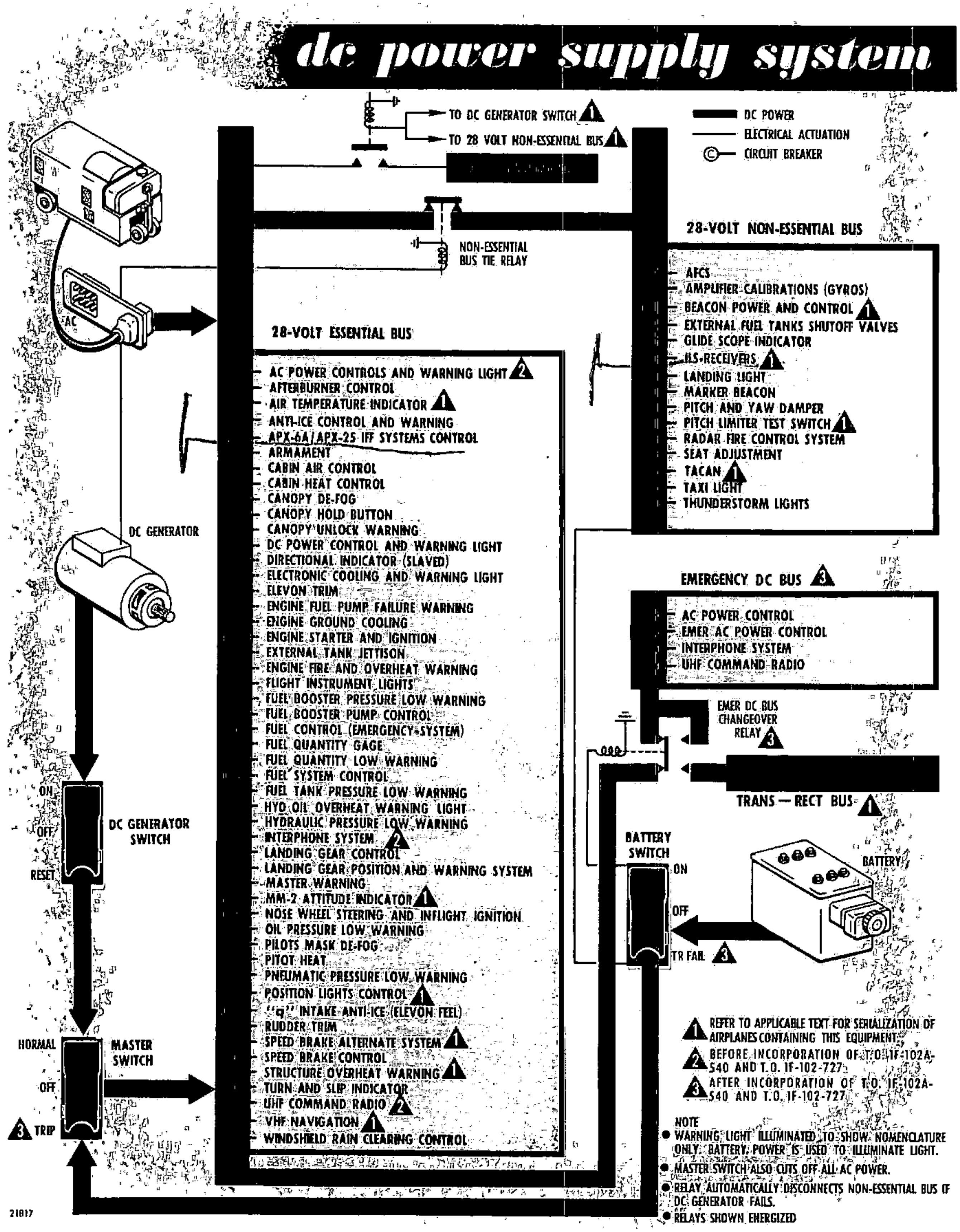

dc power supply system
TO DC GENERATOR SWITCH 1
TO 28 VOLT NON-ESSENTIAL BUS 1
DC POWER
ELECTRICAL ACTUATION
CIRCUIT BREAKER
NON-ESSENTIAL BUS TIE RELAY
28-VOLT NON-ESSENTIAL BUS
AFCS
AMPLIFIER CALIBRATIONS (GYROS)
BEACON POWER AND CONTROL 1
EXTERNAL FUEL TANKS SHUTOFF VALVES
GLIDE SCOPE INDICATOR
ILS RECEIVERS 1
LANDING LIGHT
MARKER BEACON
PITCH AND YAW DAMPER
PITCH LIMITER TEST SWITCH 1
RADAR FIRE CONTROL SYSTEM
SEAT ADJUSTMENT
TACAN 1
TAXI LIGHT
THUNDERSTORM LIGHTS
AC
28-VOLT ESSENTIAL BUS
AC POWER CONTROLS AND WARNING LIGHT 2
AFTERBURNER CONTROL
AIR TEMPERATURE INDICATOR 1
ANTI-ICE CONTROL AND WARNING
APX-6A/APX-25 IFF SYSTEMS CONTROL
ARMAMENT
CABIN AIR CONTROL
CABIN HEAT CONTROL
CANOPY DE-FOG
CANOPY HOLD BUTTON
CANOPY UNLOCK WARNING
DC POWER CONTROL AND WARNING LIGHT
DIRECTIONAL INDICATOR (SLAVED)
ELECTRONIC COOLING AND WARNING LIGHT
ELEVON TRIM
ENGINE FUEL PUMP FAILURE WARNING
ENGINE GROUND COOLING
ENGINE STARTER AND IGNITION
EXTERNAL TANK JETTISON
ENGINE FIRE AND OVERHEAT WARNING
FLIGHT INSTRUMENT LIGHTS
FUEL BOOSTER PRESSURE LOW WARNING
FUEL BOOSTER PUMP CONTROL
FUEL CONTROL (EMERGENCY SYSTEM)
FUEL QUANTITY GAGE
FUEL QUANTITY LOW WARNING
FUEL SYSTEM CONTROL
FUEL TANK PRESSURE LOW WARNING
HYD OIL OVERHEAT WARNING LIGHT
HYDRAULIC PRESSURE LOW WARNING
INTERPHONE SYSTEM 2
LANDING GEAR CONTROL
LANDING GEAR POSITION AND WARNING SYSTEM
MASTER WARNING
MM-2 ATTITUDE INDICATOR 1
NOSE WHEEL STEERING AND INFLIGHT IGNITION
OIL PRESSURE LOW WARNING
PILOTS MASK DE-FOG
PITOT HEAT
PNEUMATIC PRESSURE LOW WARNING
POSITION LIGHTS CONTROL 1
"q" INTAKE ANTI-ICE (ELEVON FEEL)
RUDDER TRIM
SPEED BRAKE ALTERNATE SYSTEM 1
SPEED BRAKE CONTROL
STRUCTURE OVERHEAT WARNING 1
TURN AND SLIP INDICATOR
UHF COMMAND RADIO 2
VHF NAVIGATION 1
WINDSHIELD RAIN CLEARING CONTROL
EMERGENCY DC BUS 3
AC POWER CONTROL
EMER AC POWER CONTROL
INTERPHONE SYSTEM
UHF COMMAND RADIO
EMER DC BUS CHANGEOVER RELAY 3
TRANS — RECT BUS 1
DC GENERATOR
ON
OFF
RESET
DC GENERATOR SWITCH
NORMAL
OFF
3 TRIP
MASTER SWITCH
BATTERY SWITCH
ON
OFF
TR FAIL 3
BATTERY
1 REFER TO APPLICABLE TEXT FOR SERIALIZATION OF AIRPLANES CONTAINING THIS EQUIPMENT.
2 BEFORE INCORPORATION OF T.O. 1F-102A-540 AND T.O. 1F-102-727.
3 AFTER INCORPORATION OF T.O. 1F-102A-540 AND T.O. 1F-102-727.
NOTE
• WARNING LIGHT ILLUMINATED TO SHOW NOMENCLATURE ONLY. BATTERY POWER IS USED TO ILLUMINATE LIGHT.
• MASTER SWITCH ALSO CUTS OFF ALL AC POWER.
• RELAY AUTOMATICALLY DISCONNECTS NON-ESSENTIAL BUS IF DC GENERATOR FAILS.
• RELAYS SHOWN ENERGIZED
21817

Figure 1-12

CIRCUIT BREAKERS AND FUSES

Individual circuit protection is provided by thermal-type circuit breakers throughout the airplane, with the exception of the pitot heat control circuit. These circuit breakers are called "trip-free" and manually holding the circuit breaker in the depressed position will not complete the circuit if it remains faulty or overloaded, thereby reducing fire hazard. The pitot heat control circuit is provided with a switch-type circuit breaker which can be held in the ON position, in case of emergency. The circuit breakers which are accessible to the pilot are on the circuit breaker panels (figure 1-15) located at the aft end of the left and right consoles and at the forward side of the left console. Fuses are used in the boost pump motor circuit, and spare fuses for this circuit are located adjacent to the boost pump relays on the aft side of each main wheel well.

MASTER SWITCH

The guarded master switch (figure 1-14), located on the electrical power control panel, is used to shut off all generator and battery power during emergency conditions. The switch is placarded "Master" and has NORMAL and OFF positions and, on some airplanes*, a TRIP position. In the NORMAL position the switch connects the battery and the generators to the airplane buses if the battery and generator switches are ON. The TRIP position is a momentary position spring-loaded to OFF, and provides battery power for tripping the generator control panels before dropping out the battery relay. The master switch receives power for disconnecting the generators from the ac generator exciter circuit on some airplanes, and from the dc essential bus on other airplanes*.

CAUTION

The master switch should be used in cases of emergency only. When in OFF position, generator disconnect relays are energized (on some airplanes**) which would drain the battery if left in this position.

BATTERY SWITCH

The guarded battery switch (figure 1-14), located on the electrical power control panel, is used to disconnect the battery from the airplane electrical system. The switch is placarded "Bat" and has ON and OFF positions, which control the circuit accordingly. On some airplanes*, the battery switch also has a TR FAIL position which energizes the emergency dc bus changeover relay directly from the battery. The TR FAIL position is used to connect the emergency dc bus to the dc essential bus (battery) when the dc generator and the transformer-rectifier have failed or when the ac, dc, and emergency ac generators have failed.

**Airplanes modified by TCTO 1F-102-727.*
***AF 53-1791 thru -1818.*

CAUTION

Placing the battery switch in the TR FAIL position at any other time may result in complete electrical power failure after battery power is depleted.

The battery control circuit receives power directly from the battery.

DC GENERATOR SWITCH

The guarded dc generator switch (figure 1-14), located on the electrical power control panel, is used to control dc generator operation. The switch is placarded "DC Gen" and has three positions, ON, OFF, and RESET. When the generator switch is ON, generator output is supplied to the dc electrical system. Placing the generator switch to OFF disconnects the generator from the essential bus which causes the nonessential bus tie relay to disconnect the nonessential bus from the battery. If a malfunction cuts out the generator or the master switch has been placed to OFF, then to NORMAL, the generator switch should be held momentarily at RESET then returned to ON to restore normal generator operation. The dc generator control circuit receives power from the 28-volt dc essential bus.

AC GENERATOR SWITCH

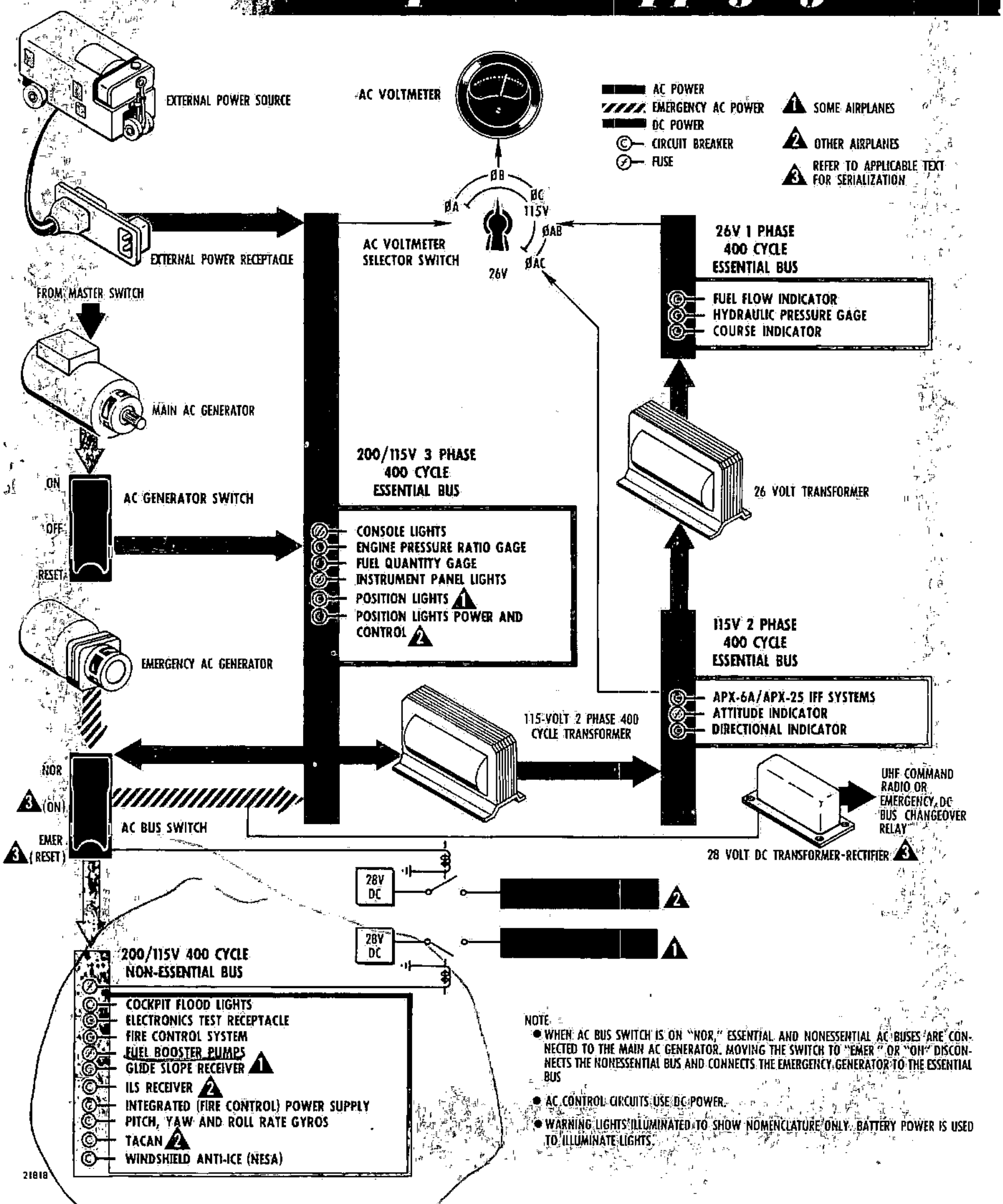
The guarded ac generator switch (figure 1-14), located on the electrical power control panel, is used to control ac generator operation. The switch is placarded "AC Gen" and has three positions, ON, OFF, and RESET. When the generator switch is ON, generator output is supplied to the ac electrical system. In the event of main ac generator failure, the ac generator switch should be turned to OFF to preclude the possibility of a fire hazard from a faulty generator. The emergency ac generator will energize directly from the ac bus switch. Placing the switch to OFF disconnects the generator from the buses. If a malfunction cuts out the generator or the master switch has been placed to OFF, then to NORMAL, the generator switch should be held momentarily at RESET, then returned to ON to restore normal generator operation. The ac generator control circuit receives power from the 28-volt dc essential bus or, on some airplanes*, from the emergency dc bus.

AC BUS SWITCH

The guarded ac bus switch (figure 1-14), located on the electrical power control panel, is used to energize the hydraulically driven emergency ac generator. The switch is placarded "AC Bus." On some airplanes the switch has two positions, NOR and EMER. Placing the switch to NOR connects the main ac generator to the buses, and

**Airplanes modified by TCTO 1F-102-727.*

ac power supply system



NOTE:

- WHEN AC BUS SWITCH IS ON "NOR," ESSENTIAL AND NONESSENTIAL AC BUSES ARE CONNECTED TO THE MAIN AC GENERATOR. MOVING THE SWITCH TO "EMER" OR "ON" DISCONNECTS THE NONESSENTIAL BUS AND CONNECTS THE EMERGENCY GENERATOR TO THE ESSENTIAL BUS
- AC CONTROL CIRCUITS USE DC POWER.
- WARNING LIGHTS ILLUMINATED TO SHOW NOMENCLATURE ONLY. BATTERY POWER IS USED TO ILLUMINATE LIGHTS.

Figure 1-13

when placed to EMER the emergency ac generator is energized, connected to the essential bus and the main ac generator and nonessential bus is disconnected from the essential bus. On these airplanes, control circuit power is supplied from the 28-volt dc essential bus through the main ac generator switch. On other airplanes*, the ac bus switch is placarded "Emer Gen" and has three positions, NOR, ON, and RESET. In the NOR position, the main ac generator is connected to the ac buses. Actuation of the switch to the spring-loaded RESET position completes a circuit to the ac emergency disconnect relay and the emergency generator shutoff valve. The completed circuit starts the emergency ac generator, ties it to the ac essential bus, and disconnects the main ac generator. When released, the ac bus switch will return to the ON position which ties the emergency disconnect relay and emergency shutoff valve to the emergency dc bus. On these airplanes, the ac bus switch receives power through the emergency dc bus and, in the RESET position, directly from the dc essential bus.

AC VOLTMETER AND SELECTOR SWITCH

The ac voltmeter (figure 1-14), located above the electrical power control panel, provides a means of determining ac generator output as selected by the voltmeter selector switch. The six-position voltmeter selector switch (figure 1-14), located on the electrical power control panel, provides a means of connecting the voltmeter to the essential ac bus circuits as follows: Phases A, B, and C of the main ac generator output (approximately 115 volts); phase AB and AC of the 115-volt transformer; and the single phase output of the 26-volt transformer.

DC POWER FAILURE WARNING LIGHT

The dc power failure warning light (3, figure 1-26), located on the warning light panel, illuminates and displays "DC POWER FAILURE" if the dc generator or the dc power disconnect relay fails. On some airplanes** the warning light will come on when the battery switch is ON or when the external power unit is connected for starting and will remain on until external power is disconnected after the engine is running. On other airplanes the light will remain out when power (external or airplane power) is applied to the buses. The dc power failure warning circuit receives power from the 28-volt dc essential bus (battery power).

Note

DC power is used to operate the electrical control panel which contains the master switch. In the event of both dc generator and battery failure, this panel will become inoperative on on some airplanes and ac power will not be available. On other airplanes*, dc power will be provided by the transformer-rectifier for the control of the ac system.

*Airplanes modified by TCTO 1F-102-727.
**AF 56-1275 thru -1316, -1332 & on.

electrical power control panel

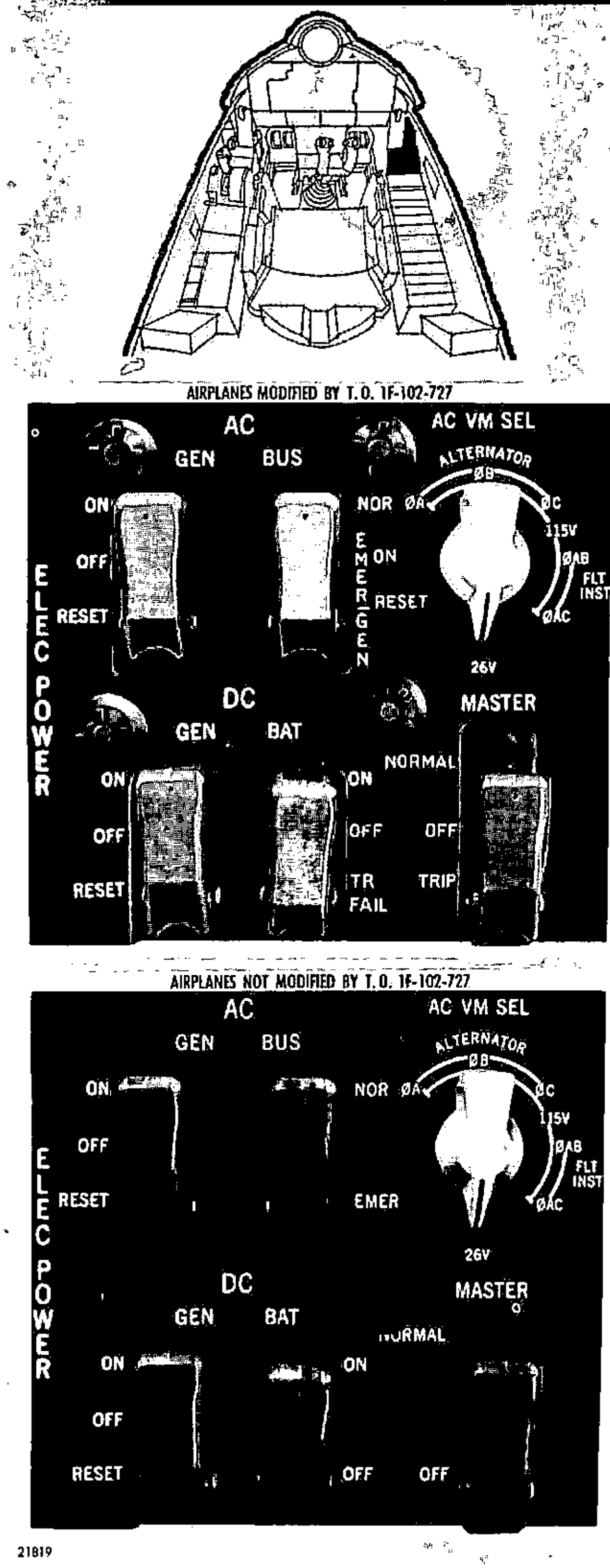


Figure 1-14

circuit breaker panels (typical)

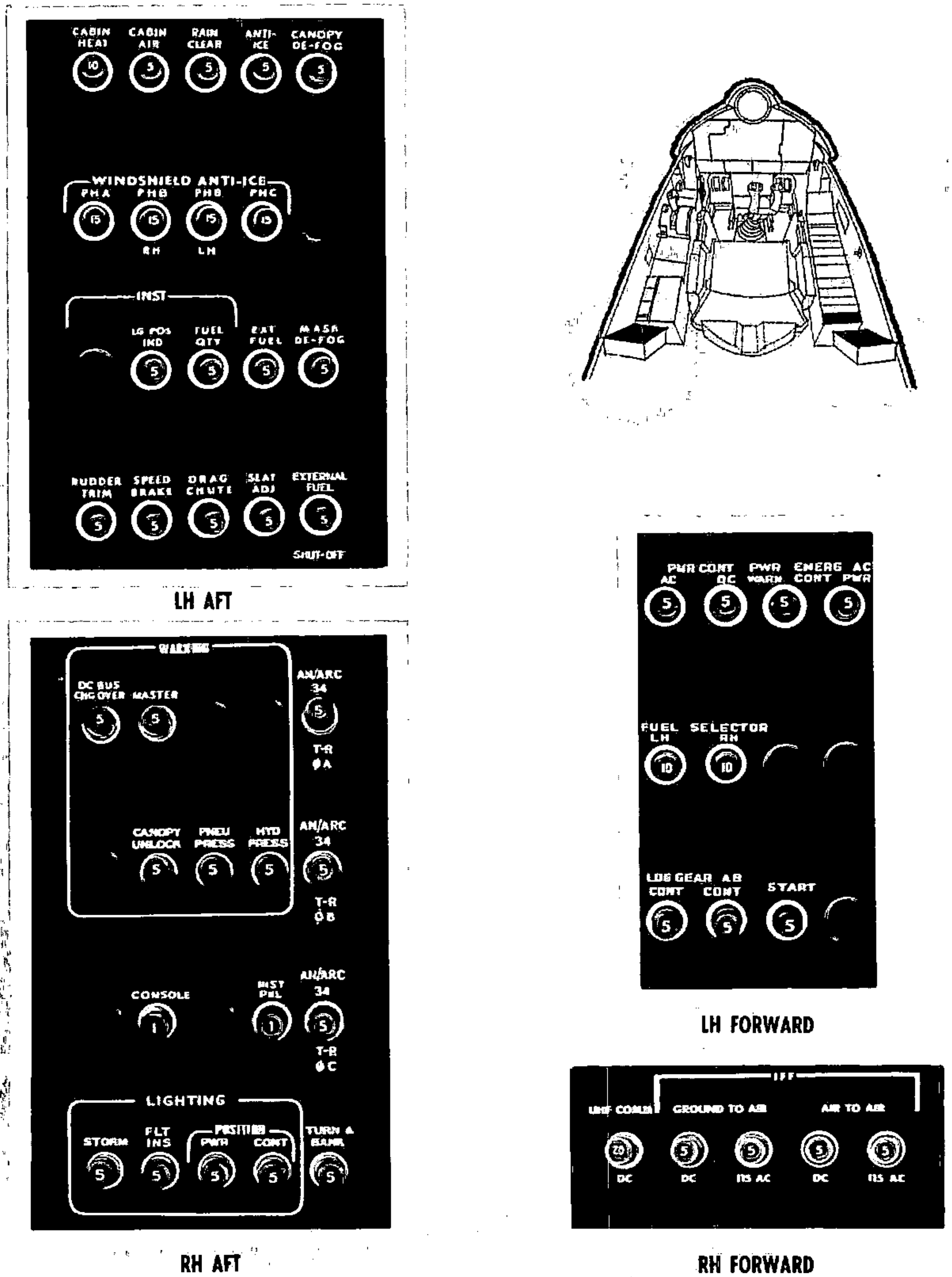


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Figure 1-15

AC POWER FAILURE WARNING LIGHT

The ac power failure warning light (2, figure 1-26), located on the warning light panel, illuminates and displays "AC POWER FAILURE" if the main ac generator or the ac power disconnect relay fails. On some airplanes* the warning light will come on when the battery switch is ON or the external power unit is connected for starting and will remain on until external power is disconnected after the engine is running. On other airplanes the light will remain out when power (external or airplane power) is applied to the buses. The ac power failure warning circuit receives power from the 28-volt dc essential bus.

HYDRAULIC POWER SUPPLY SYSTEM

The hydraulic power supply system (figure 1-16) consists of two separate constant-pressure type systems, the primary and secondary, which supply power to actuate most of the major operating components of the airplane. Normal operation of both hydraulic systems is automatic whenever the engine is running. An emergency system is also provided to supplement the primary system in event of an emergency. Pressure in either of these systems can be read on a single hydraulic pressure gage by the use of a hydraulic pressure gage selector switch. A flashing red light is provided to warn of a single system failure, or steady illumination of the same light indicates failure of both systems. On some airplanes**, a hydraulic fluid overheat warning light is also provided to warn of overheated hyraulic fluid in either the primary or secondary hydraulic systems. Illumination of the light will not differentiate between the primary and secondary hydraulic systems. See figure 1-33 for hydraulic fluid specification.

PRIMARY HYDRAULIC SYSTEM

The primary hydraulic system supplies power for operation of the flight controls only. The system is completely independent of the secondary system and consists primarily of a reservoir, a 3000 psi variable volume engine-driven pump, accumulator, supply lines, and thermal-pressure relief valves for system protection. On some airplanes† hydraulic fluid is cooled by heat exchanger coils in the right No. 3 fuel tank. The system contains conventional filters with bypass features. The reservoir has a capacity of 252 cu. in. or 1.09 gallons and is pressurized by the low-pressure pneumatic system. The piston-type accumulator contains a pressure gage (not accessible in flight) for ground checking the preload pressure.

SECONDARY HYDRAULIC SYSTEM

The secondary hydraulic system supplies power for operation of the flight controls (parallels the primary system action). The system also supplies power for operation of the landing gear and doors, nose wheel steering, speed brakes and the emergency ac generator. The secondary hydraulic system consists primarily of a reservoir, a 3000 psi variable volume engine-driven pump, accumulator, supply lines, and thermal-pressure relief valves for system protection. On some airplanes† hydraulic fluid is cooled by heat exchanger coils in the left No. 3 fuel tank. The system contains conventional filters with bypass features. The reservoir has a capacity of 420 cu. in. or 1.82 gallons and is pressurized by the low-pressure pneumatic system. The piston-type accumulator contains a pressure gage (not accessible in flight) for ground checking the preload pressure.

AF 56-1275 thru -1316, -1332 & on.

***Airplanes modified by TCTO 1F-102-847.*

†*Airplanes modifed by TCTO 1F-102-778.*

EMERGENCY HYDRAULIC SYSTEM

The emergency hydraulic system supplies power for operation of the flight controls in event of failure of the primary and secondary hydraulic systems or when the engine fails and is "frozen." The emergency hydraulic pump is driven by a variable pitch ram air turbine (RAT), which is pneumatically extended into the airstream by pulling a hydraulic emergency power handle. Since the emergency hydraulic system utilizes the same hydraulic lines as the primary hydraulic system, it is inadvisable to extend the RAT when only the secondary hydraulic system has failed, as this could cause damage and possible loss of the primary hydraulic system. When only the primary hydraulic system has failed, the RAT should be extended just after turn on final approach as a precautionary measure. Once extended the RAT cannot be retracted in flight. This emergency system will supply sufficient power for limited maneuvering and a safe approach and landing. A RAT door test hook is installed on some airplanes on the RAT door eyebolt, to insure that the RAT door is locked. The hook is equipped with a warning streamer and must be removed prior to flight.

RAT (HYDRAULIC EMERGENCY POWER) HANDLE

The RAT handle (figure 1-7), located in the outboard side of the throttle quadrant, is used to energize the emergency hydraulic system. Pulling the handle (approximately two inches) mechanically selects the pneumatic pressure to extend the RAT into the airstream. To insure satisfactory extension, the handle must be extended to full travel and held for a minimum of four seconds.

HYDRAULIC PRESSURE GAGE AND SELECTOR SWITCH

The hydraulic pressure gage (20, figure 1-4; figure 1-8) is located on the instrument panel (on early airplanes this gage is located on the right-hand console) and indicates either the primary or secondary hydraulic system pressure in increments from 0 to 4000 psi. The indicator receives system pressure information from either the

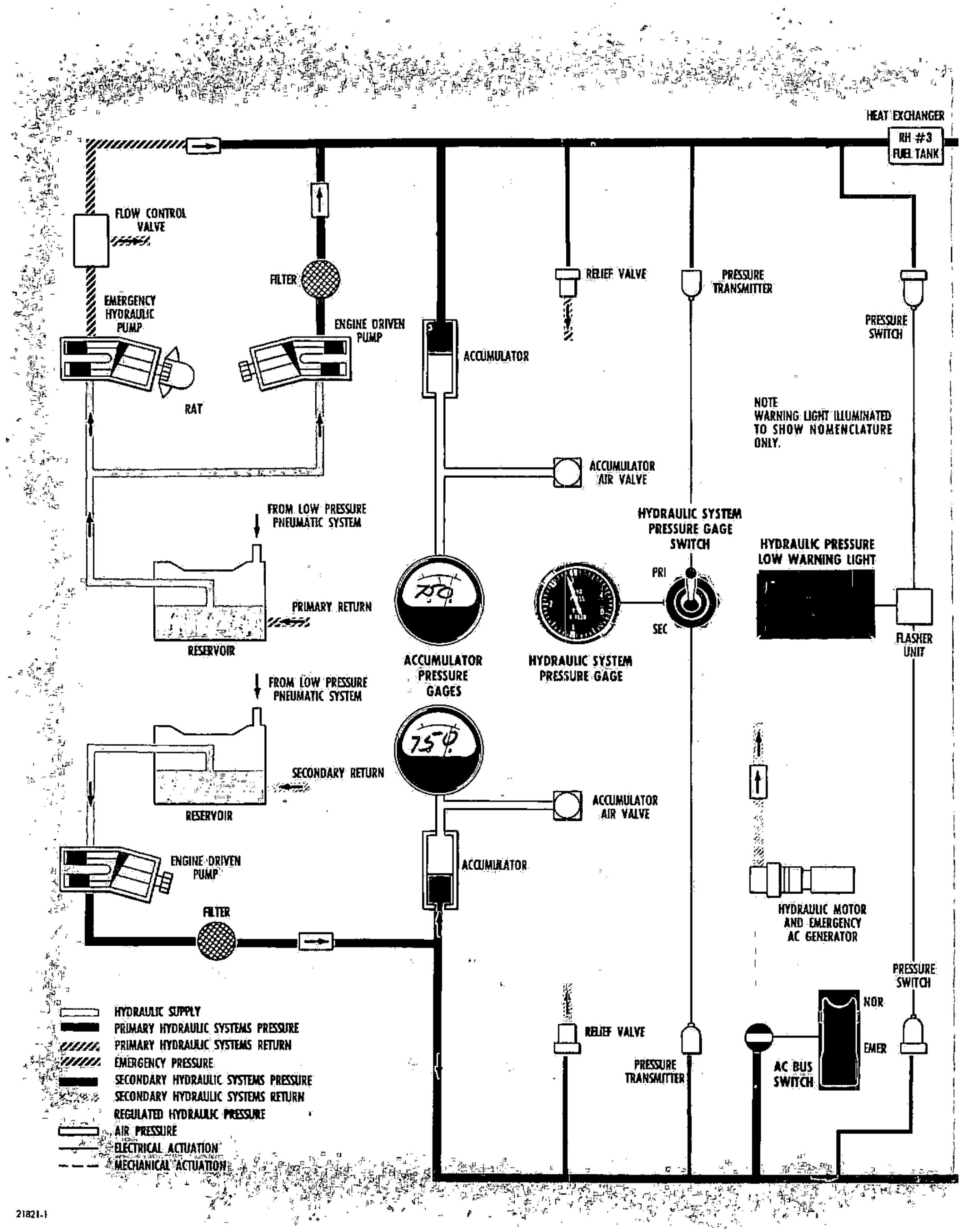
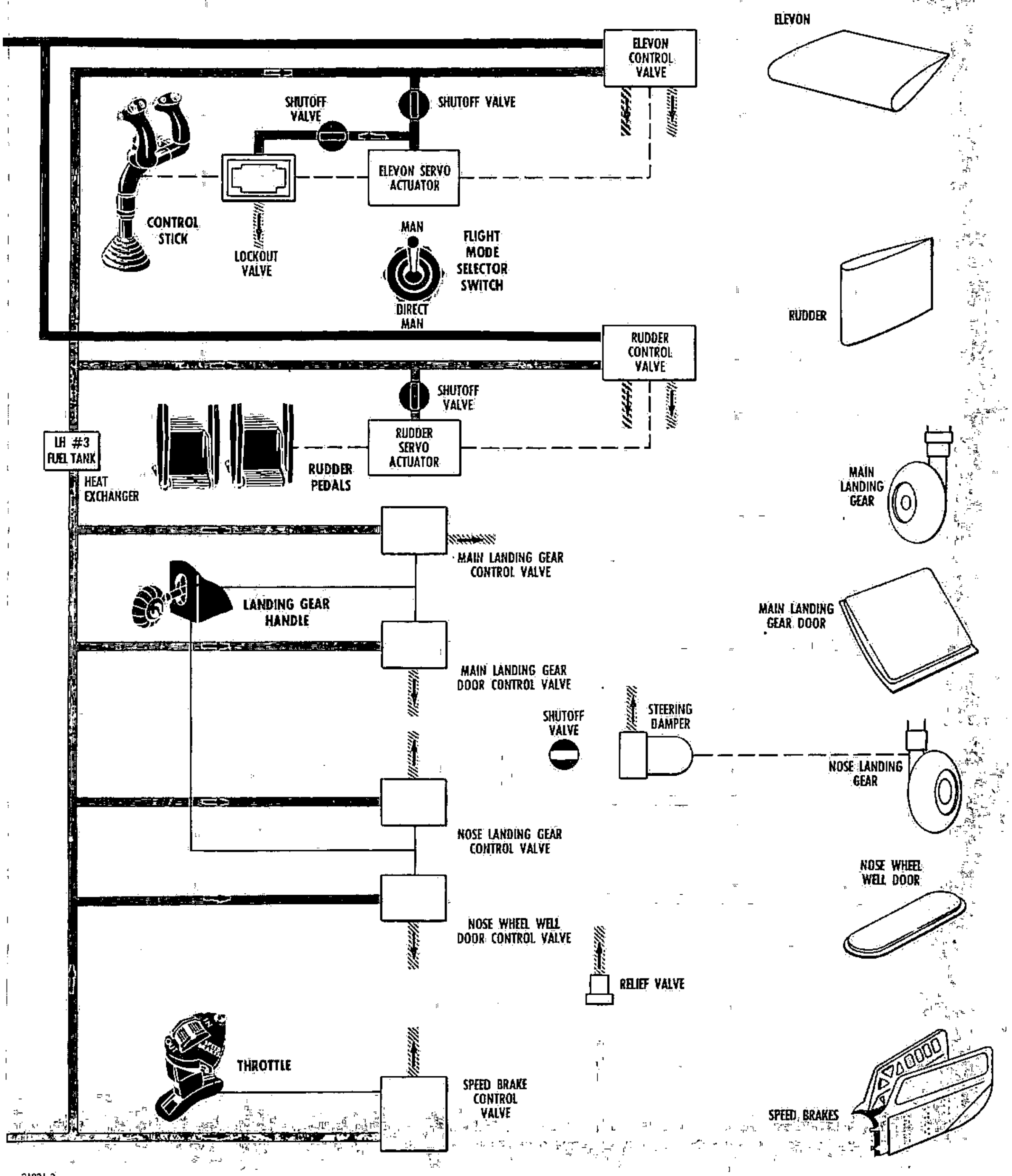

HEAT EXCHANGER
RH #3 FUEL TANK
FLOW CONTROL VALVE
FILTER
EMERGENCY HYDRAULIC PUMP
ENGINE DRIVEN PUMP
ACCUMULATOR
RELIEF VALVE
PRESSURE TRANSMITTER
PRESSURE SWITCH
RAT
NOTE
WARNING LIGHT ILLUMINATED TO SHOW NOMENCLATURE ONLY.
ACCUMULATOR AIR VALVE
FROM LOW PRESSURE PNEUMATIC SYSTEM
HYDRAULIC SYSTEM PRESSURE GAGE SWITCH
HYDRAULIC PRESSURE LOW WARNING LIGHT
PRI
SEC
PRIMARY RETURN
RESERVOIR
FLASHER UNIT
ACCUMULATOR PRESSURE GAGES
HYDRAULIC SYSTEM PRESSURE GAGE
FROM LOW PRESSURE PNEUMATIC SYSTEM
SECONDARY RETURN
RESERVOIR
ACCUMULATOR AIR VALVE
ENGINE DRIVEN PUMP
ACCUMULATOR
FILTER
HYDRAULIC MOTOR AND EMERGENCY AC GENERATOR
PRESSURE SWITCH
NOR
EMER
RELIEF VALVE
PRESSURE TRANSMITTER
AC BUS SWITCH
HYDRAULIC SUPPLY
PRIMARY HYDRAULIC SYSTEMS PRESSURE
PRIMARY HYDRAULIC SYSTEMS RETURN
EMERGENCY PRESSURE
SECONDARY HYDRAULIC SYSTEMS PRESSURE
SECONDARY HYDRAULIC SYSTEMS RETURN
REGULATED HYDRAULIC PRESSURE
AIR PRESSURE
ELECTRICAL ACTUATION
MECHANICAL ACTUATION
21821-1

Figure 1-16

hydraulic power supply system



21821-2

primary or secondary transmitter, depending upon the position of the selector switch adjacent to the indicator. This two-position hydraulic pressure gage selector switch enables the pilot to select either PRI or SEC systems. The indicating and selection systems are powered by the 26-volt ac essential bus.

HYDRAULIC PRESSURE-LOW WARNING LIGHT AND TEST-RESET SWITCH

The red hydraulic pressure-low warning light (21, figure 1-4; figure 1-8), located on the instrument panel, illuminates (flashing) and displays "HYD FAIL" if either the primary or secondary hydraulic system pressure is below approximately 800 psi. Pressure loss of both systems will cause steady illumination of this light. With rising pressure, in either system, the light will start flashing at approximately 1000 psi, or if both systems are in excess of approximately 1000 psi the light will go off. The three-position hydraulic pressure-low warning light test and reset switch (figure 1-8) has TEST and RESET positions and is spring-loaded to the center (OFF) position. If either hydraulic system has failed and caused a flashing light, placing the switch to RESET will turn the light off. RESET position will not turn the light off if both hydraulic systems have failed. Placing the switch to TEST position will illuminate the light in a flashing condition if the pressure in both systems is above approximately 1000 psi. The warning light is automatically dimmed when the instrument panel lights are on if the thunderstorm lights are off. Power is supplied by the 28-volt dc essential bus.

HYDRAULIC FLUID OVERHEAT WARNING LIGHT

The hydraulic fluid overheat warning light* (1, figure 1-26) located on the warning light panel, illuminates and displays "HYD OIL HOT" when hydraulic fluid in either the primary or secondary hydraulic system reaches a temperature of 225°(−5 +10°)F. The hydraulic fluid overheat warning light receives power from the dc essential bus.

Note

A hydraulic system overheat warning indication will not differentiate between the primary and secondary hydraulic systems.

PNEUMATIC POWER SUPPLY SYSTEM

The pneumatic power supply system consists of two separate systems, low- and high-pressure, which are used for pressurizing and actuation of system components.

LOW-PRESSURE PNEUMATIC SYSTEM

The low-pressure pneumatic system (figure 4-1) obtains bleed air from the last compression stage of the high-pressure compressor. This engine bleed air is limited to 5.5% of total engine airflow but is normally much less than 1%. The bleed air varies in temperature and pressure up to approximately 800°F and 225 psi under extreme operational and climatic conditions. A portion of this air is passed through a refrigeration unit that includes an air-to-air heat exchanger and expansion turbine to reduce the temperature and pressure. The low-pressure pneumatic system is used to pressurize and air-condition the cockpit; to pressurize the fuel tanks and hydraulic system reservoirs; to pressurize the canopy seal and the anti-g suit; to supply operating pressure to the variable air pressure regulator in the elevator feel system; and to supply warm air for anti-icing, rain-clearing and defogging.

HIGH-PRESSURE PNEUMATIC SYSTEM

The high-pressure pneumatic system (figure 1-17) supplies pneumatic pressure for actuation of various system components. Air under pressure is stored in four spherical fiberglas flasks and in the main landing gear drag braces. The air storage flasks are mounted in the left and right armament bays. The system is serviced from an external source through a filler valve located on the forward side of the right main wheel well on some airplanes and the left main wheel well on other airplanes. System pressure is indicated on a pressure gage, located adjacent to the filler valve, and when fully serviced the system contains 3000 psi of compressed air. Pressure regulators and relief valves protect the system from excessive pressure. The pressure in the main landing gear drag braces is isolated by check valves to ensure adequate brake system pressure if the high-pressure pneumatic supply is depleted in flight. On some airplanes* a check valve isolates the priority air flask to provide full operating pressure for emergency landing gear extension, ram air turbine extension, and drag chute deployment. On other airplanes** a priority valve is installed between the priority and nonpriority air flasks in lieu of the check valve. The priority valve enables priority air to augment nonpriority air in the operation of all system components until pressure drops below 1400 psi in the priority air flask. Below 1400 psi, pressure in the priority flask is available for only ram air turbine extension, drag chute deployment, and landing gear extension. The pressure gage, located adjacent to the filler valve, will not indicate pressure through the entire system unless the system is fully charged. The pressure indicated at lower pressures in various air flasks will be dependent upon the various system configurations (see figure 1-17). For servicing requirements, see Servicing Diagram, figure 1-33.

PNEUMATICALLY OPERATED EQUIPMENT

See figures 4-1 and 1-17 for complete reference to pneumatically operated equipment.

Airplanes modified by TCTO 1F-102-847.

**AF 53-1791 thru 56-1518, unless modified by TCTO 1F-102-655.*
***AF 57-770 & on, & airplanes modified by TCTO 1F-102-655.*

PNEUMATIC PRESSURE-LOW WARNING LIGHT

The pneumatic pressure-low warning light (6, figure 1-26), located on the warning light panel, illuminates to display "PNEU PRESS." Operation of the warning light is dependent upon configuration of the high-pressure pneumatic system (see figure 1-17). On all airplanes, illumination of the warning light will indicate that pneumatic pressure in the priority air flask has dropped below 1500 psi. On those airplanes using a check valve to isolate the priority air flask*, at the moment of illumination of the warning light, enough high-pressure air remains in the priority flask for emergency landing gear extension, RAT extension, and drag chute deployment, and that pressure in the nonpriority flasks is less than 1500 psi. The check valve is designated to maintain full priority flask pressure and as long as there is 1500 psi, or more, in the nonpriority flasks, there will be no illumination of the warning light. When there is 1500 psi or more pressure in the priority flask, there will be no illumination of the warning light regardless of the pressure in the nonpriority flasks, and there is, therefore, no direct indication of the condition of the nonpriority air pressure. On other airplanes**, which use the priority valve to isolate the priority air flask, an equalizing flow between the priority and nonpriority flasks is permitted until pressure in the priority flask drops to 1400 psi, indicating that at the moment of warning light illumination, there is enough pressure in the priority flask for emergency operation of the landing gear, extension of the RAT, and deployment of the drag chute, and that 1400 psi remains in the system. When the light does not illuminate, in these airplanes, it will be an indication of satisfactory pressure in both the priority and the nonpriority flasks.

FLIGHT CONTROL SYSTEM

The flight control system (figures 1-18 and 1-19) provides desirable control of the airplane throughout the speed range. The delta wing configuration utilizes elevons instead of aileron and elevator control surfaces. The system incorporates standard stick and rudder pedals with conventional control action in response to stick movement. The elevons when moved coincidentally act as elevators and differentially as ailerons. This is accomplished by a mixer assembly which consists of a bellcrank assembly which rotates to provide aileron action and moves fore and aft to provide elevator action. To induce both aileron and elevator force, the elevons are large and extend almost the full span of the wing. Each elevon consists of an inboard and outboard panel which function as one unit permitting free surface movement unimpaired by normal in-flight wing deflections. Pitch and yaw damper systems are installed to provide a stable platform for armament firing and to stabilize high speed flight. Turn coordination is also furnished through the yaw damper system if the yaw damper is engaged.

**AF 53-1791 thru 56-1518, unless modified by TCTO 1F-102-655.*
***AF 57-770 & on, & airplanes modified by TCTO 1F-102-655.*

Turn coordination is supplied primarily for fire control system attacks and is sufficient for bank angles up to 35° and roll rates up to about 30° per second. Both the elevons and rudder are actuated by two complete, independent, simultaneously operating hydraulic systems. The ram air turbine can also supply pressure to one of these systems in event of emergencies. Movement of the stick and rudder pedals mechanically position hydraulic control valves, which direct primary and secondary hydraulic pressure to the respective control surface actuating cylinders. Control surface deflection is proportioned to cockpit control movement by followup linkages which shut off hydraulic pressure to the control surface actuators. Since aerodynamic force against the control surfaces is opposed by the hydraulic action, the forces are not transmitted to the cockpit controls. An artificial feel system is therefore required to simulate the aerodynamic forces encountered. The control surfaces are not equipped with trim tabs as trimming is accomplished by changing the neutral (no load) position of each control surface and is indicated by the no load cockpit control position. The design of the flight control system eliminates the necessity for surface gust locks except during storm conditions.

ARTIFICIAL FEEL SYSTEM

As the hydraulic portion of the control system is irreversible, the pilot does not have an indication of existing aerodynamic forces on the control surfaces. An artificial feel force, therefore, is added to the flight control system to produce feel on the control stick and rudder pedals relative to airspeed and altitude. Aileron feel is provided by a feel-centering spring, and the stick force is proportional to stick deflection only. At approximately ¾ aileron deflection (5°), on airplanes which have the enlarged vertical fin, an additional 10 pounds resistance to aileron movement is imposed by added spring tension and must be overcome to obtain full aileron. Elevator feel is provided by a centering spring and a variable feel force cylinder. A large piston in the elevator feel force cylinder is attached so that movement of the stick in either direction moves the piston against ram air pressure. This ram air pressure is controlled by an elevator intelligence unit which loads the correct amount of ram air pressure into the elevator feel force cylinder. The control surfaces are most effective in the transonic speed range. Therefore, the feel force cylinder pressure is highest in this area. The feel system applies a force which results in approximately a constant stick force per g throughout the flight envelope. The elevator intelligence unit is operated by low-pressure pneumatic system pressure and controls the ram air pressure from the large impact tube on the vertical fin. Rudder feel is also provided by a centering spring and feel force cylinder. A small piston in the rudder feel force cylinder is attached so that movement of either rudder pedal moves the piston against high-pressure pneumatic system pressure. This pressure is metered by a variable air pressure regulator controlled by ram air from the small

high-pressure pneumatic

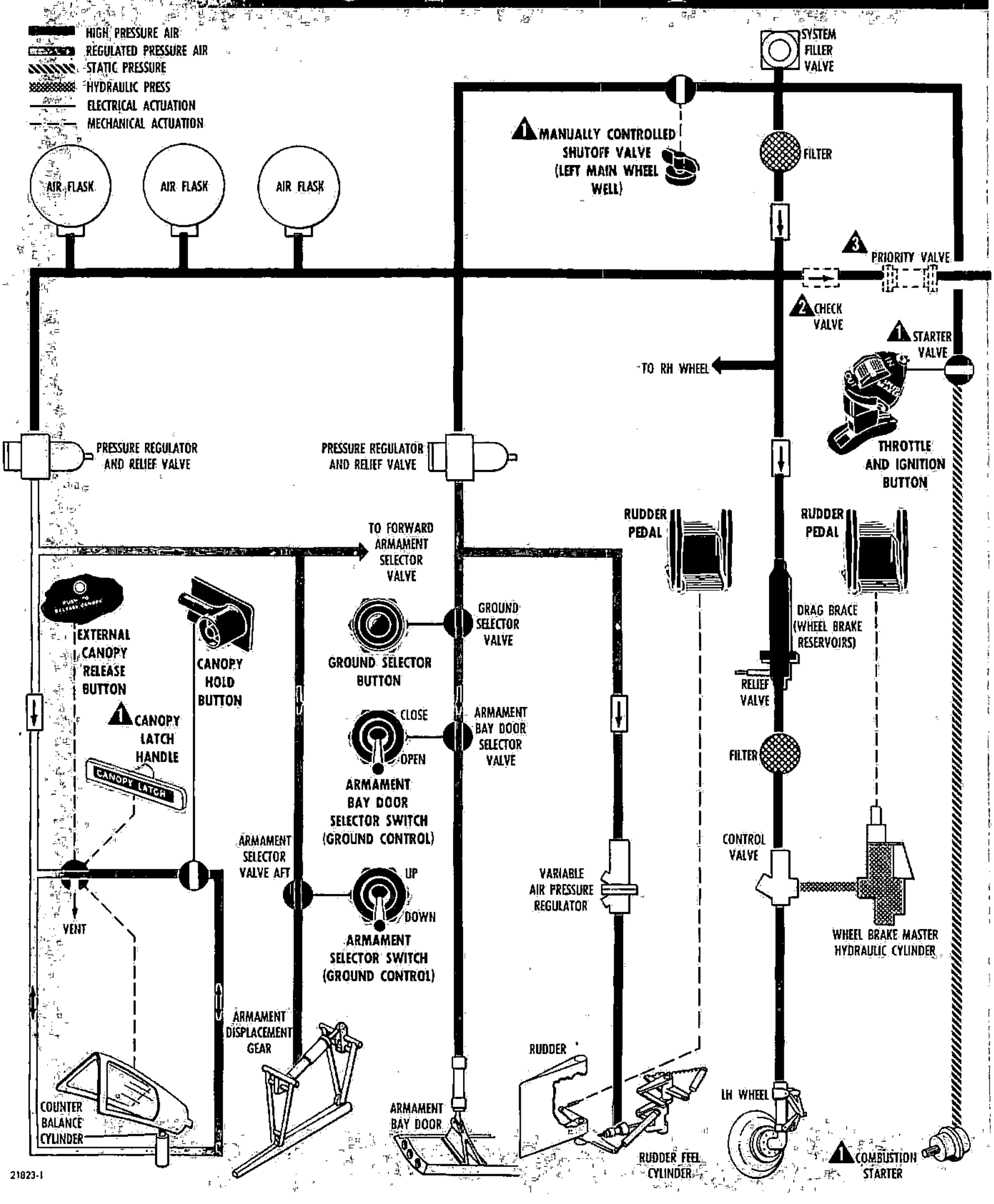
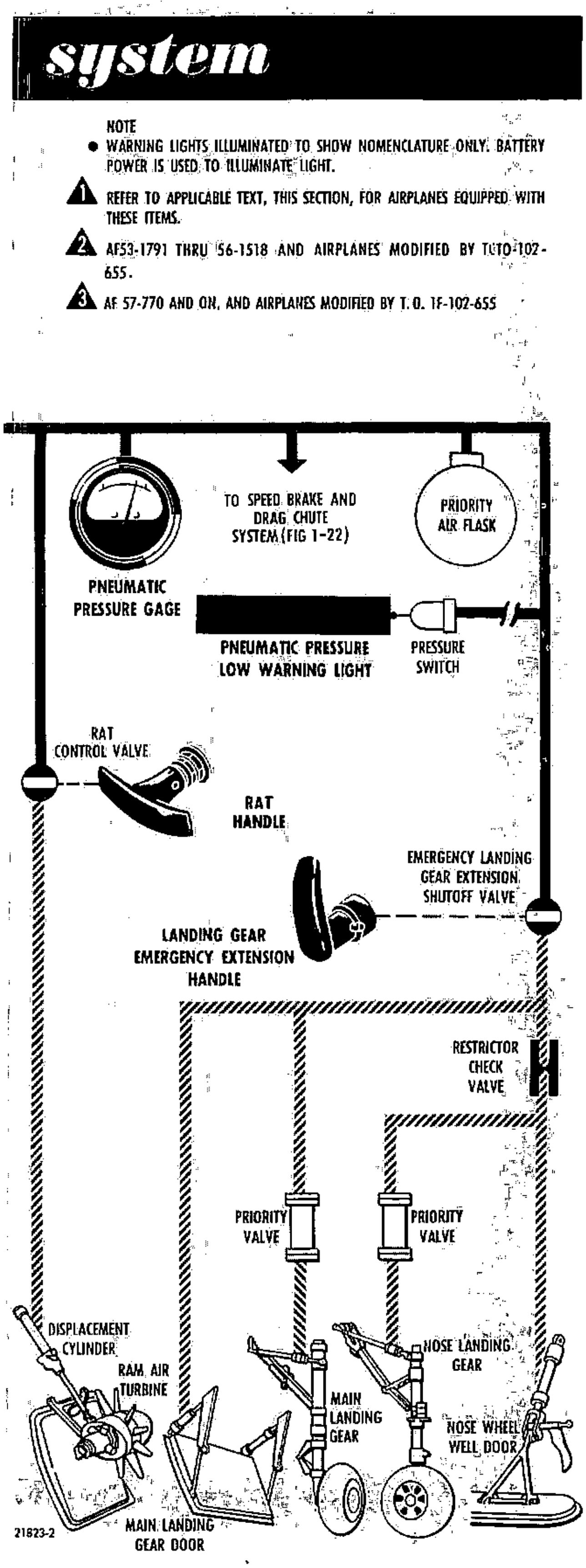


Figure 1-17



impact tube on the vertical fin. Refer to Section V for operating limitations.

TRIM SYSTEM

Trim is accomplished by deflecting the control surfaces with electrical trim actuators, installed between the feel mechanism and the mechanical control, which change the neutral (no load) position of the respective systems. Since the mechanical control system friction is less than the feel system forces, the cockpit controls and control surfaces will move in response to trim changes. Elevon (aileron and elevator action) trim is controlled by a switch on the control stick, and rudder trim is controlled by a switch on the utility switch panel or, on some airplanes, on the lower right-hand side of the throttle quadrant. Takeoff trim can be attained automatically by depressing a button, on the utility switch panel, which will reposition aileron, elevator, and rudder trim to a preset position. An indicator light illuminates when the proper takeoff trim positions are obtained. The trim system is powered from the 28-volt dc essential and nonessential busses and the 115-volt ac nonessential bus.

CAUTION

Do not operate the trim system (including takeoff trim) unless hydraulic pressure is being supplied to the flight control system, to prevent damage to the trim actuators.

PITCH AND YAW DAMPER SYSTEM

Pitch and yaw damper systems are installed to damp out short period oscillations of the airplane and to provide automatic turn coordination. The hydraulically actuated damper systems are electrically controlled. When the damper system is energized, secondary hydraulic pressure is supplied to the damper servo valve portion of the hydraulic elevon package (HEP valve) on some airplanes* or, on early airplanes, to the servo actuator (extendible link), and damper signals are imposed on pilot inputs to control surface positions. These elevons and rudder deflections are not felt at the pilot's controls. When the oscillations have been corrected, the control surfaces return to their original position. Rate gyros sense the direction and velocity of the oscillations and apply signals to the control surfaces to dampen the pitch or yaw oscillations. Aileron motion of the elevons and roll rate is measured and electrically controls the rudder servo actuator to provide automatic turn coordination when the yaw damper system is engaged. Such coordination is optimum only in level flight and deteriorates with variation in load factor. Turn coordination signals to the rudder are modified by an airspeed compensator to insure proper coordination at all airspeeds. Turn coordination is supplied primarily for fire control system attacks and is sufficient for bank angles up to

AF 56-973 & on.

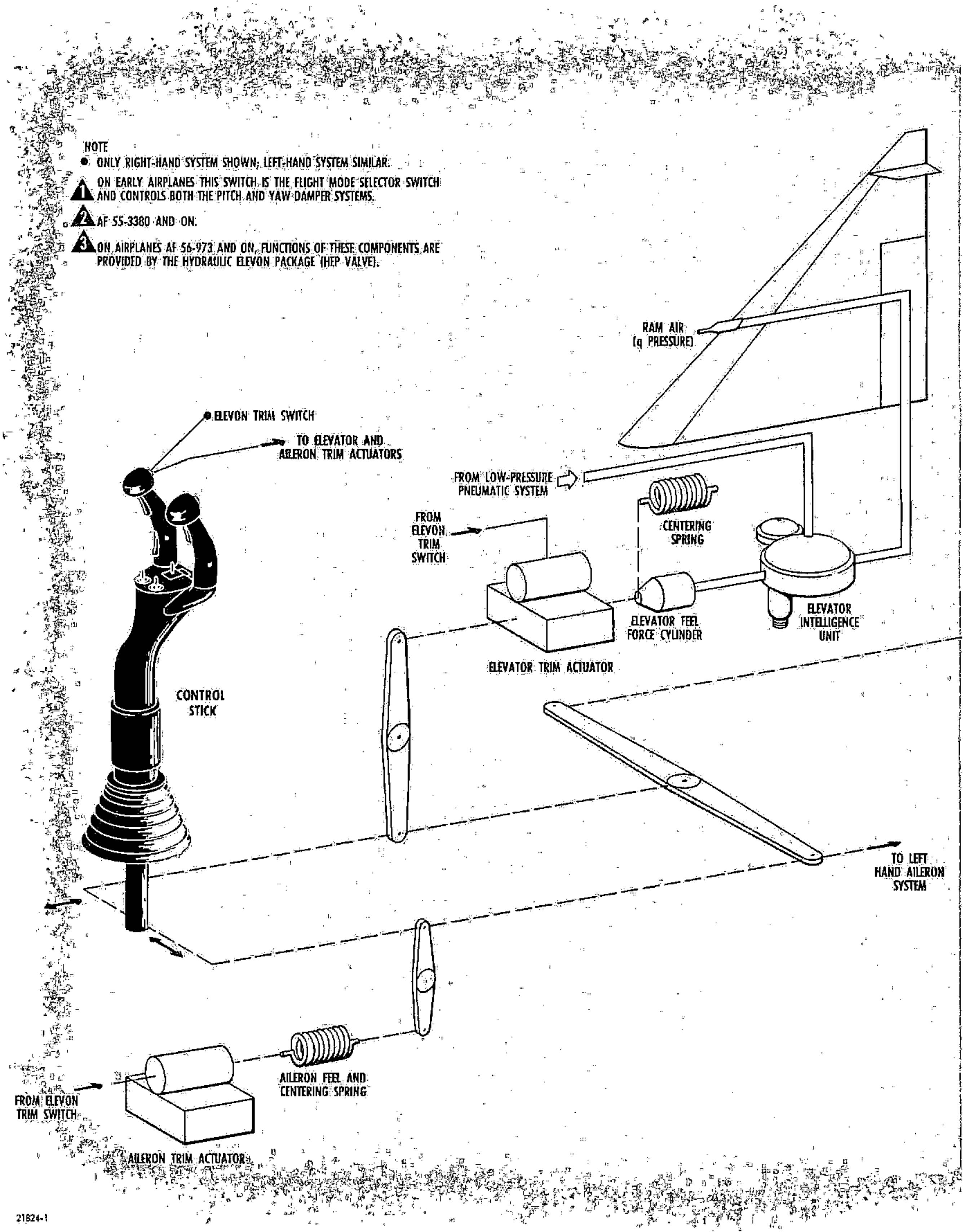
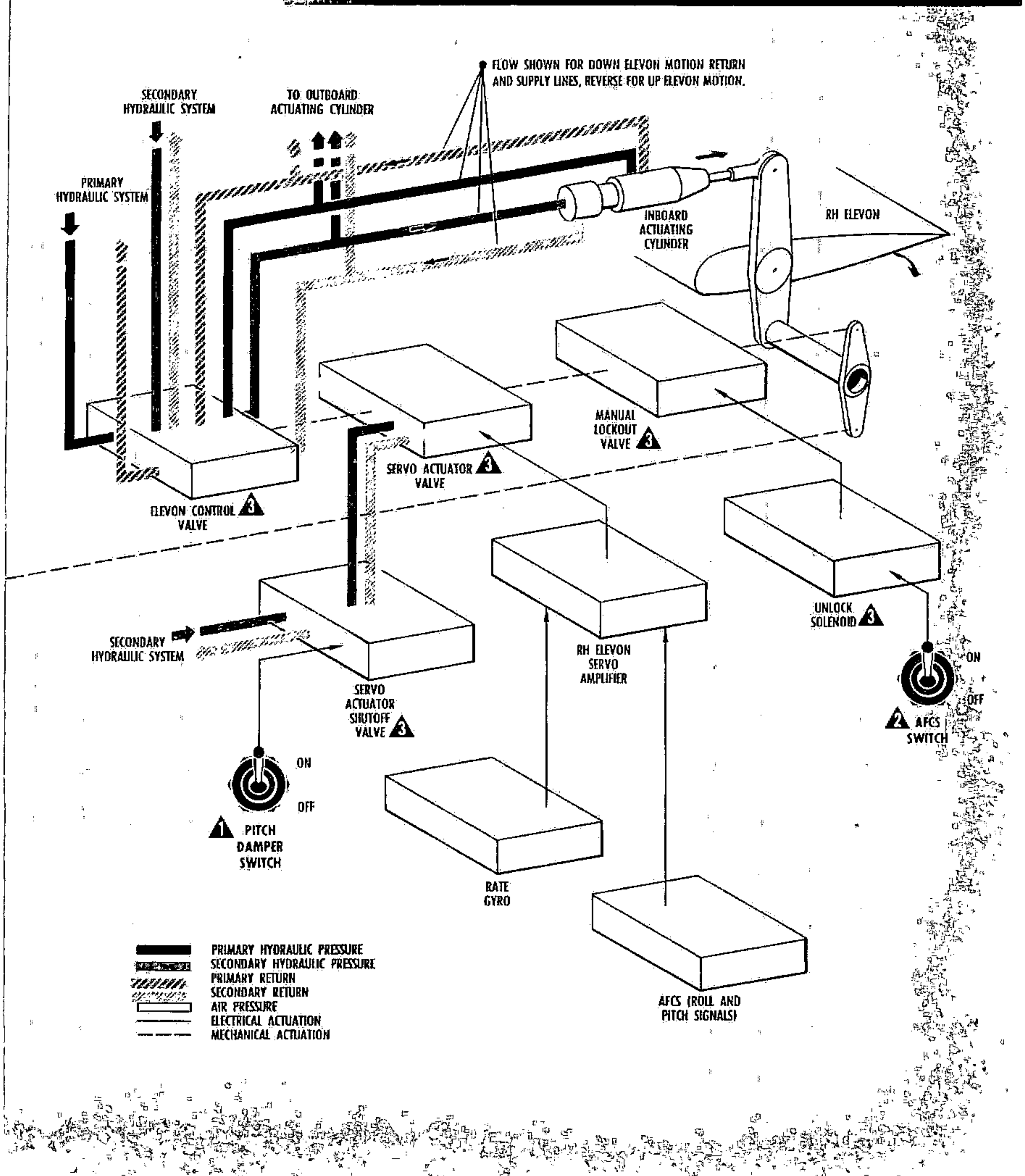

NOTE
● ONLY RIGHT-HAND SYSTEM SHOWN; LEFT-HAND SYSTEM SIMILAR.
1 ON EARLY AIRPLANES THIS SWITCH IS THE FLIGHT MODE SELECTOR SWITCH AND CONTROLS BOTH THE PITCH AND YAW DAMPER SYSTEMS.
2 AF 55-3380 AND ON.
3 ON AIRPLANES AF 56-973 AND ON, FUNCTIONS OF THESE COMPONENTS ARE PROVIDED BY THE HYDRAULIC ELEVON PACKAGE (HEP VALVE).
RAM AIR (q PRESSURE)
ELEVON TRIM SWITCH
TO ELEVATOR AND AILERON TRIM ACTUATORS
FROM LOW-PRESSURE PNEUMATIC SYSTEM
CENTERING SPRING
FROM ELEVON TRIM SWITCH
ELEVATOR FEEL FORCE CYLINDER
ELEVATOR INTELLIGENCE UNIT
ELEVATOR TRIM ACTUATOR
CONTROL STICK
TO LEFT HAND AILERON SYSTEM
AILERON FEEL AND CENTERING SPRING
FROM ELEVON TRIM SWITCH
AILERON TRIM ACTUATOR
21824-1

Figure 1-18

elevon control system



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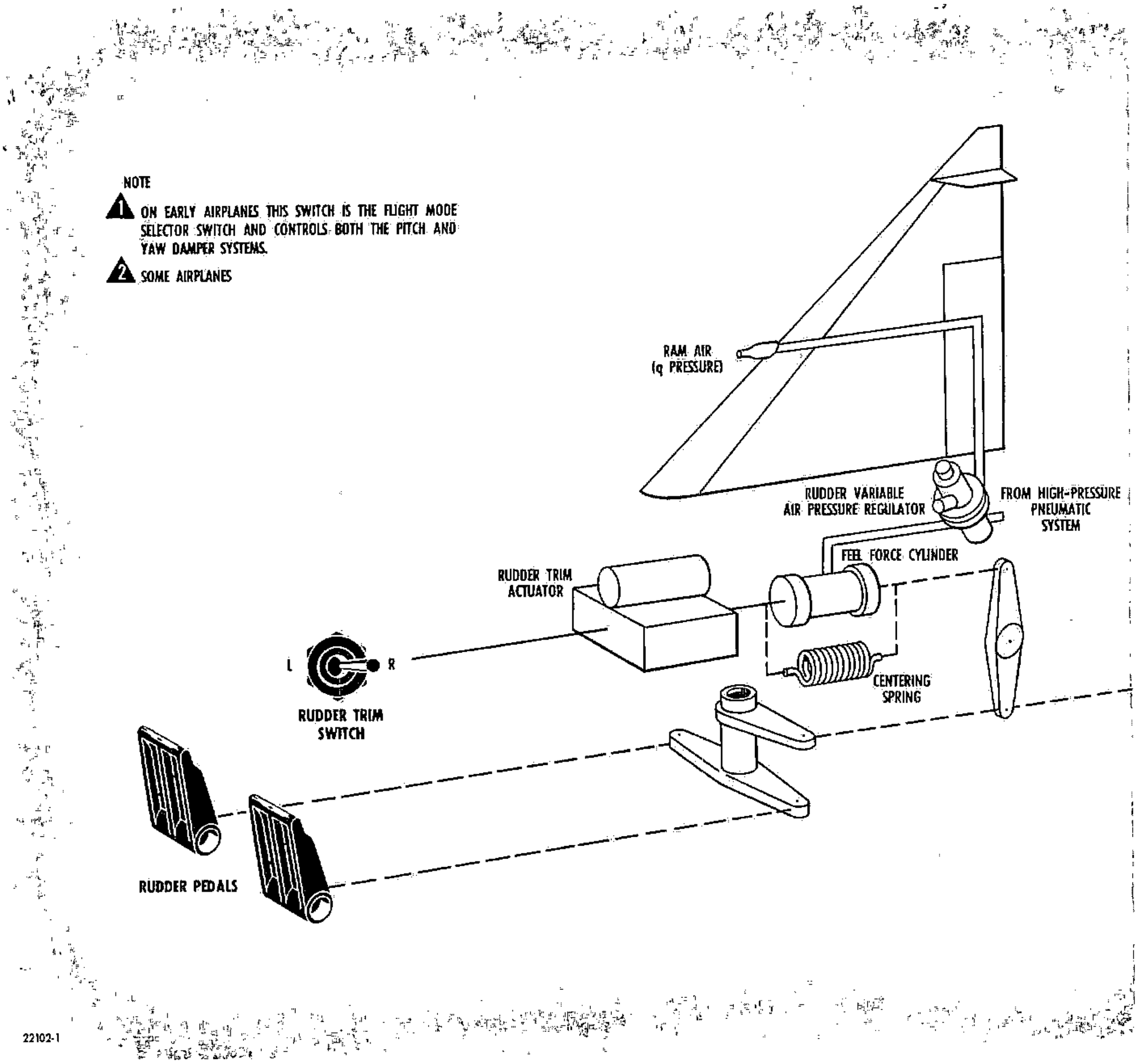


Figure 1-19

35° and roll rates up to 30° per second. The damper system receives power from the 200/115-volt ac nonessential bus and the 28-volt dc nonessential bus. Refer to Section V for operating limitations.

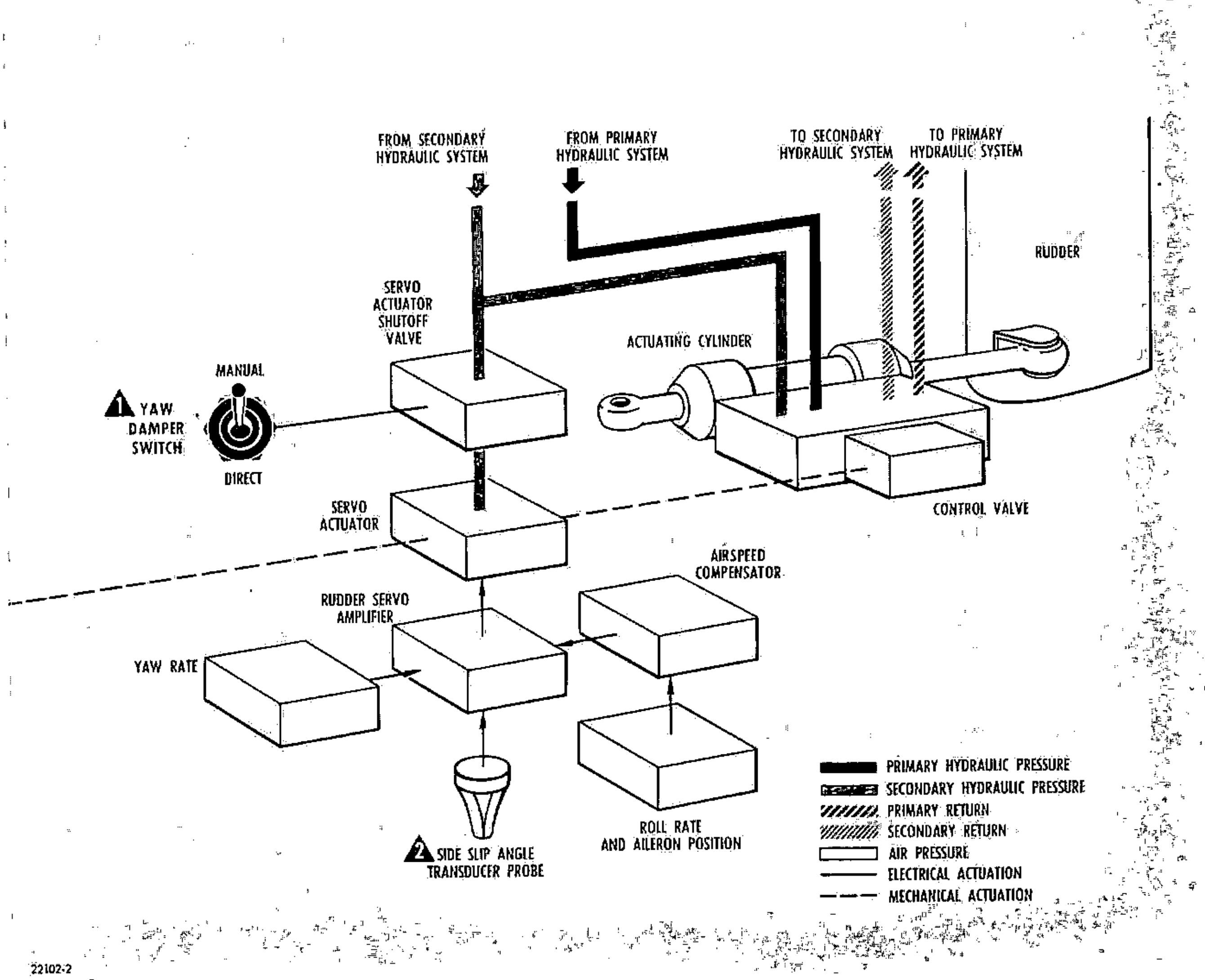
SIDESLIP ANGLE TRANSDUCER

A sideslip angle transducer is installed on early, small-tail airplanes to augment the static directional stability by applying corrective rudder against sideslip. The system incorporates a small detector probe which extends down from the fuselage centerline just forward of the nose wheel well. The detector probe contains two vertical slots which are approximately 90° apart and each positioned approximately 45° from the relative wind during flight. Any sideslip induced by the airplane is detected by this probe and the resultant signals are fed to the rudder servo actuator which initiates corrective action by rudder deflections. The detector probe is free to rotate and will align itself with the relative wind, well below takeoff speed. There are no separate cockpit controls for the sideslip angle transducer as the system is engaged by placing the flight mode selector switch in the MAN position. The sideslip angle transducer receives 200/115-volt ac nonessential and 28-volt dc power through the pitch and yaw damper system.

CONTROL STICK

The control stick (figure 1-20) controls the position of the elevon hydraulic control valves which direct primary

rudder control system



and secondary system hydraulic pressure to the actuating cylinders to displace the elevon control surfaces. Followup linkages then reposition the control valves and enable control surface deflection to be in proportion to stick movement. Conventional stick movements are used to obtain aileron and elevator action of the elevons. The control stick grips are composed of two grips mounted on a common base. The right-hand grip is the primary grip and incorporates a combination microphone and nose wheel steering button, elevon trim switch, emergency damper disconnect button, armament trigger and momentary interrupt trigger. The left-hand grip, when unlocked, serves as the radar antenna hand control and incorporates controls for the fire control system. Refer to FIRE CONTROL SYSTEM, Section IV.

RUDDER PEDALS

The rudder pedals control the position of the rudder hydraulic control valve which directs primary and secondary system hydraulic pressure to the rudder actuating cylinder. Followup linkage repositions the control valve which maintains rudder deflection in proportion to pedal movement. The rudder pedals can be simultaneously adjusted fore and aft by an adjustment crank mounted between the pedals. The wheel brakes are applied conventionally by toe action on the rudder pedals. Rudder pedal movement also controls nose wheel steering when the nose wheel steering button is depressed. Refer to WHEEL BRAKE SYSTEM and NOSE WHEEL STEERING SYSTEM, this Section.

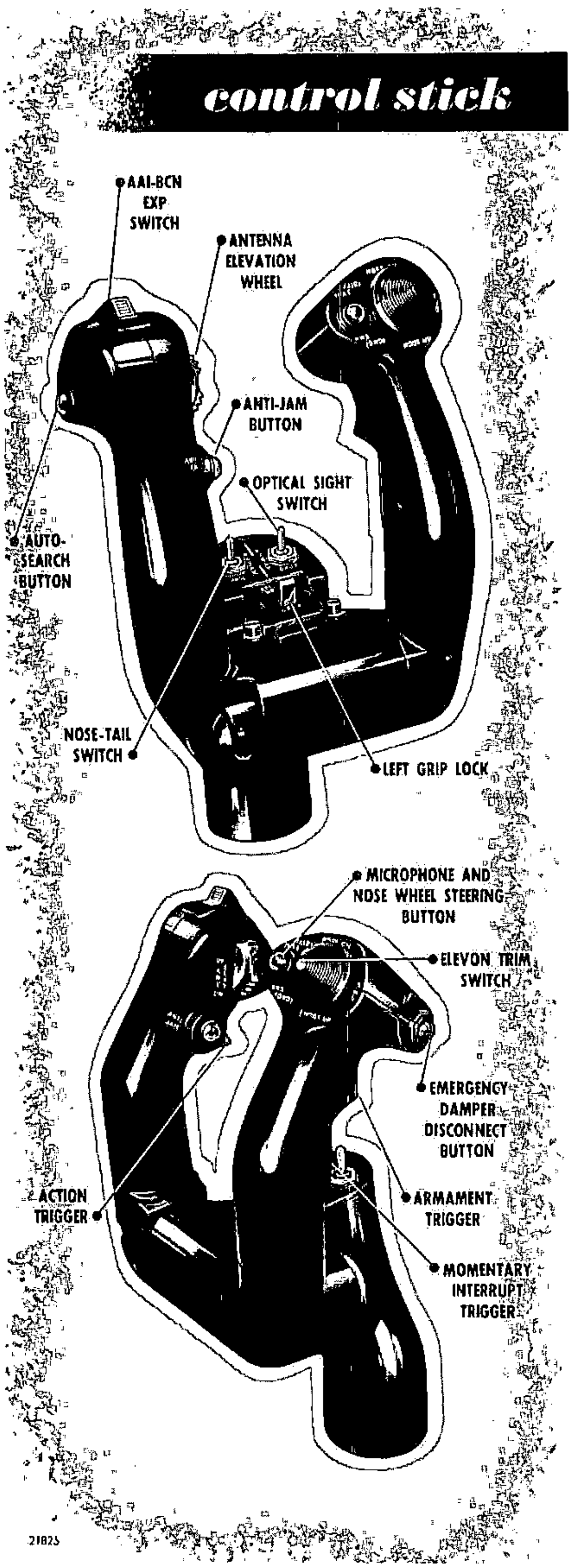


Figure 1-20

ELEVON TRIM SWITCH

Lateral and longitudinal trim is controlled by a five-position elevon trim switch (figure 1-20), located on the right-hand control stick grip. The switch has NOSE UP, NOSE DOWN, LWD, RWD and is spring-loaded to the center (OFF) position. Holding the switch in the direction of desired trim powers the ailerons or elevator trim actuator to reposition the neutral (no load) position of the respective flight control system. When released the switch automatically returns to its spring-loaded center (OFF) position. The trim switch receives power from the 28-volt dc essential bus.

CAUTION

Should the trim switch stick in an actuated position, an extreme application of trim will result. If this condition exists on preflight inspection, the airplane should not be flown until the condition is corrected. If the switch sticks in flight, it will be necessary to return to the center (OFF) position manually after the desired trim change is made. A sticking trim switch should be noted on Form 781 with a red cross.

RUDDER TRIM SWITCH

The three-position rudder trim switch (figure 1-21), located on the utility switch panel on some airplanes or on the lower right-hand side of the throttle quadrant on other airplanes, controls the position of the rudder trim actuator to reposition the neutral (no load) position of the rudder control system. The switch is placarded "Rudder Trim" and has positions L, R, and is spring-loaded to the center (OFF) position. Holding the switch to L or R applies trim in the respective direction. When released the switch automatically returns to its spring-loaded center (OFF) position. The trim switch receives power from the 28-volt dc essential bus.

TAKEOFF TRIM BUTTON AND INDICATOR LIGHT

The takeoff trim button (figure 1-21), located on the utility switch panel, when depressed, trims all control surfaces to the proper position for takeoff. The takeoff trim position of the aileron action and rudder is neutral, and the elevator action is 5° nose up. While the button is held depressed, a green indicator light illuminates when the trim system reaches the proper position for takeoff. This light displays "TAKEOFF TRIMMED" when illuminated and is automatically dimmed when the instrument panel lights are on if the thunderstorm lights are off. The takeoff trim circuit is deenergized when the nose landing gear is retracted and receives power from the 28-volt dc nonessential bus.

TRIM SERVO SWITCH

Note

If the trim servo switch is installed, it is deactivated.

FLIGHT MODE SELECTOR SWITCH

On some airplanes, a two-position flight mode selector switch* is used to engage the pitch and yaw damper system and the sideslip angle transducer (if installed). The switch, located on the utility switch panel, is placarded "Flight Mode" and has MAN and DIRECT MAN positions. The switch is spring-loaded to DIRECT MAN and utilizes a solenoid to hold the switch in MAN position. Damper system components are energized whenever power is on the airplane (nonessential buses), and placing the switch to MAN position supplies hydraulic pressure to the damper system servo (extendible link) actuators which engage the system. The MAN position also energizes the sideslip angle transducer, which is a part of the yaw damper system. The damper system will automatically disengage (switch returns to DIRECT MAN position) when the emergency damper disconnect button (on the control stick grip) is depressed, or the switch may be manually placed to DIRECT MAN position to disengage the system. The flight mode selector switch receives power from the 28-volt dc nonessential bus.

YAW DAMPER SWITCH

A yaw damper switch**, located in the lower left-hand corner of the AFCS panel (figure 4-16), is used to engage the yaw damper system. On early airplanes, the switch has a spring-loaded DIRECT (off) position to disconnect the yaw damper and a solenoid held MANUAL (on) position which engages the damper. On later airplanes, the switch positions are YAW DAMPER and OFF. Damper system components are energized when the nonessential buses are supplied with power and activated when the switch is placed in the MANUAL or YAW DAMPER position. Hydraulic pressure is supplied to the rudder damper system servo actuators to damp yaw oscillations and provide turn coordination. The yaw damper system will automatically disengage (the switch will return to DIRECT or off position) when the emergency damper disconnect button is depressed or the switch may be manually placed to DIRECT or off position to disengage the system. The yaw damper switch receives power from the 28-volt dc nonessential bus.

PITCH DAMPER SWITCH

A pitch damper switch** (figure 1-21), located on the utility switch panel, is used to engage the pitch damper system. The switch is placarded "Pitch Damper" and has

*AF 53-1791 thru 55-3379 unless modified by TCTO 1F-102A-546.
**AF 55-3380 & on, & airplanes modified by TCTO 1F-102-546.

utility switch panel (typical)

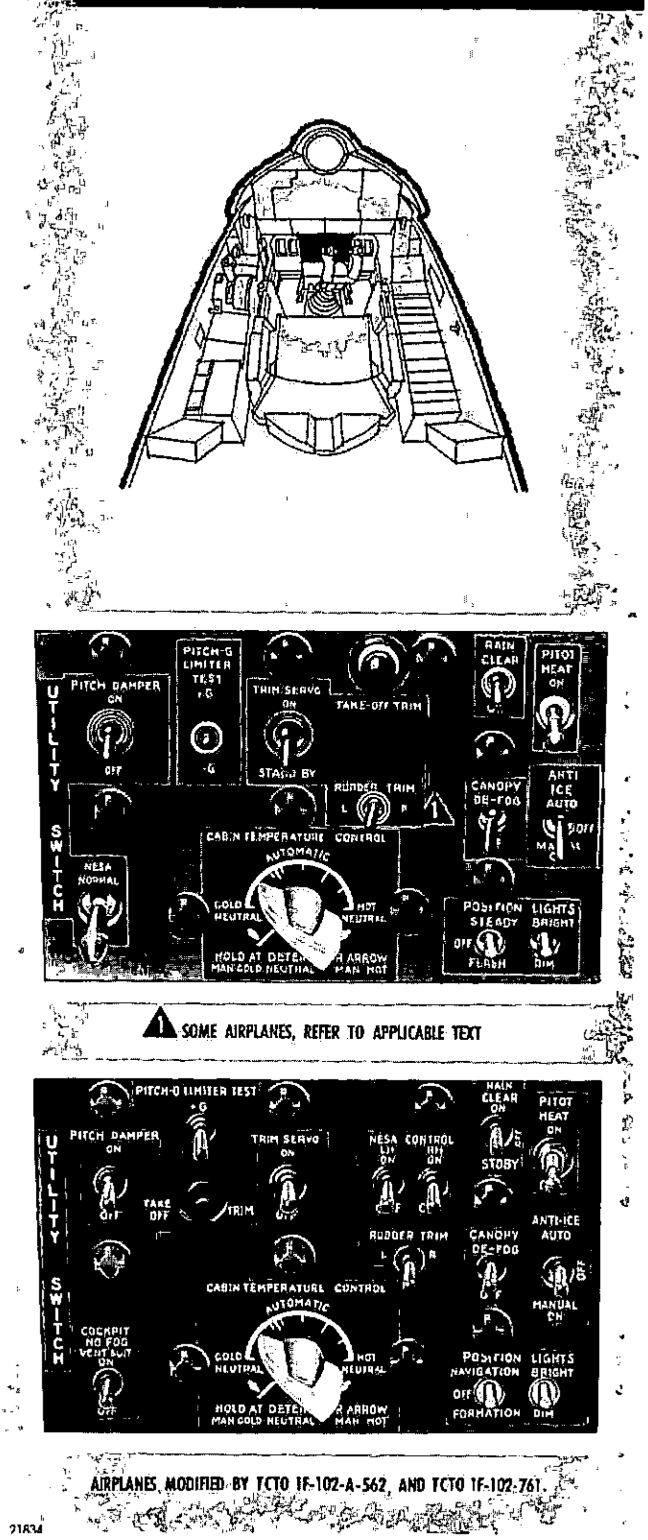


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Figure 1-21

a spring-loaded OFF position and a solenoid-held ON position. Pitch damper system components are energized when power is on the nonessential buses. Placing the switch in the ON position supplies hydraulic pressure to the elevon damper system servo actuators to activate the system.

Note

The yaw damper switch must be in MANUAL (on) position prior to placing the pitch damper switch to ON. When the yaw damper switch is in DIRECT (off), power is removed from the holding solenoid of the pitch damper switch and the switch will not engage in the ON position.

When the automatic flight control system is in operation and the preset pitch g limits are exceeded, the pitch damper switch will automatically go to the OFF position, disengaging the system. The pitch damper system will also automatically disengage (the switch will return to the OFF position) when the emergency damper disconnect button is depressed or the switch may be manually placed to OFF to disengage the system. The pitch damper switch receives power from the 28-volt dc nonessential bus.

EMERGENCY DAMPER DISCONNECT BUTTON

The emergency damper disconnect button placarded "Emer Man" (figure 1-20), located on the control stick right-hand grip, is used to simultaneously disengage the damper systems and AFCS. When the button is depressed, the circuit is broken to the holding solenoids for the flight mode selector and AFCS switches which automatically return to DIRECT MAN and OFF positions, respectively.

MOMENTARY INTERRUPT (MANUAL MODE) TRIGGER

The momentary interrupt trigger (figure 1-20), located on the control stick right-hand grip, is used to momentarily disengage the AFCS. This trigger should be used to disengage the AFCS to make large attitude or heading changes. Releasing the trigger will re-engage the AFCS to hold the attitude or heading prevailing at the time of release. The trigger when depressed interrupts power to a phase of the AFCS.

SPEED BRAKES SYSTEM

Two hydraulically operated, electrically controlled speed brakes (figure 1-22) are located above the tail cone and can be used to slow the airplane at all speeds. A relief valve in the speed brakes hydraulic system allows the speed brakes to retract, as necessary, to prevent structural damage under excessive aerodynamic loads. When extending the speed brakes, a slight nose-up trim change occurs, and when retracting the speed brakes a nose-down trim change occurs. Secondary hydraulic system pressure is supplied through a dc electrically operated selector valve to actuate a pair of hydraulic cylinders for each brake. The speed brakes are synchronized to give equal angular operation and require approximately 2 seconds to open or close. The speed brakes also serve as compartment doors for the drag chute, and when the drag chute is deployed the brakes cannot be closed until the chute is jettisoned. The speed brakes are controlled by a switch located on the throttle and are opened automatically when the drag chute handle is pulled. A speed brakes emergency opening system is incorporated and is to be used in the event of secondary hydraulic system failure to allow the drag chute to deploy. The emergency system bypasses the speed brakes switch ~~on the throttle and furnishes high-pressure pneumatic air to the speed brakes actuating cylinders~~ when the drag chute handle is pulled out (full travel), ~~then rotated~~ 90° right and pulled again.

CAUTION

- Emergency speed brakes extension should be used only in event of secondary hydraulic system failure. Use of the system when the secondary hydraulic system is pressurized and operating will not aid drag chute deployment and will result in pumping hydraulic fluid through a vent into the vertical fin structure on some airplanes. This could result in hydraulic fluid draining onto the tailpipe.
- Emergency speed brakes extension is designed for use only on the landing roll (or aborted takeoff roll) when drag chute deployment is desired. Inflight use of this system will result in loss of the drag chute when above 160 KIAS.

The speed brakes cannot be opened or closed without dc power, either by the normal or emergency systems.

SPEED BRAKES GROUND SAFETY LOCKS

Ground maintenance safety locks (figure 1-23) may be installed on the speed brakes actuating cylinders when the brakes are extended, primarily, during repacking of the drag chute and must be removed before flight.

SPEED BRAKES SWITCH

The three-position speed brakes switch (figure 1-7), located on top of the throttle, is used to control speed brake operation. The switch is placarded "Speed Brakes" and has fixed positions of IN, OUT, and a center (neutral) position which controls the selector valve accordingly. The center (neutral) position is indicated by a white alignment mark on the switch guide. When the switch is in the center position, the control valve is closed and the speed brakes are held in the selected position. The speed brakes switch receives power from the 28-volt dc essential bus.

speed brake and drag chute systems

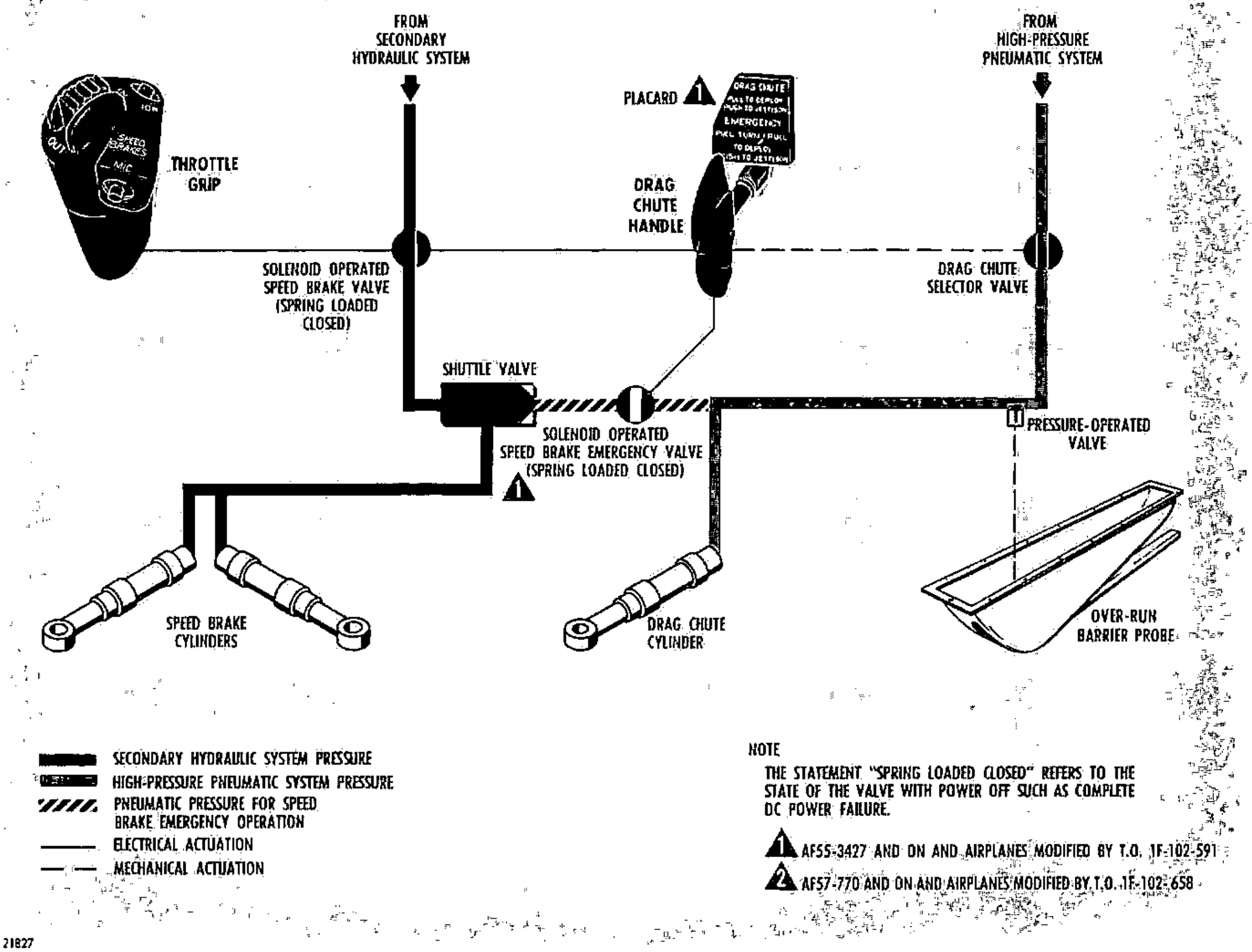


Figure 1-22

CAUTION

Damage can result to the speed brakes actuators or their electrical conduits if the speed brakes are closed after the drag chute has been jettisoned, or if the doors are closed with drag chute not installed.

LANDING GEAR SYSTEM

The tricycle landing gear and wheel well doors are electrically controlled and sequenced, and hydraulically actuated. The main gear retracts inboard into the lower surface of the wing and fuselage, and the nose gear retracts forward into the fuselage. The wheel well doors remain open when the gear is extended and fair the gear flush with the airplane contour when the gear is retracted. The main landing gear fairings are mechanically tied to the strut assembly and are actuated with gear movement. The nose gear drag brace contains a combination up- and down-lock and is unlocked by initial travel of the nose gear actuating cylinder. The main gear is locked up by the wheel well doors, and the down-lock is unlocked by initial travel of the gear actuating cylinder. Safety switches preclude normal gear retraction while the airplane is on the ground. However, an override control bypasses the safety switches and the normal landing gear handle to permit emergency gear retraction while on the ground or while airborne. Normal gear extension, retraction, and emergency retraction are actuated by secondary hydraulic system pressure. In the event of electrical or hydraulic system failure the gear can be extended by

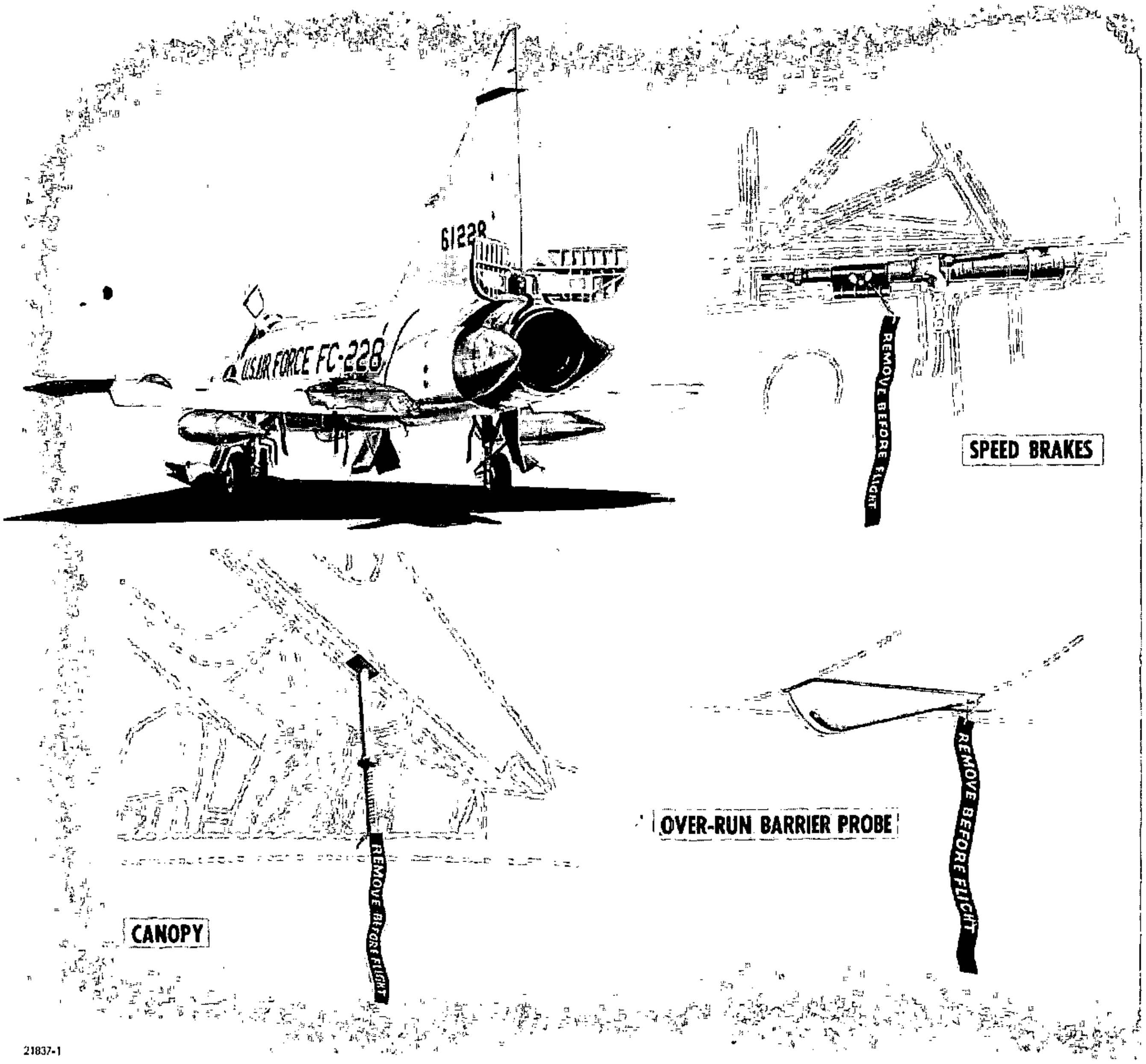


Figure 1-23

high-pressure pneumatic system pressure which is routed to the normal hydraulic actuating cylinders. During normal operation, gear extension and retraction time is approximately four to six seconds. A hydraulic steering unit is built into the nose gear assembly to provide nose wheel steering and also serve as a conventional shimmy damper. The main wheels are equipped with pneumatically operated multiple-disc type brakes. The main gear drag braces serve as pneumatic pressure reservoirs for the wheel brake system. Later airplanes* are equipped with a main landing gear which, when extended, is tilted forward at a slight angle. Therefore, while the airplane is on the ground the main gear is farther forward in relation to the airplane cg, which improves landing characteristics.

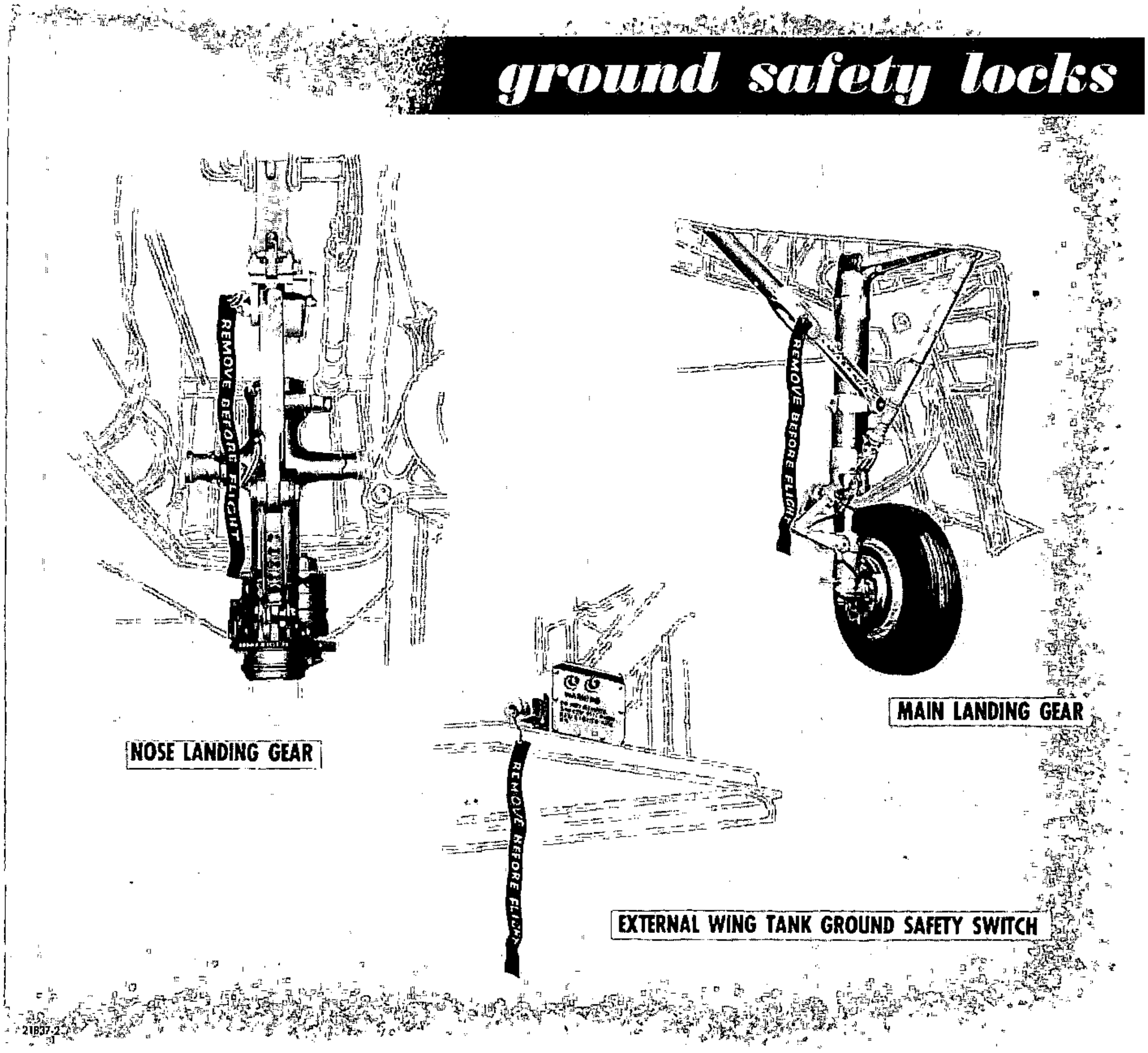
LANDING GEAR GROUND SAFETY LOCKS

Removable ground safety locks (figure 1-23) may be installed in the landing gear assemblies to prevent collapsing of the gear while the airplane is on the ground. The locks are equipped with warning streamers and must be removed before flight.

LANDING GEAR HANDLE

The landing gear handle (figure 1-24), located on the left-hand auxiliary instrument panel, electrically controls normal operation of the gear and wheel well door hydraulic selector valves. When the airplane is airborne, moving the handle to the UP position energizes the hydraulic selector valves to apply secondary hydraulic

**AF 53-1812 & on.*



system pressure to retract the gear. When the main and nose gear are fully up, the door selector valves are positioned to close the doors. When the doors are closed and locked the gear actuating system is automatically depressurized.

Note

When the weight of the airplane is on the gear, ground safety switches prevent gear retraction if the handle is inadvertently moved to UP.

When the landing gear handle is moved to the DOWN position, the hydraulic selector valves are energized to allow hydraulic pressure to unlock and open the doors, then extend the gear. Hydraulic pressure is maintained on the gear and doors when extended. The knob on the landing gear handle is wheel-shaped and contains the landing gear warning light. The landing gear circuit is powered by the 28-volt dc essential bus.

LANDING GEAR EMERGENCY-UP BUTTON

The landing gear emergency-up button is located on the left-hand auxiliary instrument panel (figure 1-24). The button is placarded "Emer Gear Up" and can be used to retract the landing gear when the airplane is in the air or moving on the ground. On the ground, depressing the button bypasses the ground safety switches and the gear will retract if the normal landing gear handle is in the UP position and the airplane is moving. When airborne, depressing the button will allow the landing gear control circuit to bypass the landing gear handle and

landing gear controls

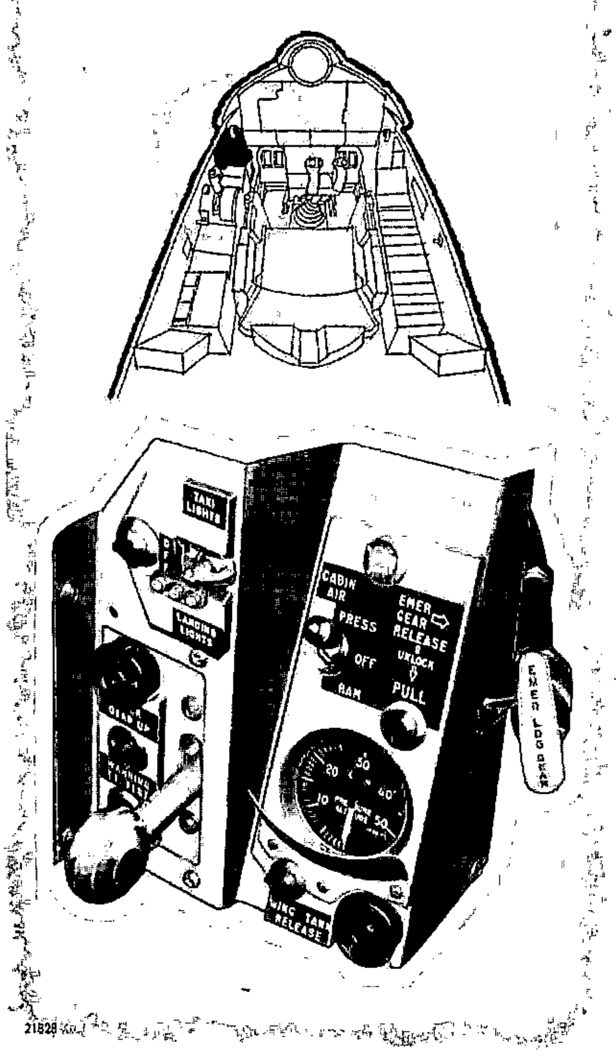


Figure 1-24

retract the gear with the normal control handle in the UP or DOWN position. To insure positive operation after take-off, the gear emergency-up button must be held depressed until the gear completely retracts. If the gear is retracted with the button while airborne with the gear handle in the DOWN position, it will be necessary to cycle the gear handle to the UP position and return to DOWN position to extend the gear. After emergency gear extension has been accomplished, the emergency-up button will be inoperative until the landing gear emergency extension handle is pushed in. In the event of ground safety switch malfunction, the emergency-up button may be used to retract the gear when airborne. The emergency-up button receives power from the 28-volt dc essential bus.

LANDING GEAR EMERGENCY EXTENSION HANDLE

The landing gear emergency extension handle (figure 1-24), suspended below the left side of the instrument panel, is used to pneumatically extend the landing gear in the event of secondary hydraulic system failure or electrical system failure or malfunction. The handle is placarded "Emer Gear Release-Pull" on early airplanes* and "Emer Gear Release-Unlock-Pull" on later airplanes**. An arrow indicates the necessity to pull down and out on later airplanes. Pulling the handle out fully (approximately two inches) mechanically opens a pneumatic shutoff valve to supply high-pressure pneumatic system air to the wheel well door and landing gear actuating cylinders which will open the doors and extend the gear. A spring clip is installed on early airplanes at the base of the handle to lock the handle in the fully extended position. On later airplanes the handle locks automatically in the fully extended position. The emergency extension handle will extend the landing gear regardless of the position of the normal landing gear handle. There are no provisions for retracting the gear after extension by the emergency systems.

LANDING GEAR POSITION INDICATORS

Three landing gear position indicators (5, figure 1-4), located on the left side of the instrument panel, show the position of the main and nose landing gear. When one of the indicators reads "UP," the respective gear and door are up and locked. Each indicator displays a symbolized wheel when the respective gear is down and locked. When there is no electrical power or the gear is in an unlocked position, the indicator displays parallel red and yellow stripes. On some airplanes† the landing gear position indicators are replaced by three green lights. These lights illuminate only when the corresponding gear is down and locked. Power to the landing gear indicators and lights is supplied by the 28-volt dc essential bus.

LANDING GEAR WARNING LIGHT AND TEST BUTTON

A red warning light, located within the wheel-shaped knob on the landing gear handle (figure 1-24) will illuminate at any time the landing gear is not in the position selected by the landing gear handle. The light will also illuminate when the landing gear is not down and locked at an altitude of 13,500 (±1000) feet climbing or 9500 (±1000) feet descending, if the throttle is retarded below FULL MIL POWER position, and the airspeed is less than 250 (±14) KIAS on some airplanes and 210 (±10) KIAS on other airplanes‡. The warning light is automatically dimmed when the instrument panel lights are on if the thunderstorm lights are off. On airplanes that have the three green landing gear indicator lights†, a

*AF 53-1791 thru 56-972.
**AF 56-973 & on.
†Airplanes modified by TCTO 1F-102-728.
‡AF 57-770 & on.

red gear unsafe warning bar light has been added. The bar light has a press-to-dim feature and when dimmed, the landing gear handle warning light will also dim. The warning bar light is wired into the circuit with the landing gear control handle warning light and both will illuminate under the above conditions. On all airplanes the gear unsafe warning light(s) can be checked by depressing the test button located on the landing gear control panel. An audible warning signal is also provided to give an audible signal in the radio headset at any time the landing gear warning lights illuminate. The light(s) receive power from the 28-volt dc essential bus.

CAUTION

On airplanes that have the red gear unsafe warning bar light, do not depress the press-to-test button on the landing gear control panel and the red warning bar light at the same time. This may result in damage to the audible signal generator.

NOSE WHEEL STEERING SYSTEM

The nose wheel steering system is provided for directional control during taxiing and for portions of the takeoff and landing roll, as desired. The system is electrically engaged, controlled by the rudder pedals, and powered by secondary hydraulic system pressure. Steering is engaged by a button on the control stick grip. The hydraulically powered nose wheel steering unit will position the nose wheel within approximately 50° each side of center, when the airplane is on the ground.

Note

Nose wheel steering is inoperative when the landing gear is extended by the emergency extension system.

A mechanically operated valve and centering cam automatically depressurizes the steering unit and centers the nose wheel as the gear retracts. The nose wheel steering system is irreversible which prevents forces applied to the nose wheel from being transmitted to the rudder pedals. When the system is not engaged or has turned in excess of 50° from center the nose wheel is free to swivel. The steering unit also serves as a conventional shimmy damper up to 50° either side of center.

NOSE WHEEL STEERING UNIT GROUND LOCK PIN

The nose wheel steering unit ground lock pin may be installed in the steering unit to facilitate jacking of the nose wheel or to provide stability of the nose unit when mooring the airplane. This lock pin must be removed before flight.

NOSE WHEEL STEERING BUTTON

The nose wheel steering button (figure 1-20), located on the control stick right-hand grip, engages the nose wheel steering system. The button is placarded "Mic (Air)-Nws (Gnd)." When the button is depressed a solenoid operated valve allows secondary hydraulic system pressure to engage the steering unit which is then controlled by the rudder pedals. If the rudder pedals are displaced and the nose wheel is centered (as during cross-wind landing roll) it is necessary to neutralize the rudder pedals to obtain control of the steering unit. Likewise, if the nose wheel is not centered when the button is depressed, the pedals must be positioned in relationship to the nose wheel before steering will engage. On some airplanes, once engaged, nose wheel steering is available on the ground as long as the button is held depressed and the wheel does not exceed 50° from center in either direction. On other airplanes* a relay is installed enabling the pilot to actuate the nose wheel steering by momentarily depressing the button. This eliminates the requirement of the pilot holding the button depressed to keep the system engaged. Depressing the button again will subsequently disengage the system.

Note

The nose wheel steering button functions as a secondary microphone control when the weight of the airplane is off the nose gear, or the nose wheel exceeds 50° from center in either direction.

The nose wheel steering button receives power from the 28-volt dc essential bus.

WHEEL BRAKE SYSTEM

The multiple-disc type, pneumatically operated brakes are installed on the inboard side of the main wheels. Conventional toe action on the rudder pedals individually applies independent hydraulic pressure to control the position of spring-loaded metering valves which, in turn, control pneumatic pressure to actuate the brakes. The main landing gear drag braces serve as pneumatic reservoirs for the brake system and are connected to the high-pressure pneumatic system through check valves which maintain braking pressure in the event high-pressure pneumatic system pressure is depleted.

Note

There is no method of checking brake system pneumatic pressure in flight. As the brakes are hydraulically controlled and pneumatically actuated, the feel of pressure in the rudder pedals is not a definite indication that pneumatic pressure is available to the brakes. However, pneumatic brake system pressure should always be equal to or greater than high-pressure pneumatic system pressure.

**AF 56-1430 & on, & airplanes modified by TCTO 1F-102A-557.*

Relief valves are installed to protect the brake system against excessive pressure. No emergency, antiskid, or parking brake system is provided.

DRAG CHUTE AND OVERRUN BARRIER PROBE SYSTEM

A drag chute is provided to reduce landing roll distance and is designed to be used after touchdown. The ringslot type parachute, packed in a deployment bag, is stowed in a compartment below the rudder. The speed brakes serve as compartment doors. The drag chute is deployed and jettisoned by a drag chute control handle which supplies penumatic pressure to operate the chute deployment mechanism. Protective strips are also added to the chute risers to prevent heat from aft section of the airplane from melting the risers. The chute pack is secured to the airplane to prevent deployment in flight when the speed brakes are opened. Should this feature fail and the chute accidentally deploy in flight (without pulling the drag chute control handle) the deployment mechanism will release the entire chute assembly from the airplane.

Note

- If the speed brakes are closed and dc essential bus power is available, secondary hydraulic system pressure, or in the event of secondary hydraulic system pressure failure, high-pressure pneumatic system pressure will open the brakes for drag chute deployment.
- Pulling the drag chute handle will open the speed brakes regardless of the speed brakes switch position unless the jettison pin is dislodged or a malfunction exists in the speed brakes circuit. If speed brakes are open, dc power is not required for drag chute deployment.

The chute mechanism incorporates a shear pin to prevent structural damage if the chute is deployed above approximately 160 KIAS. On some airplanes an extendible overrun barrier probe is installed and will extend simultaneously with the drag chute activation.

DRAG CHUTE HANDLE

The drag chute handle (2, figure 1-4), located to the left of the instrument panel, is formed to resemble a parachute and is used to deploy and jettison the drag chute. Pulling the handle fully straight out (approximately 1.5 inches) actuates a switch that electrically controls the speed brakes selector valve which applies secondary hydraulic system pressure to open the speed brakes. Simultaneously, the handle mechanically operates a selector valve to supply high-pressure pneumatic system pressure to the chute deployment mechanism securing the chute risers and pulling the ripcord pins which deploys the chute. The drag chute handle controls high-pressure pneumatic system air to both the speed brakes and the drag chute and is used in the event of secondary hydraulic system failure to operate the speed brakes for drag chute deployment. Pulling the drag chute handle fully out, rotating 90° to the right, then out again, supplies 28-volt dc power to a solenoid operated pneumatic selector valve which supplies high-pressure pneumatic air to the speed brakes cylinders. Mechanical sequencing prevents release of the chute until the speed brakes have opened sufficiently to clear the chute as it deploys. Pushing the drag chute fully in releases pneumatic pressure to the deployment mechanism which jettisons the chute. Speed brakes cannot close until the chute is jettisoned. If the speed brakes switch is IN when the chute is jettisoned the speed brakes will close; otherwise they will remain in the extended position.

CAUTION

- To prevent loss of drag chute or to prevent chute from slowing airplane to excessively slow speeds, do not deploy drag chute in flight.
- Do not jettison the drag chute unless the speed brakes switch is in the neutral position. This will prevent venting hydraulic fluid from the secondary hydraulic system if inadvertent emergency speed brakes extension had occurred during drag chute deployment.
- Damage can result to the speed brakes actuators or their electrical conduits if the speed brakes doors are closed after the drag chute has been jettisoned, or if the doors are closed with drag chute not installed. This happens when the lower drag chute restraining strap falls down so that the bolt and nut on the strap are in the direct path of the protruding bolts on the speed brakes electrical conduits.

On airplanes equipped with overrun barrier probe*, pulling the drag chute handle will mechanically extend the overrun barrier probe at the same time the drag chute is activated.

PITOT-STATIC SYSTEM

The pitot-static system (figure 1-25) supplies pitot pressure to the airspeed indicator, the pressure switch of the landing gear warning system, the fire control system, AFCS, and the engine pressure ratio gage. Static pressure is applied to the airspeed indicator, the vertical velocity indicator, AFCS, and the altimeter. The pitot-static tube is mounted on the end of the nose boom. Ram air (q) pressure, is supplied from two tubes located on the leading edge of the vertical fin, to control the elevator and rudder artificial feel systems. All pitot tubes are anti-iced by electrical power from the 28-volt dc essential bus.

**AF 57-770 & on, & airplanes modified by TCTO 1F-102-658.*

INSTRUMENTS

Note

This paragraph covers only those instruments which cannot be considered to be parts of complete systems, such as fuel system, engine, etc.

MM-2 ATTITUDE INDICATOR

The airplane is equipped with the MM-2 remote attitude indicator to give visual indication of the flight attitude of the airplane in pitch and roll. This indicator is a remote indicating instrument having the gyroscopic control unit located on the centerline of the airplane at the aft end of the upper electronics compartment with the indicating phase located on the instrument. The system is powered from the 115-volt, 3 phase, 400 cycle essential bus and the 28-volt dc essential bus. Erection of the gyro requires approximately 2½ minutes after application of power and can be observed by disappearance of the "OFF" power failure flag visible through the cover glass of the indicator. The "OFF" flag will appear in case of complete ac or dc power failure. However, a slight reduction in ac or dc power, or failure of certain electrical or mechanical components within the system, will not cause the "OFF" flag to appear, even though the system is not operating properly.

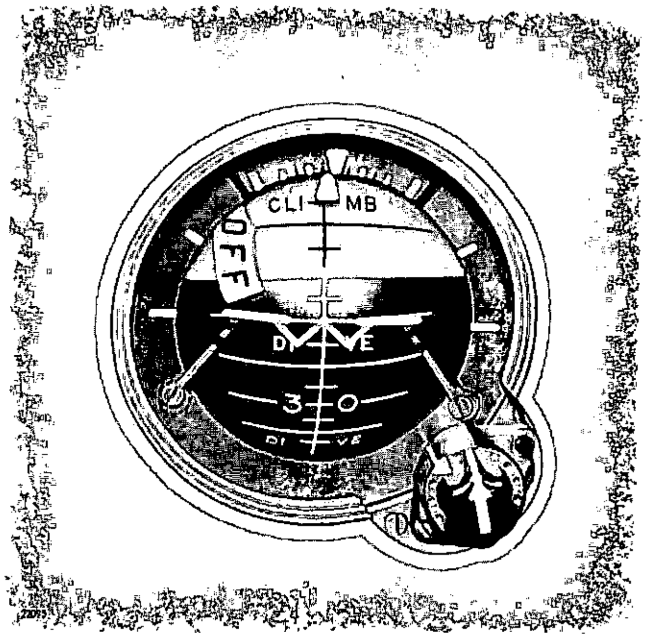
WARNING

- During flight, it is possible that a malfunction of the attitude indicator might be determined only by checking it with the other flight instruments and with the artificial horizon on the radar scope.
- If the "OFF" flag requires longer than 2½ minutes to retract or any oscillations are noted on the indicator after the "OFF" flag retracts, a possibility of a malfunction exists. Either of the above is cause for rejection of the indicator and should be noted on Form 781.

The instrument is operative through 360° of roll, 82° of climb, and 82° of dive, and it is not likely to tumble even during extreme maneuvers. However, should the gyro tumble, it will require approximately 15 minutes to erect. Indicator error is less than ½° in level flight, and, up to a turn rate of 40° per minute, the indication error compares to that of a conventional gyro. In turns of 40° or more per minute a compensating mechanism in the instrument limits turn error indication to 2°

WARNING

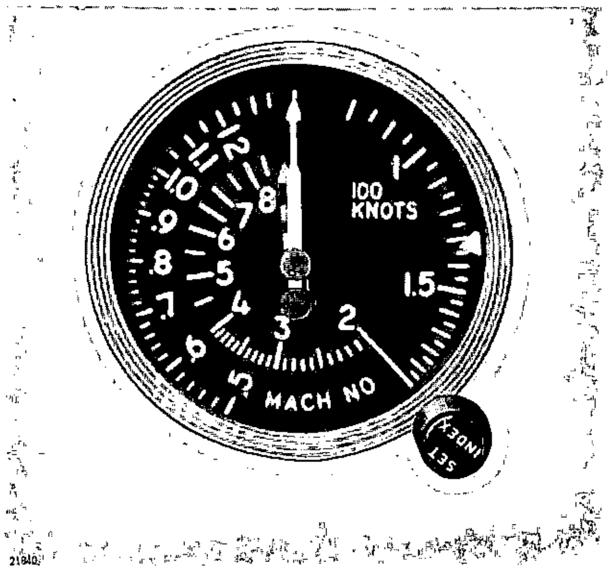
A slight amount of pitch error in the indication of the MM-2 attitude indicator will result from accelerations or decelerations. It will appear as a slight climb indication after a forward acceleration and as a slight dive indication after deceleration when the airplane is flying straight and level. This error will be most noticeable at the time the airplane breaks ground during the takeoff run. At this time, a climb indication error of approximately 1½ bar widths will normally be noticed; however, the exact amount of error will depend upon the acceleration and elapsed time of each individual takeoff. The erection system will automatically remove the error after the acceleration ceases.



The indicator does not have a manual caging handle. When power is turned off, a snubber automatically grips the gimbal and keeps it from tumbling. When power is turned on, the snubber is released after a 15-second time delay. As level flight pitch attitude of the airplane varies with different loadings and speeds, a pitch trim knob is provided on the indicator to center the horizon bar after the airplane has been trimmed for level flight.

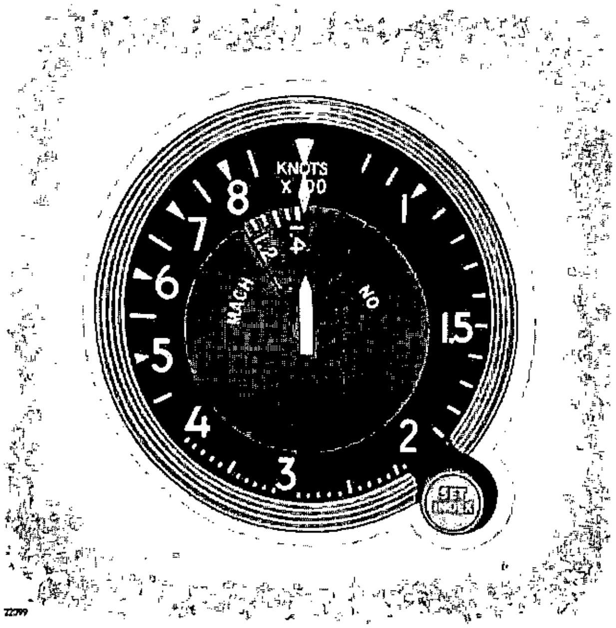
MACHMETER-AIRSPEED INDICATOR

The Machmeter airspeed indicator (8, figure 1-4) will indicate airspeeds up to 850 knots and up to Mach 2.2. Early airplanes are equipped with an ME-2 indicator and later airplanes are equipped with an ME-4 indicator. The instrument contains a dual-pointed needle which points to a movable Mach scale that rotates with altitude changes to show the Mach number that is equivalent to indicated airspeed for the particular flight altitude. For example, at sea level, the Mach 1.0 scale might be opposite the 650 knot graduation of the IAS scale. A climb to 40,000 feet would cause the Mach scale to rotate so that Mach 1.0 would be opposite the 310 knot scale. A knurled knob, located at the lower right side of the



instrument, allows setting of a movable index marker (this marker moves along the perimeter of the dial) to reference a desired speed. Mach number is read from the airspeed pointer on the Mach number dial (inside the cut-out on the ME-4 indicator and on the outer left-hand edge on the ME-2 indicator). The indicators use impact and static pressures from the pitot-static system. The red and black limiting pointer is not used on this airplane and has been set to the full limit of its upward travel. Early airplanes* have an airspeed correction card in a holder, located on the left side of the cockpit, above the console and forward of the throttle quadrant.

AF 53-1791 thru 55-3357.



ACCELEROMETER

A three-pointer accelerometer located at the lower left corner on the main instrument panel, on some airplanes, shows positive and negative g-loads. In addition to the conventional indicating pointer, there are two recording pointers (one for positive g-loads and one for negative g-loads) which follow the indicating pointer to its maximum attained travel. The recording pointers remain at the maximum travel position reached by the indicating pointer, thus providing a record of maximum g-loads encountered. To return the recording pointers to the normal (one g) position, it is necessary to press the knob on the lower left corner of the instrument ring.

CAUTION

Approach g limits slowly as difference in location of accelerometer and airplane center of gravity can introduce a lag as much as one g when making rapid changes in attitude.

ALTIMETER

A conventional altimeter (36, figure 1-4) is installed for use in determining pressure altitude of the airplane above sea level. Some airplanes are equipped with an MB-2 altimeter which is conventional in indication except for warning hash-marks in the lower portion of the instrument. The hash-marks are covered above 16,000 feet and become visible when descending below 16,000 feet. Other airplanes are equipped with the MA-1 altimeter which is similar to the MB-2 except for greater accuracy. Later airplanes* have an altimeter calibration card in a holder, located on the left side of the cockpit, above the console and forward of the throttle quadrant.

WARNING

The barometric setting knob can be turned so that the barometric disc will rotate through 360°. If the correct altimeter setting is then established, the altimeter will indicate 10,000 feet in error.

TURN-AND-SLIP INDICATOR

A conventional turn-and-slip indicator (37, figure 1-4) located on the instrument panel is rated for a four-minute turn (1½° per second). On some airplanes** the turn-and-slip indicator is mounted perpendicular to the longitudinal axis of the airplane and presents correct turn indications.

**AF 55-3358 & on.*
***Airplanes modified by TCTO 1F-102-759.*

WARNING

On other airplanes the turn-and-slip indicator presents erroneous turn indications due to mounting the instrument flush with the inclined instrument panel. This mounting results in inclination of the longitudinal axis of the turn-and-slip indicator with respect to the roll axis of the airplane, which causes false turn indications. For example, if the instrument were mounted in the floor of the cockpit, it would no longer indicate rate of turn but would indicate rate of roll (due to gyroscopic action this would be indicated by a left needle deflection when rolling to the right, or vice versa.) The inclined installation combines both roll and turn indications to the instrument which cause false indications. Since a turn is induced by first rolling into the turn, the turn needle will indicate roll (in the opposite direction), and once the turn is established the needle will reverse its movement and indicates a turn in the proper direction. The faster the rate of the roll the greater the erroneous indication in the opposite direction. In the event the attitude indicator and the radar scope reference are inoperative, and the turn-and-slip indicator is used as the primary flight instrument, excessive rate of roll should be avoided when establishing the desired bank angle.

The turn-and-slip indicator is powered by the 28-volt dc essential bus.

STANDBY COMPASS

A conventional magnetic compass, suspended from the canopy, is furnished for navigation purposes in the event of failure of navigation equipment or electrical system failure. Illumination of a light within the compass case is controlled by a switch on the right side of the cockpit. A standby compass correction card and holder are located on the left side of the cockpit, above the console and aft of the throttle quadrant.

OUTSIDE AIR TEMPERATURE GAGE

An outside air temperature gage is installed on early airplanes* and is located on the right-hand console. The indicator has a range from −50° to +50°C. It is an electrical resistance type instrument and measures ambient temperature. The temperature sensing unit is a resistance bulb located in the engine inlet duct. The temperature indicating system also receives power from the 28-volt dc essential bus.

AF 53-1791 thru 54-1383.

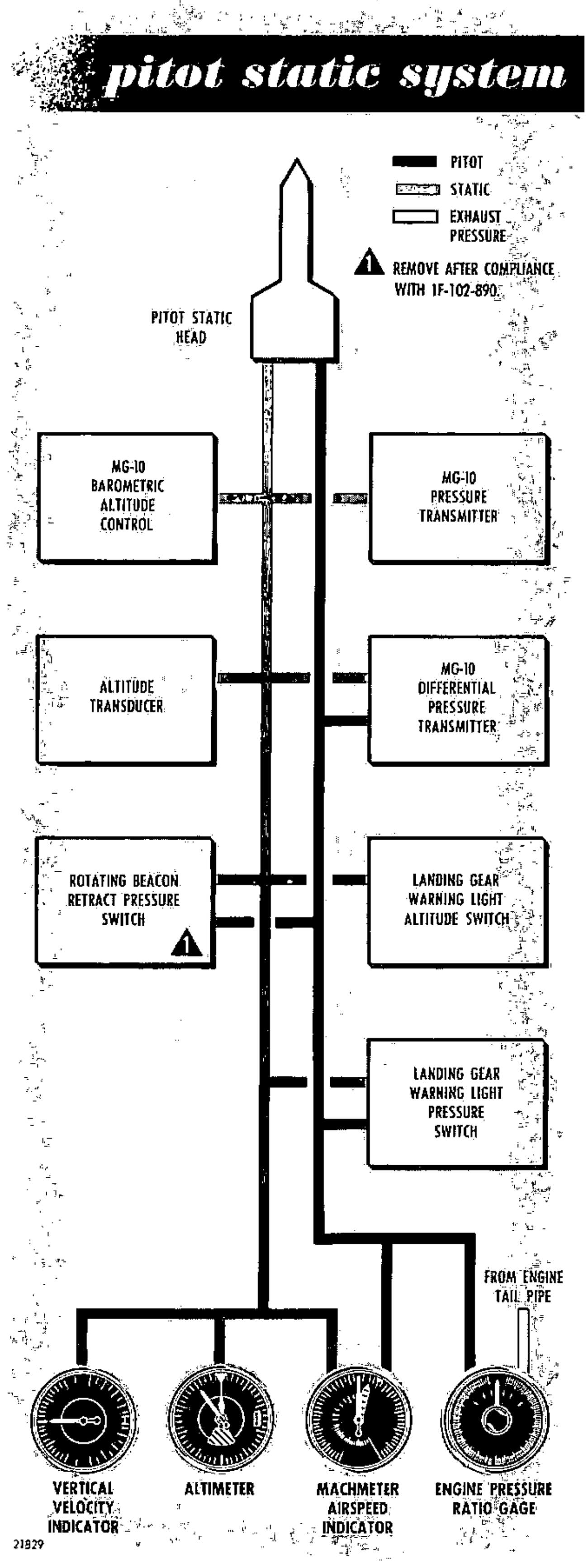


Figure 1-25

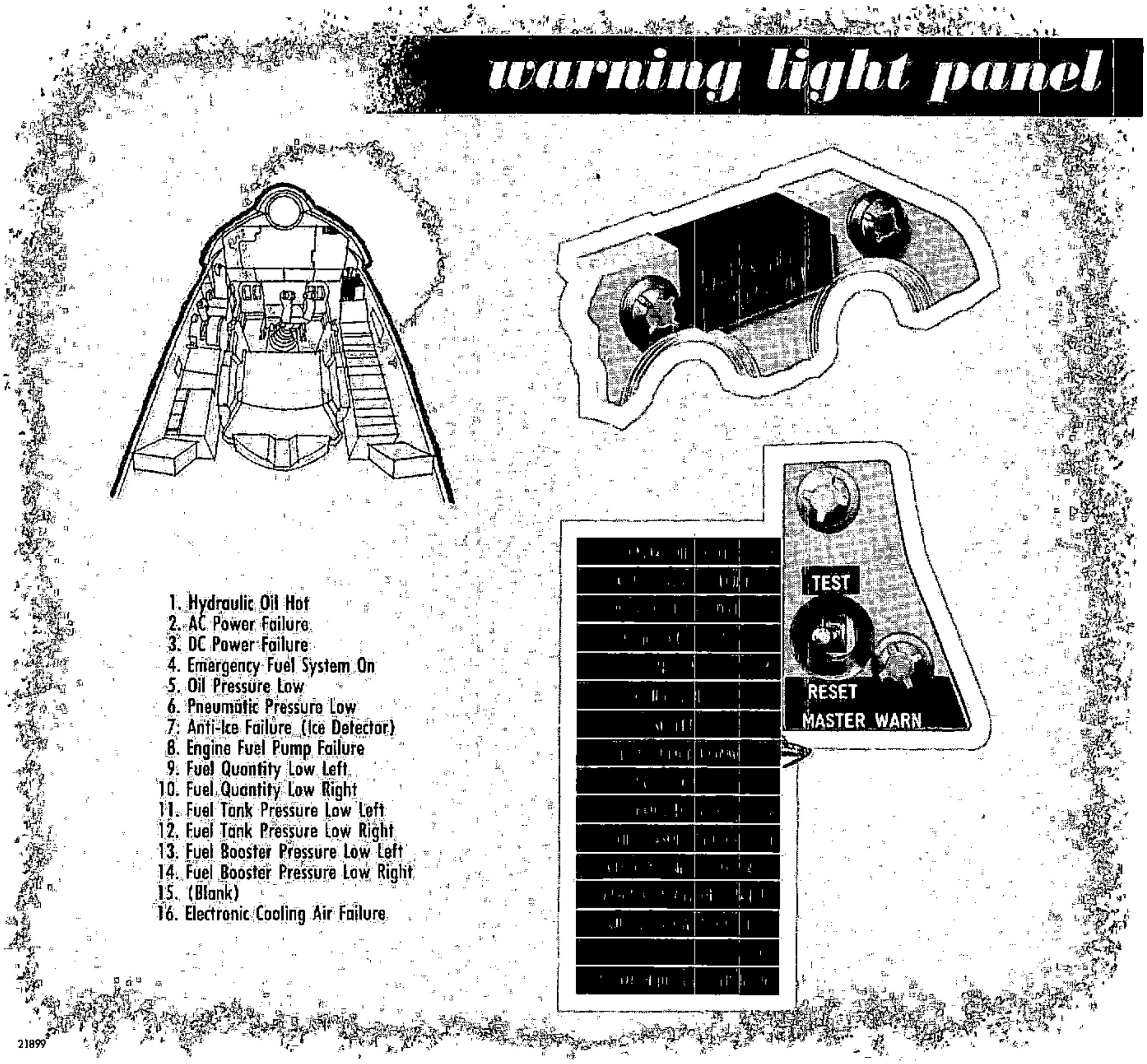


Figure 1-26

CLOCK

The clock (6, figure 1-4), located on the instrument panel, is an eight-day spring-winding type. It contains an elapsed-time mechanism which uses a sweep-second hand. The elapsed-time mechanism is started, stopped, and reset by pushing in on the elapsed time button.

DIRECTIONAL INDICATOR (SLAVED)

Refer to NAVIGATION EQUIPMENT, Section IV.

EMERGENCY EQUIPMENT

MASTER WARNING SYSTEM

A warning light panel (figure 1-26), located at the forward end of the right-hand console has 16 individual amber warning lights which indicate malfunctions or failures of various systems and equipment. Illumination of any individual light also illuminates an amber master warning light on the main instrument panel, within the pilot's normal line of vision. Once illuminated, the master warning light can be extinguished with a master warning test and reset switch. However, the individual warning light will remain illuminated until the malfunction is cleared. Subsequent malfunction will again illuminate the master warning light. Warning lights are automatically dimmed when the flight instrument lights are on, if the thunderstorm lights are off. The master warning system does not include the landing gear unsafe warning light, canopy unlocked warning light, hydraulic pressure-low warning light, or fire and overheat warning light.

fire warning system

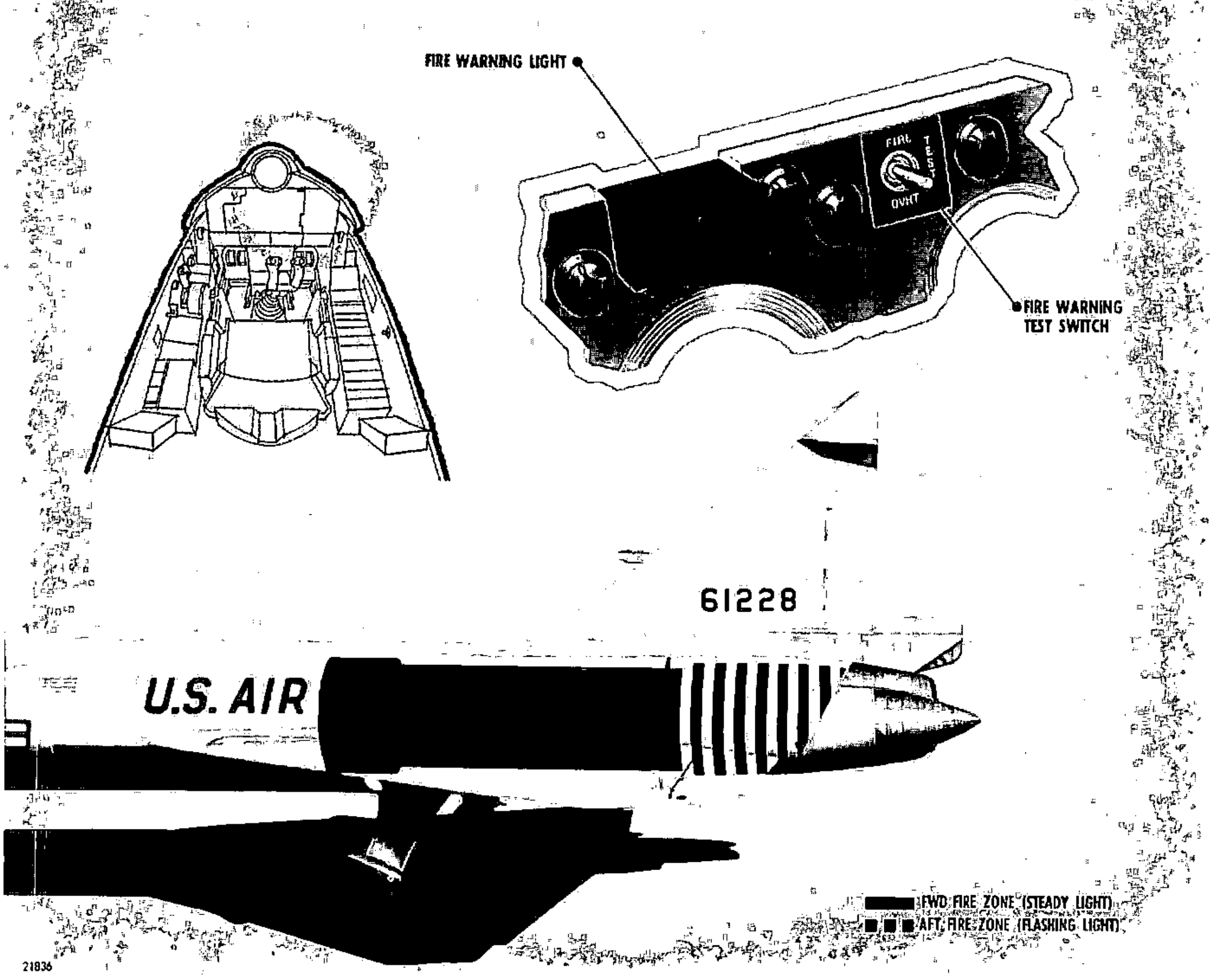


Figure 1-27

Master Warning Test and Reset Switch

A three-position master warning test and reset switch (figure 1-26), placarded "Master Warn," is located at the right of the master warning light panel. It has positions TEST and RESET, and is spring-loaded to a center (OFF) position. When held in the TEST position all lights in the master warning system should illuminate.

Note

Placing the switch to TEST is a functional check of lights only; not of complete warning circuits.

In the event of a system malfunction, both individual and master warning lights will illuminate. The RESET position is used to extinguish and reset the master warning light only so that it will relight again in case of a subsequent malfunction. The master warning system receives power from the 28-volt dc essential bus.

ENGINE FIRE WARNING SYSTEM

A fire warning system (figure 1-27) is installed to detect and indicate fire and overheat conditions in the forward or aft engine compartments. The forward engine compartment (which includes the compressor and accessory sections) and the aft compartment (which includes the combustion chambers and the turbine) have separate detector loops and detector units. The detector loops are of resistance-type coaxial construction. A hot spot anywhere along the length of the loop completes the circuit (between a center conductor and the outside tube) which the detector unit senses. A fire burning through the loop would not affect its operation. Both forward and aft detector networks are electrically connected to a warning light in the cockpit.

Engine Fire Warning Light and Test Switch

An abnormally high temperature in the forward or aft engine compartments is indicated by the red fire warning light (13, figure 1-4), located on the instrument panel. When illuminated the light displays "FIRE." To identify the zone of the fire, the light will illuminate steadily for the forward engine compartment and flash for the aft engine compartment. A three-position test switch (figure 1-8), located at the right of the warning light, is placarded "Test" and is spring-loaded to the center (OFF) position. When the switch is held to FIRE position, the warning light should illuminate steadily, indicating proper operation of the forward loop, detector, and light. When the switch is held to OVHT position, the warning light should flash, indicating proper operation of the aft loop, detector, flasher, and light. Failure of the flasher unit will not affect forward loop operation but the aft loop would be inoperative. The warning system receives power from the 28-volt dc essential bus.

SURVIVAL KIT

A survival kit* (figure 1-28) is furnished with some airplanes and is designed to fit in the ejection seat. The kit, fitted with a rubber cushion which snaps to the kit, is of fiberglas construction and serves as a seat cushion with a back-type parachute. The survival kit attaches to the parachute harness by means of an adjustable strap on each side and consists of an oxygen regulator, two bailout oxygen bottles, personal equipment leads, a one-man life raft (if required), a provision kit, and a reflector on back of the survival kit lid. The bundle of personal equipment leads is inserted into a receptacle in the right rear corner of the kit. The receptacle contains connections for oxygen, partial pressure suit, mask defog, communications leads, and the green knob for bailout bottle manual actuation. The oxygen regulator in the kit controls all oxygen used by the pilot (breathing and partial pressure suit) both in the event of ejection and during normal flight and ground operation. An oxygen system press-to-test button is located on the front panel of the kit (refer to LIQUID OXYGEN SYSTEM, Section IV). In event of ejection, oxygen from the bailout bottles provides breathing oxygen and pressure suit oxygen for a minimum of 12 minutes. The bailout bottles utilize one pressure gage which is visible through a small window in the rear portion of the kit. Pressure in the bottles should read 1800 psi and should be checked prior to each flight. A bailout bottle reducer lowers the pressure to 40-60 psi and delivers this pressure to the regulator. The bailout bottles are actuated automatically as the ejection seat leaves the airplane during ejection, or may be manually actuated during flight whenever the bailout bottles are required. Manual actuation of the bottles is accomplished by pulling the round green knob attached to a cable in the personal equipment lead bundle. Manual actuation may be desired at any time the ship's oxygen system is depleted or is not supplying oxygen for breathing or partial pressure suit operation. The life raft is inflated by an automatically actuated carbon dioxide bottle mounted on the inside of the kit. The provision kit is a waterproof packet strapped in the bottom of the survival kit and should contain the items necessary for survival in the area of operation.

**AF 56-1275 thru -1316, -1332 & on, & airplanes modified by TCTO 1F-102-642.*

CAUTION

Do not step or stand on the survival kit as damage to the case or personal leads can result. Dirt and grease can also be introduced into the oxygen regulator, rendering it inoperative and possibly creating a fire hazard.

Emergency Airway Valve

An emergency airway valve, located on the reverse side of the partial pressure suit bladder lead (figure 1-28), is provided on some survival kits. This valve is being deactivated** in all airplanes and should be capped.

Survival Kit Emergency Release Handle

The yellow emergency release handle (figure 1-28) is located on the right side of the survival kit and is placarded "Emergency Release" on some airplanes or "Kit Release" on other airplanes. The handle is hinged at the rear, and when raised following ejection and chute deployment, will release the kit. The handle also releases the kit for quick ground egress. When fully raised, the handle will separate from the kit. Raising the handle causes the following to occur simultaneously after ejection and parachute deployment:

- Disconnects kit from personal equipment lead bundle.
- Lower portion of kit drops but remains attached to parachute harness by lanyard.
- Actuation of carbon dioxide bottle for life raft inflation.

For quick ground egress, raising the handle will cause the following to occur:

- Disconnects kit from personal equipment lead bundle.
- Releases parachute attaching wedges, completely separating the pilot from the survival kit.

***In accordance with TCTO 1F-102-678.*

survival kit

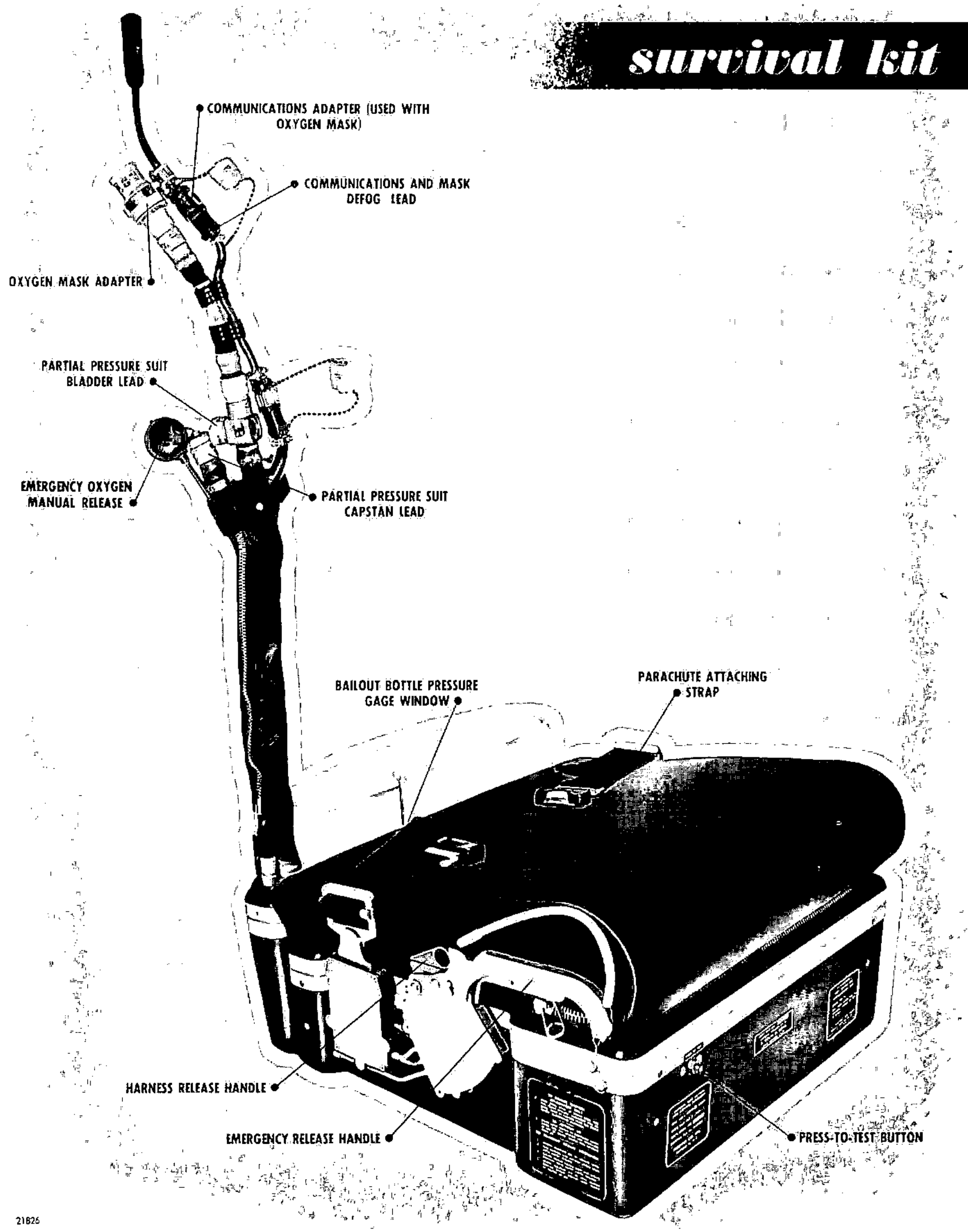


Figure 1-28

WARNING

On survival kits — Part No. 8-09353-7 — the 25-foot lanyard will remain attached to the parachute and the life raft will inflate when the emergency release handle is pulled. When equipped with the 8-09353-7 survival kit, the parachute should be removed prior to abandoning the airplane.

When released the kit will fall away from the pilot and remain attached by means of a nylon lanyard (approximately 25 feet long) attached to the right-hand parachute attaching strap. The life raft and kit will remain attached throughout the descent and will strike the surface approximately 25 feet ahead of the pilot. The emergency release handle should be raised and the kit released during the descent after parachute is fully deployed and stabilized, and a safe altitude for breathing without supplemental oxygen is reached.

WARNING

- On some airplanes, pulling the emergency release handle will always inflate the rubber life raft (if installed), whether or not the kit is in the seat. On other airplanes*, the raft will not inflate in the seat.
- Do not raise the emergency release handle during descent until after parachute deployment to prevent the kit or the lanyard from fouling the parachute.
- Do not raise the emergency release handle until after descent to an altitude not requiring oxygen. The oxygen supply will be cut off when the survival kit is released.

Survival Kit Harness Release Lever

The harness release (figure 1-28) is a lever aft of the emergency release handle on the right-hand side of the survival kit. It is placarded "Harness Release" and has a hole into which a finger may be inserted for raising. The lever is hinged at the rear and, when raised, releases the survival kit from the parachute attaching straps. This lever is designed for use when an emergency escape other than ejection is desired, such as escape from the airplane after a crash landing. When the lever is raised the pilot is released from all connections on the survival kit except the personal leads.

**Airplanes modified by TCTO 1F-102-642 or TCTO 1F-102-679.*

CANOPY

The metal-reinforced plexiglas canopy (7, figure 1-2) is a clamshell type which is manually opened and closed. The canopy is hinged at the rear and opens to provide a maximum opening of approximately 40 degrees. To facilitate manual raising and lowering of the canopy, a pneumatic counterbalance cylinder is installed which equalizes the weight of the canopy. An exterior canopy grip is installed on the forward left side of the canopy to aid in opening and closing the canopy when the pneumatic system is not serviced. A clamping or cinch-down force (from high-pressure pneumatic system) is applied by the counterbalance cylinder during the final movement of closing the canopy to compress the canopy seal and facilitate engagement of the latches. The counterbalance cylinder incorporates a ballistic charge to jettison the canopy in emergencies. Manually operated latches secure the canopy in the closed position and a warning light illuminates whenever the latches are not fully engaged. An inflatable canopy seal is incorporated around the base of the canopy to permit pressurization of the cockpit. Emergency jettisoning of the canopy is accomplished by raising the canopy jettison handle (located on the forward end of the left-hand armrest), raising either ejection seat handgrip, or by pulling the canopy external jettison handle. When the canopy is jettisoned by the above methods, it arms the ejection seat arming initiator. On some airplanes, if the canopy cannot be jettisoned in flight and it is manually raised and allowed to be removed by windblast, the seat arming initiator becomes armed. However, on other airplanes*, the seat arming initiator will not arm when the canopy is manually released in flight, and ejection from the airplane will not be possible. On early airplanes** to facilitate taxiing with the canopy open, a canopy hold-open rod is secured to the forward right-hand edge of the canopy. With the canopy open, the hold-open rod can be secured to the airplane structure to prevent small airloads from shifting the canopy position. The hold-open rod is secured by a spring-loaded sleeve which is raised to fasten or unfasten the rod. A spring clip, along the base of the canopy, is used to stow the rod when not in use. On later airplanes† a solenoid-operated valve is installed in the pneumatic pressure line that connects the upper and lower portions of the canopy counterbalance cylinder. This valve is controlled by a pushbutton switch located on the internal left-hand canopy grip, and when the valve is closed the canopy is locked in any desired open position.

CAUTION

- The canopy is not designed to be opened in flight as it would be completely removed by the wind blast.

**AF 55-3427 thru 56-1429 unless modified by TCTO 1F-102A-565.*
***AF 53-1791 thru 54-1400.*
†AF 54-1401 & on, & airplanes modified by TCTO 1F-102A-571.

- If the canopy is open during taxi operations, observe canopy limit speeds. The canopy hold-open rod must be used on airplanes which do not have a canopy hold button. Keep hands away from canopy sill unless the canopy support tool is in place.

CANOPY SEAL

An inflatable rubber seal is installed around the base of the canopy to provide sealing of the canopy to the fuselage and windshield. Engine compressor bleed air is used to inflate the seal to permit cockpit pressurization during flight. A valve operated by the canopy latch mechanism automatically admits air pressure to inflate the seal when the canopy latch handle is pushed fully in. The initial pull of the canopy latch handle dumps pressure from the canopy seal.

CANOPY GROUND SUPPORT TOOL

A removable canopy ground support tool (figure 1-23) is provided for use during ground operations. The tool fits between the canopy and canopy sill on the left side of the airplane.

WARNING

The canopy ground support tool should be in place before entering or leaving the cockpit to prevent the possibility of serious personal injury should inadvertent closing of the canopy occur. When the tool is not in position, special care should be taken to keep the area between the canopy and the canopy sill clear.

CANOPY COUNTERBALANCE CYLINDER

To facilitate manual raising and lowering of the canopy a pneumatic cylinder is installed and connects to the canopy forward of the canopy hinge line. The cylinder (in counterbalance condition) has air pressure on each side of a piston which, due to larger area on the lower side, applies an upward force that equalizes the weight of the canopy. The counterbalance cylinder is pressurized by the high-pressure pneumatic system through a regulator which reduces the operating pressure to approximately 1500 psi. During the final movement of closing the canopy a valve is mechanically opened which automatically relieves pressure from the lower side of the piston. Then the existing pressure on the upper side of the piston applies a downward force (cinch-down force) on the canopy which compresses the canopy seal and enables the canopy latches to be manually engaged.

WARNING

The final movement (one to two inches) of closing the canopy is rapid and is cinched down with considerable force; therfore, care should be taken that the area beneath the canopy is clear to prevent personal injury or damage to equipment.

To open the canopy it is necessary to push the canopy latch handle fully in and return to fully out position or depress the canopy external release button which will mechanically open a valve allowing high pressure (at a regulated value) to enter the lower side of the counterbalance cylinder restoring the counterbalance condition, and permitting the canopy to be raised manually. On later airplanes* a solenoid-operated valve is installed in the counterbalance air line between the upper and lower sides of the counterbalance cylinder to lock the canopy in any desired open position for taxi operations. This valve is spring-loaded to the open position; therefore, whenever power is being supplied to the 28-volt dc essential bus and the canopy is opened, the valve will close forming an air lock within the upper and lower portions of the counterbalance cylinder which will lock the canopy in its existing position. To facilitate desired movement of the canopy, a canopy hold button is installed on the internal left-hand canopy grip. This button, when depressed, will open the circuit to the solenoid-operated valve allowing the canopy to be manually raised or lowered. In addition, to facilitate complete closure and initial opening of the canopy from the outside when 28-volt dc power is connected, a limit switch is installed to disconnect dc power to the canopy hold button which permits free canopy movement for approximately six inches from the fully closed position. Depletion of high-pressure pneumatic pressure during flight will prevent restoring complete counterbalance action; however, the canopy can be raised manually with partial aid of counterbalance action.

CANOPY EXTERIOR RELEASE BUTTON

When the canopy is manually closed (on a parked airplane) the counterbalance cylinder applies the cinch-down force to seal the cockpit from the elements. To gain access again, it is necessary to restore counterbalance action before the canopy can be raised. This is accomplished by an exterior release button (figure 2-2) placarded "Push to Release Canopy" located on the outside of the left-hand intake duct near the base of the canopy. Pushing the button mechanically controls a pneumatic valve which supplies pneumatic pressure to the counterbalance cylinder thus relieving the cinch-down force. The canopy can then be raised manually.

**AF 54-1401 & on.*

CANOPY LATCH HANDLE

The T-shaped canopy latch handle (26, figure 1-4), located on the right side of the instrument panel, is used to mechanically engage and release the canopy latches. When the canopy is completely closed, pushing the handle fully in will engage the canopy latches and allow inflation of the canopy seal. A warning light above the handle will go out when the latches are fully engaged. Initial movement of the handle from the fully in position causes the canopy unlocked warning light to illuminate. Subsequent aft movement of the handle unlatches the canopy latch hooks and, when the handle is fully out (approximately eight inches), relieves canopy seal pressure and places the pneumatic control valve in the counterbalance condition. The canopy can then be raised manually.

CANOPY HOLD BUTTON

The canopy hold button (figure 2-2) located on the internal left-hand canopy grip, is spring-loaded to the out position which supplies power to the solenoid-operated valve, located on the canopy counterbalance cylinder. Whenever power is being supplied to the 28-volt dc essential bus and canopy movement is desired, it is necessary to depress this button permitting free flow of pneumatic pressure within the counterbalance cylinder, and allowing the canopy to be manually raised or lowered. The canopy hold button receives power from the 28-volt dc essential bus whenever the canopy is opened in excess of approximately six inches.

CANOPY JETTISON HANDLE

The yellow canopy jettison handle (figure 1-29) may be used to jettison the canopy by a ballistic charge. The handle is located on the forward end of the left armrest and incorporates a safety latch which is released by pressing a button located on the outboard end of the handle. Raising the handle will unlatch and jettison the canopy with a ballistic charge. This system functions through the same mechanisms as used by the ejection seat handgrips and should be used when it is desired to jettison the canopy without exposing the handgrip triggers.

EJECTION SEAT HANDGRIPS

The canopy can be jettisoned by raising either the right or left ejection seat handgrips. Refer to EJECTION SEAT, this Section.

CANOPY EXTERNAL RELEASE HANDLE

In the event of rescue, if the canopy fails to jettison or if the presence of fuel fumes makes jettisoning inadvisable, the canopy may be unlatched and manually raised. The external canopy release handle is located below the right-hand windshield and is covered by a small access door (figure 3-6). When the access door is opened, the spring-loaded handle pops out. Pushing the handle aft releases canopy cinchdown pressure and unlatches the canopy which may then be raised manually.

CANOPY EXTERNAL JETTISON HANDLE

During emergency conditions, the canopy can be jettisoned from outside the airplane for emergency rescue by the external jettison handle (figure 3-6). The handle is located in a small compartment on the left side of fuselage forward of the wing intersection. Removing the access door exposes the handle which has approximately six feet of excess cable, allowing the operator to stand at a safe distance from the airplane before jettisoning the canopy. Pulling the cable outboard six feet, then applying a steady pull of approximately 30 pounds, initiates the sequence by mechanically pin-firing the canopy initiator (figure 1-30) which gas-fires the thruster. This releases the canopy latches and cinchdown pressure and gas-fires the canopy remover initiator which in turn fires the ballistic cartridge located at the base of the canopy remover cylinder. The expanding gases from the ballistic cartridge open the canopy, then jettison it. The canopy should travel up and aft and will probably strike the tail. The trailing wire will fire the seat arming initiator and arm the seat ejection system, preparing it for operation. Pulling the external jettison handle actuates the same mechanisms as does the canopy jettison handle or the ejection seat handgrips.

WARNING

The external canopy jettison handle should not be pulled except for emergency reasons. The ground safety lock pin which is installed in the ejection seat right handgrip linkage prevents jettisoning of the canopy from the cockpit but will not prevent jettisoning if the external handle is pulled.

CANOPY UNLOCKED WARNING LIGHT

The red canopy unlocked warning light (22, figure 1-4), located above the canopy latch handle, illuminates and displays "CANOPY UNLOCKED" when both canopy latches are not fully engaged. The warning light receives power from the 28-volt dc essential bus.

EJECTION SEAT

The ejection seat (figure 1-29) permits bailout at high speeds and any flight attitude. A catapult fired by a ballistic charge supplies the necessary force to eject the seat and pilot upward from the airplane. Some airplanes are equipped with the M-3 ballistic catapult and other airplanes* are equipped with the MK-1 rocket catapult. The seat has an automatic-opening safety belt and accommodates a back-type parachute. A seat cushion is furnished; however, a one-man life raft or a survival kit may be used instead of the seat cushion. A rigid seat style survival and oxygen kit container is used on some

**AF 56-1513 & on.*

ejection seat (typical)

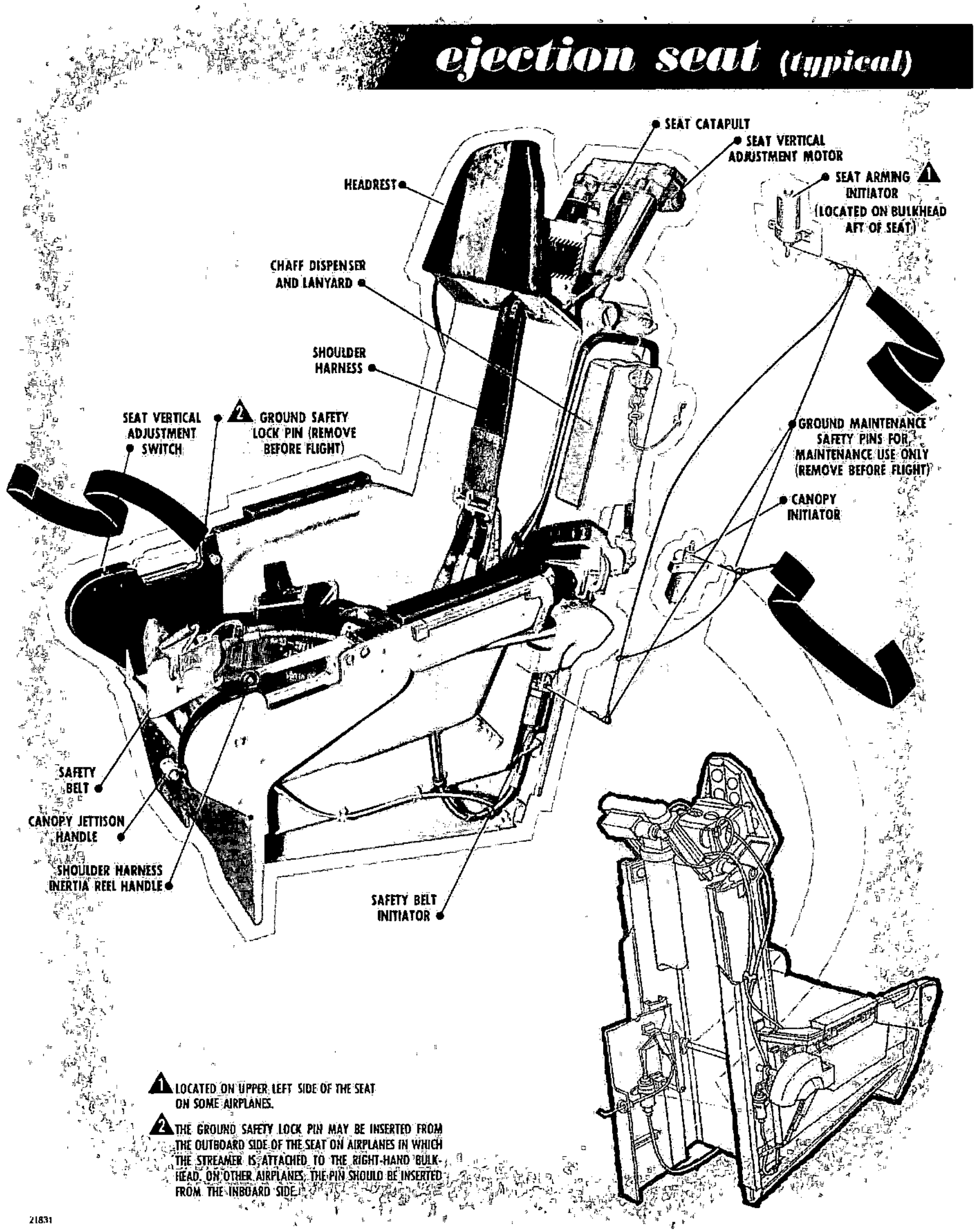


Figure 1-29

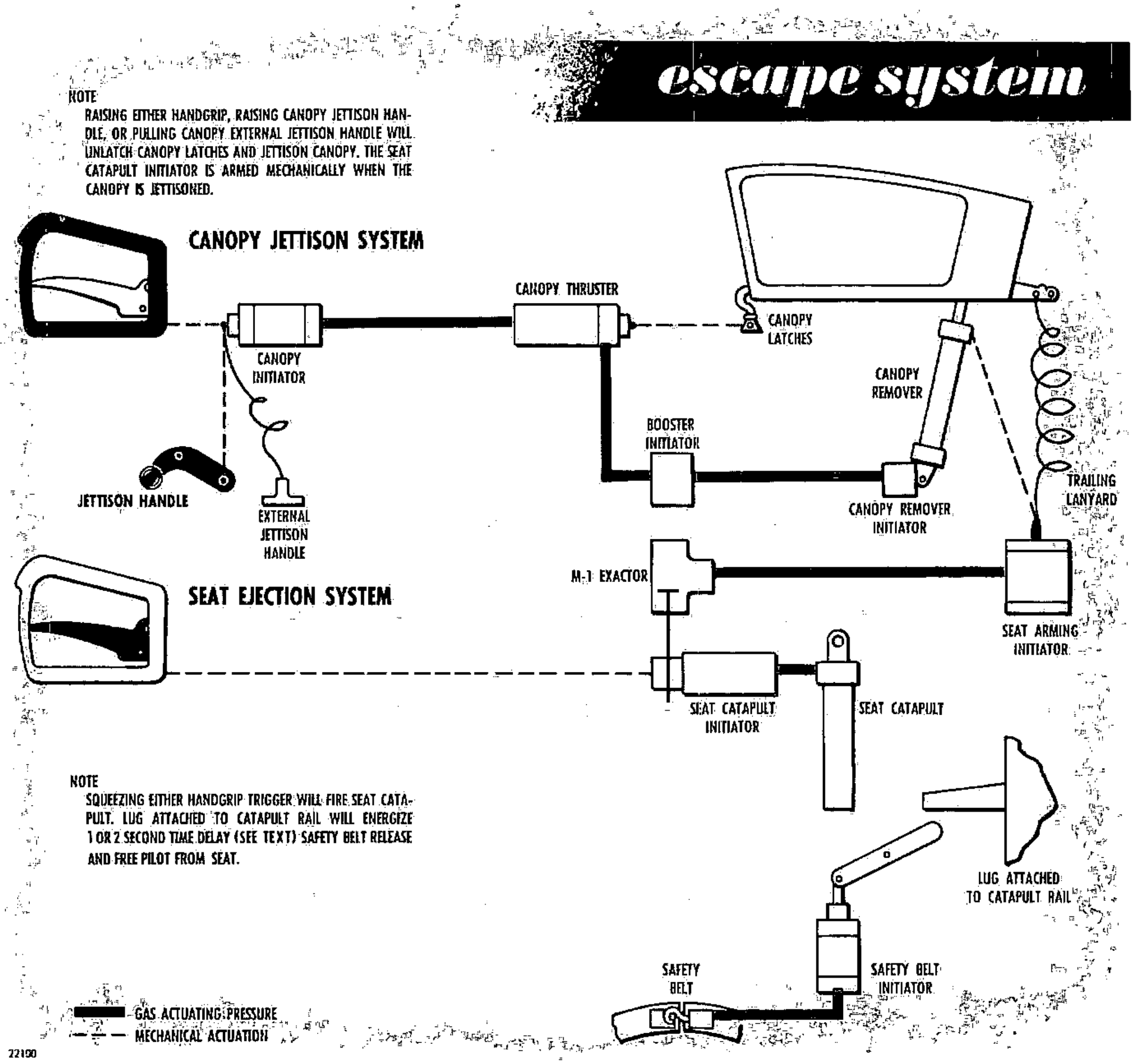


Figure 1-30

airplanes. No additional parachute support block is required as the survival kit container has its own support. On other airplanes, either the MC-2 seat cushion, or the MD-1 contoured seat style survival kit container should be used. When the MC-2 seat cushion is used, a parachute support block is required.

WARNING

Do not use the A-5 cushion, or any similar sponge rubber cushion, when equipped with a one-man life raft or survival kit. If ejection becomes necessary, serious spinal injuries can result when the ejection force compresses the cushion and enables the seat to gain considerable momentum before exerting a direct force on the pilot. Chance of injury during forced landing is also increased.

Vertical adjustment of the seat can be accomplished by an electric actuator. The shoulder harness inertia reel, located on the back of the seat, locks automatically when a rapid pull (equivalent to two to three g's deceleration) force is exerted on the harness assembly. The inertia reel is equipped with a manual control. On airplanes not equipped with a survival kit, the pilot's mask defog, headset and microphone leads, oxygen and anti-g suit hoses are attached to a disconnect unit on the lower left side of the seat and disconnect from the airplane automatically when the seat is ejected. On airplanes equipped with the survival kit, the pilot's mask defog, headset, microphone leads, and oxygen hoses are attached to the right-rear of the survival kit. The anti-g suit and vent are attached to a disconnect unit on the lower left side of the seat. All connections automatically disconnect from the airplane when the seat is ejected. Elbow guards on the armrests are folded down out of the way during normal flight but are raised automatically to a protective

position when the seat handgrips are raised prior to ejection. The ejection sequence is so designed that the seat cannot be ejected until the canopy has been jettisoned. On some airplanes*, an automatic-opening chaff dispenser (figure 1-29) is installed on the upper left-hand side of the ejection seat. The dispenser is opened on ejection by a lanyard attached to the ejection seat rail structure. The dispenser cover latch pin ring is safety-wired to the dispenser housing to prevent inadvertent opening of the dispenser. On ejection, the lanyard breaks the safety wire and opens the dispenser. The chaff dispenses to aid ground radar in locating the bailout area for rescue operations.

WARNING

Ground maintenance safety pins (figure 1-29) are inserted in the canopy initiator, the seat arming initiator and the safety belt initiator during maintenance operations. A ground safety lock pin is also inserted in the right-hand seat ejection grip. On some airplanes a fitting is provided at the right-hand side of the seat on the bulkhead for stowing the ejection seat safety lock pin. If any of the pins are left in place, canopy jettisoning, seat ejection and/or automatic opening of the safety belt is prevented.

AUTOMATIC-OPENING SAFETY BELT

An automatic-opening safety belt (figure 1-31) is provided to extend the maximum and minimum safe altitudes for seat ejection. In a high-altitude bailout, the automatic belt, combined with the automatic parachute, delays parachute deployment until a safe altitude is reached. In a low-altitude bailout, the automatic system reduces the altitude required for safe ejection by reducing the time required for separation from the seat and parachute deployment. Thorough testing of the automatic-opening belt has determined that the system is completely reliable and allows faster separation from the seat than does manual operation. It has also been determined that under no circumstances should the belt be manually opened prior to ejection. Manual opening of the belt prior to ejection precludes actuation of the automatic timing and release mechanism on the parachute and permits immediate separation from the seat upon contact with the airstream. If immediate separation occurs, the pilot is subjected to far greater deceleration forces and the wind shock could open the parachute pack prematurely. Therefore, a delay of one second is incorporated into the automatic belt release to allow a more satisfactory deceleration and provide wind blast protection for the parachute. Release of the automatic-opening belt is accomplished either by manual operation or by gas pressure from an automatically controlled initiator. The initiator supplies approximately 1500 psi gas pressure through a high-pressure hose and actuates a piston inside the belt release,

*Airplanes modified by TCTO 1F-102-874.

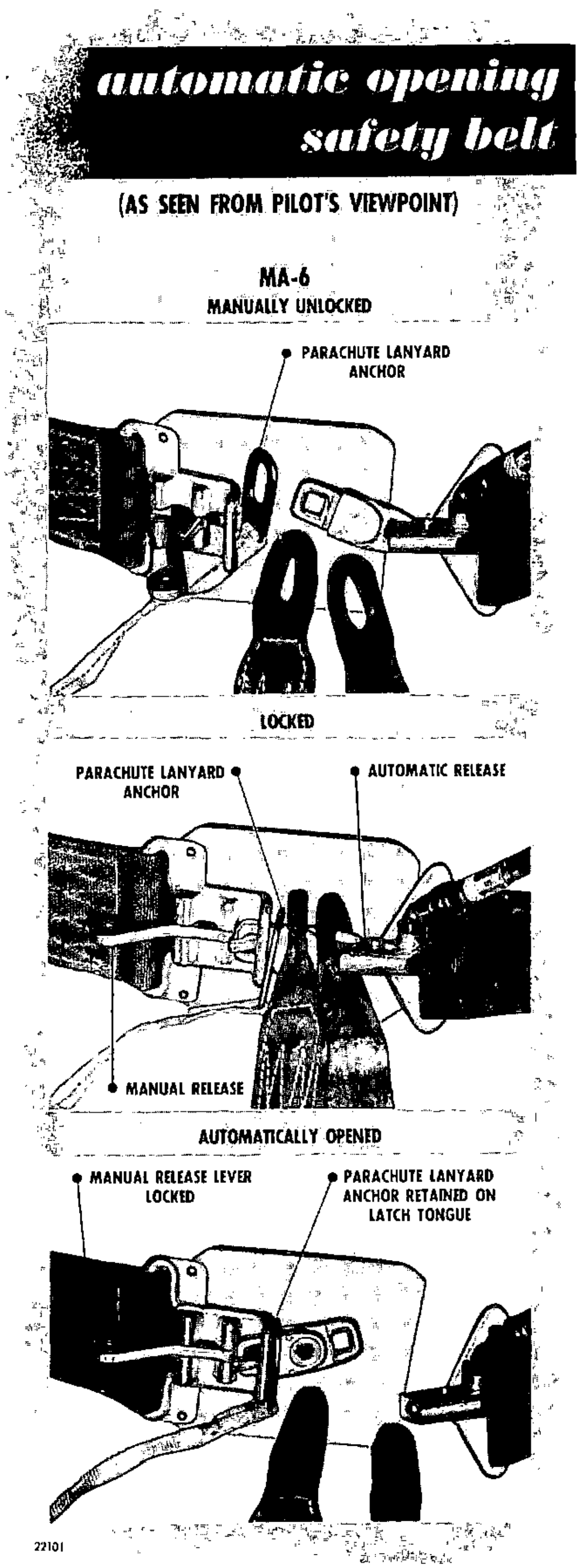


Figure 1-31

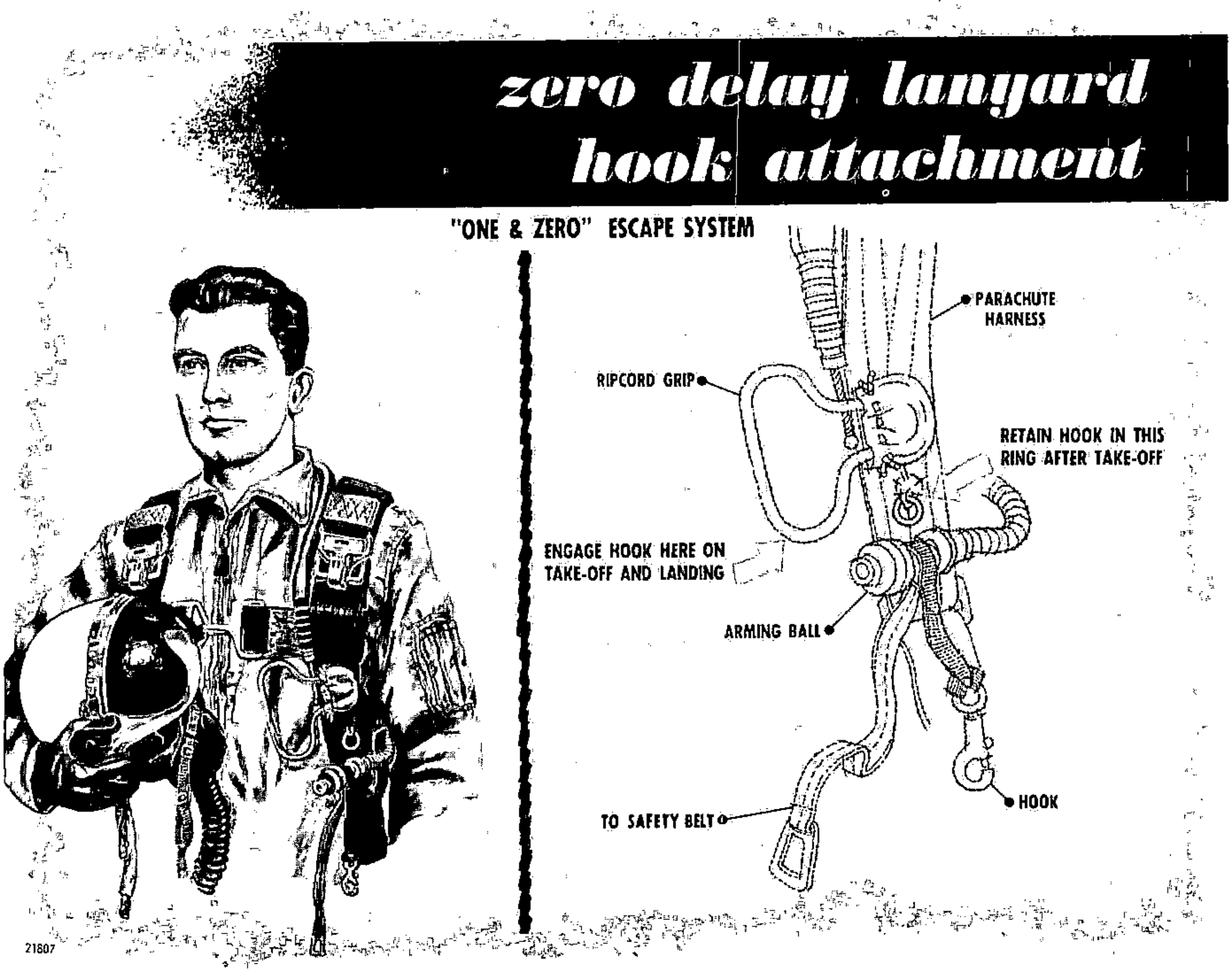


Figure 1-32

separating the belt at the latch. The release incorporates a means of attaching a lanyard which connects the belt with the static line to the timer of the automatic parachute. The MA-6 type belt connects the parachute release mechanism by inserting the safety belt tongue through a metal lanyard anchor. To close the MA-6, the shoulder harness loops must be placed on the safety belt tongue before the lanyard anchor. When the initiator is fired, the portion of the lap belt tongue holding the anchor is retained in the locked position and the portion holding the shoulder harness loops separates to allow separation from the seat and actuation of the parachute release. After the parachute release has been actuated by any of the automatic belts, a preset delay of two seconds must elapse, then chute deployment begins if ejection was at a safe altitude. In the case of high altitude ejection, the aneroid action of the parachute release will delay deployment until a preselected safe altitude is reached by free falling, then deployment begins after a two-second delay. Automatic operation of the safety belt systems can be overridden at any time by manual operation.

WARNING

If the automatic-opening safety belt is opened manually, the automatic parachute release will not be actuated and the parachute ripcord must be pulled manually.

"ONE-AND-ZERO" ESCAPE SYSTEM

A system incorporating a one-second safety belt delay and a zero-second parachute delay ("one-and-zero" system) is provided to improve low-altitude ejection seat escape capability. This system makes use of a detachable lanyard that connects the parachute timer knob to the parachute ripcord. (A sketch of the hook, figure 1-32, depicting the "hooked" and "unhooked" conditions is provided for illustrative purposes only.) At very low altitudes and airspeeds, this lanyard must be connected, thus providing parachute actuation immediately after separation from the ejection seat. At other altitudes and airspeeds, the lanyard must be disconnected from the ripcord, thus allowing the parachute timer to actuate the parachute below the critical parachute opening speed and below the parachute timer altitude setting. A ring attached to the parachute harness is provided for stowage of the lanyard hook when it is not connected to the parachute ripcord.

Note

See figure 3-3, Section III for the Zero Delay Lanyard Engagement Requirements Chart.

EJECTION SEAT HANDGRIPS

Raising either seat handgrip (figure 1-29) will lock the shoulder harness, raise the elbow guards, and jettison the canopy. The linkage is such that either handgrip may be raised without raising the other.

WARNING

- The canopy jettison and seat ejection system is safetied by a ground safety lock pin inserted through the right handgrip linkage. This pin must be removed before flight and replaced after flight by the pilot. Be certain the canopy jettison handle is in the detent position with the release button out when the pin is inserted to prevent accidentally jettisoning the canopy.
- The ground safety lock pin does not safety the canopy jettison system if the external canopy jettison handle is pulled.

When either handgrip is raised to the fully up position (handgrips will lock in up position exposing the seat catapult triggers) mechanical linkage will lock the shoulder harness, raise the elbow guards and fire an initiator unit. The expanding gases produced are routed to a thruster unit which also fires to disengage the canopy latches; and these gases which are bypassed fire an additional initiator unit. The resulting expanding gases are then routed to energize the canopy remover which will fire and jettison the canopy. A canopy trailing wire attached to the canopy fires an initiator unit and the expanding gases produced are routed to remove a pin, thus arming the seat catapult initiator.

SEAT CATAPULT TRIGGERS

The seat catapult triggers (figure 1-29) are located within the ejection seat handgrips, and are accessible only when the handgrips are in the fully up position. Squeezing either trigger fires an initiator. The expanding gases produced are routed to the seat catapult which fires the catapult, ejecting the seat.

Note

The seat ejection system is dependent upon the canopy jettison system. If raising the handgrips does not fire the canopy, the seat catapult initiator will not be armed; therefore, it cannot be fired by squeezing the triggers.

SEAT VERTICAL ADJUSTMENT SWITCH

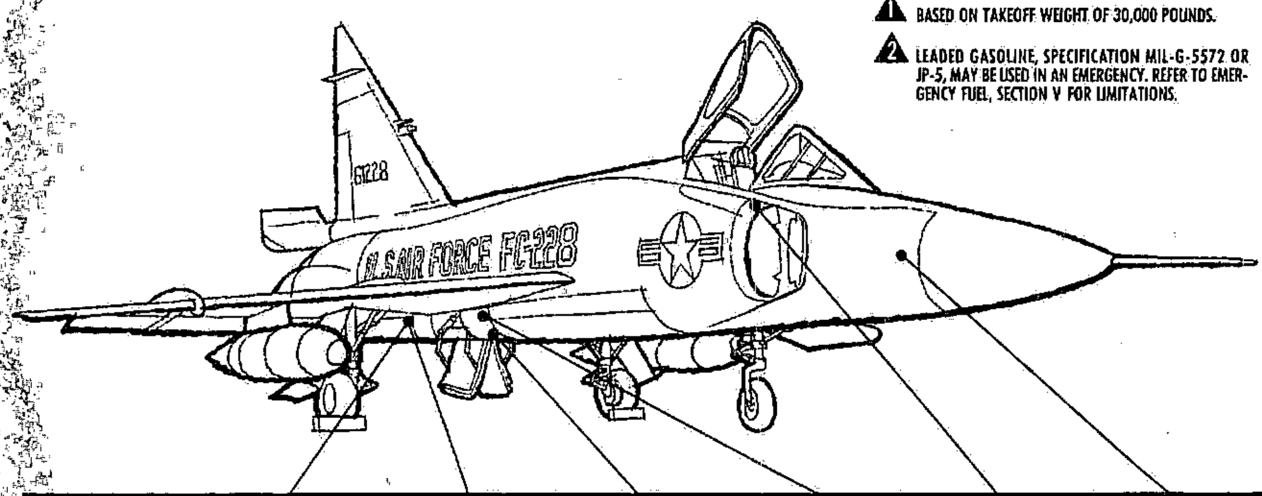
The three-position seat vertical adjustment switch (figure 1-29), located on the forward end of the right armrest, has positions UP, DOWN, and is spring-loaded to the center (OFF) position. Holding the switch to the desired position energizes an electric motor-driven actuator which moves the seat vertically in the direction selected. The seat vertical adjustment system receives power from the 28-volt dc nonessential bus.

WARNING

On airplanes without the survival kit, the oxygen lead may separate from the quick-disconnect fitting when the seat is adjusted to the fully up position, and must be checked after seat adjustment.

SHOULDER-HARNESS INERTIA REEL HANDLE

Manual control of the shoulder-harness inertia reel is provided by the shoulder-harness inertia reel handle (figure 1-29), located on the forward end of the left handgrip. The handle is placarded with an arrow pointing forward to the LOCKED position, and has overcenter detents which restrain it from slipping out of either LOCKED or aft (UNLOCKED) position. When the handle is in the UNLOCKED position, the reel harness cable will extend to allow the pilot to lean forward in the cockpit. Sudden forces applied by crash landing impact, turbulence, or rapid maneuvers, which tend to rapidly separate the pilot from the seat (including forward, upward, or sideward motion), will automatically lock the harness reel. The reel locks within approximately one-half inch of cable travel. When the reel is locked in this manner, it will remain locked until the handle is moved to the LOCKED and then returned to the UNLOCKED position. When the handle is in the LOCKED position, the reel harness cable is manually locked so that the pilot is prevented from bending forward. The LOCKED position is used only when a crash landing is anticipated and provides an added safety precaution over and above that of the automatic reel lock. If the harness is automatically or manually locked while the pilot is leaning forward, this harness retracts with him as he straightens up, moving into successive locked positions as he moves back against the seat. To unlock the harness, the pilot must be able to lean back enough to relieve tension on the lock. Therefore, if the harness is locked while the pilot is leaning back hard



NAME	Fuel Flask (Combustion Starter)	Constant Speed Drive Accumulator	Hydraulic System Reservoirs	Hydraulic System Accumulators (Filler Valves)	Survival Kit Emergency Oxygen Bottles	Radome Anti-Ice
REPLENISHING AGENT	Engine Fuel	Dry Air Or Nitrogen	Red Hydraulic Fluid	Dry Nitrogen Or Clean Dry Air	Gaseous Oxygen	Water (40%) Glycol (60%) (By Volume)
SPECIFICATION	Refer To Combustion Start Second Attempt, Section VII		Mil-H-5606		BB-O-925 Grade A Type-I	Mil-A-8243A
CAPACITY		Placarded	Fill To 1/4 Inch Below Full Mark	750 Psi At 21°C (70°F)	1800 Psi	2 U.S. Gal.
LOCATION	Right Engine Access Compartment	Right Engine Access Compartment	RAT Compartment Right Side	RAT Compartment Right Side	Survival Kit Assembly	Forward Electronics Compartment Right Side

21832-1

Figure 1-33

against the seat, he may not be able to unlock the harness without first loosening it slightly by using the adjustment buckles.

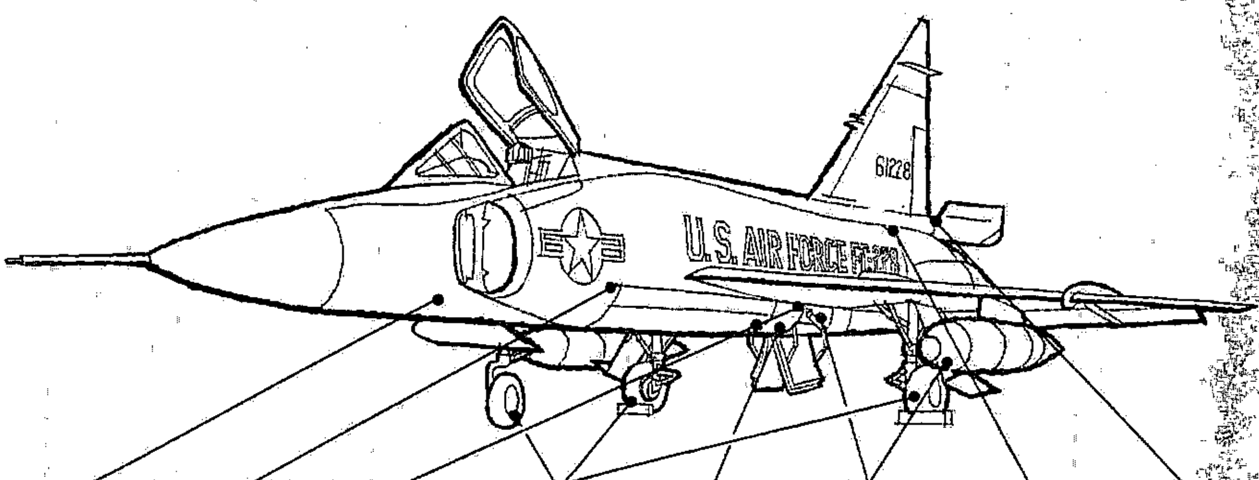
Note

A preflight check of the shoulder harness inertia reel can be made by simply giving the harness a quick jerk. The reel should lock within approximately one-half inch or less of cable movement. Pulling the harness in any direction, up, down, or sideways will lock the reel.

AUXILIARY EQUIPMENT

Information concerning the following auxiliary equipment is supplied in Section IV: Air-Conditioning and Pressurization System, Defogging System, Anti-Icing Systems, Communications and Associated Electronic Equipment, Lighting Equipment, Liquid Oxygen System, Automatic Flight Control System, Navigation Equipment, Armament Equipment, Fire Control System, Miscellaneous Equipment.

servicing diagram



Battery	Oxygen System (Liquid)	High-Pressure Pneumatic System	Tires	External Power	Fuel Tanks	Engine Oil Tank	Drag Chute
Distilled Water	Oxygen	Pneumatic Pressure	Pneumatic Pressure	115/200V 400 Cps 3 Ph AC 28V DC	Fuel	Oil	Drag Chute Pack
	BB-O-925 Grade A Type II	Ma-1 Cart	Nose—175 Psi Main—195 Psi ▲1		Grade JP 4 Mil-J-5624 ▲2	Mil-L-7808	Pack In Accordance With T.O. 14D 1-3-93
	4-4½ Liters	3000 Psi			1126 U.S. Gal. (Int.) 436 U.S. Gal. (Ext.)	5.5 U.S. Gal.	
Nose Wheel Well	Fuselage Forward Left Side			Main Wheel Well Left Side	Main Wheel Well Aft Outboard (External) Left Side (Internal)	Upper Fuselage Left Side	Above Tail Cone

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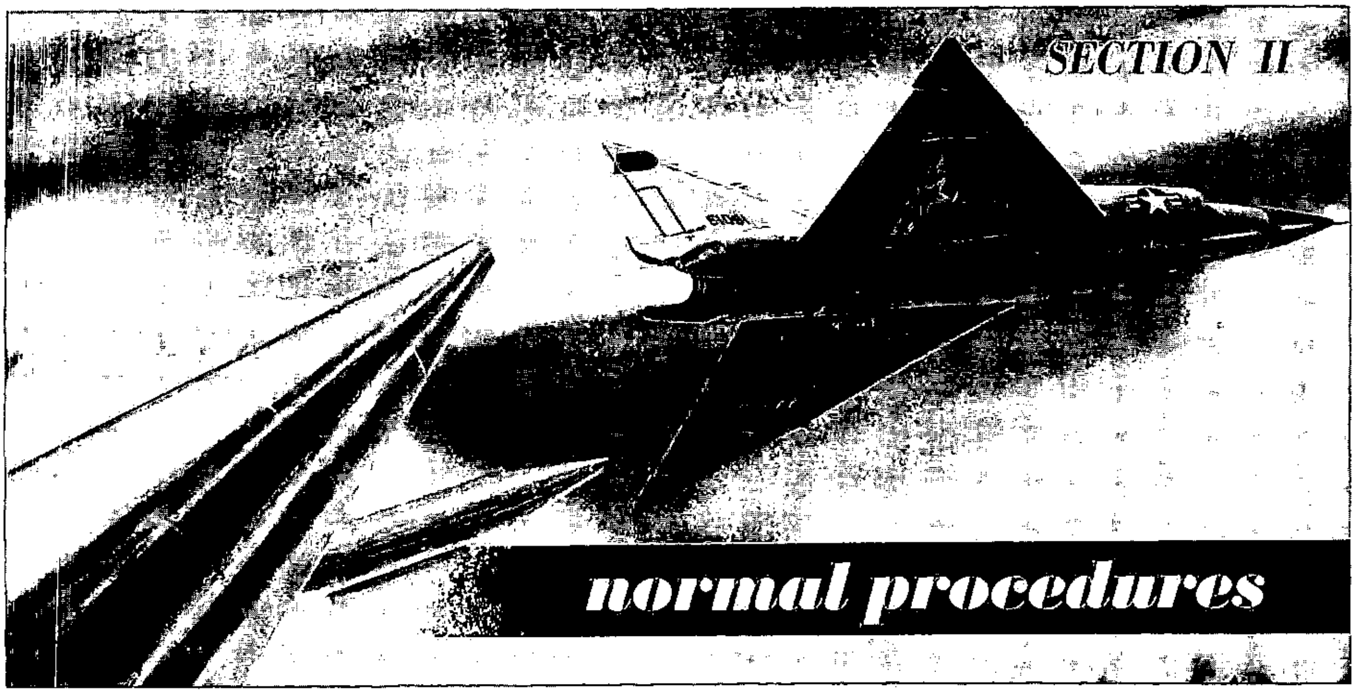

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PREPARATION FOR FLIGHT

The normal procedures in this Section have been presented to include items to place the airplane in an alert status and cocked for a scramble. In order to reduce the resulting bulk of the pilot's check list, the INTERIOR INSPECTION (ALL FLIGHTS) is arranged to include the alert cocking check list items so as not to sacrifice clarity or standardization. The airplane when cocked is ready for STARTING ENGINE (SCRAMBLE) with either external power or battery. However, external power is required to place the airplane in alert status and cocked for STARTING ENGINE (SCRAMBLE). All other phases of the check list remain unchanged.

FLIGHT RESTRICTIONS

Refer to Section V for operating restrictions and limitations.

FLIGHT PLANNING

Refer to the Appendix for information on required fuel, airspeed, thrust settings, etc. necessary to complete the proposed mission.

TAKEOFF AND LANDING DATA CARD

Complete Takeoff and Landing Data Cards in the Pilot's Abbreviated Check List, T.O. 1F-102A-(CL)1-1, by following instructions in the Appendix.

WEIGHT AND BALANCE

Refer to Section V for WEIGHT LIMITATIONS. For detailed loading information, refer to the Handbook of Weight and Balance Data, T.O. 1-1B-40. Before each flight check the following:

1. Takeoff and anticipated landing gross weights.
2. Weight and balance clearance (Form 365F).

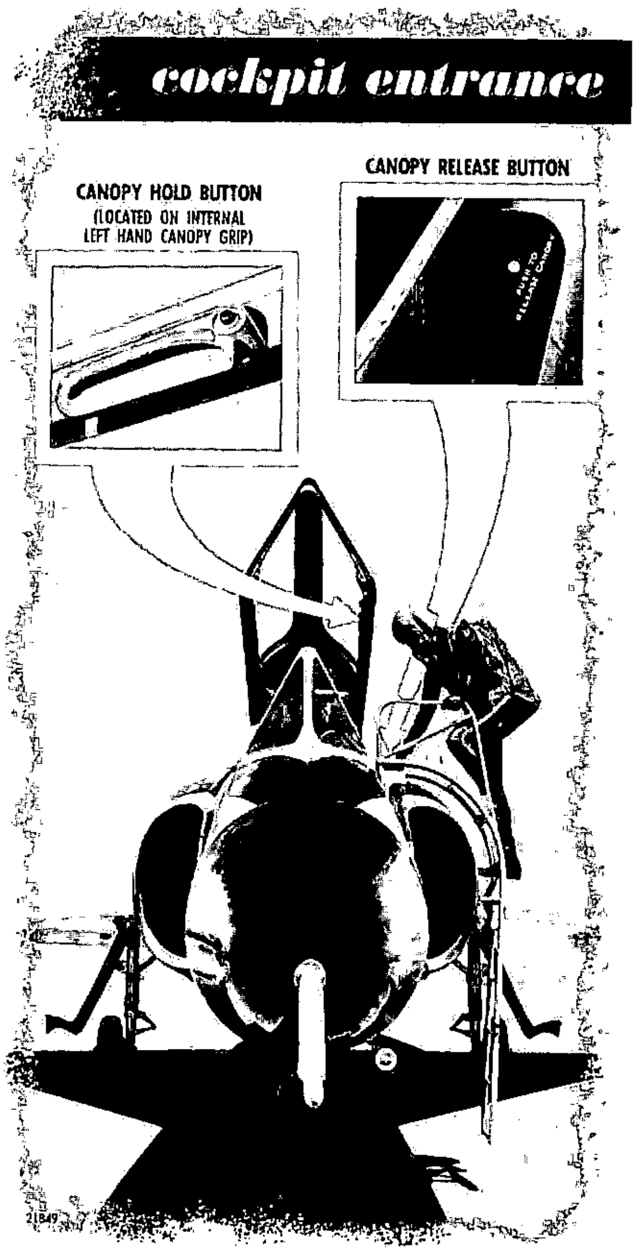


Figure 2-1

PREFLIGHT CHECK

ENTRANCE

Cockpit entry is accomplished from the left side of the airplane (figure 2-1) by use of a ladder. To open the canopy, press the external canopy release button and manually raise the canopy. On later airplanes, the canopy should be raised prior to connecting external power. However, if external power is applied before opening, the canopy can be manually raised approximately six inches. It will then be necessary to reach inside and depress the canopy hold button to completely raise the canopy.

BEFORE EXTERIOR INSPECTION

WARNING

Do not store any luggage, papers, personal effects, etc., in any of the electronic compartments at any time.

1. Canopy support tool — In place.
2. Form 781 — Check.
   Check Form 781 for engineering status.
3. Servicing — Check.
   Make certain the airplane has been serviced with fuel, hydraulic fluid, oxygen, air, armament and any special equipment required to complete the proposed mission. For detailed replenishing requirements see figure 1-33.
4. FLIPS — Check.
   Check that flight information publications required for the mission, are available.
5. Windshield — Check.
   Check that windshield delamination (marred windshield glass) does not extend more than two inches from the outer edge of the glass panels or one inch from sectional tension straps. If delamination exceeds these limits, the windshield should be considered unsatisfactory for flight.
6. Upper electronics compartment doors — Secured.
7. Canopy trailing wire — Coiled (if installed).
   Check canopy trailing wire for proper coiling.

WARNING

If canopy trailing wire is not properly coiled, ejecting the canopy may not arm the ejection seat, preventing seat ejection.

8. Throttle — OFF.
9. Survival kit — Check (if installed).
   a. Check pressure in emergency bailout bottle—1800 psi.
   b. Survival kit lid—Fully locked.
   c. Green knob assembly—Intact.
   d. Personal equipment bundle connection—Check.
10. Ejection seat safety pin—Installed, streamer visible.
11. Armament safety switch — SAFE.
12. Armament selector switch — SNAKE.
13. Igniter control switch — TRAINING (safety-wired).
14. Radar master switch — OFF.

exterior inspection

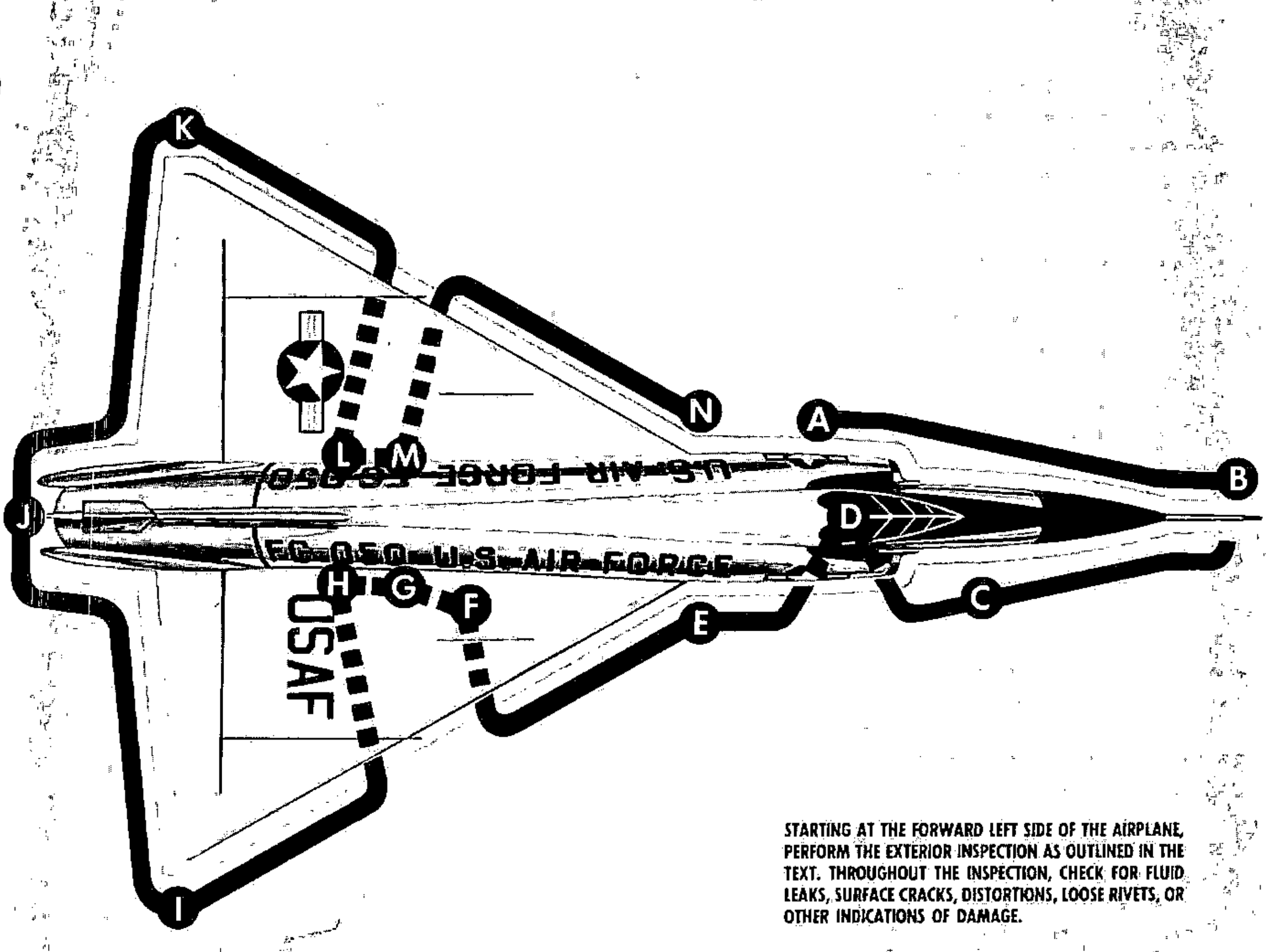


Figure 2-2

15. Canopy jettison handle — DETENT position, release button OUT.
16. RAT handle — Fully in.
17. Canopy ground maintenance safety pin—Removed.
18. Safety belt ground safety pin – Removed.
    Check that the M-12 safety belt initiator safety pin has been removed. This pin is provided for use during maintenance and, if installed, maintenance personnel should be consulted regarding the status of the ejection system before occupying the ejection seat.
19. Battery switch — OFF.
20. Master switch — NORMAL.
21. Flashlight — Check (if required).

EXTERIOR INSPECTION

Perform the following checks in accordance with figure 2-2.

A. Forward Left Side

1. Intake duct — Condition & no loose articles.
2. Boundary layer duct — Clear.
3. Forward electronic bay doors — Secure.
4. Rocket jump angle vane — Condition.

B. Nose

1. Static ports — Clear.
2. Radome — Condition & security.
3. Glycol ring — Undamaged & general condition.

4. Mast and pitot tube — Security, condition, & cover removed.
5. Sideslip angle transducer probe — Free & clear*.

C. Forward Right Side

1. Forward electronic bay doors — Secure.
2. Boundary layer duct — Clear.
3. Intake duct — Condition & no loose articles.

D. Nose Wheel Well

1. Tire — Check slippage, wear, cuts, and inflation.
2. Nose gear strut — Check five to six inches extension.
3. Steer damper scissors pin — Removed.
4. Nose wheel steering hinge pin — Check for absence of lateral movement.
5. Left-hand aft circuit breakers — In.
6. Left-hand dust cover — Secure.
7. Nose gear door lock — Remove (if installed).
8. Radar ground cooling door — Secure.
9. Battery — Secure.
10. Right-hand forward circuit breakers — In.
11. Right-hand dust cover — Secure.
12. Taxi light — Condition and security.
13. Nose door seal — Condition.
14. Nose landing gear safety lock pin — Removed.

E. Right Side

1. Intermediate electronic bay door — Secure (punch marks aligned with stripe).
2. Armament bay doors — Secured.
3. Position lights — Condition & security of glass.

F. Ram Air Turbine Compartment

1. RAT door — Check for free play.
2. Primary & secondary hydraulic fluid levels — Check.
3. Primary & secondary hydraulic accumulator pressure — 750 psi.
4. Caps on reservoirs, safety pins in and clipped (if applicable).
5. RAT rotor — Turns freely.
6. RAT — Condition of blades.
7. RAT door — Close.
   Close the RAT door by a gradual increase in force until click is heard (40-60 lbs).
8. RAT door test hook—Pull (approximately 80 lbs).
   Pull the RAT door test hook to check the RAT door for security.
9. RAT door test hook — Remove.

G. Right Main Wheel Well

1. Circuit breakers — In.
2. Emergency ac generator — Check.
3. Ground static wire — Check for security.
4. Chocks — In place.
5. Tire — Check slippage, wear, cuts, inflation, & wheel tie bolt torque marks.
6. Brake & hydraulic lines — Check.
7. Gear strut extension — Check five to six inches extension between scissors torque arm, center to center.
8. Landing gear fairing & door — Condition & security.
9. Landing light — Condition & security.
10. Gear ground safety lock pin — Removed.
11. Combustion starter exhaust port — Clear.
12. Oil overboard vent — Clear.

H. Engine Access Compartment

1. Constant-speed drive accumulator pressure — Check.
2. Hydraulic & fuel leaks — Check.
3. Bellcrank cable and drum — Check visually.
4. Elevator intelligence unit — Pin flush with case.

Note

Refer to OTHER OPERATIONAL LIMITATIONS, Section V, for limitations if pin is not flush with case.

5. Starter and duct installation — Free of contamination.
6. General condition — Check.

I. Right Wing

1. Fuel drain — Unobstructed.
2. External tank — Check (if installed).
   Visually check fuel level. If quantity is questionable, use dip stick. Cap secured, ground safety lock pin removed (if installed).
3. Wing leading edge & tip — Condition.
4. Position lights — Condition & security of glass.
5. Trailing edge & elevon — Condition.

J. Tail Section

1. Speed brakes — Condition.
2. Drag chute — Stowage & pin.
3. Rudder — Condition.
4. Position lights — Condition & security of glass.
5. Tailpipe & exhaust nozzles — Check.

CAUTION

Check for puddles at drain lines and in the tailpipe as unburned fuel creates a fire hazard.

*Some airplanes.

K. Left Wing

1. Trailing edge & elevon — Condition.
2. Position lights — Condition & security.
3. Engine oil cap access door — Secure.
4. Wing tip & leading edge — Condition.
5. External tank — Check fuel level, cap secured, pin removed.

   Visually check fuel level. If quantity is questionable, use dip stick. Cap secured, ground safety lock pin removed (if installed).
6. Fuel drain — Clear.

L. Left Main Wheel Well

1. Chocks — In place.
2. Landing gear fairing & door — Condition & security.
3. Landing light — Condition & security.
4. Ground static wire — Check.
5. Tire—Check slippage, wear, cuts, inflation, & wheel tie bolt torque marks.
6. Brakes & hydraulic lines — Check.
7. Gear strut extension — Check five to six inches extension between scissors torque arm, center to center.
8. Ground safety switch — Check.
9. Armament control panel — Check.
10. Fuel filler cap — Secure.
11. Engine ignition disconnect switch — ON.
12. Combustion starter manual air valve — GROUND (if external air source is available).
13. Circuit breakers — IN.
14. High-pressure pneumatic system pressure — 3000 psi, filler cap secure.
15. Gear ground safety lock pin — Removed.
16. External tank jettison warning lights — Check (if installed).

    Apply dc power and check that external tank jettison warning lights are not illuminated, then press-to-test lights for satisfactory operation. Disconnect dc power.
17. External tank safety pin — Removed (if installed).

    If warning lights are not illuminated and are press-tested satisfactorily remove safety pin.
18. Overrun barrier probe* — Physically check in fully up and locked position.

WARNING

To prevent inadvertent actuation of the overrun barrier probe, do not remove the ground safety lock pin unless barrier probe is in the fully up and locked position.

19. Overrun barrier probe ground safety lock pin — Removed (if installed).*

M. Left Side

1. Aft electronic bay door — Secure.
2. Armament bay doors — Secure.
3. Emergency canopy jettison lanyard — Stowed & secure.
4. Position lights — Condition & security of glass.
5. Ram air intakes — Covers removed & condition.
6. Liquid oxygen build-up & vent handle — BUILD-UP.
7. Oxygen filler access door — Secured.

INTERIOR CHECK (ALL FLIGHTS)

Note

Each of the following checks should be performed when an external power source is available for interior inspection. Items marked with the symbol ▲ preceding the step cannot be performed if making the interior check with battery power prior to battery start. These items (▲) should be checked after battery start. (Refer to INTERIOR CHECK AFTER BATTERY START, this Section.) The alert cocking and the scramble start as presented in this check list are predicated on the following: The preflight and cocking procedures must be performed with external power available. The scramble start can be accomplished on either external or battery power.

General

1. Survival kit — Attach (if installed).

   Attach survival kit parachute attaching straps to parachute harness if survival kit is installed. Pull straps snug. Pull fastener end of straps. If slippage occurs, the attachment straps are installed backwards.
2. Safety belt & shoulder harness — Secure.
   a. Do not accomplish when cocking airplane.

WARNING

The force required to open the safety belt should not be less than five pounds pull nor exceed 20 pounds pull to insure proper operation.

**AF 57-770 & on, & airplanes modified by TCTO 1F-102-658.*

Note

The safety belt must be fastened snugly to prevent the possibility of the survival kit oxygen quick-disconnect from separating from the oxygen connection on the seat during negative g maneuvers.

3. Shoulder harness handle — UNLOCKED.

   Check for freedom of movement of the shoulder harness. If the harness has been locked by a rapid pull it is necessary to cycle the shoulder harness inertia reel handle to LOCKED, then to UNLOCKED.

4. Zero delay lanyard hook — Attach (if installed).

   If "one & zero" escape system is installed, attach lanyard hook to ripcord.

5. Emergency bailout bottle hose — Connect.
6. Personal equipment leads — Connect.
7. Personal equipment lead bundle — Strapped to parachute harness (if survival kit is installed).
8. ▲ External power — Connected (if available).

   When external power is connected on some airplanes*, the dc power failure warning light and the ac power failure warning light will illuminate and will remain on until the engine is started and the external power unit is disconnected.

9. ▲ AC voltage — 105-125 volts.

   To determine that all three ac phases of the external power source are available for booster pump operation, place the voltmeter selector switch to PHASE A, B, and C and note 105-125 volts indication on each phase.

CAUTION

Do not operate the fuel boost pumps if the voltmeter readings are not as specified.

10. Battery switch — ON (if no external power source is available).
11. Seat & rudder pedals — Adjust.

    Adjust seat vertical position by use of the vertical adjustment switch and rudder pedals by crank.

WARNING

On airplanes without the survival kit, the oxygen lead may separate at the quick-disconnect fitting when the seat is adjusted to the fully up position and must be checked after seat adjustment.

Left-Hand Console

1. LH aft circuit breakers — In.
2. Spare lamps — Check.
3. Anti-g suit valve — Set.
4. Mask defog — Set.

   If partial pressure suit is worn, set mask defog as desired.

5. ▲ Fuel quantity — Check.

   Select each position on the fuel quantity selector switch and check fuel quantity gage for correct readings.

6. ▲ Fuel quantity gage test button — TEST (some airplanes).
7. Fuel selector switch – OFF (unless safety-wired†) if external power is available, or ENGINE if battery start is anticipated (some airplanes).

   Fuel shutoff valve switches — CLOSE if external power is available, or OPEN if battery start is anticipated (other airplanes).

8. ▲ Fuel system — Check.

   a. Fuel boost pump pressure-low warning lights — On.

   b. Fuel boost pump — fuel shutoff valve electrical interlock — Satisfactory operation** (do not accomplish this step if the fuel selector switch is safety-wired†).

      (1) To determine that fuel boost pump — fuel shutoff valve electrical interlock is operating satisfactorily, place the left-hand fuel boost pump switches ON and note that left-hand boost pump pressure-low warning light remains on.

CAUTION

If the boost pump pressure-low warning light does not remain on, it is an indication of a malfunction in the electrical interlock, since the pumps should not operate unless the shutoff valves are fully open. If a malfunction exists, the cause should be established and corrected before proceeding further.

      (2) Repeat step (1) for right-hand boost pumps.

   c. Fuel boost pump switches — OFF.

   d. Fuel selector switch — ENGINE (some airplanes).

      Fuel shutoff valve switches — OPEN (other airplanes).

   e. Fuel boost pumps — Check.

**AF 56-1275 thru -1316, -1332 & on.*
***AF 56-1206 & on, & airplanes modified by TCTO 1F-102-651.*
†In accordance with TCTO 1F-102-686.

(1) Observe that both boost pump pressure-low warning lights are illuminated.

(2) Left-hand forward boost pump switch ON and check that left-hand boost pump pressure-low warning light extinguishes within five seconds.

CAUTION

The warning light will normally extinguish within two seconds. However, if the light is still illuminated after five seconds, the boost pump switch should be turned OFF and the cause established and corrected before attempting to operate the other pumps.

(3) Left-hand forward boost pump switch — OFF.

(4) Repeat steps (1) through (3) for the left-hand aft, right-hand aft, and right-hand forward pumps.

f. Left-hand aft boost pump switch — ON.
At the completion of the above checks, the left-hand aft boost pump should be turned ON. Do not turn on the other boost pumps until the airplane generator is on the line and three-phase ac has been checked.

▲ 9. UHF radio — ON (if needed).

10. Speed brakes switch — NEUTRAL.
 a. Speed brakes switch – IN (for cocking).

11. Fuel control switch — NORMAL (guard closed).

12. Oxygen system — Check as outlined in Section IV. (Refer to OXYGEN SYSTEM, Section IV, for additional information and detailed checks on the liquid oxygen systems.)

Left-Hand Auxiliary Panel

1. Landing gear handle — Down.
2. Landing gear warning light(s) — Check.
 Check that the landing gear handle light and on some airplanes,* a red bar warning light is out. Depress landing gear warning test button and lights should illuminate. On airplanes with the red bar warning light, the audible signal should be heard in the radio headset.

CAUTION

Do not depress the landing gear warning light test button and the red bar warning light at the same time as damage may result to the audible signal generator.

3. Landing gear position indicators — Wheels or green lights.

▲ 4. Landing & taxi lights — Check (if required).

5. LH forward circuit breakers — In.
6. Cabin air switch — PRESS.
7. Landing gear emergency extension handle — In & secure.
8. Cabin altimeter — Check.
9. Drag chute handle — In & secure.
10. External tank fuel transfer switch — OFF (if installed).

Instrument Panel

1. Directional indicator (slaved) slaving switch — NORMAL (if installed).
2. Clock — Set.
3. Flight instrument—Check & Set.
 a. Directional indicator (slaved) (stabilizing).
 b. Attitude indicator—Check as follows:
  (1) Power warning flag for retraction within specified time limit.
  (2) Horizon line for proper attitude and freedom from oscillation (energized).
  (3) Horizon line for response to trim knob.

WARNING

If the "OFF" flag requires longer than 2½ minutes to retract or any oscillations are noted on the indicator after the "OFF" flag retracts, possibility of a malfunction exists. Either of the above is cause for rejection of the indicator and should be noted on Form 781.

 c. Vertical velocity indicator, Machmeter-airspeed indicator, and turn-and-slip indicator indicating proper static conditions.
 d. Altimeter and cockpit pressure altitude gage readings correspond to pressure conditions.

WARNING

The barometic setting knob can be turned so that the barometric disc will rotate through 360°. If the correct altimeter setting is then established, the altimeter will indicate 10,000 feet in error.

4. Engine fire warning test switch — FIRE, then OVHT.
 Place engine fire test switch to FIRE and check fire warning light on steadily. Place switch to OVHT and check for flashing light.

Airplanes modified by TCTO 1F-102-728.

5. Engine instrument — Check.
   Check tachometer, exhaust gas temperature gage, fuel flow indicator, and pressure ratio gage all reading zero.
6. Hydraulic pressure-low warning light — On.

Right-hand Auxiliary Panel

▲ 1. Thunderstorm lights — Checked, then OFF (if not required).
2. Canopy unlocked warning light – On.
3. Master warning system — Test.
   Place master warning light test switch to TEST and check that all lights on the warning light panel and the master warning light illuminate.

Right-hand Console

1. AC generator switch — ON; guard closed.
2. AC bus switch — NOR; guard closed.
3. DC generator switch — ON.

▲ 4. Navigation receiver — ON and check.

▲ 5. J-4 compass — Check.
   a. Latitude — Set.
   b. Function selector switch DG.
   c. Precess compass card approximately 45°.

CAUTION

Do not operate "set" switch continuously for more than 30 seconds to avoid overheating the slew motor.

   d. Function selector switch MAG, heading checked.

▲ 6. IFF — STBY; check mode code.
   a. IFF — NORMAL (for cocking).
7. Standby compass light — Check (if required).
8. RH forward and aft circuit breakers — In.

Utility Switch Panel

1. Flight mode selector switch — DIRECT MAN*.
2. Pitch damper switch — OFF**.
   Check pitch damper OFF and observe that pitch damper, AFCS, altitude hold, and AILAS switches are in the OFF positions.
3. Nesa switch — NORMAL (some airplanes); both Nesa switches — ON (other airplanes).
4. Cabin temperature control knob — AUTOMATIC.
5. Canopy defog switch — Set.
6. Windshield rain clear switch — OFF (some airplanes); STBY (other airplanes).
7. Cockpit no-fog/vent suit switch — OFF (if installed).

▲ 8. Pitot heat switch – Check, then OFF.
   a. Pitot heat switch — ON (for cocking).
      Have crew chief check pitot tube for heating.
9. Anti-ice switch – OFF.

Lighting Control Panel

▲ 1. Position lights — Check (if required).
   Test position light switches for FLASH, STEADY, DIM, and BRIGHT (some airplanes) or FORMATION and NAVIGATION (other airplanes).

Note

The anticollision lights will illuminate when the position light switch is placed in NAVIGATION position. Use of the anticollision lights on the ground shall be kept to an absolute minimum. The excessive heat created on the ground is detrimental to bulb life and increases maintenance problems, and during ground emergencies the operating light could confuse rescue operations since emergency ground vehicles use a similar light.

▲ 2. Cockpit lights — Check (if required).
   Check operation of cockpit lights and set rheostats as desired (OFF or ON).
3. Left-hand aft boost pump switch – OFF (for cocking).
4. UHF radio – OFF (for cocking).
5. External power — Disconnected (for cocking).

BEFORE STARTING ENGINE

An outside air source is normally utilized for starting; however, the start may be accomplished by using the airplane high-pressure pneumatic system by opening the combustion starter manual air valve in the main wheel well. An external electrical power source should be connected for starting to conserve battery power.

Note

- If it is necessary to start and no external ac power source is available, a start may be accomplished by turning the boost pumps off and following the normal start procedure. There will be no indication of fuel flow, hydraulic pressure, or fuel quantity. Starting without ac power may be harmful to the engine-driven fuel pump because of cavitation and the resulting lack of lubrication.
- Using the battery for starting will cause a heavy drain of battery power.

*AF 53-1791 thru 55-3379 unless modified by TCTO 7F-102A-546.
**AF 55-3380 & on, & airplanes modified by TCTO 1F-102A-546.

CAUTION

- If continuous ground operation at low rpm (below 80%) is required, it will be necessary to pressurize the hydraulic reservoirs to 50 (±5) psi before starting the engine in order to prevent cavitation of the hydraulic pumps.
- Before starting engine, determine that wheels are firmly chocked and hold wheel brakes on.
- Before starting engine, insure that no loose items of personal equipment or airplane forms are on instrument hood and all loose items on operator are secured. This is required to prevent ingestion of loose items into the engine as a result of engine operation when the canopy is open.

STARTING ENGINE

Note

- Each of the following steps should be performed when starting engine with an external power source. Items marked with the symbol ▲ preceding the step cannot be performed if starting with battery.
- For engine starting restrictions at remote bases where a compressor air source is not available refer to HIGH PRESSURE PNEUMATIC CAPABILITY, Section VII.
- When making a battery start using the combustion starter, if the start is aborted for any reason and a second start is desired, it will be necessary to recharge the starter fuel flask prior to the second start. The fuel flask will automatically charge if external power is supplied and the left-hand aft fuel boost pump is turned on. However, if no external power is available, refer to COMBUSTION START—SECOND ATTEMPT, Section VII.

Note

It is recommended, if practical, to preheat the starter compartment when ambient air temperature is at 0°C or below. This will thaw out any possible frozen condensation which may have accumulated in the starter overspeed switch mechanism since last engine shutdown.

CAUTION

Before starting engine, determine that wheels are firmly chocked, and hold wheel brakes on.

1. Starting air — Connected.
   Signal crew chief to connect compressed air for starting.

DETERMINE THAT DANGER AREAS FORE AND AFT OF THE AIRPLANE ARE CLEAR OF PERSONNEL, AIRCRAFT AND VEHICLES, REFER TO FIGURE 2-3. SUCTION AT INTAKE DUCTS IS SUFFICIENT TO KILL OR SERIOUSLY INJURE PERSONNEL PULLED AGAINST OR DRAWN INTO THE DUCTS. THE DANGER AREA AFT OF THE AIRPLANE IS CREATED BY THE EXHAUST VELOCITY AND TEMPERATURE.

22075

Note

If no external compressed air source is available, the combustion starter manual air valve should be placed in the AIRCRAFT position immediately prior to starting to provide starting air.

2. Danger areas — Clear.

WARNING

Prior to any ground start make sure that all ground personnel are forward of the main landing gear and outboard of either wing tip before moving the throttle out of the OFF position. Ground crew should not re-enter the wheel well area until the starting cycle is completed.

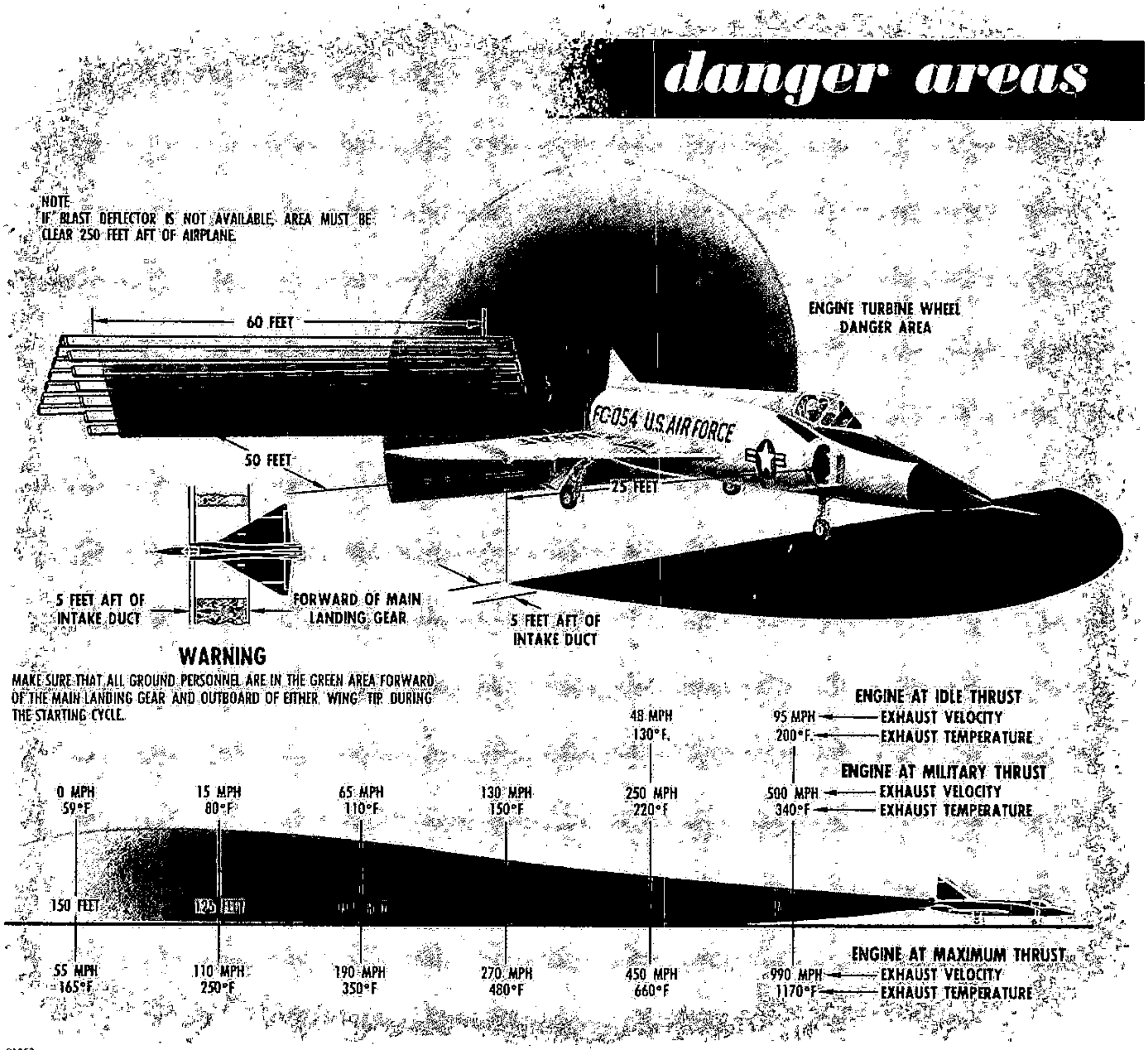


Figure 2-3

3. Throttle — START.
   Move throttle outboard to START position.
4. Ignition button — Depress.
   Hold throttle outboard and ignition button depressed until an indication of rpm is evident on the tachometer or until positive indication of hydraulic pressure. This time should not exceed six seconds, as holding for a longer period may cause condensation to form on the starter igniter plugs and thus reduce the possibility of ignition. The engine is being air-motored only at this point. For air motoring restrictions at remote bases where a compressed air source is not available, refer to HIGH-PRESSURE PNEUMATIC CAPABILITY, Section VII.

CAUTION

Once depressed the ignition button must be held depressed until the starting sequence is complete to avoid closing the starter shutoff valve and terminating ignition. If ignition button is inadvertently released, the throttle must be returned to OFF to shut off fuel flow to the combustion chambers, as starter and ignition operation cannot be reinstated without repeating the starting procedure. Do not attempt another start until fuel drainage has ceased and engine is visually checked for trapped fuel.

5. RPM — Positive rpm or hydraulic pressure indication.

WARNING

Release ignition button immediately after starter light-off if an rpm reading is not evident on the tachometer. Do not move the throttle inboard to the OFF position with the ignition button depressed. This could result in disintegration of the combustion starter. No rpm reading indicates the starter failed to engage the engine. A maximum of two attempts should be made, but if still unsuccessful, the operation should be discontinued until the cause has been established and corrected.

6. Throttle — OFF, then IDLE.
   As soon as a definite rpm indication is noted on the tachometer, move throttle inboard to the OFF position, then to IDLE.

WARNING

If starter ignition is not heard almost immediately or if a rapid increase in engine rpm is not noted, the start should be aborted immediately in accordance with the above procedure.

▲ 7. Fuel flow — Check indication.

8. Exhaust gas temperature gage — Shows increase.
   During a satisfactory start, a lightup occurs within 20 seconds after throttle is advanced to IDLE. Lightup can be noted by an indication of exhaust gas temperature.

9. Oil pressure low warning light — Out (approximately 30% rpm; at least by idle rpm).

10. Ignition button — Release at 33% rpm.

CAUTION

- If the engine does not light up when 33% rpm has been reached, or within 20 seconds after throttle is advanced to IDLE, the start should be aborted. If engine rpm of less than 25% is obtained, this indicates possible starter malfunction. Abort start and have malfunction corrected prior to attempting another start. The starter may be stopped at any time by releasing the ignition button located on the throttle. Shut down the engine by using the UNSUCCESSFUL START procedure, this Section.
- Ignition operation is limited to three minutes of continuous operation to prevent overheating and subsequent damage to the ignition unit.

11. Exhaust gas temperature — Stabilized.
    Check that exhaust gas temperature rises normally and is within limits.

CAUTION

A possible hot start can be anticipated by observing a rapid increase to 500°C and temperature is still rising. When a hot start occurs, record on Form 781 the maximum exhaust temperature reached and the duration the exhaust temperature exceeded engine operating limits shown in figure 5-2.

12. Idle rpm — 55 to 65% — Check.
    Check that engine accelerates to idle rpm (approximately 55 to 65% rpm).

Note

Starter will automatically cut out at approximately 35% rpm by means of a centrifugal switch or when the ignition button is released.

13. Engine fuel pump failure warning light — Off.

CAUTION

- If the engine fuel pump failure light comes on, shut down the engine. The light will indicate that the engine stage of the fuel pump has failed and the afterburner stage of the fuel pump is supplying fuel to the engine fuel system. Investigate cause of light indication.
- If airplane pneumatic system air was used in starting, the combustion starter manual air valve must be returned to GROUND position after start is completed to prevent loss of pneumatic pressure through leakage and to prevent actuation and possible disintegration of the starter in the event that an air start becomes necessary.

▲ 14. LH aft fuel boost pump switch — OFF.
    Fuel boost pumps should be turned OFF prior to disconnecting external power and should remain off until all phases of the airplane's ac voltage have checked satisfactorily.

15. UHF radio — OFF.

▲ 16. Compressed air and external power — Disconnected.

17. Throttle — 80%.

Note

Throttle must be advanced to 80% momentarily to insure that the hydraulic reservoirs are pressurized to 50 (±5) psi as required to prevent pump cavitation. As an alternative, pressure from an external source can be used to pressurize reservoirs before starting engine. If the engine must be operated at low rpm for an extended period of time, the throttle should momentarily be advanced to 80% at approximately five-minute intervals.

UNSUCCESSFUL START

1. Throttle — OFF.

   Throttle should be placed in OFF position to shut off fuel flow to the engine and reduce the possibility of fire.

2. Check for fire.

   Check instruments and have crew chief check for visible evidence of fire.

Note

Investigate cause of the unsuccessful start. Do not attempt another start until cause has been determined and corrected. If it is determined another start can be made, wait 30 seconds for fuel drainage before attempting another start. Have engine visually checked for trapped fuel. Check for zero rpm before re-starting.

HUNG START OR SLOW START

A hung start is indicated by failure of rpm to increase after lightup with exhaust temperature remaining within limits. A slow start is similar to a hung start and is evidenced by a slow but continuous acceleration of rpm to idle after lightup. When a hung start or slow start is experienced, proceed as follows:

1. Tachometer–Between 25 and 55% rpm.
2. Fuel control switch–EMERGENCY.

CAUTION

Do not place fuel control switch to EMERGENCY if engine rpm is below 25%. This could introduce an excess of fuel into the engine and cause exhaust gas temperature to exceed limits. It may be necessary to advance the throttle slightly past IDLE while in EMERGENCY to obtain IDLE rpm of 55 to 65%.

3. Fuel control switch — NORMAL at 55 to 65% rpm.

CLEARING ENGINE

1. Compressed air — Connected.

   If no external compressed air source is available, the combustion starter manual air valve should be in AIRCRAFT position to provide air for clearing.

2. Engine ignition disconnect switch — OFF.

   Have crew chief place the engine ignition disconnect switch in the left-hand main wheel well to OFF.

3. Throttle — OFF.
4. Fuel selector switch — ENGINE (some airplanes).

   Fuel shutoff valve switches — OPEN (other airplanes).

   Both fuel valves should be open unless internal engine fire is suspected. This permits lubrication of the fuel control unit.

5. Throttle — START.
6. Ignition button — Depress and hold.
7. RPM — Check positive indication.

WARNING

Release ignition button immediately if an rpm reading is not evident on the tachometer. Do not move the throttle inboard to the OFF position with the ignition button depressed. This could result in disintegration of the combustion starter. No rpm reading indicates the starter failed to engage the engine. A maximum of two attempts should be made, but if still unsuccessful, the operation should be discontinued until the cause has been established and corrected.

8. Throttle — OFF; insure that engine clears.

   Position throttle inboard to OFF while holding the ignition button depressed to energize the starter. Hold ignition button depressed until combustion starter has completed its cycle.

CAUTION

The following cooling periods must be observed to prevent damage to the starter from overheating:

a. A second start may be attempted any time after returning the throttle to OFF for approximately 15 seconds.

b. A second start within a 15-minute period is considered a double start. A double start shall be followed by a one-hour cooling period before the next start.

c. A second start between 15 and 40 minutes from the first shall be followed by a 40-minute cooling period before the next start is attempted.

d. Any start after a prescribed cooling period or 40 minutes after a first start shall be considered as a first start.

9. Ignition button — Release.

After engine clears, release ignition button. Have crew chief place the combustion starter manual air valve to GROUND if airplane compressed air was required for clearing.

ENGINE GROUND OPERATION

Engine warmup is not required under normal conditions. After the engine stabilizes at idle, it may be operated at full thrust; however, cooling limitations (ground operation) must not be exceeded.

CAUTION

- Insufficient tire traction requires the use of an airplane restraining bridle when using afterburner on the ground.
- If the throttle is inadvertently retarded to OFF, a flameout occurs immediately. Do not reopen throttle, as relight is impossible and resultant flow of unburned fuel into the engine creates a fire hazard.

BEFORE TAXIING

ELECTRICAL POWER SUPPLY SYSTEM

DC System

1. DC generator output — Check.

   After external power has been disconnected, check that dc operated equipment or lights are operating from the dc generator and that dc power failure light is out. The dc generator should be operating when engine speed is above approximately 30% rpm.
2. Battery switch — ON.
3. DC generator switch — OFF.
4. Battery output — Check.

   Check that dc operated equipment or lights are operating and that dc power failure light is illuminated.
5. DC generator switch — ON.

AC System

1. AC voltage (all phases) — 105-125 volts.

   To check operation of main ac generator, the ac bus switch should be in NOR position. All positions of the ac voltmeter selector switch should give the same voltmeter readings in 105-125-volt range except that the 26v position should read above 0 but not over 50 volts.
2. AC generator switch—OFF.
3. AC bus switch—EMER (some airplanes); RESET, then ON (other airplanes); 103-140 volts.

   Check operation of emergency ac generator. All positions of the ac voltmeter selector switch should give voltmeter readings of 103-140 volts except that the 26v position should read above 0 but not over 50 volts.
4. AC bus switch—NOR.
5. AC generator switch—RESET, then ON.
6. Transformer-rectifier—Check (some airplanes*).

   Determine that the transformer-rectifier is operative and providing dc power to the emergency dc bus as follows:

Airplanes modified by TCTO 1F-102-727.

a. AC generator switch — OFF.

Placing the ac generator switch to OFF, prevents premature energizing of the emergency dc bus by the transformer-rectifier.

b. DC generator switch — OFF.

With the dc generator OFF, the dc nonessential bus and the emergency dc bus are disconnected from the dc essential bus. Check that the battery is functioning.

c. AC bus switch – EMER (check for 0 volts).

With the ac bus switch in EMER, (do not hit the RESET position) check for 0 volts. If ac power is available, a malfunction is indicated.

d. AC bus switch – RESET, then EMER.

All positions of the ac voltmeter selector switch should give voltmeter readings of 103-140 volts except that the 26v position should read above 0 but not over 50 volts.

Note

Emergency ac control is powered by the emergency dc bus but is reset by dc power from the essential dc bus. Once reset, the emergency ac generator continues to power the emergency dc bus through the transformer-rectifier, and the emergency dc bus becomes self sustaining. Failure to reset indicates a malfunction.

e. AC bus switch — NOR (voltage should be 0).

f. Battery switch —TR FAIL.

The TR FAIL position reconnects the emergency dc bus to the dc essential bus.

g. AC generator switch—RESET, then ON. Check 105-125 volts.

Note

AC power control and reset are powered by the emergency dc bus. If the TR FAIL position of the battery switch does not reconnect the emergency dc bus to the dc essential bus, the main ac generator will not reset.

h. Battery switch — ON.

i. DC generator switch – RESET then ON.

7. Fuel boost pump switches—ON (if external power source was used for start).

Fuel boost pumps should be turned ON, one at a time allowing five seconds for each pump to start, when a successful check of all phases of ac voltage has been completed after starting with external power. If battery start was made, boost pumps should not be turned ON until after completion of INTERIOR CHECK AFTER BATTERY START.

8. Radar master switch—STBY.

INTERIOR CHECK AFTER BATTERY START

Except for the following items, the interior check should have been completed prior to starting engine. Starting at the left-hand console, perform the following checks:

1. Fuel quantity — Check.

Select each position on the fuel quantity gage selector switch and check fuel quantity gage for correct indications.

2. Fuel quantity gage test button — TEST.

Depress fuel quantity gage test button; check gage for decrease.

3. Fuel boost pump pressure-low warning lights — ON.

4. Fuel boost pump — Fuel shutoff valve electrical interlock — Satisfactory operation* (do not accomplish this step if the fuel selector switch is safety-wired**).

a. Fuel selector switch — R (some airplanes).

Rotate the fuel selector switch to R in order to close the left-hand fuel shutoff valve.

Left-hand fuel shutoff valve switch — CLOSE (other airplanes).

b. Left-hand fuel boost pump switches — ON; left-hand boost pressure-low warning light remains on.

CAUTION

If the boost pressure-low warning light does not remain on, it is an indication of a malfunction in the electrical interlock, since the pumps should not operate unless the shutoff valve is fully open. If a malfunction exists, the cause should be established and corrected before proceeding further.

c. Fuel selector switch — L (some airplanes).

Rotate the fuel selector switch through ENGINE to L position in order to close the right-hand fuel shutoff valve.

Left-hand fuel shutoff valve switch — OPEN (other airplanes).

Right-hand fuel shutoff valve switch — CLOSE (other airplanes).

d. Repeat step "b" above for the right-hand boost pumps and pressure-low warning light.

5. Fuel boost pump switches — OFF.

6. Fuel selector switch — ENGINE (some airplanes). Fuel shutoff valve switches — OPEN (other airplanes).

**AF 56-1206 & on, & airplanes modified by TCTO 1F-102-651.*
***In accordance with TCTO 1F-102-686.*

7. Fuel boost pumps — Check.
   a. Observe that both boost pressure-low warning lights are illuminated.
   b. Left-hand forward boost pump switch ON and check that left-hand boost pressure-low warning light extinguishes within five seconds.

CAUTION

The warning light will normally extinguish within two seconds. However, if the light is still illuminated after five seconds, the boost pump switch should be turned OFF and the cause established and corrected before attempting to operate other pumps.

   c. Left-hand forward boost pump switch — OFF.
   d. Repeat steps "a" through "c" for the left-hand aft, right-hand forward and right-hand aft boost pumps.
8. Boost pump switches — ON.
   The boost pump switches should be turned on one at a time allowing approximately five seconds for each pump to start.
9. UHF radio — ON.
10. Landing & taxi lights — Climatic.
11. Thunderstorm lights — Climatic.
12. Navigation receiver — ON and check.
13. J-4 compass — Check.
    a. Latitude set.
    b. Function selector switch — DG.
    c. Precess compass card approximately 45°.

CAUTION

Do not operate "set" switch continuously for more than 30 seconds to avoid overheating the slew motor.

    d. Function selector switch — MAG, heading checked.
14. IFF — STBY; check mode code.
15. Pitot heat switch — Check, then OFF.
16. Position lights — Climatic.
    Test position light switches for FLASH, STEADY, DIM, and BRIGHT (some airplanes) or FORMATION and NAVIGATION (other airplanes).

Note

The anticollision lights will illuminate when the position lights switch is placed in NAVIGATION position. Use of the anticollision lights on the ground shall be kept to an absolute minimum. The excessive heat created on the ground is detrimental to bulb life and increases maintenance problems, and during ground emergencies the operating light could confuse rescue operations since emergency ground vehicles use a similar light.

17. Cockpit lights — Climatic.
    Check operation of cockpit lights and set rheostats as desired (OFF or ON).

HYDRAULIC AND FLIGHT CONTROL SYSTEMS CHECK

CAUTION

Rapid and abrupt movement of the control stick should be avoided because of the possibility of damage to system components.

1. Throttle — IDLE.
2. Speed brakes switch — IN, then center (neutral) position.
   Retract speed brakes and check with crew chief for proper speed brake operation. Secondary hydraulic system pressure may drop momentarily but must return to normal within two seconds.
3. Pitch and yaw dampers — Engage.
4. Flight control surface movement — Check.
   Check rudder surface movement. Then, with the stick in the fully aft position, check elevon surfaces for proper movement (aileron as well as elevator position).

Note

During the above check there may be jerky rudder pedal motions caused by the turn coordinator. This is a normal condition.

5. Hydraulic system recovery (primary and secondary) — Check.
   With the hydraulic pressure selector switch in the PRI position, check primary hydraulic system pressure for 2950 (±100) psi. Check operation of the primary hydraulic system by moving the control stick from any corner to the diagonally opposite corner in approximately three seconds. If system pressure drops, the hydraulic pressure must return to system pressure within two seconds, or less, after control stick movement has stopped. Then place the hydraulic pressure selector switch in the SEC position and repeat the above procedures for a check on the secondary hydraulic system.
6. Momentary interrupt trigger — Depress.
   Note that AFCS disengages.

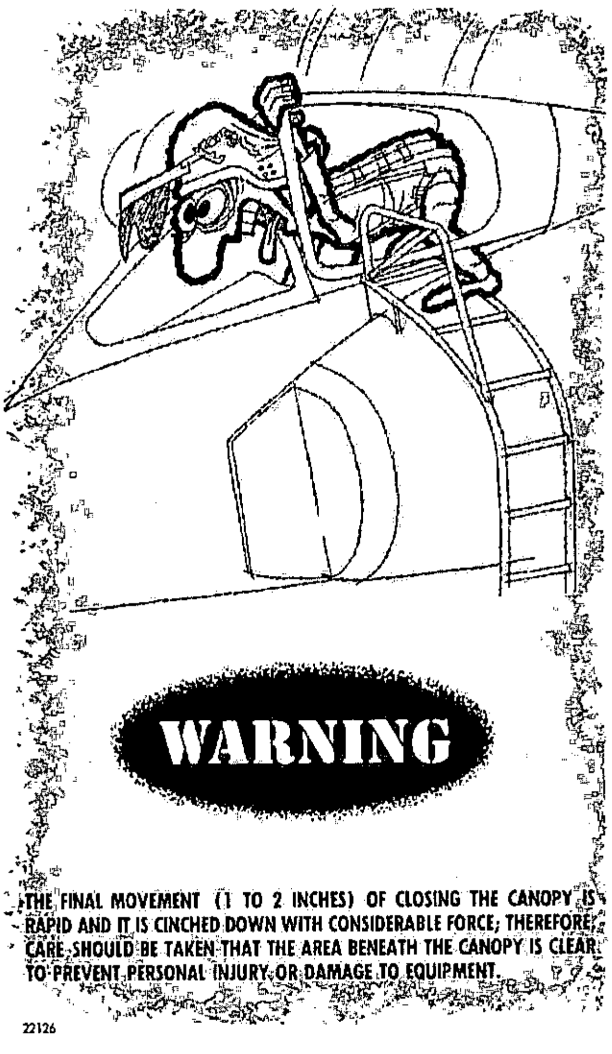
7. Emergency damper disconnect button — Depress.
   Check that pitch and yaw dampers disengage.
8. Trim — Check and set for takeoff trim.
   Trim nose-down, right wing down, and right rudder; then depress takeoff trim button. Control surfaces should return to neutral. Takeoff trim light should illuminate.

GENERAL

In addition to the above checks, observe the following instructions before taxiing:

CAUTION

If the canopy is open during taxi operations, observe canopy limit speeds. The canopy hold-open rod must be used on airplanes which do not have a canopy hold button. Keep hands away from canopy sill unless the canopy support tool is in place.



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1. Ejection seat safety pin — Remove.
   Remove ground safety lock pin from right arm rest and stow. If pin is not visible, ascertain that the pin streamer has not dropped down beside the seat or become disconnected from the pin.
2. Canopy support tool — Remove.
3. Canopy — Close and lock.
4. Canopy warning light — Out.
   Check canopy warning light out indicating latches are fully engaged.
5. IFF master switch — As desired.
6. Radar master switch — As desired.
7. Anti-ice switch — AUTOMATIC.
8. Radio call — Accomplished.
   Obtain taxi-takeoff instructions and set altimeter.
9. Engine pressure ratio gage — Set.
   Check outside air temperature and set engine pressure ratio gage to indicate takeoff limits.

TAKEOFF CHECK TABLE — J57-P-23 ENGINE

TEMP		PRESSURE RATIO SETTING
°F	°C	
32	0	2.30
37	2.5	2.28
41	5	2.26
45	7.5	2.24
50	10	2.22
54	12.5	2.20
59	15	2.18
63	17.5	2.17
68	20	2.15
72	22.5	2.13
77	25	2.11
81	27.5	2.09
86	30	2.07
90	32.5	2.06
95	35	2.04
99	37.5	2.02
104	40	2.00
108	42.5	1.98
113	45	1.96

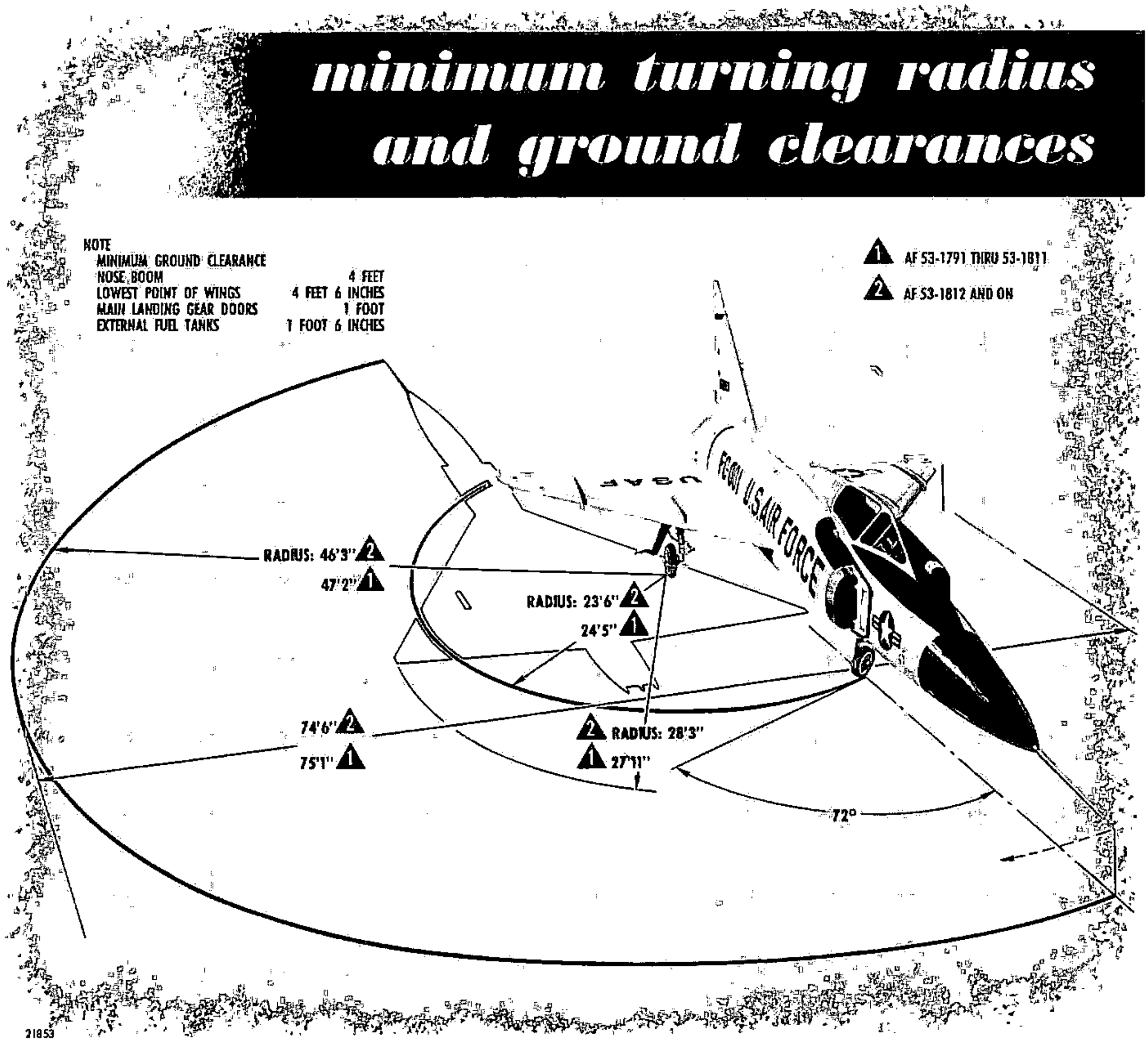

Figure 2-4

10. Armament reset switch — Actuated by crew chief (airplanes with missiles aboard).

    Have crew chief actuate the armament reset switch and obtain illumination of the armament reset light to insure that the armament system is ready for operation and that snubbing pressure is being applied to the missile bay area.

11. Chocks — Removed.

    Signal crew chief to remove wheel chocks.

STARTING ENGINE (SCRAMBLE)

1. Starting air — Connected.
2. Danger areas — Clear.
3. External power — Connected or Battery switch — ON.
4. Throttle — START.
5. Ignition button — Depress.
6. RPM — Positive indication.
7. Throttle — OFF, then IDLE.
8. ▲ Fuel flow — Check indication.
9. Exhaust gas temperature gage — Shows increase.
10. Oil pressure-low warning light — Off (approximately 30% rpm; at least by idle rpm).
11. Ignition button — Release at 33% rpm.
12. All personal equipment — Attach.
13. Canopy support tool — Remove.

14. Exhaust temperature — Stabilized.
15. Idle rpm — 55 to 65% rpm.
16. Engine fuel pump failure warning light — Off.
▲17. LH aft fuel boost pump — OFF.
18. UHF radio — OFF.
19. Compressed air and external power — Disconnected.
20. Throttle — 80%.

BEFORE TAXIING (SCRAMBLE)

ELECTRICAL POWER SUPPLY SYSTEM

DC System

1. DC generator output — Check.
2. Battery switch — ON.
3. DC generator — OFF.
4. Battery output — Check.
5. DC generator — ON.

AC System

1. AC voltage (all phases) — 105-125 volts.
2. AC generator switch — OFF.
3. AC bus switch — EMER (some airplanes); RESET, then ON (other airplanes); 103-140 volts.
4. AC bus switch — NOR.
5. AC generator switch — Reset then ON.
6. Transformer-rectifier — Check (some airplanes*).
   a. AC generator switch — OFF.
   b. DC generator switch — OFF.
   c. AC bus switch — EMER. (check for 0 volts).
   d. AC bus switch — RESET then EMER.
   e. AC bus switch — NOR (voltage should be 0).
   f. Battery switch — TR FAIL.
   g. AC generator switch — RESET then ON. Check 105-125 volts.
   h. Battery switch — ON.
   i. DC generator switch — RESET then ON.
7. Fuel boost pump switches — ON (if external power source was used for start).
8. Radar master switch — WARM.

INTERIOR CHECK AFTER BATTERY START (SCRAMBLE)

1. Fuel boost pumps — Check, then ON.
2. UHF radio — ON.

HYDRAULIC POWER SUPPLY SYSTEM

1. Throttle — IDLE.
2. Flight controls — Check.
3. Hydraulic system recovery — Check.
4. Trim — Check and set for takeoff.

GENERAL (SCRAMBLE)

1. Ejection seat safety pin — Remove.
2. Canopy — Close and lock.
3. Canopy warning light — Out.
4. Radar master switch — As desired.
5. Radio call — Accomplished.
6. Engine pressure ratio — Set.
7. Armament reset switch — Actuated by crew chief (airplanes with missiles aboard).
8. Chocks — Removed.

TAXIING

WARNING

If airplane is to be operated on the ground under possible conditions of carbon monoxide contamination, such as taxiing directly behind another operating jet airplane, or during operation with tail into wind, use oxygen with regulator diluter lever at 100% OXYGEN (some airplanes) or ON (other airplanes).

The taxiing and ground handling qualities of the airplane are normal and require no special techniques. There may be a slight delay in the braking between the pedal release and braking action release. Idle thrust is sufficient for all taxi operations under most conditions on concrete except initial movement from standstill; fuel consumption at idle thrust setting is approximately 840 pounds per hour. Nose wheel steering is not difficult but slightly higher rudder forces are noticeable which, while tending to keep the airplane aligned down the runway, cause the airplane to straighten more quickly than first expected during recovery from a turn. Initial reaction to this effect results in a "snaking" motion of short duration.

Note

Beyond 50 degrees left or right from center, or when airplane weight is off the gear, nose wheel steering is inoperative and the steering button will be keying the microphone.

Visibility during taxiing is very good except that vision to the rear is blocked through an arc of approximately 60 degrees; the wing tip and approximately one foot of the elevon can be seen. The following checks should be accomplished during taxi:

1. Nose wheel steering and brakes — Check.
   Engage nose wheel steering, commence taxiing and maintain directional control by operating rudder pedals as required. While rolling straight, release nose wheel steering and check

*Airplanes modified by TCTO 1F-102-727.

the brakes individually, noting that nose wheel steering disengages. See figure 2-4 for Minimum Turning Radius.

CAUTION

Excessive pressure on the rudder pedals with nose wheel steering engaged and the airplane not in motion may cause damage to steering mechanism.

2. Flight instruments — Check & set.
3. Navigation equipment — Check.
   Check operation of navigation equipment and directional indicator (slaved) for indicator changes during taxiing.

BEFORE TAKEOFF

PREFLIGHT AIRPLANE CHECK

1. Warning lights — Out.
2. Takeoff trim — Set.
3. Flight controls — Checked & free.
   Check rudder surface movement. Then, with the stick in the fully aft position, check elevon surfaces for proper movement (aileron as well as elevator position).
4. Safety belt — Fitted snugly.
5. Zero delay lanyard hook–Attached (if installed).
6. Shoulder harness inertia reel handle — UNLOCKED.
7. Cockpit no-fog vent suit switch – ON (if installed).
   Cabin temperature control knob – AUTOMATIC, midway or hotter (airplanes without cockpit no-fog vent suit switch).

WARNING

Excessive moisture condensation may occur through the cabin pressurization system. This condensation may become so dense when operating in conditions of high dew point temperature as to make it impossible to read the instrument panel presentation. On airplanes that do not have a cockpit no-fog vent suit switch, the temperature control knob should be placed to the hottest level possible that permits comfort and still prevents fog formation.

8. Pitot heat – As required.

Taxi into takeoff position, center nose wheel, and hold wheel brakes.

CAUTION

In order to insure that nose wheel steering is engaged for takeoff, the steering should be used to line up and should not be disengaged until the rudder becomes effective on the takeoff roll.

PREFLIGHT ENGINE CHECK

Fuel Control Emergency System Check

1. Throttle — IDLE.
2. Fuel control switch — EMER.
3. Emergency fuel control warning light — On.
4. Fuel flow 750 pph minimum; 1050 pph maximum.
   Check fuel through the emergency system with the engine at idle rpm.
5. Fuel control switch — NORMAL (guard closed).
6. Emergency fuel control warning light — Out.

Thrust Check

When ready to roll accomplish the following checks:

1. Throttle — TAKEOFF (some airplanes) or FULL MIL POWER (other airplanes).
   If a takeoff lock trigger is installed, advance throttle to TAKEOFF. On airplanes not equipped with a takeoff lock trigger, advance to FULL MIL POWER.
2. Engine instruments — Check.
   Check tachometer, exhaust gas temperature gage, fuel flow indicator, and engine pressure ratio gage for normal operating limits.

Note

Engine rpm will vary for individual engines for military thrust. The rpm for takeoff should be in relation to the rated rpm as specified on the engine data plate. Engine rpm will be less when obtaining military thrust with temperatures below standard.

TAKEOFF

WARNING

If excessive moisture condensation occurs during takeoff, place the cabin air switch to the RAM position. Return to the PRESS position after becoming airborne and above approximately 3000 feet.

NORMAL TAKEOFF

A typical takeoff is illustrated in figure 2-5. Refer to Appendix for takeoff charts showing distances required at varying gross weights, temperatures, and field elevations. Use the following procedures for normal takeoff:

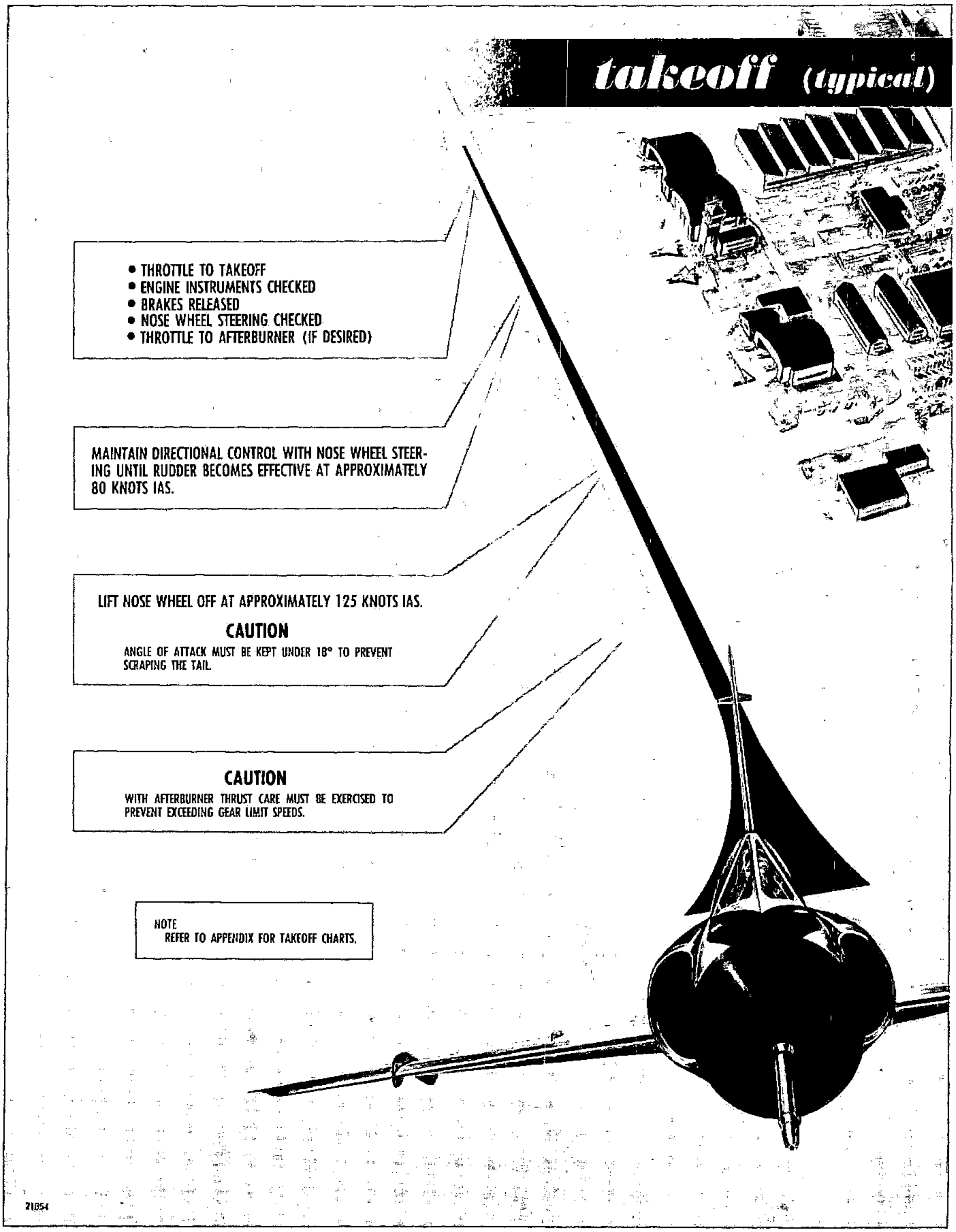
takeoff (typical)
• THROTTLE TO TAKEOFF
• ENGINE INSTRUMENTS CHECKED
• BRAKES RELEASED
• NOSE WHEEL STEERING CHECKED
• THROTTLE TO AFTERBURNER (IF DESIRED)
MAINTAIN DIRECTIONAL CONTROL WITH NOSE WHEEL STEERING UNTIL RUDDER BECOMES EFFECTIVE AT APPROXIMATELY 80 KNOTS IAS.
LIFT NOSE WHEEL OFF AT APPROXIMATELY 125 KNOTS IAS.
CAUTION
ANGLE OF ATTACK MUST BE KEPT UNDER 18° TO PREVENT SCRAPING THE TAIL.
CAUTION
WITH AFTERBURNER THRUST CARE MUST BE EXERCISED TO PREVENT EXCEEDING GEAR LIMIT SPEEDS.
NOTE
REFER TO APPENDIX FOR TAKEOFF CHARTS.
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Figure 2-5

1. Throttle — TAKEOFF.
2. Engine instruments — Check.
   Check tachometer, exhaust gas temperature gage, fuel flow indicator, and engine pressure ratio gage for normal operating limits.
3. Brakes — Release.
   Release brakes and establish a straight takeoff roll.
4. Nose wheel steering — Check.
   Move rudder pedals until it is ascertained that nose wheel steering is engaged.

CAUTION

Because of the initial rapid acceleration, it is important that nose wheel steering be engaged and the nose wheel centered prior to starting takeoff roll.

5. Throttle — AFTERBURNER (if desired).

WARNING

Takeoff should be aborted immediately if any directional change is noted when the afterburner is ignited. A directional change at this time could indicate possible afterburner nozzle malfunction which could cause side forces to be applied to the extent that rudder would be insufficient to control the airplane immediately after leaving the ground.

Maintain directional control with nose wheel steering until the rudder becomes effective at approximately 80 KIAS. Beginning at 125 KIAS, raise the nose to the horizon with maximum thrust, or slightly below the horizon with military thrust, and allow the airplane to fly off the ground. Lower angles of attack will result in higher speed, longer run takeoffs. Higher angles of attack will result in lower airspeeds, and lower rates of climb immediately following takeoff. Do not prematurely raise the nose during takeoff as increased angle of attack at low speeds will result in excessive ground roll during takeoff.

CAUTION

- Angle of attack must be kept under 18° to prevent scraping the tail.
- With afterburner thrust, care must be exercised to prevent exceeding gear limit speeds before landing gear is fully retracted.

If EGT is increasing, without engine acceleration, so as to exceed allowable limits, reduce thrust to maintain temperature within limits. Monitor tachometer. Any engine speed in excess of maximum rpm limits should be noted on Form 781.

TO PREVENT AIRLOADS FROM INFLICTING STRUCTURAL DAMAGE, THE LANDING GEAR SHOULD BE UP AND LOCKED BEFORE LIMIT SPEED IS REACHED.

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Note

In maximum thrust takeoffs, the altimeter may lag or even indicate a decrease in altitude just before breaking ground and during the initial climbout. This altimeter error is the result of disturbed pressure ahead of the airplane due to acceleration and high angle of attack. The altimeter will indicate correctly after the airplane reaches approximately 300 feet of altitude.

MILITARY THRUST TAKEOFF

If a takeoff is to be made without use of the afterburner, normal takeoff procedures should be utilized up to the point of breaking ground. Due to the possibility of increasing drag excessively by a high angle of attack immediately after takeoff, the nose should be raised to a point just below the horizon and the airplane allowed to fly off. After breaking ground, the airplane should be allowed to accelerate until certain the airplane will remain airborne before retracting the landing gear.

MINIMUM RUN TAKEOFF

A minimum run takeoff can be accomplished by using the normal takeoff procedures and engaging afterburner as soon as possible after brake release.

CAUTION

Angle of attack must be kept under 18° to prevent scraping the tail.

CROSS-WIND TAKEOFF

Cross-wind takeoffs present no particular problems in this airplane. In addition to the Takeoff Distances and Speeds Charts, check the Takeoff and Landing Cross-wind Chart in the Appendix to determine the cross-wind component and best nose wheel lift-off speed. After lift-off, establish the crab angle necessary to maintain the desired flight path.

AFTER TAKEOFF—CLIMB

Note

There are no provisions for automatic transfer from normal to emergency fuel flow within the fuel control unit.

When airplane is definitely airborne:

1. Landing gear handle—Up.

Note

Exercise care not to knock the throttle aft and out of AFTERBURNER position when retracting the landing gear.

2. Gear checked—Up.

   Check that landing gear position indicators indicate gear up and landing gear warning light off.
3. EGT—Monitor.

WARNING

If EGT exceeds the limits shown in figure 5-2, reduce thrust and increase airspeed. Abort the mission and land as soon as practicable. Make an entry on Form 781 indicating time and temperature peaks of overtemperature operation.

4. Takeoff locks—Release before 7000 feet (if installed).

   To release the takeoff locks in the fuel control unit, retard the throttle aft of TAKEOFF position before reaching an altitude of 7000 feet, flight conditions permitting.

Note

On later airplanes, without the fuel control takeoff locks incorporated, it is not necessary to retard the throttle out of TAKEOFF when climbing above 7000 feet.

5. Oxygen—Check.

   If 100% OXYGEN was used for takeoff and partial pressure suit is not worn, return oxygen regulator diluter lever to NORMAL OXYGEN, unless carbon monoxide contamination is suspected. If such is the case, continue use of 100% oxygen as long as considered necessary.

WARNING

On airplanes not equipped with a survival kit, oxygen regulator diluter lever must be returned to NORMAL OXYGEN as soon as possible to prevent premature depletion of the oxygen supply unless it is determined that the supply is adequate for the duration of the flight.

6. Zero delay lanyard hook—Detach and stow (if installed).

   If "one & zero" escape system is installed, detach zero delay lanyard hook from ripcord after reaching 2000 feet above terrain in accordance with limitations established in the Zero Delay Lanyard Engagement Requirements Chart (figure 3-3) and stow on parachute harness. For minimum ejection altitudes, see the Emergency Minimum Ejection Altitude Table, figure 3-4.
7. Damper systems—Engage.

WARNING

Initial engagement of the pitch damper should be avoided at less than 5000 feet above the terrain. This altitude will provide adequate ground clearance in the event of any phasing errors that could occur during the initial engagement.

8. External tank fuel transfer switch—ON, when internal fuel quantity indicates 5500 pounds or less.
9. Cockpit no-fog vent suit switch – OFF (if installed).

   Place cockpit no-fog vent suit switch to OFF to return temperature control to the automatic temperature controller.

10. IFF — Checked.

    If positive operation of the normal mode of IFF has not been established during departure with an air traffic facility, a check should be made with such a facility as soon after takeoff as flight conditions will permit. This check must be made prior to entering a radar advisory area. If IFF is inoperative consult the appropriate navigation publications.

11. Altimeter — Reset to 29.92 (above 23,500 feet).

CLIMB

The recommended climb speeds as given in the Appendix should be followed.

CRUISE

Note

- During operation in high humidity conditions (ground dew point 20°C or higher) operate canopy defog continuously at altitude in order to insure fog-free canopy during descent. For less severe humidity conditions, canopy defog need not be turned on until just before descent.
- If EGT is increasing, without engine acceleration, so as to exceed allowable limits, reduce thrust to remain within limits.

Refer to Appendix for recommended cruise procedures.

FLIGHT CHARACTERISTICS

Refer to Section VI for flight characteristics of the airplane.

DESCENT

Note

Refer to Section V for limitations applicable to descent. Refer to the Appendix for the recommended descent speeds, time required fuel consumption, etc.

1. Canopy defog switch—CANOPY DEFOG.
2. Cockpit no-fog vent suit switch—ON (if installed). Cabin temperature control knob—AUTOMATIC, midway or hotter (airplanes without cockpit no-fog vent suit switch).

WARNING

On airplanes that do not have a cockpit no-fog vent suit switch, the cabin temperature control knob should be placed in the hot range prior to entering the traffic pattern to prevent cockpit fog during landing or go-around.

3. IFF — Checked.

   Check the IFF within one hour prior to estimated time of landing.

4. Altimeter — Reset to point of descent setting (passing through 24,000 feet).

BEFORE LANDING

1. Zero delay lanyard hook — Attach (if installed).

   If "one & zero" escape system is installed, attach zero delay lanyard hook to ripcord prior to reaching 2000 feet above terrain in accordance with limitations established in the Zero Delay Lanyard Engagement Requirements Chart (figure 3-3) and stow on parachute harness. For minimum ejection altitudes, see the Emergency Minimum Ejection Altitude Table, figure 3-4.

Note

After landing it is not necessary to disconnect the zero delay lanyard, since it connects the ripcord to the timer knob and is not attached to the safety belt. The parachute may be removed from the airplane with the lanyard in the hooked-up condition.

2. Fuel quantity — Check.
3. Boost pump switches — ON.
4. Fuel selector switch — Check.
5. Hydraulic pressures — Check.
6. Radar master switch — STBY.
7. Armament safety switch — SAFE (guard closed).
8. Armament selector switch — SNAKE.
9. Shoulder harness inertia reel handle — UNLOCKED.
10. Entering traffic pattern.
    a. Airspeed — 300-325 KIAS.
    b. Speed brakes — As desired.

Note

In the event of a speed brakes system malfunction, it is possible for one brake to extend while the other remains retracted, resulting in violent yawing. Consequently, it is desirable to open the speed brakes sufficiently early in the pattern to permit immediate retraction and recovery should this condition occur.

    c. RPM 80 to 85%.

       Reduce thrust to decelerate to safe landing gear extension speed.

Note

At no point in the pattern should rpm be less than 75% to assure sufficient engine response in the event of a go-around.

d. Altitude — 1500 feet above field elevation.

11. Downwind.

a. Altitude — 1500 feet above field elevation.

b. Airspeed — 220 KIAS.

12. Before turning base.

a. Landing gear handle — Down; gear checked down; warning light out.

Note

- Refer to Section V for landing gear maximum extension speed and tire ground speed limits.
- Above 200 KIAS with landing gear extended a high frequency buffet occurs. This is normal and is due to airflow in wheel well areas.

b. Altitude — 1500 feet above field elevation.

c. Airspeed — 220 KIAS.

d. RPM — 85-90%.

13. Base (descending).

RECOMMENDED SPEEDS

GROSS WEIGHT	(KIAS) BASE	APPROACH	PRIOR TO FLARE	TOUCH-DOWN
20,000	173	164	159	130
22,000	182	172	167	137
24,000	191	180	175	143
26,000	199	188	182	149
28,000	206	195	189	155
30,000	214	202	196	160

a. Airspeed — 185 KIAS minimum.
Establish descent during turn on base.

14. Final approach.

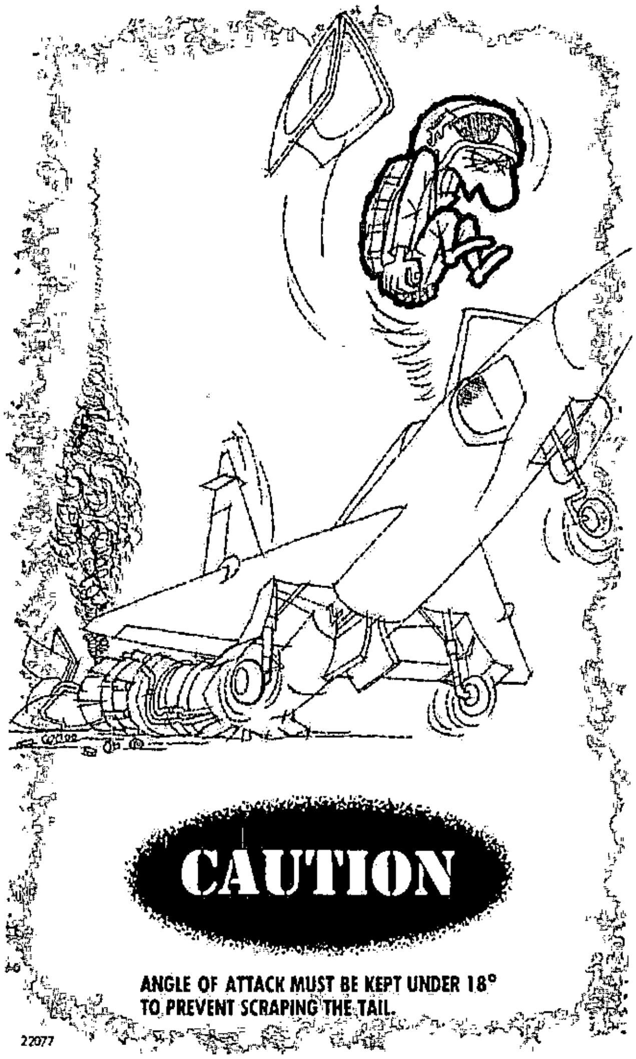
a. Airspeed — 175 KIAS minimum.

CAUTION

If airspeed becomes excessively low, a high sink rate may develop, resulting in a hard landing. See figure 6-1, Minimum Speeds, for information concerning rate of sink vs. airspeed for other than standard conditions. Normally, 85% rpm will be required on final approach to maintain sufficient airspeed.

15. Prior to flare.

a. Airspeed — 170 KIAS minimum.



LANDING

NORMAL LANDING

A typical landing pattern and recommended procedures are shown in figure 2-6. Refer to Appendix for recommended approach and touchdown speeds at varying gross weights. The landing should be accomplished in a wings-level attitude with the airplane aligned with the runway and an angle of attack approximately 15° (nose on the horizon). Typical landing procedures are as follows:

1. Flare out.

a. Throttle — IDLE.
Reduce thrust to idle during flareout.

WARNING

If engine fails to decelerate to idle rpm (approximately 55 to 65% rpm) when throttle is reduced to IDLE, place the fuel control switch to EMERGENCY to reduce landing roll.

2. Touchdown.
   a. Airspeed — 140 KIAS minimum.
   b. Drag chute — Deploy.

      Drag chute may be deployed immediately after touchdown. There is a slight nose-down pitch when the drag chute is deployed. The drag chute will deploy slowly in some cases if the nose is held off. For faster deployment, the nose may be lowered.
   c. Lower nose wheel to runway.

The nose should be lowered to the runway at approximately 105 KIAS on early airplanes*. On later airplanes** sufficient elevator control will remain at 90 KIAS, and the nose should be lowered at this speed. Wheel braking should be applied as necessary.

Note

- Full length of the runway should be used during landing to reduce brake wear. If conditions permit, delay the use of wheel brakes until the airplane has slowed to 90 KIAS or less to ensure against skidding the tires.
- There may be a slight delay between pedal release and release of braking action.

Directional Control During Landing Roll

Good directional control during the landing roll can be maintained by use of the ailerons. After the main landing gear is firmly on the runway, the airplane will turn in the direction of aileron selection. Rudder, wheel brakes, and nose wheel steering should normally be used for directional control during the landing roll. Nose wheel steering is sensitive at high speed and should not be used until the airplane has been slowed to 80 KIAS or below.

CROSS-WIND LANDING

Before Touchdown

The traffic pattern for a cross-wind landing should be normal, making proper allowances for strength and direction of the cross-wind. Proper runway alignment on the final approach can be maintained by crabbing or dropping one wing; however, a combination of the two is recommended just prior to flare. For wet or dry runway landing, maintain cross-wind correction to touchdown to prevent side drift. Reduce sink rate to a minimum to accomplish smooth touchdown. At increased cross-wind components, sink rate must be minimized due to increase of side loads imposed on the landing gear.

After Touchdown

Accomplish touchdown on upwind side of runway. The drag chute should be deployed at touchdown. Drag chute deployment at touchdown tends to counteract the downwind weather vaning tendency of the airplane.

**AF 53-1791 thru 53-1811.*
***AF 53-1812 & on.*

CAUTION

Be prepared to jettison chute if excessive turning into the wind occurs as the airplane slows down.

Prior to nose wheel touchdown, directional control should be accomplished by use of rudder and aileron.

Note

After nose wheel touchdown, directional control can be maintained by using rudder, aileron, brakes, and nose wheel steering. Avoid use of nose wheel steering above 80 KIAS due to sensitivity. The use of aileron to "steer" (left aileron to turn left) is effective. Rudder or coordinated control is preferable in high cross-winds, particularly when the nose wheel is on the ground.

Refer to the Takeoff and Landing Cross-Wind Chart in the Appendix to determine cross-wind components and recommended nose wheel touchdown speed for a particular cross-wind condition.

LANDING ON SLIPPERY RUNWAYS

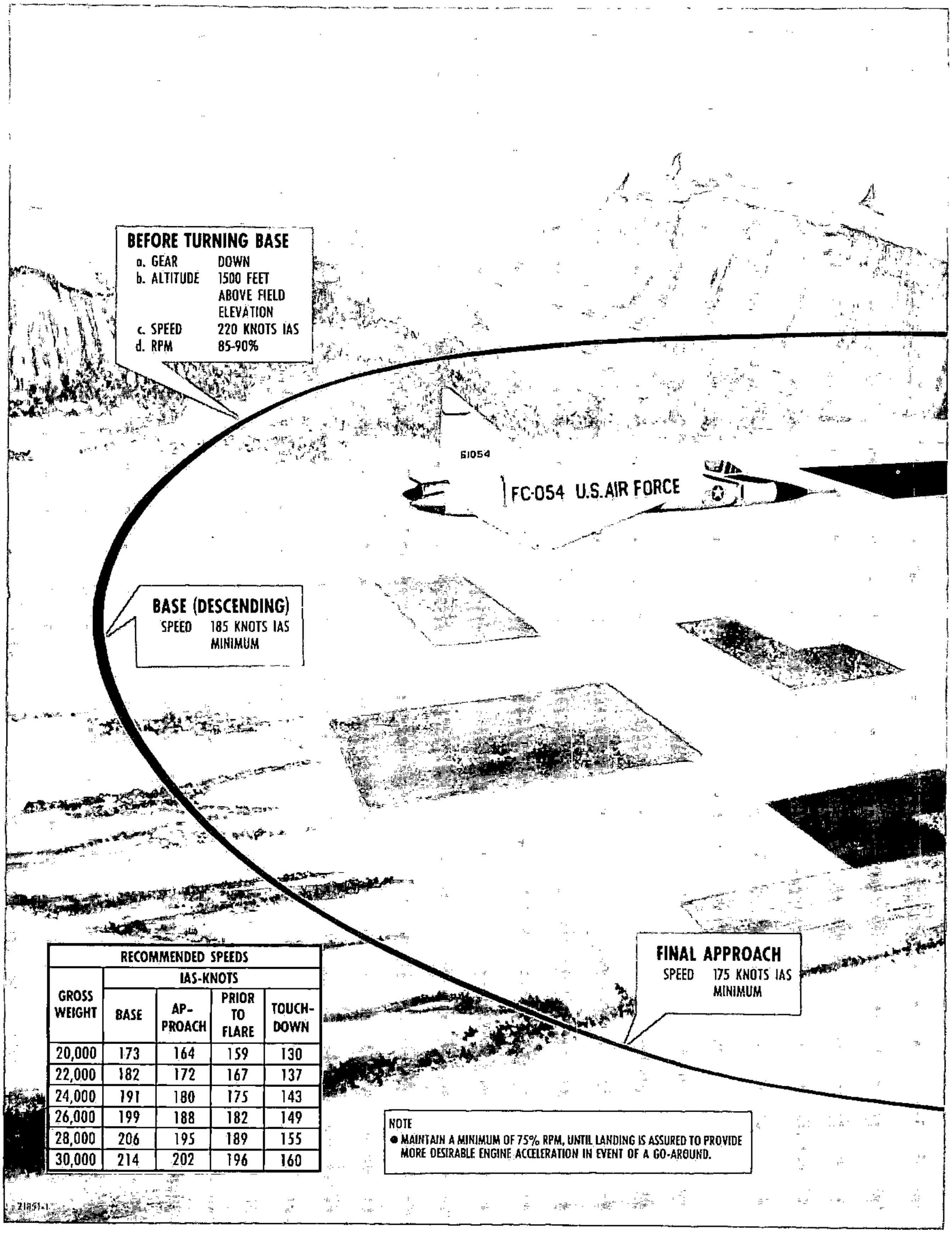
A normal landing pattern should be flown when landing on a slippery runway, planning the touchdown to allow maximum distance for the landing roll. The drag chute should be deployed immediately after touchdown.

Wet Runways

A normal landing pattern should be used for landing on wet runways. Excessive touchdown speed should be avoided. Establish the final approach speed for the airplane gross weight in accordance with the landing distance charts in the Appendix. Deploy the drag chute at touchdown. During the initial ground roll, maintain a nose-high attitude and use rudder and aileron for directional control. At approximately 90 KIAS, lower the nose wheel to the runway and commence intermittent braking. Avoid use of nose wheel steering above 80 KIAS, due to sensitivity.

CAUTION

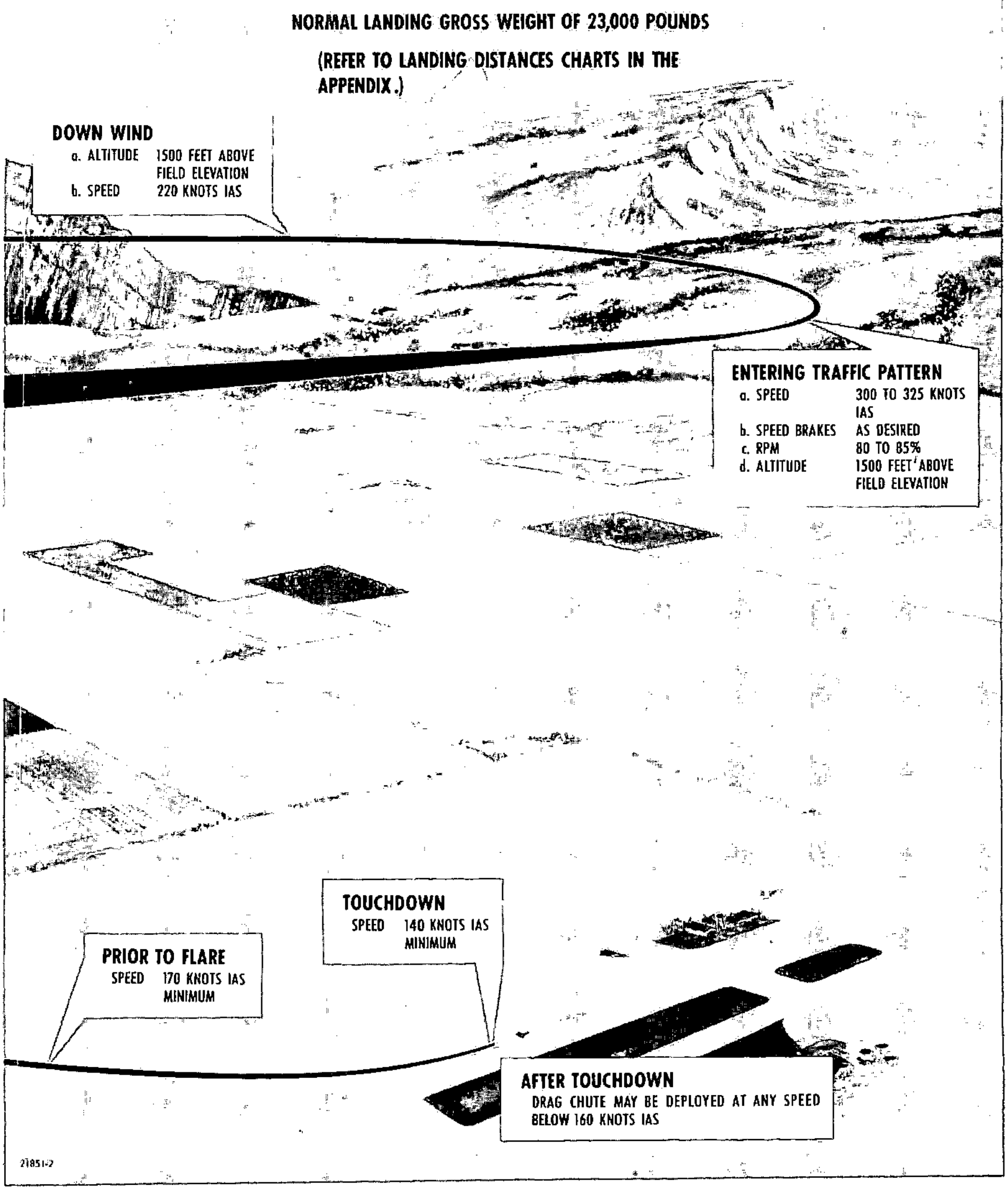
Wheel brakes are relatively ineffective until the lift of the wing has dissipated. During the initial ground roll on a wet runway, it is virtually impossible to determine when a wheel has stopped rotating. Therefore, the best assurance against blowing a tire is intermittent braking, with equal pressure applied to both brakes. If the airplane starts to yaw, release both brakes until directional control is regained, then resume intermittent braking. For additional information, refer to USE OF WHEEL BRAKES, Section VII.



RECOMMENDED SPEEDS				
	IAS-KNOTS			
GROSS WEIGHT	BASE	AP-PROACH	PRIOR TO FLARE	TOUCH-DOWN
20,000	173	164	159	130
22,000	182	172	167	137
24,000	191	180	175	143
26,000	199	188	182	149
28,000	206	195	189	155
30,000	214	202	196	160

Figure 2-6

normal landing pattern (typical)



If the drag chute fails to deploy, hold the nose up as long as possible to achieve maximum aerodynamic braking. Without the chute the landing roll will be greatly increased.

Icy Runways

The technique for landing this airplane on an icy runway is the same as for a wet runway. However, the reduced friction between the tires and the runway makes directional control more difficult and results in further reduction in brake effectiveness. Consequently it is of the utmost importance to touch down at the minimum speed for the particular airplane gross weight. Use of the drag chute is essential and ice grip tires are recommended.

HEAVY-WEIGHT LANDING

A heavy-weight landing is accomplished by using the same procedures as a normal landing; however, additional power may be required on final approach. Refer to the Appendix for prior to flare and touchdown speeds at varying gross weights.

MINIMUM-RUN LANDING

Use normal approach patterns and speeds. Touch down with the maximum angle-of-attack which will not damage the tail cone. Retard throttle to OFF at touch down, deploy drag chute, and immediately lower the nose and apply maximum braking.

Note

Although the drag chute is not designed for inflight use, it is considered practical to deploy the chute while in the landing attitude at the instant before touchdown.

CAUTION

Braking is permissible above 90 KIAS. However, extreme caution is required to prevent blowouts from skidding the tires, since it is very difficult to feel when a wheel is locked at high speeds.

After a minimum-run landing, or any time the brakes are used excessively, the airplane should be parked away from congested areas until the brakes and tires have cooled. Refer to USE OF WHEEL BRAKES, Section VII.

TOUCH-AND-GO LANDING

Touch-and-go landings are not recommended in this airplane. The usual hazards in takeoff and landing are compounded during touch-and-go landings due to the rapid actions required while rolling at high speeds and flying at low altitude. In addition, tire temperatures may become excessive after a series of touch-and-go landings.

GO-AROUND

A go-around may be made from any point in the approach. See figure 2-7 for typical go-around procedures. The descent attitude is approximately the same as the climb attitude, and the difference between climb and descent is thrust. Inasmuch as relatively high engine rpm is maintained during descent, thrust is almost instantaneously available when the throttle is advanced. No appreciable trim change is experienced when applying power or retracting landing gear. Some yaw may be experienced due to uneven speed brakes retraction at speeds below 180 knots.

Note

- If a go-around is attempted after unsuccessful or successful deployment of drag chute, push drag chute handle in, retract speed brakes, and advance throttle using afterburner as required.
- If drag chute deployment was unsuccessful, the speed brakes should not be opened until after touchdown on subsequent landing.

CAUTION

Be prepared to counteract yawing in the event a speed brakes malfunction causes unequal speed brakes retraction.

AFTER LANDING

Use extreme care when applying brakes immediately after touchdown, or at any time when there is considerable lift on the wings, to prevent skidding the tires and causing flat spots and possible blowout. If maximum braking is required after touchdown, lift should first be decreased as much as possible by lowering the nose before applying brakes. A heavy brake pressure can result in locking the wheels more easily if brakes are applied immediately after touchdown than if the same pressure is applied after the full weight of the airplane is on the wheels.

1. Pitch and yaw dampers — Disengage.
2. Speed brakes switch — Neutral.

   The speed brakes switch should be in the "neutral" position prior to drag chute jettison. This will prevent venting hydraulic fluid from the secondary hydraulic system if inadvertent emergency speed brakes extension had occurred during drag chute deployment.
3. Drag chute — Jettison.

   After taxiing clear of the runway the drag chute should be jettisoned.

Note

Drag chute should be jettisoned at slowest speed necessary to maintain parachute inflation.

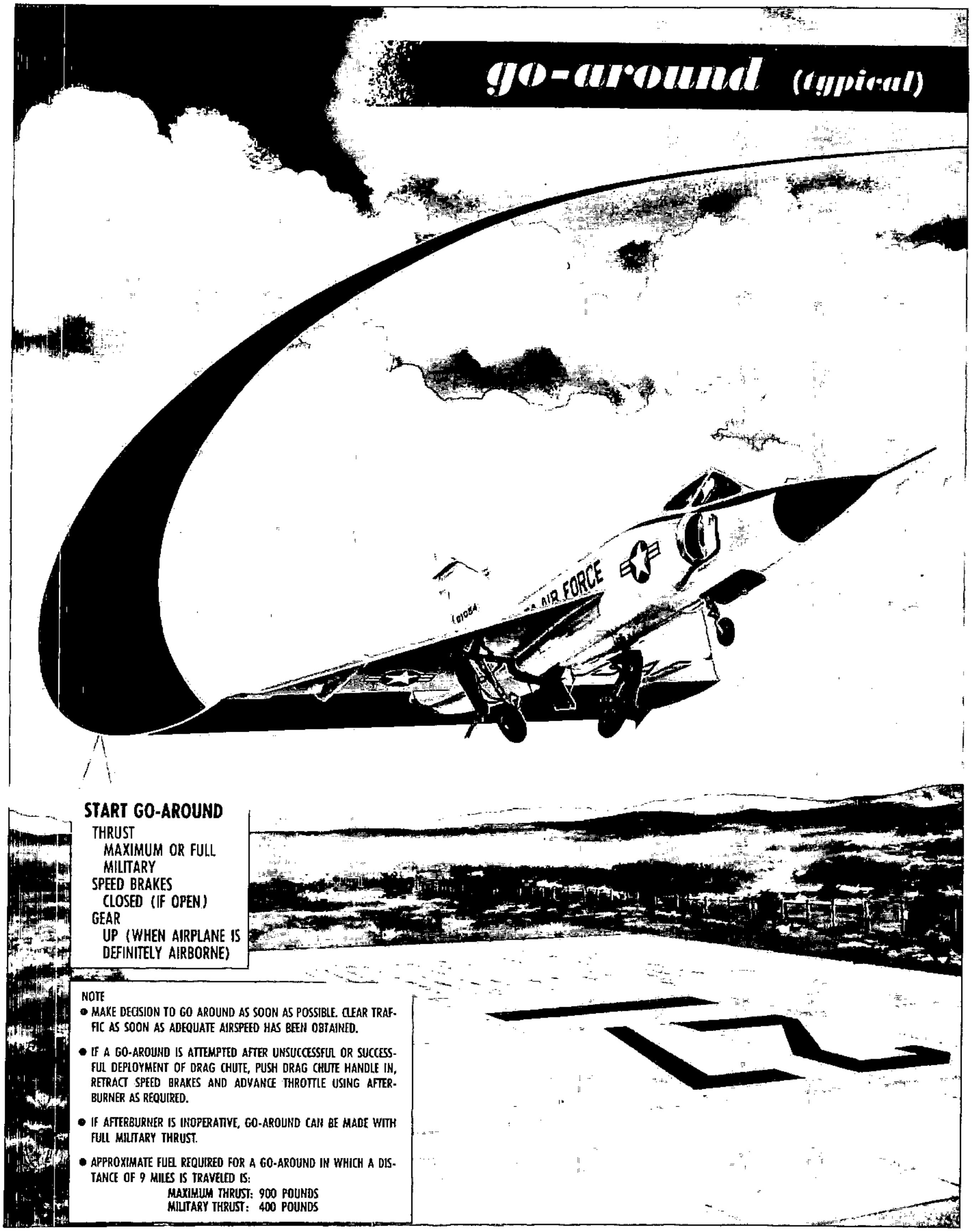

go-around (typical)
U.S. AIR FORCE
START GO-AROUND
THRUST
MAXIMUM OR FULL MILITARY
SPEED BRAKES
CLOSED (IF OPEN)
GEAR
UP (WHEN AIRPLANE IS DEFINITELY AIRBORNE)
NOTE
• MAKE DECISION TO GO AROUND AS SOON AS POSSIBLE. CLEAR TRAFFIC AS SOON AS ADEQUATE AIRSPEED HAS BEEN OBTAINED.
• IF A GO-AROUND IS ATTEMPTED AFTER UNSUCCESSFUL OR SUCCESSFUL DEPLOYMENT OF DRAG CHUTE, PUSH DRAG CHUTE HANDLE IN, RETRACT SPEED BRAKES AND ADVANCE THROTTLE USING AFTERBURNER AS REQUIRED.
• IF AFTERBURNER IS INOPERATIVE, GO-AROUND CAN BE MADE WITH FULL MILITARY THRUST.
• APPROXIMATE FUEL REQUIRED FOR A GO-AROUND IN WHICH A DISTANCE OF 9 MILES IS TRAVELED IS:
MAXIMUM THRUST: 900 POUNDS
MILITARY THRUST: 400 POUNDS

Figure 2-7

CAUTION

- The drag chute should be jettisoned before taxiing downwind in winds exceeding 15 knots because of the possibility of the chute collapsing and risers burning by contact with the hot areas of the exhaust nozzle or the chute being damaged from dragging on the ground. The drag chute should also be jettisoned in high wind or rain if the chute collapses because there is a possibility of the risers burning by contact with the hot areas of the exhaust nozzle or damage from dragging on the ground.
- Insure that helmet, oxygen mask, gloves or any other loose items are not placed in a location that could allow them to be dislodged from the cockpit and sucked into the engine intakes when the canopy is open and the engine running.
- If the canopy is open during taxi operations, observe canopy limit speeds. The canopy hold-open rod must be used on airplanes which do not have a canopy hold button. Keep hands away from canopy sill unless canopy support tool is in place.

4. RAT handle – Pull.

   The RAT should be extended while taxiing in to prevent possible injury to ground personnel.
5. Speed brakes switch — OUT.

   The speed brakes should be in the extended position prior to engine shutdown to facilitate turnaround.
6. Navigation receiver — OFF.
7. IFF — OFF.
8. Nesa switches — OFF.
9. Anti-ice switch — OFF.
10. Windshield rain clear switch — OFF.
11. Cockpit no-fog vent suit switch — OFF (if installed).
12. Canopy defog switch — OFF.
13. Pitot heat switch — OFF.
14. Anticollision light — OFF.
15. Takeoff trim — Set.

ENGINE SHUTDOWN

To shut down engine, proceed as follows:

1. Wheel chocks — Installed.

   Hold wheel brakes on; have wheel chocks installed.
2. RPM — Idle for five minutes.

   Operate engine at idle rpm for five minutes to stabilize engine temperature (taxi time at idle may be included).

Note

The five-minute stabilization period applies only when the engine has been operated above 85% rpm for periods exceeding one minute during the five-minute period prior to shutdown.

CAUTION

If engine temperatures are not allowed to stabilize for approximately five minutes before shutdown, damage can result from interference between engine rotating and stationary parts which have differences in cooling rates.

3. Compressed air — Connected.

   Signal crew chief to connect compressed air for shutdown. If no external compressed air source is available, the combustion starter manual air-valve should be placed in AIRCRAFT position to provide air for clearing.

Note

In the event of engine fire during shutdown, compressed air should be provided to motor the engine.

4. Engine ignition disconnect switch — OFF.

   Have crew chief place the engine ignition disnect switch in the left-hand main wheel well to OFF.
5. Hydraulic systems check – Two-second recovery.

   With the engine operating at idle, check operation of each hydraulic system by smoothly positioning stick fully forward and fully aft. The hydraulic system pressure gage should show a pressure drop and then return to system pressure within approximately two seconds.
6. Radar master switch — OFF.
7. Boost pump switches — OFF.
8. UHF radio — OFF.
9. DC and ac generator switches — OFF.
10. RPM — 70% or above for 30 seconds.

    Immediately prior to shutdown, operate the engine at 70% rpm for 30 seconds to assure complete scavenging of oil system.
11. Throttle — OFF.

Note

Check that engine decelerates freely by listening for any excessive engine noises during shutdown. Leave fuel selector switch to ENGINE (fuel shutoff valve switch to OPEN on some airplanes).

12. RPM zero, fuel selector switch — OFF (some airplanes) if not safety-wired*.

**In accordance with TCTO 1F-102-686.*

Fuel shutoff valve switches — CLOSE (other airplanes).

Allow engine to stop prior to shutting off fuel to provide lubrication for the fuel control unit.

13. Canopy support tool — In place.

WARNING

With low pneumatic air pressure, canopy will drop when dc power is removed if canopy support tool is not in place.

14. Battery switch — OFF.
15. Move elevons to remove hydraulic system pressure.

BEFORE LEAVING AIRPLANE

Before leaving the airplane check the following:

1. Ejection seat safety pin — Installed, streamer visible.

   Insert the seat safety pin from the inboard side** of the right arm rest with the streamer visible.
2. Oxygen supply switch — OFF (if equipped with survival kit) before removing mask or faceplate.
3. Personal equipment leads — Disconnect.

CAUTION

If wearing an automatic opening parachute that has a key attached to the aneroid arming lanyard, make sure key does not foul when leaving cockpit, to prevent chute from being opened inadvertently.

4. Survival kit — Detach from parachute (if installed).
5. Form 781 — Completed.

   Enter any discrepancies which were noted during flight.

CAUTION

Make appropriate entries on Form 781 covering any system defects or any limits in the Flight Manual that have been exceeded during the flight. Entries must also be made when the airplane has been exposed to unusual or excessive operations such as hard landings, excessive braking action during aborted takeoffs, long and fast landings, and long taxi runs at high speeds, etc.

6. Chocks, landing gear safety pins, and external tank safety switch pin (Part #SE 1100) — Install.

***Outboard side on AF 56-1275 & on & earlier airplanes modified by TCTO 1F-102-751.*

STRANGE FIELD TURN-AROUND PROCEDURES

The following procedures supplement the Normal Abbreviated Check List, providing servicing instructions in check list form for use where ground crew personnel are not familiar with the F-102A airplane. This check list should be used by the pilot only after a thorough briefing by maintenance personnel on all aspects of servicing this airplane. Servicing fluid specifications are contained in the Servicing Diagram, Section I.

Note

These procedures may be removed from the Flight Manual for convenient use as they will not appear in the cardboard check list, T.O. 1F-102A-(CL)1-1.

When required to direct or accomplish servicing of the airplane at a strange field, proceed as follows:

ENGINE OIL TANK SERVICE

Note

The engine oil tank should be filled as soon as possible after engine shutdown (not to exceed five minutes) or oil will drain from the tank into the accesory gear case in sufficient quantities to prevent an accurate oil quantity check.

The oil filler cap is located under the access panel on the top left side of the fuselage just forward of the vertical stabilizer.

1. Check oil level with clean dipstick.
2. Service with oil.
3. Replace and check filler cap for proper installation.

ARMAMENT DOOR OPENING

CAUTION

All armament doors must be fully closed before operation.

1. Check cockpit for the following:
   a. Battery switch — OFF.
   b. Armament selector switch — SNAKE.
   c. Armament safety switch — SAFE (guard closed).
   d. Igniter control switch — Training (safety-wired).

WARNING

Warn personnel and clear armament bay area.

2. Connect 28-volt dc external power or turn battery switch ON.
3. Pneumatic pressure-low warning light — Out (if the warning light is illuminated, have the high-pressure pneumatic system serviced).
4. Armament reset circuit breaker — In (do not push power circuit breaker in).
5. Armament door switch — CLOSE, then release (do not actuate launcher reset switch).
6. Armament door reset switch — RESET and hold. Do not actuate launcher reset switch.

Note

If air is heard entering the door cylinders and the green reset light illuminates, the system is reset and armament door operation can be continued.

7. Armament door reset switch — Release.
8. Armament door switch — OPEN (doors should open).
9. Armament reset circuit breaker — Out (air exhaust should be loud).
10. External power — Disconnect (or) battery switch — OFF.
11. Check that armament doors can be moved with little resistance.
12. Armament bay door safety locks — Installed (if available).

ARMAMENT DOOR CLOSING

To close armament doors, check for launchers up and locked and armament doors fully open. Repeat steps 1 through 3 above, then proceed as follows:

1. Armament bay door safety locks — Removed (if installed).
2. Armament door switch — OPEN, then release (do not actuate the launcher reset switch).
3. Armament door reset switch — RESET and hold.

Note

If air is heard entering the door cylinders and the green reset light illuminates, the system is reset and door operation can be continued.

4. Armament door reset switch — Release.
5. Armament door switch — CLOSE (doors should close).
6. Armament reset circuit breaker — Out (air exhaust should be loud).
7. External power — Remove (or) battery switch — OFF.

BARRIER PROBE OPERATION

To release the barrier probe when in the down position, depress the latch just inside the forward probe housing, raise probe to up position until it catches, then check security.

PRESSURIZING CONSTANT-SPEED DRIVE ACCUMULATORS

The pressurization valve is located in the right-hand engine access compartment by the constant-speed drive gage.

1. Remove valve cover.
2. Connect low-pressure air source.
3. Loosen hex nut approximately one-half turn.
4. Pressurize until gage reads in the green.
5. Tighten hex nut.
6. Disconnect air source.
7. Replace valve cover and check for leaks.

LANDING GEAR STRUT SERVICING

Lowering Strut

1. Remove valve cap cover and loosen valve one-half turn.
2. Depress valve stem, releasing small amount of air at a time, then rock the wing briefly. Check for proper strut extension.

Extending Strut

1. Remove valve cap cover and connect air source.
2. Loosen valve one-half turn and pressurize to proper extension.
3. Tighten valve and remove air source. Replace valve cap, then check for leaks.

DRAINING OVERFULL HYDRAULIC RESERVOIR

Primary Reservoir

The drain line is located inside rear bulkhead of right-hand armament bay.

Note

Open armament doors using procedures previously outlined.

1. Depressurize reservoir by depressing button on top of reservoir cap.
2. Remove drain line cap cover.
3. Hold button on top of reservoir depressed or remove cap to allow excess fluid to drain.
4. Drain until fluid level is ¼ inch below full mark.
5. Release button. Replace reservoir cap.
6. Replace drain line cap cover and check for leaks.
7. Pressurize reservoir and recheck fluid level.

Note

If fluid level decreases more than ¼ inch, excessive air is in the system.

8. Close armament doors as previously outlined.

Secondary Reservoir

The drain line is located on the forward side of the right-hand wheel well. The procedure for draining the secondary reservoir is the same as for the primary except that the fluid level should not decrease more than 1/8 inch when the reservoir is pressurized.

SERVICING HYDRAULIC RESERVOIR AND ACCUMULATORS

The reservoir and accumulator air charge fittings are located in the RAT compartment.

1. Operate control stick to bleed system pressure.
2. Extend the RAT.
3. Primary and secondary accumulator air gages — 750 (±25) psi. If air pressure is low, charge with dry air or nitrogen to 750 (±25) psi and check for leaks.

Note

To pressurize accumulators, remove cap from fitting, connect air source, loosen fitting one-half turn, then apply pressure. Tighten fitting prior to removing air source.

4. Check hydraulic reservoir sight gages for correct fluid level.
5. If fluid level is at refill, release air pressure by depressing the button on top of the reservoir cap. Remove reservoir cap and service with hydraulic fluid to 1/4 inch below full mark.
6. Replace reservoir cap.
7. Pressurize reservoirs with low-pressure air source. The quick-disconnect is located at the top center of the RAT compartment.

Note

The airplane is unacceptable for flight if the fluid level in the primary reservoir decreases over 1/4 inch or the secondary decreases over 1/8 inch after pressurization.

REFUELING (SINGLE-POINT ONLY)

Note

If refueling source with single-point adapter is not available and the airplane is equipped with external tanks, the airplane may be fully serviced by filling the external tanks, then starting the engine and transferring fuel into the internal wing tank system. Repeat this procedure until the airplane is fully serviced.

Refuel the airplane with JP-4. Refer to Section V for emergency fuel and limitations. The filler adapter is located on the rear bulkhead of the left wheel well. Remove the red dust cap from refueling adapter and attach fuel hose nozzle. Shutoff valve on nozzle shall be in off position. Start pump on fuel truck and adjust fuel pressure to 55-60 psi. Open shutoff valve on fuel hose nozzle and maintain 55 to 60 psi. Check for normal exhaust of air from left or right fuel tank vents at the beginning and during refueling.

CAUTION

If air does not flow from each wing vent, shut off fuel flow and investigate the trouble.

A normal discharge of fuel from refueling adapter drain will occur after refueling hose is removed. Tank should be allowed to completely drain before filler cap is installed. Check fuel filler cap installed with arrows aligned.

Note

Check fuel service against remaining fuel to insure full load.

HIGH-PRESSURE PNEUMATIC SYSTEM SERVICING

The high-pressure pneumatic system filler valve and pressure gage are located in the left main wheel well. Use high-pressure compressor (MC-1 or like equipment) which will produce 3200 psi. Service to 3000 psi. All precautions must be exercised to prevent contamination of the high-pressure pneumatic system.

CAUTION

Insure that combustion starter manual air valve in the left main wheel well is in the GROUND supply position. If high-pressure air is not available, air-motoring of starter is to be minimized.

WEATHER PROCEDURES

Canopy

The canopy should be left open (ground support tool installed) when exposed to direct rays of the sun at temperatures above 85°F. If the ground support tool is not available, leave the canopy cracked.

RADOME ANTI-ICING TANK SERVICE

The radome anti-icing tank is located in the right forward electronics compartment.

1. Release air pressure by depressing the release valve on filler cap.
2. Fill tank with anti-ice fluid (40% water — 60% glycol).

DRAG CHUTE INSTALLATION

1. Speed brakes — OPEN.
2. Speed brakes ground safety locks — Installed.
3. Inspect drag chute compartment for metal tears, cleanliness, screws flush and tight.

4. Square drag chute pack (risers should be 26 inches long).
5. Check riser gathering strap fully against "D" ring and check to see safety pin is properly aligned in pilot chute.
6. Place speed brake safety pin in position.
7. Place "D" ring in jaws.
8. Drag chute handle — Out. This will hold "D" ring securely in the jaws during installation).
9. Lay risers on bottom of drag chute compartment and install chute on top of risers. (Hold pressure against jaws to keep risers from jamming or batting up against "D" ring.)
10. Remove ground safety aft pilot chute pin.
11. Place top retainer strap over cone.
12. Place bottom retainer strap over cone.
13. Drag chute handle — In.
14. Install ball of ripcord pin in cable guide. Assure ball is in race and fingers on top of ball. Install ripcord pin in pilot chute cone.

Note

Do not pull pilot chute inner safety pin.

15. Drag chute handle — Out. If ripcord pin comes out (normal operation), repeat steps 13 and 14.

Note

If ripcord pin does not come out on operational check, a cocked "D" ring or malfunction exists and installation should not be continued until malfunction is corrected.

16. Pull pilot chute safety pin and do not touch drag chute thereafter.

COMBUSTION START — SECOND ATTEMPT

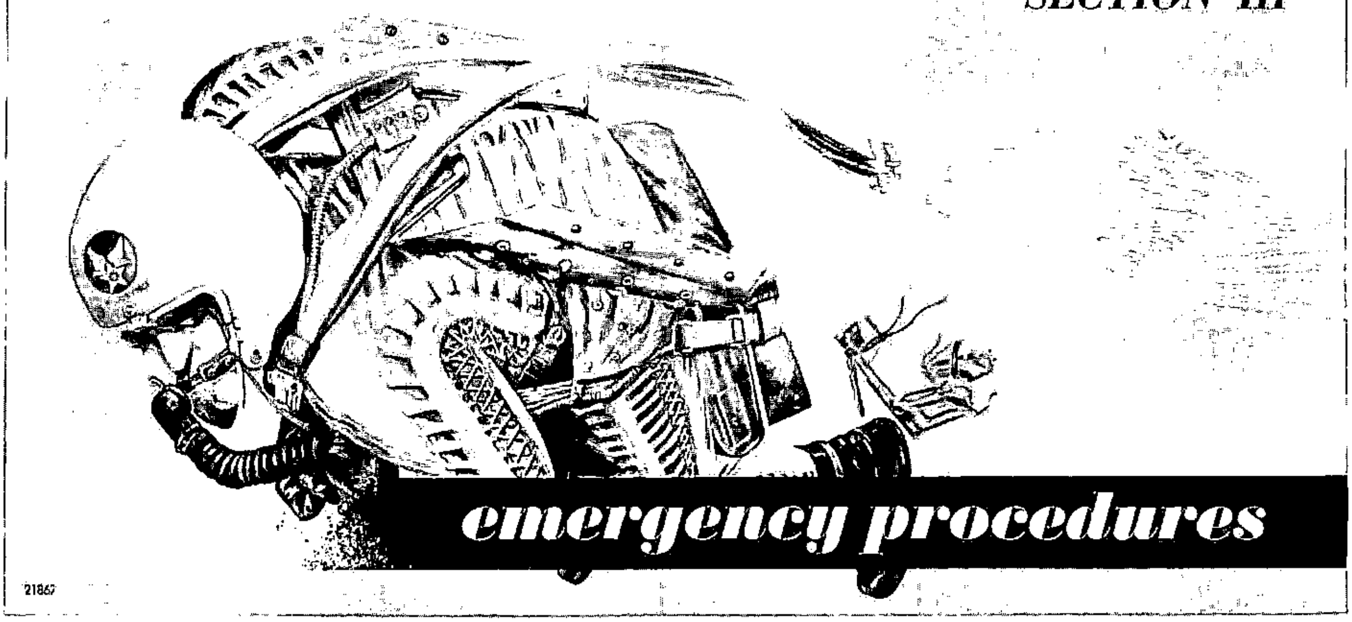
The following procedure should be followed to manually replenish the starter fuel flask. Gain entrance to the fuel flask through the right-hand engine access door.

1. Remove bleed line cap cover from starter flask bleed line.
2. Apply approximately 10 psi of air to fuel flask bleed line.
3. Listen for bottoming of the plunger in starter flask.
4. Remove air source and allow fuel to gravity feed from fuel tanks to fill starter fuel flask. Use a suitable container to collect fuel drainage from fuel flask.
5. Install bleed line cap cover.
6. Check for leaks.

Note

The Normal Abbreviated Check List is now contained in T.O. 1F-102A-(CL)1-1.

SECTION III



emergency procedures

TABLE OF CONTENTS

Note

Critical actions, which must be performed immediately and instinctively if the emergency is not to be aggravated or damage is to be avoided, are listed in bold face capital letters.

ENGINE FAILURE

Generally flameouts occur from improper fuel scheduling. Failure of components outside of the engine proper can often be detected by reference to the engine instruments or warning lights before a flameout occurs. Air starts should be attempted to restore engine thrust unless mechanical failure occurs within the engine. Should an engine mechanical failure occur, the fuel supply should be shut off by retarding the throttle to OFF, placing the fuel selector switch to OFF (fuel shutoff valve switches to CLOSE on some airplanes), and placing boost pump switches to OFF. Pilot's discretion should determine the action to be taken following an engine failure. Refer to EJECTION VS. FORCED LANDING, this Section.

Note

The airplane has normal flight characteristics with a dead engine except above speeds of approximately 220 KIAS. Above these airspeeds, the interruption of airflow through the engine inlet ducts creates a duct rumble which causes a very startling buffet and heavy vibration within the airframe. The severity of this rumble increases proportionally with IAS and is only slightly perceptible at best glide speed of 220 KIAS.

COMPRESSOR STALL

A breakdown of compressor airflow is known as a compressor stall. The compressor stall is induced by a change in the pattern of airflow being fed to the engine. The intensity of the stall is determined by the magnitude of the airflow pattern change and may affect a number, or all of the compressor blades. Stalls may be encountered at any speed. Retarding or advancing the throttle too rapidly can cause either deceleration or acceleration stalls, respectively. Without throttle movement, rapid changes of flight condition which distort the inlet duct airflow pattern can induce compressor stall under certain atmospheric conditions. Compressor stalls may be experienced during operation at high altitudes in areas of heavy ice crystal concentration such as found in or around thunderstorms.

Note

Refer to TURBULENCE AND THUNDERSTORMS, Section IX, for applicable procedures.

The ice crystals entering the inlet duct go into the engine with the air where they are heated during the compression process and become effectively, ingested water. The ingested ice crystals reduce the compressor stall margin, and compressor stall or flameout follows. These compressor stalls may occur as individual loud reports or as a series of loud reports in rapid order. Compressor stalls are accompanied by airframe vibration and in some cases, emission of vapor from the engine air intake duct. RPM drops to 80% or below, fuel flow drops to 800 pph and temporary loss of EPR occurs when a series of stalls are experienced. Fuel flow and rpm will fluctuate when individual stalls are encountered.

Note

The engine can sustain numerous rapid order type compressor stalls without engine damage.

Compressor stalls may be recognized by loss of thrust reflected through engine instruments, rapid reduction or fluctuation of engine pressure ratio at a constant throttle position, or failure of rpm to increase during acceleration. A compressor stall may be accompanied by a rise in exhaust gas temperature. The possibility of encountering compressor stall is increased during high-altitude operations as the thinner air does not conform as easily to smooth airflow through the compressor section as does the heavier air at low altitude. This results in a more easily disturbed compressor airflow pattern. There are several things that can be done to avoid compressor stall or to reduce its intensity. Erratic and abrupt throttle movements should be avoided. Rapid throttle advances during periods of high distortion of the air entering the air inlet duct, such as at low airspeeds, can cause acceleration stalls. Coordinated flying increases the efficiency of the compressor inlet air duct. Airspeeds should be maintained above the acceptable minimum. Following a single loud explosion, immediately check for signs of fire (trailing smoke or flames and fire warning system indications) to determine whether a compressor stall or engine failure has occurred. When compressor stalls occur, or when operating in heavy ice crystal concentrations, proceed as follows:

1. Ignition button — Depress and hold (airplanes with all-points ignition.) Move throttle inboard if in AFTERBURNER.

CAUTION

Continuous use of the ignition system in excess of ten minutes, or consecutive continuous usages without a ten-minute interim cooling period may result in damage to the ignition system which will render it inoperable for restarts. Energizing the ignition system will not prevent compressor stall, but it will aid in preventing flameout.

2. Check engine instruments for engine mechanical failure and overtemperature.
3. Do not retard throttle unless engine failure or overtemperature is evident.

Note

If overtemperature condition exists, retard throttle slowly to prevent exceeding EGT limits. During operation above 40,000 feet, avoid retarding throttle below 85% rpm.

4. Maintain normal coordinated straight and level flight attitudes until engine stall condition is relieved.

Note

Maintain unaccelerated flight until engine stall condition is relieved, if possible. If not, maintain altitude and heading until airspeed drops to 220 KIAS then establish glide speed of 220 KIAS and maintain heading. Do not open speed brakes.

5. If an overtemperature persists, fuel control switch to EMERGENCY.
6. **IF EGT EXCEEDS MAXIMUM ACCELERATION LIMIT, THROTTLE TO OFF.**

   Retard throttle to OFF if above procedures do not keep EGT with maximum acceleration limits (refer to ENGINE OPERATING LIMITS, Section V).
7. Restart engine.

   After a stabilized condition is obtained, restart the engine. Refer to ENGINE AIR START, this Section, for restarting procedures.

Note

Record on Form 781, any compressor stall and indicate duration and peak temperatures of any overtemperature operation.

Off-idle compressor stalls (sometimes referred to as "choo-choo" or acceleration stalls) may be experienced between idle and approximately 80% rpm while accelerating during engine ground runup. If a stall occurs as the throttle is being advanced in this range, momentarily stop or slow throttle advancement as required to correct the stalled conditions. Off-idle stall or "choo-choo" is of no consequence unless it is severe to the degree that more than 15 seconds are required for the engine to accelerate from 65% to FULL MIL POWER. If more than 15 seconds are required, the engine should be shut down and the cause determined before flight.

ENGINE FAILURE DURING TAKEOFF

Engine Failure During Takeoff Run (Before Airborne)

If engine failure occurs before the airplane is airborne, abort takeoff; refer to ABORT, this Section.

Engine Failure During Takeoff (With Airplane Airborne)

If engine failure occurs during takeoff but after becoming airborne, an immediate relight may be possible on airplanes with all points ignition*. If this is unsuccessful, the proper course of action depends on altitude at time of failure, automatic escape equipment capabilities (see figure 3-4), and terrain features of the available landing area. If the decision is to land, proceed as follows:

1. **THROTTLE — OFF.**
2. RAT handle — Pull & hold for four seconds.

   Pull the RAT handle to displace the ram air turbine. Hold the handle in the fully extended position for four seconds to insure a satisfactory extension of the turbine.
3. **LANDING GEAR HANDLE — DOWN.**
4. External tanks jettison button — Depress (if required).

   Depress the external tanks jettison button to jettison the external tanks if tanks are installed and contain fuel. If external tanks are empty, it is recommended that they be retained to cushion the impact.
5. **CANOPY — JETTISON (IF NECESSARY).**

   Raise the canopy jettison handle on the left-hand armrest if it is necessary to jettison the canopy.

Airplanes modified by TCTO 1F-102-746.

WARNING

- If canopy is to be jettisoned, make sure it is jettisoned before touchdown. Otherwise, sparks from the canopy remover may cause a fire if fuel is spilled in the vicinity of the airplane, or the canopy may become jammed.
- If the canopy is jettisoned below 175 KIAS while airborne it may strike the fin but probably will not result in loss of control.

6. Shoulder harness inertia reel handle — LOCKED.

CAUTION

The pilot is prevented from bending forward when the shoulder harness is locked; therefore, all switches not readily accessible should be "cut" before moving the shoulder harness inertia reel handle to LOCKED position.

7. Fuel selector switch — OFF (some airplanes).

   Fuel shutoff valve switches — CLOSE (other airplanes).
8. **DRAG CHUTE HANDLE — PULL (UPON CONTACT).**
9. Master switch — OFF (some airplanes); TRIP, then OFF (other airplanes).
10. **ABANDON AIRPLANE AS SOON AS POSSIBLE.**

WARNING

Because of the height of the cockpit above the ground (eight feet), care should be exerted when abandoning the airplane without a ladder to prevent bodily injury.

ENGINE FAILURE DURING FLIGHT

(Air Start Not Probable)

Note

If engine failure occurs but there is no indication of mechanical failure within the engine, an air start may be attempted. Refer to ENGINE AIR START, this Section.

When an air start is not probable, use the following procedure:

1. Throttle—OFF.

2. Airspeed 220 KIAS (gear up, speed brakes closed).

   Establish a glide speed of 220 KIAS with gear up and speed brakes closed.

3. AC bus switch—EMER (some airplanes); RESET, then ON (other airplanes).

   Engine windmilling rpm is insufficient to provide normal ac generator operation. The emergency ac generator should be energized to supply power for flight and engine instruments.

Note

A "frozen" engine will result in complete loss of ac power.

4. RAT handle—Pull (if necessary) and hold for four seconds.

   Pull the RAT handle to displace the RAT if immediate emergency flight control hydraulic pressure is necessary. The handle should be held in the fully extended position for four seconds to insure a satisfactory extension of the ram air turbine.

WARNING

- If engine failure results in a frozen engine, it will be necessary to immediately displace the RAT to obtain hydraulic pressure for flight control operation.
- If airspeed is above RAT maximum extension speed and engine is frozen, do not extend the speed brakes to slow the airplane. Speed brakes extension will cause immediate depletion of remaining secondary hydraulic system pressure. Deceleration should be accomplished by moving the flight controls a minimum amount to establish a flight attitude that will provide deceleration.

Note

If engine failure results in a windmilling engine, there will normally be sufficient hydraulic pressure available for flight control operation, unless airspeed is low, without extending the RAT. Rapid movement of the flight controls should be avoided. Once extended the RAT cannot be retracted during flight.

5. Fuel control switch—EMER (if required).

   If fuel control failure is suspected, place fuel control to EMER and check that emergency fuel control warning light illuminates.

6. Fuel selector switch—ENGINE (some airplanes).

   Fuel shutoff valve switches—OPEN (other airplanes).

7. Nonessential electrical equipment—OFF.

   Turn nonessential electrical equipment off to reduce battery load.

CAUTION

At engine speeds below approximately 40% rpm, dc generator output is not available and the battery becomes the only source of power to the dc essential bus. Usable battery power is available for approximately 5 to 20 minutes. On airplanes equipped with an emergency dc bus*, the transformer-rectifier will automatically be connected to the emergency dc bus when dc generator output is lost. The ac system is then self-sustaining and will not be affected by battery power depletion.

Note

Attempt an air start if there is no indication of mechanical failure within the engine. Refer to ENGINE AIR START, this Section.

ENGINE FAILURE DURING FLIGHT AT LOW ALTITUDE

If the engine fails during flight at extremely low altitude, and sufficient airspeed is available, the airplane should be pulled up (zoom-up) to exchange airspeed for an increase in altitude. This will allow more time for accomplishing subsequent emergency procedures (air start, establishing forced landing pattern, ejection, etc.).

Note

The point at which climb should be terminated will depend on whether the pilot intends to eject or whether he intends to continue attempting air starts, establish forced landing pattern, etc. In any event, it is recommended that air start be attempted immediately upon detection of engine flameout and repeated as many times as possible during the zoom-up. If the decision is to eject, the airplane should be allowed to climb as far as possible. Ejection should be accomplished while the nose of the airplane is above the horizon but prior to reaching a stall or sink. If the decision is to continue attempting air starts, the climb should be terminated before the airspeed drops below best glide speed in order that engine windmilling rpm will not drop below the minimum required for air start.

In the zoom-up maneuver, more altitude can be gained if external tanks are jettisoned. Maximum altitude gain can be achieved by jettisoning external tanks prior to zoom-up. However, when jettisoning external tanks, consideration must be given to such factors as sufficient airspeed to

**Airplanes modified by TCTO 1F-102-727.*

allow time for pilot reaction and an unpopulated area where the tanks will fall. In any event, the decision to jettison or retain external loads must be made by the pilot on the basis of his evaluation of the above factors and conditions existing at the time of the emergency.

ENGINE AIR START

Engine air starts are accomplished by utilizing the same basic sequence as a normal ground start but using windmilling effect rather than the starter to motor the engine. An air start should be attempted as soon after flameout as possible regardless of airspeed. If this immediate attempt is unsuccessful, there are certain conditions that should be established. Without starter operation, airspeed must be controlled to obtain optimum rpm for an air start. The best engine windmilling speed for obtaining an engine air start is between 15 and 30% rpm. The best glide speed of 220 KIAS will provide this rpm between sea level and approximately 35,000 feet. Above this altitude, at 220 KIAS, the minimum stabilized rpm will increase by approximately one percent per 1000 feet. Due to excessive altitude loss at other than best glide speed, it is recommended that 220 knots be used for all air starts, regardless of altitude. The probability of a relight at 220 KIAS is increased below approximately 35,000 feet. A windmilling engine will not deliver ac power directly through the main ac generator, but will produce and maintain sufficient hydraulic pressure to drive the emergency ac generator. In the event of a flameout condition, it is necessary to switch to the emergency ac generator to maintain power for flight and engine instruments and under night conditions, instrument panel and console lights. See figure 3-1 for engine air start procedure.

Unsuccessful Air Start

1. Throttle — OFF.

   Retard the throttle to OFF if during an air start any one of the following conditions prevail:

   a. Light up does not occur within 20 seconds after throttle has been advanced to IDLE if there was positive fuel flow indication.

   Note

   If source of flameout is due to a temporary interruption of fuel flow from the fuel tanks, restart may take up to four minutes. This condition can be detected by absence of fuel flow indication.

   b. Engine fails to accelerate to idle within approximately 45 seconds after lightup.

2. Attempt another air start.

If the engine cannot be restarted on the normal or emergency fuel systems, prepare to make a forced landing or abandon the airplane. Refer to applicable procedures, this Section.

MAXIMUM GLIDE

Maximum glide distances with a windmilling engine are obtained with an airspeed of 220 KIAS. See figure 3-2 for additional information.

FIRE

A steady illumination of the fire warning light is an indication of a fire in the forward engine compartment. A flashing fire warning light is an indication of a fire in the aft engine compartment. The hot section is enclosed in an engine-mounted fireproof cannular shroud and there is no lateral firewall; thus, a flashing light could be due to a fire between the engine shroud and the airplane fuselage structure and not merely an overheat of the engine and/or airplane structure. Therefore, a warning from one compartment can be considered no less serious than a warning from the other. A fire warning may be the result of hot gases leaking from the engine or fire fed by fuel leaking from the engine and thus is a function of engine thrust. Following a fire warning in flight, engine thrust should be immediately reduced below the operating point of the afterburner mechanical shutoff (approximately 80% thrust) and then eased off until the fire warning light extinguishes or the minimum thrust necessary to maintain safe ejection altitude is reached. In flight the throttle should be stop-cocked only after a positive verification of fire has been made. Following a fire warning on the ground, however, where power is not required, the engine should be stop-cocked immediately.

ENGINE FIRE DURING STARTING

Fire Warning Light Illuminated During Start

If the engine fire warning light illuminates or there is visible evidence of fire during starting or ground operations proceed as follows:

1. **THROTTLE — OFF.**
2. Fuel selector switch — OFF (some airplanes).

   If the switch is safety-wired and has been rotated or safety wire broken, it must be re-established that both valves are open before the next flight and before safety wire is replaced.

   Fuel shutoff valve switches — CLOSE (other airplanes).
3. Fire fighting equipment — Summon.
4. Master switch — OFF (some airplanes); TRIP, then OFF (other airplanes).
5. Abandon the airplane.

Excessive Exhaust Gas Temperature During Start

1. Starting air — Connected.

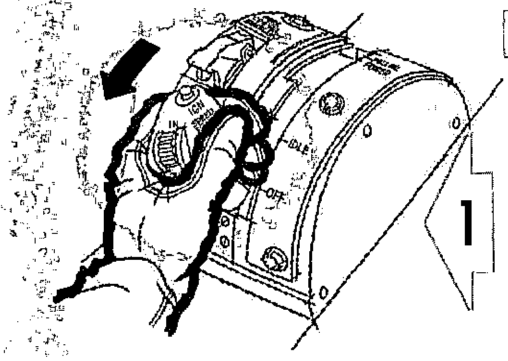
   Signal crew chief to connect a supply of compressed air to the high-pressure pneumatic ground connection for motoring.

SFS 1-5

IMMEDIATE AIRSTART

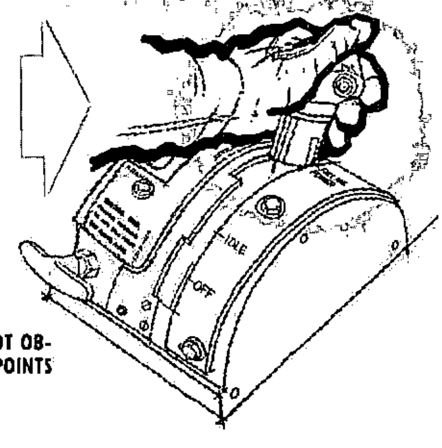
IGNITION BUTTON—DEPRESS (AIRPLANES WITH ALL POINTS IGNITION).

ATTEMPT IMMEDIATE AIR START BY DEPRESSING AND HOLDING THE IGNITION BUTTON. MOVE THROTTLE INBOARD (IF IN AFTERBURNER). SLOWLY RETARD TO IDLE AS RPM DROPS.



RESTARTING

(IMMEDIATE AIRSTART NOT POSSIBLE)



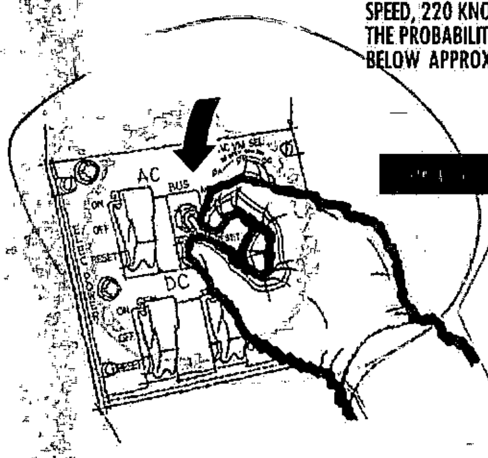
1

THROTTLE—OFF (IMMEDIATE RELIGHT NOT OBTAINED AND AIRPLANES WITHOUT ALL POINTS IGNITION).

2

AIRSPEED 220 KIAS

DUE TO EXCESSIVE ALTITUDE LOSS AT OTHER THAN BEST GLIDE SPEED, 220 KNOTS IAS IS RECOMMENDED FOR ALL AIR STARTS. THE PROBABILITY OF A RELIGHT AT 220 KNOTS IAS INCREASES BELOW APPROXIMATELY 35,000 FEET.



[redacted]

3

AC BUS SWITCH—EMER (SOME AIRPLANES); RESET, THEN ON (OTHER AIRPLANES).

ENGINE WINDMILLING RPM IS INSUFFICIENT TO PROVIDE NORMAL AC GENERATOR OPERATION. THE EMERGENCY AC GENERATOR SHOULD BE ENERGIZED TO SUPPLY AC POWER FOR FLIGHT AND ENGINE INSTRUMENTS.

NOTE

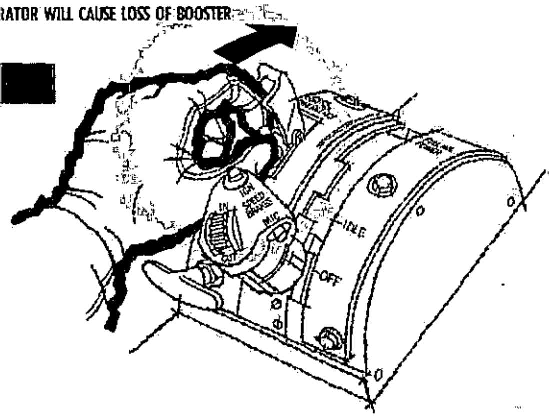
LOSS OF MAIN AC GENERATOR WILL CAUSE LOSS OF BOOSTER PUMP OPERATION.

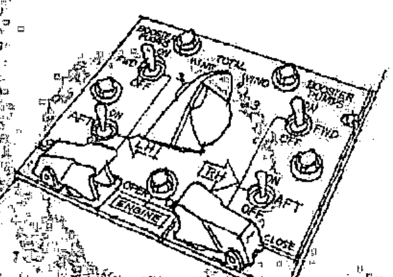
[redacted]

4

FUEL CONTROL SWITCH—EMERGENCY (IF REQUIRED)

IF FUEL CONTROL FAILURE IS SUSPECTED, PLACE FUEL CONTROL SWITCH TO EMERGENCY AND CHECK THAT EMERGENCY FUEL CONTROL WARNING LIGHT ILLUMINATES.





5

FUEL CONTROL PANEL—CHECK

FUEL SELECTOR SWITCH—ENGINE (SOME AIRPLANES).

FUEL VALVE SHUTOFF SWITCHES—OPEN (OTHER AIRPLANES).

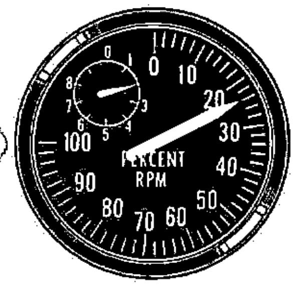
21868-1

Figure 3-1

engine air start procedures

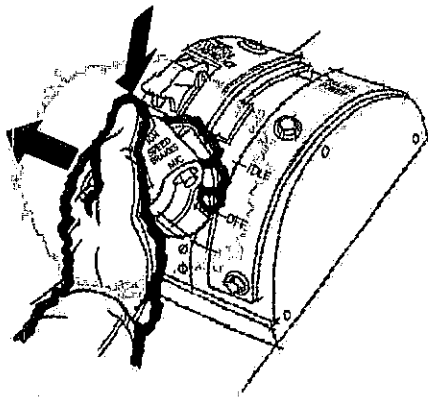
6 RPM 15% TO 30%

AIRSTARTS SHOULD BE MADE WITH ENGINE WINDMILLING BETWEEN 15 AND 30 PERCENT RPM—AN AIRSPEED OF 220 KNOTS IAS WILL PROVIDE THIS RPM BETWEEN SEA LEVEL AND APPROXIMATELY 35,000 FEET. AS ALTITUDE INCREASES, RPM WILL INCREASE.



WARNING

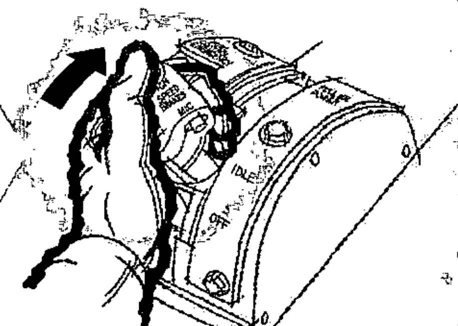
IF HYDRAULIC PRESSURE FOR FLIGHT CONTROL OPERATION BECOMES MARGINAL AT LOW ENGINE RPM, EXTEND THE RAT

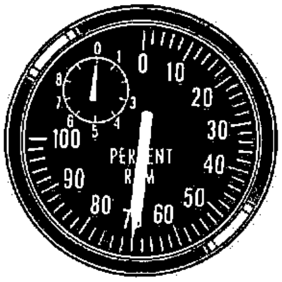


7 DEPRESS AND HOLD IGNITION BUTTON: THROTTLE OUTBOARD TO START. (AIRPLANES WITHOUT ALL POINTS IGNITION.)

NOTE
IT IS NECESSARY TO MOVE THE THROTTLE FULL OUTBOARD TO START POSITION TO ARM THE IGNITION CIRCUIT ON SOME AIRPLANES.

8 THROTTLE—IDLE



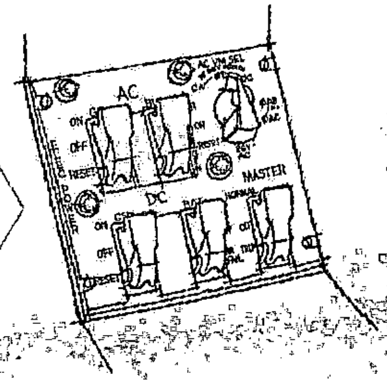


9 RPM 60% TO 80%, THEN RELEASE IGNITION BUTTON

AFTER RESTART

1. DC GENERATOR SWITCH—RESET THEN ON
2. AC BUS SWITCH—NORMAL
3. BOOSTER PUMP SWITCHES—OFF

   NOTE
   TURN BOOSTER PUMPS OFF TO REDUCE LOAD ON AC GENERATOR.
4. AC GENERATOR SWITCH—RESET THEN ON
5. BOOSTER PUMP SWITCHES—ON (ONE AT A TIME)



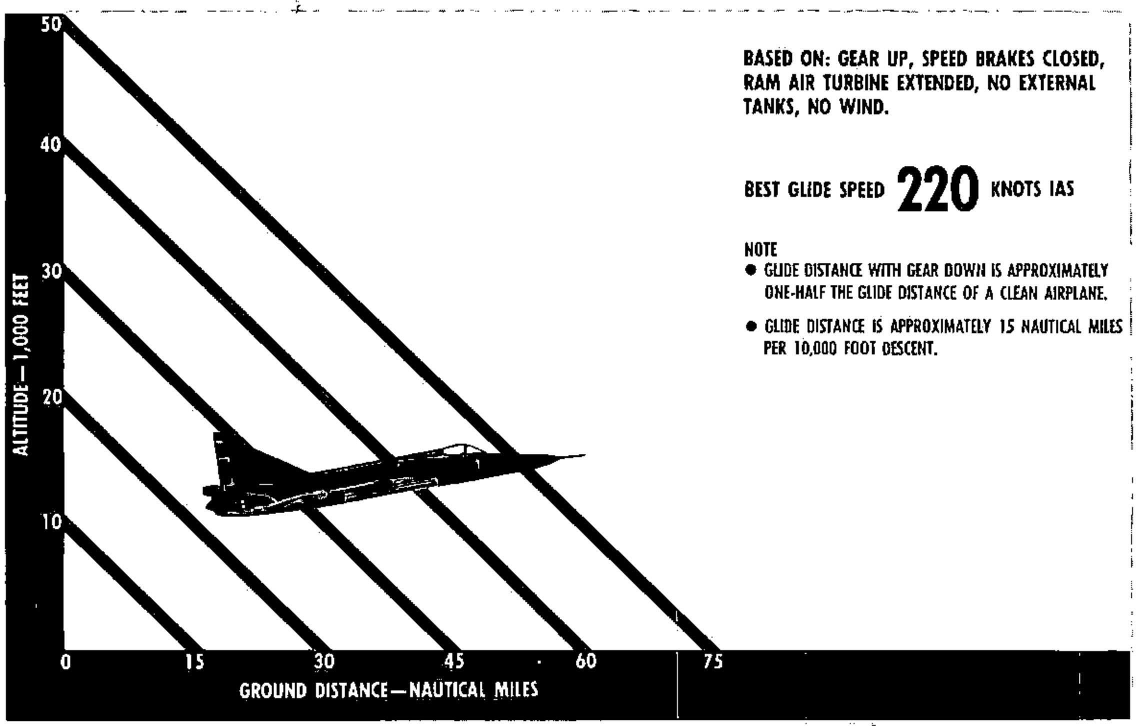
21868-2

maximum glide distances

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
FUEL GRADE: JP-4

WINDMILLING OR FROZEN ENGINE



21869

Figure 3-2

Note

In the event that a ground cart is not available, air for motoring the engine may be taken from the airplane high-pressure pneumatic system supply flasks by opening the manual shutoff valve in the left main landing gear wheel well.

2. Engine ignition disconnect switch — OFF.

   While in the main wheel well, crew chief lifts the switch guard and actuates the engine ignition disconnect switch.

Note

With the switch in the OFF position, the combustion starter may be fired but the engine will not light up since the engine ignition has been disconnected.

3. **THROTTLE — OFF.**
4. Boost pump switches — OFF.
5. Fuel selector switch — OFF (some airplanes).

   If the switch is safety-wired and has been rotated or safety wire broken, it must be re-established that both valves are open before the next flight and before safety wire is replaced.

   Fuel shutoff valve switches — CLOSE (other airplanes).
6. Throttle — START.

   Move throttle outboard to START to air motor the starter. Hold this position until rpm indication is evident on the tachometer.
7. Ignition button — Depress.
8. Throttle — OFF.

   As soon as definite rpm indication is noted, hold ignition button and move throttle inboard to the OFF position. This will fire the combustion starter and aid in clearing the engine.

9. Ignition button — Release.

   Release the ignition button after the engine clears or if the fire cannot be extinguished by clearing.

10. Abandon the airplane.

ENGINE FIRE DURING TAKEOFF

The exact procedure to follow for a fire warning during takeoff depends on the condition of the emergency. Airspeed, altitude, length of runway and overrun available, location of populated areas, etc., have to be considered before the required action is taken. The following procedures are recommended:

1. If runway and overrun area permit, abort takeoff; refer to ABORT, this Section.
2. If committed to takeoff:
   a. External tanks jettison button — Depress (if required).

      Jettison the external tanks if tanks are installed and contain fuel and drop area is unpopulated.

   b. **MAINTAIN MAXIMUM THRUST UNTIL SAFE EJECTION ALTITUDE IS REACHED.**

      Adjust thrust to maintain safe ejection altitude.

   c. Check for fire.

      Check for positive indications of fire, such as abnormal engine instrument readings, smoke in cockpit, trailing smoke or flame, or report from ground or another airplane.

   d. **IF FIRE EXISTS — EJECT.**

   e. If fire cannot be confirmed, make decision to land or eject.

Note

See figure 3-4 for emergency minimum ejection altitudes.

ENGINE FIRE IN FLIGHT

1. **REDUCE THRUST.**

   Reduce thrust to minimum necessary to maintain safe ejection altitude.

CAUTION

At high altitude, compressor stall may occur as a result of large thrust reductions. Exhaust gas temperature may rise as a result of compressor stall and should not be taken as a positive indication of fire.

2. If warning light extinguishes:
   a. Check warning light.
   b. Continue flight at minimum safe thrust.
   c. Land as soon as practicable.
3. If warning light remains on — Check for fire.

   Determine whether a fire actually exists by a report from another airplane or the ground, fumes, heat, cockpit smoke, trailing smoke following a turn, abnormal airplane responses, or abnormal engine instrument readings.

4. Fire not evident by check:
   a. Continue flight at minimum safe thrust.
   b. Land as soon as possible.
5. Fire evident:
   a. Throttle — OFF.
   b. Fuel selector switch — OFF (some airplanes). Fuel shutoff valve switches — CLOSE (other airplanes).
   c. Master switch — OFF (some airplanes); TRIP, then OFF (other airplanes).

Note

The fire warning system is deactivated by turning off the master switch and the fire warning light will go out.

6. If fire ceases — Eject or make forced landing.
7. **IF FIRE CONTINUES — EJECT; REFER TO EJECTION PROCEDURES, THIS SECTION.**

ENGINE FIRE AFTER SHUTDOWN

If engine fire is suspected after engine shutdown on the ground, use the following procedure for clearing the engine:

1. Starting air — Connected.

   Signal crew chief to connect a supply of compressed air to the high-pressure pneumatic ground connection for motoring.

Note

In the event that a ground cart is not available, air for motoring the engine may be taken from the airplane high-pressure pneumatic system supply flasks by placing the manual shutoff valve in the left main landing gear wheel well to the AIRCRAFT position.

2. Engine ignition disconnect switch — OFF.

   While in the main wheel well, crew chief lifts the switch guard and actuates the engine ignition disconnect switch.

Note

With the switch in the OFF position, the combustion starter may be fired but the engine will not light up since the engine ignition has been disconnected.

3. **THROTTLE — CHECK OFF.**
4. Boost pump switches – Check OFF.
5. Fuel selector switch — OFF (some airplanes).
   Fuel shutoff valve switches — CLOSE (other airplanes).
6. Ignition button – Depress and hold.
7. Throttle — START.
   Move throttle outboard to START to air motor the starter. Hold this position until rpm indication is evident on the tachometer.
8. Throttle — OFF.
   As soon as definite rpm indication is noted, hold ignition button and move the throttle inboard to the OFF position. This will fire the combustion starter and aid in clearing the engine.
9. Fire fighting equipment — Summon (if necessary).
10. Ignition button — Release.
    Release the ignition button after the engine clears or if the fire cannot be extinguished by clearing.
11. Abandon the airplane (if fire continues).

ELECTRICAL FIRE

There is no system to warn of electrical fire in this airplane. Circuit breakers protect most of the circuits and tend to prevent electrical fire. If an electrical fire occurs, however, and its source cannot be readily determined visually, attempt to isolate and eliminate the fire as follows:

1. AC and dc generator switches—OFF.
   Turn the generator switches OFF to eliminate electrical power to the nonessential buses.
2. Emergency AC generator—ON.
   Turn the emergency ac generator on to energize the essential buses and the emergency dc bus.
3. Master switch—OFF (if fire or smoke persists).
4. Land as soon as practicable if fire subsides.

Note

It will be necessary to turn the master switch on long enough to facilitate speed brakes operation if landing is to be made and drag chute operation is desired.

5. **IF FIRE CONTINUES AND BECOMES SEVERE — EJECT.**

SMOKE, FUMES, OR FOG ELIMINATION

Should smoke, fumes, or fog enter the cockpit, proceed as follows:

1. Oxygen regulator diluter lever — 100% OXYGEN.
2. Oxygen regulator emergency toggle lever pushed either way from center.
3. **CABIN AIR SWITCH — RAM (BELOW 25,000 FEET IF POSSIBLE).**

EJECTION

Note

For considerations affecting the decision to eject, refer to EJECTION VS FORCED LANDING, this Section.

Every emergency in which ejection is considered will have its particular set of circumstances, involving such factors as airplane speed, attitude and control, as well as altitude. However, a decision should be made before takeoff as to what action is to be taken in the event of an emergency, particularly at low altitude. The ejection seat should be used to abandon the airplane in flight. The airplane should be slowed as much as possible if at high airspeeds and wings should be level. In any low-altitude ejection (below 2000 feet) the possibility of success can be materially improved by "zooming" the airplane and ejecting while the nose of the airplane is above the horizon (wings level) and the airspeed is above 120 knots. The zoom will exchange airspeed for altitude, thus providing maximum terrain clearance at time of ejection as well as reducing airspeed within safe limits for ejection. Ejecting while the nose of the airplane is above the horizon results in a more nearly vertical trajectory of the seat and crew member, thus providing more altitude and time for seat separation and parachute deployment. At low altitudes, a minimum airspeed of 120 KIAS is recommended to assure rapid deployment of the chute.

WARNING

- When the airplane is in a descending attitude and cannot be leveled, ejection should not be delayed as this will reduce the possibility of a successful ejection.
- Under level flight conditions, ejection should be accomplished above 2000 feet whenever possible.
- Under spin or dive conditions, ejection should be accomplished above 10,000 feet.
- Eject at the lowest practical airspeed above 120 KIAS (lowest practical would be that speed below which level flight cannot be maintained).

If possible, ejection should be made at a speed between 120 and 525 KIAS since relatively minor forces are exerted on the body. Betwen 525 and 600 KIAS, appreciable forces will be exerted on the body. Above 600 KIAS, ejection is extremely hazardous since excessive forces are exerted on the body. The structural limits of

zero delay lanyard engagement requirements

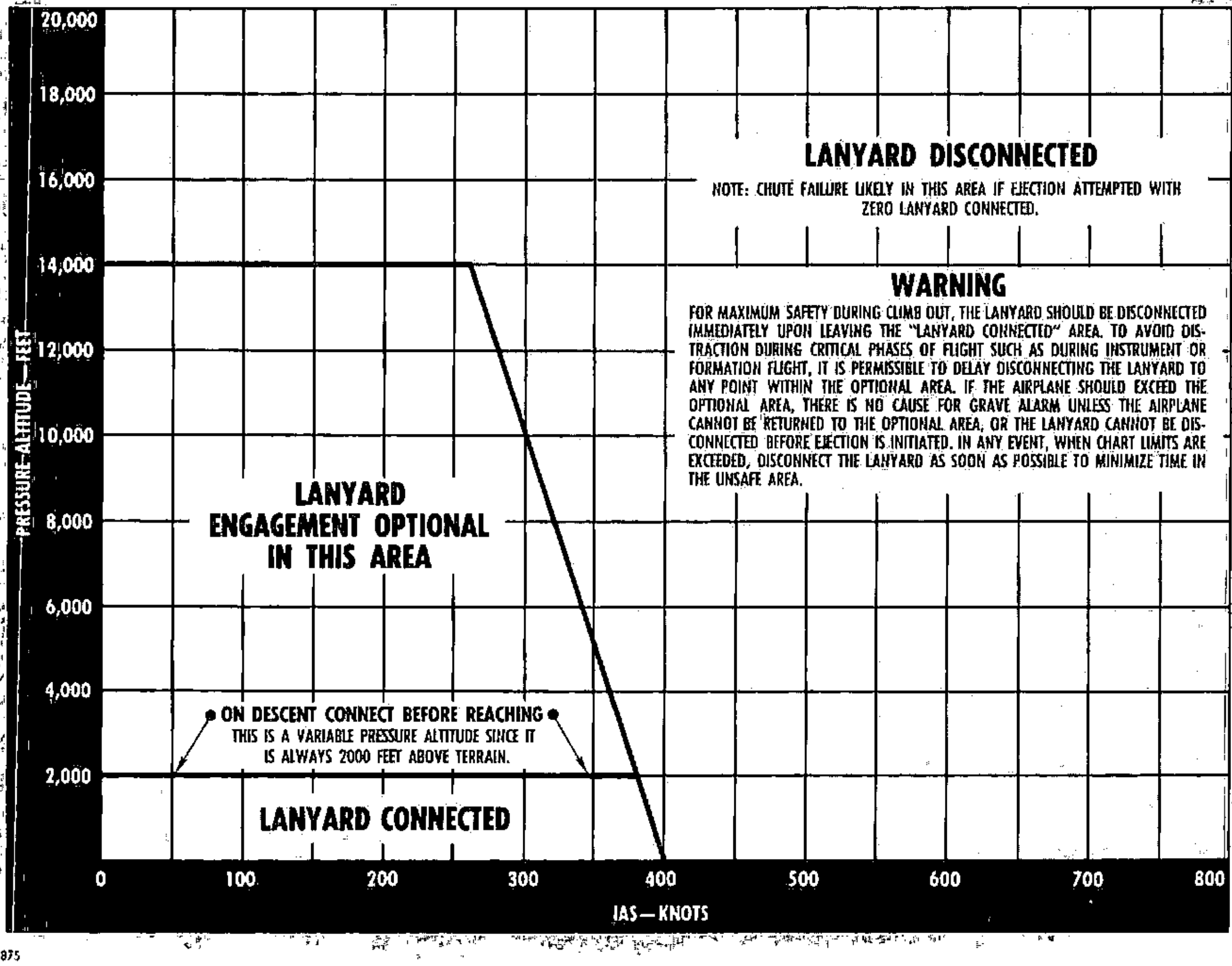


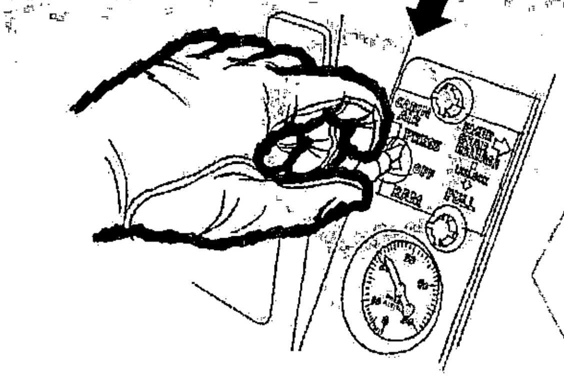
Figure 3-3

the seat may be exceeded at speeds in excess of 650 KIAS. There is no need to quote maximum airspeeds for ejection. If the airplane is controllable, airspeed will be reduced to as near 120 KIAS as practicable, which eliminates any high-speed problem. If the airplane is not controllable, ejection must be accomplished at whatever airspeed exists at the time, since ejection offers the only opportunity for survival. Speeds and altitudes in which the zero delay lanyard is connected or disconnected are outlined in the Zero Delay Lanyard Requirements Chart, figure 3-3. The chart is self explanatory in that it is divided into three positive areas; lanyard connected area, lanyard engagement optional area, and lanyard disconnected area. The pressure altitude feet scale indicates pressure altitude above sea level.

CAUTION

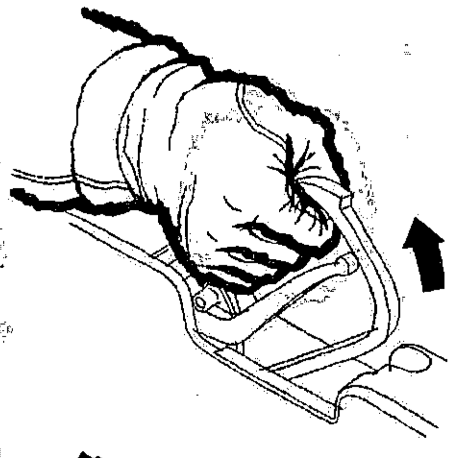
- The lanyard connected area is a variable pressure altitude area the top of which is always 2000 feet above the surrounding terrain.
- With the zero delay lanyard connected, the maximum speed for ejection must be as indicated on the Zero Delay Lanyard Engagement Requirements Chart, figure 3-3, to avoid parachute failure.

The Emergency Minimum Ejection Altitudes Table is presented on figure 3-4 and covers all possible combinations of seats, belts, and parachutes. The figures given

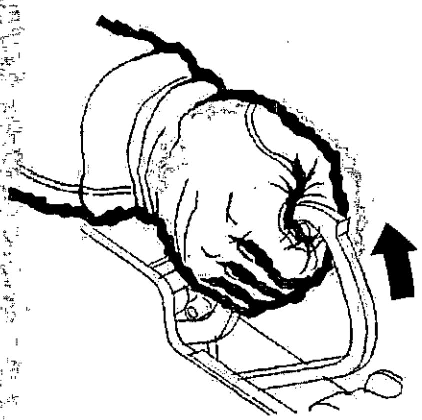


NOTE
- DO NOT MANUALLY OPEN THE AUTOMATIC OPENING SAFETY BELT PRIOR TO EJECTION AT ANY ALTITUDE.
- AFTER A 1 SECOND TIME DELAY THE SAFETY BELT WILL AUTOMATICALLY OPEN, THE FORCE REQUIRED TO SEPARATE THE PILOT FROM THE SEAT IS SUFFICIENT TO BREAK ALL PERSONAL LEAD CONNECTIONS.

1 TO PREVENT EXPLOSIVE DECOMPRESSION AT HIGH ALTITUDE, PLACE CABIN AIR SWITCH TO RAM.



2 PLACE ARMS ON ARMRESTS AND PULL EITHER OR BOTH HANDGRIPS TO JETTISON CANOPY. (SHOULDER HARNESS AUTOMATICALLY LOCKS WHEN HANDGRIPS ARE RAISED.)



3 SQUEEZE EITHER OR BOTH TRIGGERS TO EJECT SEAT.

WARNING
- THESE ARE EMERGENCY MINIMUMS. EJECTION SHOULD BE STARTED ABOVE 2000 FEET IF POSSIBLE.
- AT LOW ALTITUDES, A MINIMUM AIRSPEED OF 120 KIAS IS RECOMMENDED TO ASSURE RAPID DEPLOYMENT OF THE CHUTE.

EMERGENCY MINIMUM ALTITUDES FOR EJECTION (LEVEL FLIGHT)

	2 SECOND PARACHUTE (F-1A TIMER) B-5 PACK / C-9 CANOPY	1 SECOND PARACHUTE (F-1B TIMER) B-5 PACK / C-9 CANOPY	0 SECOND PARACHUTE (LANYARD TO "D" RING) B-5 PACK / C-9 CANOPY
AUTOMATIC SAFETY BELT WITH 1 SECOND (M12) INITIATOR	300	125	0

21872-1

Figure 3-4

ejection procedures

BEFORE EJECTION, IF TIME AND CONDITIONS PERMIT

1. THROTTLE—OFF.
2. STOW ALL LOOSE EQUIPMENT.
3. IFF TO EMERGENCY.
4. ACTUATE BAILOUT OXYGEN BOTTLE. (SURVIVAL KIT OXYGEN WILL BE FURNISHED AUTOMATICALLY UPON EJECTION.)
5. TIGHTEN CHIN STRAP OF HELMET AND LOWER VISOR.
6. SIT ERECT, BRACE ARMS IN ARMRESTS, HOLD UPPER ARMS AND ELBOWS TIGHTLY AGAINST BODY, HEAD BACK HARD AGAINST HEADREST WITH CHIN TUCKED IN.

WARNING

- UNDER LEVEL FLIGHT CONDITIONS, EJECTION SHOULD BE ACCOMPLISHED ABOVE 2000 FEET WHENEVER POSSIBLE.
- UNDER SPIN OR DIVE CONDITIONS, EJECTION SHOULD BE ACCOMPLISHED ABOVE 10,000 FEET.
- EJECT AT THE LOWEST PRACTICAL AIRSPEED ABOVE 120 KIAS (LOWEST PRACTICAL WOULD BE THAT SPEED BELOW WHICH LEVEL FLIGHT CANNOT BE MAINTAINED).
- DO NOT MANUALLY OPEN THE AUTOMATIC OPENING SAFETY BELT PRIOR TO EJECTION.
- IF POSSIBLE, PRIOR TO EJECTION, THE PILOT SHOULD ATTEMPT TO TURN THE AIRPLANE TOWARD AN AREA WHERE INJURY OR DAMAGE TO PERSONS OR PROPERTY ON THE GROUND OR WATER IS LEAST LIKELY TO OCCUR.
- WHEN EJECTING AT LOW ALTITUDES, (BELOW 2000 FEET), PULL THE NOSE OF THE AIRPLANE ABOVE THE HORIZON, IF AT ALL POSSIBLE, AND USE EXCESS SPEED TO GAIN ALTITUDE.

IF CANOPY FAILS TO JETTISON, RELEASE CANOPY AS FOLLOWS

1. CANOPY JETTISON HANDLE—RAISE.

IF CANOPY IS STILL ON

1. MASTER SWITCH—OFF (SOME AIRPLANES); TRIP, THEN OFF (OTHER AIRPLANES).
2. CANOPY LATCH HANDLE—PULL FULL OUT.
3. PUSH CANOPY INTO AIRSTREAM.
4. MASTER SWITCH—ON AT PILOT'S DISCRETION.

WARNING

IF THE CANOPY CATAPULT FAILS TO FIRE, THE EJECTION SEAT WILL NOT EJECT ON AIRPLANES AF 55-3427 THRU 56-1429 UNLESS MODIFIED BY T.O. 1F-102A-565.

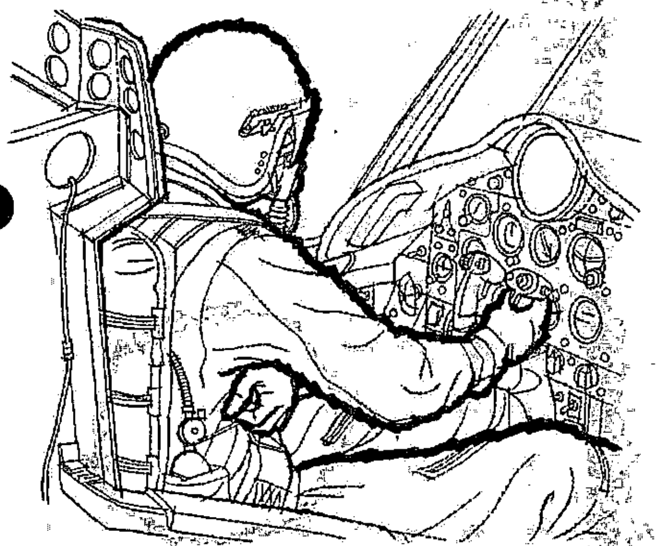
AFTER EJECTION

1. AFTER EJECTION, ATTEMPT TO "BEAT" THE AUTOMATIC SAFETY BELT, THEN IMMEDIATELY PUSH AWAY FROM SEAT WITH A POSITIVE ACTION.
2. IF AUTOMATIC SAFETY BELT FAILS AND SAFETY BELT IS RELEASED MANUALLY, PULL PARACHUTE ARMING LANYARD IF ABOVE 14,000 FEET.
3. MANUALLY PULL RIPCORD HANDLE IMMEDIATELY FOLLOWING SEPARATION FROM SEAT FOR ALL EJECTIONS BELOW 14,000 FEET.
4. IF WEARING A SURVIVAL KIT, AFTER PARACHUTE HAS OPENED AND STABILIZED, RAISE SURVIVAL KIT EMERGENCY RELEASE HANDLE. THIS WILL SUSPEND LIFE RAFT AND SURVIVAL KIT BELOW PILOT AND PREVENT INJURY ON LANDING.

WARNING

- DO NOT RAISE EMERGENCY RELEASE HANDLE UNTIL AFTER PARACHUTE DEPLOYMENT TO PREVENT THE KIT OR LANYARD FROM FOULING THE PARACHUTE.
- DO NOT RAISE EMERGENCY RELEASE HANDLE UNTIL AFTER DESCENT TO AN ALTITUDE NOT REQUIRING OXYGEN. THE OXYGEN SUPPLY WILL BE CUT OFF WHEN THE SURVIVAL KIT IS RELEASED.

4



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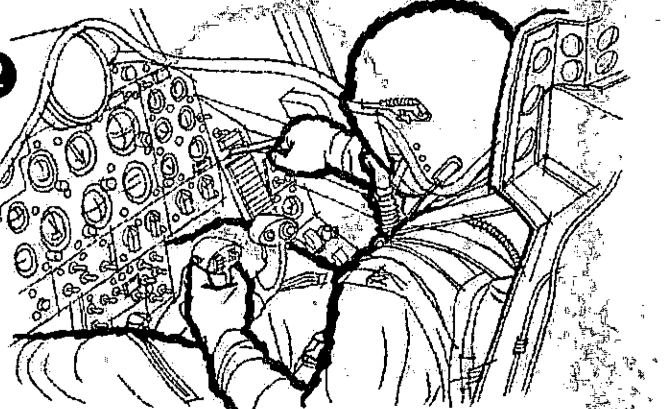

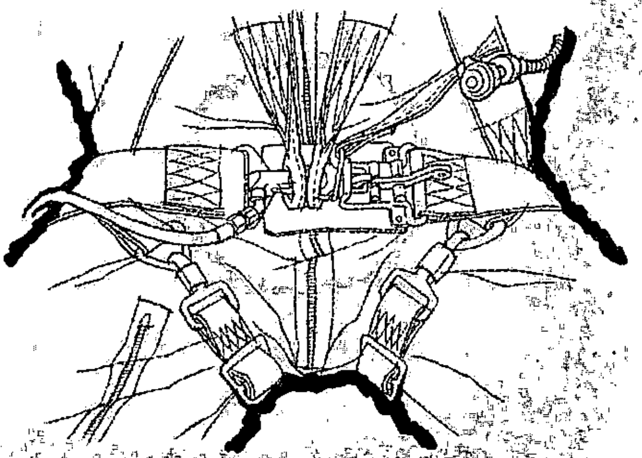

21872-2

are for level flight attitudes and are amply safe for climbs but inclined to be too optimistic for descending flight attitude. In order to use the chart, the style of automatic parachute and type of canopy, pack, and automatic release must first be determined. These are defined in T.O. 14D1-1-1. Once a minimum altitude has been determined for a particular configuration of equipment, the decision whether or not to eject in an emergency should be made in conjunction with the circumstances at hand and not by the fact that the airplane is above or below the minimum altitude as determined from these figures.

WARNING

Emergency minimum ejection altitudes presented on figure 3-4 were determined through extensive flight tests and are based on distance above terrain on initiation of seat ejection (i.e., time seat is fired). These figures do not provide any safety factor for such matters as equipment malfunction, delays in separating from the seat, etc. These figures are quoted only to show the minimum altitude that must be achieved in the event of such low altitude emergencies as fire on takeoff. They shall not be used as the basis for delaying ejection when above 2000 feet since accident statistics show a progressive decrease in successful ejections as altitude decreases below 2000 feet. Therefore, whenever possible, eject above 2000 feet.

See figure 3-4 for ejection procedures. On ejection, the seat and pilot will have a component of thrust provided by the airplane. The automatic opening belt and parachute are timed to provide the most desirable sequence under these conditions and will achieve the desired results faster than manual operation. Therefore, the automatic belt shall not be opened prior to ejection regardless of altitude because of several serious disadvantages, the most important of which are that the automatic opening feature of the parachute is eliminated, and crew member separation from the seat may be too rapid at high speeds.

CAUTION

Immediately after ejection, attempt to manually open the seat belt. This is strictly a precautionary measure in case the belt fails to open automatically. If the belt is operating normally, it will be impossible to "beat" the automatic opening action.

As soon as the safety belt releases, a determined effort must be made to separate from the seat to obtain full parachute deployment at maximum terrain clearance. THIS IS EXTREMELY IMPORTANT FOR LOW-ALTITUDE EJECTIONS.

WARNING

- If the seat belt is opened manually, the automatic feature of the parachute is eliminated. Therefore, under these circumstances, the parachute arming lanyard must be pulled if above 14,000 feet or the ripcord handle must be pulled if below 14,000 feet.
- Manually pull the parachute ripcord handle immediately following seat separation for all ejections below 14,000 feet. This is strictly a precautionary measure since the parachute should deploy automatically.
- Positive seat separation must be achieved prior to pulling the parachute ripcord handle to preclude parachute entanglement with the ejection seat.

Release survival kit after parachute is fully deployed and stabilized, and a safe altitude for breathing without supplemental oxygen is reached.

Note

Normally, the kit may be released as soon as the parachute stabilizes since the automatic opening device will not deploy the chute until a safe breathing altitude is reached. However, if for any reason the parachute deploys at a high altitude, do not release the survival kit until reaching a safe altitude for breathing without supplemental oxygen because releasing the kit cuts off the emergency oxygen supply.

TAKEOFF AND LANDING EMERGENCIES

ABORT

An aborted takeoff, regardless of cause, should be accomplished with landing gear extended. Use the following procedures if a takeoff is aborted prior to becoming airborne.

1. Throttle — OFF.
2. Drag chute handle — Pull.
3. External tanks jettison button — Depress (if required).

   If it is apparent that the airplane cannot be stopped on the runway, the external tanks should be jettisoned immediately.
4. Brakes — As required.
5. Fuel selector switch — OFF (some airplanes). Fuel shutoff valve switches — CLOSE (other airplanes).
6. Master switch — OFF (some airplanes); TRIP, then OFF (other airplanes).
7. Shoulder harness inertia reel handle — LOCKED.

CAUTION

The pilot is prevented from bending forward when the shoulder harness is locked; therefore, all switches not readily accessible should be "cut" before locking the shoulder harness inertia reel.

8. Canopy — Jettison (if necessary).

   Raise the canopy jettison handle on the left-hand armrest if it is necessary to jettison the canopy.

WARNING

If canopy is to be jettisoned, make sure it is jettisoned before the airplane comes to a complete stop. Otherwise, sparks from the canopy remover may cause a fire if fuel is spilled in the vicinity of the airplane, or the canopy may become jammed.

9. Abandon the airplane as soon as possible.

WARNING

The cockpit side rails are approximately eight feet above the ground. Exercise care when abandoning the airplane to prevent bodily injury.

FLAT TIRE DURING TAKEOFF

If a main landing gear tire is blown on takeoff, and sufficient runway is available, abort the takeoff (refer to ABORT, this Section). Deploy the drag chute and use differential braking and nose wheel steering to maintain directional control. As speed decreases, if vibration or shimmy increases, lock the brake on the wheel with the blown tire. If the nose wheel tire has blown out, the brakes should be used for primary directional control and the drag chute should be deployed to reduce landing roll and nose wheel shimmy. Full up elevator should be used to relieve as much pressure as possible on the nose wheel.

WARNING

With a blown tire, directional control is more difficult and braking efficiency is greatly reduced at high gross weights.

If an abort cannot be accomplished because of insufficient runway, continue the takeoff.

WARNING

Do not raise the gear until it has been determined visually from the ground or another airplane that no fire exists, and that further damage will not be incurred by raising the gear.

Landing gross weight should be reduced, if possible, and landing accomplished as directed in FLAT TIRE LANDING, this Section.

EJECTION VS. FORCED LANDING

Normally, ejection is the best course of action with a windmilling or frozen engine, or failure of both the primary and the secondary hydraulic systems. However, because of the many variables encountered, the final decision to attempt a flameout landing or to eject must remain with the pilot. It is impossible to establish a predetermined set of rules and instructions that would provide a ready made decision applicable to all emergencies of

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this nature. The basic conditions listed below, combined with the pilot's analysis of the condition of the airplane, type of emergency, and his proficiency are of prime importance in determining whether to attempt a flameout landing or to eject. These variables make a quick and accurate decision difficult. If the decision is made to eject, prior to ejection, if possible, the pilot should attempt to turn the airplane toward an area where injury or damage to persons or property on the ground or water is least likely to occur. Before a decision is made to attempt a flameout landing, the following basic conditions should exist:

a. Flameout landings should be attempted only by pilots who have satisfactorily completed simulated flameout approaches in this airplane.

b. Flameout landings should be attempted only on a prepared or designated suitable surface.

c. Approaches to the runway should be clear and should not present a problem during flameout approach.

WARNING

No attempt should be made to land a flamed-out airplane at any field where approaches are over heavily populated areas. If possible, prior to ejection, the pilot should attempt to turn the airplane toward an area where injury or damage to persons or property on the ground or water is least likely to occur.

d. Weather and terrain conditions must be favorable. Cloud cover, ceiling, visibility, turbulence, surface wind, etc., must not impede in any manner the establishment of a proper flameout landing pattern.

WARNING

Night flameout landings, or flameout landings under poor lighting conditions, such as at dusk or dawn, should not be contemplated regardless of weather or field lighting.

e. Flameout landings should be attempted only when either a satisfactory "high key" or "low key" position can be achieved.

f. If at any time during the flameout approach, conditions do not appear ideal for successful completion of the landing, ejection should be accomplished. Eject no later than the "low key" altitude.

FORCED LANDING

All landing emergencies involving landing on prepared or unprepared surfaces should be made with the landing gear extended. The extended gear, even on reasonably rough terrain, provides an absorption of the initial shock resulting in less injury to the pilot and damage to the airplane. The inherent nose-high landing attitude of this airplane will result in severe "slap" to the ground if the tail section is permitted to take the initial shock of a wheels-up landing. Whenever the terrain is unknown or unsuited for forced landings, or whenever the landing gear cannot be extended, consideration should be given to the use of the ejection seat. This recommendation is made because of the increasing incidence of vertebral injuries (especially spinal compression fractures) to pilots during forced landings of other high-performance aircraft. If a crash landing is to be made, the canopy should be jettisoned just before touchdown to preclude jamming in the event of fuselage buckling. Whenever the canopy is jettisoned in the normal landing speed range during a landing emergency, no noticeable change in flight characteristics should be experienced and wind blast in the cockpit should be mild. At speeds lower than 175 KIAS the canopy may strike the tail. However actual emergency experiences indicate that airplane control is not affected.

Note

- Salvo firing is the only method of jettisoning the armament load as no mechanical release is provided.
- If it is desired to salvo armament prior to making a forced landing or ditching, the area must be clear for sufficient distance to allow the armament propellant to burn out and the armament to fall to earth in the clear area.
- The helmet visor should be lowered during any emergency landing. The visor provides eye protection from impact or flying objects and from wind blast after the canopy is jettisoned.

The handling characteristics of the airplane during a flameout landing are satisfactory. Immediately after flameout, jettison external tanks and establish and maintain best glide speed of 220 KIAS until sure of reaching the field. Turn off all electrical equipment not essential for flight. The hydraulic systems operate satisfactorily, maintaining adequate pressure and capacity with windmilling engine.

Note

If a frozen engine has resulted, the secondary hydraulic system will not supply pressure to operate the following systems:

- Emergency ac generator
- Speed brakes
- Landing gear retraction

If sufficient hydraulic pressure is available with a windmilling engine the RAT is not extended at the time of flameout, it should be extended at 2500 feet in the forced landing pattern. Gear extension should be accomplished when entering the 10,000-foot high key point.

Note

- In planning the landing when the gear is to be lowered by the emergency system, one minute should be allowed for gear extension time on some airplanes.* On other airplanes** emergency extention should not require more than ten seconds.
- Nose wheel steering is inoperative when gear is extended by emergency system.

If secondary hydraulic pressure is available and the gear is extended using normal extension, the main gear will extend and lock immediately and nose gear will lock in approximately 15 seconds. During the glide approaching pattern with the gear up, rate of descent is approximately 3000 feet per minute. After the gear is extended, rate of descent will range from about 4500 feet per minute in wings-level flight to about 6000 feet per minute in a 45-degree bank. Because of this relatively high rate of descent, the flare should be started approximately 200 feet above the ground. Windmilling engine rpm at touchdown speed of 160 knots is 10 to 12 percent. Brake effectiveness does not noticeably decrease during the landing roll even with excessive use of brakes. At seven to eight percent rpm and 90 to 100 knots during the ground roll, hydraulic capacity becomes too low to recover from control requirements and the stick becomes sluggish and freezes. For the purpose of making simulated forced landings, it has been determined that a power setting of 76 percent rpm with speed brakes fully extended will simulate flameout conditions. Figure 3-5 contains recommended procedures and techniques for a forced landing on suitable terrain.

BELLY LANDING

If forced to make a gear-up landing, proceed as follows:

CAUTION

If conditions permit, salvo the armament.

1. External tanks jettison button — Depress (if required).
   Depress the external tanks jettison button to jettison the external tanks if tanks are installed and contain fuel. If external tanks are empty, it is recommended that they be retained to cushion the impact unless landing is to be made on an unprepared surface.

*AF 53-1791 thru 56-1518 unless modified by TCTO 1F-102-655.
**AF 57-770 & on, & airplanes modified by TCTO 1F-102-655.

2. Make normal approach.
3. **LANDING GEAR HANDLE — DOWN.**
4. Shoulder harness inertia reel handle — LOCKED.

CAUTION

The pilot is prevented from bending forward when the shoulder harness is locked; therefore, all switches not readily accessible must be "cut" before moving the shoulder harness inertia handle to LOCKED position.

5. Immediately before touchdown:
   a. Speed brakes switch — OUT.
      Open speed brakes to allow drag chute deployment.
   b. Throttle — OFF.
   c. Fuel selector switch — OFF (some airplanes). Fuel shutoff valve switches — CLOSE (other airplanes).
   d. Canopy jettison handle — Raise.
      Raise canopy jettison handle on the left armrest to jettison the canopy.

WARNING

Make sure canopy is jettisoned before touchdown. Otherwise, sparks from the canopy remover may cause a fire if fuel is spilled in the vicinity of the airplane, or the canopy may become jammed.

6. Normal landing attitude for touchdown.
7. Drag chute handle — Pull.
8. Master switch — OFF (some airplanes); TRIP, then OFF (other airplanes).
9. Abandon the airplane.
   When the airplane stops, abandon immediately.

ANY ONE GEAR UP OR UNLOCKED

If landing is to be made with partial gear extension, proceed as follows:

Note

If time and conditions permit, request runway be foamed as soon as possible to assist directional control and decrease fire hazard.

1. External tanks jettison button — Depress (if required).
   Depress the external tanks jettison button to jettison the external tanks if tanks are installed and contain fuel. If external tanks are empty, it

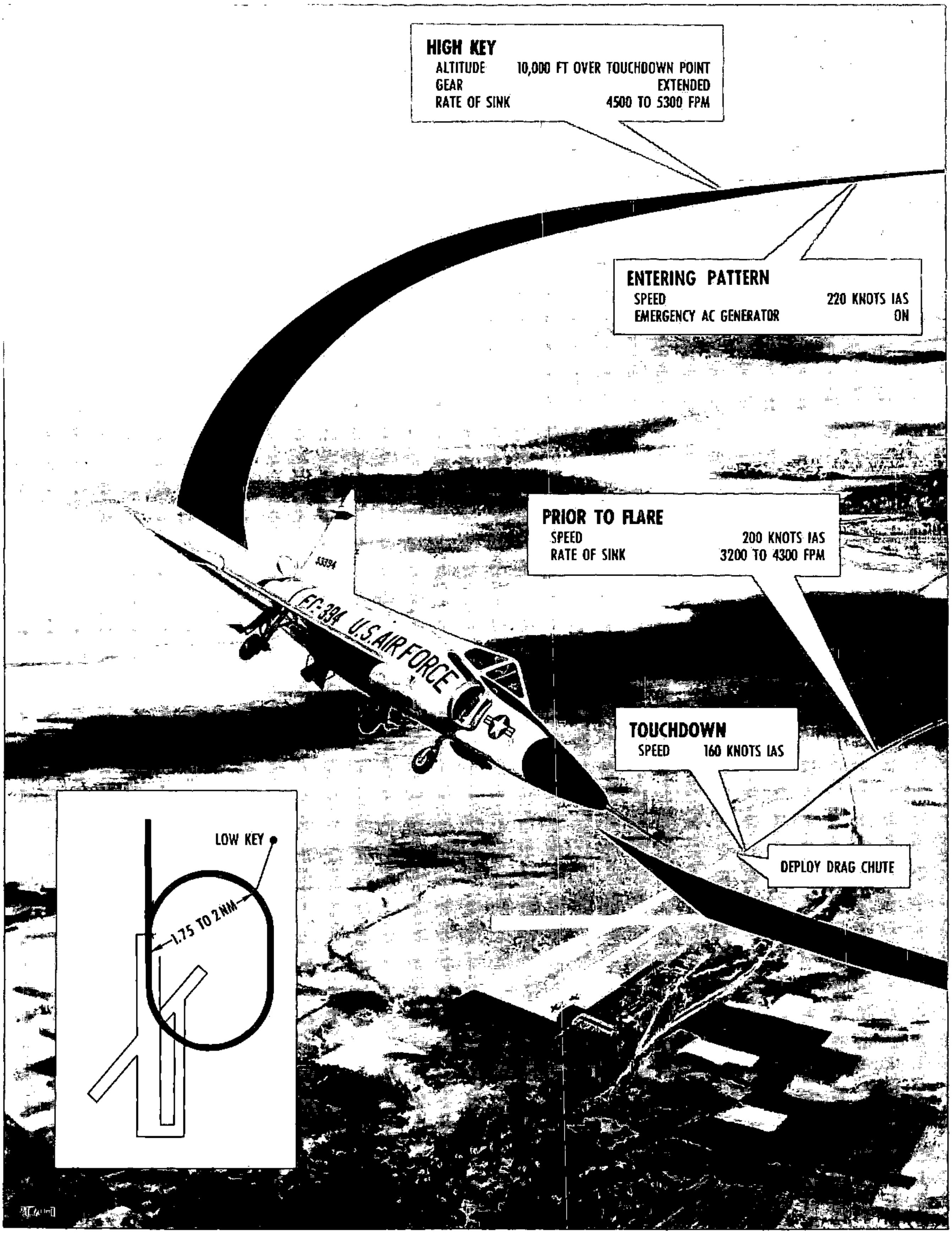
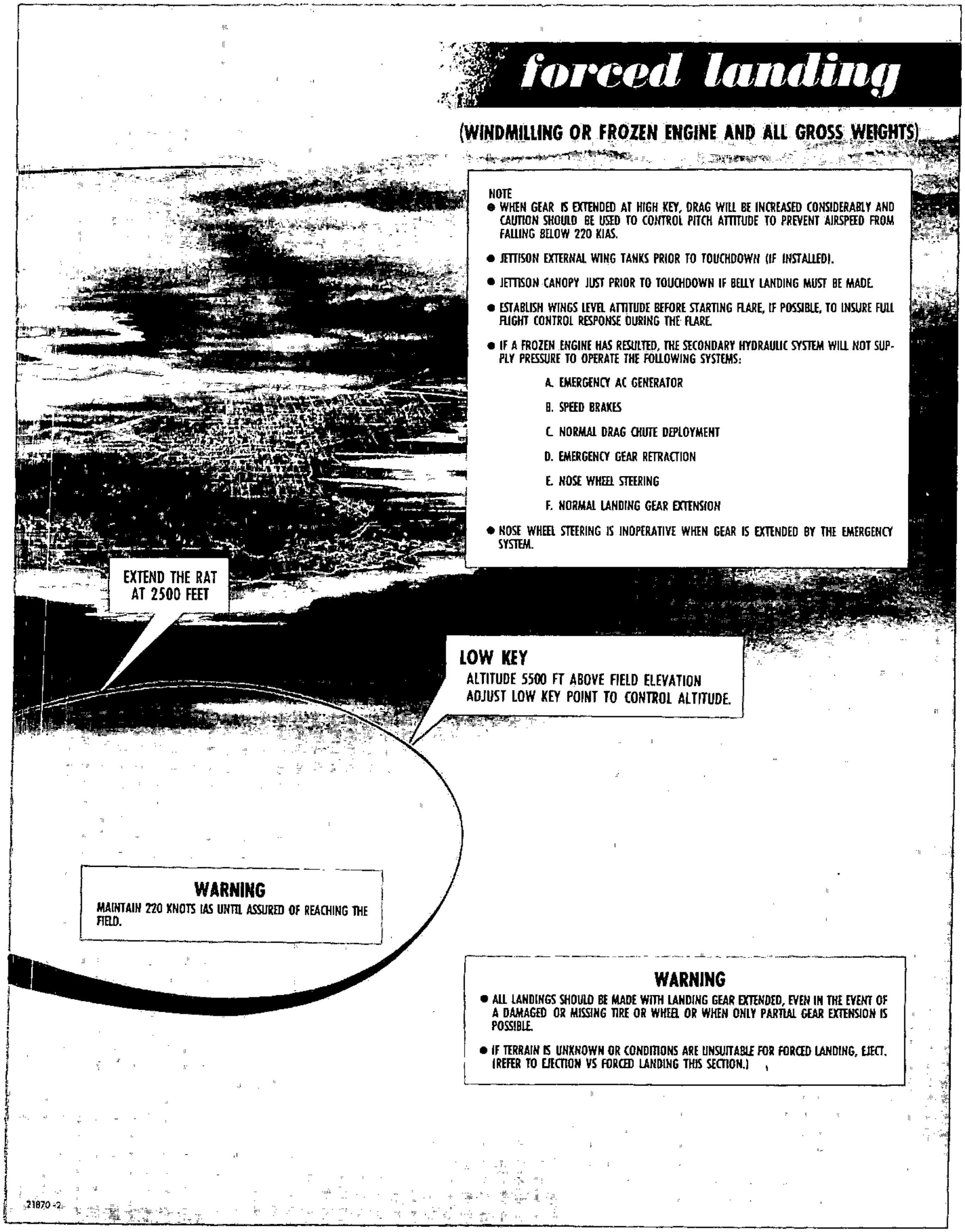

HIGH KEY
ALTITUDE 10,000 FT OVER TOUCHDOWN POINT
GEAR EXTENDED
RATE OF SINK 4500 TO 5300 FPM
ENTERING PATTERN
SPEED 220 KNOTS IAS
EMERGENCY AC GENERATOR ON
PRIOR TO FLARE
SPEED 200 KNOTS IAS
RATE OF SINK 3200 TO 4300 FPM
TOUCHDOWN
SPEED 160 KNOTS IAS
DEPLOY DRAG CHUTE
LOW KEY
1.75 TO 2 NM
FC-394 U.S. AIR FORCE

Figure 3-5


forced landing
(WINDMILLING OR FROZEN ENGINE AND ALL GROSS WEIGHTS)
NOTE
● WHEN GEAR IS EXTENDED AT HIGH KEY, DRAG WILL BE INCREASED CONSIDERABLY AND CAUTION SHOULD BE USED TO CONTROL PITCH ATTITUDE TO PREVENT AIRSPEED FROM FALLING BELOW 220 KIAS.
● JETTISON EXTERNAL WING TANKS PRIOR TO TOUCHDOWN (IF INSTALLED).
● JETTISON CANOPY JUST PRIOR TO TOUCHDOWN IF BELLY LANDING MUST BE MADE.
● ESTABLISH WINGS LEVEL ATTITUDE BEFORE STARTING FLARE, IF POSSIBLE, TO INSURE FULL FLIGHT CONTROL RESPONSE DURING THE FLARE.
● IF A FROZEN ENGINE HAS RESULTED, THE SECONDARY HYDRAULIC SYSTEM WILL NOT SUPPLY PRESSURE TO OPERATE THE FOLLOWING SYSTEMS:
A. EMERGENCY AC GENERATOR
B. SPEED BRAKES
C. NORMAL DRAG CHUTE DEPLOYMENT
D. EMERGENCY GEAR RETRACTION
E. NOSE WHEEL STEERING
F. NORMAL LANDING GEAR EXTENSION
● NOSE WHEEL STEERING IS INOPERATIVE WHEN GEAR IS EXTENDED BY THE EMERGENCY SYSTEM.
EXTEND THE RAT AT 2500 FEET
LOW KEY
ALTITUDE 5500 FT ABOVE FIELD ELEVATION
ADJUST LOW KEY POINT TO CONTROL ALTITUDE.
WARNING
MAINTAIN 220 KNOTS IAS UNTIL ASSURED OF REACHING THE FIELD.
WARNING
● ALL LANDINGS SHOULD BE MADE WITH LANDING GEAR EXTENDED, EVEN IN THE EVENT OF A DAMAGED OR MISSING TIRE OR WHEEL OR WHEN ONLY PARTIAL GEAR EXTENSION IS POSSIBLE.
● IF TERRAIN IS UNKNOWN OR CONDITIONS ARE UNSUITABLE FOR FORCED LANDING, EJECT. (REFER TO EJECTION VS FORCED LANDING THIS SECTION.)
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is recommended that they be retained to cushion the impact unless landing is to be made on an unprepared surface.

2. Plan normal landing.

   Plan normal approach and touchdown and provide maximum distance for ground roll.

3. Shoulder harness inertia reel handle — LOCKED.

CAUTION

The pilot is prevented from bending forward when the shoulder harness is locked; therefore, all switches not readily accessible should be "cut" before moving the shoulder harness inertia reel handle to LOCKED position.

4. Immediately before touchdown:
   a. Speed brakes switch – OUT.

      Open the speed brakes to allow drag chute deployment.
   b. Throttle — OFF.
   c. Fuel selector switch — OFF (some airplanes). Fuel shutoff valve switches — CLOSE (other airplanes).
   d. Canopy jettison handle – Raise.

      Raise canopy jettison handle on the left armrest to jettison the canopy.

WARNING

Make sure canopy is jettisoned before touchdown. Otherwise, sparks from the canopy remover may cause a fire if fuel is spilled in the vicinity of the airplane, or the canopy may become jammed.

5. After touchdown:
   a. Drag chute handle – Pull.
   b. Master switch – OFF (some airplanes); TRIP, then OFF (other airplanes).
   c. Hold faulty gear off the ground.

      Hold the up or unlocked gear off the ground; then before flight controls become ineffective, ease the faulty gear down.
   d. Braking technique – As required.

      If the faulty gear is the nose gear and does not collapse upon touchdown, do not use brakes if a safe stop can be made without them.

6. Abandon the airplane.

   When the airplane stops, abandon immediately.

FLAT TIRE LANDING

Use normal landing pattern if blown tire is known prior to landing. If either main landing gear tire blows out, differential braking and nose wheel steering should be used to maintain directional control. Lower the nose wheel to the runway and deploy the drag chute as soon as possible. As speed decreases, if vibration or shimmy increases, lock the brake on the wheel with the blown tire. If the nose wheel tire blows out on landing, the brakes should be used for primary directional control and the drag chute deployed to reduce landing roll and nose wheel shimmy. Because of the extreme shimmy of the nose wheel with a blown tire, nose wheel steering may be uncontrollable. When a nose wheel tire blows out, trim full up elevator to relieve as much pressure as possible on the nose wheel.

CAUTION

- If it is known before landing that a main landing gear tire is flat, the airplane should be landed on the left side of the runway when the right main landing gear tire is flat or on the right side of the runway when the left main landing gear tire is flat. The airplane will veer in the direction of the flat tire and complete directional control may not be possible.
- When it is known before landing that the nose wheel tire is flat, lower the nose wheel to the runway at approximately 110 knots to permit making a controlled nose wheel touchdown.

LANDING WITH ARMAMENT BAY DOORS OPEN

If it becomes necessary to land with the armament bay doors open due to an in-flight malfunction, the door-open configuration will create a considerable increase in drag. With the exception of the increased drag, no unusual flight characteristics will be evident. Additional thrust will be required around the pattern to maintain desired airspeeds. Do not increase pattern airspeeds for landing with armament doors open.

RUNWAY OVERRUN BARRIER ENGAGEMENT

The runway overrun barrier provides an effective means of stopping the airplane on the runway after an aborted takeoff run or in an emergency landing situation. External tanks may foul the arresting cable and therefore should be jettisoned prior to engagement. On some airplanes, when entering the barrier, there is a possibility of the nose wheel overriding the webbing at any engagement speed. The chances of successful engagement at low speed are also reduced because of the long distance between the nose wheel and the main gear. On later airplanes a probe has been added on the nose landing gear and a deflector over the taxi light to insure that the nose gear engages the webbing. Also, some airplanes* have an extendible barrier probe mounted on the centerline of the airplane just aft of the missile bay doors. This probe is extended simultaneously with the actuation of the drag chute and serves as a guide to insure that the arresting cable engages the main landing gear. However, the probability of engagement increases as speed increases. Below 60 knots ground speed the airplane cannot be expected to engage the barrier and between 60 and 70 knots ground speed engagement is marginal. Above 70 knots ground speed the airplane can be expected to engage the barrier if the webbing is not overridden. During successful engagement, structural failure of the gear is not likely, and the deceleration forces are negligible. The airplane should enter the barrier at a 90 degree angle, although some deviation will not preclude successful engagement. Directional control of the airplane is normally not difficult to maintain if both main landing gear are engaged. Just prior to engagement, the throttle should be OFF and nose wheel steering engaged.

CAUTION

Above 80 knots ground speed, nose wheel steering becomes sensitive and directional control is very difficult to maintain.

If only one main landing gear is engaged, the airplane will veer in the direction of the engaged gear. This tendency increases as the engagement speed increases. Directional control should be maintained with nose wheel steering and differential braking. Brakes should not be applied as the barrier is entered because braking will lower the nose of the airplane and the pitot boom will possibly engage the webbing.

EMERGENCY ENTRANCE

The procedure to be used by the rescue personnel when assisting a disabled pilot from the airplane following a crash landing is contained in figure 3-6.

DITCHING

Do not attempt to ditch this airplane except as a last resort. Ditching should be attempted only when altitude is insufficient for a successful bailout or in the event the ejection seat fails to function. (refer to EJECTION, this Section.) This recommendation is made because of the danger of vertebral injury (especially spinal compression fracture) to the pilot during a water landing. The helmet visor should be lowered prior to ditching. The visor provides eye protection from impact or flying objects and from wind blast after the canopy is jettisoned. If ditching is unavoidable, proceed as follows:

1. Follow radio distress procedure.
2. External fuel tanks button – Depress.
3. RAT handle – Pull (if necessary) & hold for four seconds.

   Extend the RAT by pulling the RAT handle fully out. Hold the handle in the fully extended position for four seconds to insure a satisfactory extension of the turbine.
4. Oxygen diluter lever – 100% OXYGEN.
5. **LANDING GEAR HANDLE — UP.**
6. Shoulder harness inertia reel handle – LOCKED.
7. Master switch – OFF (some airplanes); TRIP, then OFF (other airplanes).
8. Canopy jettison handle – Raise.

   Raise canopy jettison handle on the left armrest to jettison the canopy just before touchdown.
9. Normal approach and touchdown.

   Land parallel to swells unless the wind is in excess of 25 knots, in which case it is recommended to land into the wind, touching down on the falling side of the wave, if possible. Maintain nose-high attitude and touch down as slowly as possible with a small rate of descent. As the nose of the airplane settles into the water, it may tend to flip over due to the intake ducts acting as water scoops.
10. Throttle – OFF at touchdown.
11. Abandon the airplane as soon as forward motion stops. Upon contact with the water, the airplane will usually bounce, during which time the sensation of being stopped may be experienced. Do not unfasten the seat belt until deceleration forces have stopped, water spray is noticeable, and water begins to enter the cockpit. When leaving the airplane retain the survival kit if possible (if survival kit is installed).

Note

The diluter-demand type oxygen regulator is a suitable underwater breathing device when the regulator is set at 100% OXYGEN. If for some reason an immediate escape is not possible this equipment may be used for underwater breathing provided all connections are tight. The bailout bottle can be used under water only on airplanes equipped with a survival kit. On these airplanes, pulling the "green knob" will supply oxygen for approximately 12 minutes.

AFTERBURNER FAILURE

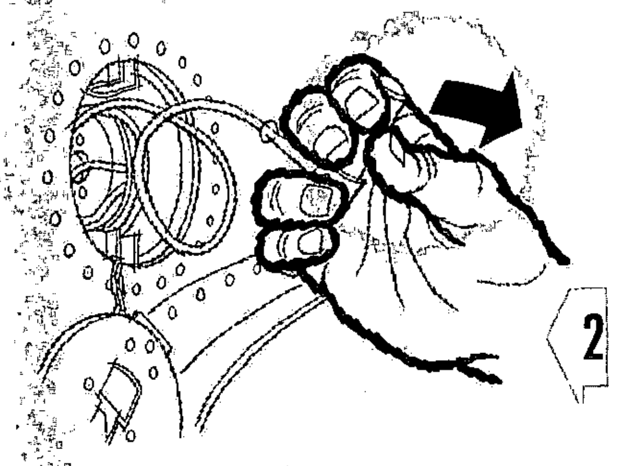
AFTERBURNER FAILURE DURING TAKEOFF

Afterburner failure may be noted by a sudden loss of acceleration.

1. If sufficient runway remains to make a safe stop or insufficient runway available to continue takeoff at military thrust, abort takeoff; refer to ABORT, this Section.

**AF 57-770 & on, & airplanes modified by TCTO 1F-102-658.*

REMOVE ACCESS DOOR. 1



2

GRASP "T" HANDLE AND PULL OUTBOARD APPROXIMATELY 6 FEET TO JETTISON CANOPY.

NOTE

CANOPY WILL BE JETTISONED BY SAME SYSTEM AS INSTALLED FOR CANOPY JETTISON BY PILOT. CANOPY WILL TRAVEL UP AND AFT AND WILL PROBABLY STRIKE THE TAIL. RESCUE PERSONNEL SHOULD WATCH PATH OF CANOPY TO AVOID BEING HIT.

WARNING

IF THE CANOPY HAS NOT BEEN RELEASED AND IF SPILLED FUEL IS IN THE VICINITY OF THE AIRPLANE, A FIRE MAY RESULT FROM A HOT POWDER SPARK WHEN THE CANOPY IS JETTISONED.

IF CANOPY FAILS TO JETTISON OR IF PRESENCE OF FUEL FUMES MAKES JETTISON INADVISABLE, GAIN ENTRANCE EITHER BY:

a. OPENING THE LATCH ACCESS DOOR LOCATED BELOW THE RIGHT-HAND WINDSHIELD AND PUSHING THE EXTERNAL CANOPY RELEASE HANDLE TO THE REAR TO UNLATCH THE CANOPY. AN EXTERNAL GRIP ON THE FORWARD LEFT-HAND SIDE OF THE CANOPY IS PROVIDED TO ASSIST IN RAISING THE CANOPY.

b. BREAKING CANOPY GLASS IF EXPEDIENCY OF ENTRY IS OF MAJOR IMPORTANCE. CANOPY GLASS SHOULD BE BROKEN IN THE LOWER FORWARD CORNER ON EITHER SIDE. IT IS ESSENTIAL TO STRIKE HARD BLOWS AS NEAR THE EDGE OF CANOPY FRAME AS POSSIBLE.

3

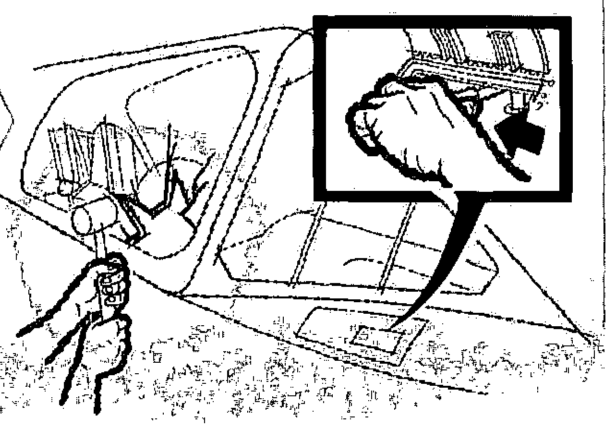


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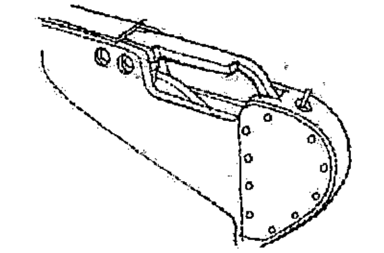
Figure 3-6

emergency entrance

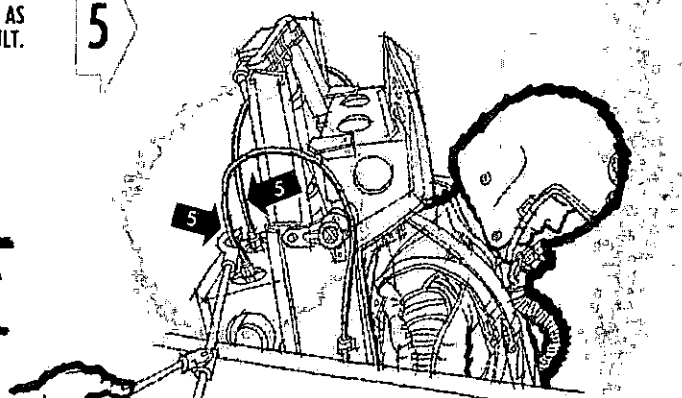
4 WHEN ACCESS TO COCKPIT IS GAINED, CHECK POSITION OF EJECTION SEAT HANDGRIPS.

NOTE

IF PILOT JETTISONED CANOPY IN PREPARATION FOR CRASH LANDING, THE CANOPY JETTISON HANDLE SHOULD HAVE BEEN USED. HOWEVER, RAISING EITHER HANDGRIP WILL ALSO JETTISON THE CANOPY. IF THE HANDGRIPS ARE RAISED, SEAT CATAPULT TRIGGERS ARE EXPOSED. SUBSEQUENT MOVEMENT OF EITHER TRIGGER WILL FIRE THE CATAPULT AND EJECT THE SEAT FROM THE AIRPLANE.

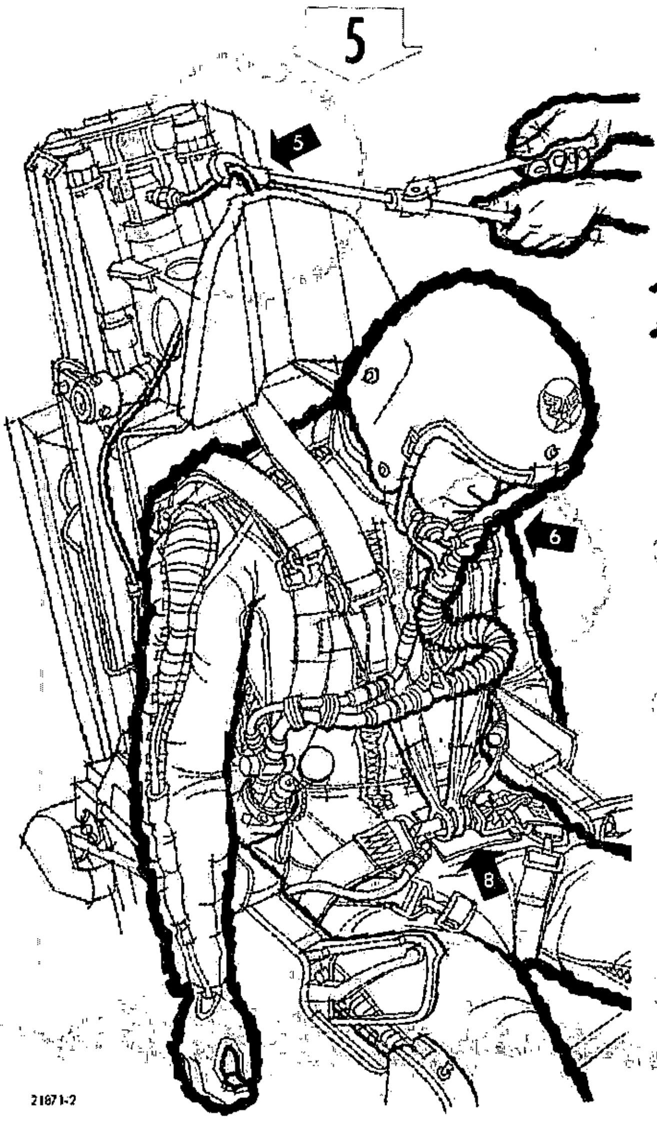


5 DISARM SEAT CATAPULT BY CUTTING WITH HEAVY DUTY CABLE CUTTERS, AS SHOWN, OR DISCONNECTING HOSE LEAD TO FITTING AT TOP OF SEAT CATAPULT.



NOTE

ON AIRPLANES WITH ROCKET CATAPULT INSTALLED, DISARM THE SEAT BY PUSHING THE QUICK DISCONNECT FITTINGS, OR BY CUTTING WITH HEAVY DUTY CABLE CUTTERS AS SHOWN ABOVE.



6 IF PILOT IS WEARING A PARTIAL PRESSURE SUIT AND HELMET, THE HELMET FACE PLATE MUST BE REMOVED PRIOR TO RELIEVING ANY PRESSURE TO THE SUIT. IF THE SUIT PRESSURE IS RELIEVED FIRST, IT IS POSSIBLE TO RUPTURE THE PILOT'S LUNGS BY FORCING AIR UNDER PRESSURE FROM THE STILL-PRESSURIZED HELMET. THE HELMET FACEPLATE MAY BE REMOVED BY PULLING DOWNWARD ON THE GREEN CORD BENEATH CHIN AND LIFTING FACEPLATE FREE.

7 PULL YELLOW SURVIVAL KIT "EMERGENCY RELEASE" HANDLE TO DISCONNECT THE HARNESS AND PERSONAL LEADS BUNDLE.

NOTE

- ON SOME AIRPLANES, THE RUBBER LIFE RAFT MAY INFLATE (IF INSTALLED). THE RAFT SHOULD BE PUNCTURED IF IT HINDERS REMOVAL OF THE PILOT.
- THE "HARNESS RELEASE" LEVER, AFT OF THE "EMERGENCY RELEASE" HANDLE, MAY ALSO BE USED TO DISCONNECT THE SURVIVAL KIT PARACHUTE ATTACHING STRAPS. HOWEVER, THE PERSONAL LEADS MUST THEN BE DISCONNECTED MANUALLY.

8 RELEASE SEAT BELT AND SHOULDER HARNESS.

21871-2

2. If there is insufficient runway available to make a safe stop and takeoff can be safely continued at military thrust:
   a. **THROTTLE — INBOARD TO FULL MIL POWER.**

CAUTION

If the takeoff is to be continued with an afterburner blowout, the throttle must be moved out of the AFTERBURNER position immediately to enable the afterburner nozzle to close. Non-afterburner engine operation with the nozzle in the open position will result in an appreciable loss of thrust.

AFTERBURNER EXHAUST NOZZLE FAILURE DURING TAKEOFF

If the afterburner exhaust nozzle fails to open at the time of afterburner ignition, a rapid rise in exhaust gas temperature and a reduction of approximately four percent rpm will be noted. Partial failure of the exhaust nozzle to open may not be noted on the engine instruments. However, failure of several adjacent nozzle flaps when the afterburner ignites during takeoff may result in a side force which may cause a sudden lurch, or tendency to change direction of the airplane.

WARNING

If several adjacent nozzle flaps have failed and takeoff is continued, the side force may be so great that there will be insufficient rudder to control the airplane immediately after leaving the ground.

1. If exhaust gas temperature rises rapidly and rpm drops or if any directional change is noted when the afterburner is ignited during the takeoff roll:
   a. **THROTTLE — OFF.**
   b. Drag chute handle — Pull.
   c. Brakes — As required.

AFTERBURNER FAILURE DURING FLIGHT

1. Throttle – Inboard and hold five seconds.

   Move the throttle inboard to shut off fuel to the afterburner. Wait five seconds to clear afterburner.
2. **THROTTLE — AFTERBURNER TO RE-IGNITE AFTERBURNER (IF DESIRED).**
3. If afterburner fails to light – Throttle inboard.

WARNING

Do not make a second attempt to relight the afterburner if cause of afterburner failure is unknown, to prevent possibility of fire or explosion.

AFTERBURNER CUTOFF FAILURE

If the afterburner cannot be cut off through use of normal throttle motion, retard the throttle smartly to some point below approximately 90% rpm to mechanically terminate afterburner; then advance slowly to the minimum thrust which will sustain flight and land as soon as practicable.

CAUTION

- Do not pause at the afterburner actuation point when retarding the throttle; move the throttle smartly through this range to insure that the exhaust nozzle will close.
- During operation above 40,000 feet, avoid rapid throttle movement and do not reduce the throttle below 85% rpm to reduce the probability of compressor stalls.

Note

Selection of thrust just below the afterburner range is important for fuel saving considerations; however, afterburning is available in the case of a "go-around" or other critical flight condition by advancing the throttle smartly above approximately 90% rpm.

ENGINE OIL SYSTEM FAILURE

Failure of the engine oil system to supply sufficient operating pressure is indicated by the oil pressure-low warning light (master warning system). Illumination of this light does not necessarily mean that the airplane should be abandoned immediately. In most instances, the engine can be operated at reduced power for several minutes before ultimate engine failure. In the event of an oil pressure low indication, the following procedure should be followed:

1. **THROTTLE — RETARD (MINIMUM NECESSARY FOR FLIGHT).**

Note

Once the throttle is retarded to minimum necessary for flight, additional throttle movement could result in the engine freezing. Subsequent throttle movement should be used only in the interest of safety.

2. **AIRSPEED — 345 KIAS OR BELOW.**

   Airspeed should be reduced below the maximum RAT speed in the event the engine freezes and it is necessary to extend the RAT.

3. Avoid high g maneuvers.
4. **LAND AS SOON AS POSSIBLE. A FLAMEOUT PATTERN CAN BE USED IF THIS IS THE MOST EXPEDITIOUS MEANS OF GETTING ON THE RUNWAY.**

   Landing should be made on the nearest suitable runway, and the engine shut down immediately after taxiing off active runway.

5. If engine vibrations become excessive during flight, shut down engine.

   Complete engine failure will normally be indicated by a steadily increasing vibration. At this indication, the engine should be shut down to preclude such a destructive failure as to jeopardize a successful ejection or a forced landing.

Note

Maneuvers at less than one g or at negative g may cause the oil pressure-low warning light to illuminate during flight. This is permissible provided the duration of the maneuver does not exceed limitations specified in PROHIBITED MANEUVERS, Section V.

WARNING

- If engine oil system fails, expect ac and dc generator failure.
- Engine failure resulting from insufficient oil pressure may result in a frozen engine. As soon as the engine fails or is shut down, the RAT should be extended to supply flight control hydraulic pressure.

FUEL SYSTEM FAILURE

ENGINE STAGE FUEL PUMP FAILURE DURING TAKEOFF

In the event of failure of the engine stage fuel pump, as indicated by the engine fuel pump failure warning light (master warning system) the takeoff should be aborted if a safe stop can be made. If committed to takeoff, the afterburner stage fuel pump will provide all the fuel required by the engine and part of the fuel required by the afterburner; therefore, partial thrust will be obtained from continued afterburner operation.

ENGINE FUEL CONTROL FAILURE

Engine fuel control failure may be recognized by a drop in exhaust gas temperature, rpm, and fuel flow. As in any takeoff emergency, the takeoff should be aborted if conditions permit (runway length, overrun condition, barrier availability, etc.). If, however, the takeoff has progressed to the point that an abort is not feasible, use the following procedure:

1. Throttle — As required.

   Attempt to match throttle position to engine rpm.

WARNING

Compressor stall or engine flameout may occur due to introduction of excess fuel if the throttle position does not approximately correspond to engine rpm prior to selection of the emergency fuel system.

2. **FUEL CONTROL SWITCH — EMERGENCY.**

WARNING

- Avoid rapid throttle movements.
- Following an inflight normal fuel system failure, do not return the fuel control switch to NORMAL for the duration of the flight or flameout will result.

3. Control engine speed as necessary.
4. Land as soon as possible.

ENGINE STAGE FUEL PUMP FAILURE IN FLIGHT

In the event of failure of the engine stage fuel pump as indicated by the engine fuel pump failure warning light (master warning system), the afterburner stage fuel pump will supply fuel to the engine and afterburner operation should not be attempted.

AFTERBURNER FUEL PUMP FAILURE DURING TAKEOFF

In the event of failure of the afterburner stage fuel pump, afterburner failure will occur. The engine stage fuel pump will not supply fuel to the afterburner. Refer to AFTERBURNER FAILURE DURING TAKEOFF, this Section.

FUEL BOOST PUMP FAILURE

In the event complete boost pump failure is experienced during takeoff but tank pressurization can be maintained, maximum engine and afterburner thrust can be sustained at any altitude from sea level to 36,000 feet with any fuel temperature up to 100°F in the tanks at takeoff. With complete fuel boost pump failure, observe the following:

1. Avoid negative g maneuvers.
2. Under low fuel quantity conditions (low level warning light illuminated):
   a. Operate in a nose-high attitude (normal traffic pattern attitude is nose high).
   b. Avoid uncoordinated maneuvers.
   c. Avoid rapid decelerations (combinations of large thrust changes, speed brakes, and landing gear extensions).
3. Land as soon as practicable.

The above steps are to prevent uncovering the aft fuel intakes. A complete fuel boost pump failure will occur with loss of ac power.

FUEL BOOST PRESSURE-LOW WARNING

Fuel boost pressure L or R warning light will illuminate under the following conditions:

- An empty No. 3 tank.
- A dual failure of boost pumps on one side.
- Fuel shutoff valve not opened. (An electrical interlock makes boost pump operation dependent upon a fully open shutoff valve.)

If the fuel boost pressure-low warning light illuminates, in flight, proceed as follows:

1. Fuel selector switch — ENGINE (some airplanes).
   Check that fuel selector switch is selected to ENGINE position.
   Fuel shutoff valve switches — OPEN (other airplanes).
2. **PLAN THE FLIGHT FOR ONE-HALF THE INDICATED FUEL BEING AVAILABLE.**
   Initial action should be based on the assumption that only one-half of the total fuel on board the airplane at the time of fuel boost pump pressure-low warning will be available, if the warning is due to a fuel valve which is not fully open.
3. Verify quantity of remaining fuel in the No. 3 tank on the side which has indicated the warning.
   a. If the No. 3 tank is empty, shut off the empty tank by placing fuel selector switch to the other side, or, on some airplanes, by moving the appropriate fuel shutoff valve switch to CLOSE.

WARNING

If operation with a known empty tank is required, the fuel shutoff valve on the empty side should be closed. This is a precautionary measure to prevent air from entering the system in the event of a pump system malfunction on the side delivering fuel. On airplanes which do not incorporate the boost pump/shutoff valve interlock system, the boost pump switches should be turned off on the empty side.

   b. If the No. 3 tank indicates fuel remaining:
      (1) Fuel selector switch — Leave at ENGINE (some airplanes).
      Fuel shutoff valve switches — OPEN (other airplanes).
      (2) Failed boost pump switches — OFF.
      (3) **LAND AS SOON AS PRACTICABLE.**

Note

If continued flight is necessary, it may be possible to use the indicated fuel remaining. Turn off boost pumps on the good side and monitor remaining fuel carefully to determine if fuel is feeding from the failed side. If it is, conduct flight in accordance with complete FUEL BOOST PUMP FAILURE paragraph above. Prior to penetration, turn good boost pumps ON. (See system description, Section I.)

FUEL FLOW EQUALIZER FAILURE

500# DIFFERENCE

Asymmetrical fuel flow can result from unequal fuel feeding through the fuel flow equalizer or from fuel flow equalizer failure. Minor unbalances from this source can be expected owing to allowable tolerances. Should extreme unbalanced fuel loading become apparent (500-lb. differential or greater), it is desirable to equalize the fuel load. If the unbalanced flow is allowed to continue without corrective action the airplane will exhaust the fuel from one side first. If an empty tank condition exists, refer to FUEL BOOST PRESSURE-LOW WARNING, this Section. To preclude the above, fuel loading may be balanced as follows:

1. Fuel selector switch — ENGINE (some airplanes).
   Fuel shutoff valve switches — OPEN (other airplanes).
2. Fuel quantity — Check (select each position on the fuel quantity selector switch).
   In the event the fuel low warning light illuminates on one side only, or abnormal wing heaviness is noted, the fuel quantity should be verified by the fuel quantity gage.
3. Boost pump switches — OFF on the low side. This will permit high rates of flow from the high side.
4. Boost pump switches — ON on the high side.
5. Fuel quantity gage selector switch — Monitor both low and high sides to insure that the low side has stopped feeding and the high side is feeding.

6. When fuel load is equalized, or when fuel low level warning light on the high side illuminates (less than approximately 570 lbs. of fuel remaining), proceed as follows:
   a. All boost pump switches — ON.
   b. Fuel quantity gage selector switch — TOTAL.

WARNING

If penetration or descent below 10,000 feet is initiated prior to correcting asymmetrical fuel flow, turn all boost pump switches ON.

FUEL TANK PRESSURIZATION FAILURE

If fuel tank pressurization fails, as indicated by illumination of either fuel tank pressure-low warning light, normal fuel transfer cannot be expected. Some fuel may transfer from the No. 1 and No. 2 tanks into No. 3 but the quantity will depend on flight attitude and altitude. Normally the only fuel which will feed to the engine will be that in No. 3. On airplanes which have No. 3 tank placarded on the fuel quantity gage selector switch, it is possible to read the amount of fuel remaining in the No. 3 tank on each side. On these airplanes, it is possible to monitor the No. 3 tanks and determine if fuel is feeding from the other two tanks without pressurization. The flight should be adjusted accordingly in event that only No. 3 tank is feeding the engine. On airplanes which have positions of L and R placarded on the fuel quantity gage selector switch, the fuel quantity gage indicates total fuel in either wing and it is not possible to determine if fuel is feeding from No. 1 and No. 2 tanks into No. 3. On these airplanes, if pressure fails, it is necessary to assume that only No. 3 tank fuel is available. Momentary illumination of the fuel tank pressure-low warning light will occur during rapid descent or high g acceleration. If the warning light illuminates steadily, proceed as follows:

1. Land as soon as practicable.
2. **DEPEND ON NO. 3 TANK FUEL ONLY.**
3. Monitor low level warning lights and fuel quantity in each wing.
4. Do not use afterburner.

ELECTRICAL POWER SYSTEM FAILURE

Note

- The airplane can be returned to base with complete electrical failure and a safe landing accomplished under visual flight conditions. When operating from the battery, all equipment not needed to maintain flight should be turned off. Usable battery power for continuous operation is approximately five to 20 minutes.
- When operating from battery power only, the UHF will not function after approximately five minutes due to lowering battery power. When the UHF stops functioning, it should be shut off to conserve battery power.

If ac and dc generators fail, the emergency ac generator will supply power to the ac essential bus and the battery will supply power to the dc essential bus. Since ac electrical power control relays which connect the ac generator to the ac bus are energized from the dc essential bus, complete electrical power failure will occur after battery power depletion (approximately five to 20 minutes). On airplanes with self-sustaining ac system*, a dc generator failure followed by an ac generator failure would also result in complete electrical power failure (except battery power). However, by momentarily placing the ac bus switch in the RESET position the emergency ac generator will start and supply power to the ac essential bus and the transformer-rectifier will again supply power to the emergency dc bus. When the ac bus switch is released, it returns to the ON position tying the ac emergency disconnect relay and emergency generator shutoff valve to the emergency dc bus.

WARNING

To conserve battery power, turn battery OFF if not required.

Should the transformer-rectifier fail after dc generator failure the battery may be used to power the emergency dc bus. Actuation of the battery switch to the TR FAIL position, ties the emergency dc bus to the dc essential bus (battery). Power will be supplied to the dc essential functions until battery power is depleted (five to 20 minutes).

AC GENERATOR FAILURE

If the main ac generator fails, the ac power failure warning light will illuminate. In the event of a failure, use the following procedure:

1. AC bus switch — EMER (some airplanes); RESET, then ON (other airplanes).
2. Radar master switch — OFF.
3. Boost pump switches — OFF.
4. Nesa switch — OFF.
5. Rain clear switch—RAINCLEAR (some airplanes); STDBY (other airplanes).
6. AC generator switch — RESET, then ON.

Note

If generator will not reset, check circuit breakers.

**Airplanes modified by TCTO 1F-102-727.*

If warning light extinguishes:

1. AC bus switch – NOR.
2. Boost pump switches – ON (one at a time).
3. Nesa switch – NORMAL (some airplanes); ON (other airplanes).
4. Radar master switch – As required.

If warning light remains on:

1. AC generator switch – OFF.
2. **LAND AS SOON AS PRACTICABLE.**

Note

In the event of main ac generator failure, the following primary flight aids, in addition to the pitot-static instruments, will be operable:

- All gyro flight instruments
- Gyro compass
- Interphone, UHF command, VOR, and marker beacon radio
- IFF radar

Refer to Section I for detailed list of electrically operated equipment.

DC GENERATOR FAILURE

If the dc generator fails, the dc power failure warning light will illuminate. In the event of failure, use the following procedure:

1. DC generator switch – RESET, then ON.

Note

If generator will not reset, check circuit breakers.

If warning light remains on:

1. DC generator switch – OFF.
2. Turn off all equipment not needed.
3. Battery switch – ON.

Note

- Airplane trim should be used as little as possible as it imposes a heavy drain on the battery.
- The battery will provide usable power for continuous operation for approximately five to 20 minutes.

4. Land as soon as practicable.

Note

After failure of the dc generator, the following primary flight aids, in addition to the pitot-static instruments, will be available until battery power depletion:

- All gyro flight instruments
- Gyro compass
- Interphone and UHF command
- VOR radio
- IFF radar

The UHF command radio and interphone will be operable after battery power depletion on airplanes with the self-sustaining ac system. Refer to Section I for a detailed list of electrically operated equipment.

AC AND DC GENERATOR FAILURE

If both normal ac and dc generator fail, follow same procedures prescribed for individual generator failure.

Note

The emergency ac generator will furnish power to the ac essential bus and 28-volt dc transformer-rectifier. The TR unit in turn furnishes dc power to the UHF command radio and with a failed dc generator, the battery furnishes all other dc power.

If the emergency ac generator or transformer-rectifier fails:

1. Battery switch – TR FAIL position.

Note

- On airplanes with self-sustaining ac system, placing the battery switch in TR FAIL position connects the dc emergency bus to dc essential bus (battery). UHF command radio and interphone will continue to operate until battery power is depleted.
- Complete loss of UHF and interphone occurs on airplanes prior to incorporation of self-sustaining ac system.

INSTRUMENT FAILURE

Failure of the main ac generator is indicated by the ac failure warning light and power can be restored to the primary flight and engine instruments by energizing the emergency ac generator. Failure of either of the transformers will also result in failure of these instruments.

CAUTION

Loss of ac power results in failure of the hydraulic pressure gage, fuel quantity gage, fuel flow indicator, directional indicator (slaved), engine pressure ratio gage and attitude indicator. These instruments will continue to register the condition that existed at the time of power failure.

There is no indication of transformer failure other than the loss of operation of equipment and no alternate source of power is provided.

HYDRAULIC POWER SYSTEM FAILURE

Note

It should be noted on Form 781 if the RAT has been extended in flight.

FAILURE OF ONE HYDRAULIC SYSTEM

Airplane flight characteristics encountered during flight with one hydraulic system inoperative reveal that high speed maneuvering capabilities are reduced. With one hydraulic system inoperative, elevator hinge moment is limited to the extent that the airplane is capable of pulling only a small percentage of maximum g limit. RAT extension is not required while flight control system operating pressure is available in either hydraulic system. In the event of primary pump failure and/or frozen engine, the RAT will provide immediate pressure to the primary hydraulic system and should be extended prior to entering the landing pattern. If the hydraulic pressure-low warning light flashes, indicating failure of one hydraulic system, use the following procedure:

1. Reduce airspeed.

   Reduce airspeed to below RAT maximum extension speed by immediate power reduction. Maintain airspeed below RAT maximum extension speed so that the emergency system will be readily available in the event of failure of the remaining system.

CAUTION

With one hydraulic system inoperative, hinge moment limitations prevent pulling the maximum g limit. For supersonic dive recovery, this becomes especially critical. Refer to Section VI for maximum g's available with one system inoperative and for dive recovery information.

2. Restrict flight to avoid need for speed brakes.

   Avoid need for speed brakes to conserve hydraulic pressure for flight control operation with primary hydraulic system inoperative, and because pressure will not be available for speed brakes operation with secondary hydraulic system inoperative.

3. Determine which system failed.

   Select PRI, then SEC with the hydraulic pressure gage selector switch and check hydraulic pressure gage to determine which system gives the low pressure indication.

CAUTION

If the secondary hydraulic system fails, the force required for initial stick movement may increase to as much as double the force normally required, and positive stick centering may be lost. When secondary hydraulic system failure occurs, the following will be inoperative:

- Pitch and yaw damper systems
- Emergency ac generator
- Speed brakes
- Nose wheel steering
- Normal landing gear extension system.

4. Land as soon as practicable.
5. If primary hydraulic system has failed, RAT handle — Pull (just after turn on final approach) & hold for four seconds.

   Hold the handle in fully extended position for four seconds to insure a satisfactory extension of the RAT.

6. If secondary hydraulic system has failed, emergency landing gear extension handle — Pull.

   If secondary hydraulic system has failed, use the emergency pneumatic system to extend gear. Use the normal gear extension if primary hydraulic system has failed.

CAUTION

- Normal landing gear extension with the primary hydraulic system inoperative should be made at a time when minimum use of flight controls is required.
- After emergency landing gear extension, do not attempt to retract gear.

FAILURE OF BOTH HYDRAULIC SYSTEMS

If the hydraulic pressure-low warning light illuminates steadily, use the following procedure:

1. Hydraulic pressure gage — Check.

   Verify that both systems have failed by checking the hydraulic pressure gage.

Note

In event of a frozen engine the hydraulic pressure gage will be inoperative. If the ac generator has failed and the engine is not frozen, the emergency ac generator should be energized. If flight instruments fuel quantity gage or fuel flow indicator operate, then secondary hydraulic pressure is available to operate the emergency ac generator hydraulic motor, and pressure indications should be noted.

2. **RAT — PULL AND HOLD FOR FOUR SECONDS.**

   Extend the RAT by pulling the hydraulic emergency power handle. Hold the handle in the fully extended position for four seconds to insure a satisfactory extension of the RAT.

WARNING

If airspeed is above RAT maximum extension speed and engine is frozen do not extend the speed brakes to slow the airplane. Speed brakes extension will cause immediate depletion of remaining secondary hydraulic system pressure. Deceleration should be accomplished by moving the flight controls a minimum amount to establish a flight attitude that will provide deceleration.

3. Check flight control operation by:
   a. Checking that hydraulic pressure-low warning light starts to flash indicating that the RAT is providing pressure through the primary system.
   b. Checking hydraulic pressure gage (if ac power is available).
   c. Moving the control stick and checking control response.

Note

In the event that hydraulic system failure is the result of a frozen engine, the hydraulic pressure gage will be inoperative due to complete ac power failure and, therefore, will not indicate hydraulic pressure even though the emergency system is operating. In this event the pressure gage will not necessarily return to a reading of zero pressure. At this time, the dc (battery) operated hydraulic pressure-low warnng light will be the only instrument for indication of hydraulic pressure.

4. **IF FLIGHT CONTROL OPERATION IS NOT POSSIBLE — EJECT (REFER TO EJECTION, THIS SECTION).**
5. If flight control is still possible but the pressure gage indicates both systems are below 800 psi and the hydraulic failure warning light is still illuminated steadily — Eject.
6. If pressure is available for flight control operation, and either ac or dc electrical power is available to give an indication that at least one system is operating, use the following procedure:
   a. Restrict flight to avoid violent maneuvers and need for speed brakes.
   b. Extend landing gear by emergency system.
   c. Land as soon as practicable.

      Plan landing to provide for straight-in final approach using a minimum of control movement for a few seconds before flareout. This is necessary to assure adequate hydraulic pressure for normal elevon operation during the flareout.

HIGH-PRESSURE PNEUMATIC SYSTEM FAILURE

If the "PNEU LOW" warning light illuminates in flight prior to usage of the high-pressure pneumatic system, land as soon as practical. Either wheel brakes or drag chute may not be available after landing. If a high-pressure pneumatic system failure is suspected, land as short as possible and use nose wheel steering as soon as the airplane has been slowed sufficiently. Apply brakes cautiously to determine if both brakes operate.

FLIGHT CONTROL EMERGENCY OPERATION

"HYD OIL HOT" WARNING LIGHT ILLUMINATION

On some airplanes* a "HYD OIL HOT" warning light may give advance warning of hydraulic fluid overheat conditions and possible subsequent flight control oscillations. If the "HYD OIL HOT" warning light illuminates, proceed as follows:

Before takeoff —

- Engine — Shut down.

  If the "HYD OIL HOT" warning light illuminates before takeoff, shut down the engine to prevent further damage to hydraulic system components.

During takeoff —

- Takeoff — ABORT.

  If the "HYD OIL HOT" warning light illuminates during takeoff, and runway and overrun conditions permit, abort the takeoff; refer to ABORT, this Section.

In flight —

1. Reduce airspeed to 220-240 KIAS and reduce engine rpm to minimum required to maintain this speed.
2. Flight controls and other hydraulically operated components. Use minimum necessary to maintain safe operating conditions.

**Airplanes modified by TCTO 1F-102-847.*

3. Do not extend the RAT if hydraulic pressure is available.

Note

Extension of the RAT will serve only to increase hydraulic temperatures in the primary system.

4. Land as soon as practicable.

FLIGHT CONTROL SYSTEM OSCILLATIONS

Flight control oscillations are primarily due to hydraulic fluid overtemperatures, but may also be due to failure of the damper system or failure of the artificial feel system. Gradual failure of either hydraulic system may be accompanied by excessive generation of heat within the system. Elevon surface oscillations can result from overtemperature of the hydraulic fluid at the elevon control valve. If the oscillations are due to overtemperature, there will be a movement of the control stick; however, if the oscillations were due to the damper system there will be no accompanying movement of the control stick. On some airplanes* a warning light, located on the master warning light panel, is provided to give an indication of hydraulic system overtemperatures. If the "HYD OIL HOT" warning light illuminates and uncontrollable oscillations are encountered, proceed as follows:

1. **REDUCE AIRSPEED TO 220-240 KIAS AND REDUCE ENGINE RPM TO MINIMUM NECESSARY TO MAINTAIN THIS SPEED.**
2. Flight mode — DIRECT MAN. Use minimum control stick movement; attempt to fly hands off for at least 30 seconds.

Note

Pilot induced oscillations, which may occur at transonic speeds when the flight mode is DIRECT MAN, are a phenomenon not related to flight control system oscillations.

3. Do not extend the RAT.

Note

Extension of the RAT will serve only to increase the hydraulic temperature in the primary system.

4. Select alternate flight modes (DIRECT MAN, DAMPER and PILOT ASSIST) and remain in the flight mode where oscillations are at a minimum.

Note

To aid in reducing hydraulic system fluid temperatures, use minimum control movement at all times. Oscillations may be more rapidly reduced in magnitude by releasing the control stick than by correcting for the oscillations with stick input.

5. If oscillations continue, utilize the following additional procedures:
   a. Use rudder to counteract roll.
   b. Maintain safe altitude while severe oscillations persist.
6. During low fuel conditions, keep nose attitude level.

Note

The heat exchanger utilizes fuel from the No. 3 fuel tank to cool the hydraulic fluid; therefore, to maintain the most effective cooling during low fuel conditions (1000 pounds or less) the airplane should not be flown in a 5° or greater nose-low attitude except as required for descent to a landing.

7. Maintain safe altitude to evaluate nature of the system overheat.

Note

The system may cool if flight is continued under cautions dictated by the foregoing procedures.

8. When oscillations have been reduced to a safe minimum, land as soon as practicable.

Note

The oscillations may be caused by progressive failure of one hydraulic system, in which case complete failure of the malfunctioning system will usually terminate the oscillations.

TRIM FAILURE

If any of the three trim actuators should fail to operate the force required to perform normal maneuvers should be well within the normal physical capabilities of the pilot. Complete loss of dc power will cause the trim system to be inoperative.

PITCH AND YAW DAMPER SYSTEM FAILURE

The damper systems become inoperative if the main ac generator fails, the dc generator fails, or when secondary hydraulic system pressure fails. If any of these conditions occur the rudder and elevon servo actuators (extendible links) which are spring-loaded to center position will return to center and then serve as a fixed link. If a malfunction of the pitch and yaw damper system (i.e. failure

*Airpalnes modifed by TCTO 1F-102-847.

to remain engaged or unusual oscillations) makes flight with the system off mandatory, the following precautions shall be observed:

- Roll maneuvers shall be restricted to positive quadrant rolls only.

Note

A positive quadrant roll is a roll to a 90-degree bank either side of level flight, or from a 90-degree bank in one direction to a 90-degree bank in the opposite direction passing through normal flight position.

- No uncoordinated maneuvers shall be performed, except during takeoff and landing.
- Rapid application of aileron shall be avoided, except during takeoff and landing.
- No intentional rolling push-downs shall be performed.

ARTIFICIAL FEEL SYSTEM INOPERATIVE

Failure of the artificial feel system may be caused by a leak in the pneumatic pressure line, a leak in the feel force cylinder, or by electrical failure to either or both of the ram air ("q") intakes, allowing them to become clogged by ice while operating in icing conditions. Failure of the artificial feel system may be recognized by low stick and/or rudder forces, excessive maneuvering, or pilot-induced oscillations. Clogging of the small intake will normally cause the rudder pedal forces to be low, being about the same as the pedal forces encountered on the ground. The abnormally low rudder pedal forces will require caution when operating at high indicated airspeeds to prevent overcontrolling, but will probably appear normal during approach and landing. However, it is possible for the intake to clog at such a time as to trap high "q" pressure, resulting in high rudder pedal forces for approach and landing. Turn coordination, as normally applied through the yaw damper system, will not provide the desired amount of rudder to coordinate turns during flight with the intakes plugged. Flight under these conditions will probably require disengaging the yaw damper system. Elevator stick forces will be normal or low in all cases of "q" intake clogging. With the small intake clogged, elevator will be normal below .5 Mach, and low above that range, requiring caution at higher airspeeds to avoid overcontrolling. With both "q" intakes clogged, elevator stick force will be low, with the extreme low force being the same as that experienced while on the ground. The low elevator stick force will feel normal during approach and landing. In the event of artificial feel system failure, proceed as follows:

1. **AIRSPEED — REDUCE TO 220-240 KIAS.**
2. Use minimum control stick movement; attempt to fly hands off for at least 30 seconds.
3. Yaw damper system—Disengage.

LANDING GEAR EMERGENCY OPERATION

EMERGENCY GEAR LOWERING PROCEDURE

The emergency landing gear lowering procedure is outlined on figure 3-7. Refer to Section VII for a discussion of landing gear operation.

WINDSHIELD FAILURE

If a panel cracks in flight and is still transparent (indicating outer layer failure only), cabin pressure need not be dumped. However, if the panel becomes white or opaque (indicating failure of the inner layer, which is the stress-bearing part of the windshield), then cabin pressure should be dumped immediately. If cabin pressure is released at an altitude of approximately 43,000 feet, the airplane oxygen system will automatically supply pressure to the partial pressure suit. The mission can be completed safely so long as there is sufficient oxygen in the airplane to furnish pressure to the partial pressure suit and oxygen for pilot breathing.

Note

Additional oxygen is available to the partial pressure suit through the oxygen bailout bottles, which can be actuated by manually pulling the green knob. Because of the limited duration of the bailout bottles, an immediate descent to a safe altitude is necessary.

In the event of windshield failure, proceed as follows:

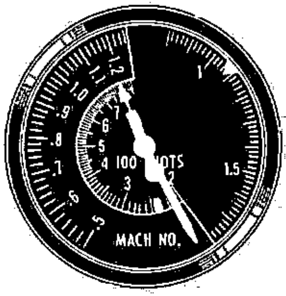
1. **CABIN AIR SWITCH — RAM (IF PANEL BECOMES OPAQUE).**
2. Nesa switches — OFF.

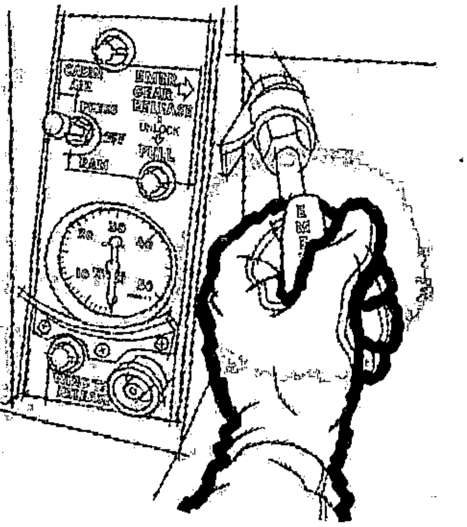
Note

If it is apparent during flight that the windshield has been or is being damaged, distorted, or discolored from uneven heat distribution, the cause may be due to failure of the power relay or control box. If such is the case, turning the Nesa switch to OFF will not remove electrical power from the windshield heating elements. It will be necessary to either pull the three-phase circuit breakers placarded "Windshield Anti-Ice," which are located on the left-hand aft circuit breaker panel, or place the ac bus switch to EMER, thus removing power from the Nesa heaters. The circuit breakers are difficult to reach in flight. Before moving the ac bus switch to EMER it must be determined whether the nature of the emergency is such that flight may be continued safely without additional items connected to the nonessential bus as they also will be rendered inoperative.

landing gear emergency lowering

1 REDUCE SPEED BELOW GEAR DOWN LIMIT SPEED.





2 PUSH DOWN (UNLOCK) THEN PULL THE LANDING GEAR EMERGENCY HANDLE APPROXIMATELY 2 INCHES.

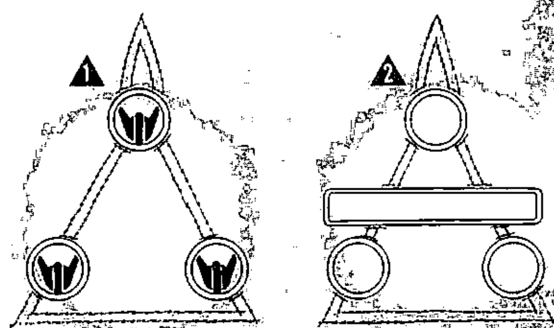
NOTE

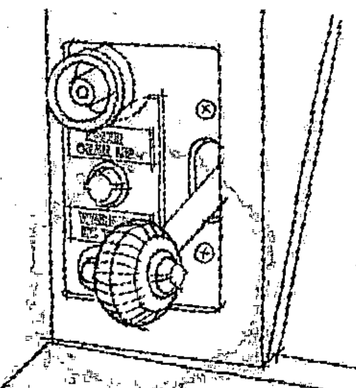
- ON AIRPLANES AF53-1791 THRU 56-972, THIS HANDLE IS PLACARDED "EMER GEAR RELEASE PULL" AND MAY BE PULLED STRAIGHT OUT. CHECK THAT SPRING CLIP OR DETENT ENGAGES IN BACK SIDE OF EMERGENCY LANDING GEAR EXTENSION HANDLE TO PREVENT INWARD MOTION OF THE HANDLE.
- ON AIRPLANES AF56-973 AND ON, THE HANDLE IS PLACARDED "EMER GEAR RELEASE—UNLOCK—PULL" AND AUTOMATICALLY LOCKS WHEN IN THE OUT POSITION.

3 CHECK LANDING GEAR POSITION INDICATORS FOR GEAR DOWN (WHEELS).

CAUTION

IN PLANNING THE LANDING WHEN THE GEAR IS TO BE LOWERED BY THE EMERGENCY SYSTEM, ONE MINUTE SHOULD BE ALLOWED FOR GEAR EXTENSION TIME ON AIRPLANES AF53-1791 THRU 56-1518 UNLESS MODIFIED BY T.O. 1F-102-655. ON AIRPLANES AF57-770 AND ON AIRPLANES MODIFIED BY T.O. 1F-102-655, EMERGENCY EXTENSION SHOULD NOT REQUIRE MORE THAN 10 SECONDS.





4 LANDING GEAR HANDLE DOWN.

NOTE

THE NORMAL LANDING GEAR HANDLE SHOULD BE IN THE DOWN POSITION PRIOR TO LANDING. IF THE REQUIREMENT FOR EMERGENCY GEAR EXTENSION IS DUE TO FAILURE OF THE SECONDARY HYDRAULIC SYSTEM, OR ENGINE FAILURE, THE EMERGENCY GEAR EXTENSION HANDLE SHOULD BE PULLED FULLY OUT PRIOR TO PLACING THE NORMAL GEAR HANDLE DOWN. THIS ACTION WILL CONSERVE ALL AVAILABLE HYDRAULIC PRESSURE FOR FLIGHT CONTROL OPERATION.

CAUTION

- NORMAL LANDING GEAR EXTENSION WITH THE PRIMARY HYDRAULIC SYSTEM INOPERATIVE SHOULD BE MADE AT A TIME WHEN MINIMUM USE OF FLIGHT CONTROLS IS REQUIRED.
- NOSE WHEEL STEERING IS INOPERATIVE WHEN GEAR IS EXTENDED BY THE EMERGENCY SYSTEM.

1 BEFORE MODIFICATION BY T.O. 1F-102-728.
2 AFTER MODIFICATION BY T.O. 1F-102-728.

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Figure 3-7

CANOPY WARNING LIGHT ILLUMINATED IN FLIGHT

Illumination of the canopy unlocked warning light while in flight indicates that the canopy latch hooks are not fully engaged and loss of the canopy could ensue. If the light comes on in flight, observe the following:

1. Do not move the canopy latch handle.
2. Reduce airspeed to 230-240 KIAS.
3. Cabin air switch to OFF.
4. Return to base as soon as possible.

Note

The Emergency Abbreviated Check List is contained in T.O. 1F-102A-(CL)1-1.

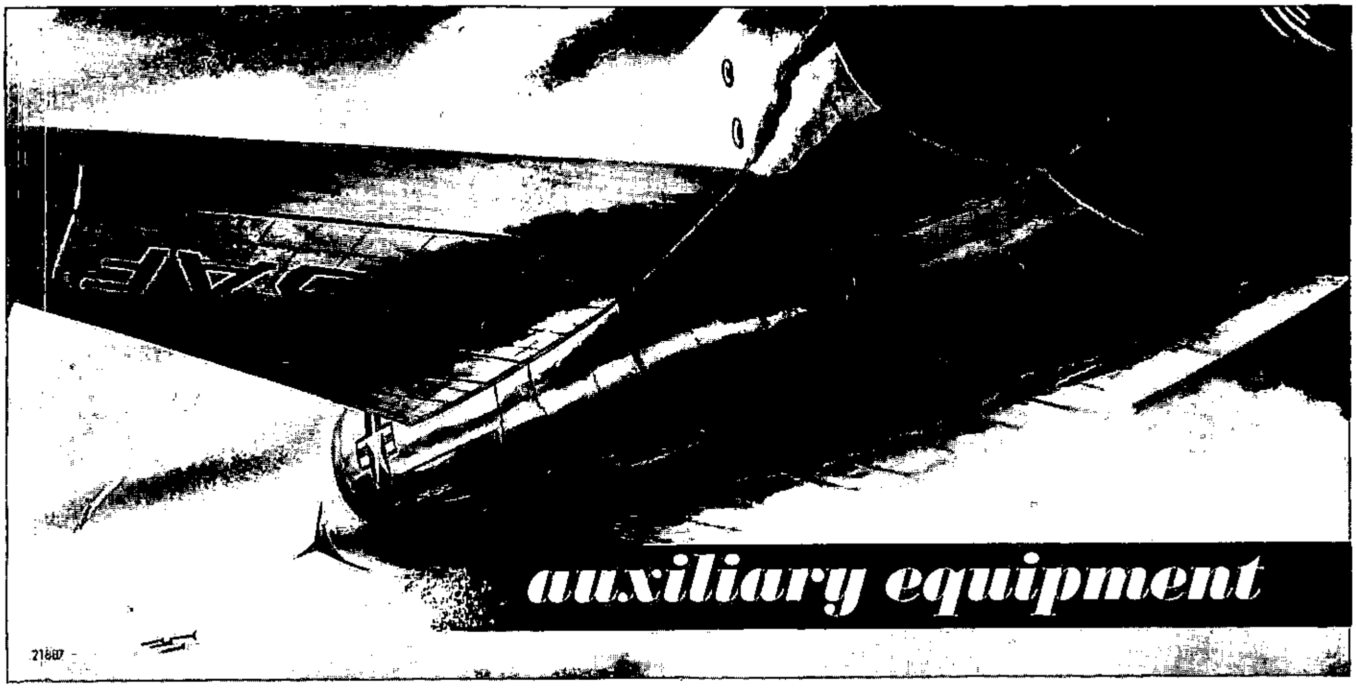

TABLE OF CONTENTS

COCKPIT AIR-CONDITION AND PRESSURIZATION SYSTEM

The cockpit air-conditioning system (figure 4-1) uses air from the low-pressure pneumatic system to provide the pilot with a pressurized, temperature-controlled cockpit. For air-conditioning and cockpit pressurization, hot compressed air (engine bleed air) is ducted to the cockpit by a system which routes the air through a refrigeration unit and also provides a bypass around the refrigeration unit. In the refrigeration unit, the hot air is first routed through an air-to-air heat exchanger and then through a cooling turbine where it is cooled and expanded. A centrifugal flow fan, mounted on a common shaft with the cooling turbine wheel, absorbs the energy of the turbine wheel by drawing cooling ram air across the heat exchanger. Cold air from the refrigeration unit is mixed with hot bypassed air and then discharged into the cockpit through two discharge tubes on each side of the cockpit. Cockpit temperature control is achieved by automatically modulating a bypass valve which mixes the proportion of hot and cold air required to maintain the temperature selected by the pilot. Temperature sensing is accomplished by two thermostats, one located in the cockpit air inlet duct and the other in the outlet duct. When maximum cooling is required, all the air is routed through the refrigeration unit. Because full ram air is not flowing over the air-to-air heat exchanger, cockpit cooling is limited while the airplane is on the ground. In the event the normal air-conditioning system malfunctions during flight, a ram air system can be used to provide cockpit ventilation. This ram air ventilation system does not provide cockpit pressurization. A ram air pressure regulator and shutoff valve shuts off the ram air supply if ram air temperature or pressure becomes excessive. A check valve located in the air duct entering the cockpit on some airplanes* will prevent rapid depressurization of the cockpit in the event of engine flameout or duct failure. Pressurization of the cockpit is provided by regulating the discharge of air from the cockpit. The pressure regulator in the floor of the cockpit automatically controls the outflow of cockpit air to maintain a preset cockpit pressure schedule. See figure 4-2. This schedule provides for an unpressurized cockpit up to an altitude of 10,000 feet. As the airplane continues to climb, the cockpit altitude remains at

**AF 56-1275 thru -1316, -1332 & on, & airplanes modified by TCTO 1F-102-656.*

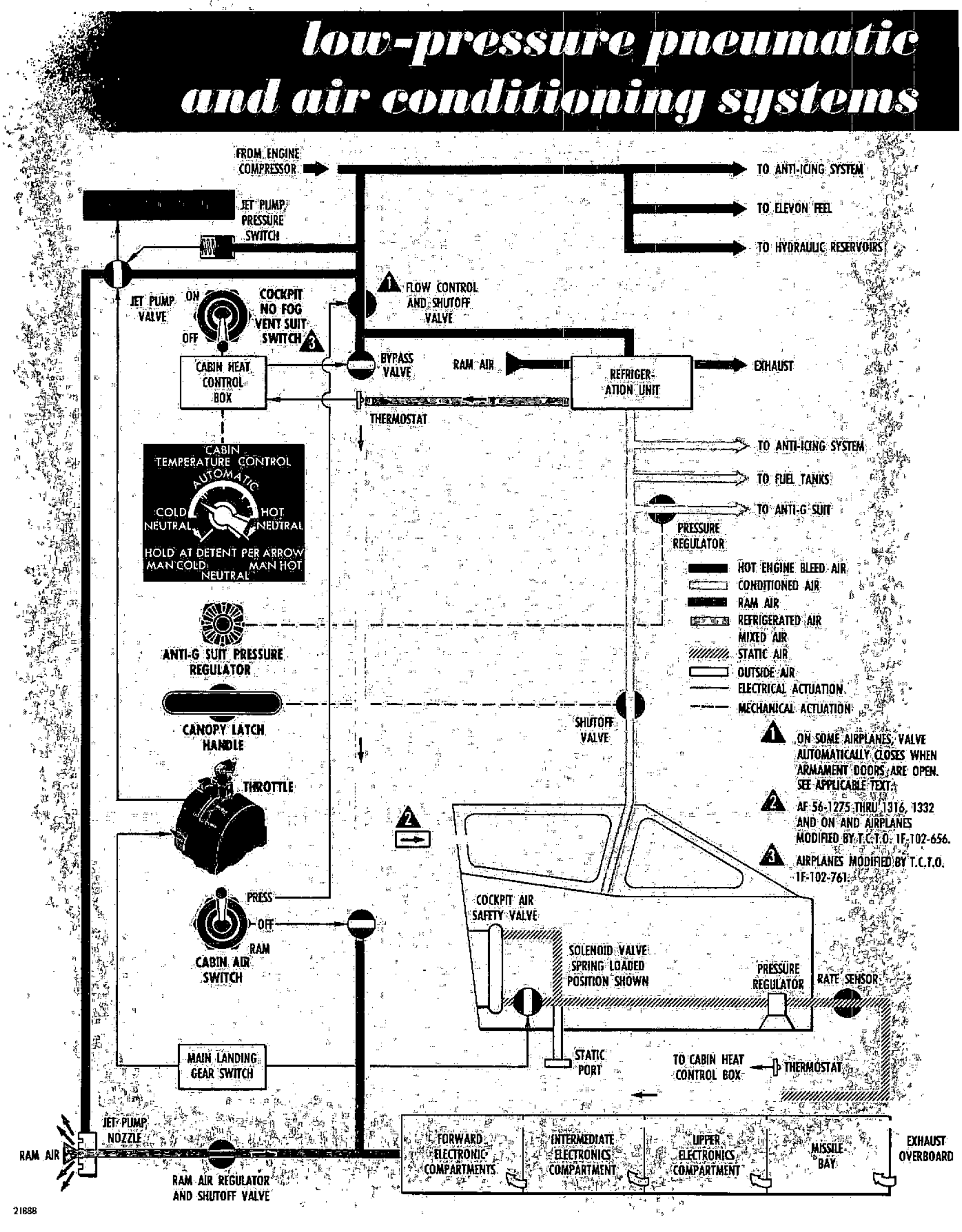
low-pressure pneumatic and air conditioning systems
FROM ENGINE COMPRESSOR
TO ANTI-ICING SYSTEM
TO ELEVON FEEL
TO HYDRAULIC RESERVOIRS
JET PUMP PRESSURE SWITCH
JET PUMP VALVE
ON
OFF
COCKPIT NO FOG VENT SUIT SWITCH
FLOW CONTROL AND SHUTOFF VALVE
CABIN HEAT CONTROL BOX
BYPASS VALVE
RAM AIR
REFRIGERATION UNIT
EXHAUST
THERMOSTAT
CABIN TEMPERATURE CONTROL
AUTOMATIC
COLD NEUTRAL
HOT NEUTRAL
HOLD AT DETENT PER ARROW
MAN COLD
MAN HOT
NEUTRAL
TO ANTI-ICING SYSTEM
TO FUEL TANKS
TO ANTI-G SUIT
PRESSURE REGULATOR
HOT ENGINE BLEED AIR
CONDITIONED AIR
RAM AIR
REFRIGERATED AIR
MIXED AIR
STATIC AIR
OUTSIDE AIR
ELECTRICAL ACTUATION
MECHANICAL ACTUATION
ANTI-G SUIT PRESSURE REGULATOR
CANOPY LATCH HANDLE
SHUTOFF VALVE
1 ON SOME AIRPLANES, VALVE AUTOMATICALLY CLOSES WHEN ARMAMENT DOORS ARE OPEN. SEE APPLICABLE TEXT.
2 AF 56-1275 THRU 1316, 1332 AND ON AND AIRPLANES MODIFIED BY T.C.T.O. 1F-102-656.
3 AIRPLANES MODIFIED BY T.C.T.O. 1F-102-761.
THROTTLE
COCKPIT AIR SAFETY VALVE
PRESS
OFF
RAM
CABIN AIR SWITCH
SOLENOID VALVE SPRING LOADED POSITION SHOWN
PRESSURE REGULATOR
RATE SENSOR
MAIN LANDING GEAR SWITCH
STATIC PORT
TO CABIN HEAT CONTROL BOX
THERMOSTAT
JET PUMP NOZZLE
RAM AIR
RAM AIR REGULATOR AND SHUTOFF VALVE
FORWARD ELECTRONIC COMPARTMENTS
INTERMEDIATE ELECTRONICS COMPARTMENT
UPPER ELECTRONICS COMPARTMENT
MISSILE BAY
EXHAUST OVERBOARD
21888

Figure 4-1

10,000 feet until an airplane altitude of approximately 26,500 feet is reached. Above this altitude a constant 5.0 psi differential pressure is maintained.

Note

Cockpit pressure fluctuations may be experienced at altitudes between 5000 feet and 15,000 feet in a speed range of 200 KIAS to 400 KIAS.

To prevent smoke and fumes from entering the cockpit, the bleed air shutoff valve shuts off the conditioned air supply to the cockpit on some airplanes* while the armament bay doors are opened for armament firing. Due to leakages, cockpit pressure may drop while the air supply is shut off. When pressurization is resumed, a rate sensor controls the speed at which the cockpit is repressurized. This prevents an initial pressure surge.

Note

Pressurization is resumed three to five seconds after the armament bay doors close or, if door closing is delayed by a misfire, by action of a time delay.

On other airplanes, in order to maintain cockpit pressurization during armament firing, cockpit pressurization is not shut off and some smoke or fumes in the cockpit may be experienced during firing.

CABIN AIR SWITCH

The cabin air switch (figure 1-24) is located on the left-hand auxiliary instrument panel. The switch is placarded "Cabin Air" and has three positions, PRESS, OFF, and RAM. When the switch is in the PRESS position, the ram air shutoff valve is closed to shut off the flow of ram air to the cockpit and the bleed air shutoff valve is open to route engine bleed air through the refrigeration and bypass unit to the cockpit. When the switch is in the RAM position, the ram air shutoff valve is open to route ram air to the cockpit, the bleed air shutoff valve is closed to shut off the engine bleed air supply, and the cabin safety valve is open to dump cockpit pressure. When the switch is in the OFF position both the ram air shutoff valve and the bleed air shutoff valve are closed to shut off the entire system. The cabin air switch receives power from the 28-volt dc essential bus.

CABIN TEMPERATURE CONTROL KNOB

The rotary cabin temperature control knob (figure 1-21) on the utility switch panel provides for automatic control of cockpit temperature within a range of 45°F to 100°F

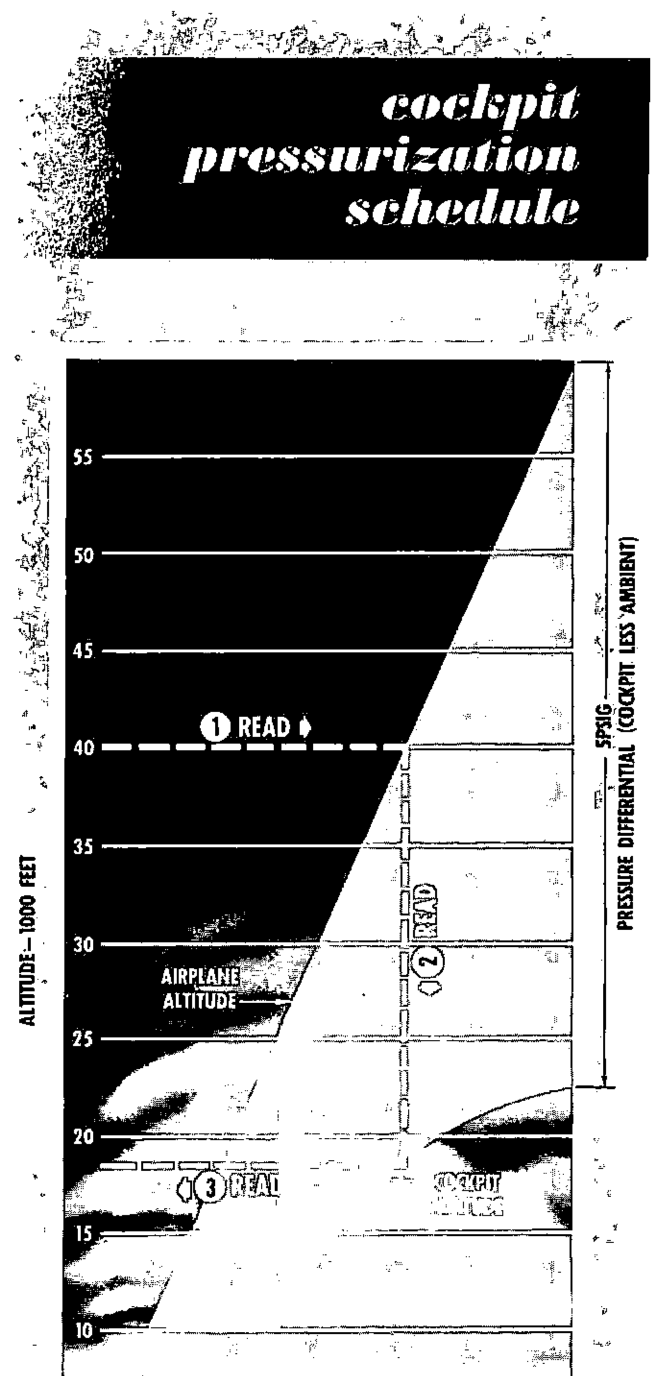

Figure 4-2

AF 53-1792, -1796, -1798 thru -1805, -1807 thru -1811, -1814 thru -1818, 54-1371 thru -1379, -1381 thru -1387, -1389, -1391 thru -1397, -1399, -1400, -1402 thru -1407, 55-3357 thru -3438, -3440 thru -3464, 56-957 thru -975, -977 thru -1088, -1090 thru -1098 unless modified by TCTO 1F-102-647.

or for manual control of the system in the event of malfunction of the automatic control. Temperature control is accomplished by the positioning of the refrigeration unit bypass valve to mix hot bleed air with refrigerated air in the required proportions to meet either the automatic or manual demands of the system. The temperature control switch is ineffective when the cabin air switch is in RAM position. The temperature control switch has an AUTOMATIC range with a NEUTRAL position at both extremes of automatic selection. The knob can be held against light spring tension to the MAN HOT or MAN COLD position to override the automatic system. When the knob is released it will return to the NEUTRAL position.

Note

If too much pressure is applied to the switch during manual operation, the switch will pass through the manual position into a common neutral range. The switch will not function from this range and should be returned to its previous position.

The switch controls power from the 28-volt dc essential bus.

COCKPIT NO-FOG VENT SUIT SWITCH

The cockpit no-fog vent suit switch* is located on the utility switch panel (figure 1-21) and has ON and OFF positions. When the switch is placed in the ON position, cockpit temperature control is governed by the cockpit air inlet duct thermostat only. A steady temperature of 80° (±10°)F is then maintained in the cockpit and in the ventilated flight suit (if used), regardless of the position of the automatic temperature control knob. This temperature range will minimize the probability of cockpit fog from the air conditioning system. Power to the cockpit no-fog vent suit switch is supplied by the 28-volt dc essential bus.

CABIN PRESSURE ALTITUDE GAGE

The cabin (cockpit) pressure altitude gage (43, figure 1-4) is located on a panel forward of the left console and indicates cockpit pressure in feet above sea level. It is graduated from zero to 50,000 feet pressure altitude. The gage is vented only to pressure within the cockpit.

NORMAL OPERATION OF COCKPIT AIR-CONDITIONING AND PRESSURIZATION SYSTEM

1. Place cabin air switch in PRESS position.
2. Close canopy.
3. Cabin temperature knob as desired within the AUTOMATIC range.
4. Cockpit no-fog vent suit switch as required.

WARNING

During high humidity conditions, the cooling effect produced by the refrigeration turbine of the air-conditioning system can create heavy condensation in the cockpit inlet air. This condensation forms a very dense fog within the cockpit. Under extreme conditions, cockpit fog can reduce visibility to a point where it is impossible to see the cockpit instruments, and outside visibility may also be completely obscured. Cockpit fog is most likely to occur on takeoff, but may also occur during landing or go-around. To prevent fog formation, the temperature control knob should be placed in the AUTOMATIC HOT range, or the cockpit no-fog vent suit switch should be placed to ON (if installed), prior to takeoff or landing.

EMERGENCY OPERATION OF COCKPIT AIR-CONDITIONING AND PRESSURIZATION SYSTEM

If automatic temperature control system becomes inoperative or if temperatures beyond the normal limits of the system are desired:

1. Hold cabin temperature control knob momentarily to MAN HOT or MAN COLD.
2. Repeat as necessary until desired change occurs.

If cabin pressurization fails and cabin altitude increases appreciably over actual altitude:

1. Determine cabin altitude.
2. If cabin altitude is above the actual altitude of the airplane, place cabin air switch to RAM, and/or reduce speed.

Note

When the cabin air switch is in RAM position, the following additional systems become inoperative: anti-g suit pressurization, radome anti-icing, and canopy sealing.

WARNING

If excessive cockpit fog occurs during takeoff, landing, or go-around, immediately place cabin air switch to RAM position to shut off the source of cockpit fog and to aid in more rapidly dissipating the fog. After the fog has cleared, or the possibility of fog formation no longer exists, resume normal operation of the cockpit air-conditioning and pressurization systems.

**Airplanes modified by TCTO 1F-102-761.*

DEFOGGING SYSTEM

Warm air from the low-pressure pneumatic system is used to defog the canopy panels and, on some airplanes*, the forward one-third of the windshield. Flow of this air is controlled by a switch on the utility switch panel. The windshields are defogged by electrically heating the Nesa glass panels. Refer to WINDSHIELD ANTI-ICING, this Section.

CANOPY DEFOG SWITCH

The canopy defog switch (figure 1-21) on the utility switch panel has CANOPY DEFOG and OFF positions. The switch controls 28-volt dc essential bus power to a motor driven valve in the hot air duct. When the switch is in CANOPY DEFOG position the valve is open and warm air is directed to the canopy and, on some airplanes*, to the forward one-third of the windshield. When the switch is in OFF position the valve is closed. Limit switches shut off the motor when the valve reaches the fully open or fully closed positions.

OPERATION OF CANOPY DEFOG SYSTEM

Note

During operation in high humidity conditions (ground dew point 20°C or higher) operate canopy defog continuously at altitude in order to insure a fog-free canopy during descent. For less severe humidity conditions, canopy defog switch need not be turned ON until just before descent.

The canopy defog system is either fully on or fully off, having no provisions for regulation by the pilot. To operate the system, proceed as follows:

1. To turn on the system, place the canopy defog switch in CANOPY DEFOG position.
2. To turn the system off, return the canopy defog switch to OFF.

Note

In the event of canopy defog system failure, hold the cabin temperature control knob to MAN HOT as necessary when canopy fogging occurs.

ANTI-ICING SYSTEMS

The airplane is equipped with an automatic anti-icing system that will provide automatic protection for the engine, intake duct leading edges, engine inlet guide vanes, radome, and ram air, "q" intakes on the fin if the system is armed. The automatic system may be overridden and operated manually in the event it malfunctions. Separate controls also provide for operation of the windshield anti-icing system, windshield rain clearing system, and the pitot-static head anti-icing system. Wing and fin anti-icing is not provided.

AF 56-1275 thru -1316, -1332 & on.

Note

The airplane configuration is such that wing and tail ice formations have no effect on stability and control. Engine thrust available is adequate to compensate for any drag increase.

ICE DETECTOR

An automatic ice detector probe is mounted in the engine intake duct to sense icing conditions and automatically turn on the ice protection system if the system is armed. When the ice protection system turns on, the detector immediately starts to de-ice. De-icing of the detector usually takes less than a second. If the probe becomes clogged or anti-ice heat to the probe remains on longer than approximately 20 seconds, the pilot is informed through the master warning system, and anti-ice detection is inoperative until the probe is unclogged. If, after the detector is de-iced, there is no further indication of ice for a period of one minute, the ice protection system will turn off automatically until the next time the detector senses ice.

ENGINE ANTI-ICING

The engine inlet guide vanes at the face of the compressor are anti-iced by engine bleed air from the low-pressure pneumatic system. The thermostatically controlled air flows through the hollow guide vanes. This system is a component part of the engine and is controlled through the automatic ice protection system.

INTAKE DUCT ANTI-ICING

The engine intake ducts are supplied with hot engine bleed air from the low-pressure pneumatic system. The thermostatically controlled bleed air flows through the double skin structure of the duct leading edges and is exhausted on the inside surface aft of the leading edge. The system is controlled through the automatic ice protection system.

RADOME ANTI-ICING

The radome anti-icing system uses glycol to prevent ice formation. The glycol supply tank has a storage capacity of two gallons. Low-pressure pneumatic system air is used to force the glycol from the tank and through the porour metal ring at the base of the nose boom. Airflow then spreads the glycol over the surface of the radome. The system is controlled through the automatic ice protection system. See figure 1-33 for anti-icing fluid specifications.

RAM AIR "q" PRESSURE INTAKE ANTI-ICING

The ram air pressure intakes on the fin are electrically anti-iced by 28-volt dc power from the essential bus. The system is controlled through the automatic ice protection system, and is operative when the anti-ice switch is in MANUAL ON or AUTO. The intake heaters receive only partial power until the right main landing gear door is closed.

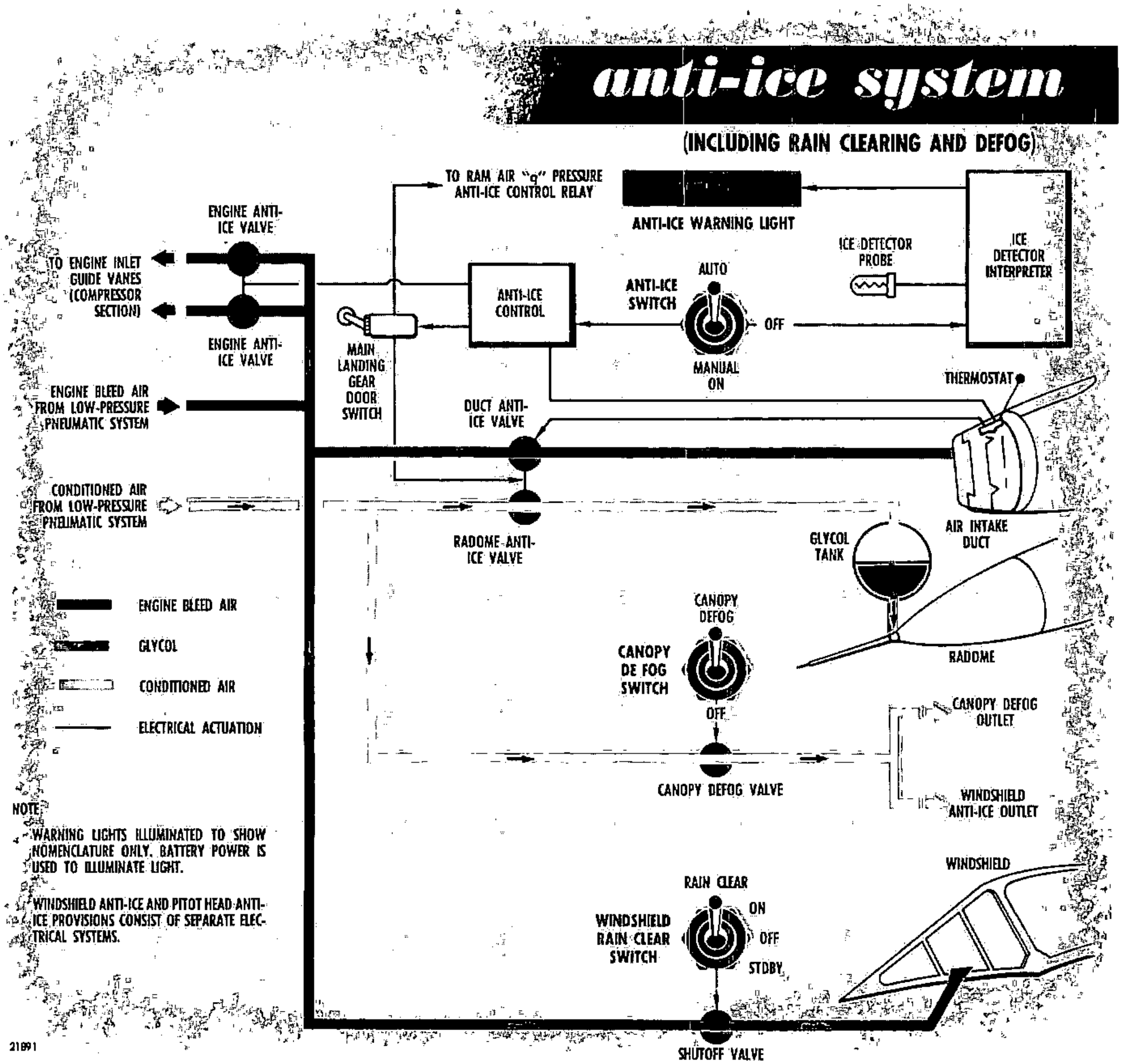

Figure 4-3

WINDSHIELD ANTI-ICING AND DEFOGGING

Anti-icing and defogging of the windshield panels is accomplished by electrical heating units imbedded in the Nesa panels. These units have temperature sensing elements to prevent overheating of the glass. On some airplanes anti-icing and defogging of the aft two-thirds is accomplished by electrical heating, the forward one-third uses hot air sent through ducts and distribution nozzle to the apex and is controlled by the canopy defog switch.

WINDSHIELD RAIN CLEARING SYSTEM

The windshield rain clearing system provides a high-velocity stream of hot air from the low-pressure pneumatic system to clear rain or snow from the left windshield panel. A switch in the cockpit enables the pilot to control the electrically operated shutoff valve in the rain clear hot air duct.

ANTI-ICE SWITCH

The anti-ice switch (figure 1-21) on the utility switch panel has three positions, AUTO, MANUAL ON, and OFF. When the switch is in AUTO position, anti-icing of the engine inlet guide vanes, intake duct leading edges, radome, and ram air pressure intakes on the fin is controlled by the automatic ice detection system through the ice detector. When the switch is in MANUAL ON position, the automatic features of the ice detection system are bypassed and all of the anti-icing systems which are controlled by the automatic ice detector system are activated regardless of icing conditions. On some airplanes*, it is necessary to pull the toggle switch out and then down to place the switch in MANUAL ON. On some airplanes** the system is deenergized if the anti-ice switch is in OFF position. On other airplanes, however, only the anti-ice warning light is deenergized and the anti-ice circuit remains energized as if in AUTO position. The anti-ice switch receives power from the 28-volt dc essential bus.

WINDSHIELD ANTI-ICE (NESA) SWITCH

The windshield anti-ice and defog system is controlled by switches located on the utility switch panel (figure 1-21). Some airplanes have a single switch, placarded "Nesa," with NORMAL and OFF positions. Placing the switch in NORMAL position connects 200/115-volt ac nonessential power to the Nesa transformers for both panels. Other airplanes† have two Nesa switches, one for the left and one for the right windshield panel. These switches are placarded "Nesa Control LH-RH" and provide the same functions for their respective panels as the single switch on the earlier configuration.

WINDSHIELD RAIN CLEAR SWITCH

The windshield rain clear switch is located on the utility switch panel (figure 1-21). On some airplanes, the switch has two positions, RAIN CLEAR, and OFF. In the RAIN CLEAR position, the switch supplies 28-volt dc essential bus power to the solenoid valve in the rain clear hot air duct. Other airplanes† have a third switch position, STDBY, which provides for automatic operation of the windshield rain clearing system if the Nesa system fails due to an ac power failure.

PITOT HEAT SWITCH

The pitot-static head anti-icing switch (figure 1-21) on the utility switch panel is placarded "Pitot Heat" and has ON and OFF positions. When in the ON position, the switch supplies 28-volt dc essential power to the pitot-static head on the nose boom.

ANTI-ICE SYSTEM WARNING LIGHT

A light on the warning light panel (figure 1-26) will illuminate to display "ANTI-ICE" when the anti-ice switch is OFF or when the airflow through the engine intake duct is under 40 knots with the switch in AUTO. The light will illuminate during normal operation (anti-ice switch AUTO) when the ice detector probe heater is on for 17 to 20 seconds and the probe openings remain clogged to indicate automatic anti-ice system is inoperative. The light will remain on until the probe is no longer clogged or the anti-ice switch is moved to MANUAL ON. The warning light is part of the master warning system and operates from the 28-volt dc essential bus.

*AF 56-1234 thru 56-1316, 56-1321 & on, & airplanes modified by TCTO 1F-102-664.

**AF 56-1045 & on.

NORMAL OPERATION OF ANTI-ICING SYSTEMS

The windshield anti-icing system, which also provides defogging, should be turned on as soon as ground electrical power is connected to the airplane and left on continuously until engine shutdown after completion of flight. This is necessary to insure proper defogging.

1. Place anti-ice switch in AUTO position. Under known or suspected icing conditions, place anti-ice switch in MANUAL ON.
2. Place windshield anti-ice (Nesa) switch in NORMAL (some airplanes); both Nesa switches ON (other airplanes)†.
3. Place rain clear switches in STDBY (some airplanes)†.
4. Move pitot heat switch to ON.

EMERGENCY OPERATION OF ANTI-ICING SYSTEMS

Automatic Anti-Icing System.

1. Place anti-ice switch to MANUAL ON position.

Windshield Anti-Icing System.

In the event of loss of ac electrical power to the laminated windshield, windshield defogging may be obtained by the following procedure:

1. If windshield is heated and power failure is noted immediately, place windshield rain clear switch to RAIN CLEAR.

   If the system has functioned normally, maintaining a heated windshield, and ac power failure is noted, the rain clearing system should be turned on immediately. Activation of the rain clearing system will provide an effective "heat block" for the heat energy stored in the left-hand windshield panel at the time of the power failure and will maintain the inside surface temperature of the left-hand panel.

2. If ac power failure is not noted until condensation has formed, proceed as follows:
   a. Windshield rain clear switch to RAIN CLEAR. Activation of the rain clearing system will initiate a heat addition process on the left-hand windshield outer surface.

†Airplanes modified by TCTO 1F-102A-562.

communications and associated electronic equipment

⚠1 SOME AIRPLANES—REFER TO APPLICABLE TEXT

TYPE	DESIGNATION	FUNCTION	PRIMARY OPERATOR	RANGE	LOCATION OF CONTROLS
Interphone Equipment	AN/AIC-10	Connects audio of radio and navigation systems and pilot's headset. Also provides communication between pilot and ground crew when plane is on the ground.	Pilot and ground crew.	Cockpit to ground crew.	Ground crew connection and amplifier in the nose wheel well.
UHF Command Radio	AN/ARC-34	Communications from airplane to airplane and from airplane to ground by UHF communications radio.	Pilot	Line of sight.	Left-hand console.
Visual Omni-range Receiver ⚠1	AN/ARN-14	Provides information for navigation and instrument low approach.	Pilot	Localizer approx. 45 miles and omni-directional 100 miles, depending on the flight altitude.	Right-hand console. Indicator on instrument panel.
Glide Slope Receiver ⚠1	AN/ARN-18	Indicates glide angle for automatic approach.	Pilot	Approximately 25 miles.	Right-hand console. Indicator on instrument panel.
Tacan ⚠1	AN/ARN-21	Used in conjunction with Tacan surface beacon to provide bearing and range information.	Pilot	Approximately 195 miles, line of sight.	Right-hand console. Indicators on instrument panel.
ILS Receiver ⚠1	AN/ARN-31	Receives ILS localizer and glide path transmission.	Pilot	Localizer approximately 45 miles, glide path approximately 25 miles.	Right-hand console. Indicator on instrument panel.
Marker Beacon Receiver	AN/ARN-12 or AN/ARN-32	Receives location marker signal on navigational beam.	Pilot		Indicator light on instrument panel.
IFF ⚠1	AN/APX-6A AN/APX-25	Automatic radar identification returned if challenged by surface or airborne sets. Can transmit emergency identification signal.	Pilot	Line of sight.	Right-hand console.

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Figure 4-4

b. Cabin temperature control knob to MAN HOT. Regulate cabin air temperature to maximum tolerable. This action will start a heat addition process on the inside surface of the windshield.

Note

- The desired heating on a windshield that has cooled, or that was not preheated electrically, will require from two to four minutes when using the above procedure.
- On some airplanes*, the rain clearing system will come on automatically after an electrical power failure if the rain clear switch is in STDBY position.

COMMUNICATIONS AND ASSOCIATED ELECTRONIC EQUIPMENT

TABLE OF COMMUNICATION AND ASSOCIATED ELECTRONIC EQUIPMENT

See figure 4-4.

INTERPHONE — AN/AIC-10

An interphone system is installed to permit communications between the pilot and a ground crew through a nose wheel well connection, when the airplane is on the

**Airplanes modified by TCTO 1F-102A-562.*

ground, and also to provide an amplifier for the dynamic microphone for transmitting. The dynamic microphone is the push-to-talk type. The pushbuttons are located on the throttle grip (figure 1-7) and on the control stick grip (figure 1-20). There is no cockpit interphone control panel. The system is on whenever power is available from the 28-volt dc essential bus or, on some airplanes*, from the emergency dc bus.

UHF COMMAND RADIO — AN/ARC-34

Note

> No transmission will be made on emergency (distress) frequency channels except for emergency purposes in order to prevent transmission of messages that could be construed as actual emergency messages.

The AN/ARC-34 command radio set provides voice transmission and reception on 1750 frequencies in the range of 225.0 to 399.9 megacycles. The control panel for the set (figure 4-5) is on the left console and permits selection of any of 20 frequencies which can be preset in any order. A remote indicator** (44, figure 1-4) on the instrument panel, will indicate the frequency of the preset channel selected. In addition, an operating frequency can be set up manually without disturbing any of the preset frequencies. The set uses two receivers, a main and a guard receiver. The guard receiver is capable of covering the frequency range of 238.0 to 248.0 megacycles. Tuning of the guard receiver is set and fixed prior to installation. The functions of the set are selected by the four-position main control switch on the right side of the control panel. The switch has four positions: OFF, MAIN, BOTH, and ADF. When the switch is on MAIN position, the transmitter and receiver are operative on the same selected main frequency. The BOTH position allows transmission and reception on the main selected channels and simultaneous reception on the guard channel. The ADF position of the switch is inoperative in this installation. A button placarded "Tone" is adjacent to the main control switch. Holding the tone button down interrupts reception and provides a continuous tone transmission to aid ground stations in obtaining a direction finding bearing. A selector control above the channel selector knob is used to select the desired operating mode. The MANUAL position of the control permits the operating frequency to be changed to any desired frequency in the operating range of the set. The GUARD position selects the fixed guard frequency for the main receiver and transmitter, and PRESET is used to allow selection of any of the 20 preset frequencies. When the selector control is in PRESET, subsequent movement of the channel selector knob (at the center of the control panel) changes the frequency

*Airplanes modified by TCTO 1F-102-727.
**Airplanes modified by TCTO 1F-102-860.

uhf command radio control panel

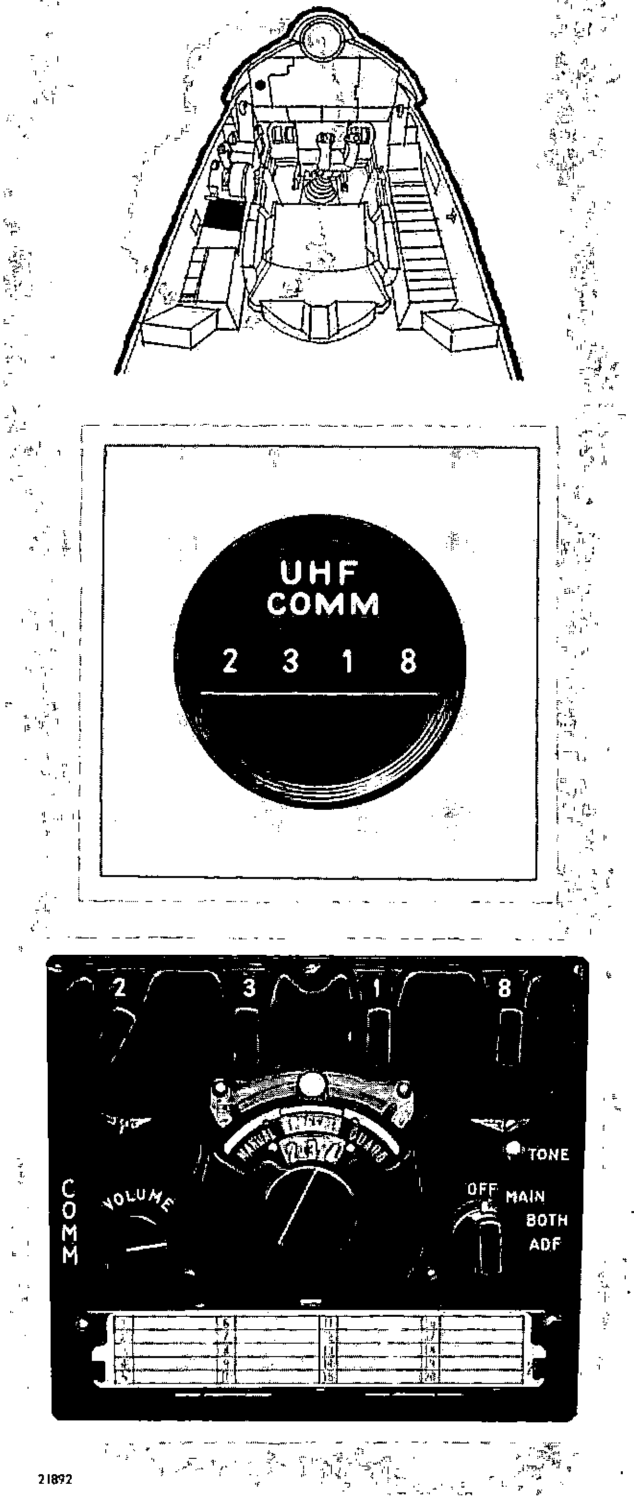

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Figure 4-5

to the desired preset channel. A numerical indication of the selected channel appears in a window above the selector knob. A record of the frequencies that have been preset and assigned to the 20 channels can be noted on a plastic card provided for this purpose and located at the bottom of the control panel. The preset frequencies can be changed in flight if necessary. Audio volume is adjusted by a knob on the left side of the control panel. The AN/ARC-34 command radio requires 28-volt direct current for operation. The power source varies on different airplanes. On some early airplanes, the set receives power directly from the dc essential bus. Other airplanes* include a transformer-rectifier which is powered by the three-phase ac essential bus to produce dc power for the command radio. An additional modification on some airplanes** provides an emergency dc bus to power this radio set and some other equipment (see figure 1-12).

Operation of Command Radio

1. Select PRESET position with the selector control (sliding button).

Note

Due to the proximity of the fuel selector switch to the ARC-34 command radio selector control, be certain that the desired control is selected prior to movement.

2. Rotate main control switch to BOTH position and allow approximately two minutes to warm up main and guard receiver units.
3. With channel selector knob, select a channel other than the desired channel. Allow tuning cycle to be completed, then return to desired channel.

Note

Full power for reception and transmission is not available if the first channel selected is used.

4. Adjust volume control for desired audio level.
5. For manual selection of a frequency that is not included in the preset channels, set selector control to MANUAL. The four windows across the top of the panel should open. Turn the four knobs at the top of the panel until the numerals indicating the desired frequency appear in the windows. (The main control switch must be at MAIN or BOTH for manual frequency selection.) This procedure places the set in receive condition. Transmission on the same frequency is obtained by depressing the microphone button.

Note

- The microphone button should be released before changing transmitter frequency. Approximately four seconds should elapse before transmission begins on a new frequency.
- A thermal relay stops the drive motor after 50 to 125 seconds of continuous operation (switching from one channel to another without stopping). If this occurs, place the main control switch to the OFF position and allow for a 30-second cooling period. Then switch to the BOTH position and again select the desired channel.
- Do not attempt to tune the receiver to any frequency below 225 megacycles, as the radio will not operate in this range. If a frequency is set manually below 225 megacycles continuous operation of the channeling drive motor results.

6. To obtain transmission and reception of the guard frequency, move selector control to GUARD.
7. To turn off receiver-transmitter, move main control switch to OFF.

VOR RECEIVER — AN/ARN-14*

The VHF navigation receiving set consists of a control panel on the right-hand console, a course indicator on the instrument panel, and a radio magnetic indicator also on the instrument panel. The dynamotor for this set is powered from the 28-volt dc essential bus. Indicator power is from the 26-volt ac bus.

VOR Navigation Control Panel

The VHF Navigation control panel (figure 4-6) has a power switch with ON and OFF positions, two frequency selector controls, a frequency window, and a volume-control knob. Rotation of the notched outer frequency control wheel selects the frequency from 108 to 136 megacycles. The inner frequency selector is in the form of a rotary switch key and selects the frequency in tenths of megacycles. The frequency selected is shown in the frequency window.

Course Indicator

The course indicator (34, figure 1-4) has a bearing selector knob, a bearing window, a "TO-FROM" window, a course deviation indicator (vertical pointer), a glide-slope indicator (horizontal pointer), and a heading pointer. The bearing selector knob permits selection of a desired bearing, which is displayed in the bearing window. The "TO-FROM" window indicates whether the

*AF 57-770 & on, & airplanes modified by TCTO 1F-102A-540.
**Airplanes modified by TCTO 1F-102-727.

*AF 53-1791 thru 56-1241, -1317 thru -1358, -1369 thru -1392, -1394 & on, unless modified by TCTO 1F-102-719.

bearing selected is a "to" or "from" radial. The course deviation indicator (CDI) shows the position of the airplane in relation to the desired bearing. The glide-slope indicator is used when the pilot is flying an instrument landing system approach. The heading pointer, actuated by the directional indicator (slaved) system, shows the heading of the aircraft relative to the selected bearing. The marker beacon light is illuminated when signals are received by the marker beacon radio. When the CDI is centered, the selected track will be maintained. If the pointer is deflected to the left or right, the airplane's heading is to the left or right the number of degrees indicated on the pointer scale. When radio signals are unreliable or too weak, an "OFF" flag appears at the end of the vertical bar or at the end of the horizontal bar, depending on which system is unreliable.

Radio Magnetic Indicator

The radio magnetic indicator (RMI) (11, figure 1-4) consists of a rotating compass card, actuated by the directional indicator (slaved) system, and a large and small pointer. Information from the flux valve compass transmitter (J-2 or J-4) turns the compass card under a heading arrow at the top of the indicator to indicate the magnetic heading of the airplane. The double-barred pointer is actuated by the VOR receiver, and points to the magnetic bearing of the omni station being received. The smaller pointer is wired in parallel with the larger one and travels with it.

Operation of VOR Receiver

1. Position power switch to ON.
2. Select desired frequency.
3. Adjust volume to desired level.
4. To turn equipment off, position power switch to OFF.

GLIDE-SLOPE RECEIVER — AN/ARN-18*

This set is used in conjunction with the localizer portion of the omni-directional range radio. Visual glide-slope indications are presented on the omni-range receiver course indicator (34, figure 1-4) by means of the horizontal pointer of that instrument. The glide-slope receiver is controlled from the navigation radio control panel (figure 4-6).

Operation of Glide-Slope Receiver

1. Place the power switch on the VHF navigation control panel, in the ON position.
2. Select the localizer frequency to be used.

vor-control panel

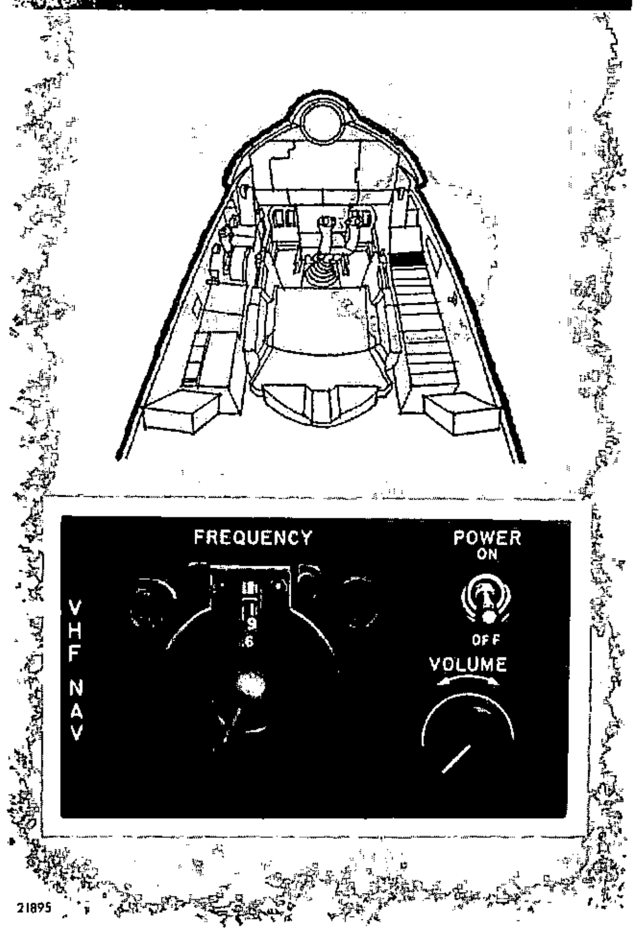


Figure 4-6

3. Observe glide-slope flight characteristics as indicated by the horizontal pointer of the course indicator.
4. To turn set off, place omni-range power switch in OFF position.

TACAN — AN/ARN-21**

This set is designed to operate in conjunction with a specifically constructed surface navigation beacon to form a radio navigation system called Tacan (TACtical Air Navigation). This system enables an equipped airplane to obtain continuous indications of its distance and bearing from any selected Tacan surface beacon located within a line-of-sight distance of approximately 195 nautical miles. The AN/ARN-21 utilizes the radio magnetic

**AF 53-1791 thru 56-1241, -1317 thru -1358, -1369 thru -1392, -1394 & on, unless modified by TCTO 1F-102-719.*

***AF 56-1242 thru -1316, -1359 thru -1368, -1393, & airplanes modified by TCTO 1F-102-719.*

tacan control panel

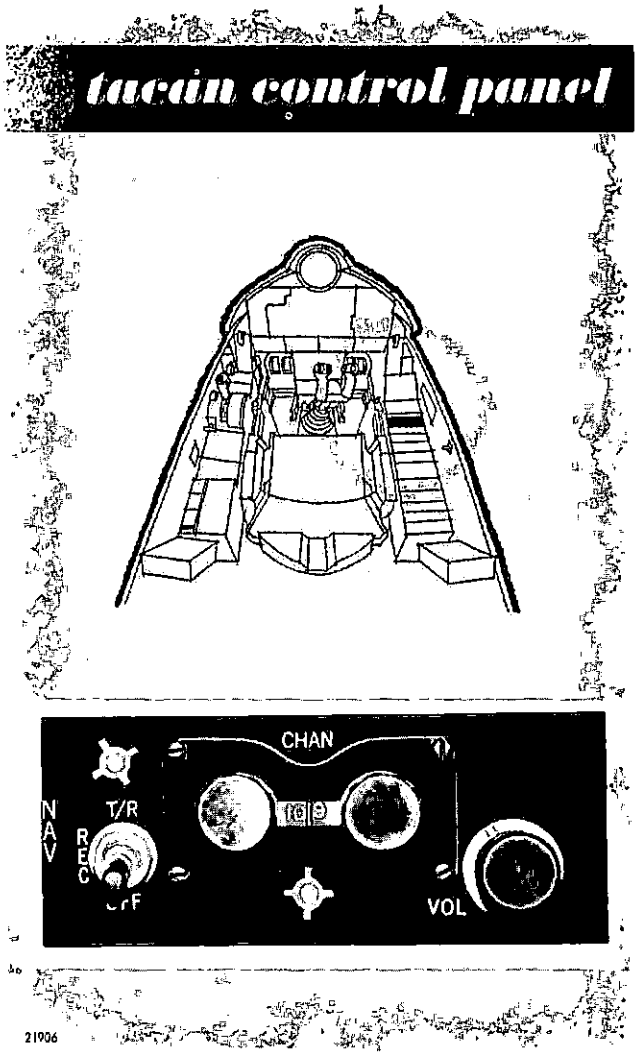


Figure 4-7

indicator, a range indicator, the course deviation indicator (CDI) (vertical pointer) of the course indicator, a control panel, and an instrument selector panel. The AN/ARN-21 transmits interrogation pulses, receives beacon pulses from the Tacan surface beacon and prepares the received information for display on the bearing and distance indicators. The system operates on the following UHF frequency ranges: transmitter—1025 to 1150 megacycles; receiver—962 to 1024 megacycles and 1151 to 1213 megacycles. There are 126 frequency channels, any one of which may be selected by setting the proper controls on the control panel. The equipment can operate at altitudes up to 50,000 feet. However, to prevent the possibility of arcing (flashover) at extreme altitudes, an altitude sensing switch is incorporated which shuts off the equipment automatically at a compartment pressure altitude of 45,000 feet ±2500 feet while climbing or restores operation at approximately 40,000 feet descending. The equipment is powered by the 28-volt dc nonessential bus and the 115-volt ac nonessential bus.

Note

Tacan will be inoperative with loss of either ac or dc generator.

Tacan Control Panel

The Tacan control panel (figure 4-7), located on the right-hand console, has a power switch with OFF, REC, and T/R positions, two channel selector knobs, a channel window and a volume-control knob. With the power switch in the REC position, the distance function of the set is disabled, and only bearing information is available. With the power switch in the T/R position, both bearing and distance information is displayed on the indicators. The left channel selector knob selects the first two figures of the Tacan beacon channel number, and the right channel selector knob selects the third number. The volume control knob is used to adjust the volume of aural identification signals received from the Tacan surface beacon.

Note

The Tacan surface beacon channels range from 00 to 129; however, the AN/ARN-21 is designed to operate only on channels 01 to 126.

Instrument Selector Panel

An instrument selector panel (figure 4-8) containing a two-position switch is located on the right-hand console. The switch controls the CDI of the course indicator. With the switch in TACAN position, the panel illuminates the area labeled TACAN and the CDI responds to AN/ARN-21 functions. With the switch in ILS position, the area surrounding ILS is illuminated, and the CDI responds to localizer signals used in conjunction with the AN/ARN-31 during an instrument landing system approach.

Radio Magnetic Indicator

The radio magnetic indicator (RMI) (11, figure 1-4) includes a rotating compass card and two pointers. The rotating card is actuated by the directional indicator (slaved) system. The pointers which are connected to function as a single unit, are actuated by the receiver portion of the AN/ARN-21. Azimuth signals from the Tacan surface beacon are received by the AN/ARN-21 and relayed to the RMI, causing the pointers to indicate the magnetic bearing of the Tacan surface beacon. With the control switch in the REC position, bearing information may be received even though the transmitter portion of the set is not energized. The set is so designed that when the correct bearing information cannot be determined or the equipment is not functioning satisfactorily, the indicator will "search" or rotate rapidly so that the pilot will be unable to derive unreliable information from the azimuth indicator.

Tacan Range Indicator and Altitude Switch Over-Ride

The Tacan range indicator panel (figure 4-9) is installed on the instrument panel and contains the range indicator and altitude switch over-ride. The range indicator displays the distance in nautical miles between the airplane and the Tacan surface beacon. The numerals in the window are controlled by the range circuits of the AN/ARN-21. These circuits measure the time elapsed between transmissions of the signal and the reception of the response signal from the Tacan surface beacon. The time difference is then converted into digital information, which is displayed on the range indicator in nautical miles. While the indicator is "searching" for the correct range or when the Tacan power switch is in REC position, the rotating numbers are partially covered by a red flag, which warns against reading incorrect distance indications. The altitude switch over-ride on the Tacan range indicator panel is provided to over-ride the automatic shutoff function of the altitude switch. The switch is placarded "Altitude SW Over-Ride" and has positions OFF and ON. Above 45,000 feet, when placed in the ON position, the altitude switch over-ride will restore operation of the Tacan equipment if the altitude switch has shut it off.

CAUTION

The altitude switch over-ride should not be used except in an emergency as arcing may occur in the Tacan equipment when operating above 45,000 feet.

Course Indicator

Signals received by the AN/ARN-21 from the surface beacon are relayed to the CDI (vertical pointer) of the course indicator (34, figure 1-4). Deviation of the airplane course either to the left or right of the transmitting beacon will be indicated by displacement of the CDI. For further information concerning course indicator, refer to "VOR Receiver—AN/ARN-14," this Section.

Operation of an AN/ARN-21

1. Power switch to either REC or T/R.
2. Instrument selector switch to TACAN.
3. Allow approximately 90 seconds warmup time after power is applied to nonessential buses. There is no delay when going from REC to T/R.
4. Select desired beacon channel.
5. Adjust volume to desired level.
6. To turn equipment off, position power switch to OFF.

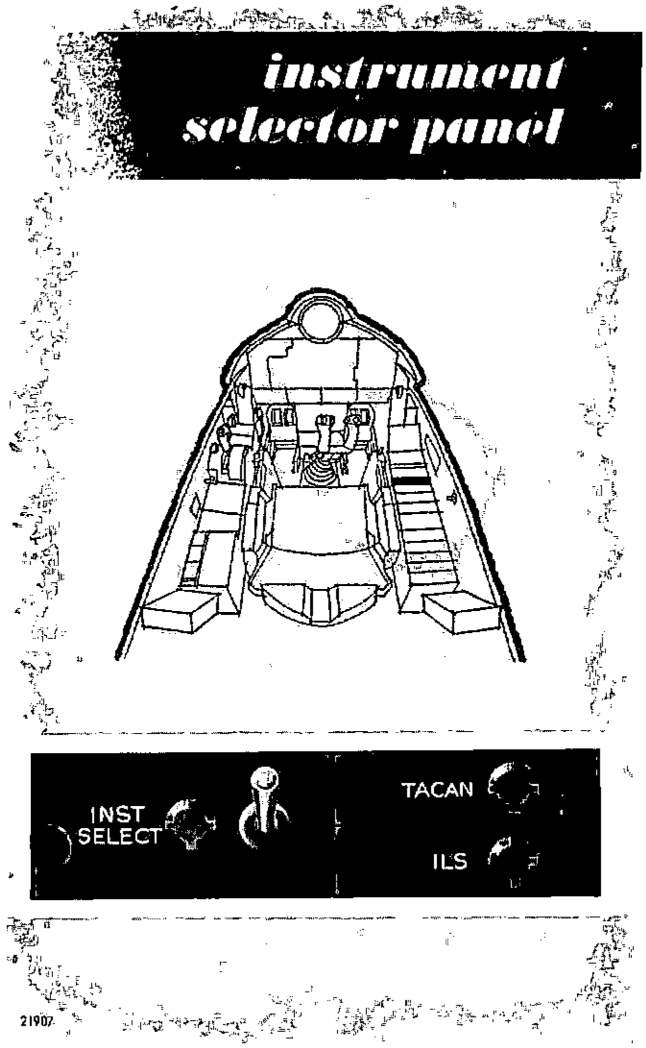


Figure 4-8

ILS RECEIVERS—AN/ARN-31*

This set consists of localizer and glide slope receivers, a control panel and an instrument selector panel. Indications from the AN/ARN-31 are displayed on the course indicator and give both vertical and horizontal guidance to a pilot making an instrument landing system approach. The localizer and glide slope operate 20 fixed frequency channels. The localizer frequencies range from 108.1 to 111.9 and the glide-slope from 329.3 to 335.0. The standard glide-slope frequency is automatically obtained when the desired localizer frequency is selected on the control panel. Two warning "OFF" flags are visible at the right end of the horizontal and at the lower end of the vertical bar on the course indicator whenever the signal level from the selected frequency is too weak to be reliable. The set is powered by the 28-volt dc nonessential bus and the 115-volt ac nonessential bus.

**AF 56-1242 thru -1316, -1359 thru -1368, -1393, 57-770 & on, & airplanes modified by TCTO 1F-102-719.*

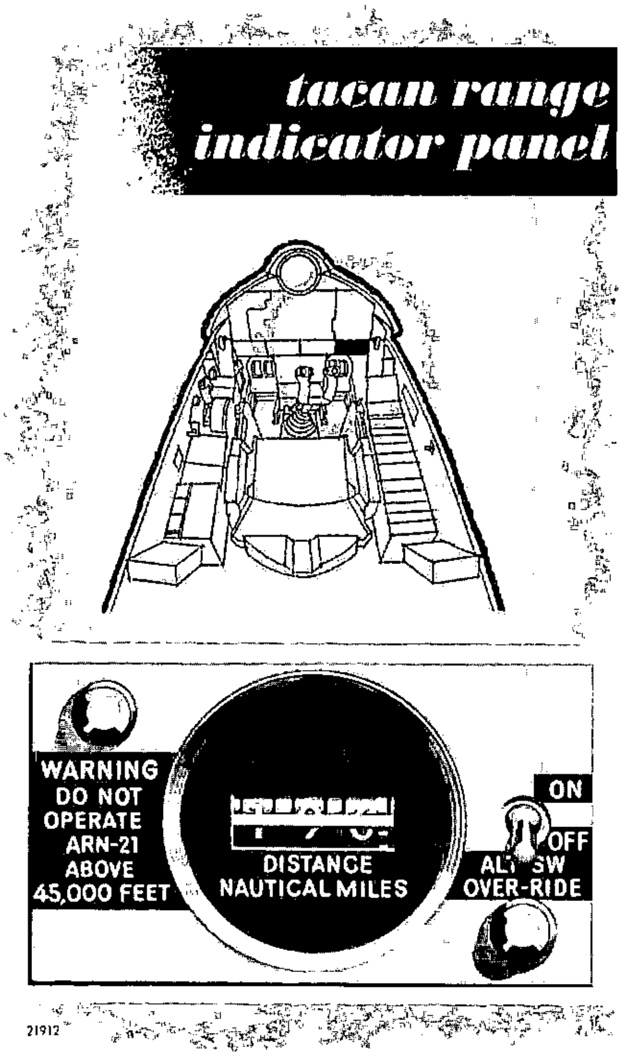


Figure 4-9

ILS Control Panel

The ILS control panel (figure 4-10) located on the right-hand console consists of a power switch, a frequency selector knob, a frequency window, and a volume control knob. Placing the power switch from OFF to POWER puts the set into operation. Turning the frequency selector knob selects any one of 20 localizer frequency channels and automatically obtains the desired glide-slope frequency. The localizer frequency selected is displayed in the window.

Instrument Selector Panel

A two-position switch (figure 4-8), on the right-hand console, controls relays that connect the course indicator to either the ILS receivers or the Tacan receiver. The switch must be in ILS position for operation of the AN/ARN-31 during an instrument landing approach. The instrument selector panel also contains a TACAN position which causes the vertical pointer (CDI) to respond to AN/ARN-21 functions. The instrument switch and relays receive power from the 115-volt ac nonessential bus.

WARNING

During an ILS approach, if disappearance of the glide-slope indicator warning "OFF" flag is delayed beyond normal expectation, it may be an indication that electrical power to the instrument selector relays is lost and the CDI information is being derived from a Tacan station. At the start of an ILS approach where the CDI warning "OFF" flag has disappeared but the glide-slope warning "OFF" flag is still visible, or at a locally prescribed check point, check to determine if there has been power loss to the relays by turning the bearing selector knob a few degrees away from the inbound heading. If the CDI responds to the bearing change, the signal was being received from a Tacan station and not the localizer. If the CDI does not respond to the inbound bearing change, the signal was being received from the localizer. After both warning "OFF" flags have disappeared, a subsequent power loss to the instrument selector relays will be detected by the horizontal needle warning "OFF" flag appearing and the horizontal needle will center itself and remain centered regardless of airplane movement.

Operation of AN/ARN-31

1. Place power switch in POWER position.
2. Place instrument selector switch in ILS position.
3. Select desired frequency.
4. Set is ready for operation when warning flags are no longer visible.
5. To turn set off, place power switch to OFF.

MARKER BEACON RECEIVER—AN/ARN-12 OR AN/ARN-32

Early airplanes* are equipped with AN/ARN-12 receiver and later airplanes** with AN/ARN-32 receiver. Both are fixed tuned receivers and perform the same navigational function. The marker beacon indicator light is on the course indicator. This equipment is in a standby condition at all times that dc power is supplied to the nonessential bus.

IDENTIFICATION RADAR (IFF-SIF)—AN/APX-6A, AN/APX-25

The AN/APX-6A, AN/APX-25 radar sets identify the airplane automatically according to prearranged modes

**AF 53-1791 thru 56-1274, -1317 thru -1331.*
***AF 56-1275 thru -1316, -1332 & on.*

of operation when challenged by suitably equipped ground or airborne units. Interrogation signals are decoded and coded signals sent in return. The identification radar sets also have means for transmitting a special distress signal. The AN/APX-25 identification radar can be set by ground technicians to decode challenges from either of two ground or airborne identification systems: the Mark X IFF, or the selective identification feature (SIF) systems. SIF provides a greater number of possible coded replies than does the Mark X system. When operating in SIF, in addition to the conventional IFF controls used with the Mark X IFF (AN/APX-6A radar installed or AN/APX-25 radar installed, but set for Mark X IFF), it will be necessary to use the coder group control panel placarded "SIF." The master switch on the IFF control panel receives power from 28-volt dc essential bus to operate a relay controlling power from the 115-volt ac essential bus which powers both radars.

IFF Control Panel

The IFF control panel (figure 4-11) is located on the right-hand console and has the IFF master control knob, a plunger button, and three toggle switches. The IFF master control knob is placarded "Master" and has positions EMERGENCY, NORM, LOW, STDBY, and OFF. Placing the IFF control knob to EMERGENCY selects the special distress signal. The OFF position deenergizes the identification radars. STDBY provides for set warmup with primary power on, but the receiver desensitized so that the set will not respond to interrogation signals. NORM position provides for normal set operation, while the LOW position provides a partial receiver sensitivity which allows the set to respond to only very strong interrogation signals. Selection is achieved by turning the IFF master control knob until the desired position appears below the selection arrow marker. To select EMERGENCY it is necessary to depress the plunger button in the lower left-hand corner of the control panel while turning the control knob. This button prevents accidental selection of the emergency position. Normal operating selections are called modes — mode 1, mode 2, and mode 3. Mode 1 is selected by placing the IFF master control knob in the NORM position. Selection of additional modes is accomplished by use of two mode selector toggle switches with positions MODE 2 — OUT, and MODE 3 — OUT.

Note

Combinations of mode selections available are: mode 1 and mode 2, mode 1 and mode 3 and modes 1, 2, and 3. Mode 1 is always used, alone or with any combination of modes.

The third toggle switch is provided to supply a special "identification of position feature." This toggle switch has three positions placarded I/P, OUT, and MIC. It is spring-loaded to OUT from the I/P position, and must be held in the I/P position. When placed in the MIC position, the microphone button must also be held depressed. The identification radar features provided by the incorporation of this switch have been supplied to aid ground and airborne air-controllers with special position identification problems and should be used only as directed.

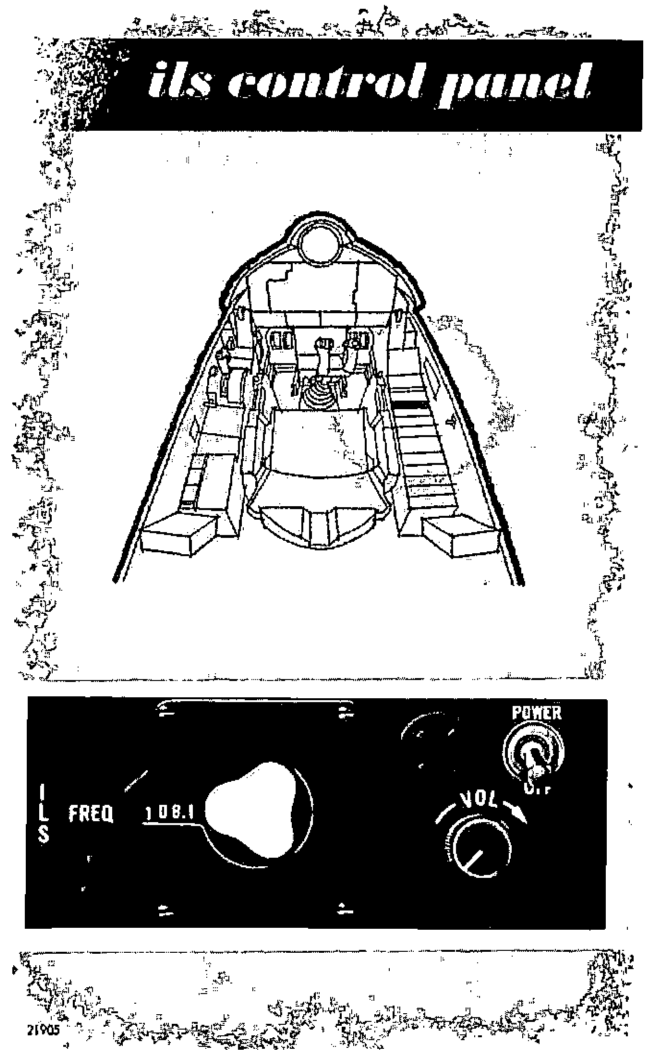


Figure 4-10

SIF Coder Control Panel

The SIF coder control panel (figure 4-11) is located immediately behind the IFF control panel, on the right-hand console, and is used in conjunction with the IFF control panel when the AN/APX-25 identification radar has been set to respond to SIF challenges. The selection of the mode of operation is done by positioning of the controls on the IFF control panel. The coder control panel is used to select the coded response to the selected mode 1, mode 3, or both. Coded SIF responses to a selected mode 2 are set on the ground and are provided automatically when mode 2 is selected on the IFF con-

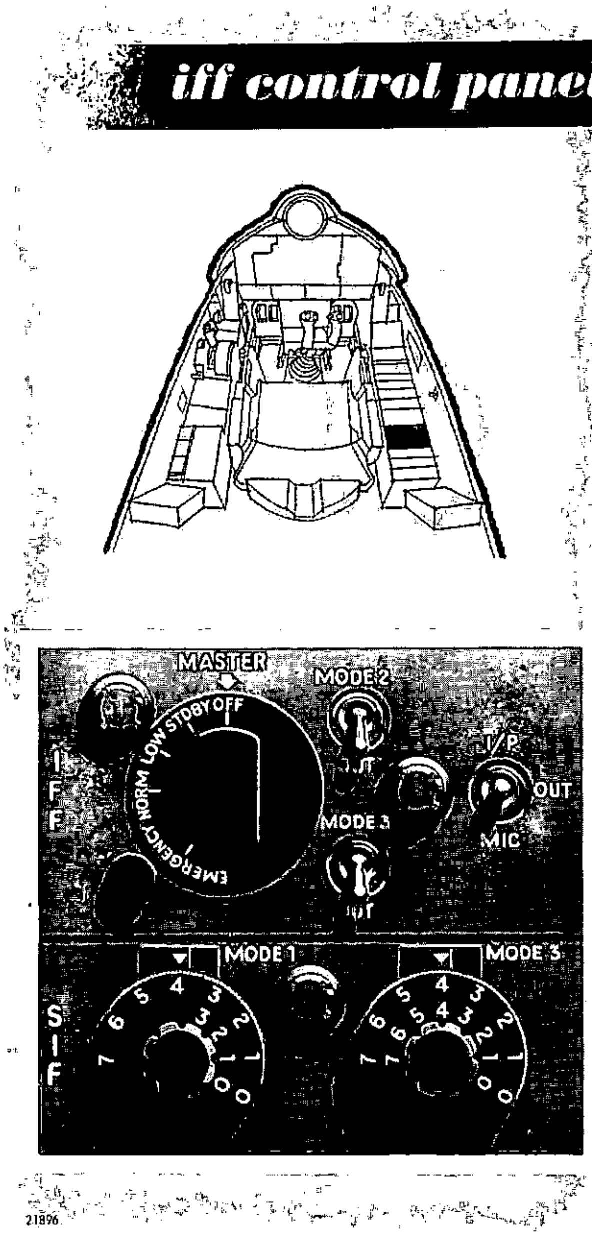


Figure 4-11

trol panel if the AN/APX-25 has been set to respond to SIF interrogations. The coder control panel consists of two dual concentric selector knobs. The left-hand knob controls mode 1 SIF responses coding, the right-hand knob controls coded response to mode 3 SIF interrogations. Combinations of numbers on the inner and outer rings of each of the two selector knobs, aligned with the index markers, set the codes.

Operation of Identification Radar (IFF-SIF)

1. IFF master control knob — As desired.
2. Mode 2 and mode 3 selector switches – OUT, unless otherwise directed.

Note

Set I/P-OUT-MIC switch to OUT unless directed otherwise for identification of position.

3. SIF coder group controls — Set for proper codes as directed.
4. If in distress — Press dial stop plunger button and rotate master switch to EMERGENCY position. Set will then automatically transmit distress signals when interrogated.

LIGHTING EQUIPMENT

EXTERIOR LIGHTING

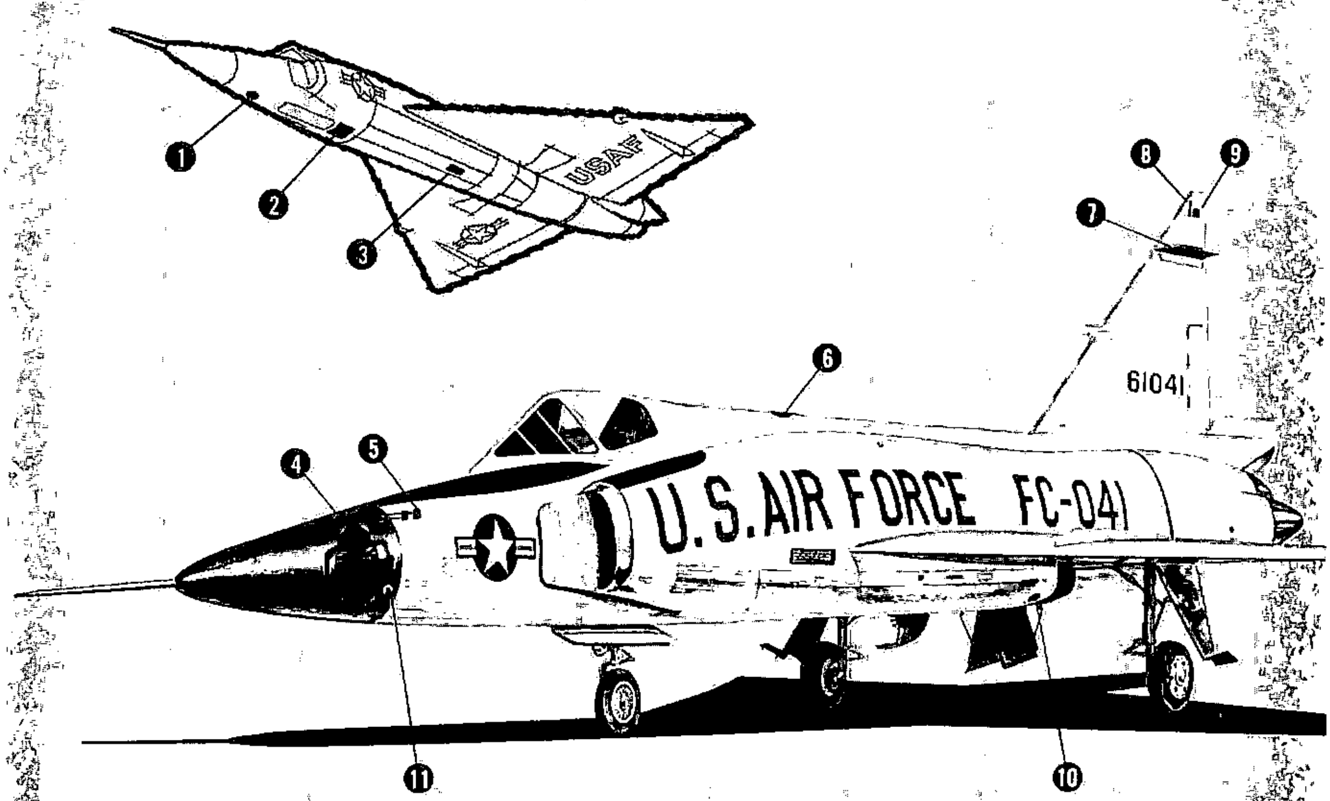
Exterior lighting* consists of 10 position lights, two landing lights and one taxi light. Three groups of position lights (fuselage, wing tip, and tail) are controlled by two switches in the cockpit, which provide for the selection of dim, bright, and flashing circuits. The landing and taxi lights are controlled by a single selector switch. A white light is located on each side of the upper fuselage above the wing root leading edge. A similar white light is located in the right and left armament bay flipper door, below the wing root leading edge. The white lights are not on a flasher circuit. One light is located in each wing tip. The right-hand light has green lenses flush with the upper and lower surfaces of the wing tip. The left-hand light has red lenses flush with upper and lower surfaces. The four tail lights, one white and one amber on each side, are located on the right and left sides of the tail cone fairings, above the inboard trailing edges of the elevons. The upper light on each side is amber and the lower one is white. The wing tip and tail lights are on an automatically cycled flasher circuit. One landing light is mounted on the inner side of each main landing gear fairing door, so that the lights are in duty position when the main landing gear is extended and the doors are fully opened. The taxi light is mounted on the inner side of the nose landing gear fairing door. Neither landing lights nor taxi light can be turned on if the nose landing gear is not down and locked.

Position Light Switches

Two position light switches* (figure 1-21) are located at the bottom right corner of the utility switch panel. All lights are turned on by selecting either STEADY or FLASH position with the left switch, and either BRIGHT or DIM position with the right switch (both must be used, as they are in series). The FLASH position energizes two wing and tail light circuits alternately at the rate of approximately 40 cycles per minute. During one cycle, each circuit is energized and deenergized once. One circuit consists of left and right white tail lights and left (red) and right (green) wing tip lights. The other cir-

**AF 53-1791 thru 56-1429.*

antenna locations



1. Lower IFF Antenna. (AN/APX-6A or AN/APX-25.)
2. Data Link Antenna.
3. Marker Beacon Antenna. (AN/ARN-12 or AN/ARN-32.)
4. Radar Antenna. (MG-10.)
5. Glide Slope Antenna. (AN/ARN-18 or AN/ARN-31.)
6. Upper IFF Antenna. (AN/APX-6A or AN/APX-25.)
7. VHF Antenna (AN/ARN-14) or Localizer of AN/ARN-31.
8. UHF Command Radio Antenna. (AN/ARC-34.)
9. Air-To-Air FIS (AN/APX-27) (PROVISIONS ONLY)
10. Tacan Antenna (AN/ARN-21).
11. Air-To-Air FIS (AN/APX-26) (PROVISIONS ONLY)

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Figure 4-12

cuit consists of left and right amber tail lights. If the flasher fails, the circuit consisting of the white tail lights and the wing tip lights will be energized and will remain on steady. The flasher switch controls the 115-volt single-phase ac essential bus power to all the lights, and also the 28-volt dc essential bus power to the flasher relays and controller. The dimmer switch controls only the 115-volt ac power to the lights.

EXTERIOR LIGHTING

Two retractable anticollision lights on the fuselage, two wing tip lights, two landing lights and one taxi light provide exterior lighting* for both inflight and ground operation. One of the red anticollision lights is located on the upper centerline of the fuselage aft of the cockpit and the other on the lower centerline of the fuselage aft of the main wheel well. The anticollision lights extend and illuminate whenever the position lights switch is in NAVIGATION position. They retract and turn off whenever the position lights switch is in FORMATION position or automatically when 335 KIAS is exceeded. On some airplanes**, to provide for more effective utilization of the rotating beacon lights as anticollision lights, the automatic extension and retraction feature has

**AF 53-1801, -1802, -1809, 56-987 and 56-1430 through 57-909.*
***Airplanes modified by TCTO 1F-102-890.*

been eliminated. The lights will extend in the NAVIGATION position or retract in the FORMATION positions respectively. With the anticollision lights retracted, a white light portion illuminates and provides lighting on both the upper and lower sides of the fuselage.

WARNING

Airplanes in the rotating beacon configuration do not provide adequate exterior lighting to meet night formation requirements. Avoid night formation flight on a lead F/TF-102A airplane with the rotating beacon configuration if mission requirements permit.

One position light is located on each wing tip, a green light on the right side and a red light on the left side. One landing light is mounted on the inner side of each main landing gear fairing door, so that the lights are in duty position when the main landing gear is extended and the doors are fully opened. The taxi light is mounted on the inner side of the nose landing gear fairing door. Neither landing lights nor taxi light can be turned on if the nose landing gear is not down and locked. The landing and taxi lights are controlled by a single selector switch.

Position Light Switches

Two position light switches* (figure 1-21) are located at the bottom right corner of the utility switch panel. The left switch has three positions: NAVIGATION, OFF, and FORMATION. In NAVIGATION position the red anticollision lights are turned on and extended, and the wing tip lights illuminate steadily. The right BRIGHT-DIM switch has no effect on the lights with the power switch in NAVIGATION position. In FORMATION position the anticollision lights are turned off and retracted, the wing tip lights illuminate steadily, and the white portion of the anticollision lights illuminate at the top and bottom of the fuselage. The lights may be dimmed in this position. There is no flasher position provided. The NAVIGATION position controls 28-volt dc nonessential bus power to the anticollision lights and the 115-volt ac essential bus power to the transformer for wing-tip lights. The FORMATION position controls 115-volt ac essential bus power to the transformer for wing tip lights and white fuselage lights.

Note

The rotating anticollision lights should be turned OFF during flight through conditions of reduced visibility where the pilot could experience vertigo as a result of the rotating reflections of the lights against the clouds. In addition, the lights would be ineffective as anticollision lights during these conditions since they could not be observed by pilots of other airplanes.

Landing and Taxi Light Switch

The landing and taxi light switch (figure 1-24) is located on the landing gear control panel, above the landing gear handle and has TAXI LIGHTS, OFF, and LANDING LIGHTS positions. The switch controls power from the 28-volt dc nonessential bus, but is inoperative if the nose wheel is not down and locked.

INTERIOR LIGHTING

Interior lighting equipment includes the edge-lighting for the instrument panels, switch panels and consoles, the thunderstorm lights, cockpit floodlights, and the standby compass light. Edge-lighting of instrument panels, switch panels, and console panels is accomplished by small lights set into the panels. The plastic panel facing has an opaque outer layer and a translucent inner layer. Light from the bulbs in the panels is conducted by the translucent inner layer to the edges of instruments and the edges of holes through which switches and controls protrude, thereby illuminating instruments and outlining switches and controls. Wherever lettering is cut in the opaque surface material, to identify or mark the positions of a switch or control, light shines through from the translucent layer and illuminates the lettering. Red floodlights are directed at the right and left sides of the instrument panels and the tops of the consoles. White thunderstorm lights provide brilliant illumination of the cockpit to counteract the blinding effects of lighting on the pilot's vision. Intensity of lighting in ac circuits is controlled by variable transformers, known as "powerstats." Intensity of dc lighting is controlled by a rheostat. Powerstat and rheostat knobs are identical in appearance.

Flight Instrument Panel Lights Rheostat

The flight instrument panel rheostat (33, figure 1-4) is located at the top center of the lighting control panel. Turning the knob clockwise from OFF to BRT controls the intensity of the fifteen panel lights that illuminate the flight instrument group in the center of the instrument panel, and the electrical control panel on some airplanes and also the standby compass light, when the compass light switch is in the ON position. Power to this rheostat comes from the 28-volt dc essential bus.

**AF 56-1430 & on.*

Forward Panel Lights Powerstat

A powerstat placarded "Fwd Panels" at the top right corner of the lighting control panel (33, figure 1-4) controls the intensity of the lighting for all of the engine instrument group on the instrument panel, and also for the armament control panel, the antenna scan control panel, the target altitude panel, the two small radar scope control panels at either side of the radar scope, the far left side of the instrument panel, the utility switch panel, and the lighting control panel. Turning the knob clockwise from OFF to BRT controls the intensity of the lighting on these panels. Power to this powerstat comes from the 115-volt ac essential bus.

Console Lights Powerstat

The console lights powerstat, located at the lower right corner of the lighting control panel (33, figure 1-4), is placarded "Console." Turning the knob clockwise from OFF to BRT controls the intensity of the lights on both consoles, and also the lights on the upright panels at the forward ends of the consoles. Power to this powerstat comes from the 115-volt ac essential bus.

Cockpit Floodlights Powerstats

Two floodlight powerstats are located at the left side of the lighting control panel (33, figure 1-4). One placarded "Inst Flood" controls intensity of the four red floodlights in two reflection shields aft of the main instrument panel. The other, placarded "Console Flood" controls the intensity of red floodlights above each console. Power to these powerstats comes from the 115-volt ac essential bus.

Thunderstorm Lights Switch

A switch placarded "Storm Lights" (figure 1-4) above the canopy unlocked warning light, at the extreme right end of the instrument panel, has ON and OFF positions and controls five white thunderstorm lights, located over the two consoles and aft of the instrument panel. When the switch is in ON position, the master warning dimming relay is cut out, causing the master warning light, master warning light panel, landing gear unsafe warning light, canopy unlocked warning light, hydraulic pressure low warning light, fire and overheat warning light, and takeoff trim light to be at full brilliance while the thunderstorm lights are on. On some airplanes* the canopy unlocked warning light is removed from the warning light dimming circuit and will appear at full brilliance any time the canopy is not locked. The thunderstorm light switch controls power from the 28-volt dc nonessential bus.

**Airplanes modified by TCTO 1F-102-773.*

LIQUID OXYGEN SYSTEM

The major components of the liquid oxygen system are a storage and converter unit, an oxygen pressure gage, a liquid oxygen content gage, an external filler valve, and an oxygen regulator. The storage and converter unit includes a five-liter (one liter equals about one quart) insulated storage container and a converter, which converts the liquid oxygen to gaseous oxygen and then supplies it to the oxygen regulator. The liquid oxygen content gage indicates, in liters, the supply of liquid oxygen in the storage container. The oxygen regulator is located on the left console on some airplanes and in the seat cushion-survival kit on other airplanes*. Gaseous oxygen is delivered from the converter to the regulator at a fairly constant pressure of about 70 psi, and the regulator, in turn, supplies the pilot with breathing oxygen. After a 24-hour boil-off period, the oxygen system supply allows for a maximum of 22.6 hours at a minimum demand and a minimum of 3.7 hours at a maximum demand. Oxygen duration at various altitudes is shown in figure 4-13. At sea level and with average temperature, a full supply of liquid oxygen dissipates through a relief valve in about five days, when the airplane remains on the ground and no demands are made on the system. The liquid oxygen system is serviced through a single-point filler valve located within an access door on the left side of the fuselage below the cockpit.

LIQUID OXYGEN REGULATOR

A combination pressure-breathing, diluter-demand MD-1 oxygen regulator (figure 4-16) is mounted on the left console. Gaseous oxygen is supplied to the regulator at approximately 70 psi. The regulator reduces the oxygen pressure, mixes air with oxygen in varying amounts depending on altitude, temperature and pilot demand, and delivers it through a flexible tube to the pilot's mask or the K-1 helmet. At high altitudes, the regulator supplies positive-pressure breathing. Control of the oxygen system is accomplished by the use of three levers: the supply lever, the diluter lever, and the emergency toggle lever. The pilot receives an indication of system operation from the flow indicator and oxygen pressure gage located on the oxygen regulator panel.

Diluter Lever

The diluter lever, aft center on the regulator panel, has two positions, NORMAL and 100%. With the lever at NORMAL, the regulator mixes air with oxygen in varying amounts, according to altitude, and delivers it through a flexible tube to the pilot's mask or helmet. With the lever at 100%, the regulator delivers 100% oxygen regardless of altitude.

**AF 56-1275 thru -1316, -1332 & on, & airplanes modified by TCTO 1F-102-642.*

liquid oxygen duration chart-hours

USING PRESSURE BREATHING OXYGEN MASK TYPE MS22001 (A-13A)

CABIN ALTITUDE—FEET	5	4	3	2	1	BELOW 1
35,000 AND ABOVE	31.4	25.2	18.9	12.6	6.3	EMERGENCY DESCEND TO ALTITUDE NOT REQUIRING OXYGEN
	31.4	25.2	18.9	12.6	6.3	
30,000	22.7	18.1	13.6	9.0	4.5	
	23.3	18.7	14.0	9.3	4.7	
25,000	17.5	14.0	10.5	7.0	3.5	
	22.0	17.6	13.2	8.8	4.4	
20,000	13.3	10.7	8.0	5.3	2.7	
	25.0	20.0	15.0	10.0	5.0	
15,000	10.7	8.6	6.4	4.3	2.2	
	30.2	24.2	18.1	12.1	6.0	
10,000	8.6	6.9	5.2	3.4	1.7	
	30.2	24.2	18.1	12.1	6.0	
5,000	6.8	5.4	4.1	2.7	1.4	
	30.2	24.2	18.1	12.1	6.0	
SL	5.5	4.4	3.3	2.2	1.1	
	30.2	24.2	18.1	12.1	6.0	

LIQUID CONTENT—LITERS

- FIGURES ON GRAY INDICATE DILUTER LEVER—NORMAL OXYGEN
- FIGURES ON WHITE INDICATE DILUTER LEVER—100% OXYGEN
- ON AIRPLANES NOT EQUIPPED WITH SURVIVAL KIT, EMERGENCY TOGGLE SET AT EMERGENCY.

USING HIGH ALTITUDE PRESSURE COVERALLS MC-3 AND PRESSURE HELMET TYPE MA-2

CABIN ALTITUDE—FEET	5	4	3	2	1	BELOW 1
30,000 AND ABOVE	12.8	10.2	7.7	5.1	2.6	EMERGENCY DESCEND TO ALTITUDE NOT REQUIRING OXYGEN
25,000	9.8	7.8	5.9	3.9	2.0	
20,000	7.6	6.1	4.6	3.0	1.5	
15,000	6.0	4.8	3.6	2.4	1.2	
10,000	4.8	3.8	2.9	1.9	1.0	
5,000	3.9	3.1	2.3	1.6	0.8	
SL	3.2	2.6	1.9	1.3	0.7	

LIQUID CONTENT—LITERS

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Figure 4-13

Emergency Toggle Lever

The emergency toggle lever, aft left on the regulator panel, has two placarded positions, EMERGENCY and TEST, and an unmarked center (neutral) position. The toggle lever should remain in the neutral position at all times, unless an unscheduled pressure increase is required. Moving the toggle lever to EMERGENCY provides continuous positive pressure to the mask for emergency use. When the toggle lever is moved to TEST, it provides positive pressure to test the mask for leaks.

CAUTION

When positive pressures are required, it is mandatory that the oxygen mask be well fitted to the face. Unless special precautions are taken to insure against leakage, continued use of positive pressure under these conditions will result in the rapid depletion of the oxygen supply and extremely cold oxygen flowing to the mask.

Supply Lever

The supply lever, aft right on the regulator panel, is safety-wired to the ON position. This lever has ON-OFF positions and controls oxygen pressure to the regulator accordingly.

Note

If the safety wire is broken there should not be concern as long as the lever is at ON.

Pressure Gage and Flow Indicator

A pressure gage and a flow indicator are located on the oxygen regulator panel. The pressure gage shows gaseous oxygen supply pressure (pressure being furnished to the oxygen regulator inlet). The flow indicator consists of an oblong opening on the face of the regulator panel and shows black and white alternately during the breathing cycle.

Liquid Oxygen Content Gage

A liquid oxygen content gage is located just aft of the oxygen regulator panel. The gage provides an indication of the content of the storage container and is calibrated in liters from 0 to 5.

Note

The liquid oxygen content gage should read between 4 and 4½ liters when the system is fully charged. Do not be alarmed that the gage does not read five liters, since it is impossible to charge the liquid oxygen converter to five liters.

Liquid Oxygen System Preflight Check

Before takeoff the oxygen system should be checked as follows:

Note

This test procedure is applicable only for an initial preflight check of the system. Inflight tests or repeated tests made within short periods may produce false or misleading indications.

1. Attach oxygen hose as outlined in figure 4-15.
2. Check oxygen pressure gage at 70 to 110 psi.
3. Check liquid content gage at three liters minimum with partial pressure suit or two liters minimum without partial pressure suit.
4. Check supply lever ON.
5. Check oxygen regulator with the diluter valve first at NORMAL and then at 100% as follows: Remove mask and blow gently into the end of the regulator hose as during normal exhalation. There should be resistance to blowing. Little or no resistance to blowing indicates a leak or faulty operation.
6. With the oxygen mask connected to the regulator and the diluter lever at 100%, breathe normally into the mask and conduct the following checks:
   a. Observe flow indicator for proper operation.
   b. place emergency lever to EMERGENCY. A positive pressure should be supplied to the mask. Return emergency lever to center (neutral) position.
   c. Hold emergency lever in TEST position. A positive pressure should result within the mask. Hold breath to determine whether there is leakage around mask. Release emergency lever; positive pressure should cease.
7. Retain diluter lever in 100% position or return diluter lever to NORMAL as required.

Normal Operation of Liquid Oxygen System

1. Before each flight, be sure oxygen pressure gage reads at least 70 psi and liquid content gage shows a minimum of three liters with partial pressure suit or two liters without partial pressure suit. If content is below this minimum, have oxygen system serviced before takeoff.
2. See that oxygen supply lever is safety-wired in ON position.
3. See that diluter lever is at 100% or NORMAL position as required.

Note

- Above 30,000 feet, a vibration or wheezing sound may sometimes be noted in the mask. This noise is a normal characteristic of regulator operation and may be overlooked.
- Steady deep breathing will cause the quantity gage indicator to momentarily drop toward zero. This is normal and does not mean the oxygen supply is depleted.

Emergency Operation of Liquid Oxygen System

WARNING

A partial pressure suit should be worn for all flights above 45,000 feet ambient altitude.

If symptoms of hypoxia develop or if smoke or fumes enter the cockpit, proceed as follows:

1. If operating on NORMAL, move diluter lever to 100% position.
2. Push emergency lever forward to EMERGENCY position.
3. If oxygen regulator becomes inoperative, actuate emergency oxygen bailout bottle (which contains about a six-minute oxygen supply). Descend to a cockpit altitude below 10,000 feet as soon as possible.

LIQUID OXYGEN REGULATOR

A pressure breathing oxygen regulator* is mounted in the aft portion of the survival kit in the ejection seat. Gaseous oxygen is supplied to the regulator during normal operation from the oxygen converter at approximately 70 psi or during emergency operation from the bailout oxygen supply in the survival kit (refer to SURVIVAL KIT Section I). Bailout bottle pressure is approximately 1800 psi when fully charged and it is therefore necessary to reduce this pressure by means of a restrictor prior to delivery to the regulator. Two control units in the regulator, one for partial pressure suit capstan pressure and one for breathing and chest bladder pressure, regulate and deliver 100% oxygen to the pilot's mask or K-1 helmet and pressure suit. Using the airplane oxygen supply the regulator supplies oxygen under increasing pressure as altitude increases. The regulator will not function as a diluter (will not mix the oxygen with air) and therefore delivers 100% oxygen at all times. When using an oxygen mask instead of the partial pressure suit and helmet, it is necessary to use an adapter to reduce the amount of oxygen pressure between the regulator and the mask. The oxygen system is controlled by a single switch on the oxygen control panel.

Liquid Oxygen Control Panel

The liquid oxygen control panel* is located on the left-hand console and is placarded "Oxygen Supply." The panel contains a supply switch, a liquid oxygen content gage, and a pressure gage. The supply switch has ON-OFF positions and controls the flow of oxygen from the airplane supply to the oxygen regulator. The liquid oxygen content gage provides an indication of the content of the storage container and is calibrated in liters from 0 to 5.

Note

The liquid oxygen content gage should read between 4 and 4½ liters when the system is fully charged. Do not be alarmed that the gage does not read five liters as it is impossible to charge the liquid oxygen converter to five liters.

The pressure gage shows gaseous oxygen pressure from 0 to 500 psi. When oxygen is being used from the system, the pressure gage will normally indicate from 70 to 80 psi. However, under static conditions and on a hot day the gage may indicate as high as 110 psi.

Press-to-Test Button

The oxygen system press-to-test button* is located on the forward edge of the survival kit. Depressing the button will provide positive pressure to the oxygen mask or pressure helmet to determine that the system is operating satisfactorily prior to takeoff. The press-to-test button is the only method of checking oxygen (other than decrease of liquid content and positive flow of oxygen through the system) as there is no blinker or flow indicator installed.

Oxygen Mask Adapter

An adapter* is provided with the survival kit and is to be used when the A-13 type oxygen mask is worn in lieu of the partial pressure helmet. The adapter consists of two separate units. One is an electrical plug which is to be plugged into the mask defog and communications leads in the personal equipment lead bundle on the survival kit. This unit will provide for a satisfactory receptacle for communications lead from the protective helmet and oxygen mask. The second unit is an adapter for the oxygen lead. This adapter provides a quick-disconnect connection for the oxygen mask hose and also serves as a pressure restrictor to reduce the oxygen pressure from the regulator to the mask.

Liquid Oxygen System Preflight Check

Before takeoff the oxygen system should be checked* as follows:

**AF 56-1275 thru -1316, -1332 & on, & airplanes modified by TCTO 1F-602-642.*

1. Before entering cockpit, check bailout bottle pressure gage (located in survival kit) at 1800 psi.
2. Connect oxygen mask adapter to end of oxygen hose in personal equipment lead bundle if oxygen mask is to be worn instead of partial pressure helmet.
3. Attach oxygen hose as outlined in figure 4-15.
4. Check oxygen pressure gage at 70 to 110 psi.
5. Check liquid content gage at three liters minimum with partial pressure suit or two liters minimum without partial pressure suit.
6. Check supply switch ON.
7. With the oxygen mask connected breathe normally into the mask. As a slight positive pressure is supplied at all times there should be no resistance to breathing.
8. If a partial pressure suit is worn, depress the press-to-test button on the front of the survival kit. A definite positive pressure should result within the face plate. Hold breath to determine whether there is leakage around face plate. Release the press-to-test button. Check for normal breathing.
9. If an A-13A oxygen mask is worn, depress the press-to-test button. Release button as soon as a buildup is felt.

Normal Operation of Liquid Oxygen System*

1. Perform oxygen system preflight check as outlined above prior to each flight. After completion of preflight check, oxygen system is ready for normal usage.
2. Oxygen supply switch ON after fastening mask or faceplate.
3. After landing when oxygen is no longer desired, turn oxygen supply switch OFF before opening mask or faceplate.

WARNING

The oxygen supply switch should neither be turned ON until immediately after the mask or faceplate is closed nor turned OFF until immediately before opening of the mask or faceplate. This will prevent a rapid depletion of the oxygen supply and also prevent low temperatures from damaging various components of the liquid oxygen system and causing personal injury.

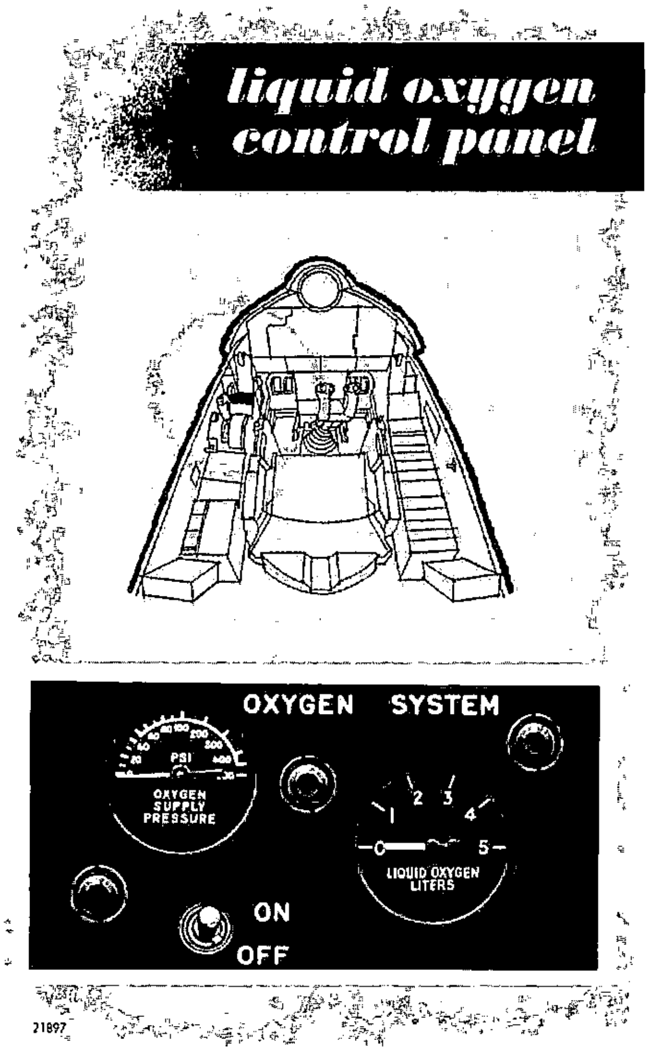


Figure 4-14

Emergency Operation of Liquid Oxygen System*

1. If symptoms of hypoxia develop, check oxygen hose connections and check supply switch ON.
2. If the ship's oxygen system is depleted or not supplying oxygen, actuate the bailout supply by pulling the green knob in the personal equipment lead bundle. Descend to a cockpit altitude below 10,000 feet within 12 minutes.

WARNING

Remove oxygen mask or faceplate when emergency oxygen supply is depleted.

AF 56-1275 thru -1316, -1332 & on, & airplanes modified by TCTO 1F-102-642.

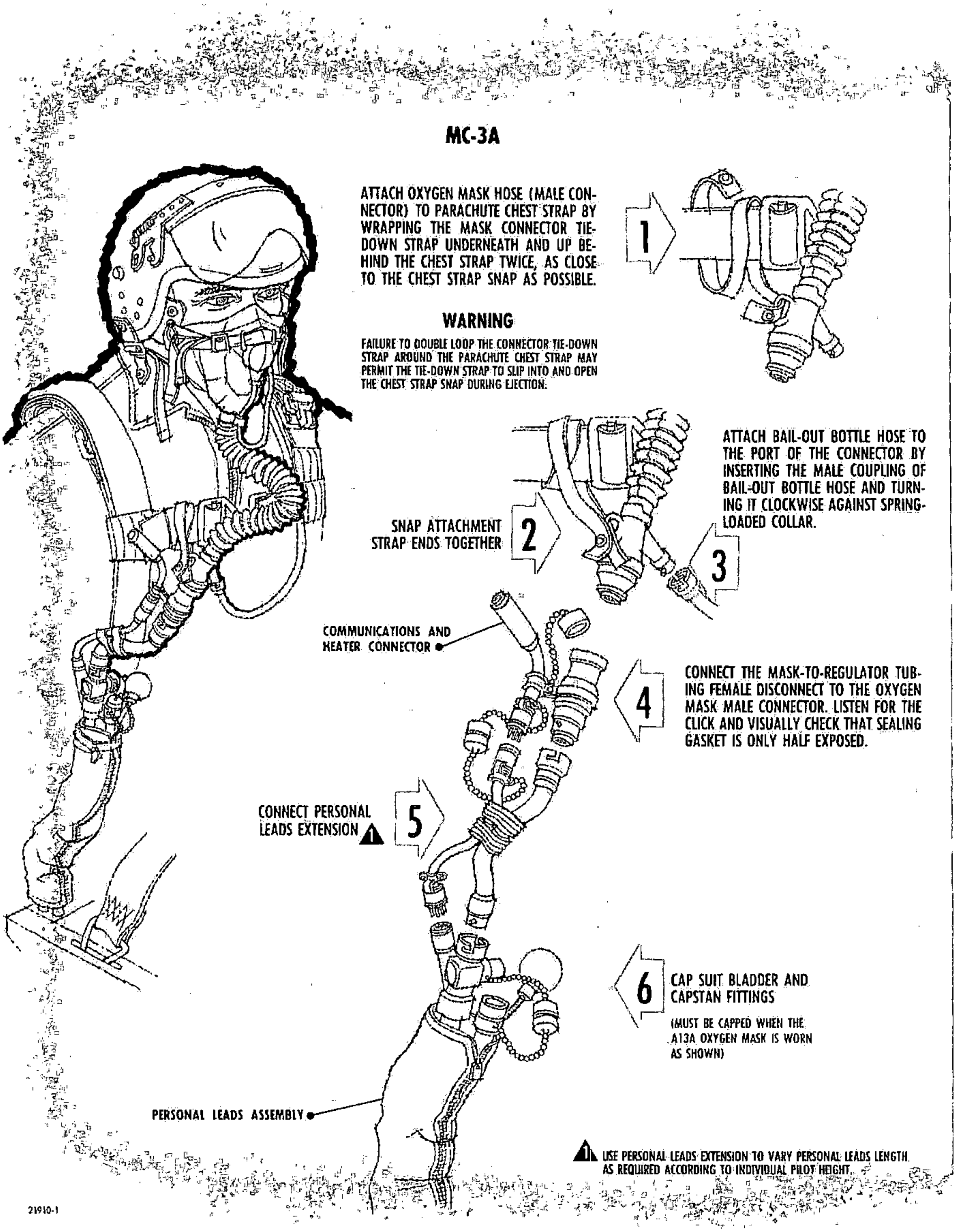
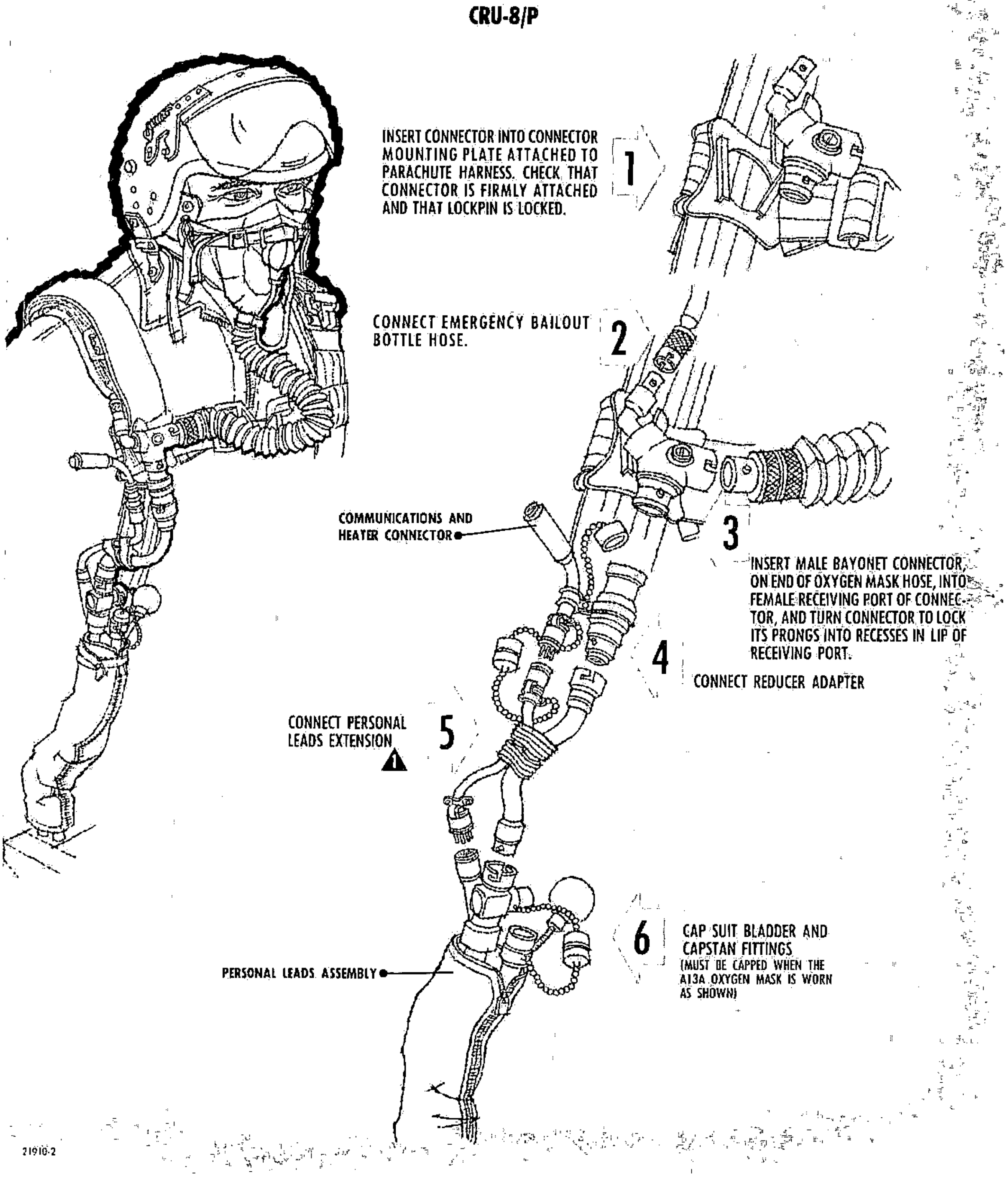

MC-3A
1
ATTACH OXYGEN MASK HOSE (MALE CONNECTOR) TO PARACHUTE CHEST STRAP BY WRAPPING THE MASK CONNECTOR TIE-DOWN STRAP UNDERNEATH AND UP BEHIND THE CHEST STRAP TWICE, AS CLOSE TO THE CHEST STRAP SNAP AS POSSIBLE.
WARNING
FAILURE TO DOUBLE LOOP THE CONNECTOR TIE-DOWN STRAP AROUND THE PARACHUTE CHEST STRAP MAY PERMIT THE TIE-DOWN STRAP TO SLIP INTO AND OPEN THE CHEST STRAP SNAP DURING EJECTION.
2
SNAP ATTACHMENT STRAP ENDS TOGETHER
3
ATTACH BAIL-OUT BOTTLE HOSE TO THE PORT OF THE CONNECTOR BY INSERTING THE MALE COUPLING OF BAIL-OUT BOTTLE HOSE AND TURNING IT CLOCKWISE AGAINST SPRING-LOADED COLLAR.
COMMUNICATIONS AND HEATER CONNECTOR
4
CONNECT THE MASK-TO-REGULATOR TUBING FEMALE DISCONNECT TO THE OXYGEN MASK MALE CONNECTOR. LISTEN FOR THE CLICK AND VISUALLY CHECK THAT SEALING GASKET IS ONLY HALF EXPOSED.
5
CONNECT PERSONAL LEADS EXTENSION 1
6
CAP SUIT BLADDER AND CAPSTAN FITTINGS
(MUST BE CAPPED WHEN THE A13A OXYGEN MASK IS WORN AS SHOWN)
PERSONAL LEADS ASSEMBLY
1 USE PERSONAL LEADS EXTENSION TO VARY PERSONAL LEADS LENGTH AS REQUIRED ACCORDING TO INDIVIDUAL PILOT HEIGHT.
21910-1

Figure 4-15

oxygen mask connection



3. If wearing an A-13A oxygen mask and cabin pressure is lost, an immediate descent to 30,000 feet or below is mandatory.

WARNING

A partial pressure suit should be worn for all flights above 45,000 feet ambient altitude.

AUTOMATIC FLIGHT CONTROL SYSTEM

The automatic flight control system (AFCS) consisting of analog computing equipment which, when coupled with the airplane damping system and the MG-10A fire control system, provides automatic flight control of the airplane in three modes: pilot assist, attack, and landing approach. Command signals from the AFCS, combined with pitch-rate, roll-rate, yaw-rate, and effective elevon position signals deflect the control surfaces, through hydraulic actuators, to steer the desired course. The airplane damping system must be engaged and the MG-10A master switch must be in STBY or ON before automatic flight is possible. During all automatic modes of operation, the control stick follows the motion of the control surfaces. To prevent application of excessive aileron, limit switches are incorporated to disengage the AFCS switch and the pitch damper switch when aileron travel exceeds 2.5°. Pitch g limiting is provided to prevent airplane acceleration limits from being exceeded and to prevent subjecting the pilot to an uncomfortable number of g's in all modes of operation. Refer to Section V for AFCS limitations.

PITCH G LIMITER

Airplanes having AFCS are equipped with a pitch g limiter which prevents the automatic flight control system from subjecting the airplane to excessive pitch g forces. The preset limits are 4½ g's positive and 1½ g's negative or a positive pitch rate of 45° per second and negative pitch rate of 15° per second. If the preset limits are reached, the pitch g limiter removes electrical power from the solenoid-held switches of the automatic flight control system (all modes) and the pitch damper circuit. The yaw damper remains engaged to aid the pilot in maintaining coordinated flight after the automatic flight control system is deactivated. A preflight check of the limiter should be made prior to all flights by means of a test switch on the utility switch panel. This check may also be performed in flight. Power to the pitch g limiter originates from the 115-volt ac essential bus.

PILOT ASSIST MODE

The pilot assist mode relieves the pilot of routine steering tasks by performing conventional autopilot functions such as: (1) altitude or pitch attitude hold, and (2) heading or bank attitude hold. Primary inputs to AFCS in this mode are from the MG-10A vertical gyro, the airplane directional indicator, and ambient air pressure signals. The pilot may select whether he wishes to hold pitch attitude or altitude by positioning a switch on the AFCS control panel. Pitch attitude will be maintained with this switch in the OFF position. Pitch attitude is held by mixing signals from the vertical gyro and a pitch memory servo which registers the attitude at the time of engaging. If altitude hold is selected, a barometric altitude control unit is engaged. This unit produces an error signal proportional to ambient air pressure changes which is used to hold the altitude as it was when the unit was engaged. Altitude hold function utilizes the airplane static pressure sensing system, and, accordingly, the static pressure sensing errors will be reflected in the form of pitch changes while operating in the transonic region. After passing through the transonic zone, pitch changes will continue as altitude hold seeks the altitude held at time of engagement. Heading hold function may be selected by means of a switch on the AFCS panel. If the airplane's wings are within 5° of level flight when AFCS and heading hold are engaged, signals from the airplane compass and heading memory servo maintain the heading. If the bank angle exceeds 5° and heading hold is engaged, or any time the heading hold is off, the vertical gyro and the bank memory servo combine to produce an error signal to hold the bank attitude. The heading hold switch must be engaged at any time heading hold is desired. Heading hold errors will be evident as a result of airplane compass (J-4) precession. Large changes of altitude, pitch attitude, bank attitude, or heading may be made manually by momentary release of AFCS by using a momentary interrupt trigger on the control stick. However, if altitude hold has been selected, depression of the trigger will return this switch to OFF. Small changes of altitude, and bank attitude (if prevailing bank attitude exceeds 5°) may be made by using the elevon trim button to "beep" in signals to the AFCS. The heading hold switch must be OFF to maintain bank angles of less than 5°. When beep trimming in bank attitude, if the trim button is released when the wings are within 5° of level flight and the heading hold switch is engaged, the airplane will return to level flight attitude. If altitude hold has been selected, trimming in pitch will return this switch to OFF. When the trim button is used, the pitch and roll references will change at a constant rate as long as the button is displaced. With AFCS engaged, the pilot assist mode will function until the radar is locked on or the AILAS is engaged. When engaging AFCS, if the system does not function properly, monitor circuits prevent engaging and return the switch to OFF.

ATTACK MODE

The AFCS attack mode automatically steers the airplane on the computed lead collision course for rocket or missile firing. With AFCS in operation, the radar attack is performed automatically after the desired armament is selected and the radar is locked on. The automatic run continues until armament firing occurs unless AFCS is manually disengaged by the pilot or a collision warning signal automatically disengages the attack mode. Loss of lock-on or appearance of the pullout signal on the radar scope returns AFCS to pilot assist (attitude hold) mode. The preselected armament will fire automatically during all automatic attack runs. During automatic operation, electronic limiting provides g limits of +3.3 g to +5. g and limits the command voltage to a value within these g limits. In the attack mode, the primary inputs to AFCS are the same as those to the steering dot on the pilot's scope. AFCS translates the azimuth and elevation steering signals into requests for control surface movement. Control surfaces are deflected to steer the airplane toward the correct lead collision course at a rate proportional to the off-course error signal. Thus, in the case of large steering error, the airplane turns rapidly initially, then as the error decreases, the rate of turn decreases proportionally. During the attack phase, the elevon trim button is inoperative and cannot be used to beep or feed trim signals to vary the airplane attitude.

AUTOMATIC INSTRUMENT LANDING APPROACH SYSTEM (AILAS)

In this mode, the airplane is automatically flown throughout an ILS approach. Throttle control, landing gear actuation, and flareout and landing are manually accomplished by the pilot. Throughout the approach, the course indicator should be monitored for proper operation of the system. A light in the AILAS pushbutton switch illuminates to indicate engagement. The automatic approach consists of two phases: constant altitude and glide-slope. The constant altitude phase extends from AILAS engagement to glide-slope entry. The glide-slope phase begins with glide-slope entry and extends to termination of the automatic approach. Preparatory to initial AILAS engagement, the pilot must set the proper runway heading into the ILS course indicator. This insures that localizer course deviation (as determined from the localizer beam signal) and the heading error signal (difference between the airplane compass heading and the selected runway heading) combine to provide horizontal steering signals for localizer beam entry, bracketing, and flying the center of the localizer beam. Engagement may be made on any heading in the localizer engage area which is a circle of four miles diameter centered 12 miles from the runway. However, an entry of 45° or less to the localizer heading is recommended to insure early localizer capture and minimum overshoot. Altitude at engagement should be 1500 feet above runway altitude as AILAS steering sensitivity during glide-slope descent is reduced as a function of sensed barometric pressure increase equivalent to 1500 feet. Prior to glide-slope entry, bank angle is limited to 33° (±3°). Glide-slope entry is indicated by airplane pitch change in desponse to the glide-slope signal. Thereafter, glide-slope deviation signals are used for vertical steering in place of pressure altitude signals which served to maintain altitude during the constant altitude phase. During the glide-slope phase, bank angle limiting is reduced to 15°. When visual contact with the runway is established, the momentary interrupt trigger should be held depressed, and the landing should be completed manually.

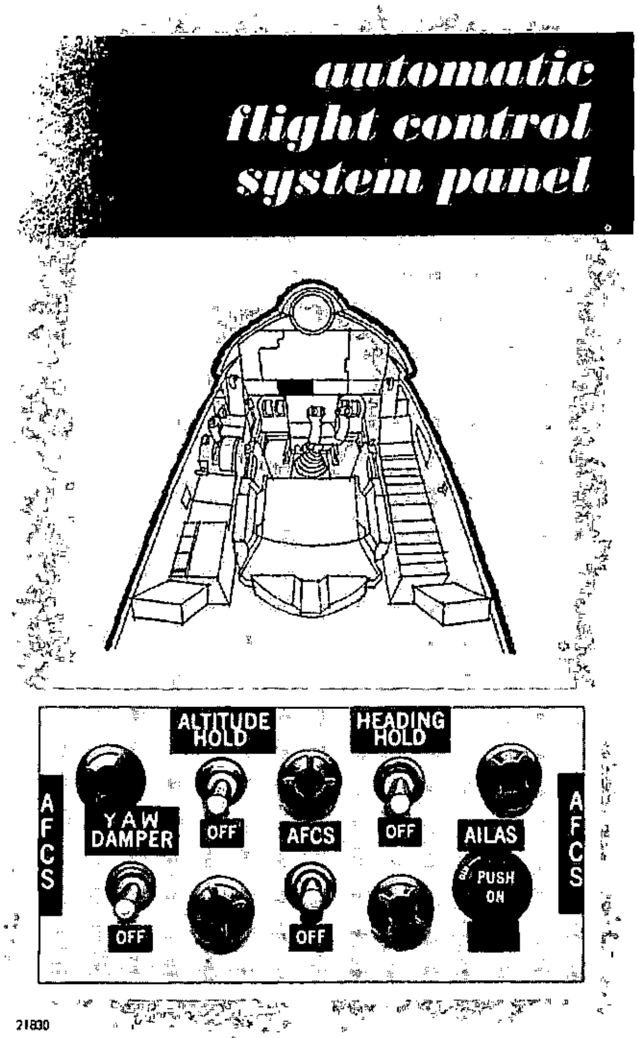


Figure 4-16

CAUTION

Flare and landing must be accomplished manually as the landing gear will not withstand the impact at 170 KIAS along the 3° glide-slope.

An ILS localizer signal interruption or failure prior to glide-slope entry or any ILS signal interruption will cause AILAS disengagement either immediately or at glide-slope coincidence, and flight control will automatically revert to pilot-assist mode. If normal ILS signals are restored before glide-slope entry, AILAS mode may be reengaged in the normal manner. Refer to AUTOMATIC APPROACH (AILAS), Section IX, for automatic approach procedures.

PITCH G LIMITER TEST SWITCH

On later airplanes, a test switch (figure 1-21) on the utility switch panel placarded "Pitch G Limiter Test" has positions +G, −G, and a spring-loaded (center) position. When the automatic flight control system is engaged, placing the g limit test switch to either the +G or −G position will actuate the g limiter, thus disengaging holding solenoids to return the pitch damper switch, and all AFCS switches to their OFF positions. When the switch is held in the +G position, an electrical input equivalent to that produced at five g's is fed to the limiter. In the −G position, an input signal equivalent to −2g's is fed to the limiter. The pitch g limiter test switch is operated by the 28-volt dc nonessential bus.

AUTOMATIC FLIGHT CONTROL SYSTEM SWITCH

The AFCS switch (figure 4-16), located on the AFCS control panel, is a two-position switch with AFCS and OFF positions. This switch supplies power from the integrated power supply for the three modes of AFCS operation and may be engaged when the pitch and yaw dampers are on and the radar master switch is in STBY or ON. With the switch in the solenoid-held AFCS position, the following conditions may be established:

1. Attack mode is automatically selected upon radar lock-on except in the SNAKE mode.
2. Landing approach mode will operate when the AILAS switch is engaged.
3. The pilot assist mode will function at all other times.

The AFCS switch receives power from the 28-volt dc nonessential bus.

ALTITUDE HOLD SWITCH

The altitude hold switch (figure 4-16), located on the AFCS control panel, is a two-position switch with ALTITUDE HOLD and OFF positions. During the pilot assist mode of operation, pressure altitude will be maintained when the switch is in the solenoid-held ALTITUDE HOLD position. With the switch in the OFF position, pitch attitude of the airplane will be maintained. The holding solenoid which engages the switch will release, and the switch will return to OFF under the following conditions:

1. Switch manually returned to OFF.
2. Momentary interrupt trigger is depressed.
3. The airplane is beep-trimmed in pitch.
4. Fire control system lock-on is obtained.
5. AILAS is engaged. During the constant altitude phase of AILAS, the switch will return to OFF but the system will continue to hold the engage altitude until glide-slope entry.
6. Emergency damper disconnect button is depressed.

The altitude hold switch receives power from the 28-volt dc nonessential bus.

AILAS BUTTON

The AILAS button (figure 4-16) on the AFCS control panel is a pushbutton switch placarded "AILAS" with instructions to PUSH ON on the face of the switch. When the switch is pushed in for engaging a green light illuminates, indicating successful engagement. The indicator light may be dimmed by clockwise rotation. When the switch is engaged, AILAS follows instrument landing system localizer and glide-slope signals to direct the airplane throughout an ILS approach. The AILAS switch can be engaged only if AFCS is engaged, and ILS signal is received, and the airplane is below the glide-slope. This switch receives power from the 28-volt dc nonessential bus.

HEADING HOLD SWITCH

The AILAS button (figure 4-16) on the AFCS control panel is a pushbutton switch placarded "AILAS" with instructions to PUSH ON on the face of the switch. When the switch is pushed in for engaging, a green light illuminates, indicating successful engagement. The indicator light may be dimmed by clockwise rotation. When the switch is engaged, AILAS follows instrument landing system localizer and glide-slope signals to direct the airplane throughout an ILS approach. The AILAS switch can be engaged only if AFCS is engaged, and ILS signal is received, and the airplane is below the glide-slope. This switch receives power from the 28-volt dc nonessential bus.

1. Switch manually returned to OFF.
2. AFCS switch returned to OFF.
3. Emergency damper disconnect button depressed.

The heading hold switch is powered by the 28-volt dc nonessential bus.

AUTOMATIC FLIGHT CONTROL SYSTEM PREFLIGHT

After starting engine, perform the following checks prior to all AFCS flights:

1. Radar master switch — STBY or ON.
2. Artificial horizon—Appears within about 30 seconds.
   Check for appearance of artificial horizon on scope and that the vertical gyro erects in about 60 seconds.
3. Pitch and yaw damper systems — Engage.
4. AFCS switch — AFCS; should engage in about 90 seconds.
   Check that no objectionable stick movement occurs when the switch is placed to AFCS.

5. Longitudinal beep trim — Check.

   Check longitudinal beep trim by trimming NOSE UP and NOSE DN, checking that control stick follows trim button displacement freely.

6. Pitch trim follow up — Check.

   Check longitudinal elevon trim followup by placing AFCS switch to OFF while stick position is trimmed to approximately three inches up or down elevon. Stick movement should be slight.

7. Lateral beep trim — Check.

   Check lateral beep trim by trimming RWD and LWD, noting that control stick follows trim button displacement freely. When aileron application exceeds 2.5° (control surface deflection), note that aileron limit switches cause pitch damper and AFCS switches to return to OFF and stick returns to neutral. Re-engage pitch damper and AFCS switches.

8. Momentary interrupt trigger — Depress; check manual flight.

   Depress momentary interrupt trigger and check that manual flight control is available. Release trigger.

Note

If aileron application exceeds 2.5° while checking for manual control, it will be necessary to re-engage the AFCS switch.

9. Override — Check.

   Check for override of AFCS by rapidly moving the control stick.

Note

If aileron application exceeds 2.5° while checking for override of the system, it will be necessary to re-engage the pitch damper switch and the AFCS switch.

10. Pitch g limiter test switch — +G; note that pitch damper and AFCS switches return to OFF. Re-engage switches.
11. Pitch g limiter test switch — −G; note that pitch damper and AFCS switches return to OFF. Re-engage switches.
12. Emergency damper disconnect button — Depress.

    Depress the emergency damper disconnect button and check that the AFCS and pitch and yaw damper switches disengage.

13. Takeoff trim — Set.

AUTOMATIC FLIGHT CONTROL SYSTEM NORMAL OPERATION

Engaging Procedure

Note

AFCS automatically disengages during vertical gyro erection. If the artificial horizon display on the pilot's scope does not function properly or will not cage, AFCS should not be engaged. The vertical gyro that furnishes the artificial horizon display also furnishes signals to the automatic flight control system for pitch and bank attitudes and, when not operating properly, may cause incorrect and erratic control surface movement.

1. Radar master switch—STBY or ON.
2. Pitch and yaw damper systems—Engage.
3. Trim for desired flight attitude.
4. Artificial horizon—Appears within about 30 seconds.

   Check for appearance of artificial horizon on scope and that vertical gyro erects in about 60 seconds.

5. With momentary interrup trigger depressed, AFCS switch — AFCS.
6. Momentary interrupt trigger — Release to engage AFCS.
7. Heading hold switch — HEADING HOLD.
8. Altitude hold switch — ALTITUDE HOLD.

Note

The altitude hold switch should be engaged in approximately level flight. Engaging while climbing or descending will result in an overshoot condition before damping to a constant altitude.

When operating in altitude hold, the altitude hold switch will return to OFF if the airplane is beep-trimmed in pitch, momentary interrupt trigger is depressed, or if auto attack or AILAS is engaged. The attack mode is automatically engaged when the pilot assist mode of AFCS is in operation, proper armament is selected, and radar lock-on is obtained. Refer to NORMAL OPERATION OF FIRE CONTROL SYSTEM, this Section, for operation of the attack mode.

Disengage Procedure

Momentary disengaging is available by depressing the momentary interrupt trigger. If it is not desired to re-engage AFCS immediately, any of the following actions will disengage the system:

1. AFCS switch — OFF.

2. Radar master switch to WARM or OFF.
3. Pitch and yaw damper systems — Disengage.
4. Emergency damper disconnect button — Depress.

Note

Depressing the emergency damper disconnect button also disengages the damper systems.

Abbreviated Check List

Refer to T. O. 1F-102A-(CL)1-1 for the Abbreviated Check List of the above procedures.

NAVIGATION EQUIPMENT

STANDBY COMPASS

Refer to INSTRUMENTS in Section I.

DIRECTIONAL INDICATOR (SLAVED)—J-2 COMPASS SYSTEM

This system* consists of a flux valve transmitter installed near the tip of the left wing, a control gyro and amplifier installed in the nose wheel well, and the rotating dial element of the radio magnetic indicator (refer to COMMUNICATIONS AND ASSOCIATED ELECTRONIC EQUIPMENT, this Section) on the instrument panel. The system is basically a gyro-stabilized compass that is automatically kept on a true magnetic north heading by signals from the flux valve transmitter, which detects the north-south flow of the earth's magnetic field. The directional gyro control contains an electrically driven gyro, having a spin axis tangent to the earth's surface. The gyro is also slaved to the flux valve transmitter, which puts in corrections referenced to the earth's magnetic meridian. The system provides an indication of magnetic headings without northerly turning error, oscillation or swinging. The compass system is operable when ac and dc essential bus power are available.

Note

Should ac power fail, dc power to the system will also be cut off by the interlock relay. However, dc power failure will not disconnect ac power to the system.

For the first three or four minutes of operation, the gyro is on a fast slaving cycle and will precess rapidly. During this time it should align with the magnetic heading. The gyro then begins a slow slaving cycle. The directional indicator (slaved) is free from drift and requires no resetting in normal operation.

Note

After the gyro reaches operating speed, the indicator should be checked against the standby compass indication to make sure the indicator does not show a 180-degree ambiguity. The system is not operating properly if such ambiguity exists.

AF 53-1791 thru 55-3379 unless modified by TCTO 1F-102A-546.

Indicator readings are incorrect if the airplane exceeds 85 degrees of climb, dive, or bank.

Directional Indicator (Slaved) Fast Slave Button

A fast slave button at the top left side of the instrument panel provides a means of stabilizing the gyro after it has been upset by overbanking or acrobatics. Depressing the button interrupts 28-volt dc essential bus power to the compass. When the button is released, power is restored and the fast slaving cycle is initiated to permit faster gyro recovery to the true heading.

CAUTION

To avoid damage to the slaving torque motor, the fast slave button should not be used too frequently. Allow ten minutes between actuations, and hold button depressed no longer than two seconds.

Directional Indicator (Slaved) Slaving Switch

A switch on the top left side of the instrument panel has two positions, NORMAL and DESLAVE. When in DESLAVE position the switch applies dc power to a relay in the amplifier unit to open the circuit that slaves the control gyro to the flux valve transmitter. This switch is used in polar regions where the excessive dip in the earth's magnetic lines of force causes compass indications to become inaccurate. With the switch in DESLAVE position the compass system may still be used temporarily as a turn indicator if conventional procedures for making gyro drift corrections are employed. Except for the special circumstances noted, the switch should always be on NORMAL.

Directional Indicator (Slaved) Correction Card

A correction card and holder are located on the left side of the cockpit above the console, and forward of the throttle quadrant.

DIRECTIONAL INDICATOR (SLAVED) — J-4 COMPASS SYSTEM

The J-4 compass system* may be used as a directional gyro corrected for apparent drift due to the earth's rotation or as a directional gyro stabilized magnetic compass. The J-4 compass system consists of a directional control gyro, an amplifier-servo assembly, a control panel (figure 4-17)

AF 55-3380 & on, & airplanes modified by TCTO 1F-102A-546.

on the right-hand console, and the rotating compass card of the radio magnetic indicator on the instrument panel. The two modes of operation, "magnetic slaved" and "directional gyro," provide accurate directional reference for all latitudes. Magnetic slaved mode may be used at all latitudes; however, a severe magnetic distortion occurs when operating near the magnetic poles. When in magnetic slaved mode the system is basically a gyro-stabilized compass slaved to the flux valve (remote compass) transmitter. This mode provides magnetic headings without northerly turning error or oscillations. Directional gyro mode may be used at all latitudes but is most useful when the magnetic field is weak or distorted or when navigating in polar regions. When in directional gyro mode, the system is free of magnetic influence and operates as a directional gyro indicating an arbitrary gyro heading (corrected for apparent gyro drift due to the earth's rotation) as selected by the pilot. At different latitudes, apparent gyro drift due to earth's rotation varies, with the smallest amount of drift being at the equator and the greatest amount in the polar regions. In directional gyro mode, with the proper latitude selection made on the control panel, the gyro is made to precess the correct amount required to overcome gyro drift at the selected latitude. The J-4 compass system also serves as a directional reference for the automatic flight control system and supplies information to the VHF navigation indicators. The system is powered by the 200/115-volt and 26-volt, 400-cycle ac essential bus and the 28-volt dc essential bus. On some airplanes the J-4 compass control panel lights will illuminate whenever the 28-volt dc essential bus is energized. On later airplanes* the control panel lights are controlled by the console light switch located on the lighting control panel (33, figure 1-4). The system is energized whenever power is supplied to the essential buses.

Directional Indicator (Slaved) Correction Card

A correction card and holder are located on the left side of the cockpit above the console, and forward of the throttle quadrant.

Function Selector Switch

The two-position function selector switch (figure 4-17), located on the J-4 compass control panel, has positions DG and MAG. DG position selects directional gyro mode; MAG selects magnetic slaved mode. The switch receives power from the 28-volt dc essential bus.

Synchronizer "SET" Switch and Synchronization Indicator

The synchronizer "set" switch (figure 4-17), located on the J-4 compass control panel, provides a manual means to fast slave, or synchronize, the rotating compass card of the radio magnetic indicator to the correct magnetic

*AF 57-770 & on.

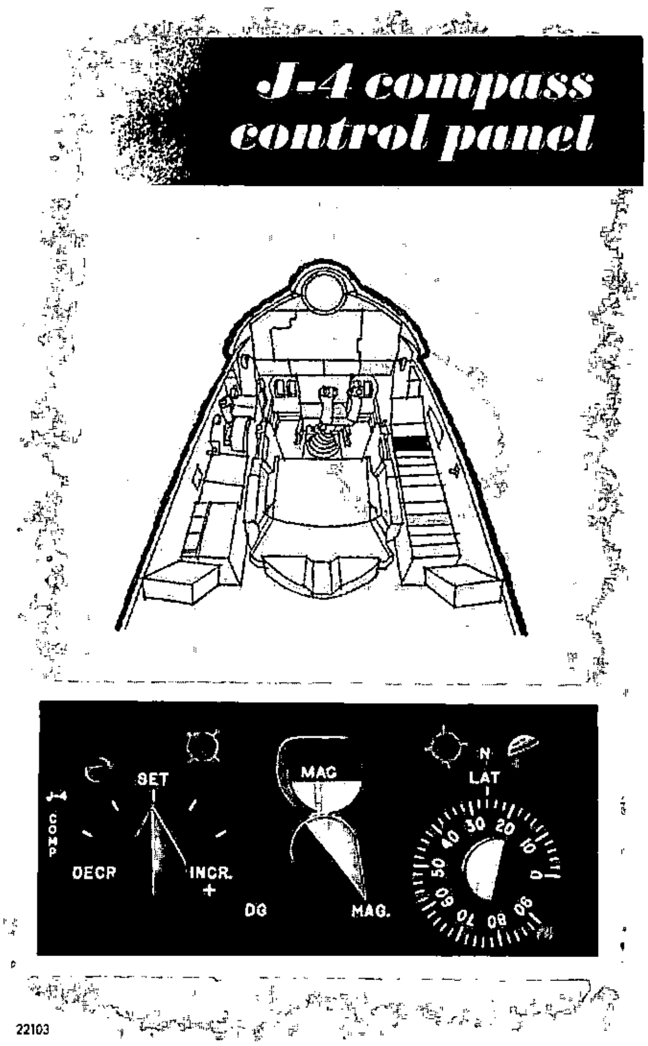


Figure 4-17

heading when the system is in magnetic slaved operation. When in directional gyro mode, the "set" switch is used to position the compass card to the desired gyro heading. The switch has a spring-loaded SET position at the top center and may be moved, against spring tension, to the left "DECR −," position or to the right, "INCR +," position. Two intermediate dash marks on either side of SET position indicate slow or fast synchronization. The first dash mark on either side provides a slewing rate of ½ rpm and the second mark provides a slewing rate of seven to nine rpm. When in DG operation, the direction of movement of the "set" switch is determined by whether an increased or decreased heading is desired. When in MAG operation the direction of displacement of the "set" switch is determined by a synchronization indicator (figure 4-17) above the function selector switch on the control panel. When the function selector switch is in MAG the synchronization indicator is exposed and is placarded MAG. A center dash mark at the bottom of the indicator is opposed by a pointer which is secured at the top (pendulum-like) and is caused to swing to either a "+" indication on the left or a "−" indication on the right by signals from the flux valve transmitter. When the pointer is off center to the "+" side, the "set" switch is moved to the "INCR +" direction to the desired rate of slewing until the synchronization pointer swings down to the center position. If the pointer is indicating "−," the knob is moved to the "DECR −" position to center of the pointer. Centering the pointer of the synchronization indicator synchronizes the rotating compass card to the correct magnetic heading. The "set" switch receives power from the 28-volt dc essential bus.

Hemisphere Selector Screw and Indicator

The hemisphere selector screw (figure 4-17) on the compass control panel is used to select the hemisphere in which the airplane is operating. A small window beside the screw displays "N" or "S" to indicate the hemisphere selected.

Latitude Selector Switch

The latitude selector switch (figure 4-17) on the compass control panel is used to rotate a circular dial at the base of the knob placarded LAT. The circular dial is numbered from 0 to 90 to indicate degrees latitude and has a mark for each two degrees. The latitude selector switch and dial are operative in DG mode and are used to select the latitude in which the airplane is operating. When in DG operation, with the operating latitude selected, the directional gyro will be corrected for apparent drift due to the earth's rotation.

Note

The proper corrections will not be made if the hemisphere selector screw is not indicating the correct hemisphere.

The latitude selector switch receives power from the 28-volt dc essential bus.

NORMAL OPERATION OF THE J-4 COMPASS

Magnetic Slaved Mode

1. Select MAG with the function selector switch.
2. Allow approximately two minutes warmup time after power is applied to the essential buses. When power is initially applied, or when switching from DG, a fast slaving action is applied for the first 15 seconds to synchronize the compass card with the flux valve (remote compass) transmitter. After the initial fast slave cycle, the system returns to the normal slewing cycle of 2° per minute.
3. Before takeoff, check the synchronization indicator to see if the system is synchronized. If the system is not synchronized prior to takeoff, use the "set" switch to center the synchronization pointer. The switch may be used at any time to obtain synchronization and will not produce a hard-over signal if used during automatic flight control system operation.

CAUTION

Do not operate the "set" switch continuously for more than 30 seconds to avoid overheating the slew motor.

Directional Gyro Mode

1. Allow approximately 12 minutes warmup time after power is applied to the essential buses.
2. Select the desired hemisphere with the hemisphere selector screw. Check the "N" or "S" in the hemisphere selector indicator window.
3. Select the latitude in which the airplane will be operating with the latitude selector switch.
4. After desired heading is established in magnetic mode, switch the function selector switch to DG. The system is then independent of the magnetic compass equipment and latitude correction, for apparent gyro drift is being given to the compass card.

Note

As the airplane passes through different latitudes in flight, the latitude selector switch should be rotated to the new latitude every two degrees of latitude change.

ARMAMENT EQUIPMENT

Note

Rerer to the Confidential Supplement, T.O. 1F-102A-1A, for armament information and for the following figures pertaining to armament:

- ***Armament Control Panel*—** **Figure 4-21.**
- ***Armament Selection Table*—** **Figure 4-22.**

FIRE CONTROL SYSTEM

All information concerning the fire control system is contained in the Confidential Supplement, T. O. 1F-102A-1, except the following:

ELECTRONIC COOLING

During normal flight conditions, the eletronics compartments are cooled by ram air, with the cockpit exhaust air aiding in cooling the forward, upper and intermediate electronics compartments. A pressure-and-temperature-controlled shutoff valve in the ram air duct maintains air pressure at approximately two inches Hg above ambient, and the shutoff valve will close if the temperature of the incoming ram air reaches approximately 160-165°F. When the ram air valve closes due to high inlet temperature, cockpit exhaust air is automatically distributed through the ram air ducts for cooling of the forward, upper, and intermediate compartments. Ground cooling of the electronic compartments with the engine operating is provided by a reverse flow of air in the ram air distribution ducts. In the aft electronics compartment and IFF units, this reverse flow of air is started by the engine compressor creating a partial vacuum in the engine intake ducts, which causes a reverse flow of air through the aft electronic compartment and IFF units. Air in the aft compartment comes from the main wheel well, and air in the tail section is drawn through the IFF units. The vacuum in the intake ducts also causes a purge valve between the left engine inlet duct and the upper electronics compartment to open and draw air from the open nose wheel well through the upper and intermediate electronics compartments. A reverse air flow created by a jet pump is the only ground cooling means for the forward electronics compartments, and the jet pump also assists in cooling the intermediate and upper compartments. When the engine rpm is approximately 72% or below, the jet pump valve opens and allows engine bleed air to flow forward and overboard through the left boundary-layer ram air duct. A partial vacuum is thus created in the ram air duct, which draws cooling outside air from the nose wheel well through the forward, intermediate, and upper electronics compartments. Operation of the jet pump is normally audible to the pilot. A pressure switch senses bleed air duct pressure and prevents the jet pump valve from opening when the duct pressure exceeds 30 to 35 psi. Structure overheat detection probes, near the jet pump nozzles, sense excessive fuselage skin temperatures and automatically close the jet pump valve when excessive temperatures exist. Malfunction of the jet pump system is indicated by an electronics cooling warning light on the warning light panel. During standby conditions with the engine not running, a fitting in the nose wheel well provides for connection of a ground cooling unit.

Note

Refer to ELECTRONICS COOLING, Section V, for cooling limitations.

ELECTRONIC COOLING WARNING LIGHT

An electronic cooling warning light is provided which will illuminate and display "ELECTRONIC COOLING" (16 figure 1-26) to indicate a malfunction in the electronic cooling system. Any time the light illuminates it indicates a malfunction, either in the electronic compartment cooling as intended, or a malfunction in the electrical circuit of the warning light itself. Illumination of the light in flight could indicate the following cooling system malfunctions:

- An electrical malfunction.
- Ram air regulator is closed and ram air pressure downstream of the regulator above two inches Hg.
- Ram air regulator open and ram air temperature above 160°F.
- Jet pump valve open.
- Structural overheat.

Note

Frequently a flashing or flickering light may be experienced; however, only a steady light is significant indication that proper electronic cooling is not provided, and the fire control system should be turned off.

Some airplanes* have no provision for warning of electronic compartment cooling system trouble while airborne, and in these airplanes illumination of the electronic cooling warning light during flight could indicate only a malfunction in the warning light electrical circuit. This illumination would impose no restriction in the operation of the electronic components in these airplanes. For most airplanes** with the electronic cooling warning light modification, armament firing will have no effect upon the electronic cooling warning light operation. On some airplanes†, however, it is normal for the electronic cooling warning light to illuminate briefly during armament firing, and if there is no malfunction in the system it will go out shortly after firing. During ground operations illumination of the electronic cooling warning light will indicate a malfunction on all airplanes** incorporating the electronic cooling warning light modification. There is one exception. With electrical power and the engine furnishing no rpm, it is normal for the electronic cooling warning light to illuminate. If there is no malfunction, the light should go out when external cooling or engine rpm is applied. With other airplanes‡, a check must be made to determine the true existence of a malfunction when the light appears. If the light illuminates when the throttle is above 72%, retard the throttle to below 72%; if the light remains on, there is malfunction in the electronic cooling system and the fire control system should be turned to WARM or OFF. If the light goes out, there is no malfunction and the throttle may be advanced to desired rpm. (Refer to ELECTRONIC COOLING, Section V, for limitations.) If the light illuminates after advancing the throttle during takeoff, disregard as it is normal and the light will go out when the landing gear is up.

AF 53-1791, -1792, -1794 thru -1796, -1798, -1800 thru -1805 53-1807 thru 54-1387, -1389, -1391 thru -1400, 54-1402 thru 55-3372.

***All airplanes after compliance with TCTO 1F-102-713.*

†*AF 53-1793, -1797, -1799, -1806, 54-1388, -1390, -1401, 55-973 & on, prior to compliance with TCTO 1F-102-713.*

‡*All airplanes not modified by TCTO 1F-102-713.*

NORMAL OPERATION OF THE FIRE CONTROL SYSTEM

Only basic operating procedures are given in this section. Data link procedures are not included. For additional information on fire control system operation and techniques, refer to applicable technical order.

Operating Procedure

Note

Referto ELECTRONICS COOLING, Section V, for cooling limitations which must be observed prior to operation of the fire control system.

1. Armament safety switch — SAFE.
2. Armament selector switch — SNAKE.
3. Igniter control switch — TRAINING (safety-wired).
4. Radar master switch — STBY.

Note

Check that the range trace and artificial horizon appear on the scope in approximately 30 seconds after turning master switch to STBY from OFF or when turning from WARM to STBY.

5. Radar mode selector switch — As desired.
6. Elevation scan control — As desired.
7. Elevation vernier wheel — DETENT.
8. Azimuth scan control — As desired.
9. Antenna elevation scan button — As desired.
10. Anti-clutter switch — OUT.
11. ATOT switch — NORMAL (center) (some airplanes).
12. Anti-jam switch — NORMAL (some airplanes).
    Auto-tune/missile anti-chaff switch — NORMAL (center) (other airplanes).
13. Estimated range rate knob — 0 (some airplanes).
14. Nose-tail switch — As required.
15. Radar master switch — ON.

    When master switch is placed to ON (after delay period has elapsed), check that range trace widens to approximately ⅛ inch and that "grass" or noise appears on the scope to indicate that the set is transmitting.
16. Search intensity — Adjust.

    After radar is determined to be transmitting, the search intensity should be adjusted as follows:
    a. IF Gain control knob — Counterclockwise to stop.
    b. Search intensity wheel — Adjust until range trace is barely visible.
    c. IF Gain control knob — Clockwise to optimum position.
17. Attack intensity wheel — As desired.

    Adjust attack intensity by adjusting brightness of artificial horizon display.

Automatic Search Operation

1. Azimuth scan knob—As desired.

   The azimuth scan knob should be set at B if azimuth of target is unknown. If azimuth is known, use scan knob to obtain best target return.
2. Elevation scan knob—As desired.

   Elevation scan knob should be set to scan desired vertical area.
3. Radar mode selector switch — As desired.
4. Anti-clutter switch — As desired.
5. Anti-jam switch — As desired (some airplanes).
   Auto-tune/missile anti-chaff switch — As desired other airplanes).

   If jamming is detected, select AUTO-TUNE position to obtain radar frequency clear of jamming.
6. Data link GCI station channel selector switch — Assigned channel.
7. Data link airplane channel selector switch — Assigned channel.

Manual Search and Lock-On

Lock-on is obtained after the target is within range by using the following procedure:

1. Action trigger—Depress.

   Squeeze the action trigger to gain manual control of the antenna after azimuth and elevation position of the target is noted.
2. Target—Spotlighted.

   By means of lateral movement of the left control grip and adjustment of the antenna elevation wheel, obtain best possible target return.
3. Range gate marker—Coincide with target.

   When the range gate marker and the target coincide, the attack display should appear showing lock-on has been obtained.
4. Action trigger — Released.

   After lock-on is obtained, release the action trigger and hand control.

Automatic Track; Snake Mode Operation

1. Armament safety switch — SAFE.
2. Armament selector switch — SNAKE.
3. Igniter control switch — TRAINING (safety-wired).

4. Steering dot — Centered.

   After lock-on, a pursuit course is flown by centering the steering dot in the reference circle. To perform an identification pass, steer with the dot centered and maintain the desired rate of closure.

WARNING

The first indication of a possible collision course is evidenced by range circles failing to shrink at the proper time. The pullout maneuver should be executed if a collision course is suspected. The pullout signal will appear when the airplane is 200 feet below and to the right and 200 yards astern of the target.

When it is desired to fly a formation course on a leading airplane, maintain the range circle at a constant diameter, keep the rate of closure gap in the range circle at zero, and maintain a constant position steering dot.

Automatic Track; Missile Firing

1. Armament selector switch — MISSILE selection as desired.
2. Armament safety switch — ARMED.

Note

Normally the desired armament is selected prior to the attack phase during automatic search operation but should be rechecked after lock-on.

3. Igniter control switch —TACTICAL.
4. Lock-on.
5. Steering dot — Centered.

   Fly lead collision course by centering the steering dot within the reference circle.
6. Armament trigger—Depress to SECOND DETENT after reference circle collapses to 1/4 inch.

   It will be necessary to depress the armament trigger for firing unless an automatic attack is accomplished.

WARNING

When using AFCS to accomplish automatic attack, armament firing is automatic and use of the armament trigger is not required. If the wrong target was selected, or if for any reason it is desired to abort the attack run, depress manual mode trigger or auto-search button and place the armament safety switch to SAFE to prevent automatic firing.

7. Estimate target range after range circle starts to shrink.

   The range circle will start to shring at 25,000-yard range. Depending on speed and altitude of the airplane, missile launch range will be approximately 3½ to 1½ miles.

WARNING

- If the range circle fails to shrink at proper rate or time with normal rate of closure or if range trace fails to move to the center of scope, a collision course may be indicated and the pullout maneuver should be executed.
- If armament firing is initiated during transient negative pitch maneuvers (not including steady-state negative g maneuvers), it is possible to strike the airplane nose section with the missiles. Therefore, missiles should not be fired during or immediately following rapid application of nose down elevator or sudden release of nose up elevator.

8. Continue to fly course until fire signal (X) appears on scope.

   The fire signal (X) appears at time of missile launch and disappears from the scope in three seconds. When the signal disappears, the pullout maneuver should be executed.

Note

During pullout, do not exceed the limits of radar lock-on or do not intentionally break lock before missile impact. The radar maintains lock-on after the pullout signal appears, to provide return radiated energy for the radar missile guidance system. If lock-on is broken, no energy will be provided to guide the missile, resulting in a probable miss and a free missile.

9. Auto-search button — Depress after missile impact.

   After missile impact, depress the auto-search button to restore automatic searching.

Automatic Track; Rocket Firing

1. Armament selector switch — ROCKET or RAD.
2. Armament safety switch — ARMED.
3. Igniter control switch —TACTICAL.

Note

Normally the desired armament is selected prior to the attack phase during automatic search operation but should be rechecked after lock-on.

4. Lock-on.
5. Steering dot — Centered.

   Fly lead collision course by centering the steering dot within the reference circle.
6. Armament trigger — Depressed to SECOND DETENT and reference circle collapses to 1/4 inch.

   On some airplanes it will be necessary to depress the armament trigger for firing unless an automatic attack is accomplished.

WARNING

When using AFCS, armament firing is automatic and use of the armament trigger is not required. If the wrong target was selected, or if for any reason it is desired to abort the attack run, depress the momentary interrupt trigger or auto-search button and place the armament safety switch to SAFE to prevent automatic firing.

7. Estimate target range after range circle starts to shrink.

   The range circle will start to shrink at 5000-yard range.

WARNING

- If the range circle fails to shrink at proper rate or time with normal rate of closure, or if range trace fails to move to the center of scope, a collision course may be indicated and the pullout maneuver should be executed.
- If armament firing is initiated during transient negative pitch maneuvers (not including steady-state negative g maneuvers), it is possible to strike the airplane nose section with the rockets. Therefore, rockets should not be fired during or immediately following rapid application of nose down elevator or sudden release of nose up elevator.

8. Continue to fly course until fire signal (X) appears on scope.

   When firing rockets, the pullout maneuver should be executed as soon as the signal appears.

WARNING

During rocket firing attacks, the rocket travel is considerably less than for a missile on missile runs and less horizontal clearance is provided. Pullout maneuvers should be planned in advance and correctly executed to avoid possible collision.

9. Auto-search button — Depress after pullout.

   After pullout, depress the auto-search button to restore automatic radar search.

AFCS Automatic Attack

An attack run using either missiles or rockets may be performed automatically by using AFCS. To make an AFCS attack, the AFCS should be operating in pilot assist mode and armament selected prior to lock-on. After lock-on, AFCS "fades" into complete control of the airplane in 1/2 to 4 1/2 seconds.

Note

During AFCS attacks, steering signals are provided from the fire control system. System noise, if present, will be reflected in AFCS steering performance.

Snap-Up Attack

A snap-up attack can be performed by using manual steering (without AFCS) and normal missile procedures. This attack is a lead collision attack except that the target can be as much as 15,000 feet above the airplane. Steering is accomplished by ignoring the elevation steering error until the reference circle collapses. At this time, as rapidly and smoothly as possible, the nose of the airplane should be pulled up to center the steering dot in elevation. At the time of snap-up, the dot should be centered in azimuth. The armament trigger should be squeezed when the reference circle collapses. After the climb attitude is established and the steering dot centered, the missiles are launched by a signal from the computer or clock timer and continue to climb for impact with the target. For a typical snap-up attack, see figure 4-18.

Ground Map Operation

In MAP mode, single bar scan of an area 200 nautical miles forward of the airplane is provided. However, close-in ground mapping can be obtained in other search modes if the airplane is low enough. With the fire control system operating in automatic search, proceed as follows for ground map operation:

1. Radar mode selector switch — MAP.
2. Azimuth scan knob — As desired.
3. Elevation scan switch — DN to provide desired range coverage.
4. IF gain control knob — As required.

   Adjust the IF gain control knob for optimum setting.

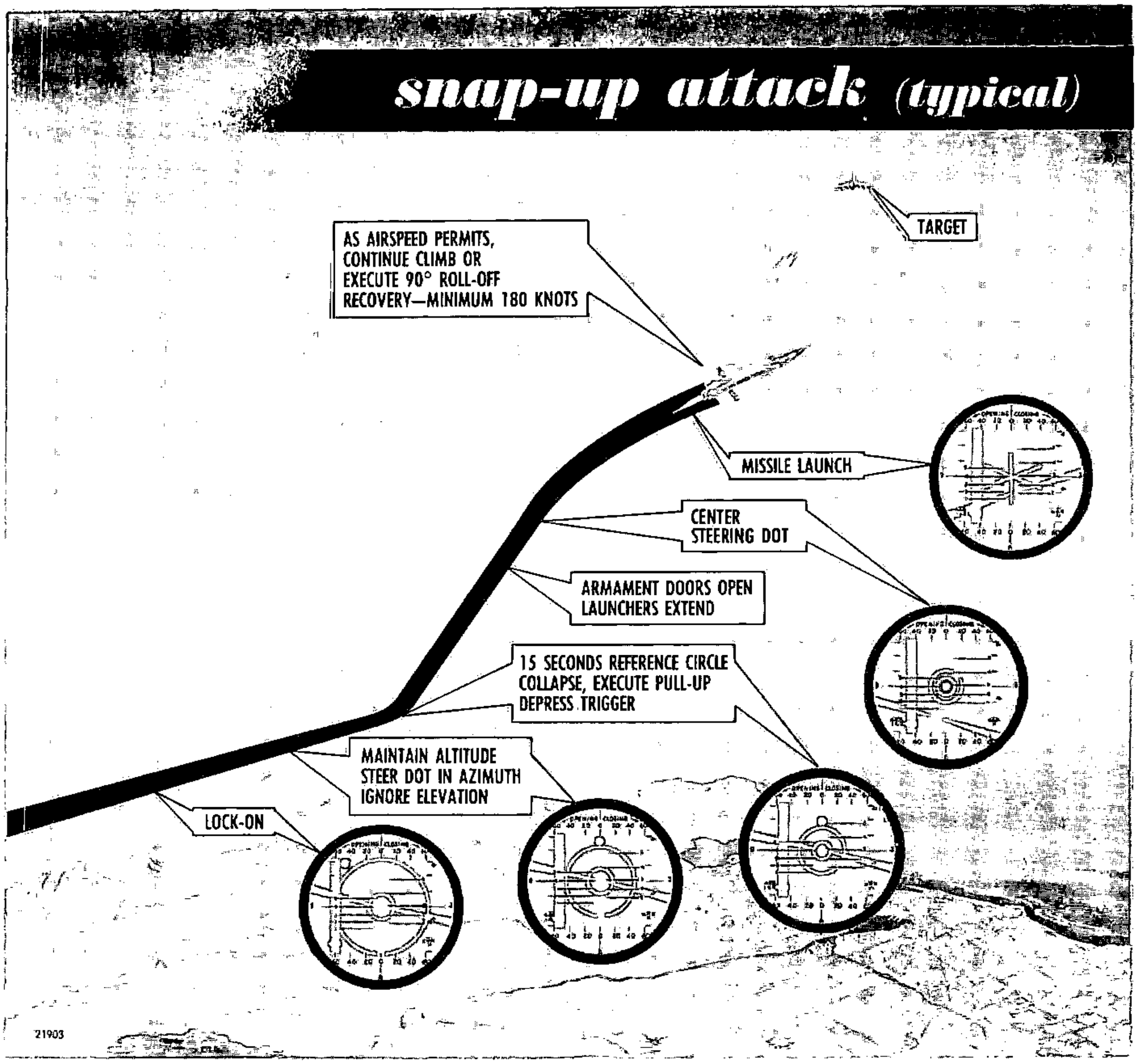


Figure 4-18

5. To expand display, position strobe then move AAI beacon expand switch to BCN EXP.

   To expand display, position strobe at the lower limit of the desired 20 mile sector by use of the hand control; then hold the AAI beacon expand switch to BCN EXP.

Beacon Operation

With the fire control system operating in automatic search, use the following procedure for beacon operation.

1. Radar mode selector switch – BCN.
2. Azimuth control knob – As desired.
3. Elevation control knob – DN to provide desired range coverage.
4. IF Gain control knob – As required.

   Adjust the IF gain control knob for maximum legibility.

5. To expand display, position strobe then move AAI beacon expand switch to BCN EXP.

   To expand display, position strobe at the lower limit of the station return by use of the hand control; then hold the AAI beacon expand switch to BCN EXP to read the station identification.

Prior to Acrobatics or Landing

1. Radar master switch—STBY.

   For acrobatics, landing pattern, and taxiing after landing, the master switch should be in STBY.

2. Armament safety switch—SAFE.
3. Armament selector switch—SNAKE.

Normal Shutdown (After Parking)

1. Search and attack intensity controls—Fully counterclockwise.

CAUTION

To prevent damage to the radar scope face, the search and attack intensity controls should be turned fully down prior to placing the radar master switch OFF.

2. Radar master switch – OFF.

ABBREVIATED CHECK LIST

Refer to T.O. 1F-102A-(CL)1-1 for Abbreviated Check List of the above procedure.

EMERGENCY OPERATION OF FIRE CONTROL SYSTEM

Misfire Warning Light Illuminated

If the misfire warning light illuminates, it is an indication that the intervalometer has produced an ignitor signal and a missile has not been launched (the doors remain open and the selected launchers remain extended). If this condition occurs, the following procedure should be followed:

1. Auto-search button – Depress to break "lock-on."
2. Armament safety switch – SAFE.
3. Armament selector switch – SNAKE.
4. Normally, launchers remain extended if a misfire occurs. If a flight condition exists wherein a clean airplane configuration is necessary, launchers may be retracted after five minutes. In an emergency, the launchers may be retracted after 30 seconds.

Incomplete Armament Cycle

If the launchers remain extended or the armament bay doors do not close for any reason other than a misfire as described above, the misfire warning light will not illuminate. Use the following procedure in attempting to close doors:

1. Depress auto-search button to break lock-on and wait 15 seconds.
2. Armament selector switch to opposite intervalometer and hold for five seconds (if missiles previously selected, select rockets or vice versa).
3. Armament safety switch – ARMED (hold for five seconds).
4. Igniter control switch – TACTICAL.
5. Armament safety switch – SAFE.
6. Armament selector switch – SNAKE.

   If this procedure does not result in closing the armament doors, land with doors open. Refer to LANDING WITH ARMAMENT DOORS OPEN, Section III.

WARNING

The armament trigger should not be depressed at any time during this procedure to preclude the possibility of armament firing.

Intervalometer Reset

Failure to properly reset the intervalometer at time of armament loading could result in failure of the armament bay doors to open or launching racks to extend, causing an aborted attack. This condition would follow a completed radar attack during which the pullout signal was received but the missiles or rockets did not fire. The following procedure should be followed to reset the intervalometer prior to subsequent radar attacks.

WARNING

The armament trigger should not be depressed at any time during this procedure to preclude the possibility of accidental armament firing.

1. Armament safety switch – SAFE.
2. Armament selector switch – ROCKET TRIGGER SALVO (if on last attack rockets were selected) or MISSILE TRIGGER SALVO (if on last attack missiles were selected). Hold for five seconds.
3. Armament safety switch – ARMED (hold two seconds).
4. Armament safety switch – SAFE.
5. Armament selector switch – SNAKE.
6. Proceed with next attack.

ABBREVIATED CHECK LIST

Refer to T.O. 1F-102A-(CL)1-1 for Abbreviated Check List of the above procedures.

COMPUTING OPTICAL SIGHT

The windshield mounted optical sight* (figure 4-19) is used for firing missiles from a pure pursuit course or rockets from a lead pursuit course. The optical sight

**AF 56-1137 & on.*

is generally utilized when radar operation is unsuccessful because of heavy ground return or enemy jamming. The sight head consists of controls for operation and a retractable reflector glass which may be lowered into position when desired. When the intended target has been identified as to type (heavy or medium bomber) the desired target reticle may be selected. Depressing armament trigger to the first detent position will cause a reticle to appear on the reflector glass with a missile selection. The reticle consists of a center dot and two concentric circles and corresponds in size to the type of target selected. The inner circle is used for missile-firing attacks and the outer circle for rocket-firing attacks. In making a rocket-firing attack, an optical selection on the armament control panel will cause the radar antenna to be taken out of automatic search and cause the reticle to appear on the reflector glass. After tracking the target for a few seconds with the antenna caged by means of a switch on the control stick yoke, angular rate voltages build up to establish the amount of lead angle required. The antenna is then uncaged and the computor output then causes the antenna and the reticle position (which is slaved to the antenna) to be processed to the correct lead angle. Tracking the target with the reticle will then fly the interceptor on a lead pursuit course. During missile firing with the optical sight, depressing the armament trigger to the first position causes the reticle to appear, and will cause the antenna to be fixed along the radar boresight line and remain there. Throughout the missile attack no lead or deflection angle is supplied and the sight operates as a fixed sight.

Note

In addition to the optical sight control discussed below refer also to ARMAMENT TRIGGER, RADAR MASTER SWITCH, and ARMAMENT SELECTOR SWITCH, this Section.

Brightness Selector Knob

The brightness selector knob (figure 4-19) on the face of the optical sight, is used to select the desired brightness of the reticle image. Turning the knob clockwise increases the brightness of the reticle. Turning the knob fully counterclockwise will remove the reticle.

Target Selector Knob

The target selector knob (figure 4-19) is located on the face of the optical sight and is placarded "Target Selector." This knob has four positions of which two are operable. The selection of either of the placarded positions, HVY or MED, provides for display of the correct size reticle on the reflector glass. Selection of HVY

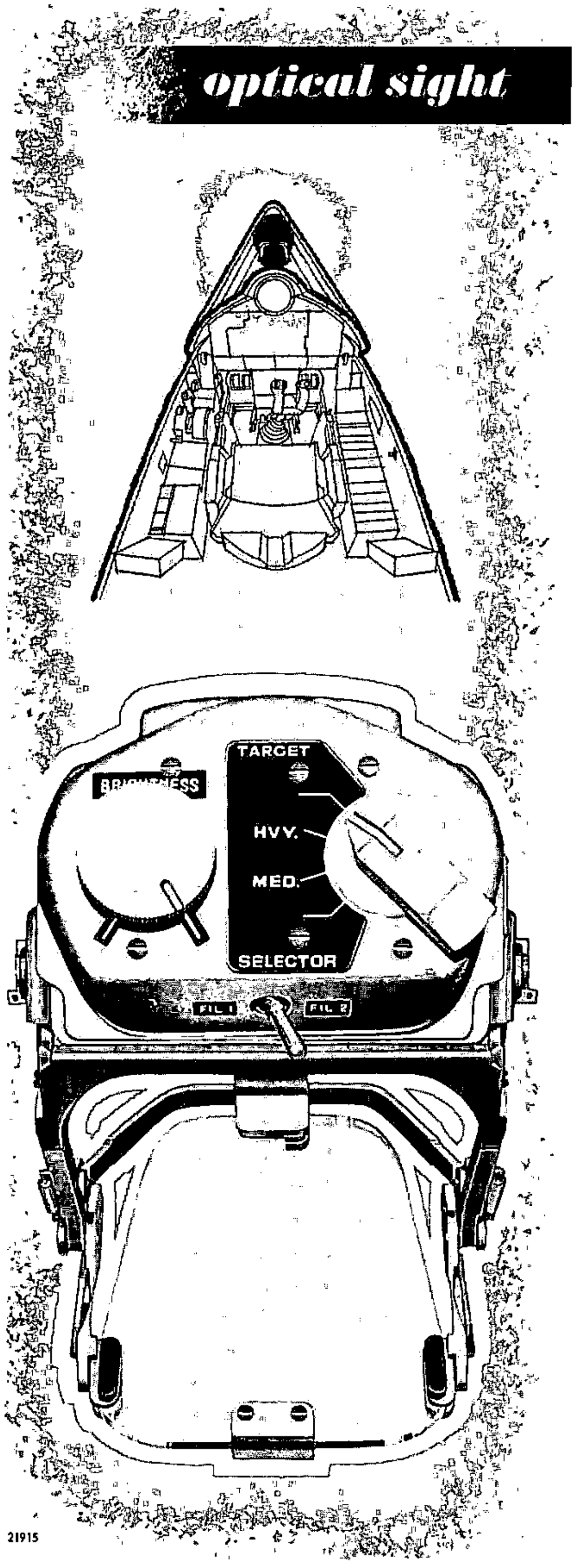


Figure 4-19

displays a reticle which is compatible to operation against a heavy bomber. Selection of MED displays the correct sized reticle for attacking a medium bomber.

WARNING

When making passes on airplanes smaller than the target selector setting, closing range will be closer than normal.

In addition to providing reticle size, selection of the desired target also introduces a fixed range voltage and an airplane-target overtake speed voltage to the computer subsystem for rocket-firing attacks, and an overtake speed voltage to the missile auxiliaries subsystem during missile-firing attacks. The target selector knob is powered by the 28-volt dc nonessential bus.

Filament Selector Switch

The filament selector switch (figure 4-19) is a two-position toggle switch located on the lower portion of the sight head. Selection of FIL 1 or FIL 2 position selects one element of the dual filament bulb.

Optical Sight Cage Switch

The optical sight cage switch (figure 1-20) is a toggle switch located on the platform extending forward from the control stick yoke. It is placarded "Sight" and has positions CAGE and UNCAGE. This switch operates only during optical rocket attacks. In CAGE position, the switch provides for caging the radar antenna along the antenna reference line. When the switch is in UNCAGE and the reticle is maintained on the target, gyros on the antenna sense the rate of change of the line-of-sight, and provides a signal to the computer to determine lead angle. The optical sight cage switch is powered by the 28-volt dc nonessential bus.

Normal Operation of Computing Optical Sight

MISSILE FIRING — OPTICAL

1. Radar master switch — STBY or ON, Delay period elapsed.
2. Armament selector switch — MISSILE selection as desired (RAD, ALL, or OPT).

   This system is primarily for use with the optical missile or for obtaining launch range information for radar missiles in ATOT pursuit attacks. If only optical tracking is available and a RAD or ALL selection is made, the radar missiles will fire without guidance.
3. Armament safety switch — ARMED.
4. Igniter control switch — TACTICAL.
5. Target selector knob — HVY or MED.
6. Armament trigger — Held depressed to FIRST DETENT — reticle appears.

   A minimum of 16 seconds is required for missile preparation prior to firing.

   Note

   To avoid potentially disabling strain on missile components, do not exceed a two-minute total warmup time. If it appears that the closing portion of the attack will require more than two minutes, let the 13-second hold period expire by releasing the armament trigger until closing time has been reduced. The 16-second preparation time must then be re-accomplished.
7. Brightness knob — As desired.
8. Track target with reticle.
9. When wings of target fill small reticle, armament trigger—Second detent.

WARNING

When making an attack on airplanes smaller than the target selector setting, firing range will be closer than normal.

ROCKET FIRING — OPTICAL

1. Radar master switch — STBY or ON, delay period elapsed.
2. Armament selector switch — ROCKETS — OPT. Reticle appears.
3. Armament safety switch — ARMED.
4. Igniter control switch — TACTICAL.
5. Target selector knob — HVY or MED.
6. Brightness knob — As desired.
7. Optical sight cage switch — CAGE.
8. Track target with reticle.

   Track the target with reticle, establishing and holding a smooth turning rate for four to six seconds minimum.
9. Optical sight cage switch — UNCAGE.
10. Track target with reticle.
11. When wings of target fill large reticle, armament trigger — Depress to second detent.

WARNING

When making an attack on airplanes smaller than the target selector setting, firing range will be closer than normal.

ABBREVIATED CHECK LIST

Refer to T.O. 1F-102A-(CL)1-1 for abbreviated check list of the above procedures.

MISCELLANEOUS EQUIPMENT

PILOT'S MASK DEFOG RHEOSTAT

The pilot's mask defog rheostat (figure 4-20) is located near the aft end of the left console. The control knob is placarded "Mask Defog." The word INCREASE and an arrow indicate that more heat is available to the mask when the knob is turned in a clockwise direction. On later airplanes,* to accommodate use of the MA-2 pressure helmet, the rheostat has been removed and a step-switch installed. The control knob and its function remains the same on these airplanes. The system is powered by the 28-volt dc essential bus.

CAUTION

Use the minimum amount of heat necessary to prevent or remove any accumulation of moisture on the faceplate. It is not necessary that heat be felt on the face. The faceplate heat rheostat should be at maximum heat just long enough to remove moisture and then returned to the minimum heat required to prevent moisture accumulation.

ANTI-G SUIT PROVISIONS

Low-pressure pneumatic system air, cooled by the cockpit air conditioning system, is used to pressurize the pilot's anti-g suit. Duct pressure is reduced to anti-g suit pressure by a type M-8 regulator in the left-hand console. The regulator maintains pressures of between 9 and 11 psi in the suit at high g loads. A valve in the regulator assembly automatically pressurizes the suit at g loads greater than 1½ to 2 g's. On some airplanes a manual control enables the pilot to control suit pressures within the pressure range of the regulator output. On other airplanes, varying pressures within the pressure range of the regulator are furnished automatically. The regulator also incorporates a manual control button. When the button is depressed, unregulated low-pressure pneumatic system air pressurizes the anti-g suit. When the button is released, the suit is depressurized. This feature of the regulator may be used by the pilot during extended flights to prevent fatigue by creating a massaging effect on the body. Operation of the manual button does not test the automatic feature of the regulator.

Anti-G Suit Control Knob

A large round knob (figure 4-20) on the left console, aft of the pilot's mask defog rheostat, is placarded "Anti-g Suit" and has arrows pointing to LO and HI positions. On some airplanes turning the knob clockwise increases g suit pressure.

AF 57-770 & on, & airplanes modified by TCTO 1F-102A-570.

mask defog and anti-g suit control panel

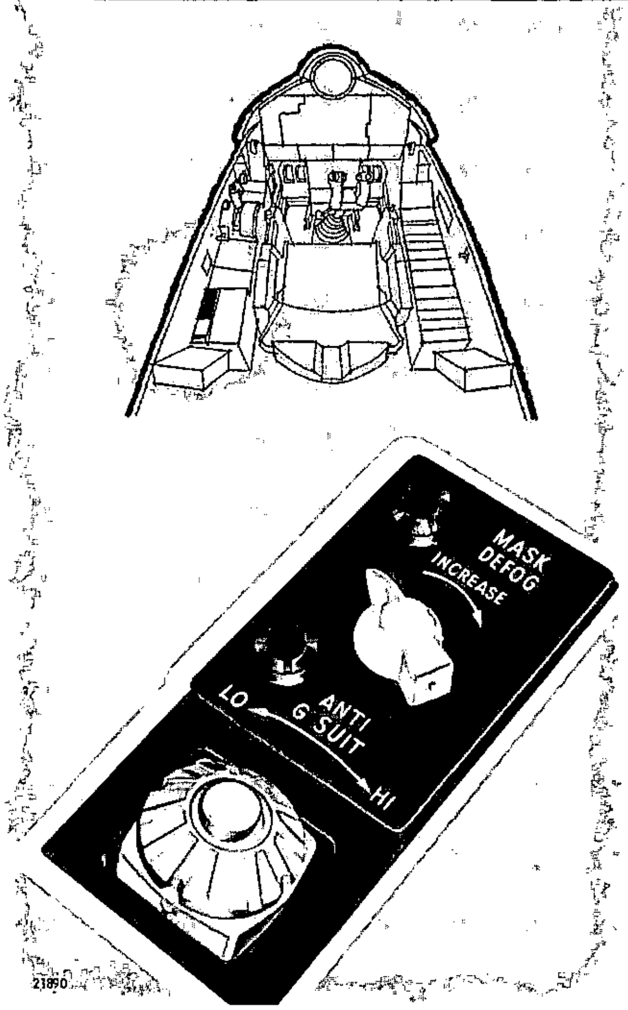


Figure 4-20

SPARE LAMPS

Spare lamps are stowed in holders at the aft end of the left console. The holder is placarded "Spare Lamps."

CHECK LIST

A check list (figure 1-3) is printed on a hinged shelf which folds out of the way when not in use. This shelf is located on the right side of the cockpit, just under the canopy sill. The check list can be illuminated at night by three lights on the shelf, controlled by the console light switch. The lights receive power from the 115-volt ac essential bus.

FLIGHT REPORT, MAP AND DATA CASE

A combination flight report holder and map and data case is located inboard of the spare lamp holder, at the aft end of the left console.

SHOULDER HARNESS STOWAGE STRAP

Some airplanes are equipped with strap assemblies on the left and right-hand sides of the cockpit sill, providing a means of stowing pilot's shoulder harness.

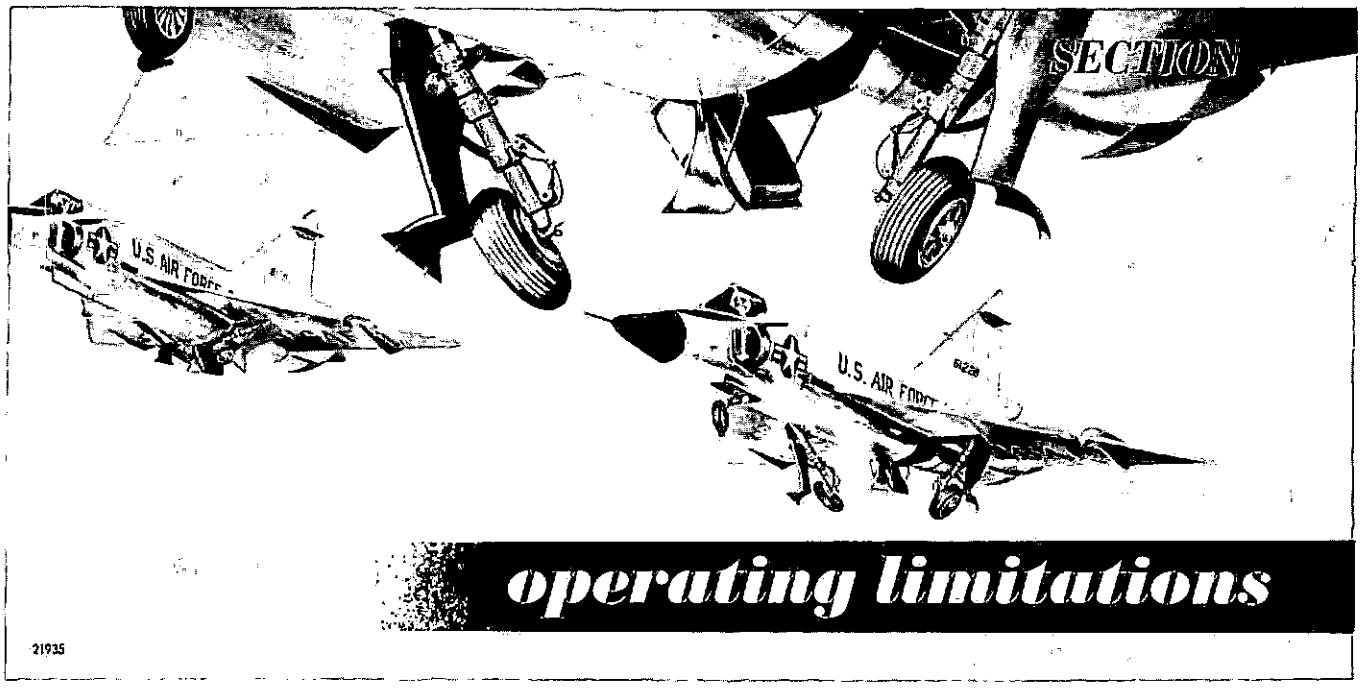


TABLE OF CONTENTS

Note

This section includes airplane and engine limitations which must be observed during normal operation. The high performance of this airplane demands that these limitations be closely adhered to.

ENGINE LIMITATIONS

Normal engine limitations are shown in figure 5-1. Additional information concerning operating limitations is given in figure 5-2 for starting, acceleration, idle, maximum continuous, military, and maximum thrust. On some airplanes,* these limitations are listed on a placard located on the left-hand side panel above the compass and altimeter correction cards. Maximum thrust is the thrust obtained by placing the throttle fully forward and outboard for afterburner operation. Military thrust is the thrust obtained by placing the throttle fully forward and inboard (nonafterburning).

**Airplanes modified by TCTO 1F-102-707.*

CAUTION

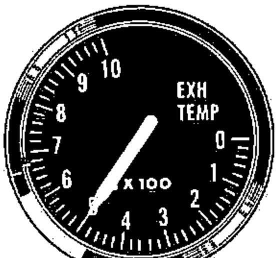
- During operation above 40,000 feet the engine may develop a rough operating condition, or compressor stall if the throttle is moved rapidly or rpm is reduced below 85%. Subsequent movement of the throttle may not eliminate this condition. The probability of a flameout also exists.
- In order to maintain engine EGT within the desired limits, all training flights should be limited to 45,000 feet as far as practical. When required to maneuver above 45,000 feet, climb must be accomplished at a minimum indicated Mach of .93.
- Under standard day conditions, the limit EGT will coincide with the ultimate ceiling of the airplane.
- When climbing above 40,000 feet, monitor EGT continuously. Any time the EGT is observed to be at or approaching limits, the throttle must be retarded or the indicated Mach number increased.

ENGINE OVERSPEED

The maximum permissible engine speed is 102% rpm. Any engine speed in excess of 102% should be noted on Form 781. If the engine speed exceeds 102% rpm, it must be inspected for damage. Speeds in excess of 104% rpm necessitate engine overhaul.

instrument markings

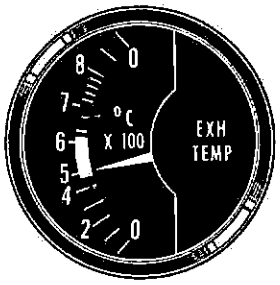
FUEL GRADE JP-4



EXHAUST TEMPERATURE
J57-23 ENGINE
AF 56-957 AND ON

- 470°C TO 610°C CONTINUOUS OPERATION.
- 670°C MAXIMUM DURING FLIGHT (EXCEPT ENGINE ACCELERATION.)
- 680°C MAXIMUM DURING ENGINE ACCELERATION. (2 MINUTE GROUND OPERATION) (2 MINUTES FLIGHT OPERATION)

REFER TO FIGURE 5-2 FOR ADDITIONAL INFORMATION.



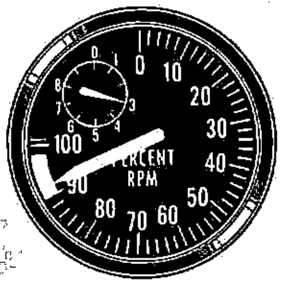
EXHAUST TEMPERATURE
J57-23 ENGINE
AF 53-1791 THRU 55-3464

- 470°C TO 610°C CONTINUOUS OPERATION.
- 670°C MAXIMUM DURING FLIGHT (EXCEPT ENGINE ACCELERATION.)
- 680°C MAXIMUM DURING ENGINE ACCELERATION. (2 MINUTE GROUND OPERATION) (2 MINUTES FLIGHT OPERATION)

REFER TO FIGURE 5-2 FOR ADDITIONAL INFORMATION.

HYDRAULIC PRESSURE

- 1000 PSI MINIMUM OPERATING PRESSURE
- 2500 PSI TO 3050 PSI CONTINUOUS OPERATION
- 3050 PSI MAXIMUM OPERATING PRESSURE



TACHOMETER

- 90% TO 98% CONTINUOUS OPERATION.
- 102% MAXIMUM PERMISSIBLE RPM.

MACHMETER-AIRSPEED

NOTE
MAXIMUM SPEED POINTER DOES NOT INDICATE ANY LIMIT.

240 KNOTS IAS MAXIMUM GEAR DOWN

REFER TO TEXT FOR MAXIMUM SPEED LIMITS.

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Figure 5-1

engine operating limits

J57-23	MAXIMUM OBSERVED EXHAUST GAS TEMP. (C)		TIME LIMITS	
	S.L. TO 30,000 FT.	ABOVE 30,000 FT.	GROUND OPERATION	FLIGHT OPERATION
OPERATING CONDITIONS: MAXIMUM (EXCEPT ENGINE ACCELERATION)	640	670	1 MINUTE	15 MINUTES
MILITARY	630	660	5 MINUTES ⚠	30 MINUTES
MAXIMUM CONTINUOUS	580	610	5 MINUTES ⚠	CONTINUOUS
IDLE	340	———	———	———
STARTING	630	630	———	———
ACCELERATION	680	680	2 MINUTES	2 MINUTES

NOTE
IF TEMPERATURE LIMITS ARE EXCEEDED, MAKE ENTRY IN FORM 781 STATING DURATION AND PEAK TEMPERATURES.

⚠ REFER TO "COOLING LIMITATIONS" TEXT THIS SECTION

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Figure 5-2

COOLING LIMITATIONS (GROUND OPERATIONS)

If the afterburner is not operated, the engine can be run for a total of five minutes at power above approximately 77% rpm. In addition, the engine can be run for one minute at maximum power (military plus afterburning). If these ground operating limits are closely approached, five minutes operation at idle thrust is required to allow for engine cooling before a schedule which will approach the limits can be repeated. If the engine has been operated for longer than one minute above 85% rpm in the five-minute period prior to shutdown, it will be necessary to operate the engine at idle rpm for a period of five minutes to stabilize engine temperatures. Immediately prior to shutdown, operate the engine at 70% rpm for 30 seconds to scavenge the oil system. Make entry on Form 781 if temperature limits are exceeded and state the time and temperature peaks.

AFTERBURNER LIMITATION

Do not attempt to start afterburner below 83% of military thrust.

EMERGENCY FUEL

Aviation gasoline (Mil-G-5572) or JP-5 (Mil-J-5624) may be used as an emergency fuel. Aviation gasoline, when used, should be the lowest available grade and the temperature of the fuel in the tanks at take-off must not be greater than 70°F.

CAUTION

When aviation gasoline is used it must be diluted 2.5% by volume with oil, Specification Mil-O-6082, grade 1100, and flight restricted to altitudes below 35,000 feet JP-5 fuel freezes at minus 55°F; therefore when using this fuel, flight must be restricted to altitudes where temperatures below this point will not be encountered.

ENGINE OIL

The maximum allowable engine oil consumption is four pints per hour.

AIRSPEED LIMITATIONS

MAXIMUM ALLOWABLE AIRSPEEDS

Some airplanes* are limited to 655 KIAS or Mach 1.25, whichever occurs first, due to inlet duct vibration at high indicated airspeeds. Other airplanes**, except AF 53-1813, are cleared of this structural deficiency and may be flown at design limit speed of 655 KIAS or Mach 1.5, whichever occurs first. AF 53-1813 is limited to 600 knots or Mach 1.25, whichever occurs first, due to a non-production vertical fin.

LANDING GEAR EXTENSION SPEED

The maximum allowable airspeed with landing gear extended is marked by a yellow radial on the airspeed indicator. The airplanes are limited to 240 KIAS with the landing gear extended.

DRAG CHUTE DEPLOYMENT SPEED

The maximum allowable airspeed at which the drag chute may be deployed is 160 KIAS. If the drag chute is deployed at speeds above this limit it will be automatically released from the airplane.

CAUTION

To prevent loss of the drag chute or to prevent slowing the airplane to excessively slow speeds, the drag chute must not be deployed in flight.

RAT LIMIT SPEED

The maximum allowable airspeed at which the RAT can be extended or operated is 345 KIAS. Extension of the RAT at speeds above 345 KIAS is likely to cause overspeed or structural damage. Satisfactory emergency hydraulic pressure will not be maintained at airspeeds below 125 KIAS.

OTHER OPERATIONAL LIMITATIONS

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS)

Operation of all modes of AFCS is temporarily prohibited below 12,000 feet at speeds above 480 KIAS on some airplanes.† Operation of the attack mode of AFCS is prohibited when external tanks containing fuel are installed.

**AF 53-1791, -1792, -1795 thru -1803, -1805 thru -1812, -1814 thru -1818, 54-1371 thru -1387, -1389, -1391 thru -1397, -1399, -1400, 55-3357 thru -3373, unless modified by TCTO 1F-102A-524.*

***AF 53-1793, -1794, -1804, 54-1388, -1390, -1398, -1401, 55-3374 & on & airplanes modified by TCTO 1F-102A-524.*

†80% limit (refer to PROHIBITED MANEUVERS, this Section).

ELECTRONICS COOLING

Ground Operation

During engine runup, the jet pump will cut out at approximately 72% rpm. Without jet pump cooling, radar equipment will begin to suffer damage within one minute. On the ground, operation of the radar with radar master switch in STBY or ON is prohibited, with engine rpm above 72%, except during takeoff. With engine rpm below 72%, operation of the radar in STBY or ON, is limited to seven minutes. When the radar is required as soon as the airplane is airborne, the following procedures should be followed:

1. If airplane is taxied with radar master switch in WARM, place master switch to ON and wait 30 seconds before advancing throttle for takeoff.
2. If airplane is taxied with radar master switch in STBY or ON (engine rpm below 72%) takeoff should be made within seven minutes of the radar master switch turned to WARM.

Inflight Operation

During hot day conditions (100°F or warmer at sea level), flight at either military power or loiter (maximum endurance) throttle settings below 5000 feet are restricted to seven minutes with the MG-10A radar master switch in ON or STANDBY positions. If flight at these throttle settings exceeds seven minutes, the radar master switch must be in OFF or WARM positions to prevent excessive heat from damaging electronics equipment.

Note

- The radar master switch must be in the WARM or OFF position during flight below 10,000 feet at maximum speed during hot weather operation to prevent damage to electronic equipment from overheating.
- If the radar master switch is placed in WARM position, the pilot assist and automatic functions of AFCS will be rendered inoperative.

EXTERNAL TANKS

External Tank Rubbing Strips

Installation of external tanks is prohibited until satisfactory rubbing strips are installed on the main landing gear fairing doors and the external tanks to prevent damage to the door and tank from contact with each other.

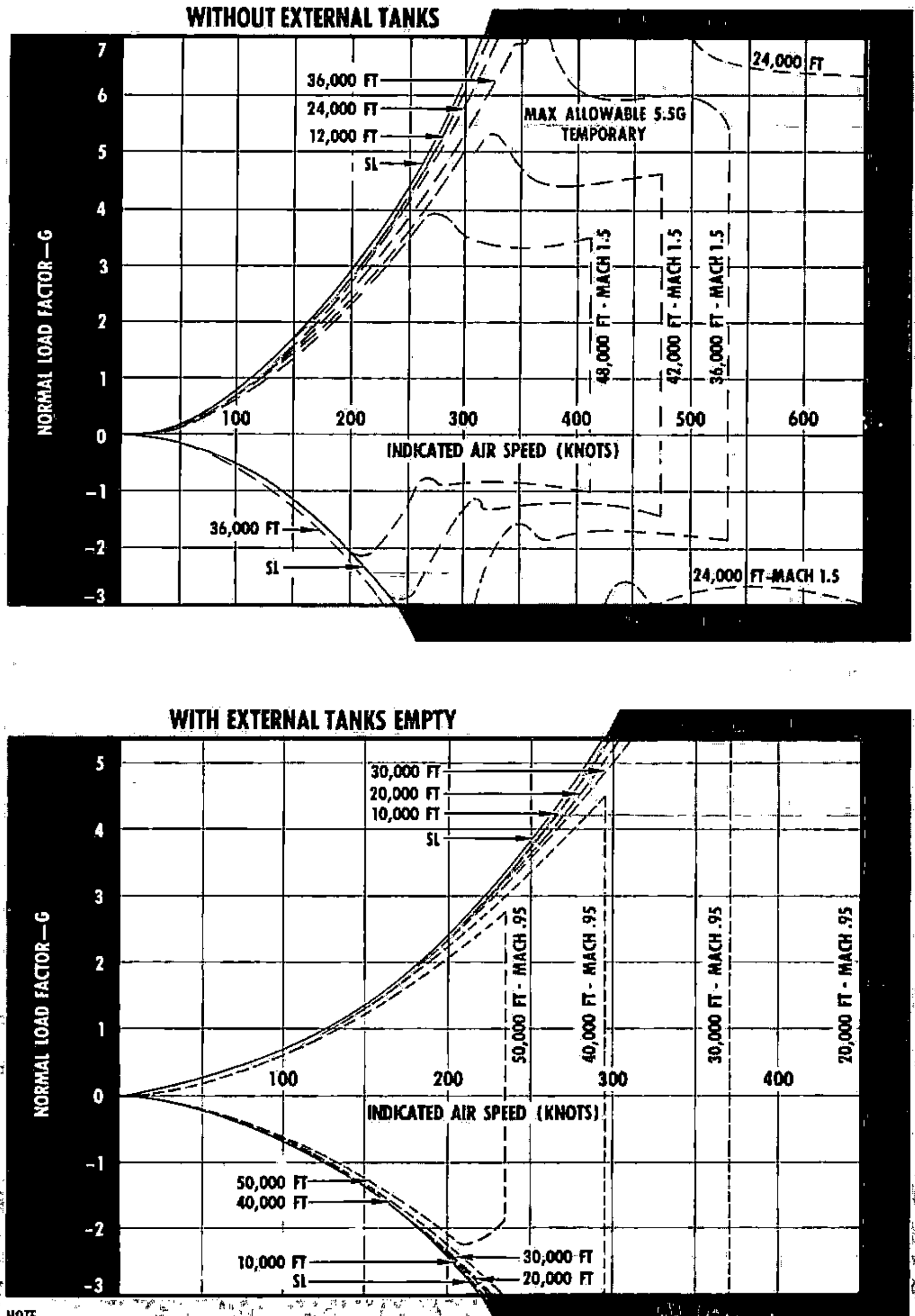
External Tank Fuel Transfer Switch

The external tank fuel transfer switch must remain in the OFF position until internal fuel quantity indicates 5500 pounds or less to maintain cg within limits. On airplanes not equipped with an external tank fuel transfer switch it will be necessary, prior to flight with full external tanks, to check center of gravity in accordance with

operating flight limits

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
FUEL GRADE: JP-4



NOTE
- FLIGHT CAPABILITY FOR VARIOUS ALTITUDES INDICATED BY DOTTED LINES.
- REFER TO TEXT FOR MAXIMUM ALLOWABLE AIRSPEEDS AND LOAD FACTORS.

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Figure 5-3

manual of weight and balance data, T.O. 1-1B-40, and either ballast the airplane or off-load fuel if it is found that the aft center of gravity is beyond the permissible limit.

External Tank Jettison Warning Lights and Safety Pin

On airplanes not equipped with external tank jettison warning lights and safety pin, inadvertent tank ejection is possible. Extreme caution will be required any time the cockpit is occupied during maintenance or ground handling. Until incorporation of the jettison warning lights and safety pin, which will prevent inadvertent tank ejection, tanks should be installed immediately prior to takeoff and removed immediately after the airplane is parked and the engine is shut down.

Maneuvering Limitations

With external tanks installed and containing fuel, maximum aileron deflection is limited to approximately 1/3 aileron (3°) above 270 KIAS. Refer to ACCELERATION LIMITATIONS, this Section, for symmetrical and asymmetrical (rolling) maneuvering limitations with external tanks installed.

Airspeed Limitation

Maximum allowable airspeed with external tanks (empty or containing fuel) is 450 KCAS or Mach .95, whichever occurs first. External tanks may be jettisoned at any speed within the operating speed range.

ARMAMENT EQUIPMENT

The following limitations must be observed when operating armament equipment:

1. Firing of the armament is prohibited at any time external tanks are installed.

Armament Gear Limitations

With missile bay doors open, avoid excessive roll and rudder induced maneuvers. Refer to ACCELERATION LIMITATIONS, this Section, for symmetrical and asymmetrical (rolling) maneuvering limitations with missile bay doors open.

Rocket Firing

Rocket firing from some airplanes* equipped with 2.75-inch rockets is prohibited until installation of a satisfactory debris deflector on each blast pan in the armament door. Possible damage which could be incurred when firing without a debris deflector is:

- Damage to the missile or missile radome.
- Dents or holes in the missile bay beams.
- Damaged electrical wiring, pneumatic lines, or hydraulic lines.

CAUTION

It is also possible for debris from a forward 2.75-inch rocket to prevent the aft rocket from leaving the tube, resulting in destruction of the armament bay door when aft rocket is ignited. For practice rocket firing missions, ensure that only the forward 2.75-inch rockets are loaded.

Note

Refer to the Confidential Supplement, T.O. 1F-102A-1A, for figure 5-3, Missile Firing Limitations.

TIRE LIMITATIONS

The maximum allowable tire ground speeds are:

1. With 18 ply rubber tread tires—174 knots.
2. With 20 ply rubber tread tires—195 knots (195 knots printed on the side wall).
3. With 20 ply fabric tread tires—217 knots (217 knots printed on the side wall).

Note

Refer to the Appendix for takeoff tire ground limit speeds.

ELEVATOR FEEL-FORCE REGULATION (SHELLY INTELLIGENCE UNIT)

If the pin on elevator feel-force intelligence unit is found to be extended or retracted (not flush with case of the unit) ±0.06 inch (approximately 1/16 inch) during exterior inspection, the airplane should not be flown in excess of Mach .90.

CAUTION

Shaft retraction directs unregulated ram air into the elevator feel-force cylinder. The increased pressure in the cylinder produces higher stick forces, but the airplane remains controllable at restricted speeds. If inspection shaft extends from hole more than 0.06 inch, the regulator must be replaced.

**AF 53-1791, -1792, -1794 thru -1796, -1798 thru -1805, -1807 thru -1818, 54-1371 thru -1389, -1391 thru -1400, -1402 thru -1407, 55-3357 thru -3464, 56-957 thru -1136 unless modified by TCTO 1F-102-609.*

SPEED BRAKES

Speed brakes operation is temporarily restricted, on some airplanes* as follows:

1. Flight with speed brakes open is prohibited at speeds above 300 KIAS.
2. Speed brakes should not be used during abrupt maneuvers below 300 KIAS.
3. Use of speed brakes should be avoided while flying in close proximity of other aircraft.

CANOPY

When taxiing airplanes equipped with a canopy hold button, do not exceed 60 knots ground speed or 90 KIAS, whichever is less. On airplanes without a canopy hold button, the canopy hold-open rod must be used and taxi speed limited to 30 knots ground speed. Taxi speeds in excess of the above may result in damage to the canopy or airplane structure.

PROHIBITED MANEUVERS

SPINS

Intentional spins are prohibited due to excessive loss of altitude and the probability of exceeding engine temperature limits. (Refer to SPINS, Section VI.)

NEGATIVE G MANEUVERS

Maneuvers at less than one g or at a negative g for more than three seconds duration are prohibited on airplanes without an oil recirculating system. The recirculating system will supply satisfactory oil pressure to the constant-speed electrical power system during negative g conditions. On modified airplanes** which incorporate a recirculating system, negative g maneuvers are limited to 15 seconds maximum, of which only 10 seconds may be at zero g. This limitation is necessary to ensure adequate engine lubrication.

OTHER MANEUVERING LIMITATIONS

AILERON DEFLECTION LIMITATIONS

Note

- There are no aileron deflection restrictions on any airplanes during takeoff and landing when airspeed is below 270 KIAS.
- At approximately ¾ aileron deflection, an additional 10 pounds resistance (feel force) is imposed by added spring tension and must be overcome to obtain full aileron.

**AF 53-1791 thru 57-823, 57-825 thru 57-830, -832, -833, -835, -837, -838, -840, & -841, unless modified by TCTO 1F-102-725.*

***AF 53-1791, -1793, 56-1083 & on, & airplanes modified by TCTO 1F-102-602.*

Airplanes With Enlarged Vertical Fin

1. Yaw damper and turn coordinator on:
   a. Above 450 KIAS below 20,000 feet—¾ aileron (½ aileron on some airplanes†).
   b. All other conditions—No restriction.
2. Yaw damper and turn coordinator off:
   a. Above 450 KIAS below 20,000 feet—½ aileron.
   b. All other conditions—¾ aileron except for takeoff and landing.

Airplanes With Small Vertical Fin

1. Yaw damper and turn coordinator on or off:
   a. Above 450 KIAS below 35,000 feet—¾ aileron.
   b. Above 480 KIAS below 12,000 feet—½ aileron.
   c. All other conditions—No restriction.

Yaw Damper and Turn Coordinator Off

When operating the airplane with the yaw damper and turn coordinator off, the following restrictions shall be observed to prevent exceeding the current structural limitations of the airplane:

1. Roll maneuvers shall be restricted to positive quadrant rolls only.

Note

A positive quadrant roll is a roll to a 90° bank either side of level flight or from a 90° bank in one direction to a 90° bank in the opposite direction passing through level flight position.

2. No uncoordinated maneuvers shall be performed except during takeoff and landing.
3. Rapid application of aileron shall be avoided except during takeoff and landing.
4. No intentional rolling pushdowns shall be performed.

Roll Maneuvers

When executing rolls through greater than 180°, elevator position should not be varied from the position selected at entry, except to prevent the airplane from exceeding limit asymmetrical load factors.

Note

If elevator is inadvertently applied during rolls and excessive pitch or yaw maneuvers are encountered, do not attempt to fight the response. Neutralize all controls until control is regained.

†80% load factor limit.

Airplanes With Small Vertical Fin

In addition to the other maneuvering limitations, the following restrictions apply to airplanes with the small tail:

1. Snap rolls or snap maneuvers are prohibited.
2. Uncoordinated maneuvers are prohibited.
3. High rates of roll must be avoided to prevent inertia coupling (see Section VI).
   a. Roll rates for positive quadrant rolls are not restricted (roll to a 90° bank either side of level flight or from a 90° bank in one direction to a 90° bank in the opposite direction passing through level flight position).
   b. For full roll maneuvers (360°) with the sideslip angle transducer inoperative (flight mode selector switch to DIRECT MAN), the airplane is limited to a ½ aileron deflection or a maximum roll rate of 80° per second. With sideslip angle transducer inoperative (flight mode selector switch to MAN), the airplane is limited to ¾ aileron deflection or a maximum roll rate of 120° per second.

ACCELERATION LIMITATIONS

Some airplanes are restricted to 80% of design load factors until certain structural modifications are accomplished, at which time they are cleared for operation at 100% design load factors. Those airplanes that are limited to 80% of design load factors are as follows:

- AF 54-1375, 54-1390, 55-3430, 56-992, and 56-995 unless modified by TCTO 1F-102-633A.
- AF 53-1799 unless modified by TCTO 1F-102-600.
- AF 53-1793, 53-1799, 53-1817, and 54-1403 unless modified by TCTO 1F-102-639.
- AF 53-1794, 53-1797, 53-1799, and 53-1806 unless modified by TCTO 1F-102A-538.

Figure 5-3 represents operating flight limits for symmetrical loads. Asymmetrical (rolling) maneuvers impose higher loads on the airplane and therefore require more restrictive g limitations. The following tables show symmetrical and asymmetrical maneuvering limitations:

Missile Bay Doors Open

Symmetrical		Rolling Pullouts
Negative	Positive	Positive
−0.8g*	+2.4g*	
−1.0g	+3.0g	

*80% Limits.

External Tanks Containing Fuel

Symmetrical		Rolling Pullouts
Negative	Positive	Positive
−0.8g*	+2.4g*	0 g to 1.8g*
−1.0g	+3.0g	0 g to 2.3g

*80% Limits.

External Tanks Empty or Over 4200 Pounds Internal Fuel

Symmetrical		Rolling Pullouts
Negative	Positive	Positive
−2.2g*	+4.2g*	0 g to 3.1g*
−2.7g	+5.3g	0 g to 3.9g

*80% Limits.

Clean Airplane, Less Than 4200 Pounds Fuel

Symmetrical		Rolling Pullouts
Negative	Positive	Positive
−2.4g*	+5.6g*	0 g to 4.0g*
−3.0g	+7.0g	0 g to 5.0g

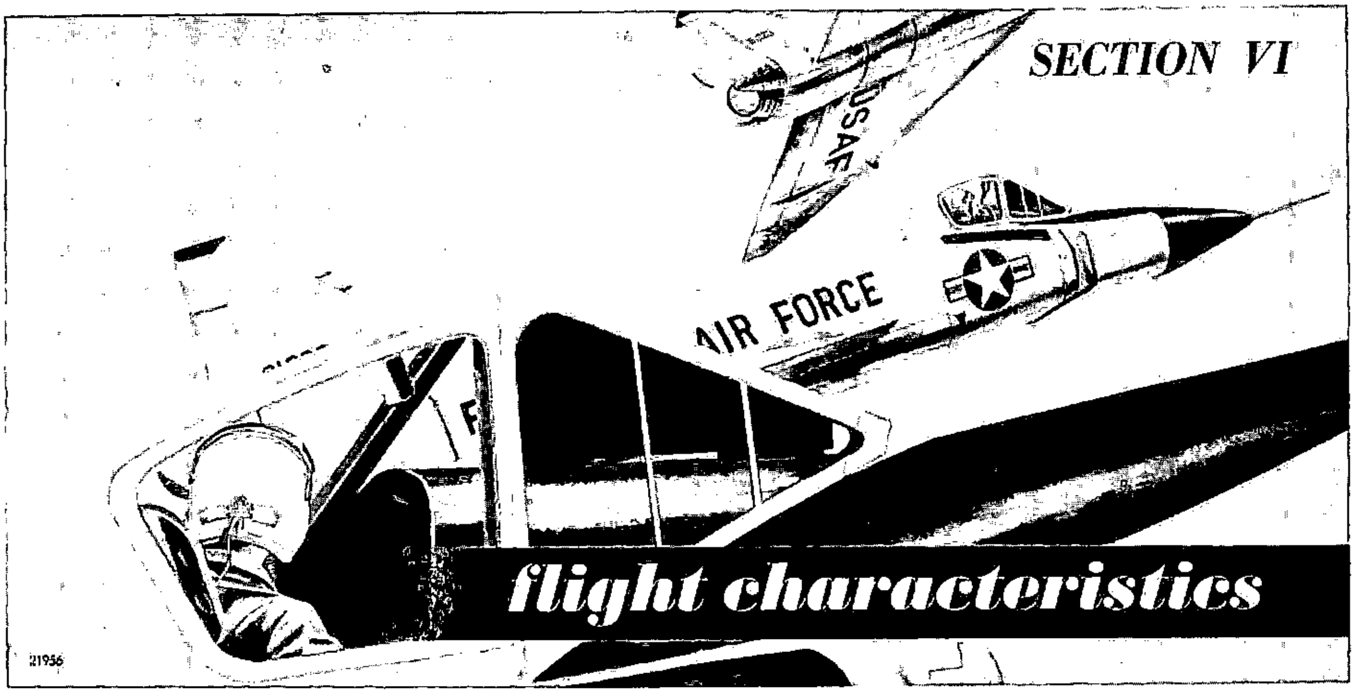
*80% Limits.

CENTER OF GRAVITY LIMITATIONS

The only in-flight control of cg position is the firing of armament and the transfer of external fuel. (Refer to EXTERNAL TANKS, this Section.) The cg position at takeoff is therefore important and should be closely checked and maintained within limits either by armament loading or ballast. CG position at takeoff approaches the aft limit and moves forward during flight and is normally near the forward limit for landing. Firing of armament soon after takeoff may result in the cg position shifting beyond the aft limits. (Refer to Handbook of Weight and Balance Data, T.O. 1-1B-40.)

WEIGHT LIMITATIONS

The design of the airplane precludes the possibility of overloading; therefore, the maximum gross weight will not be exceeded for takeoff as long as standard armament and full external tanks are carried. At landing weights (less than 22,000 pounds) the rate-of-sink at touchdown is limited to a maximum of 540 feet per minute, to prevent structural damage to the landing gear. If the gross weight at touchdown exceeds 22,000 pounds or if empty external wing tanks are still on, the designed rate of sink at touchdown is 300 feet per minute. Refer to FLIGHT WITH EXTERNAL TANKS in Section VI for additional information.



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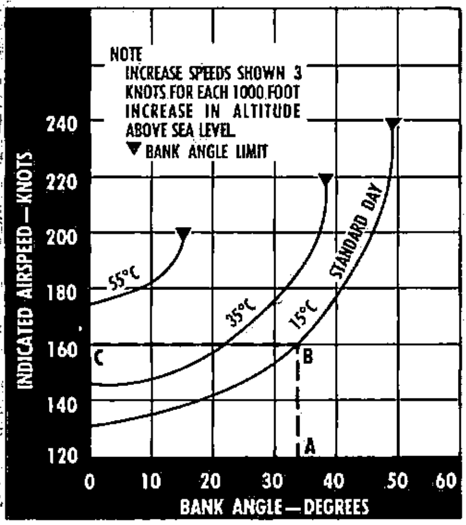
The airplane has a conventional turbojet airplane response to change of thrust; however, the change of thrust in the operating range can be relatively large for a small throttle movement due to engine bleed valve operation. Optimum performance is obtained when the cg is maintained close to the aft limit due to reduced trim drag. The use of elevons requires no unusual flying techniques as stick movement for a desired maneuver is the same as it would be if conventional elevator and ailerons were employed. Control response is good at all speeds; however, a "snaking" motion appears in the transonic speed range. This snaking or oscillating motion is damped out by the yaw damper system. In addition to the yaw damper (which incorporates a turn coordination system) a pitch damper is installed to minimize pitch oscillations. The maneuvering capabilities of the airplane are generally good; however the roll restrictions as established in Section V are imposed to provide satisfactory operation of the airplane and, at the same time, reduce the possibility of encountering inertial coupling. Later airplanes are provided with an increased area vertical fin to eliminate roll restrictions imposed because of inertial coupling dangers. Flight tests indicate that airplanes with the increase of 40% vertical fin area have less inertial coupling effects and better handling and general flying qualities than the earlier airplanes without this design improvement. With incorporation of the increased area fin, performance of the airplane is not perceptibly changed. The general handling characteristics of the airplane during flight with a dead engine are good in the low speed ranges. However, when a flameout occurs (or is induced) at airspeeds above 220 KIAS, the interruption of airflow through the engine inlet ducts will create a duct rumble which causes an annoying buffet or heavy vibration within the airplane. The severity of this disturbance will increase proportionally to the indicated airspeed, and is only slightly perceptible at best glide speed of 220 KIAS.

STALLS

The handling characteristics of this airplane at low speeds are excellent, with good control response below the recommended minimum speeds of the airplane, and no "stall," in the usual sense of the word, is encountered. A stall in this airplane may be defined as loss of control about any one axis. See figure 6-1 for the recommended minimum speeds based on normal gross weight and sea level altitude. The data are presented for maximum and military thrust and show the variation of minimum speed with bank angle, load factor and ambient air temperature. A correction factor is included to account



LEVEL FLIGHT MINIMUM SPEEDS WITH MILITARY THRUST.

BANK ANGLE	LOAD FACTOR
0°	1.0 G
30°	1.2 G
45°	1.4 G
60°	2.0 G

EXAMPLE

A INITIAL BANK ANGLE =33°.
B OUTSIDE AIR TEMPERATURE =15°C.
C MINIMUM AIRSPEED =160 KNOTS IAS.

ALTITUDE REQUIRED TO RECOVER FROM DESCENT IN ADJUSTING THRUST FROM THAT SPECIFIED TO MILITARY THRUST

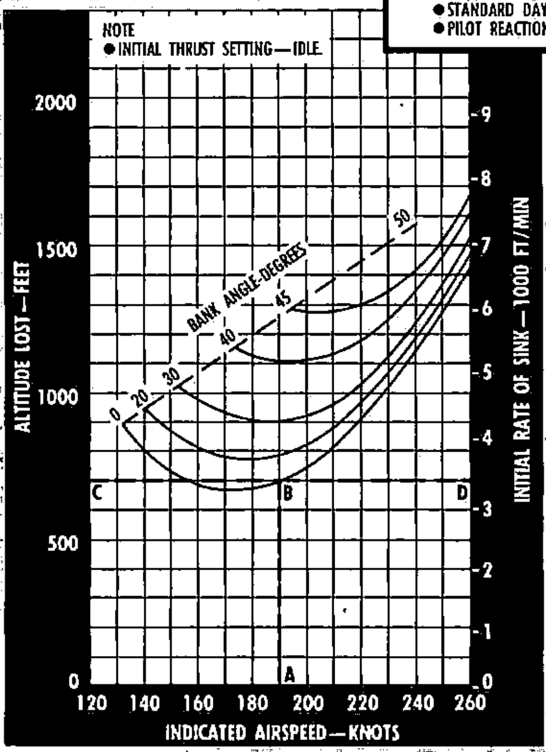
EXAMPLE

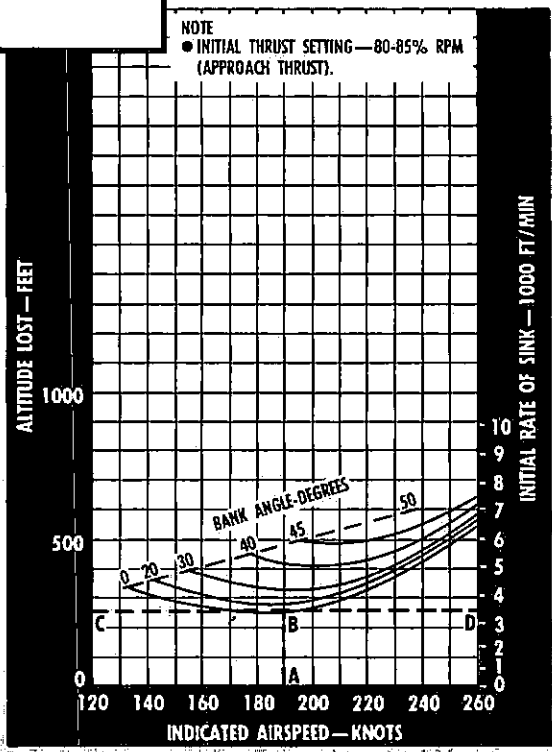
A. INITIAL AIRSPEED =190 KNOTS IAS.
B. INITIAL BANK ANGLE =0°.
C. ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT =700 FEET.
D. INITIAL SINK RATE =3300 FT/MIN.

NOTE
- ALTITUDE LOST IS HIGHLY DEPENDENT ON PILOT TECHNIQUE AND CURVES ARE THEREFORE APPROXIMATIONS.
- ----LEVEL FLIGHT MINIMUM SPEED (MILITARY THRUST).
- STANDARD DAY.
- PILOT REACTION TIME =2 SECONDS.

EXAMPLE

A INITIAL AIRSPEED =190 KNOTS IAS.
B INITIAL BANK ANGLE =0°.
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT =250 FEET.
D INITIAL SINK RATE =3,100 FT/MIN.





21957-1

Figure 6-1

minimum speeds

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
FUEL GRADE: JP-4

CONDITIONS

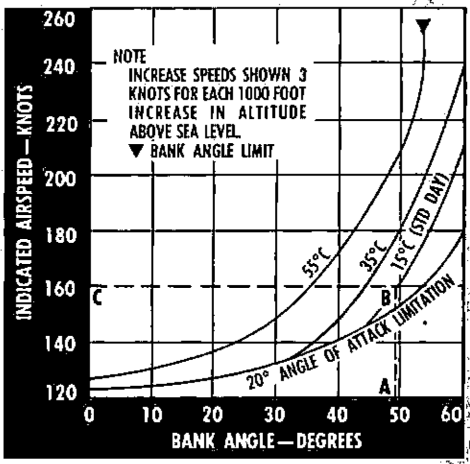
- GROSS WEIGHT EQUALS 27,500 LBS
- C G 27.5%
- AMBIENT TEMPERATURE AS INDICATED
- ALTITUDE—SEA LEVEL

CONFIGURATION

- LANDING GEAR DOWN
- SPEED BRAKES OUT

LEVEL FLIGHT MINIMUM SPEEDS WITH MAXIMUM THRUST

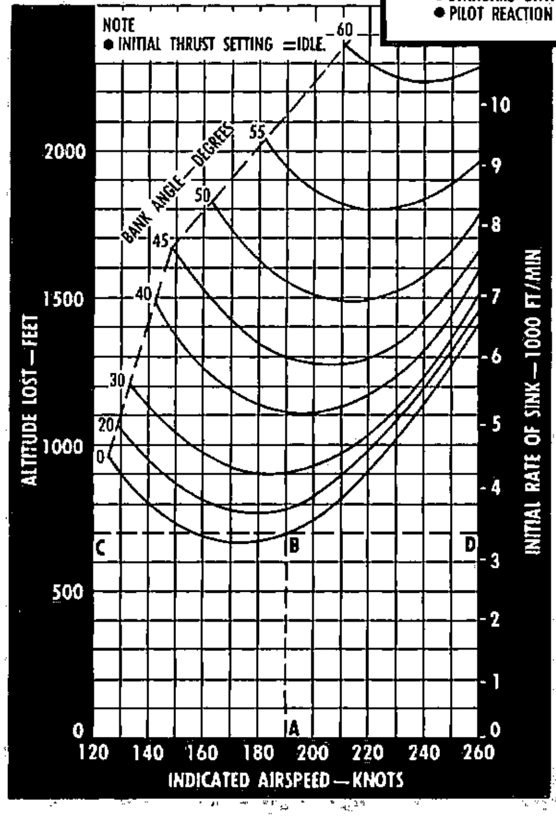
EXAMPLE
A INITIAL BANK ANGLE =49°.
B OUTSIDE AIR TEMPERATURE =15°C.
C MINIMUM AIRSPEED =160 KNOTS IAS.



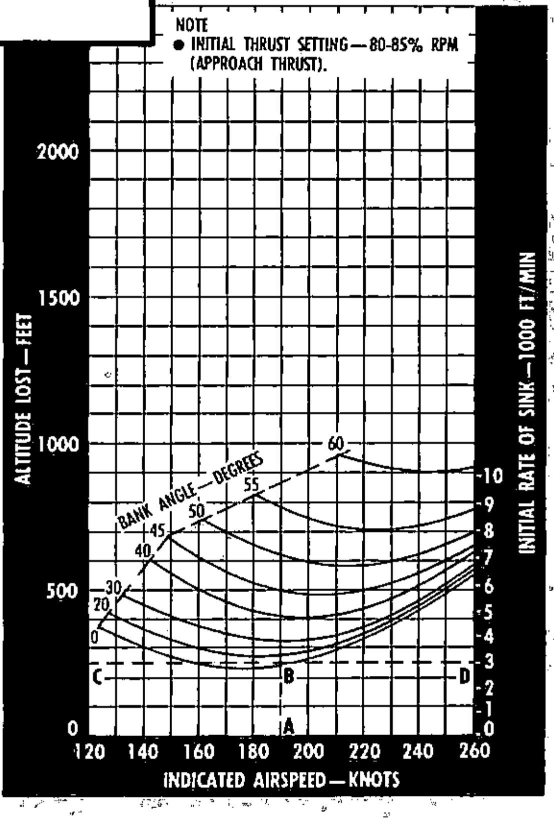
ALTITUDE REQUIRED TO RECOVER FROM DESCENT IN ADJUSTING THRUST FROM THAT SPECIFIED TO MAXIMUM THRUST

NOTE
- ALTITUDE LOST IS HIGHLY DEPENDENT ON PILOT TECHNIQUE AND CURVES ARE THEREFORE APPROXIMATIONS.
- ---LEVEL FLIGHT MINIMUM SPEED (MAXIMUM THRUST).
- STANDARD DAY.
- PILOT REACTION TIME =2 SECONDS.

EXAMPLE
A INITIAL AIRSPEED =190 KNOTS IAS.
B INITIAL BANK ANGLE =0°.
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT =700 FEET.
D INITIAL SINK RATE =3300 FEET/MIN.



EXAMPLE
A INITIAL AIRSPEED =190 KNOTS IAS.
B INITIAL BANK ANGLE =0°.
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT =250 FEET.
D INITIAL SINK RATE =3,100 FEET/MIN.



21957-2

for altitudes greater than sea level. Additional data are presented in the form of rate-of-sink and altitude required to recover from descent with idle thrust and approach thrust (80-85% rpm) using maximum thrust are military thrust for recovery. During level flight, at a gross weight of 25,000 pounds, a warning buffet starts at approximately 170 KIAS, and increases slightly to a moderate buffet at 125 KIAS. With military thrust at 30,000 feet, a rate-of-sink develops at approximately 160 KIAS, and very high rates-of-sink are encountered at lower speeds. As the airspeed is reduced below 110 knots, the ailerons should not be used for lateral control of the airplane as the adverse yaw induced may inadvertently cause a spin entry. Rudder control alone should be used to maintain lateral directional control to stall speed of 95 KIAS. At this speed, the airplane may fall off in a nose-down spiral or, if back pressure is held, enter a spin. Recovery from a stall, or the approach to a stall, is easily effected by relaxing back pressure on the stick; however, considerable altitude (up to 6000 feet) will be lost. The recommended minimum speed for this airplane is 125 knots. Below this speed, lateral control effectiveness falls off rapidly and very high sink rates are encountered (up to 10,000 feet per minute). At altitudes above 35,000 feet, engine compressor stalls are usually encountered below 115 KIAS, and if allowed to persist, the exhaust gas temperature limits may be exceeded (Refer to COMPRESSOR STALL, Section III.)

SPINS

Note

Intentional spins in this airplane are prohibited due to excessive loss of altitude and the probability of exceeding engine temperature limitations.

The possibility of entering inadvertent spins cannot be overlooked in any operational fighter-type airplane, although the pre-spin warnings of the F-102A should greatly reduce the probability of entering a spin. Spin warning characteristics, as in stalls, are airframe buffet, loss of lateral directional control, high rate of sink, and probable engine compressor stalls. As the airspeed is allowed to decrease well below the recommended minimum airspeed or a g load is held through severe buffet, the airplane will fall off laterally in either direction and spin if continued back stick pressure is held. The airplane will spin in an oscillatory manner and may change direction of rotation several times when up elevator and neutral aileron controls are held. If aileron is applied and held in one direction, the airplane will spin in the opposite direction, i.e., left aileron will induce a spin to the right. Rudder deflection at spin entry will determine the initial direction of the spin if applied or held in before the angle of attack has reached 32° nose up.

SPIN RECOVERY

The recommended procedure for recovery from an inadvertent spin is as follows:

1. Throttle IDLE.
2. Release all controls until the nose has dropped to well below the horizon and the rotation has stopped.
3. Apply power and up elevator to complete the recovery only after 130 KIAS has been reached. At this airspeed the elevons and rudder are effective and may be used to maintain desired altitude.

If the rotation continues for more than one turn after release of the controls, the recovery can be effected immediately by applying one-half aileron in the direction of the spin (left aileron to recover from a spin to the left).

Note

- Oscillation in roll may mask the direction of rotation; if the aileron does not cause recovery in one turn, reverse the direction of application.
- As the spin breaks, the nose will pitch down and rotation will stop simultaneously. As soon as this occurs, neutralize aileron to stop roll rate.

After recovery from a spin, during which engine compressor stalls were encountered, return to base at minimum power and land. If the airplane has been spun or fully stalled this fact shall be entered on Form 781 so that the engine can be inspected for damage.

FLIGHT CONTROLS

SPEED BRAKES

The hydraulically operated speed brakes may be used to slow the airplane at all speeds. When extending the speed brakes, a slight nose-up airplane trim change occurs which becomes more pronounced at higher airspeeds. When retracting the speed brakes, a nose-down trim change occurs. Refer to the Appendix for descents with speed brakes and to Section V for speed brakes limitations.

LONGITUDINAL CONTROL FORCES

Stick force is proportional to stick deflection and varies with altitude and Mach number. Caution should be used during accelerated flight conditions as some elevator stick force lightening will occur as speed decreases in the transonic region (Mach .9 to 1.05).

LATERAL CONTROL FORCES

Lateral controls are designed so that stick force is a constant function of stick displacement. However, ailerons are most effective at approximately Mach .9 and decrease in effectiveness with an increase or decrease of speed. Control response near the neutral position is inherently sensitive but not excessive. The new pilot checking out may have a tendency to overcontrol slightly before becoming accustomed to stick feel in the proximity of the neutral position.

RUDDER CONTROL FORCES

The rudder pedal force varies with indicated airspeed. Full rudder deflection is available at approach and taxi speeds with approximately 50 pounds pedal force. At high indicated airspeeds approximately five times this force is required for one-quarter rudder deflection.

LEVEL FLIGHT CHARACTERISTICS

LOW SPEED

The recommended airspeeds for takeoff, approach and landing, and their relationship to gross weight, ambient air temperature and altitude contained in this handbook are above any unsafe flight attitude. Handling characteristics within these speed ranges are excellent and control response is positive. However, high angle of attack, associated with low speeds in the approach and landing configuration, together with the effective control response may mask the development of high sink rates. Since this airplane does not exhibit conventional stalling characteristics, it is possible, in high-altitude flight, particularly in maneuvering, to generate a condition of high drag which is beyond the capability of the engine to overcome. The possibility of entering this condition must be anticipated, since recovery may require more altitude than is available, particularly in the landing approach. Figure 6-1 shows the relationship of rate-of-sink and altitude lost in recovery from descent in the low-speed region. Data are included for initial thrust settings of idle and approach thrust (80-85% rpm) using maximum and military thrust for recovery. Engine thrust is not always a substitute for airspeed, particularly in the low speed range, so do not rely on engine thrust alone to get out of trouble. As is shown in figure 6-1, there is a minimum speed associated with the various airplane and atmospheric conditions of operation. It is very important to obtain at least this speed prior to any high-altitude maneuvering flight. A speed higher than the minimum is advisable at least until more complete familiarity with the airplane characteristics is attained. Although the design of the airplane provides for good forward visibility during all normal flight, the wide range of the obtainable angle of attack (angle of attack in excess of about 20°) at low speeds makes it possible to obstruct forward visibility along the flight path with the nose of the airplane. Refer to Appendix for takeoff and landing speeds.

MEDIUM SPEED

Medium speed flight characteristics are conventional.

HIGH SPEED

Whenever flight into the supersonic region is anticipated, the pitch and yaw damper systems should be engaged to aid in maintaining coordinated flight through the high-speed stages.

CAUTION

- Flight into the transonic and supersonic speed ranges should not be attempted with the yaw damper inoperative. With the yaw damper disengaged, a snaking or oscillating motion will appear in the transonic speed range.
- If flight with the damper systems inoperative is mandatory, refer to OTHER MANEUVERING LIMITATIONS, Section V.

As speed increases through the transonic range and into the supersonic speed range, a nose-down trim change (nose tucks down slightly) occurs. Some stick force lightening may occur when decelerating through the Mach 1.0 and Mach .95 region. Because of the excellent response to aileron action throughout the entire speed range of the airplane, large aileron deflections in the high-speed range should not be attempted until the pilot is thoroughly familiar with the response. When using maximum thrust at low altitude, it is possible to exceed temporary or design speed limitations (refer to Section V).

Note

See figure 6-4 which shows maximum Mach number capabilities.

MANEUVERING FLIGHT

The present-day design requirements for an interceptor are principally for speed with reduced capabilities for maneuvering. With this design the mass of the fuselage as compared to short wingspread creates a new phenomenon. This phenomenon, inertia coupling, is discussed in the following paragraph.

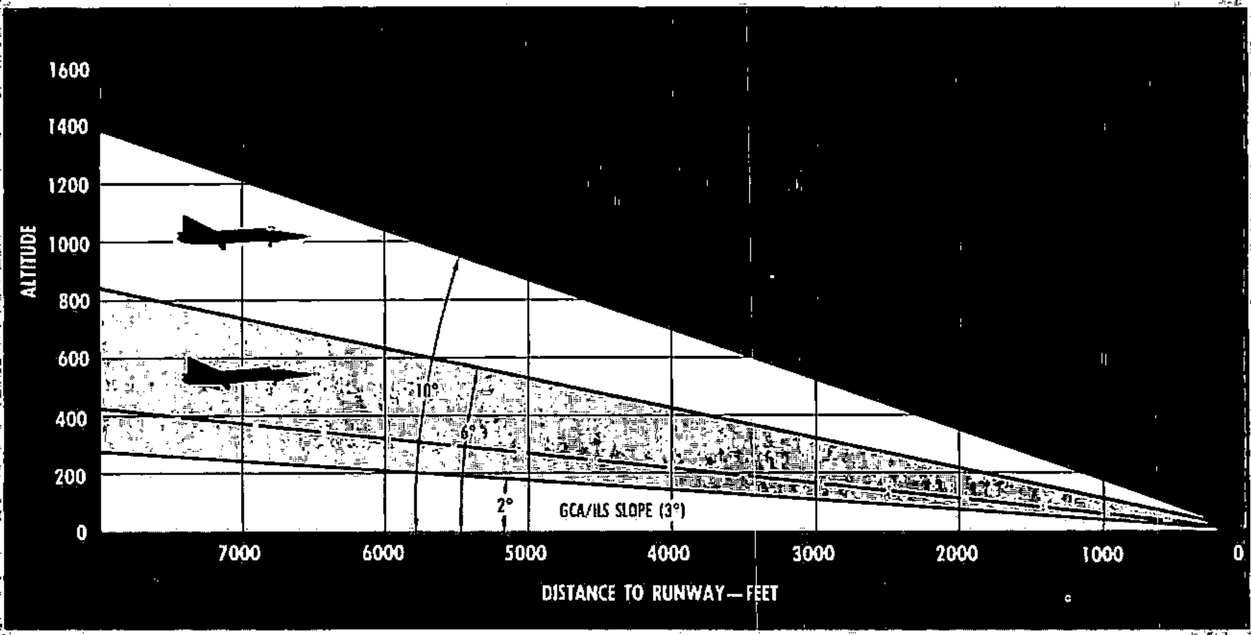
INERTIA COUPLING

The large lateral and normal load factor response during an abrupt aileron roll maneuver has been referred to as the inertial coupling effect. When a maneuver is performed so that a combined rolling and yawing rotational velocity exists, a pitching motion results due to the inertia effects. Similarly a combined pitching and rolling rotational velocity results in a yawing motion. This phenomenon can be illustrated by considering a rod to represent the distributed mass along the length of the fuselage. If the rod is rotating about an axis which is inclined to the axis of the rod at some pitch angle, the centrifugal force or inertia effects will tend to cause the rod to increase this pitch angle (i.e. becomes a fly weight). The aerodynamic characteristic which opposes the increase of this pitch angle is the static stability of the airplane. This effect is present on all airplanes but has become more pronounced on present day design. The automatic stability augmentation system of this airplane provides adequate protection from structural damage provided that the abrupt aileron maneuvers are performed

characteristics at low speeds

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
FUEL GRADE: JP-4



LANDING FLARE CAPABILITY IS INCREASED WITH HIGHER APPROACH SPEEDS. EXCESSIVE ANGLE OF ATTACK (AT LOW SPEEDS) WILL INCREASE THE DRAG TO A POINT THAT CAN BE AS GREAT AS THE THRUST AVAILABLE.

CRITICAL
TOLERABLE
DESIRED

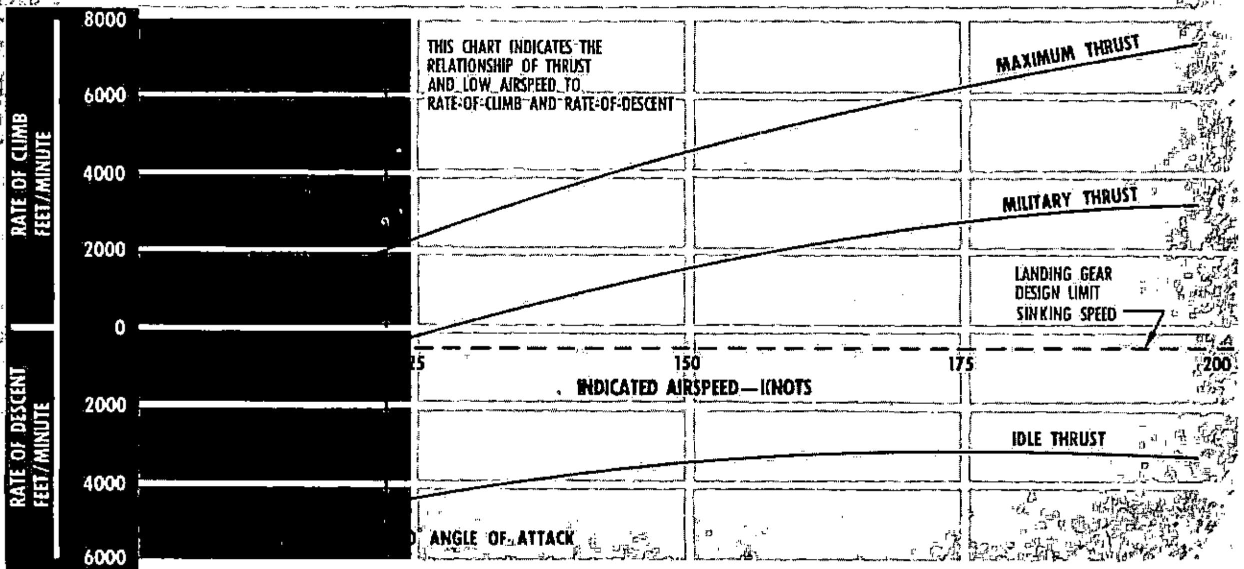


Figure 6-2

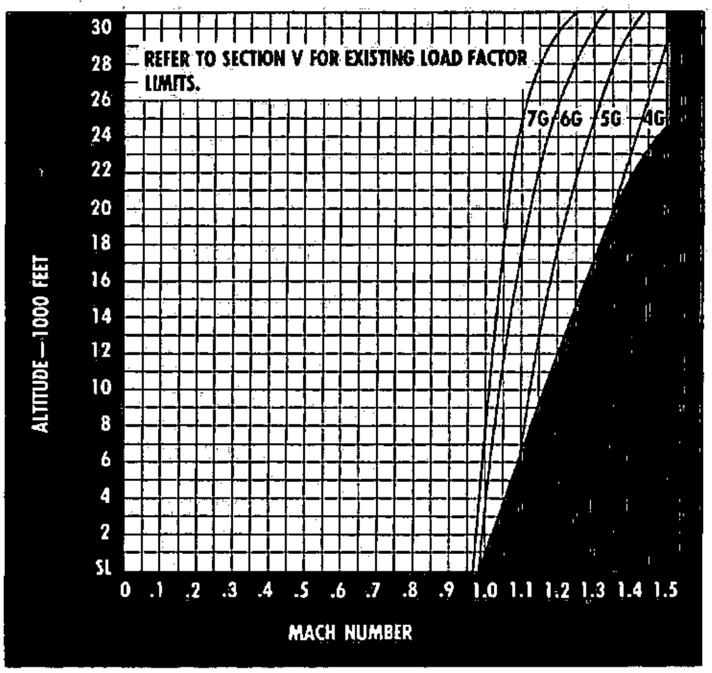
hinge moment limitation

MODEL: F-102A
DATE: 26 MARCH 1957
DATA BASIS: FLIGHT TEST

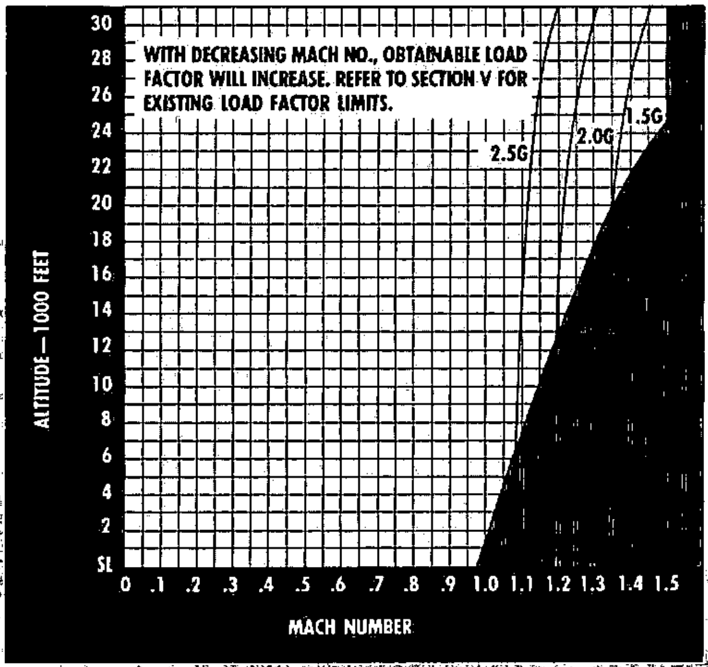
CONFIGURATION: CLEAN
GROSS WEIGHT: 25,000 POUNDS
C.G.: 28.5% M.A.C.

ENGINE: J57-23
FUEL GRADE: JP-4

IDLE THRUST



OBTAINABLE LOAD FACTORS (BOTH HYDRAULIC SYSTEMS OPERATING)



OBTAINABLE LOAD FACTORS (SINGLE HYDRAULIC SYSTEM OPERATING)

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Figure 6-3

maximum speed capabilities

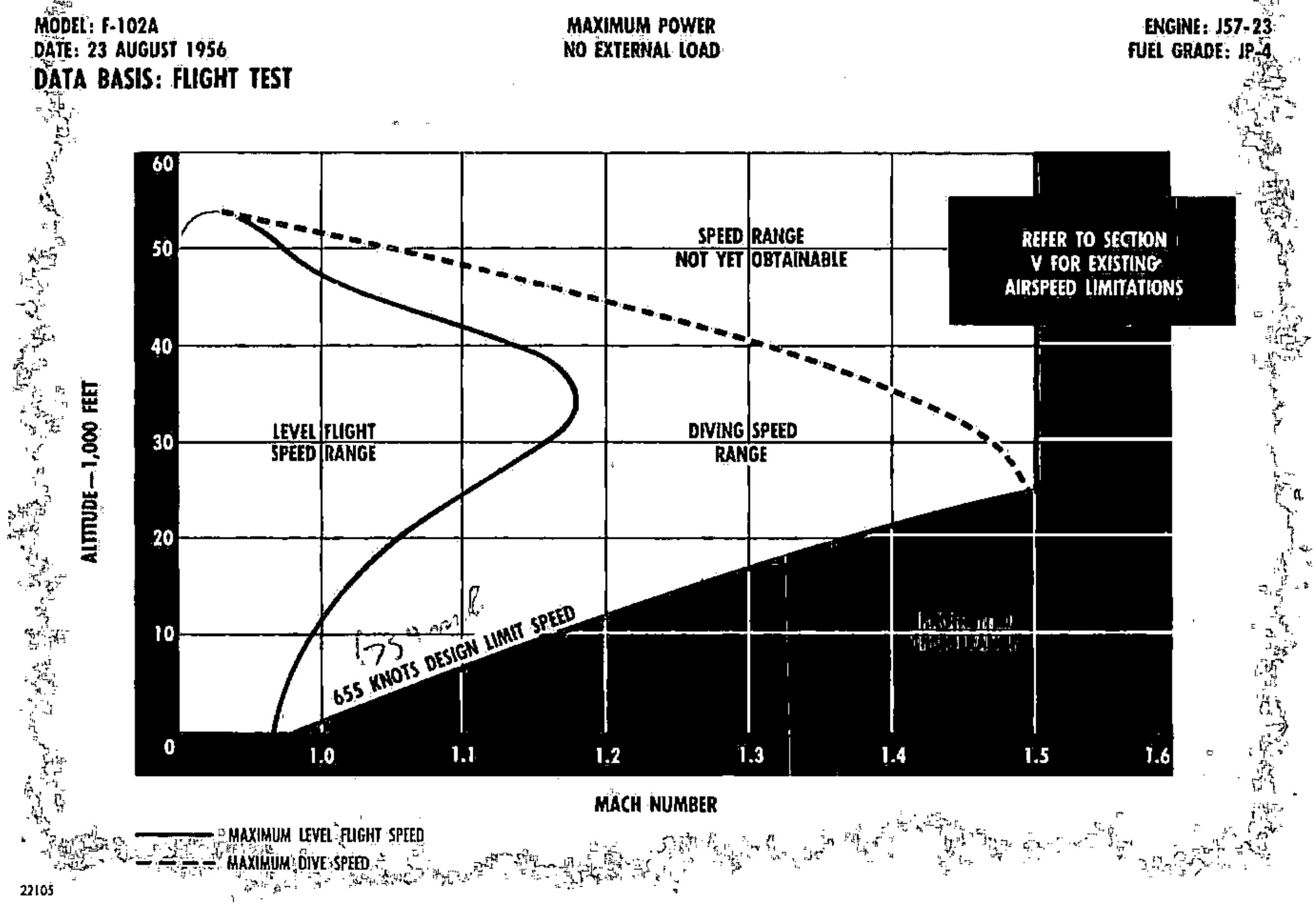


Figure 6-4

in accordance to the operating limitations. However, the following provides additional information regarding the roll behavior of the airplane:

a. Adverse yaw (against the turn) characteristics exist with nose-high angles of attack during rolling maneuvers, whereas complementary (with the turn) yaw characteristics exist with low angles of attack. The crossover between adverse yaw and complementary yaw characteristics for level flight entry conditions occur at approximately 268 KIAS.

b. Application of back stick (up elevator) aggravates the divergence tendency of the adverse yaw flight conditions, whereas forward stick (down elevator) aggravates flight conditions where complimentary yaw characteristics exist.

c. During the critical response roll maneuver, the normal load factor is opposite to the pitch rate, i.e. negative load factor with positive pitch rate or vice versa. Under these conditions the application of elevator governs the pitch rate and can greatly increase the normal load factor.

If limit roll rates are exceeded and inertia coupling is encountered, the recommended control action is to use aileron only to stop the roll rate. Do not attempt to fight the pitch and yaw, but neutralize the elevator and rudder.

DIVES

Diving flight characteristics are similar to high-speed level flight characteristics. A tuck-under tendency noted in the transonic and lower supersonic speed ranges is also present when diving. To minimize the tuck-under when dive speed approaches Mach 1.3, correction should be made with manual trim or back stick application.

DIVE RECOVERY

When diving in the speed range between maximum level flight speed and limit dive speed, a roll-off, generally to the right (approximately 20° per second), will be encountered during recovery below 35,000 feet, when attempting a maximum load factor pullout. The roll-off is caused by some airplane asymmetry which requires aileron trim.

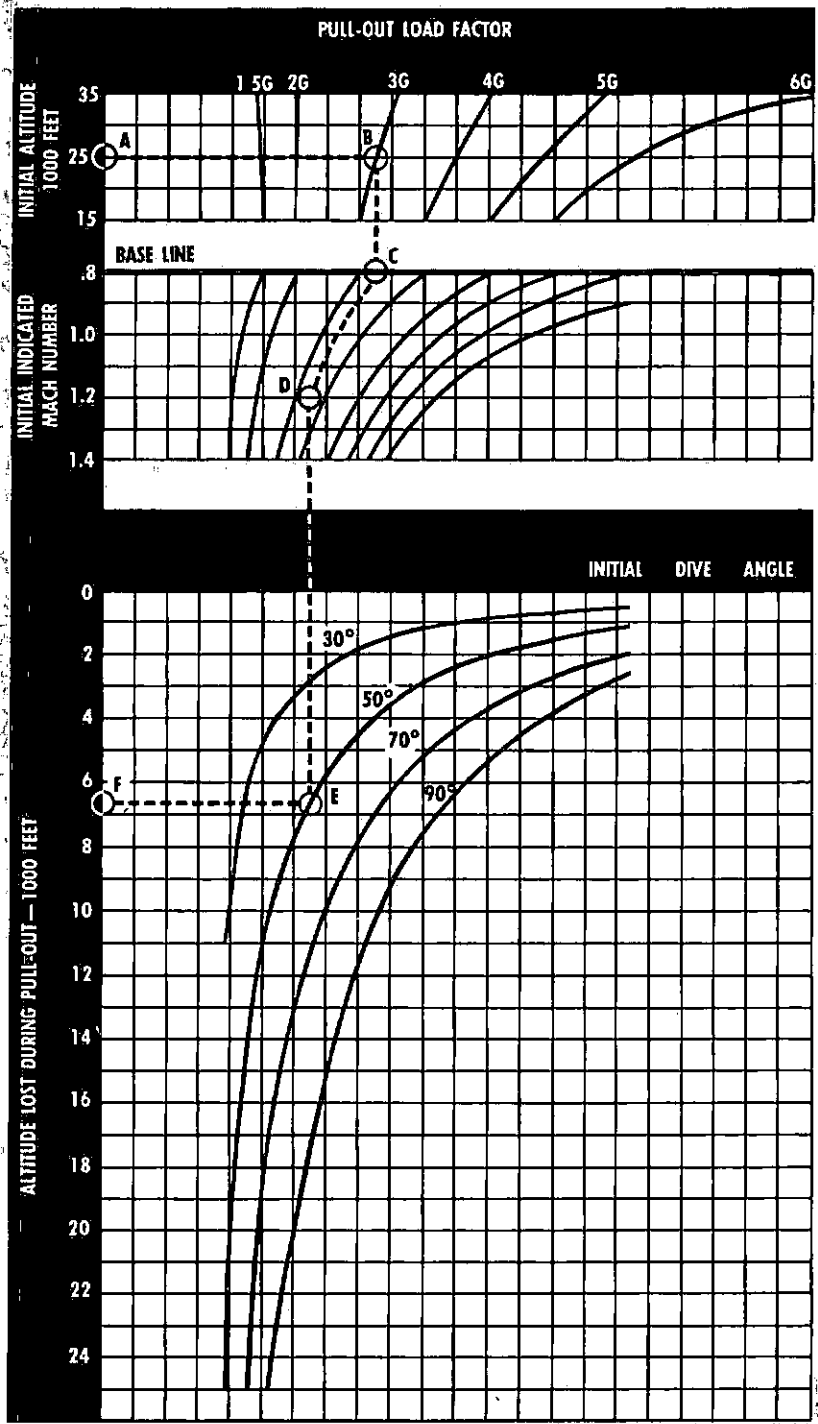
dive recovery chart

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: SPEED BRAKES OPEN
(GROSS WT.: 24,500 LB.)
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4

IDLE THRUST



EXAMPLE

A. IS ALTITUDE AT START OF PULL-OUT (25,000 FEET).
B. IS PULL-OUT LOAD FACTOR (3G).
C. IS MACH NUMBER BASE LINE.
D. IS MACH NUMBER AT START OF PULL-OUT (1.2).
E. IS DIVE ANGLE AT START OF PULL-OUT (50°).
F. IS ALTITUDE LOST DURING PULL-OUT (6,700 FEET).

21959

Figure 6-5

The trim is lost when maximum elevon hinge movement is obtained, resulting in insufficient elevator deflection to develop full g capabilities. See figure 6-3 for maximum load factors obtainable before encountering roll-off. Dive recovery should be executed with caution since an accelerometer is not installed on all airplanes, and no dive should be attempted when pullout would exceed a load factor of 4.0 g's.

Note

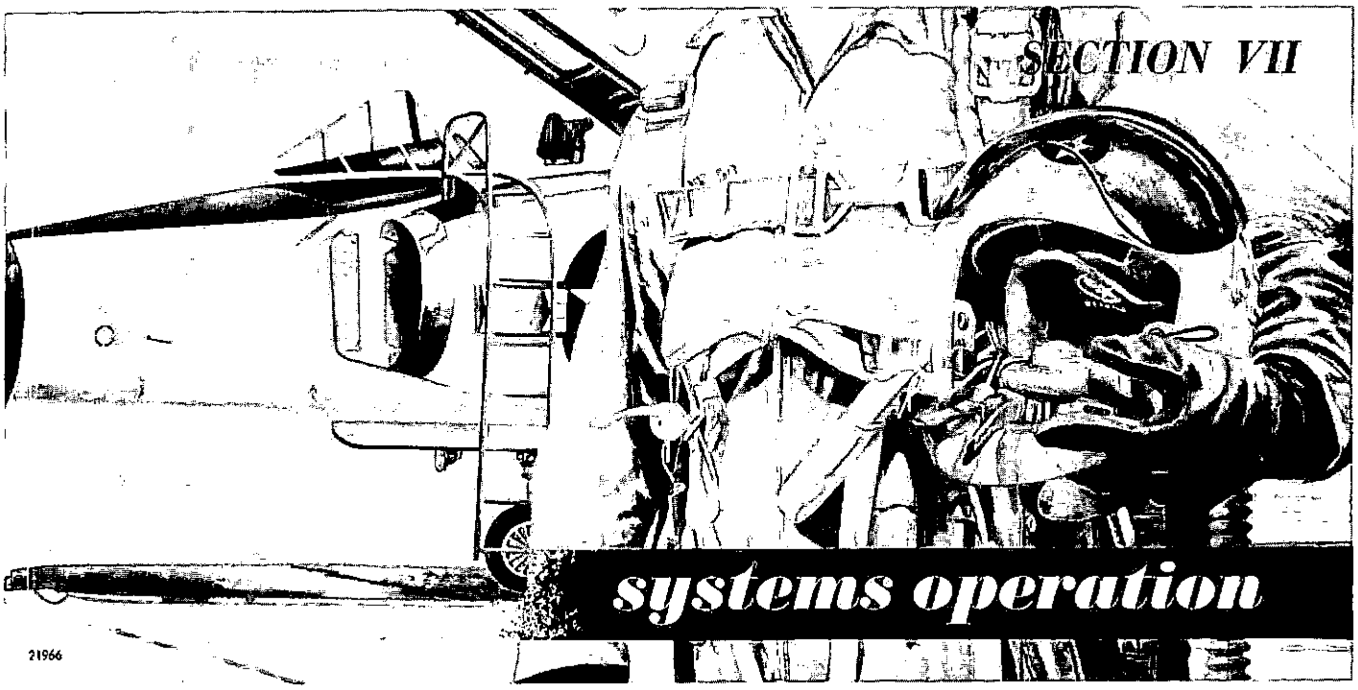
In recovery from a supersonic dive, minimum altitude loss is experienced if afterburner is turned off and speed brakes are opened.

FLIGHT WITH EXTERNAL TANKS

General flight characteristics with external tanks installed are basically the same as for the clean airplane. The normal differences associated with increased gross weight will be noted. Takeoff roll will be slightly increased and nose wheel lift-off speed should be a few knots higher. With one tank containing fuel and the other empty, no particular problem should be encountered in flight. Landing with either or both external tanks full or partially full is not recommended, and if attempted, close attention to airplane gross weight and sink rate must be observed. External tanks may be jettisoned at any speed within the tanks-on operating range. Though some lift may be experienced as the tanks release, there is no tendency to pitch-up. If either or both tanks are full when jettisoned, a sudden shock or vibration is experienced within the airplane as a result of the tank ejector mechanism reacting against the inertia of the heavy tank. This is characteristic of the ejector mechanism and should cause no concern.

Note

Refer to EXTERNAL TANKS, Section V, for airspeed and acceleration limitations with external tanks installed.



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COMBUSTION START—SECOND ATTEMPT

If a second attempt to start is to be made on airplanes equipped with a combustion starter and no external power is available, it may be necessary to manually replenish the starter fuel flask. This would be necessary if a false start had occurred. Without external power, ac power is not available and as a result the starter fuel flask would not be automatically replenished. The following procedure should be followed to manually replenish the starter fuel flask:

1. Gain entrance to the fuel flask through the right-hand engine access door.
2. Remove bleed line cap cover from starter flask bleed line.
3. Apply approximately ten psi of air to fuel flask bleed line.
4. Listen for bottoming of the plunger in starter flask.
5. Remove air source and allow fuel to gravity feed from fuel tanks to fill starter fuel flask. Use a suitable container to collect fuel drainage from fuel flask.
6. Install bleed line cap cover.
7. Check for leaks.

ENGINE COMPRESSOR BLEED SYSTEM

An automatically controlled compressor air bleed system on the engine reduces the possibility of compressor instability and resulting compressor stalls due to surges of low-speed compressor airflow acceleration. At times when the forward (low-speed) compressor supplies a greater volume of air than can be used by the aft compressor, air is vented to atmosphere from excess 9th stage air, overboard from the 9th stage manifold through a bleed valve and ducting to openings in the upper fuselage skin. A compressor bleed governor, driven by the low-speed compressor, senses changes both in rotor speed and in air pressure and temperature in the engine compressor inlet. This intelligence is transmitted to a bleed valve actuator which opens or closes the bleed valve, according to information received. The governor is mechanical, and the temperature sensing unit operates through a capillary. The valve actuating mechanism is pneumatic and uses air bled from the 16th stage of the high-speed compressor. The engine compressor bleed system is independent of any of the airplane systems.

FUEL BOOST PUMP LOW FUEL LEVEL OPERATION

On lowering the landing gear or on application of equivalent decelerating force, the fuel in the tanks will momentarily displace forward and unport the aft boost pumps and "T" check valve bell mouth. If neither forward pump is operating in fuel, air will be introduced into the fuel system and result in a flameout. Basically, the fuel system will sustain a constant flow of fuel as long as either forward boost pump is operating in fuel or both "T" check valves are submerged in fuel. If one side has been depleted of fuel as indicated by the low level warning light and the boost pump low-pressure light, the fuel valve to the empty side will be closed. In this condition, the concern is with the forward pump and "T" check valve on the side supplying fuel. In flight, boost pump failures are few but unawareness of probable consequences is a hazard. A boost pump that is a potential failure will normally continue operating satisfactorily until stopped. Once stopped, it may not restart. During descent with low fuel level in the No. 3 tanks, the following is applicable:

1. Whenever a penetration is made with a comparatively low fuel level in the No. 3 tanks, 600 pounds or less per tank, then the positive operation of the forward pumps should be established.
2. If the aft boost pumps are momentarily deactivated and the boost pump low-pressure warning light does not illuminate, this is an indication of proper functioning of respective forward boost pump.
3. If either forward pump is not available, then a relatively nose-high attitude must be maintained.

For additional information on fuel management, refer to Section III.

HIGH-PRESSURE PNEUMATIC CAPABILITY

Basic design concept in providing air supply for starting the engine is to provide a start capability at a remote base where a high-pressure compressor is not available. This concept presumes that a mission has been accomplished (including three armament passes consisting of three door cycles and two launchers cycles), and that the airplane requires only refueling before a return to the prime base, but does not allow for any emergencies or air motoring requirements of the combustion starter. When landing at a remote base where a high-pressure compressor is not available and it is essential that a start be made, air motor is to be minimized under the following conditions:

1. If the airplane has made two or three separate armament passes.
2. If a warning light indicating low pressure in the high-pressure pneumatic system is illuminated.

Note

- To minimize air motoring of the combustion starter during the starting sequence, place the throttle outboard to START, then depress the ignition button and immediately move inboard to the OFF, then IDLE position.
- There is a possibility of air motoring with the airplane's air supply system after two armament passes, but this would be marginal and may result in a "hung" start.
- Do not turn off battery power when landing at a remote base if an air compressor is not available and a combustion start is to be attempted. Do not reset the armament switch until after engine start to prevent the cylinders from refilling with air needed for the combustion starter.

Under all other conditions, air motoring of the combustion starter is mandatory. Given the most extreme conditions of air usage, leakage, and thermal loss, a starter capability should exist from 15 to 45 minutes after touchdown. Under more favorable conditions, this capability should exist over a much longer period due to a usual thermal gain after touchdown. (See figure 7-1.)

Note

- If a combustion start is made using the airplane supply prior to a firing mission, there will not be sufficient air left for armament firing.
- Low pressure in the system will affect rudder feel. Feel will not increase above that normally attained at approximately 380 knots indicated. If faster airspeeds are attained it is recommended that the rudder not be used because rudder feel will give the pilot a misleading indication of the stresses on the tail assembly.

SMOKE FROM TAILPIPE AFTER SHUTDOWN

Normally the pressurizing and dump valve in the fuel control system will prevent the accumulation of fuel in the engine after shutdown. However, if for any reason, fuel or oil should collect in the turbine housing during or immediately after shutdown, residual heat will vaporize the liquid and create a potential hazard. The situation is indicated by smoke or vapor exiting from the tailpipe. The engine should be cleared by the procedure given in Section II under STARTING ENGINES. All personnel should remain clear of the tailpipe for several minutes after engine shutdown and at all times when smoke or vapor is issuing from the nozzle.

LANDING GEAR

Emergency landing gear extension is accomplished by pulling the landing gear emergency extension handle. When the landing gear emergency extension handle is pulled, electrical power to the normal landing gear handle is shut off; therefore the landing gear handle can

high-pressure pneumatic capability

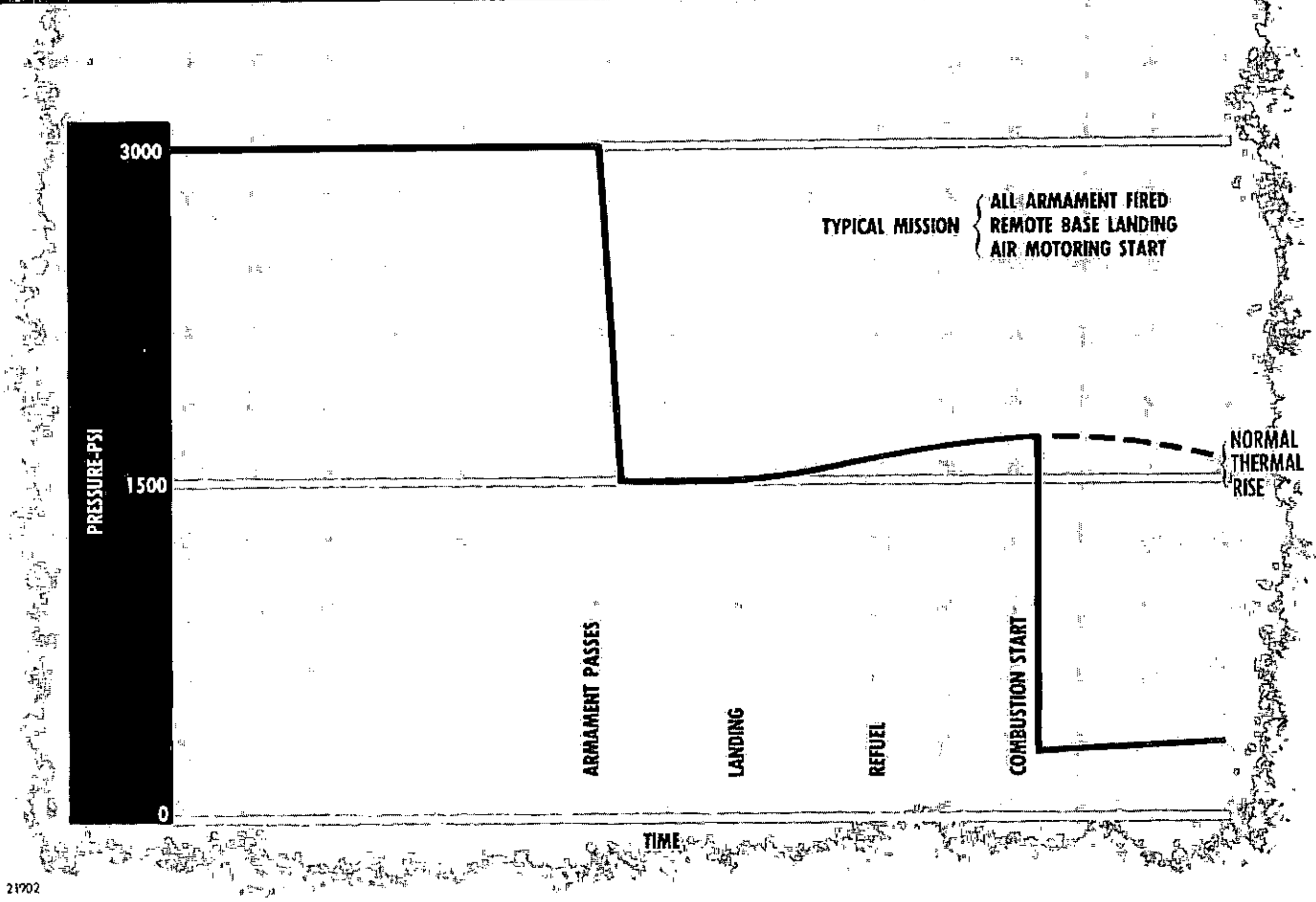


Figure 7-1

be in either the UP or DOWN position whenever the emergency extension handle is pulled, and the gear will remain extended by the emergency system. This action mechanically opens a pneumatic shutoff valve to supply high-pressure pneumatic system pressure to the wheel well door and gear actuating cylinders which will open the doors and extend the gear. The air pressure is routed through cylinder mounted shuttle valves which prevent intermixing of air and hydraulic fluid during extension. A spring clip is installed to lock the landing gear emergency extension handle in the fully pulled position. The emergency extension system should be utilized whenever the normal extension system fails to extend the gear, in event of complete electrical failure, failure of the secondary hydraulic system, or in the event of engine failure. Since the landing gear system operates from the secondary hydraulic system, failure of the primary hydraulic system would not influence normal gear extension; however, using the emergency system with only the primary hydraulic system inoperative may cause damage to the secondary hydraulic system.

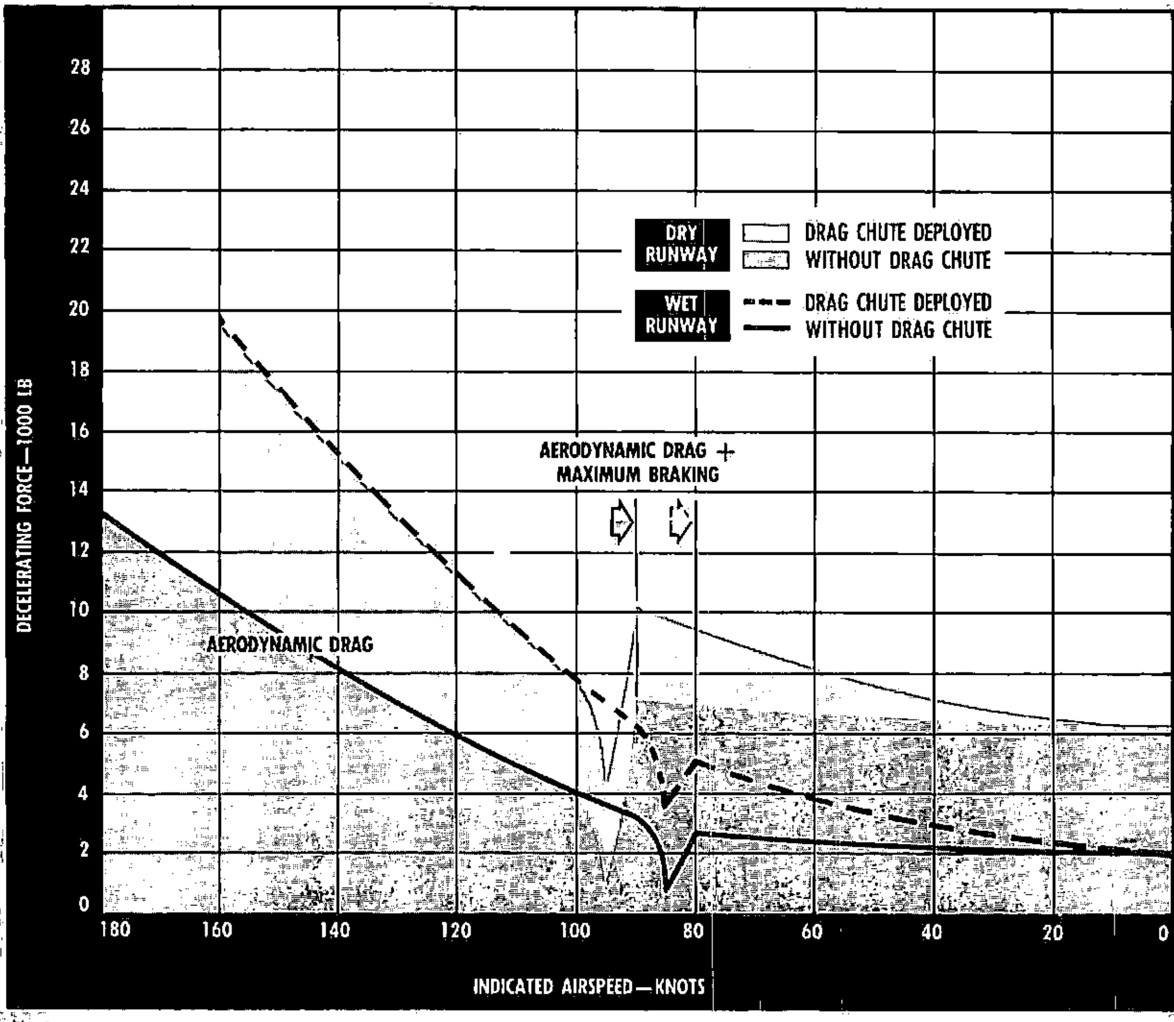
WARNING

- Normal landing gear extension with the primary hydraulic system inoperative should be made at a time when minimum use of flight controls is required.
- In planning the landing when the gear is to be lowered by the emergency system, one minute should be allowed for gear extension time on some airplanes.* On other airplanes** emergency extension should not require more than 10 seconds.

The above considerations would also apply in the event of engine failure even though the engine is windmilling and partial system pressure is available from the primary

**AF 53-1791 thru 56-1518 unless modified by TCTO 1F-102-655.*
***AF 57-770 & on, & airplanes modified by TCTO 1F-102-655.*

landing deceleration chart normal landing



CONDITIONS

- SEA LEVEL
- STANDARD DAY
- NO WIND
- 21,500 LB GROSS WEIGHT
- BRAKING COEFFICIENT OF FRICTION $(\mu) = 0.3$ DRY RUNWAY, 0.1 WET RUNWAY
- NOSE HIGH ATTITUDE TO 100 KNOTS FOR DRY RUNWAY AND TO 90 KNOTS FOR WET RUNWAY
- MAXIMUM BRAKING BEGUN AT 95 KNOTS FOR DRY RUNWAY AND AT 85 KNOTS FOR WET RUNWAY
- SPEED BRAKES OPEN

21920

Figure 7-2

and secondary hydraulic systems. There are no provisions for retracting the landing gear pneumatically; therefore, landing gear retraction should not be attempted following an emergency extension.

USE OF WHEEL BRAKES

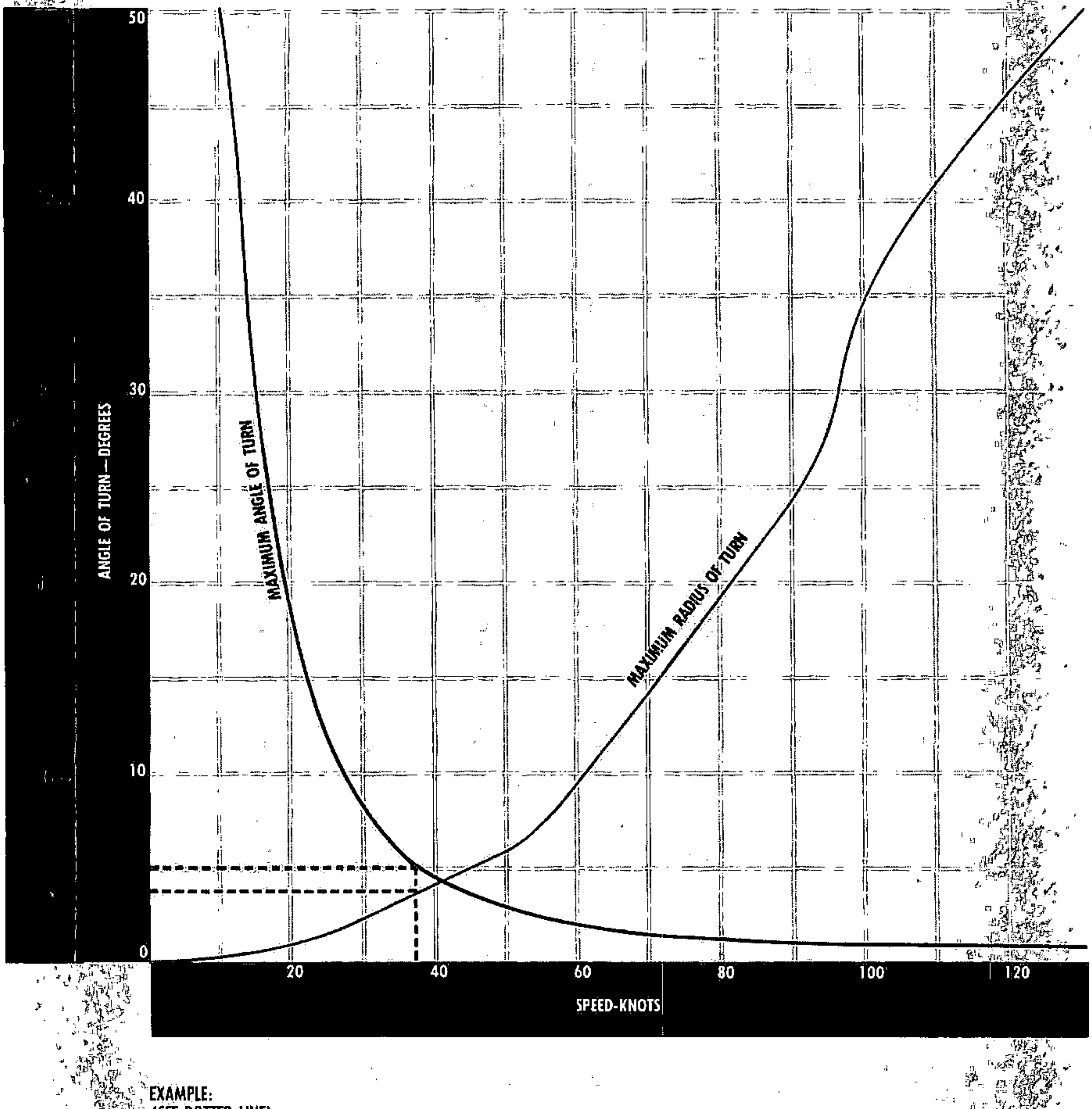
To reduce maintenance difficulties and accidents due to wheel brake failure, it is necessary that airplane brakes be treated with respect. The most common mistakes that are made that reduce brake life and reliability are stopping the airplane as quickly as possible regardless of the length of the runway; use of the brakes consistently for speeding up taxiing turns; and dragging the brakes while taxiing. When applying brakes, there may be a slight time delay between the pedal release and the release of braking action. To minimize brake wear, the following precautions should be observed insofar as is practicable.

1. Use extreme care when applying brakes immediately after touchdown or at any time when there is considerable lift on the wings to prevent skidding the tires and causing flat spots. A heavy brake pressure can result in locking the wheel more easily if brakes are applied immediately after touchdown than if the same pressure is applied after the full weight of the airplane is on the wheels. A wheel once locked in this manner immediately after touchdown will not become unlocked as the load is increased as long as brake pressure is maintained. Proper braking action cannot be expected until the tires are carrying heavy loads. Brakes, themselves, can merely stop the wheel from turning, but stopping the airplane is dependent on the friction of the tires on the runway. For this purpose, it is easiest to think in terms of coefficient of friction which is equal to the frictional force divided by the load on the wheel. It has been found that optimum braking occurs with approximately a 15 to 20% rolling skid; i.e. the wheel continues to rotate but has approximately 15 to 20% slippage on the surface so that the rotational speed is 80 to 85% of the speed which the wheel would have were it in free roll. As the amount of skid increases beyond this amount, the coefficient of friction decreases rapidly so that with a 75% skid the friction is approximately 60% of the optimum and, with a full skid, becomes even lower. There are two reasons for this loss in braking effectiveness with skidding. First, the immediate action is to scuff the rubber, tearing off little pieces which act almost like rollers under the tire. Second, the heat generated starts to melt the rubber and the molten rubber acts as a lubricant. If one wheel is locked during application of brakes, there is a very definite tendency for the airplane to turn away from that wheel, and further application of brake pressure will offer no corrective action. Since the coefficient of friction goes down when the wheel begins to skid, it is apparent that a wheel, once locked, will never free itself until brake pressure is reduced so that the braking effect on the wheel is less than the turning moment remaining with the reduced frictional force.
2. If maximum braking is required after touchdown, lift should first be decreased as much as possible by dropping the nose before applying brakes. This procedure will improve braking action by increasing the frictional force between the tires and the runway.
3. For short landing rolls, a single, smooth application of the brakes with constantly increasing pedal pressure is most desirable.
4. It is recommended that a minimum of 15 minutes elapse between landings where the landing gear remains extended in the slip stream, and a minimum of 30 minutes between landings where the landing gear has been retracted, to allow sufficient time for cooling between brake applications. Additional time should be allowed for cooling, if brakes are used for steering, cross-wind taxiing operation, or a series of landings is performed.
5. The full landing roll should be utilized to take advantage of aerodynamic braking and to use the brakes as little and as lightly as possible. Aerodynamic braking is most effective during the high-speed portion of the ground roll, when wheel brakes are least effective. Figure 7-2 illustrates the benefits derived from aerodynamic braking.
6. After the brakes have been used excessively for an emergency stop and are in the heated condition, the airplane should not be taxied into a congested parking area. Peak temperatures occur in the wheel and brake assembly from 5 to 15 minutes after a maximum braking operation. To prevent brake fire and possible wheel assembly explosion, the specified procedures for cooling brakes should be followed.
7. The brakes should not be dragged when taxiing, and should be used as little as possible for turning while taxiing.

GROUND MANEUVERING

Excessive side g-loads are often imposed on the landing gear when high speed turns are made during ground operation. This occurs most frequently while taxiing prior to takeoff and during the landing roll. Excessive side g-loads cause undue strain and eventual damage to the landing gear. As indicated in figure 7-3, the maximum permissible angle of turn decreases and the radius of turn increases sharply at higher speeds. As an example, at 15 knots the maximum angle of turn is 30 degrees and the radius of turn 35 feet. With an increase of speed to 37 knots for the same condition, the angle of turn is reduced to five degrees and the radius of turn increases to 230 feet. Because of the unreliability of the airspeed indicator at slow speeds, the airplane should be slowed down as much as possible prior to turning off the active

ground maneuvering chart



EXAMPLE:
(SEE DOTTED LINE)
KNOTS: 37
ANGLE OF TURN: 5 DEGREES
RADIUS OF TURN: 230 FEET

21917

Figure 7-3

runway after landing. As an added precaution, it is recommended the last taxi way be used whenever possible. This will decrease tendencies of making high-speed turns that would subject the landing gear to excessive side g-loads. Considerable runway is required to reduce speed sufficiently for making turns. Under certain conditions with maximum braking first applied at touchdown and with no wind effect, approximately 2100 feet is required to slow the airplane from 134 knots to 30 knots (134 knots is used as touchdown speed in this example) with the drag chute deployed, and approximately 3255 feet without drag chute assistance. More than half the landing roll with drag chute, and two-thirds without drag chute, is at speeds of 90 knots or more. It can be seen that much less braking force can be applied at high speeds, and that considerable runway is used before braking is most effective. For information concerning proper use of wheel brakes, refer to USE OF WHEEL BRAKES, this Section.

PARTIAL PRESSURE SUIT DEPRESSURIZATION

When the partial pressure suit is pressurized during ground check or inadvertently due to a crash landing or malfunction at low altitude, the pressure should always be released from the helmet before it is released from the suit. This can best be accomplished by opening (unlocking) the faceplate. The pressure is released simultaneously from helmet and suit following use of the test button on the kit and on some types of preflight test consoles. With the MC-3 and MC-4 suits, there is little immediate difficulty if pressure is first released from the suit, but with previous partial pressure suits, lung damage can result. It is the responsibility of each pilot who wears an altitude suit to be thoroughly familiar with donning and connecting his suit, conducting a preflight inflation with available testers, and being thoroughly familiar with the operation and care of his particular type suit.

SECTION VIII—CREW DUTIES

Not applicable to this airplane.

SECTION IX

all weather operation

21984

TABLE OF CONTENTS

Note

Except for some repetition necessary for emphasis, clarity, or continuity of thought, this section contains only those procedures that differ or are in addition to the normal operating instructions covered in Section II and Section IV. Any discussion relative to systems operations is covered in Section VII.

INSTRUMENT FLIGHT PROCEDURES

These procedures and techniques pertain primarily to instrument flight conditions and are in addition to normal procedures. Because navigation facilities and terrain vary at each base, this information is intended to serve as a guide to commanders in establishing instrument flight procedures. This airplane is capable of supersonic speeds during instrument flight conditions, thus demanding a relatively high degree of instrument proficiency and conscientious preflight planning. Fuel requirements for completion of instrument letdown, approach procedures, and possible diversion to alternate bases are much greater than for VFR flights and must be included in preflight planning. Only the essential navigation equipment is installed since ground-controlled guidance replaces some of the more conventional navigation aids found in most other types of airplanes. With the existing equipment, three types of instrument approaches may be made; ILS, GCA, VOR, and Tacan (some airplanes).

INSTRUMENT TAKEOFF

Complete the normal TAXI and BEFORE TAKEOFF checks as prescribed in Section II and check pitot heat ON. After aligning airplane visually with runway center line, check directional indicator (RMI) and standby magnetic compass against runway heading. Set attitude indicator at 5° nose-down. This setting will provide a more accurate indication as it will tend to offset the pitch error resulting from takeoff acceleration.

CAUTION

To insure that nose wheel steering is engaged for takeoff, the steering should be used to line up and should not be disengaged until the rudder becomes effective on the takeoff roll.

Instrument takeoffs may be made at either military or maximum thrust.

Note

Afterburner is recommended to shorten the takeoff roll in conditions of low visibility.

Maintain heading with nose wheel steering until rudder becomes effective at approximately 80 knots. The directional indicator is primarily for directional control, but reference should be made to runway centerline and runway lights if possible. Beginning at 125 KIAS apply back pressure and rotate the airplane from 5° nose-down pitch attitude to 15° nose-up pitch on the attitude indicator. Airplane will become airborne with pitch change. Raise landing gear when a positive climb indication is noted. Maintain approximately 12° pitch attitude on the attitude indicator until a positive climb is indicated on the vertical velocity indicator. Maintain 12° pitch attitude on the attitude indicator and a positive climb on the vertical velocity indicator until the climb schedule is reached.

Note

In maximum thrust takeoffs, the altimeter and vertical velocity indicator may lag or even indicate a descent in altitude just before breaking ground and during the initial climbout. This altimeter error is the result of disturbed pressure ahead of the airplane due to acceleration and high angle of attack. The altimeter will indicate correctly after the airplane reaches approximately 300 feet of altitude.

INSTRUMENT CLIMB

If the tactical situation permits, use military climb schedule when turbulence or rain are encountered. Upon reaching climb schedule, increase pitch attitude to approximately 18° on the attitude indicator.

CAUTION

Afterburner climb through rain can result in damage to the radome and wing fences. This schedule also results in a high pitch attitude for instrument flight.

INSTRUMENT CRUISING FLIGHT

Keep the airplane trimmed for straight and level flight. For ease and precision of flight, limit all turns to 30° bank angle.

RADIO AND NAVIGATION EQUIPMENT

Refer to Section IV for radio and navigation equipment installed.

DESCENT

The optimum thrust for fuel economy during descents is idle. Instrument descents can be flown without difficulty at any speed; though for maximum ease of handling, a constant-speed letdown (275 KIAS) is recommended.

WARNING

Steep descents, high indicated Mach numbers, and high angles of bank should be avoided to maintain positive control of the airplane at all times. Limiting bank angle to 30° at all altitudes and rates of descent is recommended.

RADAR RECOVERY

Radar recovery can be accomplished with a minimum amount of fuel and time. Radar recovery procedure to be used following interceptions in IFR conditions should be practiced in VFR and IFR weather to develop and improve the teamwork of the pilot and radar controller. When a GCA or ILS approach is required, the descent from the inbound cruising altitude should be started at a sufficient distance to permit a straight-in at the recommended airspeed plus three to four miles for deceleration and changing to approach configuration before turn-on point (gate) to final approach. See figure 9-3 for recommended procedures.

JET PENETRATION

Omni penetrations can be made by using various techniques. For ease of operation, letdowns using 85% rpm, 275 KIAS, gear up, and speed brakes out are recommended. The exact procedure for jet penetrations in the "Terminal Flight Information (High Altitude)" will vary at different bases because of terrain differences, airway locations, and conflicting traffic control zones. Consequently, fuel requirements may vary as conditions vary. Letdown procedure at destinations should be carefully checked and fuel allowances made a part of the preflight planning. See figures 9-1 and 9-2 for the recommended techniques for typical penetration procedures.

Note

If weather is below minimum for an omni low approach, request ILS or GCA prior to beginning letdown. Make decision to proceed to alternate while still at altitude, if possible.

HOLDING

Based upon minimum fuel consumption consistent with ease of handling, the holding data are as follows:

ALTITUDE	KIAS	% RPM	FUEL FLOW (lbs/hr)	MAXIMUM BANK ANGLE
40,000	232	88 ± 2	2200-2500	20°
20,000	225	85 ± 2	2400-2750	30°

INSTRUMENT APPROACHES

The airplane has good handling characteristics during instrument approaches. At low rpm and low indicated airspeeds, response to throttle movement and acceleration is slow. Therefore, use relatively high thrust settings in the approach configuration and maintain at least 170 KIAS. In rain or snow, use of the windshield rain clearing system will greatly increase forward visibility. If it is necessary to execute a missed approach, military thrust should be added and speed brakes closed.

Note

Afterburner should be used only if necessary, because of high fuel consumption rate.

Raise the gear when a definite climb has been established and maintain at least 200 KIAS. Execute the established missed approach procedure as published in the "Terminal Flight Information (High Altitude)" or as directed by the GCA controller.

TACAN

Establish recommended final approach airspeed with gear extended after reaching the gate. Thrust required for level flight is approximately 85% rpm. Upon reaching the gate, extend speed brakes and descend to minimum altitude.

RADIO APPROACHES

Normally an omni approach would be required only if the airplane is not VFR after reaching minimum penetration altitude and no GCA or ILS is available. Refer to the "Terminal Flight Information (High Altitude)" for the local procedures for the standard jet instrument approach.

GCA APPROACH

GCA procedures vary at each base due to location of radio fixes and local terrain. Fuel may be conserved by waiting until final approach before using speed brakes and lowering landing gear. In heavy precipitation it is difficult for the GCA operators to keep the airplane visible in the precipitation clutter on the radar scope. This may result in initiating a missed approach.

ILS APPROACH

Check the "Terminal Flight Information (High Altitude)" for local procedures. During an ILS approach, maintain airplane attitude with the basic flight instruments and monitor the ILS indicator for reference to the localizer and glide-slope. If localizer interception is from a radio fix or GCI control and is at an angle of 90° or less, allow sufficient distance out to perform final cockpit check, slow to approach speed, and establish correct beam heading. See figure 9-4.

WARNING

During an ILS approach, if disappearance of the glide-slope indicator warning "OFF" flag is delayed beyond normal expectation, it may be an indication that electrical power to the instrument selector relays is lost and the CDI information is being derived from a Tacan station. At the start of an ILS approach where the CDI warning "OFF" flag has disappeared but the glide-slope warning "OFF" flag is still visible, or at a locally prescribed check point, check to determine if there has been power loss to the relays by turning the bearing selector knob a few degrees away from the inbound heading. If the CDI responds to the bearing change, the signal was being received from a Tacan station and not the localizer. If the CDI does not respond to the inbound bearing change, the signal was being received from the localizer. After both warning "OFF" flags have disappeared, a subsequent power loss to the instrument selector relays will be detected by the horizontal needle warning "OFF" flag appearing and the horizontal needle will center itself and remain centered regardless of airplane movement.

AUTOMATIC APPROACH (AILAS)

Automatic approaches (see figure 9-4) are flown by using the automatic flight control system (AFCS). For detailed operating instructions of the automatic flight control system, refer to Section IV. Refer to Section V for AILAS mode limitations. Use the following procedures for an automatic approach:

1. Radar master switch—STBY. This positions radar antenna downward clear of the ILS glide-slope antenna.

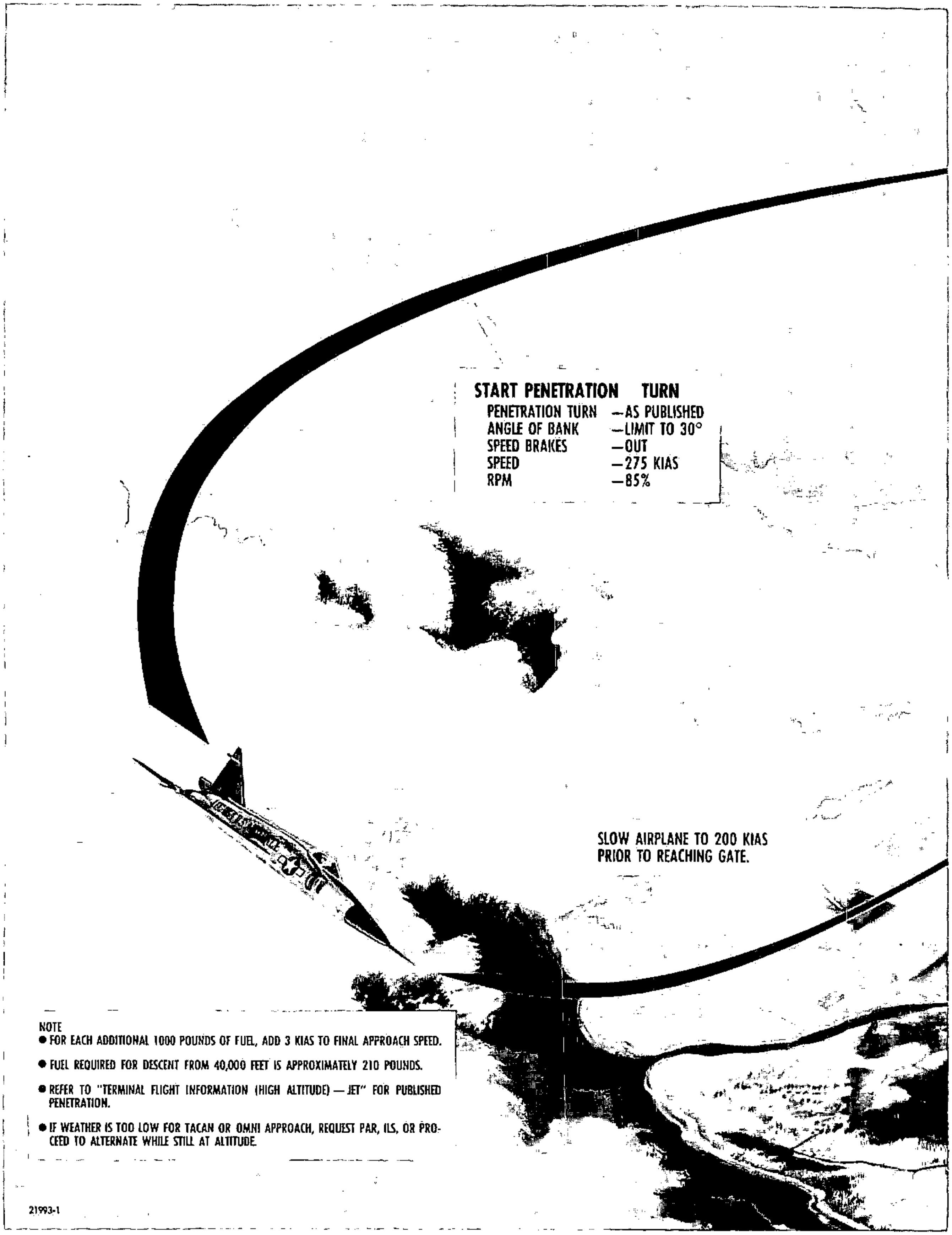
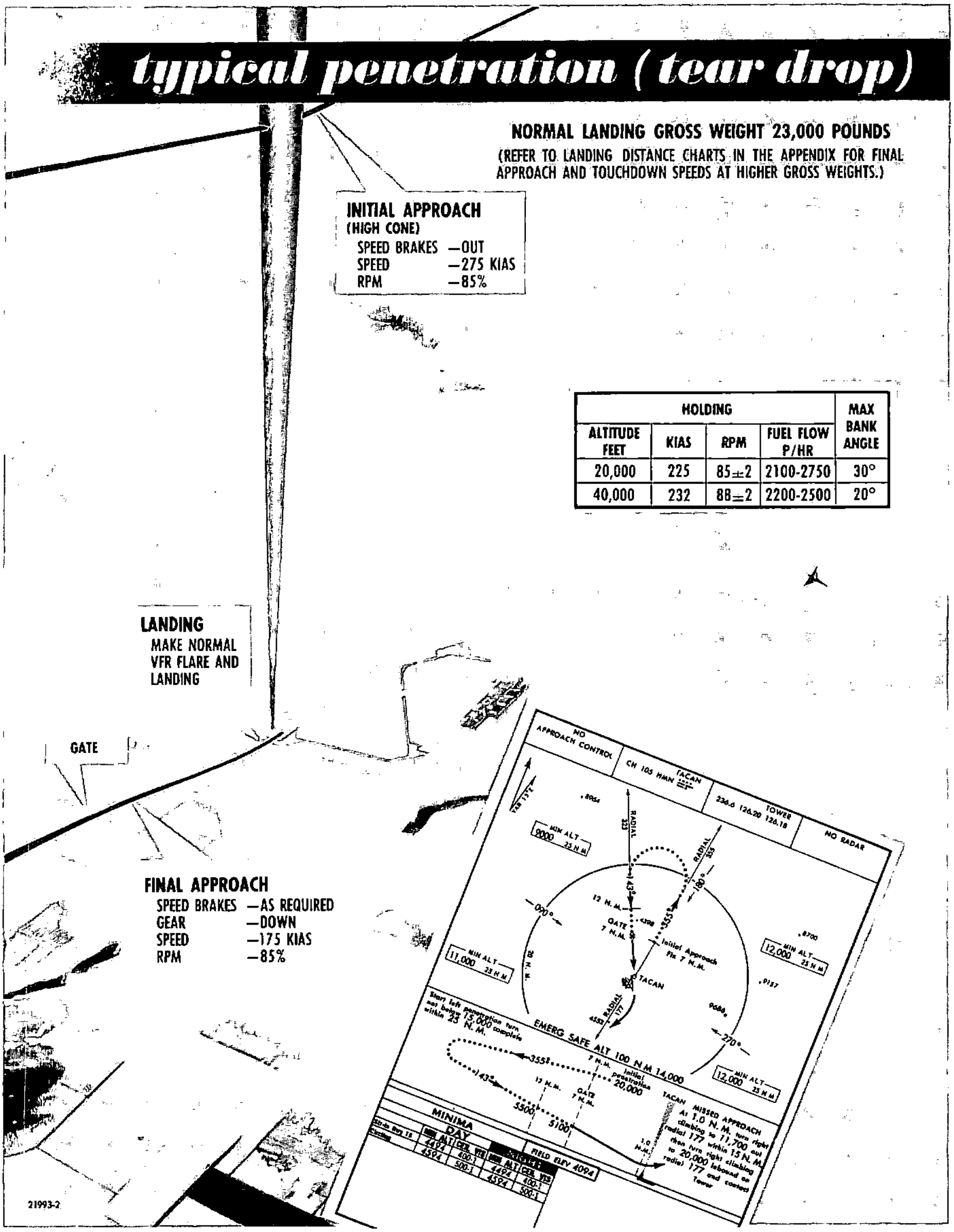

START PENETRATION TURN
PENETRATION TURN —AS PUBLISHED
ANGLE OF BANK —LIMIT TO 30°
SPEED BRAKES —OUT
SPEED —275 KIAS
RPM —85%
SLOW AIRPLANE TO 200 KIAS PRIOR TO REACHING GATE.
NOTE
● FOR EACH ADDITIONAL 1000 POUNDS OF FUEL, ADD 3 KIAS TO FINAL APPROACH SPEED.
● FUEL REQUIRED FOR DESCENT FROM 40,000 FEET IS APPROXIMATELY 210 POUNDS.
● REFER TO "TERMINAL FLIGHT INFORMATION (HIGH ALTITUDE) — JET" FOR PUBLISHED PENETRATION.
● IF WEATHER IS TOO LOW FOR TACAN OR OMNI APPROACH, REQUEST PAR, ILS, OR PROCEED TO ALTERNATE WHILE STILL AT ALTITUDE.
21993-1

Figure 9-1

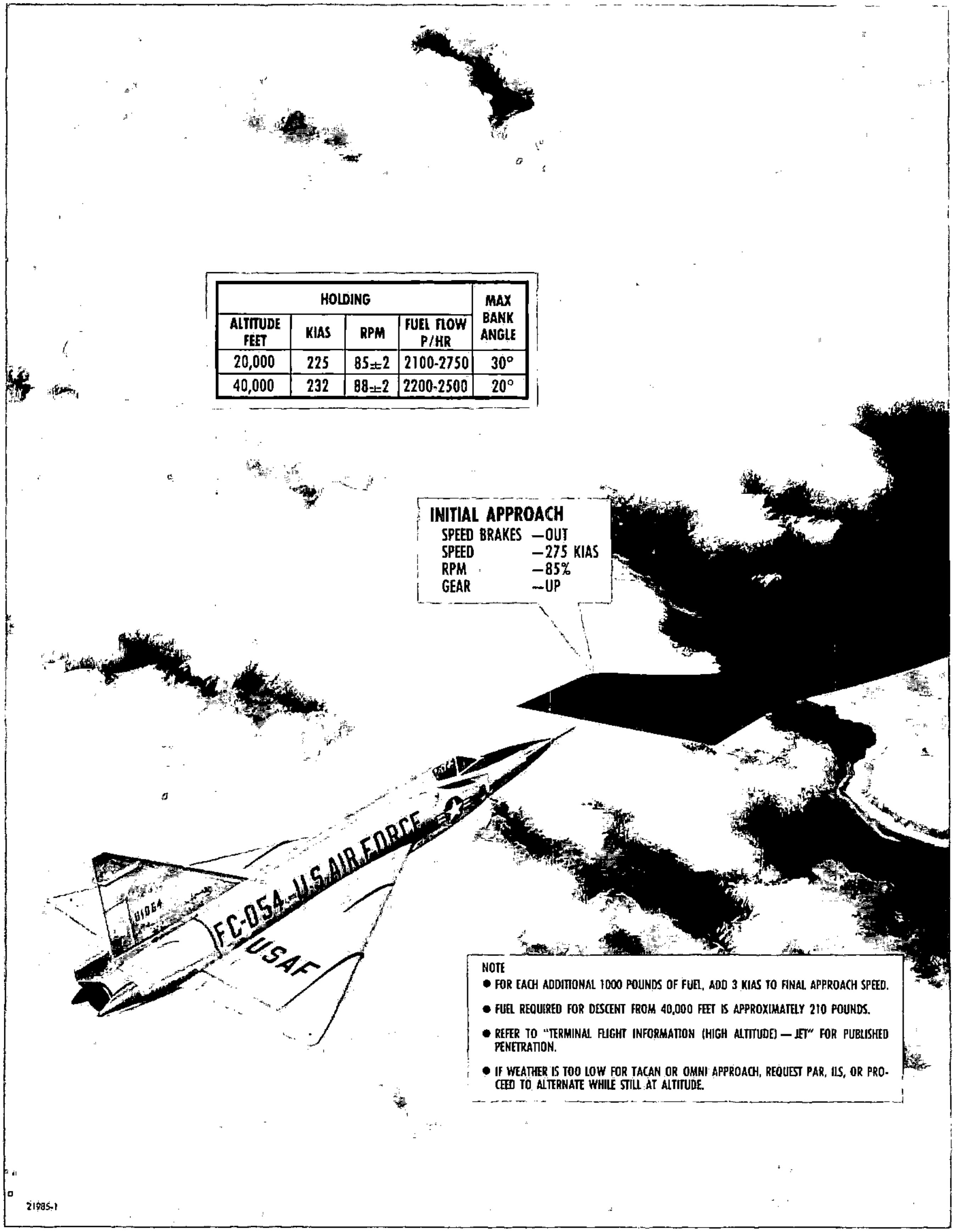
typical penetration (tear drop)

NORMAL LANDING GROSS WEIGHT 23,000 POUNDS
(REFER TO LANDING DISTANCE CHARTS IN THE APPENDIX FOR FINAL APPROACH AND TOUCHDOWN SPEEDS AT HIGHER GROSS WEIGHTS.)



HOLDING				MAX BANK ANGLE
ALTITUDE FEET	KIAS	RPM	FUEL FLOW P/HR	
20,000	225	85±2	2100-2750	30°
40,000	232	88±2	2200-2500	20°

21993-2



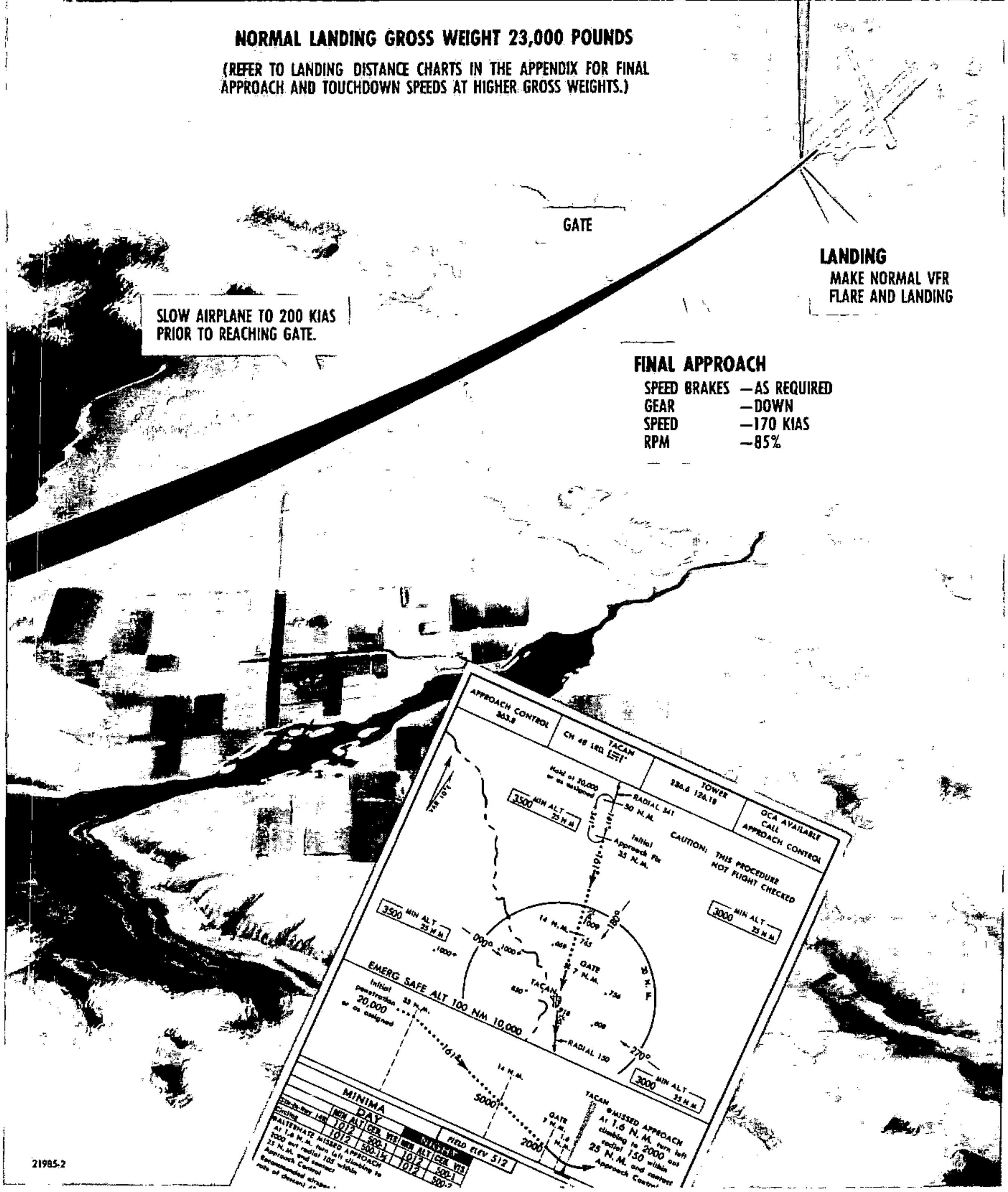
HOLDING				MAX BANK ANGLE
ALTITUDE FEET	KIAS	RPM	FUEL FLOW P/HR	
20,000	225	85±2	2100-2750	30°
40,000	232	88±2	2200-2500	20°

NOTE

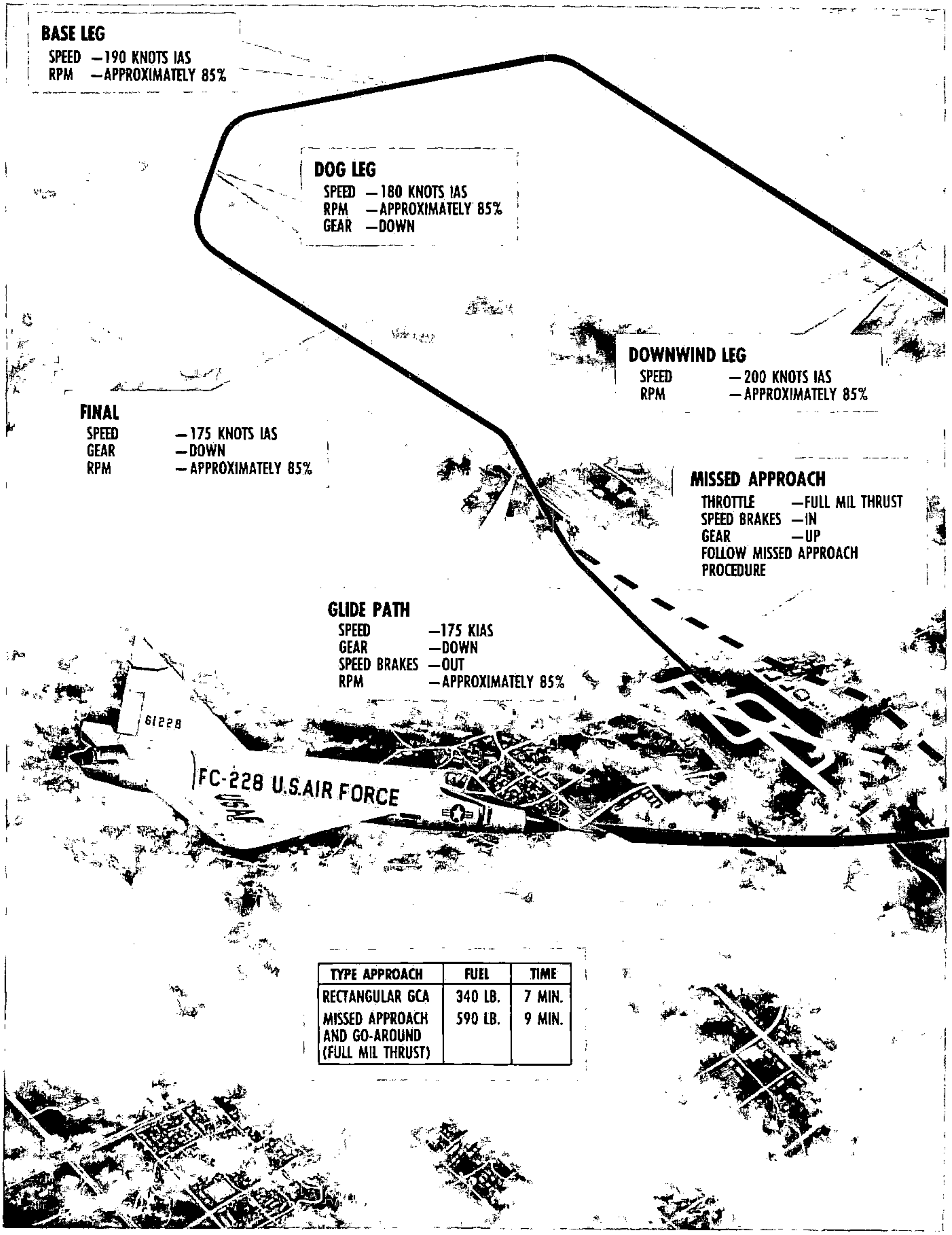
- FOR EACH ADDITIONAL 1000 POUNDS OF FUEL, ADD 3 KIAS TO FINAL APPROACH SPEED.
- FUEL REQUIRED FOR DESCENT FROM 40,000 FEET IS APPROXIMATELY 210 POUNDS.
- REFER TO "TERMINAL FLIGHT INFORMATION (HIGH ALTITUDE) — JET" FOR PUBLISHED PENETRATION.
- IF WEATHER IS TOO LOW FOR TACAN OR OMNI APPROACH, REQUEST PAR, ILS, OR PROCEED TO ALTERNATE WHILE STILL AT ALTITUDE.

Figure 9-2

typical penetration (straight in)



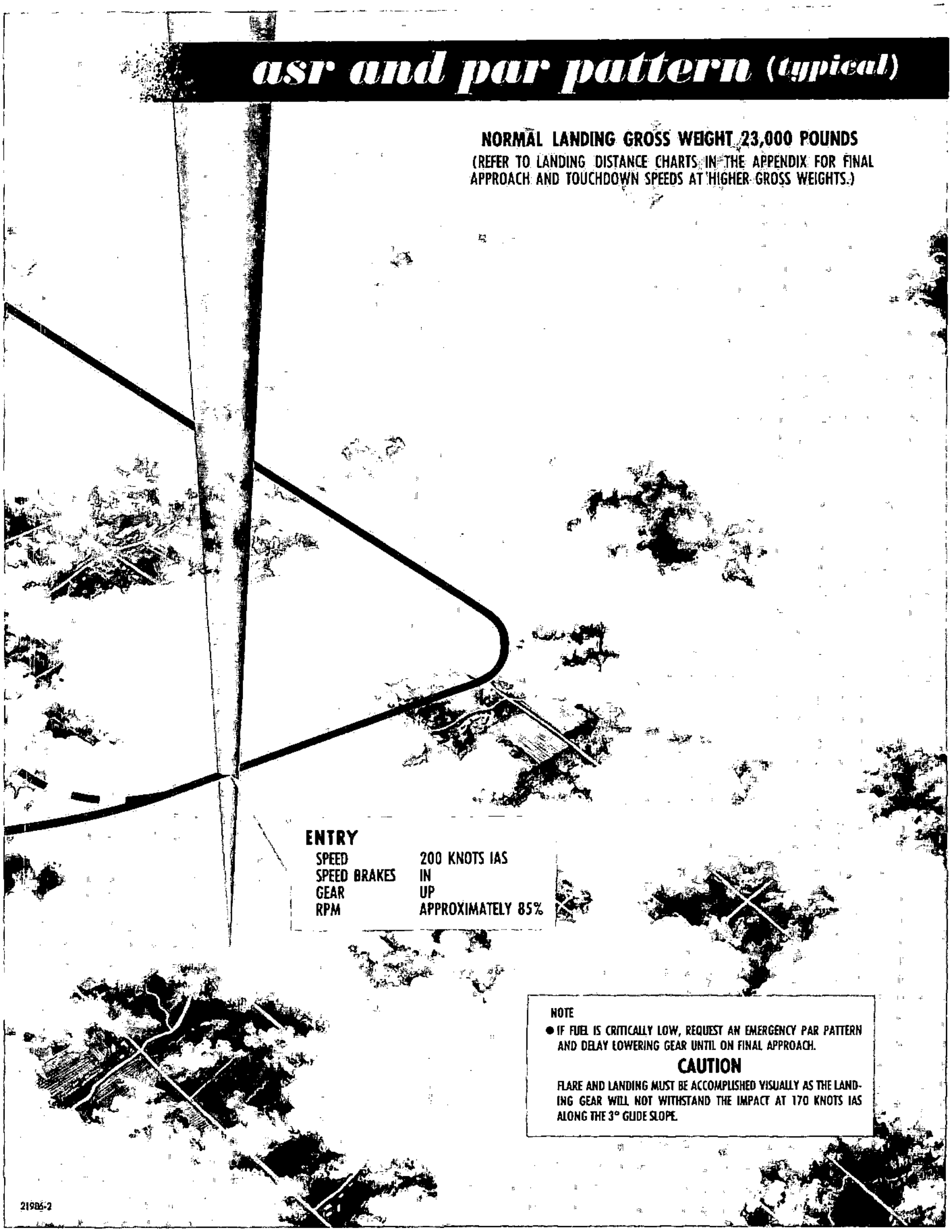
21985-2



TYPE APPROACH	FUEL	TIME
RECTANGULAR GCA	340 LB.	7 MIN.
MISSED APPROACH AND GO-AROUND (FULL MIL THRUST)	590 LB.	9 MIN.

Figure 9-3

asr and par pattern (typical)



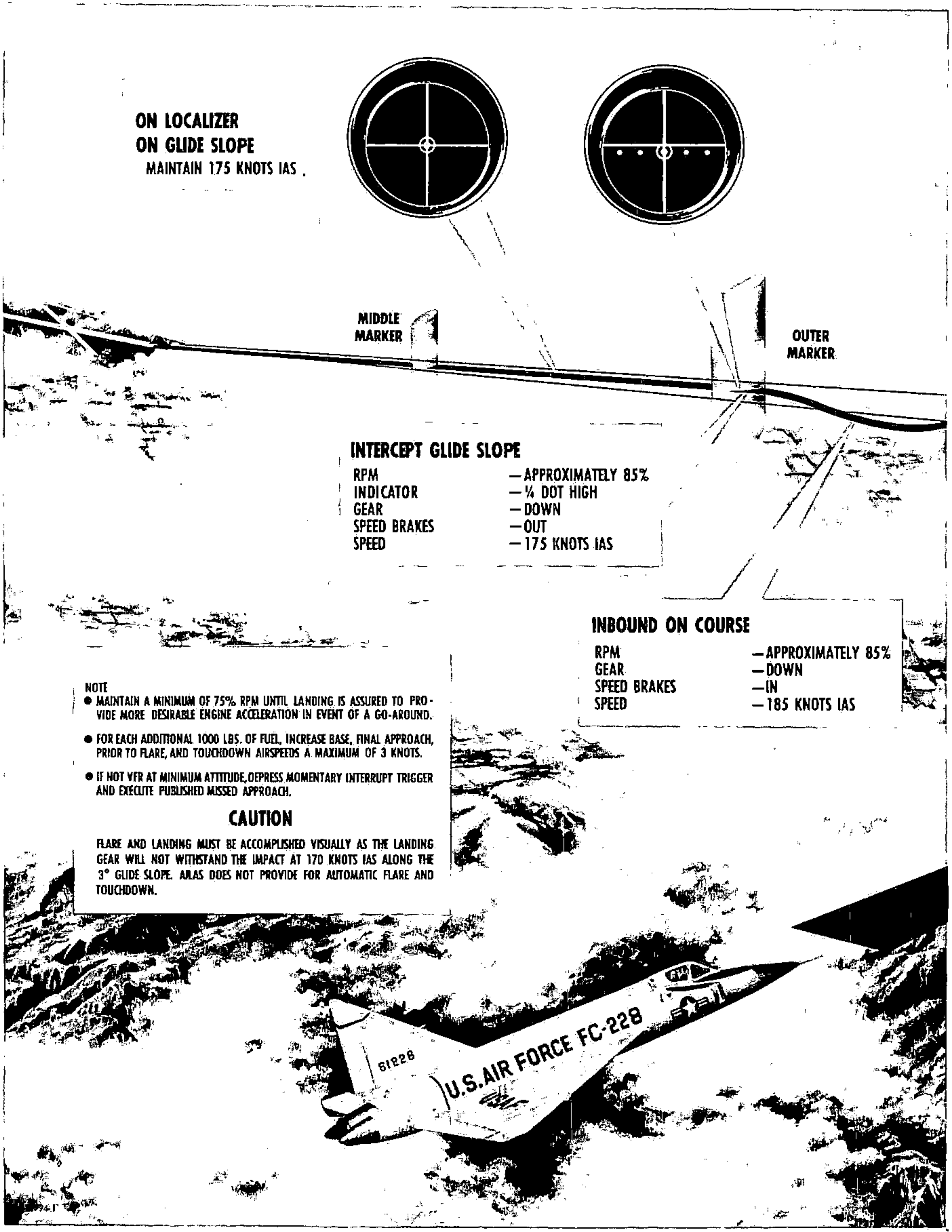
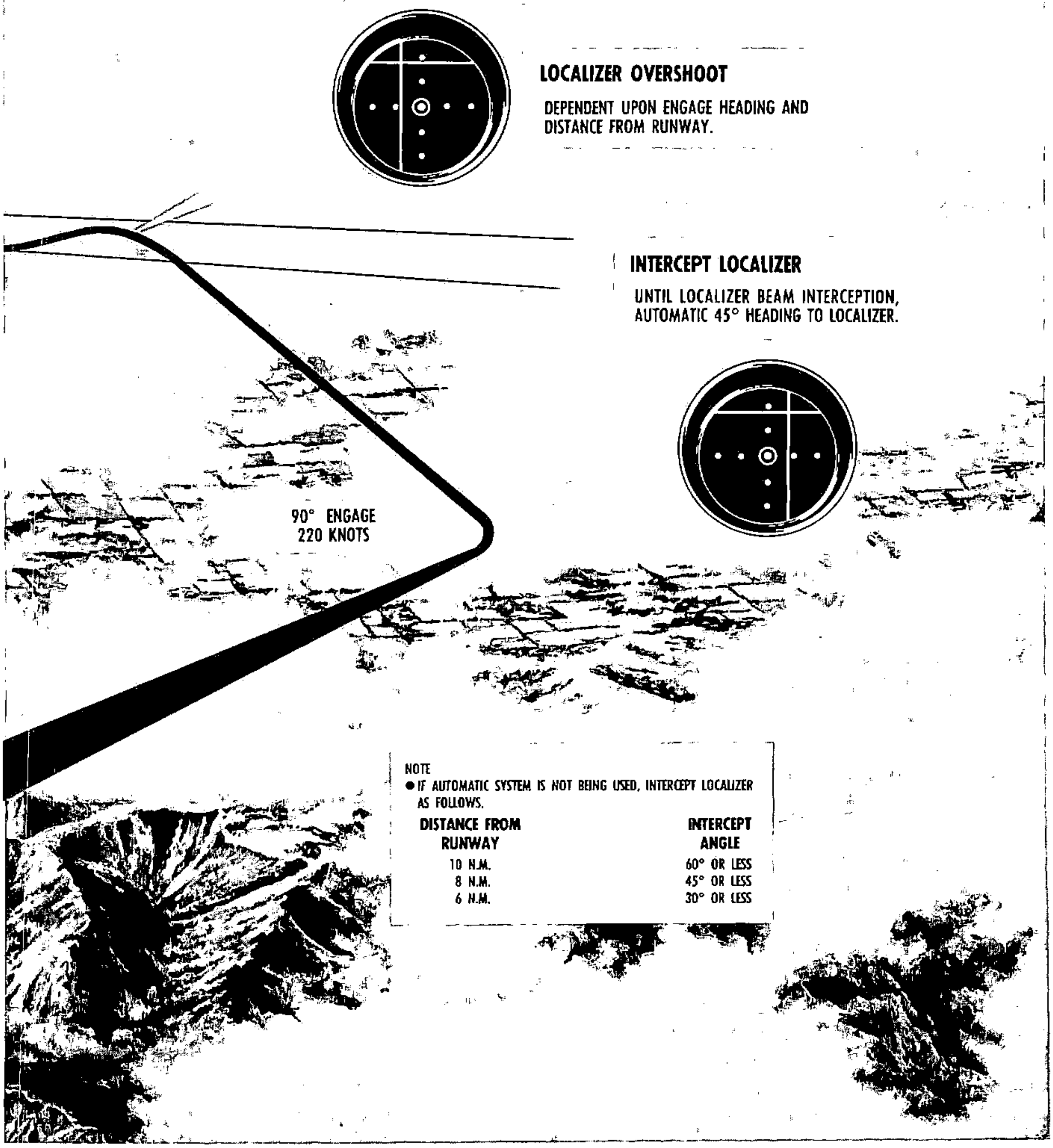

ON LOCALIZER
ON GLIDE SLOPE
MAINTAIN 175 KNOTS IAS.
MIDDLE MARKER
OUTER MARKER
INTERCEPT GLIDE SLOPE
RPM —APPROXIMATELY 85%
INDICATOR —¼ DOT HIGH
GEAR —DOWN
SPEED BRAKES —OUT
SPEED —175 KNOTS IAS
INBOUND ON COURSE
RPM —APPROXIMATELY 85%
GEAR —DOWN
SPEED BRAKES —IN
SPEED —185 KNOTS IAS
NOTE
● MAINTAIN A MINIMUM OF 75% RPM UNTIL LANDING IS ASSURED TO PROVIDE MORE DESIRABLE ENGINE ACCELERATION IN EVENT OF A GO-AROUND.
● FOR EACH ADDITIONAL 1000 LBS. OF FUEL, INCREASE BASE, FINAL APPROACH, PRIOR TO FLARE, AND TOUCHDOWN AIRSPEEDS A MAXIMUM OF 3 KNOTS.
● IF NOT VFR AT MINIMUM ATTITUDE, DEPRESS MOMENTARY INTERRUPT TRIGGER AND EXECUTE PUBLISHED MISSED APPROACH.
CAUTION
FLARE AND LANDING MUST BE ACCOMPLISHED VISUALLY AS THE LANDING GEAR WILL NOT WITHSTAND THE IMPACT AT 170 KNOTS IAS ALONG THE 3° GLIDE SLOPE. ALAS DOES NOT PROVIDE FOR AUTOMATIC FLARE AND TOUCHDOWN.
U.S. AIR FORCE FC-228
61228

Figure 9-4

automatic approach (ailas)

NORMAL LANDING GROSS WEIGHT 23,000 POUNDS
(REFER TO LANDING DISTANCES CHARTS IN THE APPENDIX FOR FINAL APPROACH AND TOUCHDOWN SPEEDS AT HIGHER GROSS WEIGHTS.)



NOTE
- IF AUTOMATIC SYSTEM IS NOT BEING USED, INTERCEPT LOCALIZER AS FOLLOWS.

DISTANCE FROM RUNWAY	INTERCEPT ANGLE
10 N.M.	60° OR LESS
8 N.M.	45° OR LESS
6 N.M.	30° OR LESS

2. AFCS engaged and operational in pilot assist mode.
3. Set ILS runway heading in course indicator.
4. Set function selector knob on J-4 compass control panel to MAG.

   This is necessary or the compass will be phased to another heading reference.
5. VOR receiver on, frequency set (some airplanes); ILS receiver (AN/ARN-31) on, frequency set, and instrument selector switch to ILS (other airplanes).
6. Check that the localizer and glide-slope warning "OFF" flags are not showing and that the airplane is below the glide-slope (glide-slope indicator above center).
7. Enter AILAS four-mile engage circle, which is centered 12 miles from the runway, at an altitude of 1500 feet above the runway. AILAS may be engaged on any heading; however, for early localizer capture and minimum localizer overshoot, the entry heading should be 45° or less to the localizer course.
8. Airspeed 220 KIAS, landing gear up.
9. AILAS button depressed. Check green light illuminated, dim as desired.

   The AILAS light will remain illuminated as long as the AILAS remains engaged; the light will go out upon disengagement. While under AILAS control, the elevon trim switch is inoperative. AILAS limits bank angle to 33 (±3°) until glide-slope entry.
10. Inbound on localizer, reduce speed to 185 KIAS. The pilot may assume manual control at any time during the approach by depressing the momentary interrupt trigger. Prior to glide-slope entry, manual control is available without disengaging AILAS by depressing the momentary interrupt trigger (engage light remains on). Upon release of the momentary interrupt trigger, the system reverts to the AILAS mode.
11. As the glide-slope indicator progresses down to ¼ width from center, extend speed brakes, lower landing gear, and reduce speed to final approach speed.
12. Check for glide-slope capture by noting airplane pitch response to the glide-slope signal following coincidence. Following glide-slope interception, AILAS limits bank angle to 15 degrees.

When visual contact with the runway is established, depress and hold momentary interrupt trigger, flare out and touch down. With AILAS engaged prior to glide-slope capture, if the localizer signal is lost (warning flag appears), AILAS will automatically disengage and the AFCS will revert to pilot assist attitude hold. If either localizer or glide-slope is lost after glide-slope entry, the AILAS will automatically disengage and the AFCS will revert to pilot assist attitude hold.

When the anti-icing systems are in operation, the airplane may be flown safely under icing conditions. No wing or vertical fin surface anti-icing systems are required as there is sufficient thrust available to overcome the increased range as increased thrust is required to maintain desired flying speed. The stability and control of the airplane will not be noticeably affected with surface ice buildups. The most probable free air icing temperatures vary from −4°C (+25°F) at sea level to −24°C (−12°F) at 20,000 feet. Above 20,000 feet, due to the inability of the drag from surface ice buildups. Surface icing will reduce air to contain moisture, the amount of icing is negligible. The mission of the airplane is so designed that most phases of a typical mission will be performed at altitudes above icing levels. The phase most susceptible to ice and most critical to operation is the instrument approach. If icing conditions are known to exist at instrument approach altitudes, the most expeditious means of recovery (normally the GCI penetration to final approach with a straight-in GCA or ILS) should be used to minimize the surface ice buildup. If icing is encountered unexpectedly and is allowed to build up, more thrust will be required to maintain desired speeds and rates of descent during the instrument approach. Flight under icing conditions with the engine and intake duct anti-icing systems inoperative could result in two forms of engine damage. Ice buildup on the engine inlet guide vanes may result in a restricted flow of inlet air, causing loss of thrust, a rapid rise in exhaust gas temperature and possible compressor stall (refer to COMPRESSOR STALL, Section III). The possibility of this occuring is reduced by the absence of inlet screens and the relatively clean, unrestricted intake. However, should sufficient ice buildup occur to restrict airflow, the first indication will be a compressor stall. Immediate action should be taken to relieve the stall and change to an altitude where icing is less severe. The second form of engine damage could result from intake duct ice breaking loose and being drawn into the compressor section of the engine resulting in compressor section failure. Because of the possible damage that may result due to engine ice, the anti-icing system should be operating at all times when flight under icing conditions is anticipated.

Note

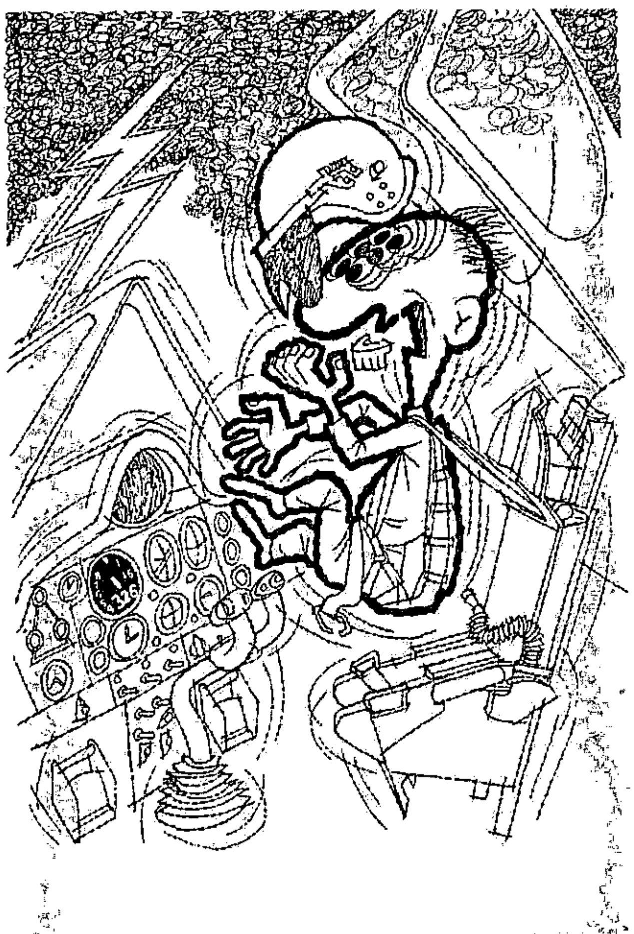
Engine operation below approximately 90% rpm may not supply sufficient heat to keep the engine compressor inlet clear of ice under severe icing conditions. When descending in icing conditions, power should be increased to provide sufficient heat.

When instrument takeoffs or approaches are to be made and rain is anticipated, the windshield rain clearing system should be in operation to increase forward visibility through the left-hand windshield panel. To insure proper operation of the rain clearing system, it is recommended that the system be energized momentarily prior to flight. Energizing the system will blow from the ducts any water that may be present. If reported rain intensity is heavy or less, good vision should be obtained through cleared area. Cleared area will be slightly smaller in heavy rain than in moderate rain. If very heavy rain is reported, some visibility will be retained but it will be substantially impaired. Successive flights through rain at supersonic speeds can cause rain erosion to occur to the radome, requiring a visual inspection of the radome after landing. Refer to Sections II and IV for procedures and operation of anti-icing and rain clearing equipment.

CAUTION

- Flight through areas of turbulent air, hailstones, or thunderstorms should be avoided whenever possible. Flight under these conditions increases the possibility of compressor stall or engine flameout and can result in damage from hail or turbulence.
- Above 40,000 feet, a rough operating condition or compressor stall may result if the throttle is moved rapidly, or rpm is reduced below 85%. Subsequent movement of the throttle may not eliminate this condition.

The most serious consequences of flight through severe turbulence and thunderstorms is the increased possibility of engine flameout. Of particular importance, is the "crystal ice" compressor stall. At high altitudes, ice crystals associated with the area around and in thunderstorms can cause compressor stall and probable flameout. The ice crystals do not settle on the intake duct but go into



"AN AIRSPEED OF 275 TO 325 KNOTS IAS, NOT TO EXCEED .88 MACH, IS RECOMMENDED FOR SAFE COMFORTABLE TURBULENT AIR PENETRATION."

22082

the engine with the air. The crystals are then heated during the compression process, and become ingested water. The ingested ice crystals reduce the compressor stall margin and compressor stall or flameout follows. There are several other factors associated with turbulence and thunderstorms which are also conducive to flameout:

- High liquid content of cumulus buildups.
- Icing of engine intake ducts or inlet guide vanes.
- Increased angles of attack caused by turbulence, resulting in marginal engine performance.
- Above 40,000 feet, the surge margin of the engine is reduced and there is poor air distribution across the face of the compressor.

CAUTION

Flying in turbulence or hail may increase inlet distortion. At higher altitudes this distortion can result in engine surge and possible flameout. However, normal engine air start may be accomplished.

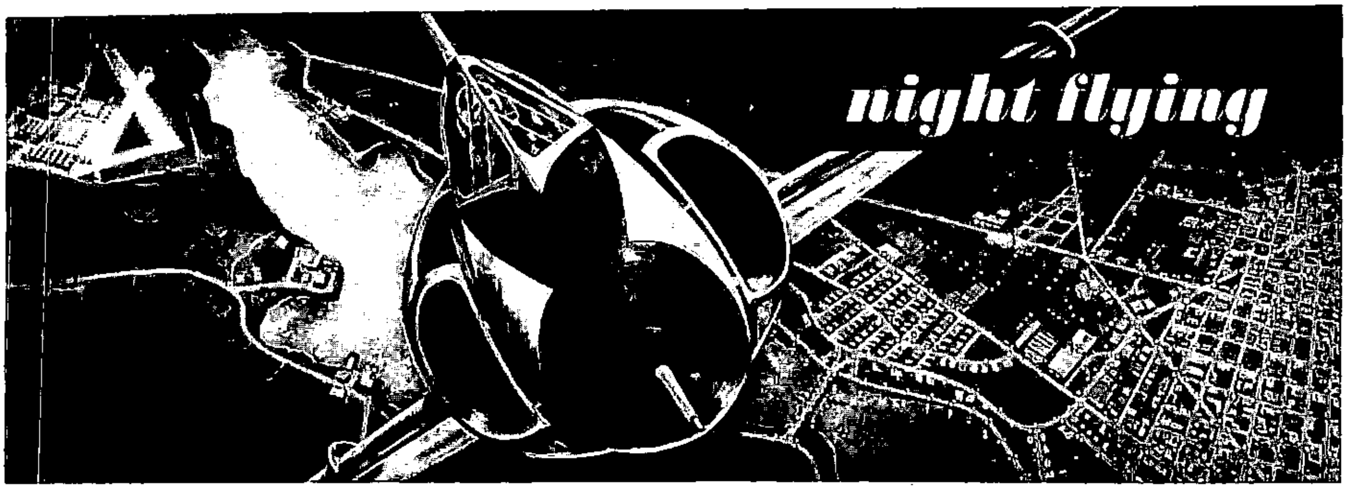
If thunderstorm areas cannot be avoided, use ground radar (GCI) or the fire control system radar (search mode) to determine and avoid the most intense storm areas. Establish a safe, comfortable penetration speed of 275 to 325 KIAS (not to exceed Mach .88) if such speed does not interfere with accomplishment of the mission. However, design limit load factors will not be exceeded at any speed below the design limit speed. Prior to weather penetration, make certain that pitot heat is on and that the anti-ice switch is in MANUAL ON position. The engine anti-ice system prevents ice formation but is not fully effective unless used before ice buildup occurs. Exhaust gas temperature and engine pressure ratio gauges should be monitored continuously during penetration to detect possible engine malfunction. Exhaust gas temperature indication alone may come too late to take timely action in preventing flameout. On airplanes having all points ignition capability, which permits engine ignition to be initiated regardless of throttle position (non afterburning), depress and hold the ignition button during the period of thunderstorm penetration. Under these circumstances, continuous use of the ignition system is permissible for a period not to exceed 10 minutes. After 10 minutes the ignition system must be turned off for a cooling period of at least 10 minutes.

CAUTION

- Continuous use of the ignition system in excess of 10 minutes, or consecutive continuous usages without a 10-minute interim cooling period may result in damage to the ignition system which will render it inoperable for restarts. Energizing the ignition system will not prevent compressor surge or stall, but it will aid in preventing flameout.
- During the period of ignition on, it is essential that EGT and EPR be monitored closely for possible increase above allowable limits. In event of high EGT and/or EPR, reduce thrust and altitude and maintain adequate airspeed for optimum engine inlet airflow.

Note

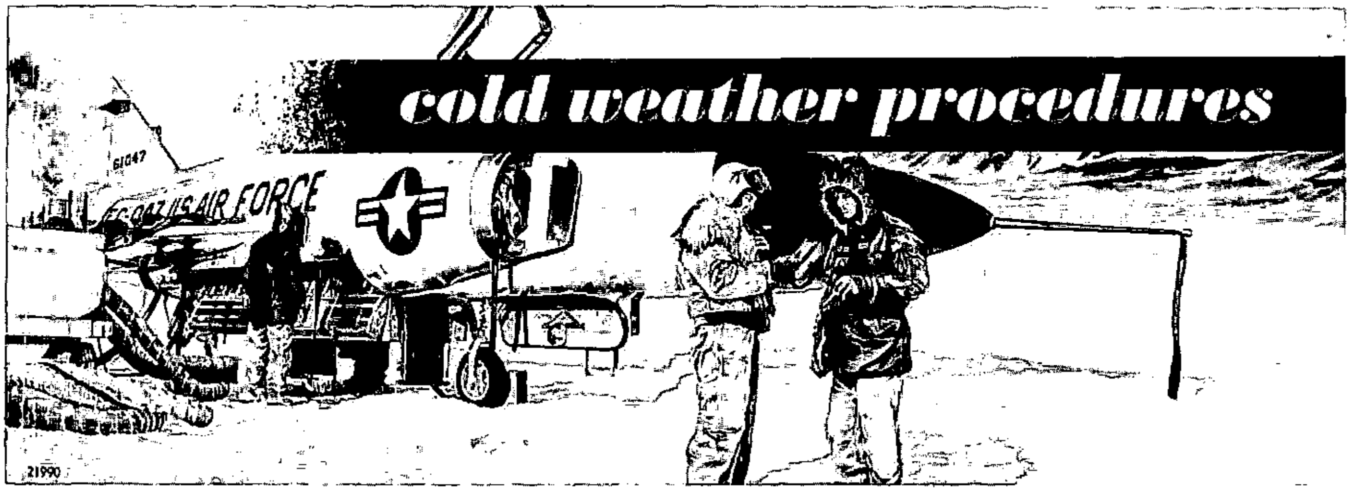
In event the ignition system is energized continuously for a period in excess of three minutes, an entry should be made on Form 781 noting the period of continuous ignition operation.



Night flights in this airplane do not present any special problems or techniques except during landing. When lowering the nose, it will be necessary to switch from landing to taxi lights to illuminate the area ahead of the airplane.

WARNING

Airplanes in the rotating beacon configuration do not provide adequate exterior lighting to meet night formation requirements.



To insure satisfactory cold weather operation, utilize the normal operating procedures outlined in Section II in conjunction with the following additions:

BEFORE ENTERING THE AIRPLANE

Failure to remove snow and ice accumulated on airplanes while on the ground can result in serious aerodynamic and structural effects when flight is attempted. Depending on the weight and distribution of the snow and ice, takeoff distances and climb-out performances can be adversely affected. This roughness, pattern, and location of the snow and ice can affect stall speeds and handling characteristics to a dangerous degree. To eliminate these hazards and insure satisfactory performance, check that all snow and ice is removed from the airplane surfaces. Assure that all protective covers are removed.

BEFORE STARTING ENGINE

When attempting a normal start, if start is not obtained, the start should be aborted and a second start initiated immediately.

Note

In order to make an immediate second start attempt, the airplane must be supplied with external ac power to resupply fuel to the starter fuel flask. If no external power is available, refer to COMBUSTION START, SECOND ATTEMPT, Section VII.

If the second start fails, the procedure may be repeated after the ground compressor has been recharged.

WARMUP AND GROUND CHECKS

Special attention should be paid to operational checks on all ice protection and defogging equipment. Refer to Section IV for anti-icing systems operation.

WARNING

In cold weather, make sure all instruments have warmed up sufficiently to insure normal operation. Check for sluggish instruments during taxiing.

TAXIING INSTRUCTIONS

Increase space between airplanes while taxiing to provide safe stopping distance and to prevent icing of airplane surfaces by melted snow and ice in the jet blast of preceding airplanes. Taxi speed should be reduced when taxiing on slippery surfaces to avoid skidding. Emergency fuel system may be used to reduce thrust if normal idle causes too high a taxi speed.

BEFORE TAKEOFF

If other airplanes are in takeoff position, taking the runway behind them should be avoided if possible. Flying debris in the form of ice, snow, and ice fog from other jet engines can considerably reduce visibility prior to takeoff or during takeoff roll. Brakes alone cannot be relied upon to prevent skidding of the airplane when operating on a slippery surface above approximately 85% rpm. It will be necessary to restrain the airplane by methods other than brakes if a normal full military thrust check is to be made before the airplane begins the takeoff roll.

CAUTION

If no means of restraining the airplane is available and the instruments are to be checked on the takeoff roll, do not hold the brakes while the engine is accelerating. It is possible to lose control of the airplane if one wheel slides ahead of the other during engine acceleration.

WARNING

Under conditions of high relative humidity, excess moisture through the air-conditioning system could cause fog condensation so dense that the instrument panel is not visible. In event this occurs, the cabin pressure switch should be placed to RAM and not returned to PRESS until above approximately 3000 feet.

TAKEOFF

During low-temperature operation, engine performance is considerably improved over normal temperature operation. Because of this improved performance, takeoff roll will be reduced and initial climb attitude will be much steeper than normal. Afterburner takeoffs may produce an uncomfortably steep initial climb angle.

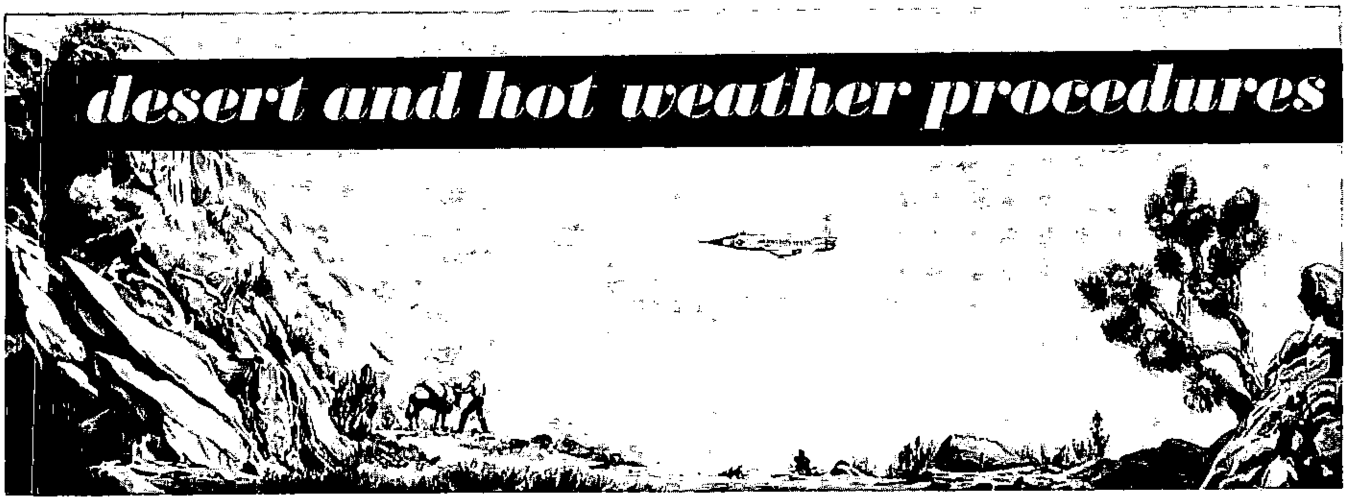
LANDING

Apply brakes carefully on icy runways, especially when some dry spots exist, to prevent skidding. Normal drag chute operation should be used.

CAUTION

When landing in a cross wind on icy runways, be prepared to jettison the drag chute should a weather-cocking tendency develop. Refer to Section II.

desert and hot weather procedures



In general, desert and hot weather procedures differ from normal procedures mainly in that added precautions must be taken to protect the airplane from damage due to high temperature and dust. Particular care should be taken to prevent the entrance of sand into the various airplane parts and systems (engine, fuel system, pitot static system, etc). All filters should be checked more frequently than under normal conditions. Units incorporating plastic or rubber parts should be protected as much as possible from wind-blown sand and excessive temperatures. Tires should be checked frequently for signs of blistering, etc. Takeoff and landing ground roll will be considerably increased in hot weather.

BEFORE ENTERING THE AIRPLANE

Check exposed portions of the shock strut pistons for dust and sand, and have them cleared if necessary. Check intake ducts for accumulations of dust or sand. Make sure crew chief has had all filters cleaned and that the airplane has been thoroughly inspected for fuel or hydraulic leaks caused by the swelling of packing or expanding of fittings. Inspect area behind the airplane to make sure sand and dust will not be blown onto personnel or equipment during starting operations. Check inflation of shock struts and tires, which may have become over inflated from the heat. Make sure that all protective covers are removed.

ON ENTERING AIRPLANE

Check the cockpit for excessive accumulation of dust or sand. Check instruments and control for moisture from high humidity, and ground heat them if necessary to dry them. Complete as much of preflight cockpit check as possible before operating, to avoid prolonged ground running.

BEFORE TAKEOFF

Note

Do not attempt takeoff in a sandstorm or dust storm. Park airplane cross-wind, shut down the engine, and have crew chief install protective covers.

The air-conditioning system should be turned on before takeoff. If, under humid climatic conditions, fog forms in the cockpit, adjust the cabin temperature control toward HOT until the fog disappears.

TAKEOFF

WARNING

Excessive moisture condensation may occur through the cabin pressurization system. This condensation may become so dense when operating under conditions of high dew point temperature that it may be impossible to read the instrument panel presentation. In the event this occurs place the cabin air switch to the RAM position, returning it to PRESS position after becoming airborne and above approximately 3000 feet.

CAUTION

It is imperative that takeoff be made at recommended speeds. Refer to Appendix for takeoff distances required at varying gross weights, temperatures, and field elevations. When outside air temperature is high, do not lift from runway too soon, as more than the usual takeoff distance will be required to obtain takeoff speed.

DURING FLIGHT

Note

During operation in high humidity conditions (ground dew point 20°C or higher), operate canopy defog continuously at altitude in order to insure fog-free canopy during descent. For less severe humidity conditions, canopy defog need not be turned on until just before descent.

DESCENT

Check that the windshield and canopy defog system is on at least four minutes before any rapid descent from altitude to prevent fogging and frosting of the windshield and canopy.

Note

Under high humidity conditions, the windshield defog system may not be capable of keeping the windshield clear of moisture.

APPROACH

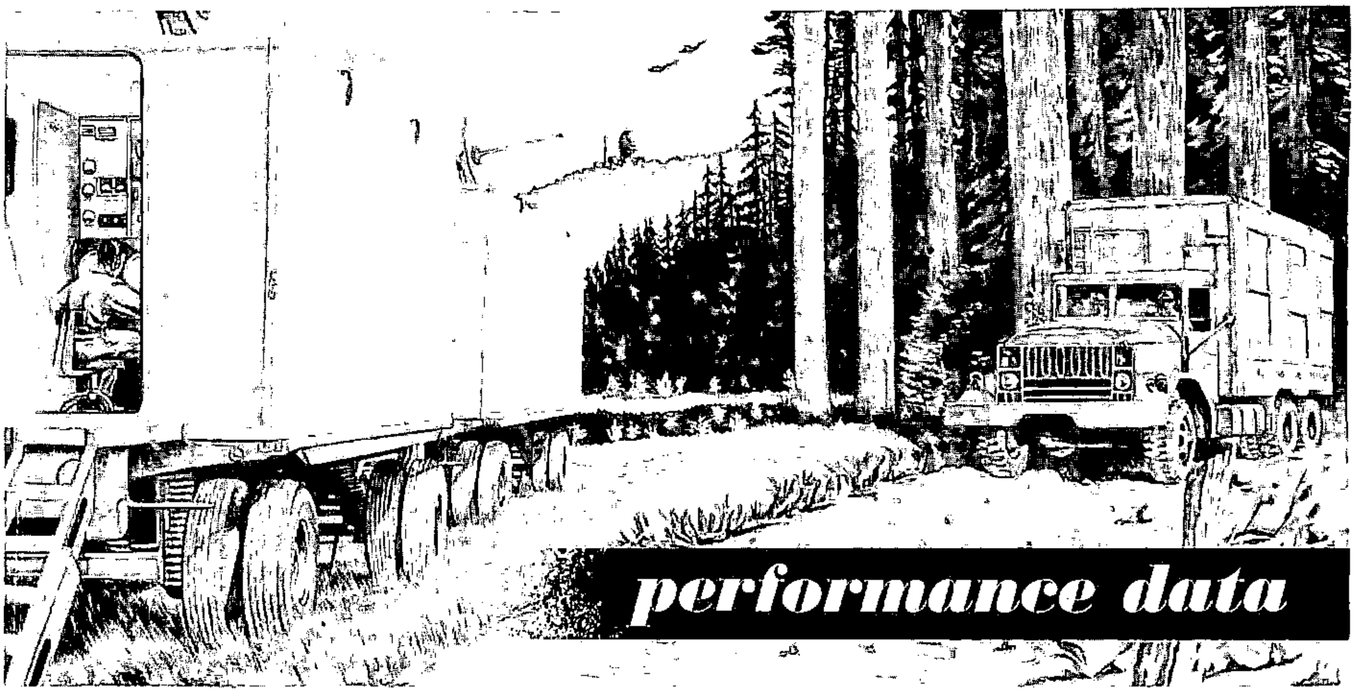
Maintain the recommended indicated airspeeds for approach and touchdown. Because of high outside air temperatures, the true airspeed will be higher than normal and longer landing roll will result.

LANDING

Avoid heavy braking during the landing roll. Small increments of braking with the drag chute deployed will stop the airplane in a reasonably short distance without excessive tire wear. Heavy braking may cause brake grabbing and tire failure.

BEFORE LEAVING AIRPLANE

Leave the canopy open to permit air circulation in the cockpit unless blowing sand or dust is expected. Have the crew chief install the protective covers from the pitot boom, ram air intakes, engine intake ducts, and tailpipe.

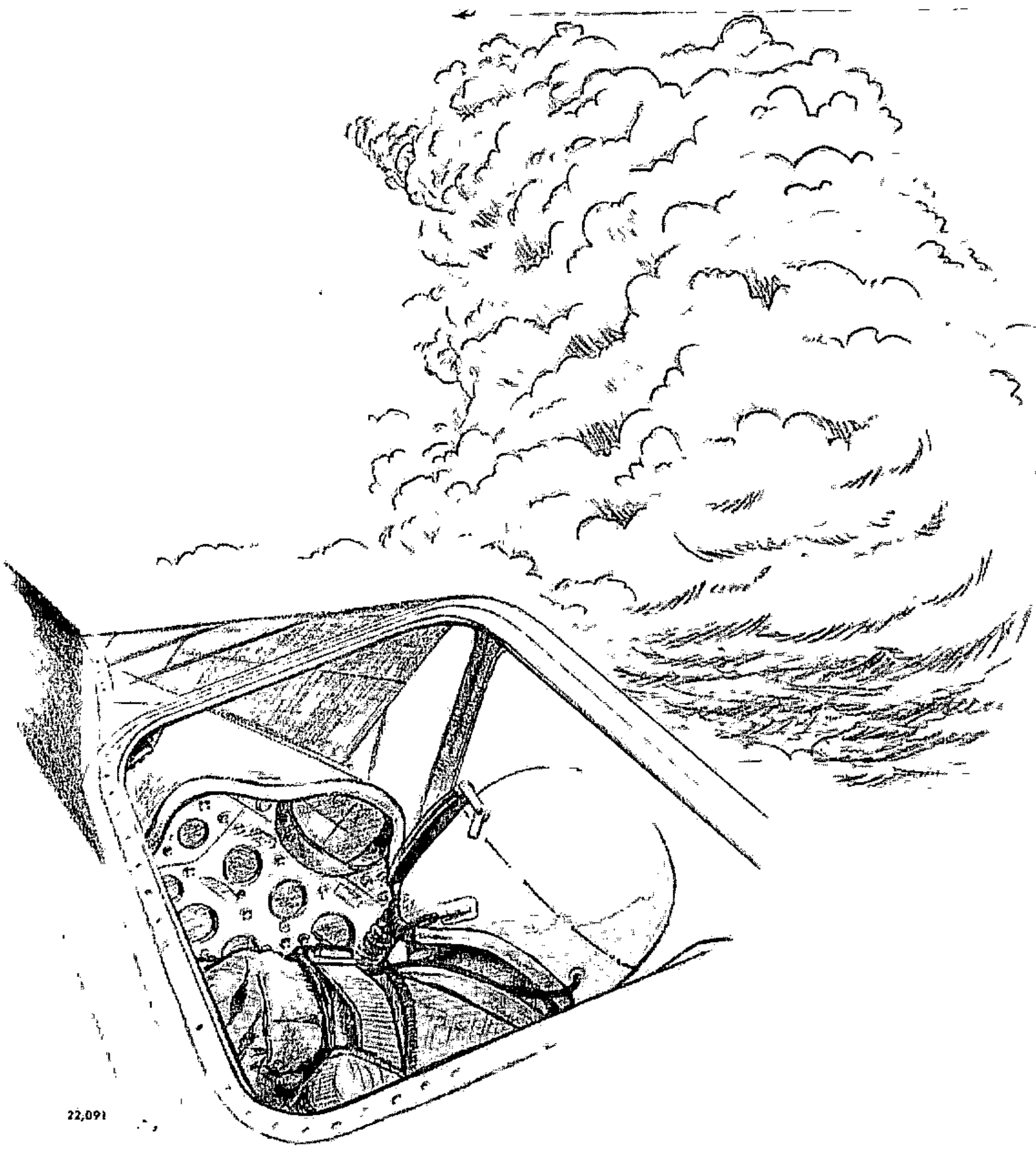


performance data

Note

Refer to the Confidential Supplement, T.O. 1F-102A-1A, for the following:

- Appendix I
- Appendix II
- Appendix III
- Appendix IV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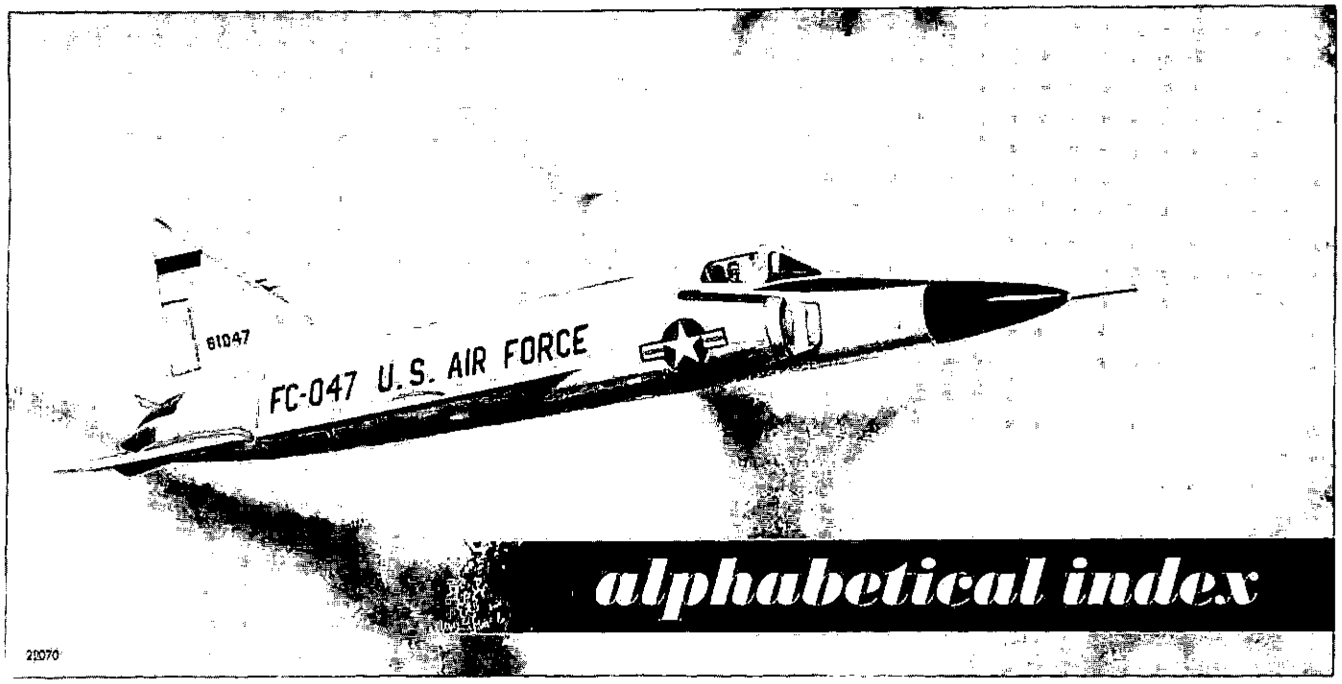

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F-102A

MAINTENANCE TRAINING SUPPLEMENT

(EXPERIMENTAL)

To Be Used in Conjunction With

T.O. 1F-102A-2-3
T.O. 1F-102A-2-7
T.O. 1F-102A-2-8

HYDRAULIC SYSTEM

CONVAIR
F-102A

MAINTENANCE TRAINING
SUPPLEMENT
HYDRAULIC SYSTEM

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE

AIR FORCE—AFPS—SAN ANTONIO, TEXAS—5-57-2600

1 JULY 1956

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Foreword

The F-102A Training Supplements have been prepared by Convair, A Division of General Dynamics Corporation, to provide you, the system mechanic or maintenance technician, with basic information concerning the F-102A airplane and its operating systems. There are ten of these supplements, each covering an airplane operating system or major component. The text is direct and informal and is supplemented with illustrations and diagrams to aid you in understanding how the systems operate. The following is a list of the Training Supplements on the F-102A airplane:

Title

Flight Control System
Hydraulic System
High-Pressure Pneumatic System
Low-Pressure Pneumatic System
Airplane General
Airframe Fuel System
Power Plant Installation
Airframe Armament System
Electrical System
Instruments

Each supplement describes an operating system or major component, how and why it operates, and maintenance problems that you may encounter. The entire major system is further simplified by describing each of its subsystems separately. Thus, when you understand how all of the subsystems operate, you will be familiar with the functions of the major system. This knowledge of the airplane systems will enable you to use other operational, service, and maintenance instructions.

Since the purpose of these publications is to familiarize and acquaint you with the airplane systems and components, specific values, measurements, and tolerances are not given. Any values that appear in these supplements are approximate only and are used to emphasize more clearly a certain operation or condition. You must refer to your 1F-102A-2-3, -2-7 and -2-8 Technical Orders and other pertinent handbooks for specific values when adjusting or checking a system and its components. These Technical Orders are revised periodically to include the latest maintenance procedures and data on the equipment installed in the F-102A.

The F-102A Maintenance Technical Orders consist of the following handbooks:

T.O. 1F-102A-2-1	**General Airplane**
T.O. 1F-102A-2-2	**Ground Handling, Servicing, and Airframe Group Maintenance**
T.O. 1F-102A-2-3	**Hydraulic and Pneumatic Power Systems**
T.O. 1F-102A-2-4	**Power Plant**
T.O. 1F-102A-2-5	**Fuel Supply System**
T.O. 1F-102A-2-6	**Air Conditioning, Pressurization, and Anti-Icing Systems**
T.O. 1F-102A-2-7	**Flight Control Systems**
T.O. 1F-102A-2-8	**Landing Gear**
T.O. 1F-102A-2-9	**Instruments**
T.O. 1F-102A-2-10	**Electrical Systems**
T.O. 1F-102A-2-11	**Radio-Communication and Navigation Systems**
T.O. 1F-102A-2-12	**Armament and Armament Electronics**
T.O. 1F-102A-2-13	**Wiring Data (F-102A)**
T.O. 1F-102A-2-13A	**Wiring Data (TF-102A)**

The Air Force is vitally interested in seeing that its equipment functions properly and is maintained as efficiently as possible. The Air Force, like a manufacturer of an automobile or commercial appliance, provides printed instructions for use with its equipment. The printed instructions you will use most frequently in maintaining the F-102A are the dash-2 Technical Orders listed above. They are available for your reference at the Maintenance Office on your base. You must refer to them frequently so that you will always have the latest maintenance infromation. The Training Supplements will help you to use these Technical Orders by answering many of the "hows" and "whys" that may puzzle you from time to time.

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Introduction

This supplement is arranged in four chapters and covers the F-102A Hydraulic System. Chapter I introduces the F-102A Hydraulic System and reviews the basic principles of hydraulics. Chapter II describes the power section of the primary and secondary hydraulic systems. Chapter III tells you how hydraulic power operates the elevon and rudder controls. Chapter IV explains the hydraulic operation of the landing gear, nose wheel steering, speed brakes, and the emergency a-c generator systems. A summary concludes this supplement.

Chapter I

FUNDAMENTALS OF HYDRAULICS

Contents *Page*

Today, we see the application of hydraulics all around us. Look in almost any repair shop and you see that the mechanic uses a hydraulic jack to lift heavy equipment. Many of you even carry a small hydraulic jack in the tool kit of your automobile. The portable lift which service station attendants use to raise your car for needed repairs gets its lifting force from hydraulic power. Dentists and barbers call on hydraulics to lift their chairs to convenient working heights when they move the hand lever on the side of the chair. Hydraulics supply the power that raise and lower some elevators from one floor level to another. Doorstops, using hydraulic principles, keep your doors from slamming.

Hydraulic power applications also produce the operating power for automobile brakes, power steering, automatic shifting of transmissions, elevating large artillery weapons, and propelling many commonly used machines. You might ask why our technological improvements have brought about this widespread use of hydraulic power or the increased use of hydraulics to operate aircraft subsystems.

The many common uses of hydraulic machinery have resulted from the fact that hydraulic systems possess many favorable characteristics. To mention a few, they eliminate the need for a complicated system of gears, cams, and levers to do work—the motion is transmitted by hydraulic systems without the slack or play that accompanies the use of mechanical linkages. Also, since hydraulic systems use a liquid to do the work, there is comparatively little wear and tear inflicted upon the mechanisms of these systems, thus resulting in many troublefree hours of operation.

When properly applied, hydraulic power provides smooth, flexible, uniform action that does not vibrate. All this is possible, plus the fact that this power can be transmitted up, down, and around corners, with little or no loss in efficiency and without the use of complicated mechanisms.

Hydraulic power, because of these many advantages, solves many of the problems of supplying power for operating components in airplanes. An airplane is such a large and complicated unit that it is not practical and often not possible to operate all of the systems with power taken directly from the engine or engines. Units such as the landing flaps and bomb bay doors may be some distance from the engine and to provide a gear train or mechanical linkage from the engine to the unit would be quite cumbersome. Hydraulic power is the perfect solution for problems such as this. In addition to the units mentioned above, landing gear and brakes are often operated hydraulically, and on most of the supersonic aircraft, hydraulic systems are often provided for nose-wheel steering and for operation of the flight controls.

The hydraulic system used in the F-102A is shown diagramatically in figure 1-1. You will note that the system actually consists of two separate systems, the primary and secondary. The primary system supplies power only to the flight control subsystem while the secondary system supplies supplemental power to the flight control subsystem as well as power to all of the other hydraulically-operated subsystems in the airplane.

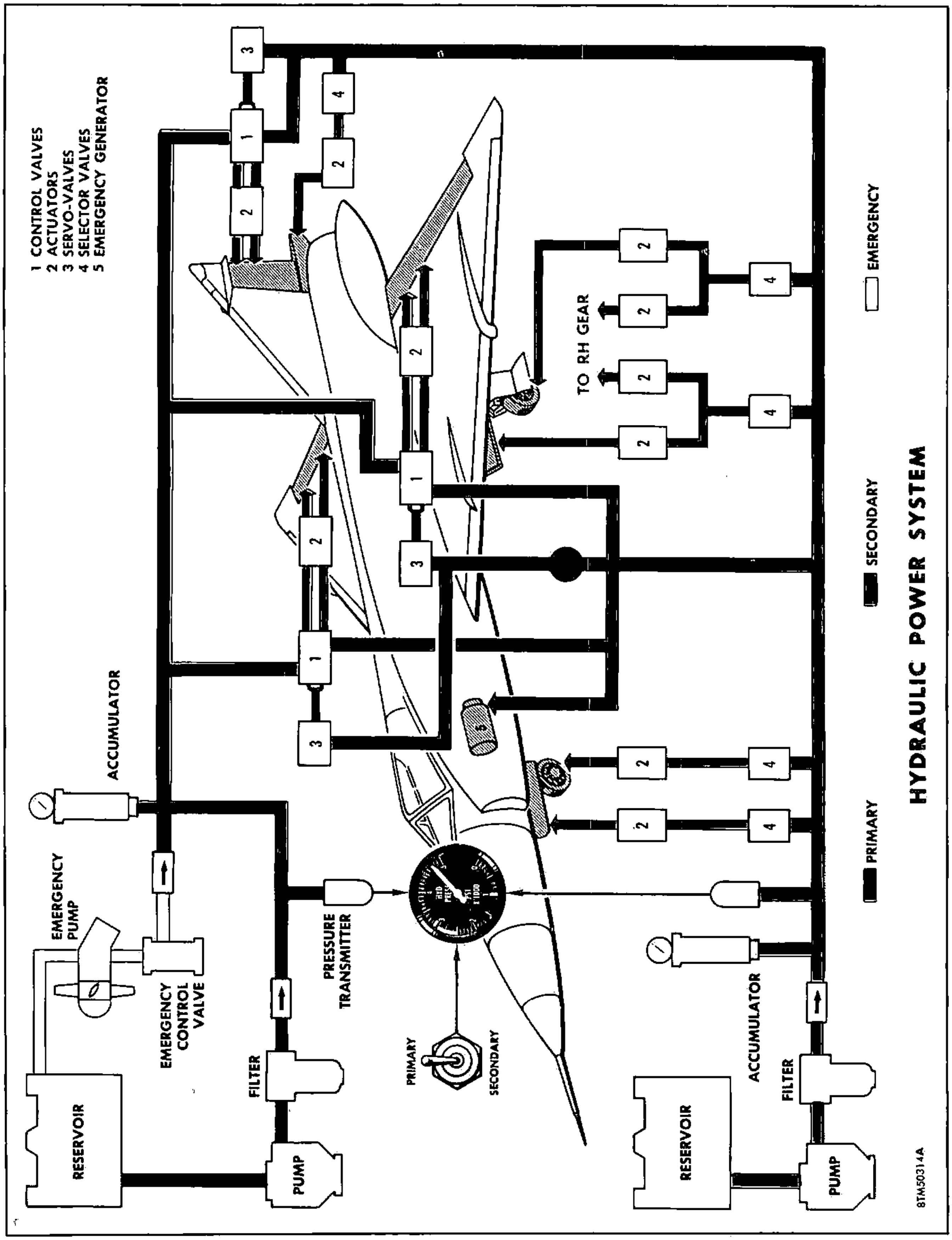

Figure 1-1. The F-102A Hydraulic System

Hydraulic pressure is supplied to these systems by two engine-driven, variable-displacement pumps. As you will note in the illustration, each pump supplies its respective system continuous hydraulic pressure. With the exception of the two engine-driven pumps and the high pressure filters, all of the components of both power supply systems are located in the hydraulic accessory compartment. This compartment is located on the right side of the fuselage just forward of the main landing gear wheel well.

The emergency hydraulic power system, shown in yellow on figure 1-1, consists primarily of a ram-air, turbine-driven pump which is connected to the primary system in such a way that it substitutes for the primary system pump in the event the primary system pump fails. The emergency system ram-air turbine unit is attached to the inboard side of the hydraulic accessory compartment door. The airplane's high pressure pneumatic system, controlled from the cockpit, supplies power to open this door and hold the ram-air turbine in the airstream during emergency flight operations. The emergency system has no hydraulic or mechanical connection with the secondary system.

The landing gear wheel brake system has a small, enclosed hydraulic system that works in conjunction with the high pressure pneumatic power system to operate the wheel brakes. This hydraulic system, however, is completely separate from the F-102A main hydraulic systems and is discussed in another Training Supplement of this series which describes the landing gear system and its operation.

This short description of the F-102A hydraulic system has given you a brief introduction to the system and made you aware of some of the system's functions and responsibilities. After we have reviewed some of the basic principles of hydraulics, we will take a longer, more detailed look at the F-102A hydraulic system. Then, in subsequent chapters, each portion of the system and its components will be discussed and analyzed in detail.

HYDRAULIC PRINCIPLES.

Before beginning the familiarization tour of the F-102A hydraulic system, let us review some of the principles of hydraulics and fundamental operations of components that are common to all hydraulic systems. No matter how complex a system might be, a sound knowledge of basic hydraulic principles helps us to understand how that system operates.

Since bulky or heavy equipment is not desired in an airplane, hydraulic system equipment is always designed to be as small and as compact as possible. In many cases, several components of a subsystem may be combined into one assembly. This particular assembly may then appear complex to you. However, keep in mind that the basic principles of operation are *always* the same, and that reduction in size, change of position, or the combination of components do not affect these basic principles. For this reason, emphasis is added to the basic principles of operation employed in the simplest units found in hydraulic systems. A complete knowledge of these fundamental units will make it easy for you to understand any of the arrangements that you may encounter in various aircraft hydraulic systems.

PASCAL'S LAW OF HYDRAULICS.

Hydraulics is based upon the discoveries of Pascal, a seventeenth century French scientist. Pascal proved that *pressure set up in a liquid acts equally in all directions,* and that *this pressure acts at right angles to the containing surfaces.* This law holds true in all of the applications of hydraulics that we will discuss in this supplement.

A good example of this law is the action of the fireman's water hose shown in figure 1-2. On the truck, this hose is folded flat and compactly placed. But when the hose is connected to the fire plug and the water is turned on, the hose opens up into a round, snake-like tube—water pressure is acting on the inside of the hose in *all directions.* If, at any point in the hose, you were to puncture a hole the same size as the nozzle opening, water would rush out of the hole at the same force that it flows through the nozzle end of the hose.

THE SHAPE OF CONTAINER HAS NO EFFECT ON THE PRESSURE.

A liquid is a fluid with many particles that have freedom of movement among themselves, but lack the ability to separate. All liquids have a definite volume that takes the shape of the container which holds it. By definite volume, we mean that any quantity, such as one quart, will remain one quart whether it is put into a one quart container, a one-gallon container, or a five-gallon container.

From Pascal's law, we derive the fact that pressure exerted on any part of a confined liquid is transmitted to all parts of the liquid, regardless of the shape of the container. In all cases, the pressure in a container of one shape is the same as the pressure in a container of any other shape if the two containers are interconnected and pressurized from the same source. This principle is shown in figure 1-3. Each container in the illustration has a different shape but, since all containers are connected, the fluid in each has the same pressure throughout.

In an aircraft hydraulic system, this principle of constant pressure in different shaped containers applies to the fluid confined in the system tubing. The fluid transmits pressure from one location to another. The pressure is the same at either end of the tubing regardless of the number of bends, turns, or the length of the tubing.

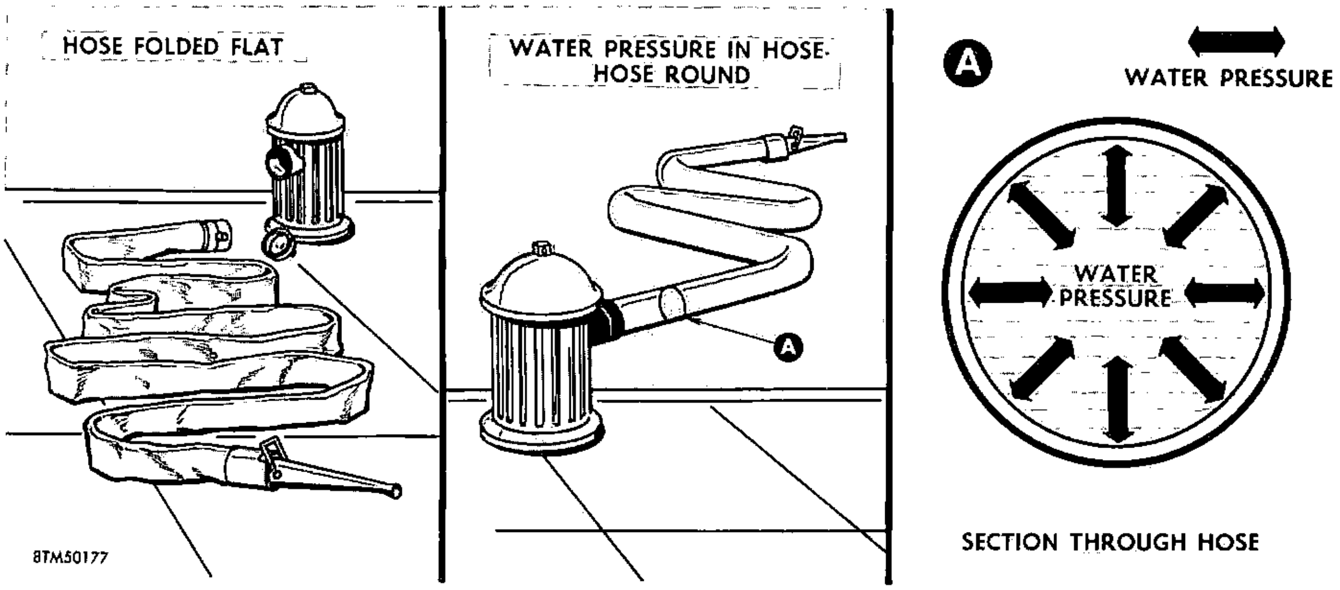

Figure 1-2. Pressure Acts Equally in All Directions

THE INCOMPRESSIBILITY OF LIQUIDS.

As just mentioned, liquids do not have a specific shape and they take the form of the container that holds them. In spite of this fact, however, liquids are even less compressible than most solids. Compressibility is the ability of a substance to occupy a smaller space, or volume, when an outside force is applied to this substance. Because liquids are not compressible, a hydraulic system has instantaneous response to the forces acting upon it. The liquid acts much like a solid steel bar—if liquid in one end of a cylinder is forced back by a piston, an equal amount of liquid on the other end is immediately displaced.

This instant response action is one of the main advantages of a hydraulic system. The rigidity of the liquid in the fluid line means that transmission of force between two hydraulic mechanisms is fast and complete. If hydraulic systems do not leak or have faulty operating components, their output efficiency is usually very high—most hydraulic systems have efficiencies above 95 percent.

HOW PRESSURE AFFECTS FLUID.

The air around us exerts a downward force on the surface of the earth. Although we do not notice this atmospheric condition, it is strong enough to support a column of water 33 feet high. This force, 14.7

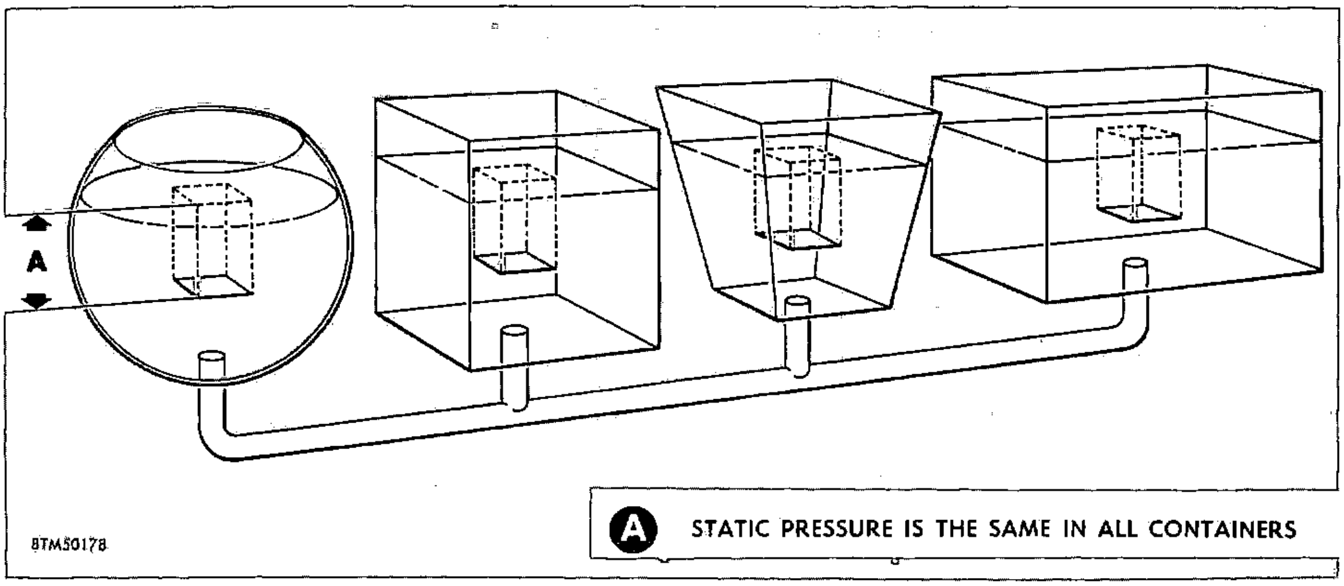

Figure 1-3. Shape of Container Does Not Affect Pressure

pounds per square inch (psi), is the standard *atmospheric pressure* at sea level. Figure 1-4 shows the effect of atmospheric pressure at sea level on water with respect to its effect on mercury. Note in the illustration that atmospheric pressure raises the column of water 33 feet in an evacuated tube while it raises the column of mercury only 29.92 inches. This 29.92 inches of mercury is the standard barometric pressure at sea level. At elevations above sea level, these readings change because the atmospheric pressure decreases as altitude increases.

Atmospheric pressure is the reason that we are able to drink soda pop with a straw. You suck the air out of the straw, thus creating a partial vacuum, or a low air pressure area, in both the straw and in your mouth. The atmospheric pressure that is constantly bearing down on the soda pop at a force of 14.7 psi, consequently pushes the soda pop up into the low pressure area that you produced in the straw and in the mouth. Immediately after the suction caused the air pressure in the straw to drop below 14.7 psi, the atmospheric pressure which is constantly bearing down on everything pushed the soda pop up through the straw and into the mouth in its effort to again balance the two pressures.

HYDRAULIC PRESSURE.

The fact that liquids are incompressible explains why the force applied to one end of a tube of liquid is immediately transmitted to the other end. However, this transmission of force is not only applied to the other end, but in all directions. As you will remember from Pascal's Law, the pressure is equal in every direction throughout the column—forward, backward, and sideways. Throughout a containing tube, the pressurization is always the same. This statement holds true for all tubes, cylinders, and other units that are filled with fluid, and acted upon by some outside force.

Pressures of liquids are measured by dividing the applied force by the *area* of liquid upon which the force acts. This is expressed in the formula:

$$\text{PRESSURE} = \frac{\text{FORCE}}{\text{AREA}}.$$

Figure 1-5 shows a practical application of this formula. If five pounds of force is applied to piston No. 1 and the area of the piston is two square inches, then the pressure delivered by cylinder No. 1 would be 2.5 psi. However, if this same five pounds of force were applied to the large piston No. 2, with an area of 5 square inches, the pressure delivered by cylinder No. 2 would be 1 psi. Although the force on both pistons is the same, the resultant pressure is less in the larger cylinder because the force on its piston is spread over a larger area.

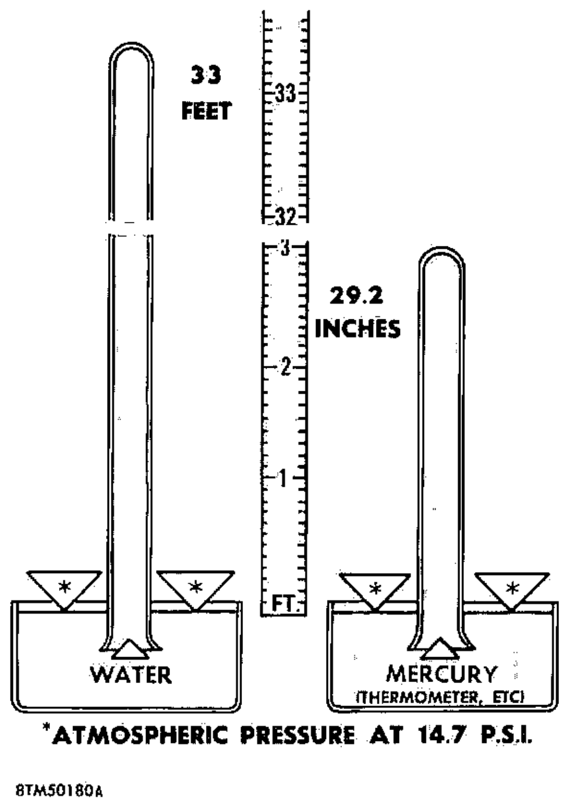

Figure 1-4. Atmospheric Pressure

Now that we have reviewed how pressure is applied to fluids, let us see how this pressure can be made to do work. In figure 1-6, piston A applies a force to the system. This force produces the pressurized fluid that moves piston B several inches in the opposite direction and thereby causes the system to do work. The surface area of piston B, the actuating piston, determines how far it is moved. The smaller the surface area of the actuating piston the farther it must travel to offset the force of the fluid that acts upon it. A rule to follow, because of Pascal's Law, is: *The force acting on a piston will be directly proportional to the area of the piston* and *the distance the piston is moved depends directly upon the amount of liquid transferred.*

MECHANICAL ADVANTAGE IN A HYDRAULIC SYSTEM.

The size of the pistons in a hydraulic system determines how much work the system is able to do. In figure 1-7 the force produced at piston No. 2 is 2½ times the force applied at piston No. 1, simply because piston No. 2 is 2½ times larger than piston No. 1. The mechanical advantage of this system is therefore 2½ to 1.

The mechanical advantage of hydraulic systems varies in direct proportion to the area of their actuating

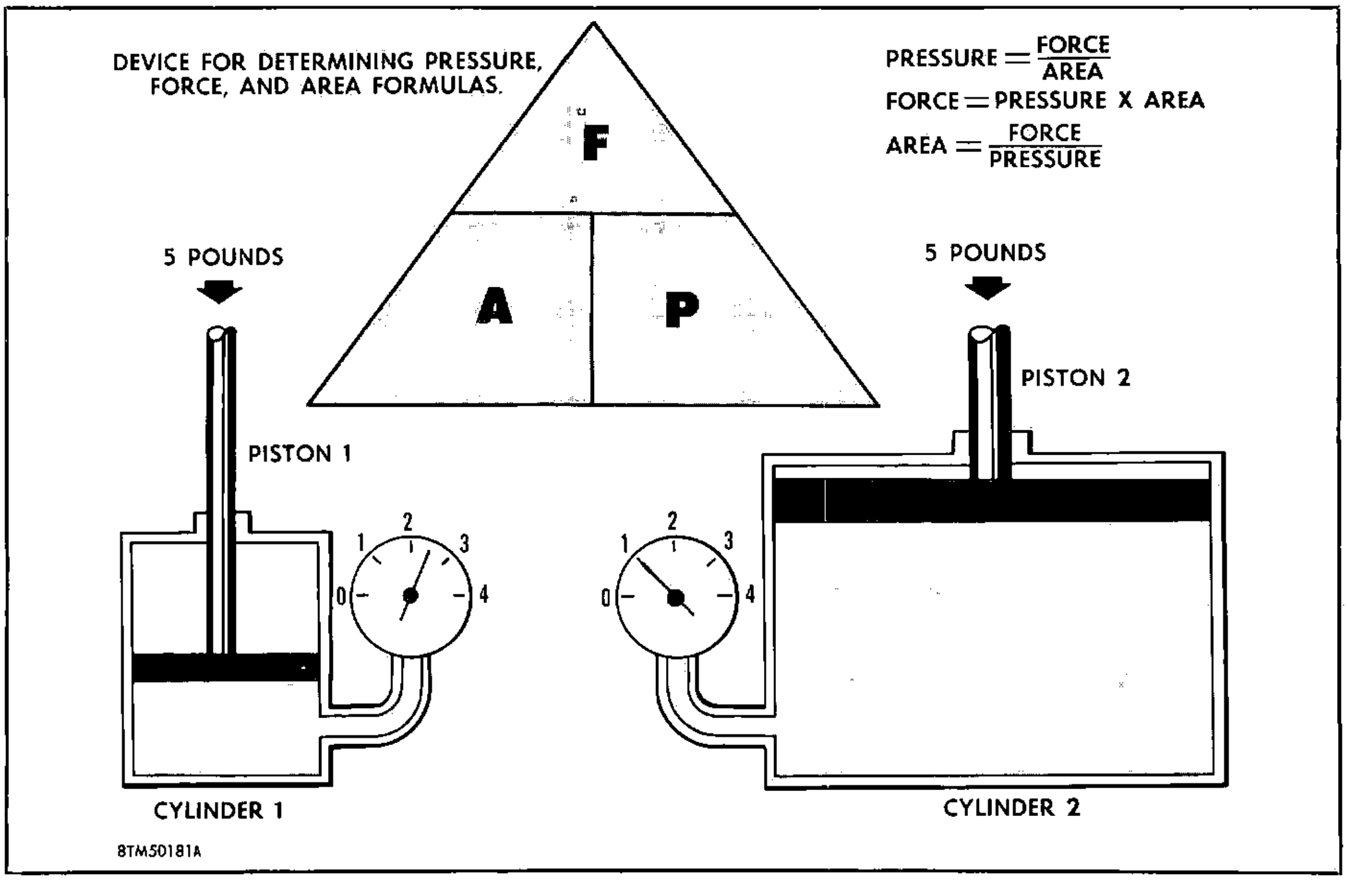

Figure 1-5. Hydraulic Pressure

pistons. Looking again at figure 1-7, imagine that piston No. 2 is twice its present size and is now 10 square inches. In such a case the mechanical advantage of the system becomes 5 to 1, but the displacement of piston No. 2 is less. However, if we were to decrease the size of piston No. 2 we would have lesss mechanical advantage than before, but at the same time we would increase the displacement of piston No. 2. In each case the input force applied to piston No. 1 is the same, although both the output and mechanical advantage were varied by varying the actuating piston area.

HOW A HYDRAULIC SYSTEM OPERATES.

A simple hydraulic system consists of a reservoir that holds the fluid, a pump or master cylinder for producing pressure, and an actuating cylinder which transforms the pressure into mechanical motion. Figure 1-8 shows one of the simplest types of hydraulic systems. Notice that a selector valve is added to the system to permit two-way actuation of the surface we want to move.

The hand pump is the key unit in this simple system. By pushing the pump handle in the direction of the arrows, valve A is closed while valve B is forced open by the fluid displaced from the cylinder of the pump. At the end of this pressure stroke, the pump piston is pulled back by the pump handle and the cylinder is again refilled with fluid from the reservoir.

In this refilling cycle, valve A is opened and valve B is closed by its spring action. Now the pump is ready for another pressure stroke. Hydraulic fluid under pressure leaving the check valve at B is routed to the selector valve. With the selector valve in the UP position, the pressure enters one end of the actuating cylinder and displaces the piston. The movement of this displaced piston moves the attached surface in the desired direction. At the same time that the piston moves the surface, it also forces the fluid on the unpressurized side of the piston back through the selector valve to the reservoir. If we were to move the selector valve to the DOWN position, the movement of the piston in the actuating cylinder would be reversed and the surface would move in the opposite direction.

A POWERED HYDRAULIC SYSTEM.

The need for more powerful, continuously operating hydraulic systems brought about the development of the powered hydraulic system. With the addition of mechanical power to the hydraulic system, automatic safety and control devices are needed to control the system pressure. These additional control units make powered hydraulic systems look complex, but, as previously mentioned, the basic operation of a hydraulic system is always the same.

In figure 1-9, which shows a powered hydraulic system, an electric motor rather than a hand pump supplies power for the system. As you will note, the powered pump in this system draws hydraulic fluid from the reservoir into the inlet side of the pump. The pump then forces the fluid out under pressure, and the fluid passes through a filter which removes any foreign matter. The fluid must then pass through several pressure and flow control devices before it starts the actuator to operating.

The check valve, shown to the right of the filter, has no effect on fluid flowing to the actuator, but it does prevent any reverse flow of fluid that might damage the pump. The relief valve, located downstream of the check valve, acts as the safety valve for the system. During normal system operation, the fluid does not pass through the relief valve, but when the system pressure becomes higher than normal, the relief valve opens and allows the excess pressure to return to the reservoir.

Now the fluid goes to an accumulator where any pressure surges in the system caused by the pump are smoothed out. Most accumulators have either a diaphragm or a movable piston which is backed up by a compressed gas (air or nitrogen). This gas acts as a shock absorber and cushions all surges in the system pressure. The fluid continues past this accumulator to the control valve. The control valve directs the fluid under pressure in one of two possible directions—to either the open or the close side of the actuating cylinder.

When the control valve is operated, the pressure then goes to one side of the actuator and moves the actuator piston causing the piston rod to move the control surface. The movement of the actuator piston forces return fluid out the other side of the actuator, back through the control valve, and into the reservoir. This cycle of the fluid pressure from the reservoir to the pump, to the control valve, to the actuator, and back to the pump by way of the reservoir, makes the system operation a continuous cycle.

Comparing this powered system with the previous illustration of the simple hydraulic system (figure 1-8), note that an emergency pump is connected in parallel with the powered pump and that an additional control valve has been added. The emergency pump supplies power to the system when the main pump malfunctions. The control valve permits this emergency power to enter the system and serves as a check valve when the emergency system is not in operation. The check valve in the main system prevents this emergency power from backing up to the main pump.

CLOSED-CENTER POWERED HYDRAULIC SYSTEM.

Closed-center hydraulic systems are continuously pressurized systems—the action of the powered pump is

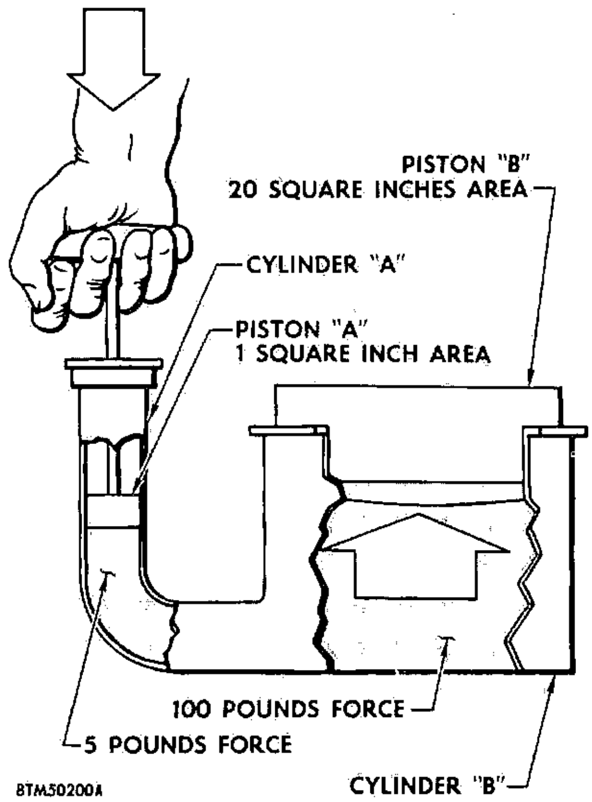

Figure 1-6. Hydraulic Pressure as a Source of Power

continuous but fluid circulation stops during the time that the hydraulic components are not in operation.

At one time, open-center hydraulic systems were also in wide use. This type of system was designed so that hydraulic fluid constantly circulated from the reservoir through the pump to the selector-type control valves, and back to the reservoir at a relatively low pressure. The hydraulic system demands of the supersonic age airplane, however, have pushed this type of system into obsolescence and all of the modern airplanes use the closed-center type of system.

The closed-center type system shown in figure 1-9 shows the basic actions of all closed-center types of hydraulic systems. Fluid pressure produced in the pump is made to pass through a filter unit before it enters the regulator. The two ball-shaped poppets in the regulator are forced to move downward—opening the left port and closing the right port. The fluid flows through the left port, passes the accumlator, and goes on through the control valve into the actuating cylinder.

After looking at this system for awhile, you have no doubt come up with several questions. First, you might ask how the pressure is stopped from acting downward in the piston when the actuation stroke is

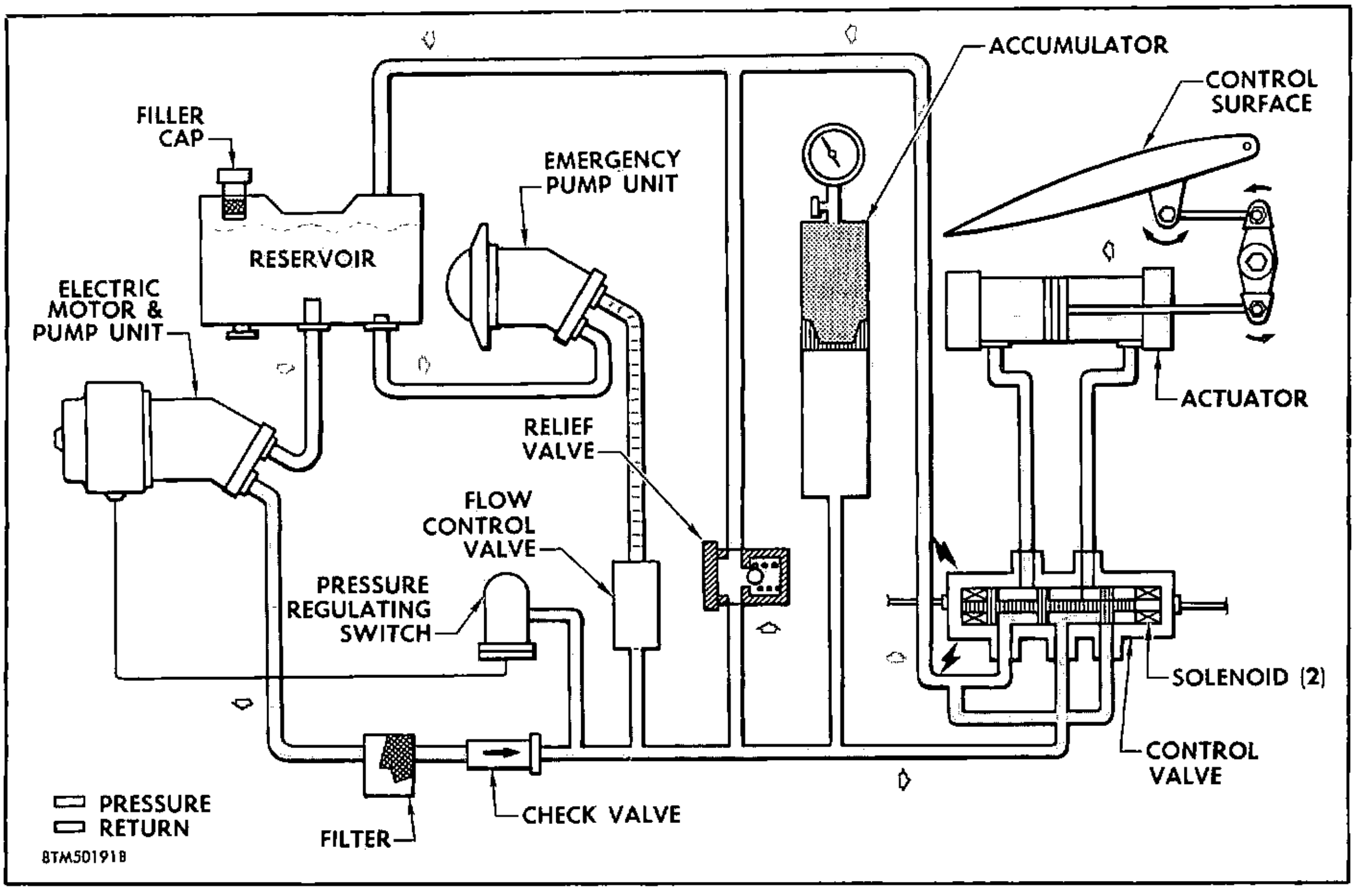

Figure 1-9. A Powered Hydraulic System

completed. And second, how is the piston forced up in the opposite direction? Once the piston has reached its full travel, the pressure acts in all directions and overcomes the *weakest* resistance. This weakest resistance is the large piston in the regulator that is being held down by the pressure on the right ball poppet. Since the area of this piston is much larger than that of the poppet, the piston is moved up, thus opening the right fluid port in the regulator.

In this case, the fluid from the pump circulates through the regulator and back to the reservoir. The pressure output from the pump is now closed to the elements downstream of the regulator and fluid circulates only from the pump to the regulator, and then back to the reservoir. Pump operation is continuous, but in this closed-center type system, pressure does not flow continuously through the control valve as it would in the open-center type system.

To reverse the direction of piston movement in the actuating cylinder, the two-way control valve must be rotated to change the direction of flow. As the valve rotates and allows pressure to the lower side of the piston, the top side of the valve becomes the return side. After the control valve has been repositioned, the action of the system is the same as before.

There are two big advantages of closed-center systems. First, this type system responds much faster than on open-center type system. Also, closed-center systems cause less wear and tear on their components since they do not circulate fluid continuously.

The F-102A has a closed-center type of hydraulic power system incorporating a constantly operating pump which automatically changes its pressure output to offset normal operating pressure changes. The effectiveness of this system and all other hydraulic systems depends to a great extent on the operation of its fluid pressure-generating pump—most maintenance problems are traced back to the faulty operation of a pump. However, the addition of electrical and mechanical devices into hydraulic power systems for control purposes, has made systems like the two in the F-102A, very efficient operating systems.

COMPONENTS OF A HYDRAULIC SYSTEM.

As hydraulic systems are called upon to do more accurate work at higher operating pressures, these systems require more components to provide proper control. Although the closed-center and the open-center hydraulic systems are entirely different in operation, they have almost identical components. To clarify the functions of the many components in hydraulic systems, the components can be separated into three subdivisions—the pressure supply group, the pressure con-

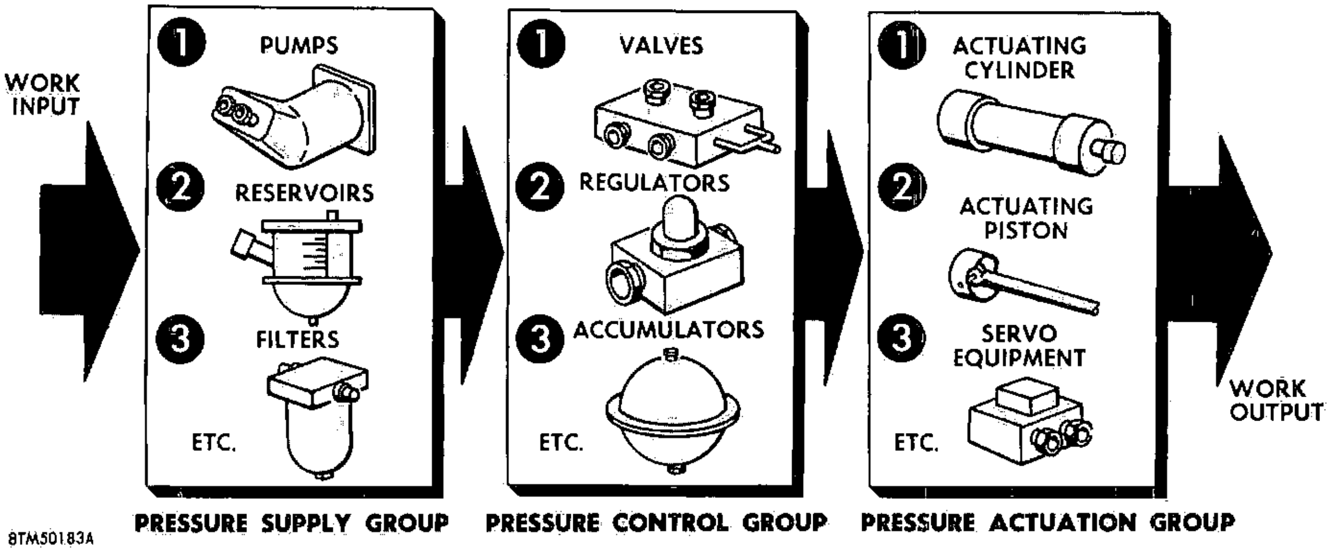

Figure 1-10. Components of a Hydraulic System

trol group, and the pressure actuation group. The components in the three different categories are shown in figure 1-10.

Because the closed-center hydraulic systems are used in most modern aircraft hydraulic systems, each component in this chapter is discussed as it would function in a typical closed-center system.

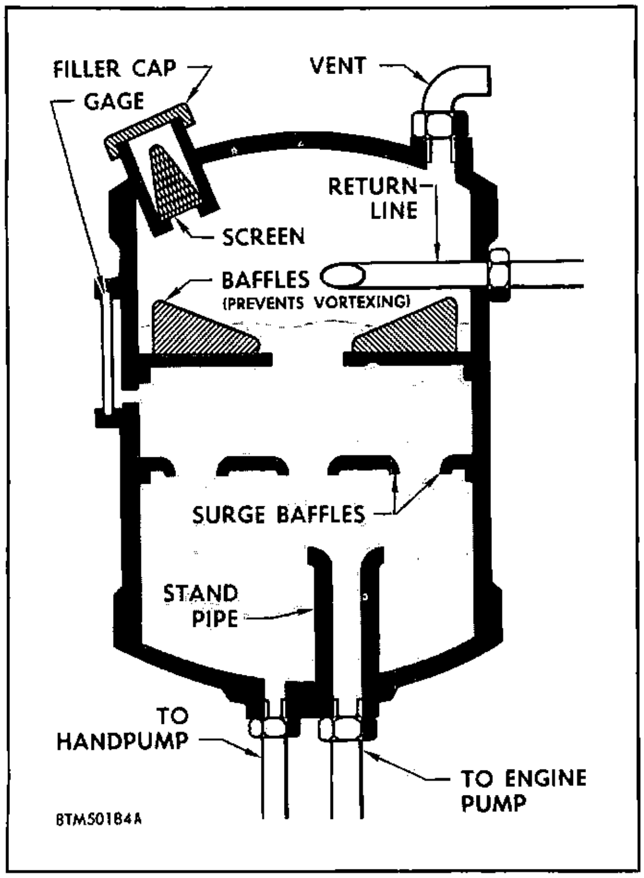

Figure 1-11. Typical Reservoir Diagram

PRESSURE SUPPLY GROUP.

The pressure supply group in a hydraulic system supplies the initial force to pressurize the system. This part of the system usually consists of a reservoir, a pump, and a filter.

Reservoirs.

Reservoirs, like the one shown, are fluid storage tanks that also serve as overflow basins for return fluid that is forced out of actuating cylinders and bled off by relief valves. The reservoir supplies clean fluid to the main system pump and to the auxiliary hand pump or emergency pump.

Figure 1-11 shows a simple hydraulic reservoir, the kind used in low-pressure hydraulic systems. This reservoir is usually located above the pumps to which it supplies fluid so that the force of gravity pushes the fluid down to the pumps. Notice that a filler screen just below the filler cap prevents foreign matter from entering the reservoir when it is being replenished with fluid. The air vent at the top opens the reservoir to atmospheric pressure which prevents a partial vacuum from forming in the tank as hydraulic fluid is supplied to the pumps.

On the left side of the reservoir, you will note the glass sight gage that indicates the fluid level in the tank. This gage is calibrated to indicate when the reservoir is full and when the fluid level has fallen to a point where additional fluid is required. The baffles in the reservoir are installed at two levels, one at the surface of the fluid and the other within the fluid. These baffles, which are small metal vanes, prevent

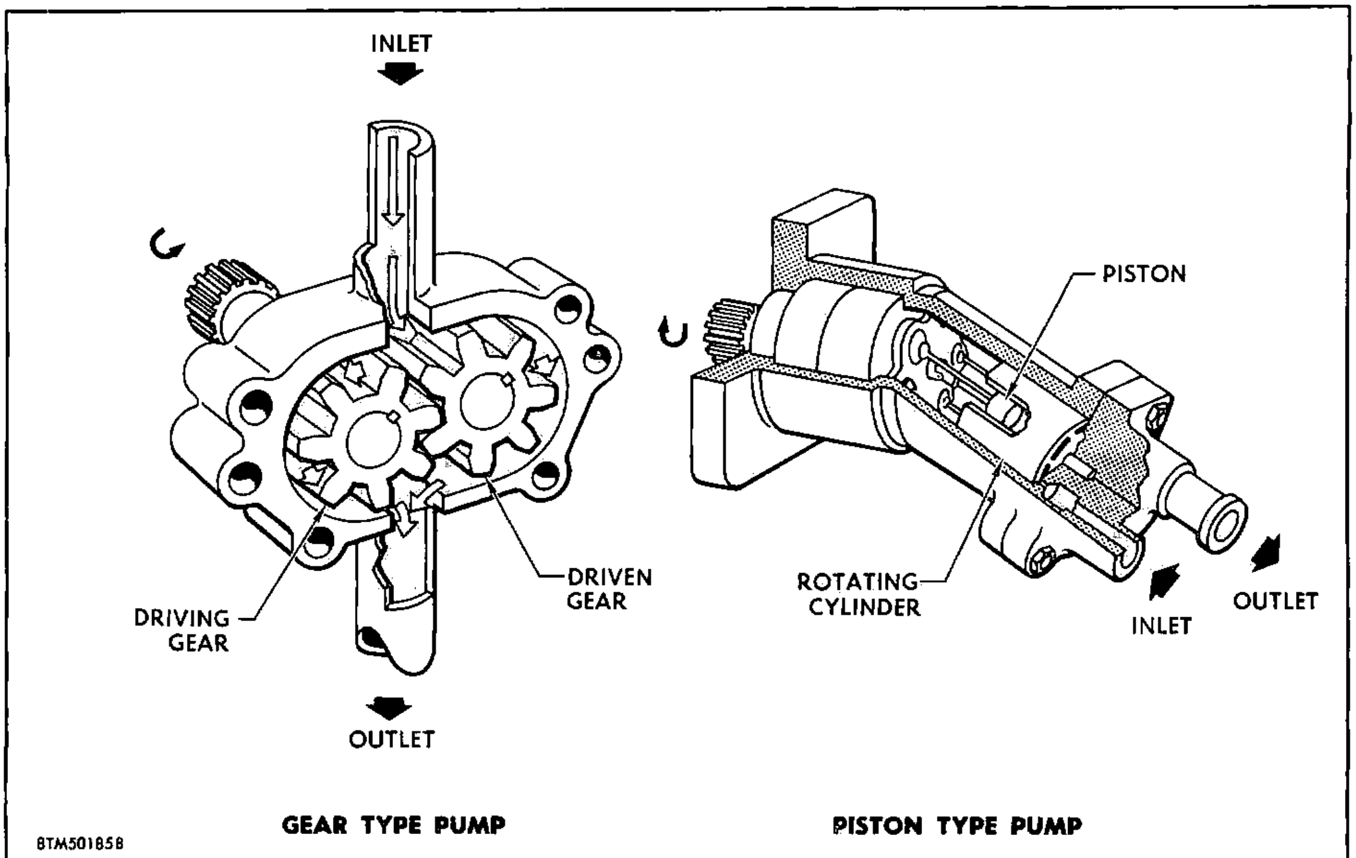

Figure 1-12. Types of Hydraulic Pumps

the fluid from vortexing and surging. Vortexing is the circular motion set up in a fluid which causes a cavity or partial vacuum in the center much the same as that created in a whirlpool.

The two lines at the bottom of the reservoir supply the engine or main system pump and the hand or emergency pump. Notice that the opening of the standpipe which feeds the engine pump is located above the bottom of the reservoir. By having the standpipe in this location, the emergency system is always assured a supply of fluid should the main system develop a leak and lose its supply of fluid.

In many hydraulic systems, the reservoir is not located in such a position as to permit the gravity forced flow of fluid to the pump. Consequently, the reservoir is pressurized to make up for the lack of gravity pressure. In this type of a reservoir, controlled air pressure would enter at the vent opening. This pressurization produces fluid pressure in the reservoir and keeps a positive pressure on the fluid line to the pump.

Pumps.

A hydraulic pump is the mechanism that produces and supplies hydraulic systems with fluid pressure, and is the key unit in any hydraulic system. The two most common types of modern hydraulic pumps are the piston-type and gear-type pumps. Although there are many variations of these pumps, each type functions upon the same general principles of operation. All hydraulic pumps are either driven by an electric motor or are mechanically driven by the airplane engine.

Of the two types of pumps, the piston type is the most satisfactory for high-pressure systems. A piston pump, like the one shown in figure 1-12, consists of a housing, a cylinder block, pistons and connecting rods, a drive shaft, and a universal drive linkage. Despite the complexity of this piece of equipment, it produces a positive fluid displacement that is smooth and free of high pressure surges.

The drive shaft of the piston pump is turned either by the airplane engine or by an independent motor. The turning of this pump shaft rotates the cylinder block which is attached to the shaft at about a 15 or 20 degree angle. This angle plays an important part in the operation of the pump. As the cylinder block and pistons rotate at an angle, the pistons move in and out of the cylinder.

When a piston is at the top of the pump it is at the extreme end of its withdrawal stroke. Then, as the cylinder and piston rotate toward the bottom of the

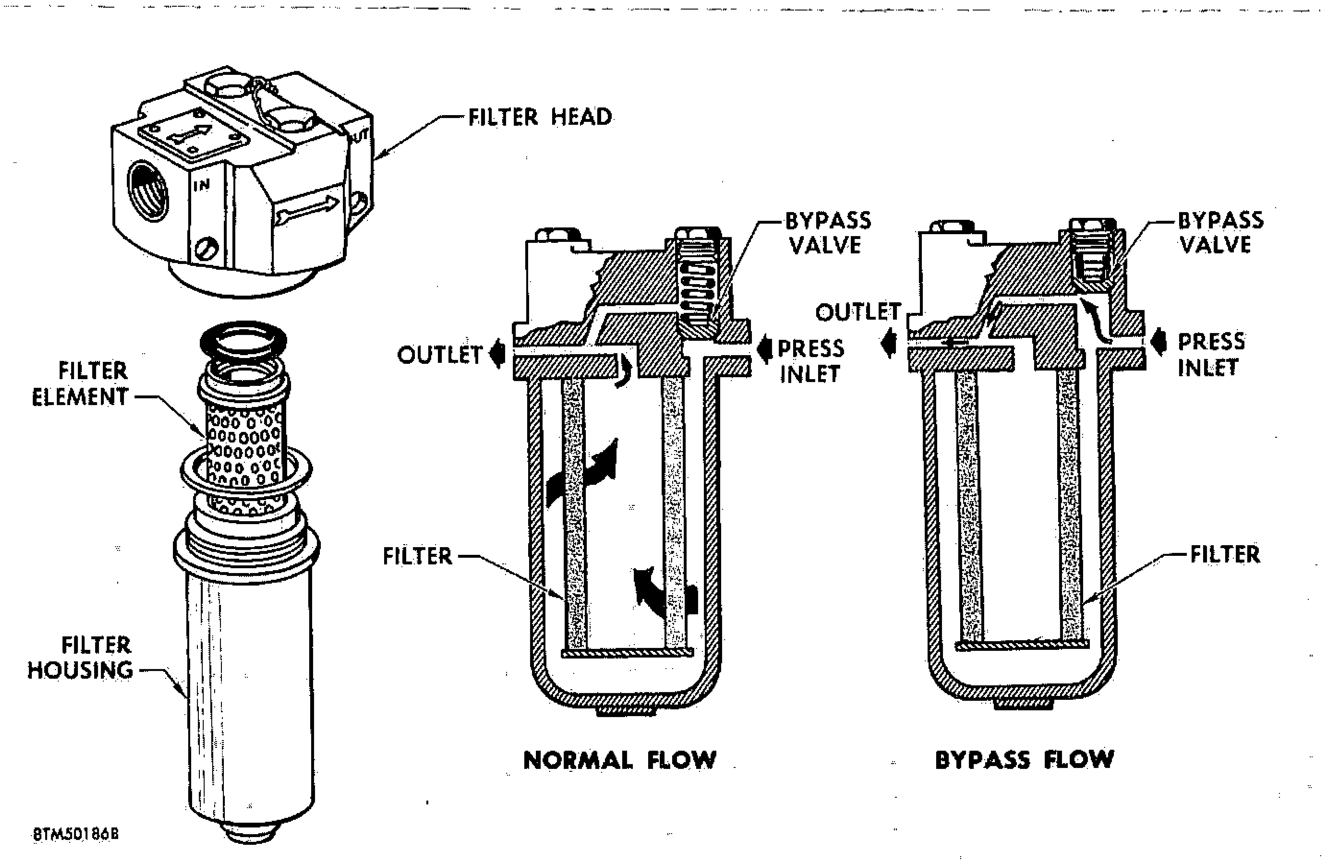

Figure 1-13. Hydraulic Filter

pump, the piston moves into the cylinder block. In one complete rotation of the cylinder, the piston has made a complete stroke. Each stroke of each piston produces the fluid pressure that the pump puts out.

Gear pumps are much simpler in operation. They produce hydraulic pressure by the meshing of two gears that revolve inside a housing. Clearance between the two gears traps fluid and as the gears turn, they force the fluid from the pump outlet port, thus producing hydraulic pressure.

Filters.

Downstream from the pump where the system is under pressure, we need assurance that any foreign matter that may be in the fluid will not travel into the delicate mechanisms of the system. This assurance of clean fluid is obtained by adding a filter, similar to the one shown in figure 1-13. This filter removes and collects foreign matter of even the smallest size. As you can see in this illustration, fluid enters the inlet port, is forced through the filtering element, then leaves the filter through the outlet port. The element in hydraulic filters may either be the parchment type or the wire wound type.

Parchment filtering elements are porous enough to permit fluid flow, but are impenetrable to solid matter. Wire wound elements consist of very fine wire wound in a shape similar to the parchment filter shown in figure 1-13. The space betweeen these windings is so small that foreign matter is trapped on the outside of the filter element. The bottom section of the filter assembly screws into the top or head section so that the filtering element can be removed easily for periodic cleaning.

Between the inlet and outlet ports of all hydraulic high pressure filters is a bypass relief valve. This relief valve permits the bypassing of fluid around the filter whenever the filter becomes clogged with foreign matter. When the hydraulic pressure in the filter exceeds the normal system operating pressure due to a clogged filter and builds up to the pressure setting of the relief valve, the relief valve automatically opens and bypasses the hydraulic fluid around the filtering element to the outlet port.

PRESSURE CONTROL GROUP.

The pressure control group in a hydraulic system maintains the system at the proper pressure, directs fluid flow, gages the operating hydraulic conditions,

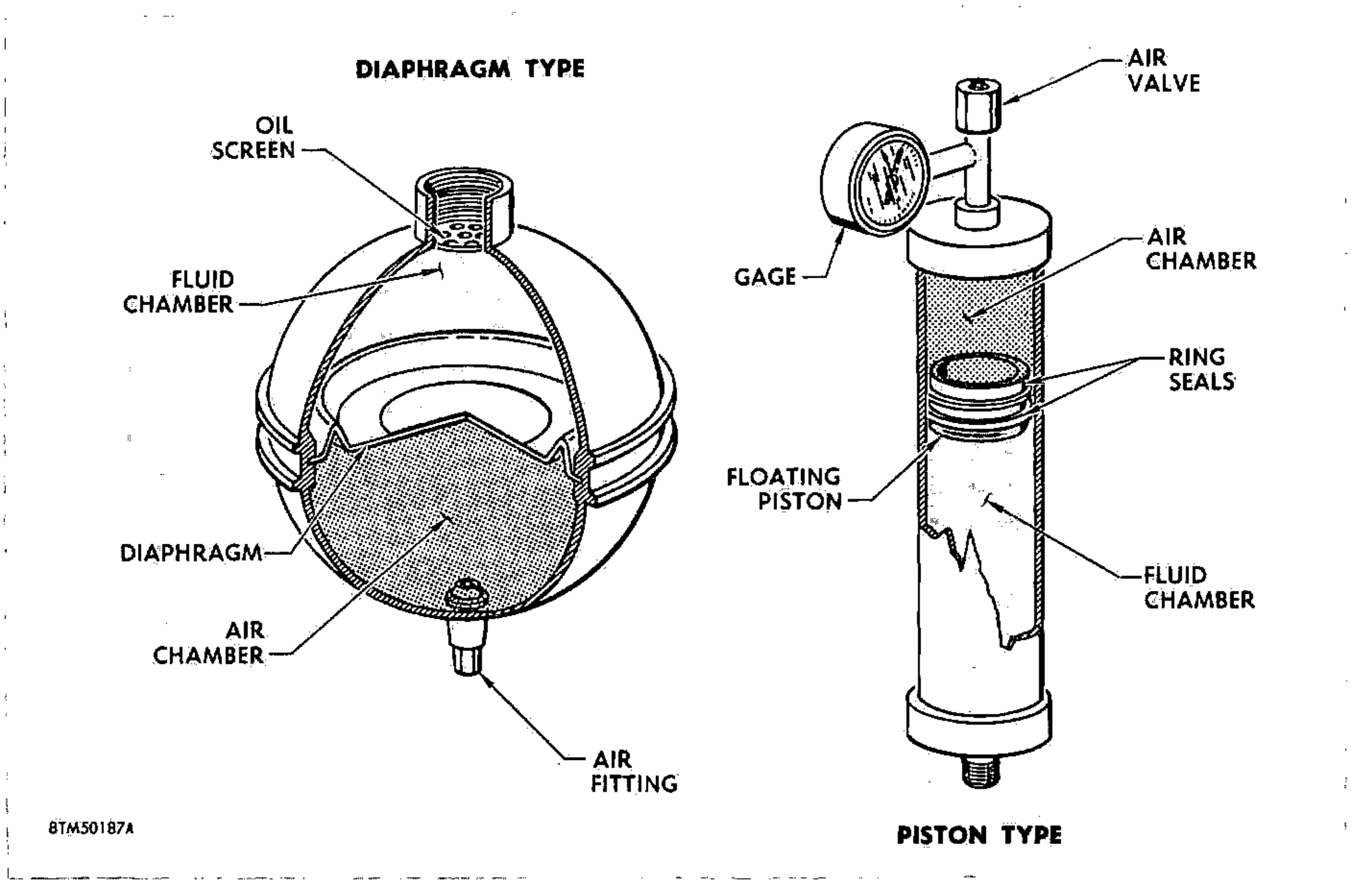

Figure 1-14. Types of Accumulators

and actuates the warning system. The devices that make up this group are regulators, transmitters, switches, snubbers, accumulators, and many types of valves that have different types of control functions.

Accumulators.

An accumulator does just what its name implies—it accumulates and stores up pressurized hydraulic fluid. However, in the process of storing this pressure, this device develops large amounts of potential (stored-up) energy. Basically, an accumulator, of which two common types exist, is a device that cushions and absorbs pressure surges in a hydraulic system.

In figure 1-14 you can see the two common types of accumulators used in hydraulic systems. The accumulator on the left is the diaphragm type which consists of a flexible diaphragm that divides the accumulator into two separate chambers, the air chamber and the oil chamber. The lower chamber is equipped with an air valve so that it can be charged with compressed air. This places the hydraulic fluid in the upper or oil chamber under pressure. Jerky or uneven fluid pressure is absorbed by this air-cushioned diaphragm so that the pressure in the system is smooth and free of surges.

The accumulator shown on the right of figure 1-14 is the piston type, and its basic operation is identical to that of the diaphragm type. In this type, the floating piston moves up and down in the cylinder, providing the same effect as the flexible diaphragm when pressures surges are encountered. The piston is sealed against leaks so that the fluid and air are always separated. The gage on the air pressure end of the cylinder measures the operating pressure in the system, or the air charge in the accumulator when the hydraulic system is not pressurized.

During system operation, the initial charge of compressed air put into the accumulator forces the flexible diaphragm upward or the piston down. When the system is pressurized, the pressure of the oil in the fluid side of the accumulator builds up higher than the air pressure and moves the diaphragm down or the piston up. This action compresses the air even more and adds more potential energy to the diaphragm or piston. During periods of peak operation, the highly compressed air will tend to force fluid back into the system. If the power pump is not running, the compressed air will supply a limited amount of fluid under pressure for the operation of a mechanism.

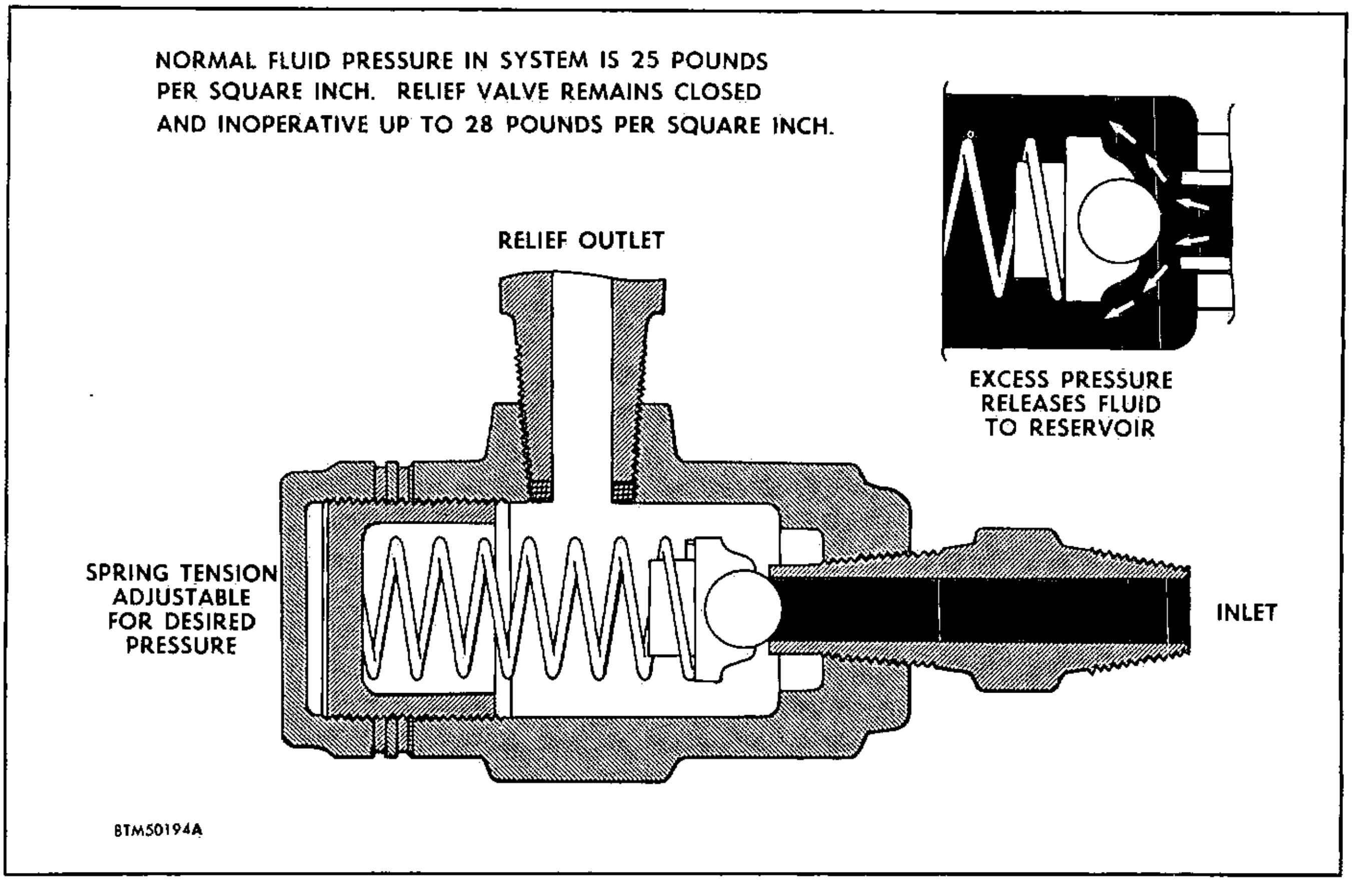

Figure 1-15. Relief Valve

Regulators.

Regulators are devices that control the amount of pressure built up in hydraulic systems. They unload pressure and relieve the hydraulic pump when the desired pressure is reached. There are many types of regulators ranging from a simple pressure relief valve to the more accurate and complex, automatic "cut-in cut-out" type regulator.

Relief Valves.

Relief valves consist of a two or four-port housing, a ball- or cone-shaped seat that is held in place by a strong spring, and an adjusting screw. Note in the illustration of a relief valve (figure 1-15) that the spring forces the ball valve flush against the inlet line of the valve and thus causes the fluid to bypass the valve. However, when the fluid pressure exceeds the pressure setting of the valve, it overcomes the spring tension and pushes the valve seat back. This allows excess pressurized fluid to leave through the relief outlet of the valve and return to the reservoir. From the description of the relief valve, you can see why relief valves are sometimes called "safety valves."

Restrictors.

Restrictors are devices that reduce the rate of flow of hydraulic fluid in one or more directions. It is possible for a high-pressure system to actuate light-weight, slow-moving machinery by placing a restrictor in the fluid line, upstream from that unit's actuating cylinder. Restrictors usually have small orifices

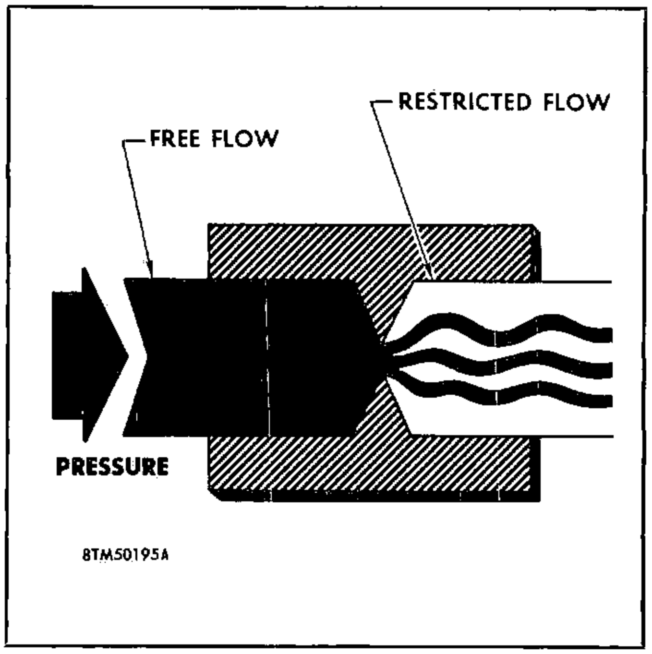

Figure 1-16. Restrictor Valve

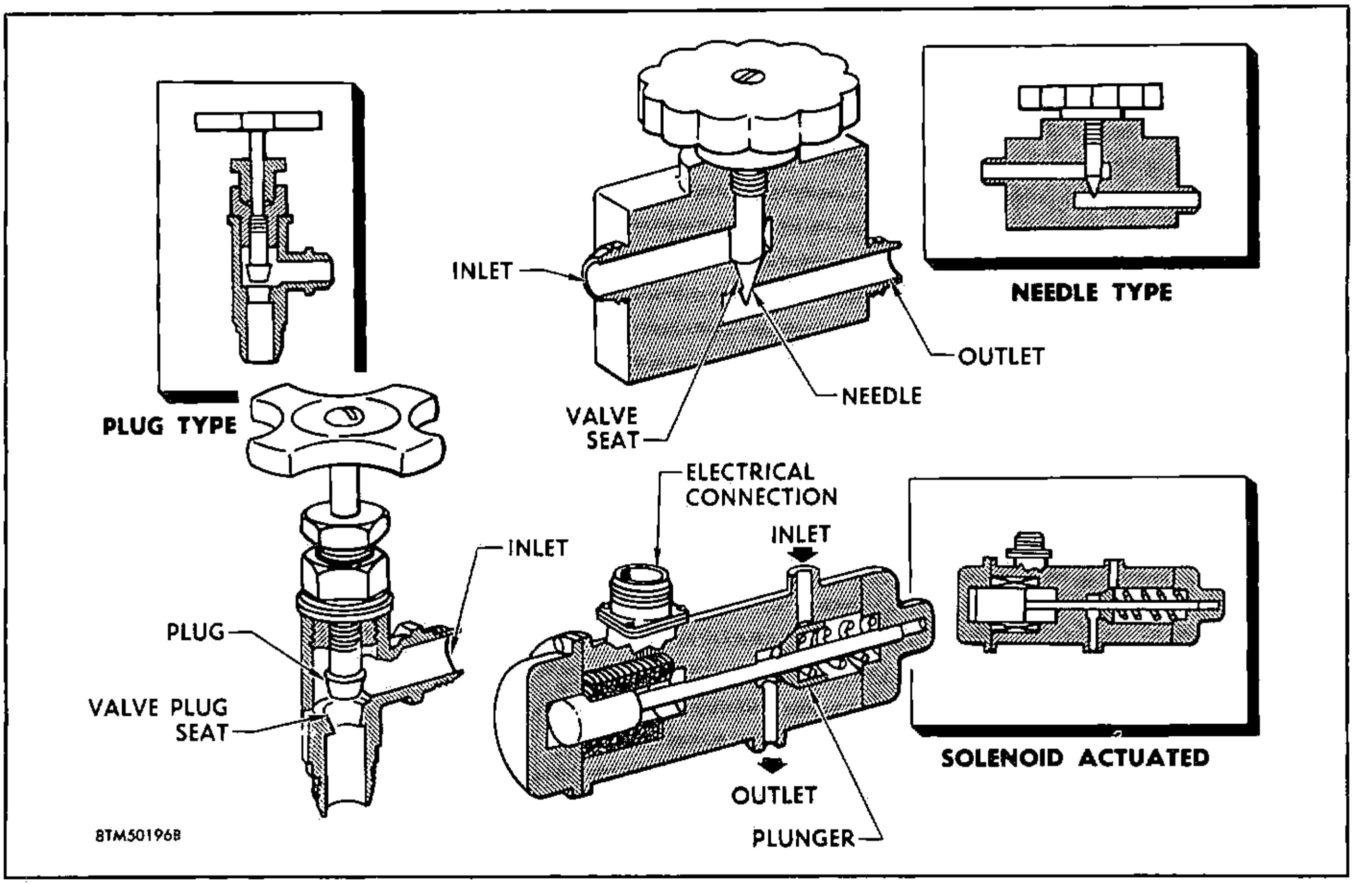

Figure 1-17. Shutoff Valve

or adjustable needles with a small clearance between a valve housing and the needle. The restrictor shown in figure 1-16 is of the non-adjustable type. Note that the opening at either end is larger than the restriction at the center. This small opening, or orifice, restricts fluid flow and thereby reduces pressure. From this description of a restrictor, you can understand that a restrictor is effective only where fluid is in motion.

Shutoff Valves.

Shutoff valves are in effect hydraulic switches. This type of valve opens to allow pressurized fluid to flow or closes to prevent fluid from flowing to a mechanism. Figure 1-17 shows the forms of some of the various shutoff valves. Note, however, that the basic operation of all of the valves is the same even though the method of actuation differs.

Most shut-off valves are operated by electrical power. The electrical power energizes a solenoid which in turn relays the power to open the fluid port of the shutoff valve. Basically, a solenoid consists of a winding around an iron core which magnetizes when energized by electricity. The resultant magnetic force of the solenoid opens the port of the shutoff valve; deenergizing the solenoid and spring tension closes it. Shutoff valves are usually placed between the main part of the hydraulic system and an emergency or auxiliary unit.

Check Valves.

A check valve in a hydraulic system allows the free flow of fluid in *one* direction, but prohibits any flow in the opposite direction. Check valves are used to trap pressure in one part of a hydraulic system. There are several variations of the check valve, but the most commonly used types are the *orifice check valve* and the *bypass check valve.*

The device in a check valve that does the fluid checking is either a ball, a cone, or a flapper plate that is held on its seat by the force of a light spring. Fluid is allowed to flow freely in one direction through this valve, but a reverse flow causes the checking device to seat. Notice in figure 1-18 how this principle works. In the normal flow view the ball is unseated, allowing the fluid to flow. In the upper view, back flow is checked by the action of the spring in seating the ball check.

PRESSURE ACTUATION GROUP.

The pressure actuation group in a hydraulic system receives the force from the pressurized fluid and

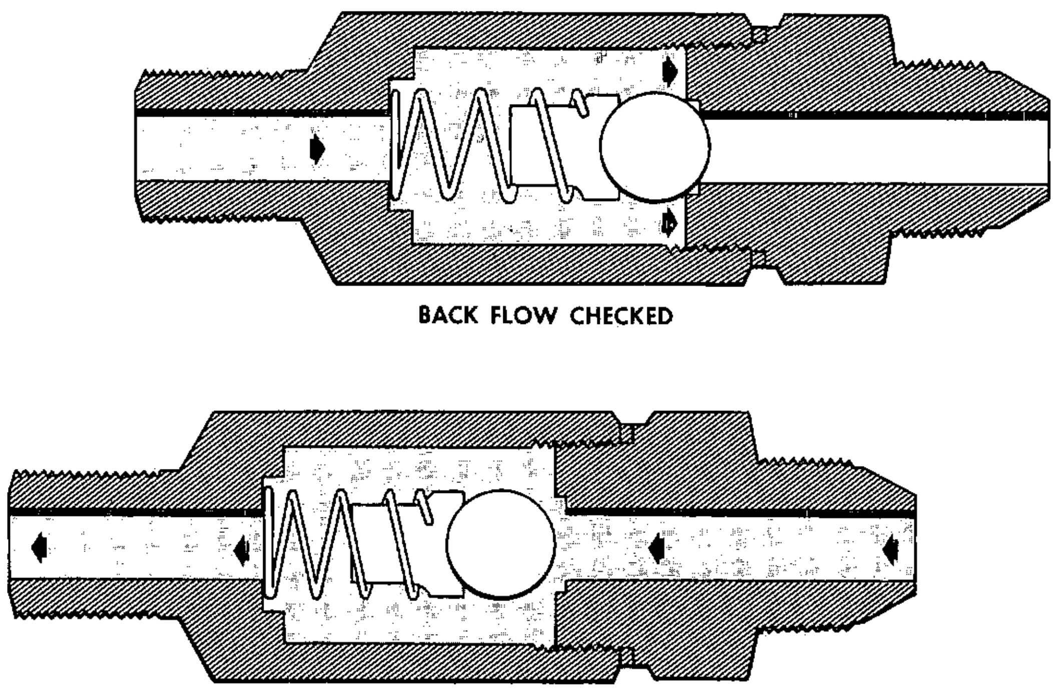

Figure 1-18. Check Valve

transforms it into work output. It is in this subdivision of hydraulic systems that the control valves, the actuating cylinders, and the servo (control booster and regulator) equipment are used.

Control Valves.

The purpose of control valves is to control the direction of fluid and to regulate the amount of fluid directed to an actuating mechanism. A control valve is similar to the selector valve mentioned earlier in the open-center system. Although selector valves may be placed in this classification, it must be understood that all control valves are not selector valves. A selector valve is one that is engaged at the will of the pilot for the purpose of directing fluid to the desired unit. A control valve is usually automatically controlled, and may be only partially opened or closed to allow a regulated amount of fluid to a mechanism.

Figure 1-19 shows a simple four-port, rotor-type selector valve in its three operating positions. The handle on this type valve is usually controlled by actuating linkage in the operating system. In the DOWN position, fluid pressure is routed from the pump through the valve to the down side of the actuating unit, while return pressure is routed through the other side of the valve back to the reservoir. When the actuating unit reaches the desired position, feedback mechanical linkage from the actuating unit returns the valve to the NEUTRAL position shown in the illustration. This shuts off fluid pressure to the actuating unit and stops the unit at the desired position. The center view shows the position of the valve when fluid pressure is directed to move the actuating unit in the UP position.

Many other types of valves may be used as control valves: such as poppet valves, spool valves, and piston valves; but all control valves work on the same principle of controlling direction and amount of fluid pressure.

Actuating Cylinders.

Hydraulic actuating cylinders (commonly called hydraulic jacks) function in a hydraulic system to transform fluid pressure into mechanical movement. These cylinders are used in all hydraulic applications that require linear (fore and aft) or angular movement.

Actuating cylinders consist basically of a cylinder case with inlet and outlet ports, a piston and piston rod, and oil seals. In figure 1-20, note how fluid pressure performs work in a simple actuating cylinder. Fluid pressure enters the actuating cylinder at the left port and forces the piston and its rod to the right.

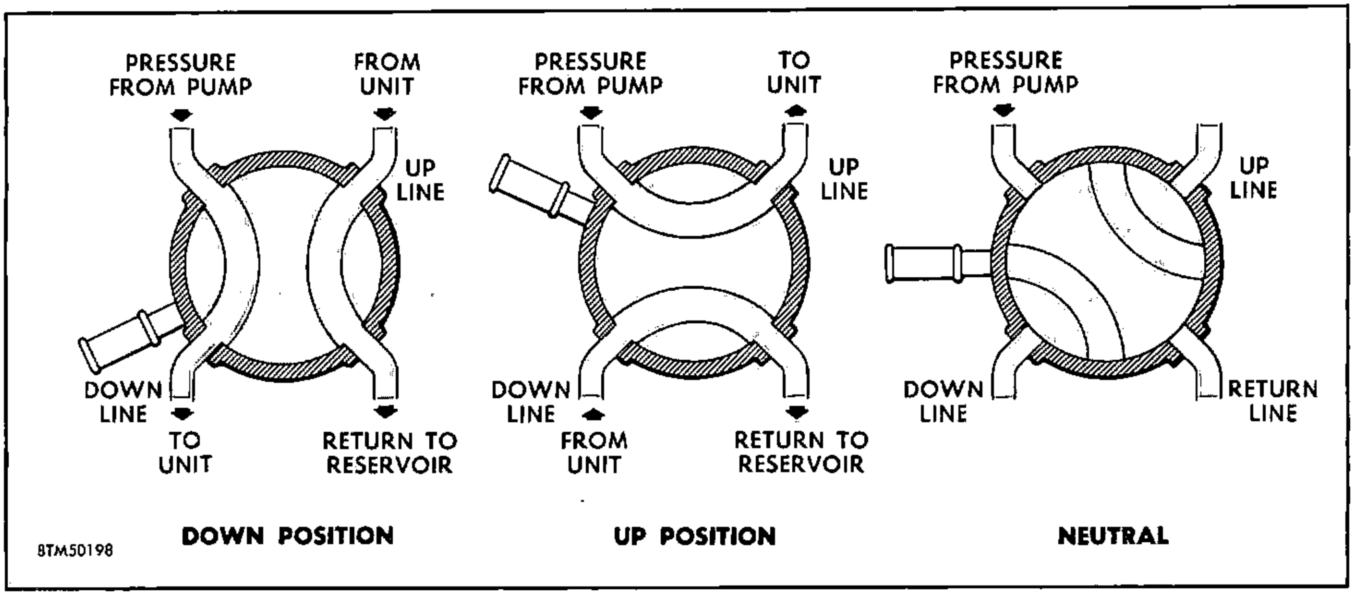

Figure 1-19. Control Valve

As this piston moves, it displaces the fluid behind the piston out the right port. A reverse operation will take place if the fluid is made to enter through the right port to actuate the piston assembly toward the left, and thereby force return fluid out the left port.

The O-ring oil seal on the piston prevents internal leakage of fluid from one side of the piston to the other. When these seals are defective, they allow fluid to leak past the piston and cause sluggish operation of the actuating cylinder. These actuating pistons may vary in construction so as to provide a mechanical advantage in an operating system. One variation of a piston assembly may have two piston rods—one on each side of the piston—and another might have two pistons operating in tandem (one after the other) in one cylinder.

The force of hydraulic fluid moves the actuating piston and since the piston has a direct linkage with the operating mechanism through its piston rod, all piston movements result in work done to the mechanism to which it is connected—the transfer of *fluid pressure into mechanical motion.*

Servo Equipment.

In this day of electronics and automatic controls, you have heard or seen the word *servo.* When we refer to servo equipment, we mean *booster* devices that operate in conjunction with a system to improve power output, time response, and control regulation. The popular modern room temperature control unit and the electrical refrigeration unit in your home operates properly through the use of servo devices. A servo unit is used to start and to boost some operating mechanism when a change or abnormal condition takes place.

In many cases we have control handles and pedals connected directly to servo booster units. When you move one of these controls, you start a booster device which supplies additional power and more accurate movements to improve the overall system operation. For example, consider a vehicle traveling at 50 miles per hour on a smooth level highway. The driver holds the accelerator down at a position that supplies the proper amount of gas to the carburetor to maintain this speed. However, the vehicle slows down when it goes up hills despite the fact that the accelerator remains in the same position.

This drop in speed is called the *error* in the power output of the vehicle's engine. Servos are also error correcting devices. As the vehicle began to lose speed near the bottom of the hill, a servo unit would have sensed this error and corrected for it by adding additional power to keep the speed at 50 mph. The opposite holds true for descending the hill, where the servo unit would have supplied power to brake the speed of the vehicle down to 50 mph.

Regulation by means of servos is used for voltage regulation, temperature control, and many other types including hydraulic fluid pressure regulation. Some servo mechanisms are very complex, but the example described above is the fundamental nature of all servo equipment. Later on in this supplement, you will learn about the servo devices in the F-102A which work in conjunction with the hydraulic system to improve the efficiency of the system.

HYDRAULIC PLUMBING.

The plumbing in the hydraulic system consists of the tubing which connects the various units, the fittings

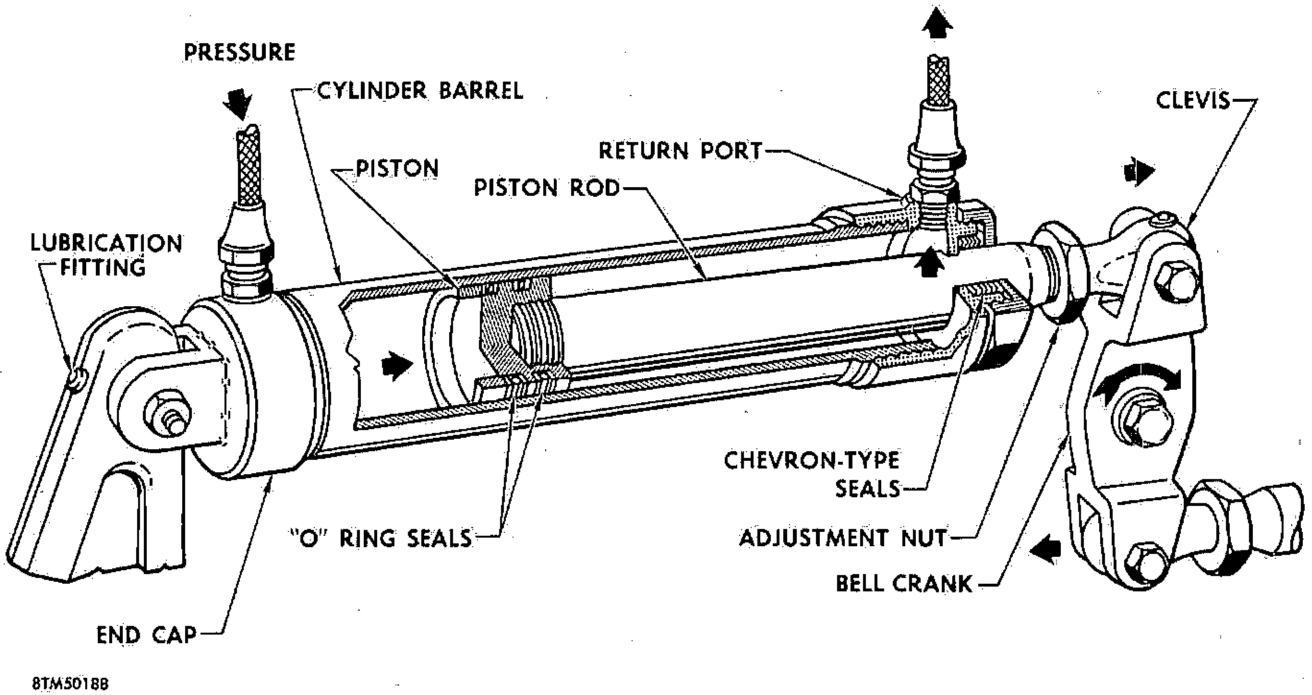

Figure 1-20. Actuating Cylinder

which connect tubing to these units, and seals throughout the system which insure a leak proof system.

Tubing.

Metal tubing, or lines, have the job of carrying fluid under pressure to the various devices and units in the hydraulic system. The volume and speed of flow required to do the work also determines how large any tube must be. Although many other types of metal are in general use, strong aluminum alloy metal makes excellent tubing for a light-weight, high pressure system.

If tubing bends are made properly so as to not start turbulent flow, they will not affect the system. However, unless it is unavoidable, never try to change the shape of a tube assembly—it usually starts troubles.

Flexible Hose.

Flexible hose for hydraulic purposes usually consists of steel or fabric braid covered by a synthetic rubber tube, with connection fittings swaged on to both ends. The fabric-reinforced type of hose is suitable only for moderate hydraulic pressures while the steel-reinforced hose can be used for all high pressures.

Fittings.

Fittings are used to connect the tubing of a hydraulic system to the various system units. Practically all fittings used in aircraft today are like the two AN types shown in figure 1-21. These fittings are designed to mate with the flared ends of the system tubing, and are made of aluminum alloy or steel. These fittings are colored so you can easily identify the type of metal—aluminum alloy fittings are blue, steel fittings are black.

The triple type connector shown in figure 1-21 is a standard AN variety that consists of three pieces; the fitting, the nut, and the sleeve. The sleeve on the connector fits directly over the tubing—one end of the sleeve flared at the same angle as the tubing flare. The nut fits over the sleeve and, when tightened, draws the sleeve and tubing flare tightly against the fittings to form a fluid-tight seal. Note that the fitting is beveled at the same angle as the tubing flare. The sleeve on the connector supports the tubing so that vibration does not concentrate at the edge of the flare and it distributes the shearing action over a wider area for strength.

The standard connection in the lower portion of the illustration is another AN type fitting consisting of a

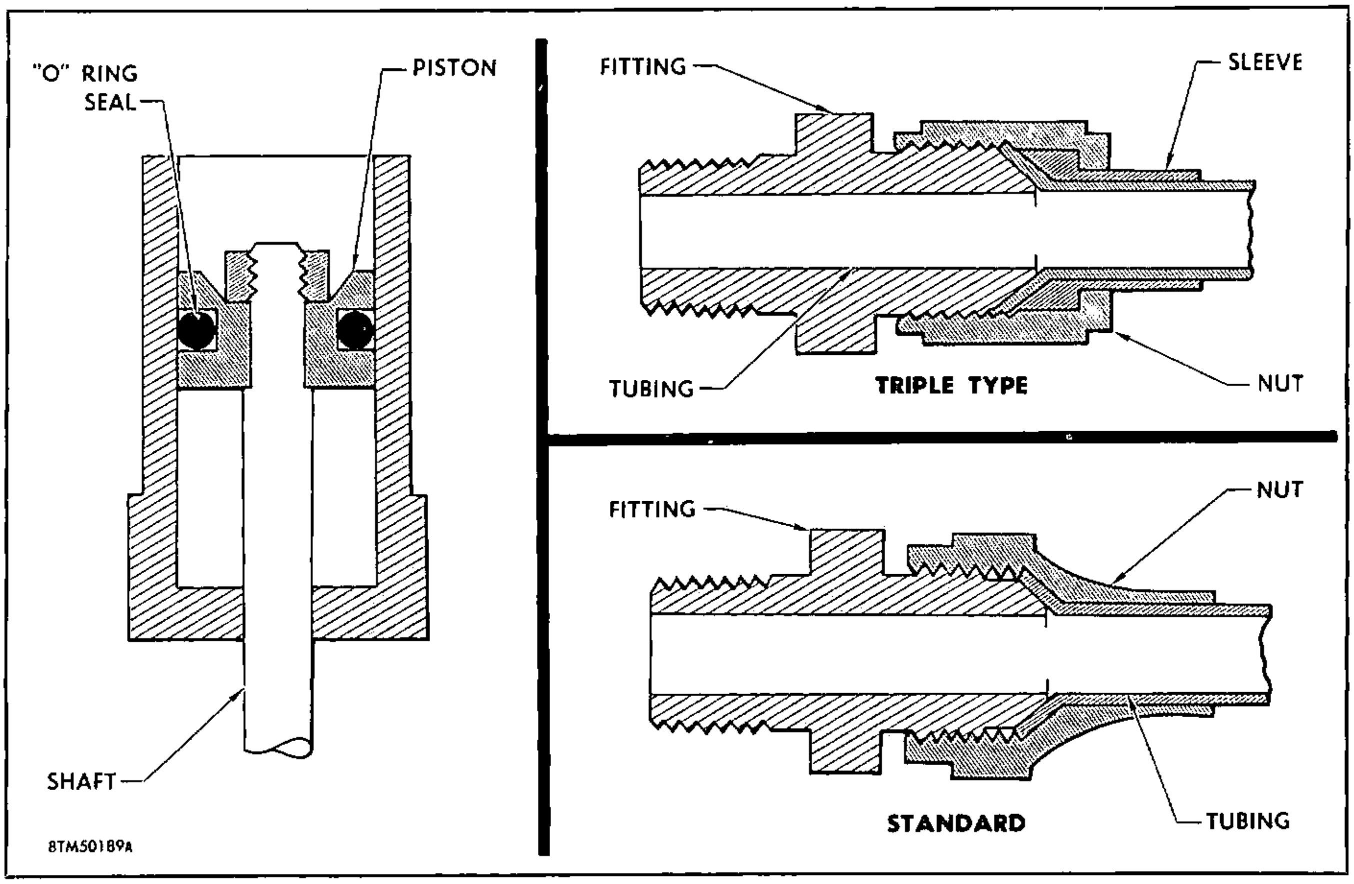

Figure 1-21. Seals and Fittings

nut and fitting. Note that this connector has no sleeve. The inside of the nut fits against the outside of the tubing flare and clamps it against the fitting. This nut is designed to support the tubing in the same manner as the sleeve described in the triple type connector.

When installing fittings, always start the nut on the male fitting by hand. This will prevent cross-threading of the fittings. If the threads bind, examine the fittings to make sure that the threads are clean and contain no imperfect threads. Always be certain that fittings are aligned when you connect them. Never force a binding fitting—additional turning will strip its threads.

Seals.

In maintaining pressure in a hydraulic system, seals are as vital to hydraulic fluid as hydraulic fluid is to the seals. Neither is complete without the other. Oil seals, gaskets, and packing rings are made of relatively soft materials. These materials are placed between the meeting surfaces of hydraulic fittings to insure a fluid-tight connection. Seals must be made of a type of material that will not affect or be affected by the hydraulic fluid. Another feature of a good seal is its ability to withstand extreme temperatures and pressures without flaking or disintegrating. Neoprene, rubber, cork, and teflon are the most widely used materials for packings in high pressure hydraulic systems.

Figure 1-21 shows an oil seal in a cylinder-piston assembly. The seal in this instance is the O-ring type oil seal that is used to seal the piston against leakage between the two sides of the piston. Seals of this type are always tightly fitted so as to give a maximum seal during movement as well as when the piston is not in motion.

When replacing any of the seals, packings, or gaskets in any hydraulic system, you should always be sure that the replacement seals are the same type as those which you remove. All O-ring seals, gaskets, and other types of seals are identified as to the type of system in which they should be used, and the manufacturer. The coding is accomplished by a series of two or three dots arranged in clockwise order. The first dot indicates the system and the second and third dots indicate the manufacturer. A blue dot indicates that seals are to be used on hydraulic systems. If you are not already familiar with the coding markings, you should refer to ANA Bulletin No. 419 or other controlling Air Force Specifications.

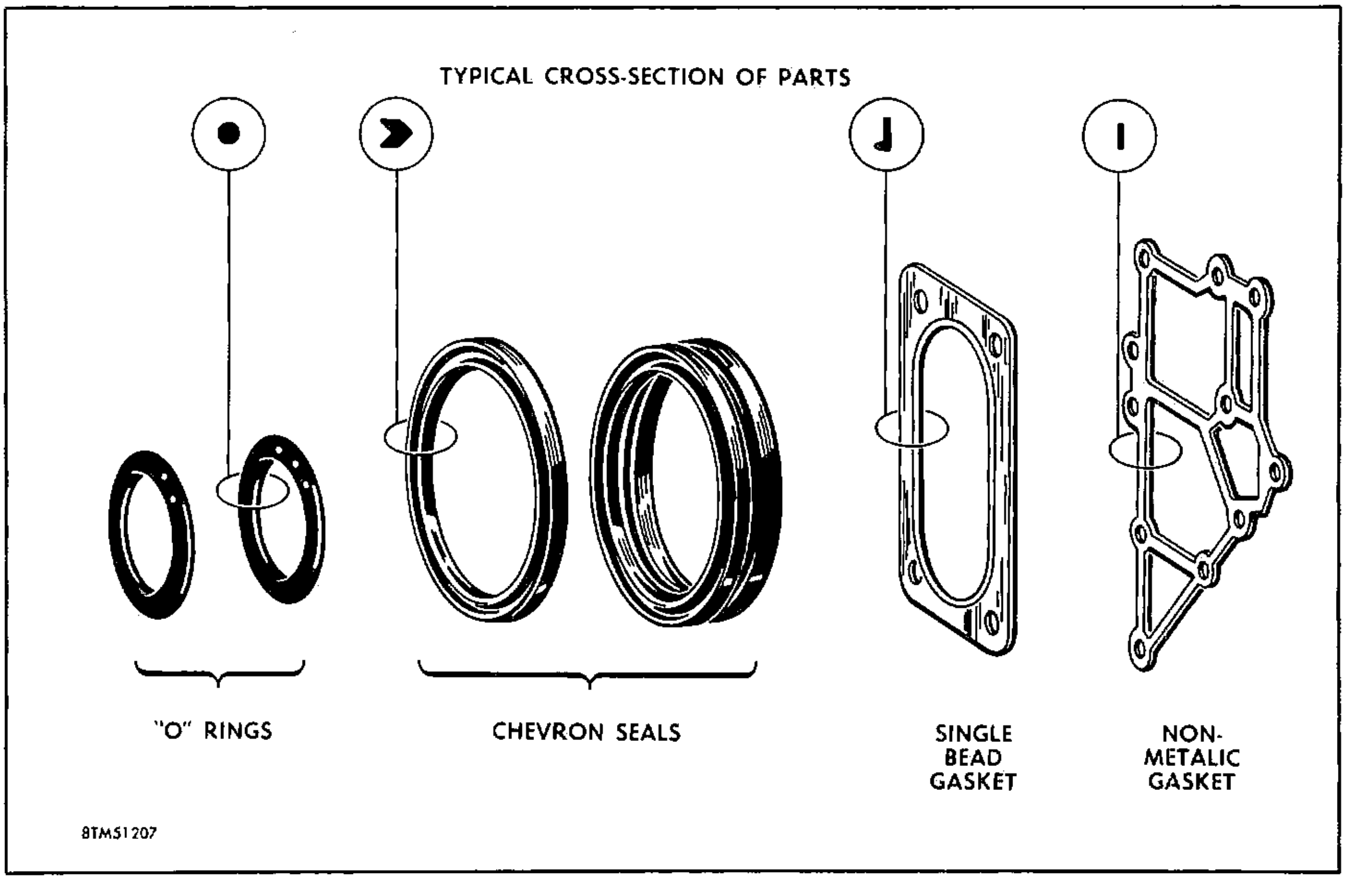

Figure 1-22. Hydraulic System Seals

HYDRAULIC FLUIDS.

There are many different types of fluids which have been used successfully in hydraulic installations. One of the most common types used in the very early hydraulic systems was just plain water. But as hydraulic systems have become more complex and are being designed to perform more critical operations, special fluids have had to be developed to insure that the systems function properly under many different operating conditions.

Hydraulic fluid, as generally known in the aircraft industry today, is a light oil that has good lubricating qualities, relatively high viscosity, chemical inertness, and a low setting point. Hydraulic fluid with good lubricating qualities reduces the wear factor in systems which utilize high speed pumps and other moving components. High viscosity reduces system leakage problems while the inertness factor avoids corrosion and other chemical attack problems. The low setting point is required, of course, to eliminate freezing problems when the aircraft is operating in cold climates or at exceptionally high altitudes.

Some of the types of hydraulic fluids used by the Air Force that meet the requirements are Specification MIL-O-5606, MIL-H-7644, MIL-O-6083, and MIL-L-6387. These different types of hydraulic fluids are *not* interchangeable, however, except when specified in the aircraft or system technical orders. Some of the fluids are mineral oil base hydraulic fluid while others are vegetable oil base and still others are synthetic oil base fluids. In most cases, hydraulic fluids are dyed for identification purposes (red for mineral base and blue for vegetable base) but because more than one type of fluid may be dyed the same color, the only sure method for determining the type of fluid is by checking the container in which the fluid is shipped from the manufacturer.

The F-102A hydraulic system is designed to use a mineral oil base hydraulic fluid, Specification MIL-O-5606. This fluid is dyed red and has an operating temperature of —65°F to +275°F. Generally speaking, this fluid is intended for use in aircraft hydraulic brake systems, shock absorbers, and in flight control surface mechanisms.

As a precautionary note, keep in mind that MIL-O-5606 hydraulic fluid is not interchangeable with vegetable base hydraulic fluids (blue colored). In the event these fluids are mixed inadvertently while servicing the F-102A hydraulic system, the affected system or component should be flushed immediately. Pertinent

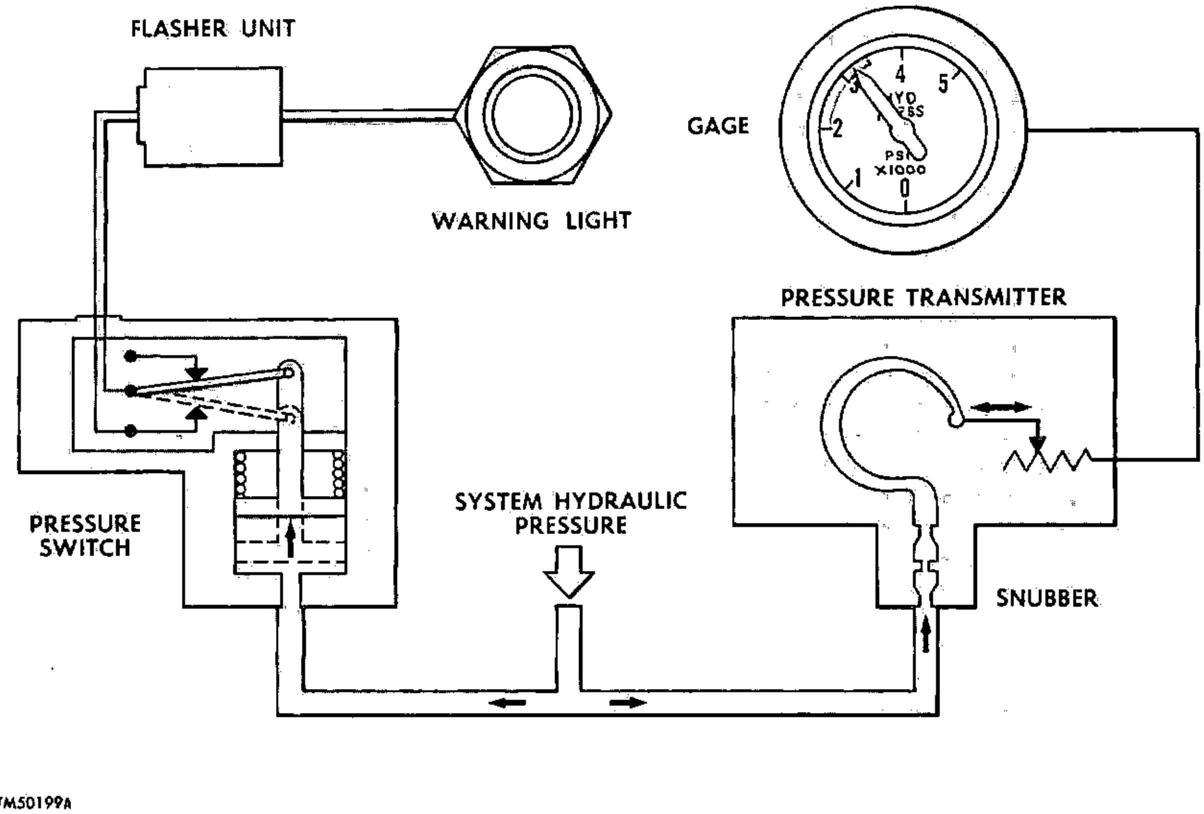

Figure 1-23. A Simple Indicating System

flushing instructions should be obtained from the maintenance manual or the controlling military specications.

PRESSURE INDICATING AND WARNING SYSTEMS.

Aircraft hydraulic systems require a pressure indicating system and a pressure warning system so that the pilot will always know just how the system is doing. The pressure indicating system tells him at a glance just how much operating pressure he has, while the warning system effectively notifies him whenever the operating pressure falls to a dangerous level. The simplest type of indicating system is one which gives a direct reading. However, almost all hydraulic systems today use a remote indicating system since the hydraulic system is too far from the cockpit. Remote indicating systems are also used to eliminating routing inflammable fluids into the cockpit area.

HOW THE HYDRAULIC PRESSURE INDICATING SYSTEM OPERATES.

The simplified schematic diagram of a pressure indicating system shown on the right of figure 1-23 is typical of all indicating systems. This system transforms hydraulic pressure into electrical signals which are then registered on the gage as hydraulic pressure.

In this indicating system, hydraulic pressure passes through a snubber and into the curved tube in the pressure transmitter. This curved tube, which is rigidly attached at the inlet end and is free to move at the other end, works on the *Bourdon tube* principle.

As hydraulic pressure enters the curved tube, it tends to straighten out. This is caused by the surface area on the inside of the tube being less than the surface area on the outside; therefore, more pressure is applied to one side than the other. The tube will straighten out until the difference in force is balanced by the elastic resistance of the material composing the tube. The free end of the tube is connected to a variable resistor, and as the tube is moved by changes in hydraulic pressure, it produces a deviation on the variable resistor. Such a variation in the resistance of an electrical circuit creates a proportional change in the amount of current that flows through this resistor. Therefore, the current in the circuit is an accurate representation of the hydraulic pressure.

The pressure gage, shown in figure 1-23, operates by remote-control electrical signals. The transmitter sends these signals to an a-c synchro device in the gage. This a-c synchro device has both a *rotor* and a *stator,* each of which is connected in parallel with similar units in the transmitter. The rotor can turn inside the stator to offset any induced electric current that develops when the coils receive power. The intensity of this induced current is relative to the transmitted electrical signal that is sent from the Bourdon tube. The pointer that you see on the face of the gage is actually attached to the rotor. The rotating motion of the pointer is really the same as the movement which the synchro rotor makes in response to the induced electric current.

The irregularity of pressure in the system may sometimes cause the Bourdon tube to vibrate and cause the needle on the gage to oscillate. Therefore, the pressure snubber mentioned above is installed at the pressure transmitter. This snubber has a restricted opening which dampens out pressure surges in the hydraulic system and eliminates any vibration of the Bourdon tube and oscillation of the gage needle.

HOW THE WARNING SYSTEM OPERATES.

The warning system, shown on the left side of figure 1-23, operates continuously whenever the hydraulic system is pressurized and electrical power is on the airplane. It warns the pilot whenever a dangerously low pressure condition exists in the hydraulic system. In the warning system shown in this schematic diagram, the warning light remains off during normal pressure conditions and flashes on and off when the pressure drops to a critically low-pressure range.

System hydraulic pressure flows to the pressure switch and acts upon the movable piston or actuator inside this switching device. The spring shown behind the pressure switch actuator holds the actuator down until fluid pressure overcomes the spring tension and pushes the actuator up—a low amount of pressure will not hold the actuator up. When hydraulic pressure is low, the actuator rod rests on the lower contact (shown in the dotted line position) and closes the electrical circuit to the warning light. This allows power to flow through the flasher unit to the warning light and flash the light on and off. The pilot, seeing this flashing light, knows that a low-pressure condition exists in the system and immediately takes the proper corrective action.

As pressure in the hydraulic system builds up and overcomes the spring tension, the actuator moves to the *up* position shown in figure 1-23, and opens the circuit in the pressure switch. With the circuit open, the light stops flashing and the pilot knows that the hydraulic system pressure is again within the normal operating range.

THE F-102A HYDRAULIC SYSTEM.

The F-102A hydraulic system is actually composed of three systems—two separate, completely independent systems and an emergency system. Each independent hydraulic system is divided into a power supply system and one or more power distribution subsystems. The subsystems distribute power to move the flight controls, raise and lower the landing gears, open and close the speed brakes, steer the nose wheel, and drive the emergency a-c generator.

One of the independent power supply systems supplies power to the flight control subsystem only and is called the *primary* hydraulic system. The other power supply system furnishes additional power to the flight control subsystem as well as power to all the other hydraulically-operated subsystems in the airplane. This system is known as the *secondary* hydraulic system. Hydraulic pressure for both systems is supplied by two engine-driven pumps. Each pump pressurizes its respective independent system.

These pumps and the other units in the two pressure systems and the emergency system are shown schematically in figures 1-24 and 1-25. All the components of power supply systems, except the engine-driven pumps and the high pressure filters, are located in the hydraulic accessory compartment. This compartment is on the right side of the fuselage just forward of the main landing gear wheel well.

The emergency hydraulic power system consists primarily of the ram-air turbine-driven pump on the inboard side of the hydraulic accessory compartment door. The emergency pump is shown on the primary hydraulic system illustration (figure 1-24). It is driven by the ram air produced by the forward motion of the airplane when the compartment door is opened into the airstream. Air pressure, furnished by the high pressure pneumatic system, opens this door and is controlled from the cockpit. The hydraulic lines of the pump connect to the primary system at the emergency flow control valve so that the pump can furnish emergency power for the flight controls. This emergency pump assures the pilot of hydraulic pressure during flight to operate the rudder and elevons.

Although the landing gear wheel brakes are hydraulically operated and pneumatically actuated, they are not a part of the hydraulic system. A detailed description of the brake hydraulic system is given in another training supplement in this series which covers the general airplane.

HYDRAULIC SYSTEM OPERATION.

Both the primary and secondary hydraulic power systems operate in the same manner. The two main differences between the systems are the tie-in of the emergency system to the primary system, and the subsystems that use the hydraulic power supplied by the systems. To understand how these power systems function, follow the operation of the primary system in schematic (figure 1-24). By referring to the two illustrations, you can see when a difference exists between the two systems.

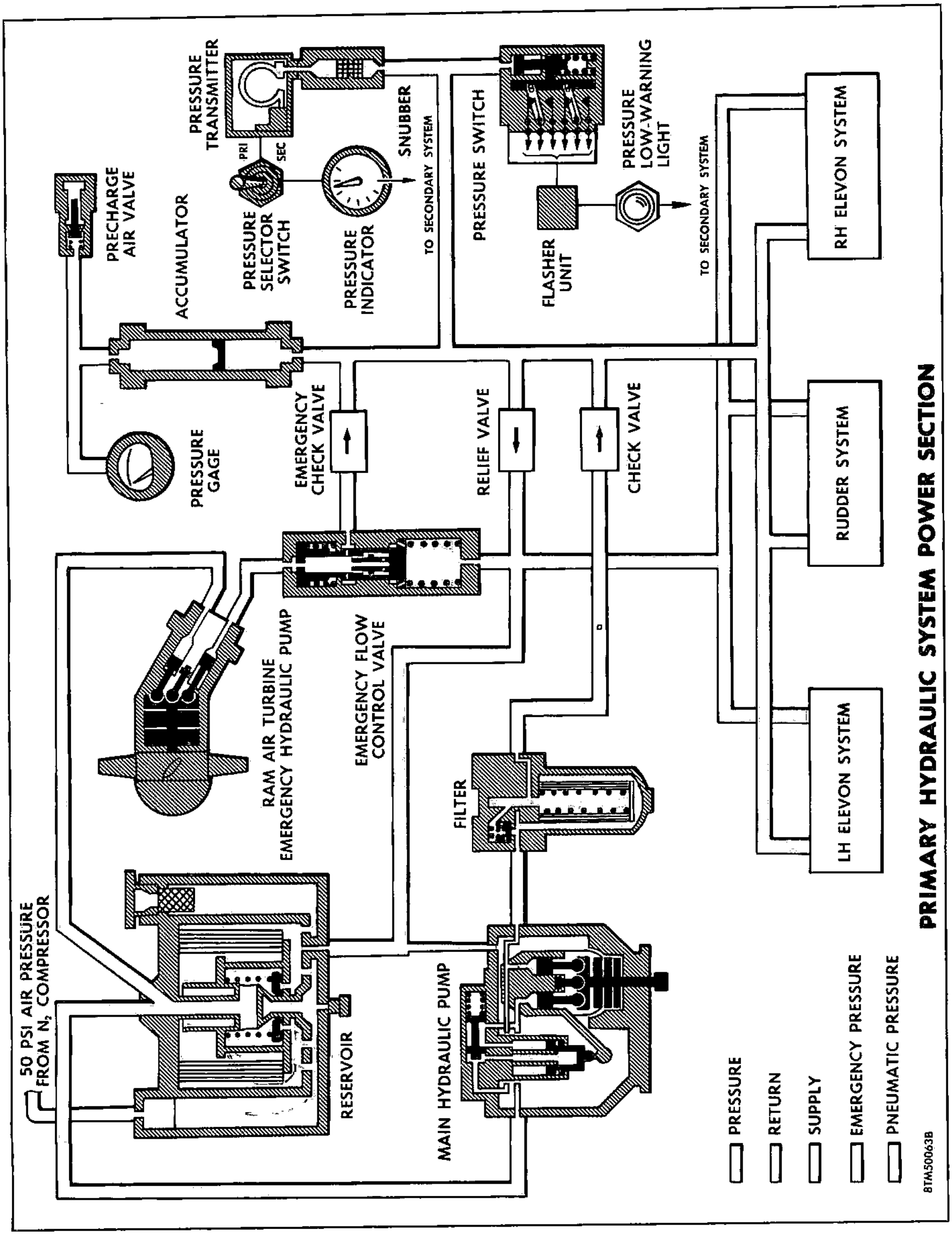

Figure 1-24. Primary Hydraulic Power Supply System

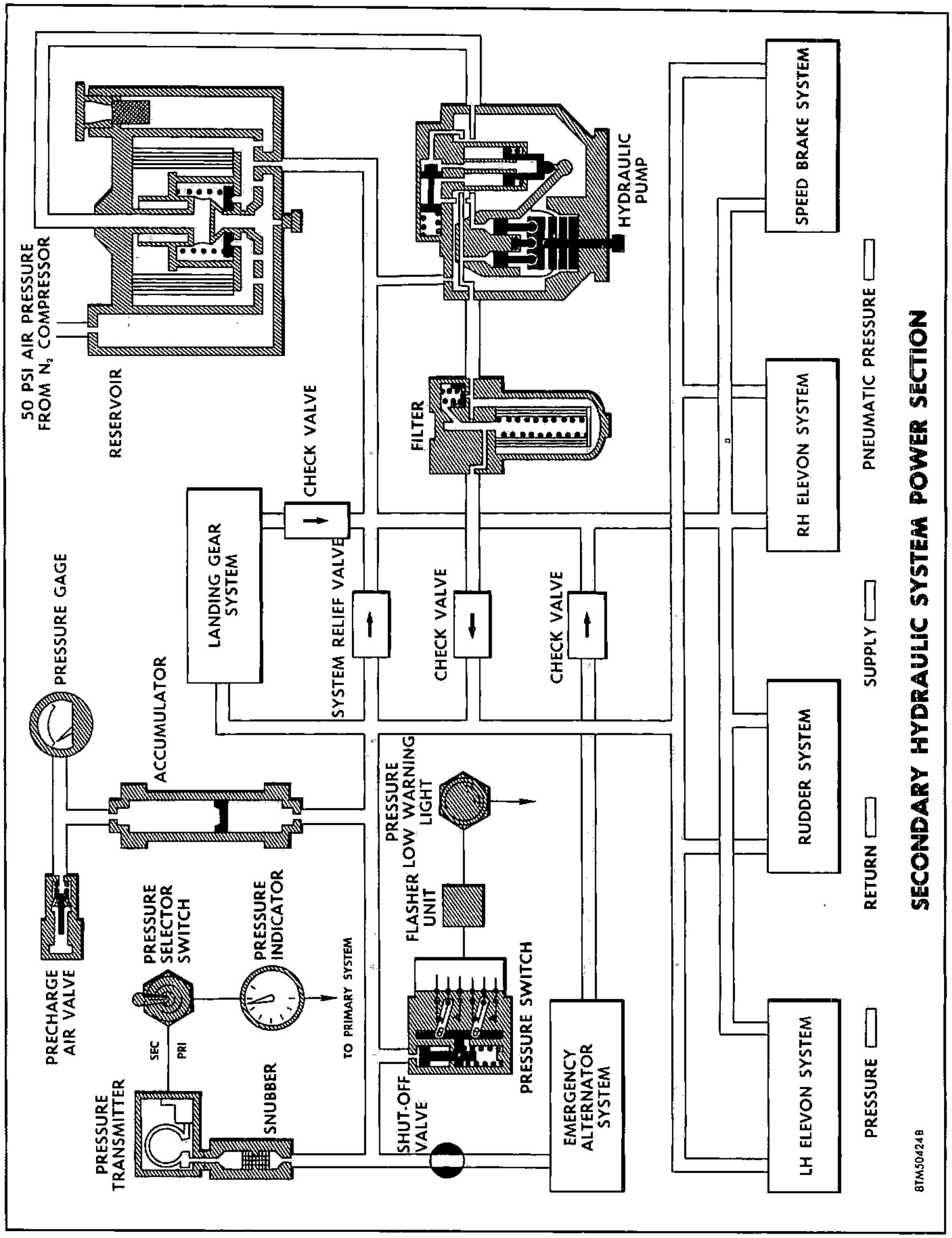

Figure 1-25. Secondary Hydraulic Power Supply System

The main hydraulic pump operates whenever the engine is running. It draws fluid through the supply line from the top of the reservoir to the inlet side of the pump. Regulated air pressure at the reservoir assists the pump in drawing fluid and also prevents cavitation at the pump. This pump is the variable displacement type and supplies sufficient pressure (3000 psi) for all subsystem operations. When pressure is not required by a subsystem, the pump merely delivers enough hydraulic pressure to keep the power system at 3000 psi. Any pressure in excess of 3000 psi is returned to the reservoir by a bypass valve.

Hydraulic fluid leaves the pump and enters a high pressure filter which removes foreign material that may be in the fluid. A bypass filter in the filter allows the pressure and fluid to continue on into the system, should the filter become clogged. The hydraulic fluid is then routed to the subsystems through a check valve, which prevents reverse flow of pressure.

As you will note, in figures 1-24 and 1-25, the hydraulic fluid branches off in two directions, one toward the accumulator and the other toward the susbsystems. Hydraulic pressure is now available at the control valves of the operating subsystems. The operation of these subsystem control valves allows pressure to enter the subsystems and move the flight controls.

In figure 1-25, showing the secondary power system, note the other subsystems that use hydraulic pressure to operate their components. Going back to the check valve next to the filter, notice that pressure is also routed to the accumulator and to the presssure switch and transmitter. The accumulator is precharged either with nitrogen or clean dry air to about 750 psi. The accumulator serves as a shock absorber to remove surges in pressure caused by the hydraulic pump and also acts as a reservoir of pressure. The relief valve, shown in figures 1-24 and 1-25, protects its respective system from excessive pressure. This relief valve opens and relieves pressure back to the reservoir whenever the pressure downstream from the main pump exceeds the setting of the valves.

The hydraulic pressure routed to the pressure transmitter first enters a snubber which protects the Bourdon tube in the transmitter. Increasing system pressure tends to open the Bourdon tube arc, while decreasing pressure allows the arc to close. This action of the tube is electrically transmitted through the pressure selector switch to the pressure indicator gage in the cockpit. When the switch is in the PRI (primary) position, the gage registers the pressure in the primary system; with the switch in the SEC (secondary) position, the gage indicates secondary hydraulic system pressure. This gage can register the pressure of only one hydraulic system at a time.

The pressure switch in either system actuates when the pressure in that system is low. When the pressure drops to approximately 800 psi, the switch sends a signal through the flasher unit to the warning light in the cockpit. When the light flashes, the pilot checks for the low system by noting the pressure on the system gage. Note in figures 1-24 and 1-25 that the pressure low-warning light is electrically connected to both systems. If the pressure in both systems is low, this light will come on steady instead of flashing. This warning light will go off when the pressure again reaches 1000 psi.

EMERGENCY HYDRAULIC SYSTEM OPERATION.

The emergency system (shown on figure 1-24) operates only when the primary system pump has failed. When the pilot determines that the primary system pump is inoperative, he operates the switch, lowering the air turbine hydraulic pump into the slipstream. As mentioned earlier, the emergency hydraulic pump and its operating turbine are attached to the hydraulic compartment door. When this door is opened into the airstream by the pilot, the turbine rotates and drives the pump. The pump draws fluid from the primary reservoir in the same manner as the main hydraulic pump.

Hydraulic pressure is routed through a flow control valve which regulates the flow either back to the reservoir or into the primary system. Hydraulic fluid is routed back to reservoir until the emergency pump has increased the flow sufficiently to open the control valve and allow the pressure to pass through the check valve and into the primary system. Actually, this pressure will not exceed about 1000 to 2600 psi, but should the pressure in the system become excessive, the relief valves shown below the emergency system check valve will open and relieve excess system pressure.

In some cases the explanation of the components in the hydraulic systems consisted of very simple devices —probably unlike any such components that you may ever see in a modern airplane hydraulic system. However, the main idea of operation, plus the understanding of the principles involved in each of these devices, makes it easy for you to understand the more efficient and complex devices in airplanes.

This chapter should be of use as an excellent reference and refresher on the elements of hydraulics. In almost every instance the *cause* and *effect* of any operation in the system is due to some principle of hydraulics discussed in this chapter. Whenever any part of the F-102A hydraulic system seems hazy to you, refer to this chapter for a quick review of the basic principles.

The next chapter covers the F-102A hydraulic power supply system and the succeeding chapters go into the operation of the hydraulically operated subsystems. Upon completion of this supplement, you will find that the F-102A hydraulic system is quite similar to the simple closed-center system discussed earlier in this chapter.

SUMMARY.

The fundamental knowledge about hydraulic principles and hydraulic equipment that you have learned and reviewed in this chapter is necessary to properly understand how and why the F-102A hydraulic system operates. In the F-102A hydraulic system many newly developed devices and instruments are used. Nevertheless, the hydraulic fundamentals outlined in this chapter are the basis for all their operations. What you have learned in this chapter is, therefore, your tool to the understanding of hydraulic systems and equipment.

Chapter II

F-102A HYDRAULIC POWER SUPPLY SYSTEMS

Contents *Page*

No device or system is any more reliable than the source of power that operates it. No automobile, regardless of its year or make, is any more reliable than the battery that supplies the power to get it started. The same is true of the generators in your home town electrical power system. The power source is always the most vital unit in any operating mechanism or system.

Hydraulic systems are known for their dependability. Naturally, our first thought then is to question the reliability of the power source in hydraulic systems. A quick answer to this is—excellent. The fluid pressure generating components of hydraulic systems are of test-proven reliability for continuous high- and low-pressure operations.

This high degree of reliability is why hydraulic systems are called upon to actuate a number of the essential and precision-operated subsystems in modern high speed aircraft. Just as in all walks of life—the more you are capable of doing, the more is expected of you. This trend of increased operating demands has led to the continuous improvement of hydraulic power supply components.

In the F-102A, the hydraulic power supply system has the all important job of operating the flight controls and a number of other systems. On such a high-performance airplane as the F-102A, with its critical mission of interception and protection, the hydraulic power system is definitely in the spotlight.

In this chapter you will learn how this much used source of power is developed and maintained in the F-102A. Hydraulic maintenance personnel always like to know what goes on inside the components they must service and replace. This chapter does just that! Here you will see in a simple and logical manner how the hydraulic power in this airplane lives up to all of its expectations in delivering dependable and efficient power.

F-102A HYDRAULIC SYSTEM.

The F-102A hydraulic system is actually composed of three systems—two separate, completely independent, main-center systems and an emergency system. Each independent hydraulic system is divided into a power supply system and one or more power distribution subsystems. The subsystems distribute power to move the flight controls, raise and lower the landing gears, open and close the speed brakes, steer the nose wheel, and drive the emergency a-c generator.

One of the independent power supply systems furnishes power only to the flight control system; this

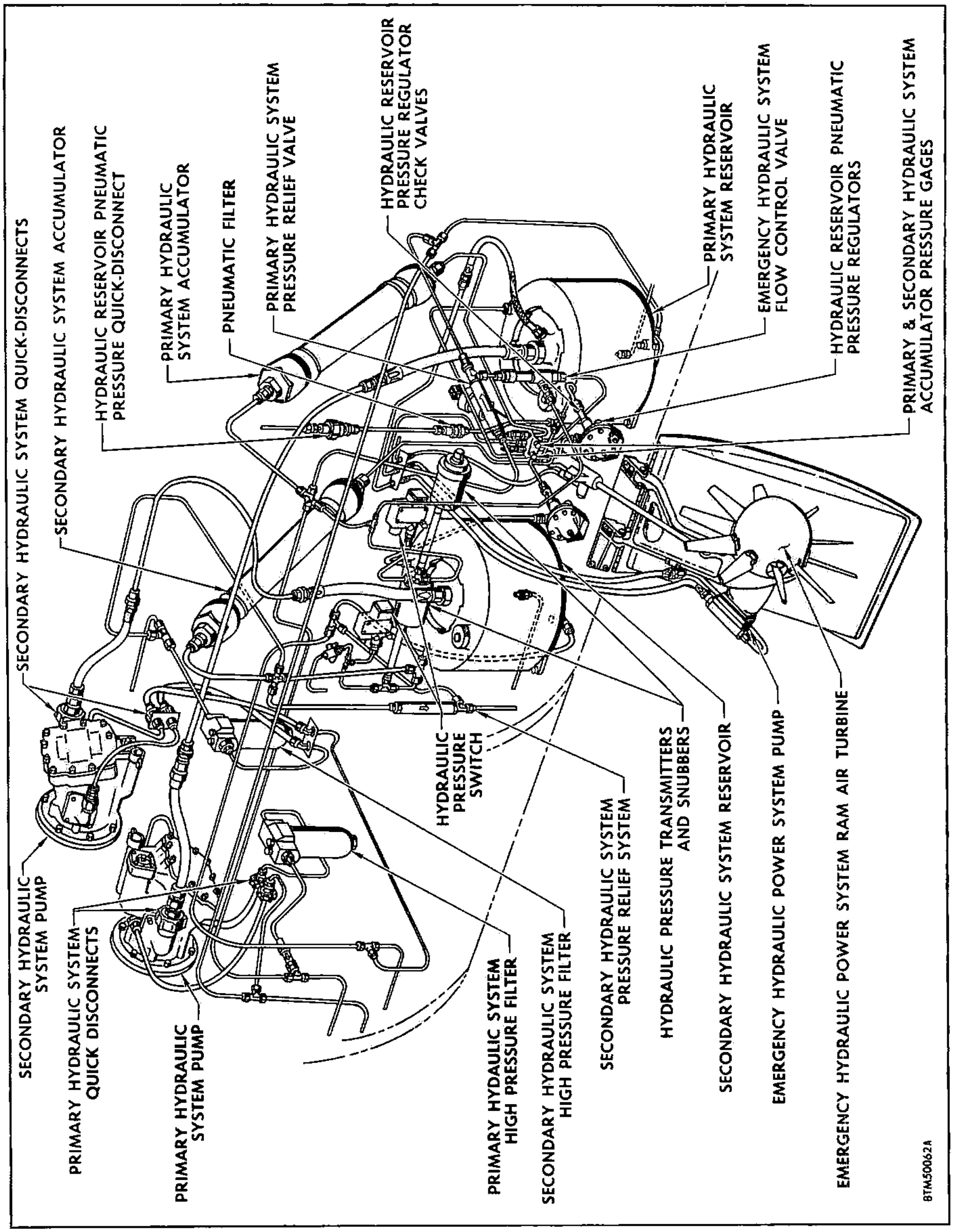

Figure 2-1. F-102A Hydraulic Power Systems

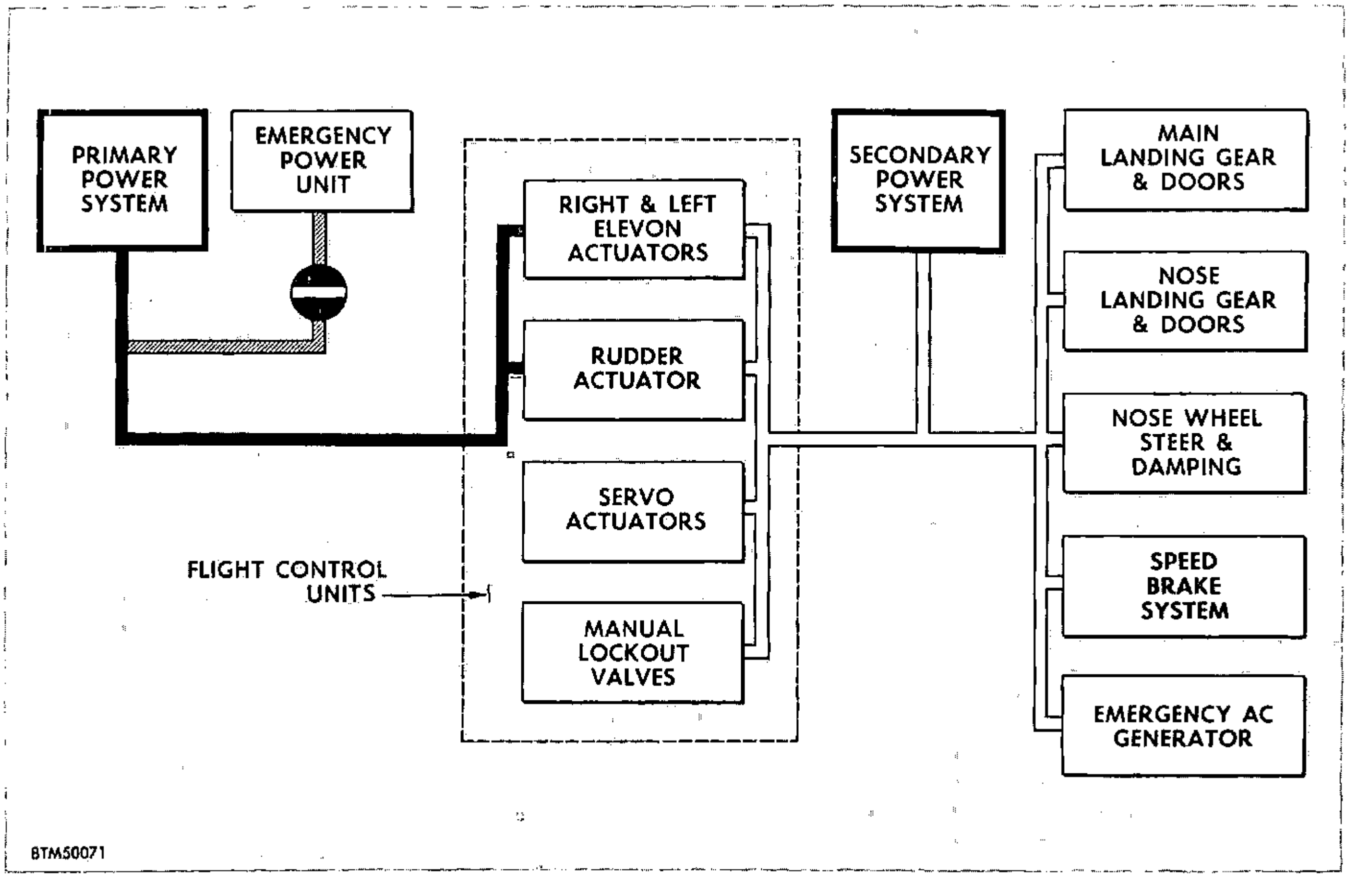

Figure 2-2. Hydraulic System Diagram

system is called the *primary hydraulic system.* The other power supply system also furnishes power to the flight control subsystem as well as power to all other hydraulically operated subsystems in the airplane; this system is known as the *secondary hydraulic system.* Hydraulic pressure for both systems is supplied by two engine driven pumps. Each pump pressurizes the hydraulic fluid in its respective independent system.

The emergency system consists primarily of a ram-air, turbine-driven pump and a flow control valve. The hydraulic lines of this emergency system connect only to the primary system. No connections are made to the secondary system. The emergency pump furnishes alternate power for the flight controls in case the primary and secondary pumps should malfunction.

Figure 2-1 shows all the components in the power supply systems just as you will find them in the F-102A. Except for the two engine-driven pumps and the two high pressure filters, all components shown on the illustration are located in the hydraulic accessory compartment. This compartment is on the right side of the fuselage just forward of the main landing gear wheel well. All units in this area are accessible through the ram-air turbine door which is shown in the open position. Through this door you can service the reservoirs with hydraulic fluid and the accumulators with high pressure air.

Note in figure 2-1 that most units in the primary and secondary systems are alike. The one main difference in these units is in the size of the reservoirs. The secondary system reservoir, although identical in construction, is larger than the primary reservoir. This larger size in the secondary reservoir is necessary since it supplies fluid to a number of subsystems, while the primary reservoir supplies only the flight control subsystem. In addition, you will also see that an extra line taps off the top of the primary reservoir. This line supplies fluid to the emergency ram-air driven pump on the compartment door.

One easy way by which you may distinguish between the components of these two power supply systems is to remember the right and left-hand location plan. As you look in the hydraulic accessory compartment you will find the primary power system components to the right (forward) side of the compartment, and the secondary power system components on the left (aft) side. This location plan also holds true in the engine accessory compartment behind the main landing gear wheel well. Here you will find the primary system pump and filter on the right and the secondary system pump and filter on the left.

Referring to figure 2-1 again, note the hydraulic lines that connect the various components of the power systems. Later in this chapter, when we discuss each system separately, you will learn how fluid is routed through these lines. So that you can distinguish primary system lines from those in the secondary system, an identification code is established. First, to show that a line carries hydraulic fluid, blue and yellow striped tape is wrapped around the lines at various intervals. Secondly, the lines are stenciled with markings that identify the system and the direction of normal fluid flow in that particular line. For the identification of these lines and those used in other systems, you should refer to the F-102A Handbook of Maintenance Instructions, T.O. 1F-102A-2-3.

WHY DUAL HYDRAULIC SYSTEMS ARE USED.

Two hydraulic systems are used on the F-102A for one main reason—the flight control surfaces are moved by full hydraulic power. Remember that the primary hydraulic system functions exclusively for the flight controls, while the secondary hydraulic system supplies power to the flight controls and other hydraulically operated systems. With two systems supplying power to move the flight control surfaces, positive actuation is always assured.

These two hydraulic systems also offer an additional safety feature. For example, if there were only one hydraulic system in the airplane, and that system failed, the pilot would have no other means of operating the flight control surfaces. As you will learn in Chapter III when we discuss the flight control hydraulic system, no direct mechanical linkage connects the pilot's controls with the flight control surfaces. Therefore, you can see that if the F-102A had only one hydraulic system and it failed, the pilot would not be able to move his controls. With a flight control system that operates by two hydraulic systems like that on the F-102A, you can visualize the importance of having added assurance against a complete hydraulic system failure.

SUBSYSTEMS OPERATED BY HYDRAULIC POWER.

You learned at the beginning of this chapter that the hydraulic power system acts as the motivating force for five important subsystems in the airplane. The block diagram below shows these subsystems and the power system that supplies the hydraulic pressure for their actuation. In the diagram note that the primary power system supplies pressure only to the right and left elevon actuators (cylinders) and the rudder actuator. The secondary power system near the right side of the diagram supplies a number of systems. You will note that in addition to other flight control units it powers the same flight control units as the primary system. During emergency conditions with the secondary system "out," the airplane can be controlled without the two bottom units (servo actuators and manual lockout valves) functioning. To the right of the secondary power system source, you can see the other subsystems powered by this source. Note the emergency power unit next to the primary power source. This is the ram-air driven pump that comes into play when he main system pumps malfunction.

Although these subsystems are discussed in detail in Chapters II and IV, let's take a quick look at them before we go into the power supply systems. This will help you understand what happens to the hydraulic pressure that the power systems produce.

FLIGHT CONTROL SYSTEM.

As you already know, the F-102A flight control surfaces consist of a conventional rudder attached to the vertical stabilizer and two elevons, one on each wing trailing edge. These elevons serve as both elevators and ailerons. Hydraulic pressure from both the primary and secondary power systems move these surfaces. When the pilot moves the control stick and rudder pedals, mechanical linkage displaces dual control valves—one for the rudder and one for each elevon. These valves in turn route hydraulic pressure from both power systems to the dual actuators that move the surfaces.

These dual actuators are actually two cylinders in one with one actuating piston rod. One end of the actuating cylinder has *pressure* and *return* chambers for the primary system, while the other end has *pressure* and *return* chambers for the secondary system. When the surfaces reach a position corresponding to the cockpit control, a feed-back system returns the dual control valves to *neutral* and shuts off pressure to the surface actuators, thus stopping surface movement. A complete description and schematic illustration of the actuating cylinders will be given in Chapter IV.

The secondary system also powers other units in the flight control system. These units are the servo actuators in the rudder and elevon systems and the lockout valve in the elevon system. When the flight control system is in the MANUAL MODE, hydraulic pressure is routed to the servo actuators. Pitch and yaw damping and turn coordination signals, which are generated in the manual mode system, cause the servo actuators to move. The servo actuators are mechanically connected to the control valves discussed in the paragraph above, and superimpose their damping impulses on the movement of the control surfaces.

When the flight control system is in the PILOT ASSIST MODE, secondary hydraulic pressure is routed to the elevon lockout valve. This lockout valve "cuts out" pilot initiated movement on the control valve and allows the *pilot assist* system to move the elevons.

A complete description how the flight control system is electrically controlled is given in another training supplement in this series which covers the F-102A

Flight Control System. In this hydraulics supplement, the discussion of the flight controls will be confined to the hydraulic portion of the system. In those cases where electrical control circuits are mentioned and you would like additional information, refer to the Flight Control Training Supplement or the F-102A Handbook of Maintenance Instructions, T.O. 1F-102A-2-7.

Landing Gears.

The three landing gears and all of the landing gear doors are actuated by secondary hydraulic system power. Four selector valves direct the fluid from the main hydraulic system lines to the gear and door actuating cylinders. These actuating cylinders raise and lower the gears and open and close the gear doors. Each gear and door operates separately, but their actions are coordinated to occur in sequence. That is, during the raising cycle the gears retract then their respective doors close, while in the lowering cycle the doors open first then the gears extend.

An electrical switching circuit controls the sequence of the gear and door movements. After the control handle in the cockpit is placed in the desired gear UP or gear DOWN position, this switching circuit takes over and controls the opening and closing of the selector valves through the use of solenoids. Restrictors installed in the gear and door hydraulic lines reduce their speed of operation and thus prevent the assemblies from slamming into position. You should understand that this description of the landing gear system is very general in nature and pointed toward the hydraulic application.

For a complete description of the landing gear system and its electrical control circuits, refer to Chapter VII of the Airplane General Training Supplement.

Nose Wheel Steer-Damper.

The nose wheel steering system provides the pilot with directional control of the airplane while on the ground. This system is actuated by hydraulic power from the secondary system. The principal component of this system is a hydraulic steer-damper unit which utilizes hydraulic power for both steering control and shimmy damping of the nose wheel when it is in a free castering position. The hydraulic steering system is controlled by a mechanical linkage system which connects the two rudder pedals and the steer-damper unit. A turn on the ground is made either to the right or left by depressing the corresponding rudder pedal. The nose wheel steer-damper system is discussed in detail later in Chapter IV.

Speed Brakes.

The two hydraulically-actuated speed brakes are located just below the rudder. They are opened by the pilot to slow the airplane in the air and to deploy the drag chute on landing. Each speed brake is powered by two actuating cylinders that operate in unison from secondary hydraulic system pressure. These speed brakes, which form a part of the drag chute housing in the island at the base of the rudder, are selectively opened and closed by means of a control switch. Hydraulic pressure is automatically removed when the speed brakes are retracted, and an integral lock mechanism holds the brakes in place during normal flight.

Emergency A-C Generator.

The a-c emergency generator is driven by a hydraulic motor that is controlled by a solenoid-operated shutoff valve. This hydraulic motor is powered by the secondary hydraulic system, and operates at a high speed. The turning of this motor rotates the emergency a-c generator. If the main electrical power supply should fail, the solenoid-operated shutoff valve will automatically be energized by the emergency a-c bus switch. Energizing this solenoid opens the valve which permits fluid to flow into the hydraulic motor which in turn drives the emergency generator.

THE PRIMARY HYDRAULIC POWER SYSTEM.

The primary hydraulic power system, shown in schematic form in figure 2-3, is a closed-center system. You learned in Chapter I that closed-center systems do not have hydraulic fluid in constant circulation, instead these systems are primed with pressure ready for use when demanded by a subsystem. That is just the way the primary hydraulic power system in the F-102A operates—it is pressurized to about 3000 psi whenever the engine is operating. When the flight control system is moved, it demands hydraulic pressure; when the controls are stationary and hydraulic pressure is not needed, hydraulic fluid is stationary in the system, except between the reservoir and pump. You will learn just how this system functions when we discuss its operation. First let's take a look at this power system.

By referring to figure 2-3 and figure 2-1, you can see what units are included in this system. In the schematic illustration you will note an emergency pump and an emergency flow control valve. Although these two units are connected to the primary power system, we will discuss them under the emergency system. Basically, the primary hydraulic power system consists of a pressurized reservoir, an engine-driven main hydraulic pump, a high-pressure filter, an accumulator, check and relief valves, a pressure indicating system, and a low-pressure warning system. As you know, all these units except the main pump and filter are in the hydraulic accessory compartment. The pump and filter are in the engine accessory compartment.

Shown at the bottom of figure 2-3 is the flight control subsystem that uses the primary system power. Now that you know the units that make up this system, we

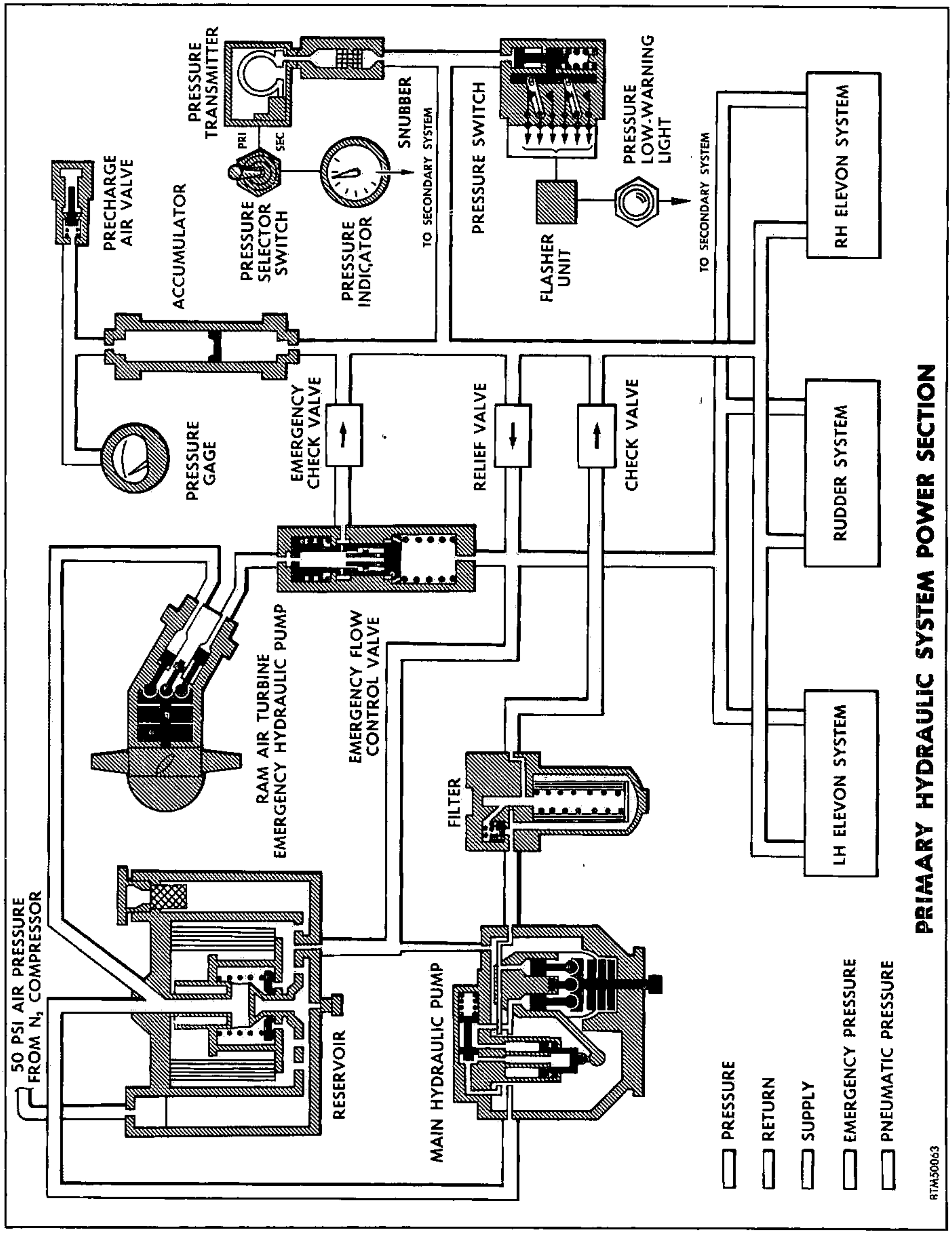

Figure 2-3. Primary Hydraulic Power System Diagram

will discuss how the system functions to keep a constant supply of pressure always available for the flight controls.

PRIMARY POWER SYSTEM OPERATION.

The main hydraulic pump operates whenever the engine is running. It draws fluid through the supply line from the top of the reservoir. Regulated air pressure, which is supplied by the N_2 section (aft section) of the engine compressor, is pumped into the reservoir to assist the engine pump in drawing fluid, and thereby preventing cavitation (starving because of air mixing with the fluid) at the pump. The pump is the variable displacement-type and supplies subsystem. When pressure is not required by the flight controls, the pump delivers just enough pressure to keep the power system at 3000 psi. Any pressure in excess of 3000 psi is returned to the reservoir by the pump bypass valve.

Pressurized hydraulic fluid leaving the pump enters the high-pressure filter where any foreign material is removed by the filtering element. Should this filter element become clogged or even partially restricted and reduce the system pressure, a bypass valve in the filter assembly head will open and permit the pressurized fluid to continue on into the system. From this filter the hydraulic fluid passes through a check valve which prevents reverse flow.

As you can see in figure 2-3, the hydraulic system branches into two directions as it leaves the check valve—one branch goes to the accumulator and pressure indicating systems, the other branch goes to the flight control subsystem. This is actually the way the lines are routed in the airplane. If you refer back to figure 2-1 you can see this branching of the lines outboard of the primary system filter. Hydraulic pressure is now available at the control valves in the flight control system. The operation of these control valves allows pressure to enter the flight control system and move the control surfaces.

Going back to the check valve where the system pressure branched, note the line leading to the accumulator and to the pressure switch and transmitter. The accumulator is precharged on the ground with either nitrogen or clean, dry air to about 750 psi. It serves as a shock absorber to remove surges in pressure caused by the hydraulic pump and also acts as a reservoir of pressure. The relief valve, shown between the two check valves, protects the system against excessive pump pressure. This valve opens and relieves excess pressure back to the reservoir whenever the pressure downstream from the main pump exceeds the setting of the valve.

The hydraulic pressure, routed to the pressure transmitter, first enters a snubber which protects the Bourdon-tube in the transmitter. Increasing system pressure tends to open the Bourdon-tube arc, while decreasing pressure allows the arc to close. This action of the tube is electrically transmitted through the pressure selector switch to the pressure indicator gage in the cockpit. When the switch is in the PRI position, the gage registers the pressure in the primary system. In the SEC position, the gage indicates secondary hydraulic system pressure. This gage can only register the pressure of one system at a time.

The pressure switch in the system actuates when the pressure in the system is low. When the pressure drops to approximately 800 psi, the switch closes and completes a circuit through the flasher unit to the warning light in the cockpit. When the light flashes, the pilot checks for the low system by noting the pressure on the system gage. In figure 2-3 note that the pressure-low warning light is connected electrically to both hydraulic systems. If the pressure in both systems is low, this warning light will come on steady instead of flashing. This light will go off when the pressure in both systems again reaches about 1000 psi. If the pressure in only one system returns to 1000 psi, the light will then go from steady to flashing.

EMERGENCY HYDRAULIC POWER SYSTEM.

The emergency hydraulic power system consists of the ram-air, turbine-driven pump on the inboard side of the hydraulic accessory compartment door, a flow control valve, and a check valve as shown in figure 2-1. This pump furnishes emergency hydraulic power for operation of the flight controls whenever the main hydraulic pumps fail or the engine stops in flight. In the schematic illustration (figure 2-3) you can see how this emergency system ties into the primary system at the reservoir and emergency flow control valve.

The emergency pump is driven by the ram-air produced by the forward motion of the airplane when the compartment door is opened into the airstream. Air pressure (furnished by the high-pressure pneumatic system) opens this door. Door movement is controlled from the cockpit. Thus this emergency pump assures the pilot of having hydraulic pressure during flight to operate the rudder and elevons.

EMERGENCY HYDRAULIC SYSTEM OPERATION.

The emergency system, shown on the primary hydraulic system schematic (figure 2-3), operates only when the primary system pump is not functioning as determined by the pilot. He then operates the switch extending the air turbine hydraulic pump into the airstream. As mentioned in the system description, the emergency hydraulic pump and its operating turbine are attached to the hydraulic compartment door. When this door is opened into the airstream by the pilot, the turbine rotates and drives the pump. The pump draws fluid from the primary reservoir in the same manner as the main hydraulic pump. The N_2

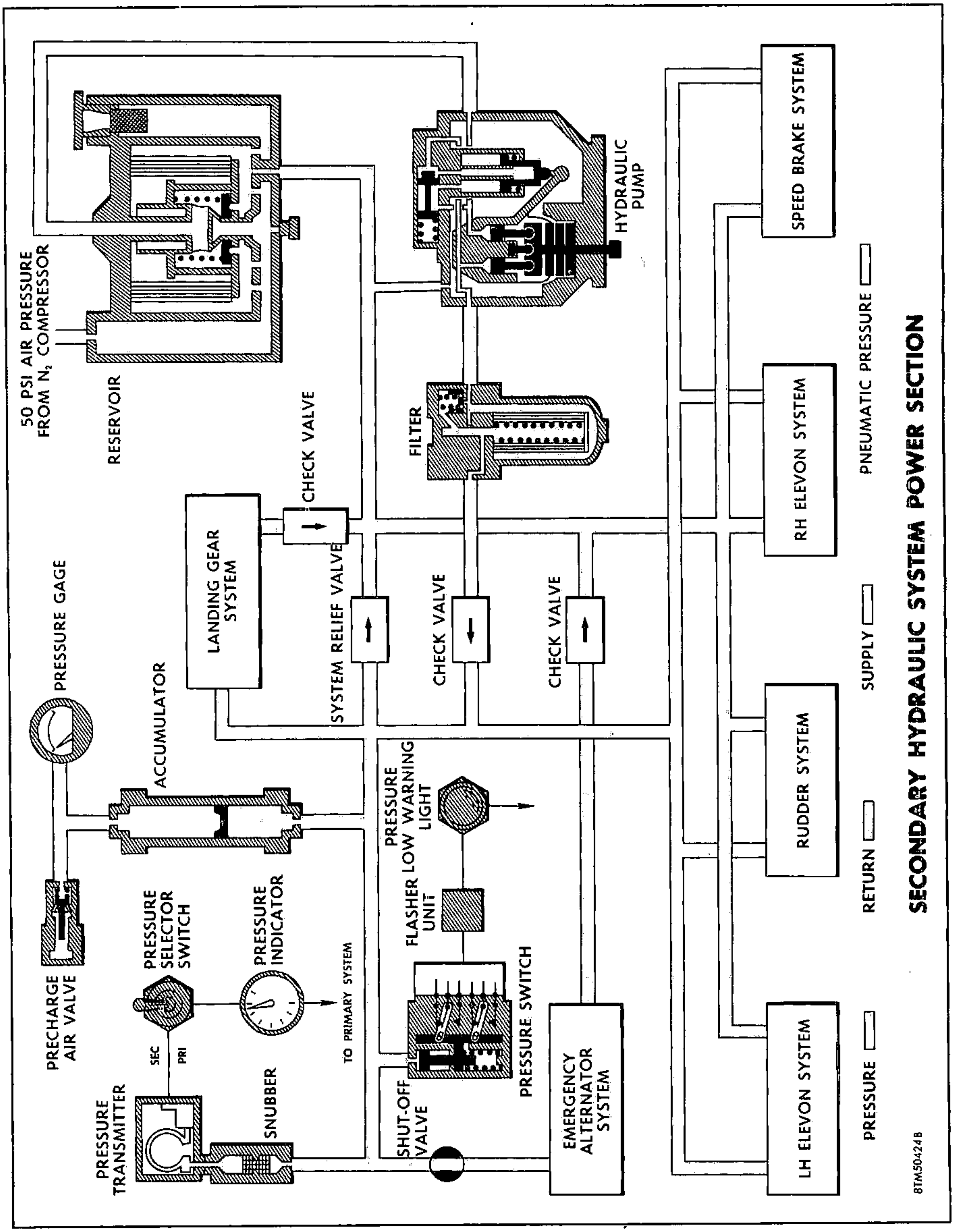

Figure 2-4. Secondary Hydraulic Power System Diagram

compressor air which pressurizes the reservoir also prevents cavitation (starving because of air mixing with the fluid) at the emergency pump just as it does at the main pump.

Hydraulic pressure is routed from the pump through a flow control valve which regulates the flow either back to the reservoir or into the primary system. Hydraulic fluid is routed back to the reservoir until the emergency pump has increased the flow sufficiently to open the control valve and allow the pressure to pass through the check valve and into the primary system. Actually, this will not exceed 1000 to 2600 psi, but should the pressure in the system become excessive, the relief valve shown below the emergency system check valve (figure 2-3) will open and relieve excess system pressure.

THE SECONDARY HYDRAULIC POWER SYSTEM.

The secondary hydraulic power system is basically the same as the primary system. In figure 2-4 you can see that this system has the same components that you found in the primary system. All these components, except the hydraulic pump and filter are in the hydraulic accessory compartment. The pump and filter, like those in the primary system, are in the engine accessory compartment. As you learned earlier in this chapter, this secondary system is the left-hand system —that is, the pump and filter are on the left side in the engine accessory compartment, and the remaining units are on the left (aft) side of the hydraulic accessory compartment.

In the secondary system, note the absence of an emergency system. This is one of the main differences between the two power systems. Since the flight control surfaces can be moved by pressure from either main system, emergency power is needed in only one of these systems. The other main differences are in the subsystems operated by the secondary system.

Where the primary system furnished power only to the essential units (actuators) in the flight control system, the secondary system furnishes power to the same units plus the servo actuators and the elevon lockout valve in the flight control systems. It also supplies power to the landing gear, the speed brakes, and the emergency a-c alternator.

SECONDARY POWER SYSTEM OPERATION.

The secondary power system operates in exactly the same manner as the primary. Therefore, if you know how one system operates, you also know how the other operates. Trace the operation of the secondary system in figure 2-4 from the reservoir through the pump to the operating subsystems; then follow the system through to the accumulator, the pressure transmitter, and the pressure switch. By doing this you will not only see how the two power systems are alike, but you will become thoroughly familiar with their operation. Note the electrical connections to the primary system at the pressure indicator and at the pressure low warning light.

You will recall in the primary system operation we said that the indicator registered pressure for both systems, and that the warning light flashed for low pressure in either system and burned steady for low pressure in both systems. In no case throughout the power systems or their subsystems does the hydraulic fluid in one system mix with the fluid in the other. This is true, however, only when the subsystems are functioning normally. It is possible for the fluid in one system to leak into the other if you should have a defective seal in one of the control surface actuating cylinders. You will learn about this possible leakage malfunction when we discuss the flight control hydraulic system in Chapter III.

HYDRAULIC POWER SYSTEM COMPONENTS.

So far in this chapter you have learned what the hydraulic power systems are like and how they operate. You also learned the functions that each of the components in these systems perform. Now you will learn just how the pumps, reservoirs, filters, accumulators, relief valves, and flow control valve operate to perform these functions. Just as anything else mechanical, these components will require maintenance from time to time. So when we discuss these units, we will also bring up some of the maintenance problems you may encounter and what you can do to remedy the condition.

MAIN HYDRAULIC PUMPS.

The F-102A has two main hydraulic pumps—one for each hydraulic power supply system. These pumps, which are identical except for position of mounting and connection of lines, are mounted on and driven by the engine accessory gear case. The primary system pump is on the forward right side of the gear case; the secondary system pump is on the forward left side. Both pumps are accessible through the engine accessory compartment doors.

Figures 2-5 and 2-6 show these pumps installed on the engine. The primary pump is shown installed and lines connected, while the secondary pump is shown exploded from its mounting and lines disconnected. This will help you to see just how these pumps are installed.

In the illustration showing the primary system pump (figure 2-5) you can see its location on the engine. Just below the location view is the detail showing the pump installed. Here you can see how the four hydraulic lines connect to the pump. In the *suction line* note the large fitting next to the pump. This is a *quick-disconnect fitting* which you will use when connecting the hydraulic test stand to the airplane.

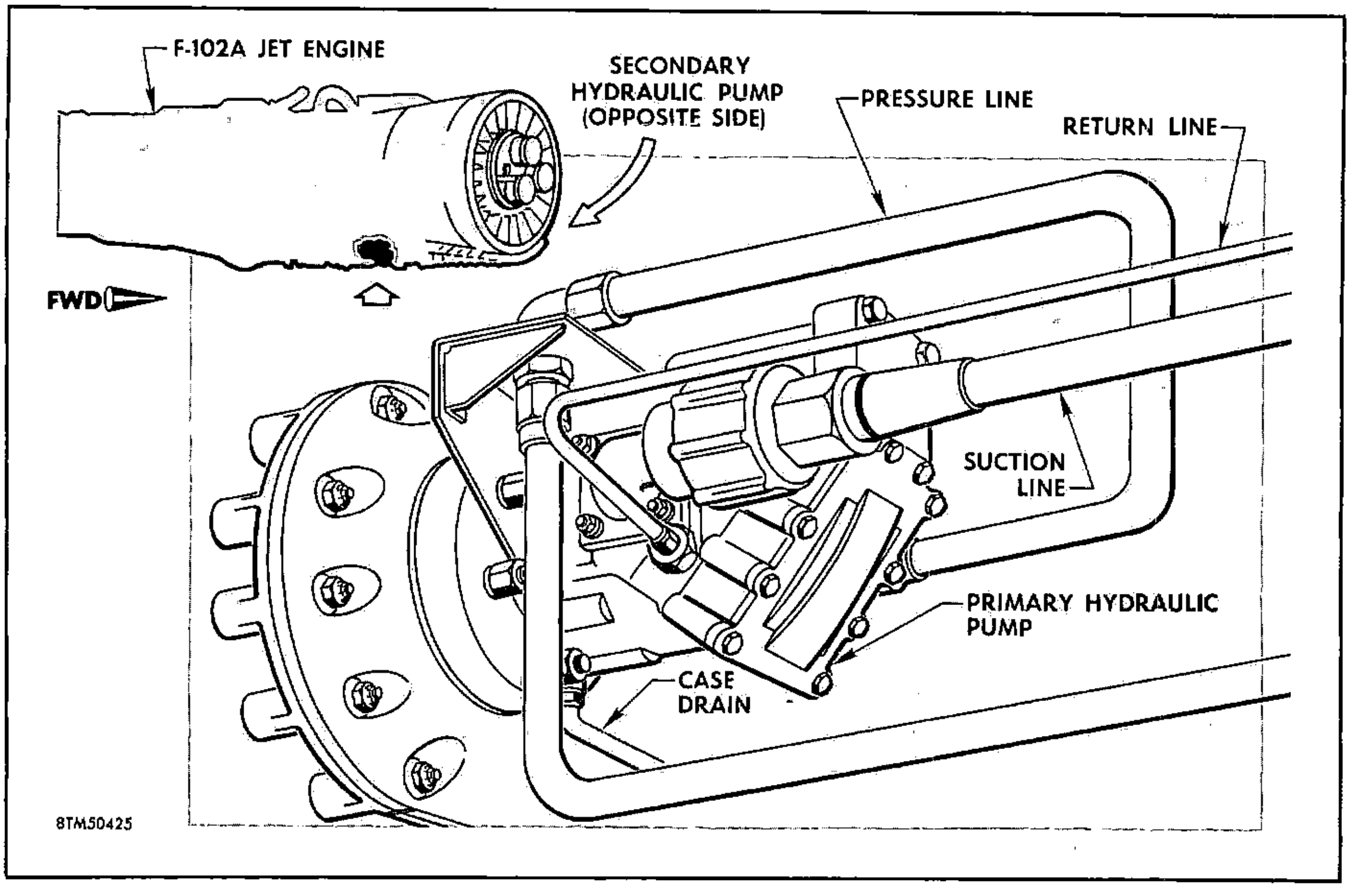

Figure 2-5. Primary Hydraulic System Pump

Although not shown, another quick-disconnect fitting is in the pressure line forward of the pump. You will learn more about these fittings later in the chapter when we discuss the hydraulic test stand.

The *suction line* supplies fluid from the reservoir to the pump. The *pressure line* routes the fluid from the pump to the high pressure filter. Just below the suction line is the *return* line which routes excess pressure at the pump back to the reservoir. Note the *case drain line* at the bottom of the pump. This is the line that drains overboard any fluid that leaks past the pump shaft seal. Now note the position of the pump —the forward slanted face is down. This is the way the pump is installed on the right (primary) side.

In figure 2-6, note how the secondary pump is installed. Note that the forward slanted face is up in this installation. Actually, the secondary pump is rotated 180° from that of the primary pump installation. All the line connections, except the case drain, are also reversed. The case drain in this mounting is now on the outside. (The mounting face of the pump has four case drains, but only one is used. The other three drains are plugged. This permits the pump to be installed on several airplanes and the most convenient drain opening can be used.)

How Pumps Produce Pressure.

The two main hydraulic pumps operate in somewhat the same manner in which the heart supplies blood to the body. As you know, when your body relaxes the heart pumps blood at a slow but sufficient output. However, if you run fast, work hard, or expend a lot of energy your heart beats faster in order to pump more blood as required by the body. The variable-displacement type hydraulic pumps in the F-102A operate on very much the same basic idea. The more pressure the subsystems require, the more pressure the pumps produce.

Internally the pumps consist primarily of a cylinder block with nine pistons, a yoke assembly, and a control piston assembly. The cutaway illustration (figure 2-7) shows the major units within the pump. Note how the yoke holds the cylinder block in place and how it is connected to the control piston. You can also see one of the pistons and some of the nine piston rods. These piston rods are connected to a rotating shaft mechanism which is turned by the spline shaft at the mounting end of the pump. When this spline shaft turns, it also rotates the attached piston rods and the universal connection which turns the cylinder block. When the yoke and cylinder blocks are at some angle other than 0°, such as the 30° angle shown in the illustration, the pistons will move up and down

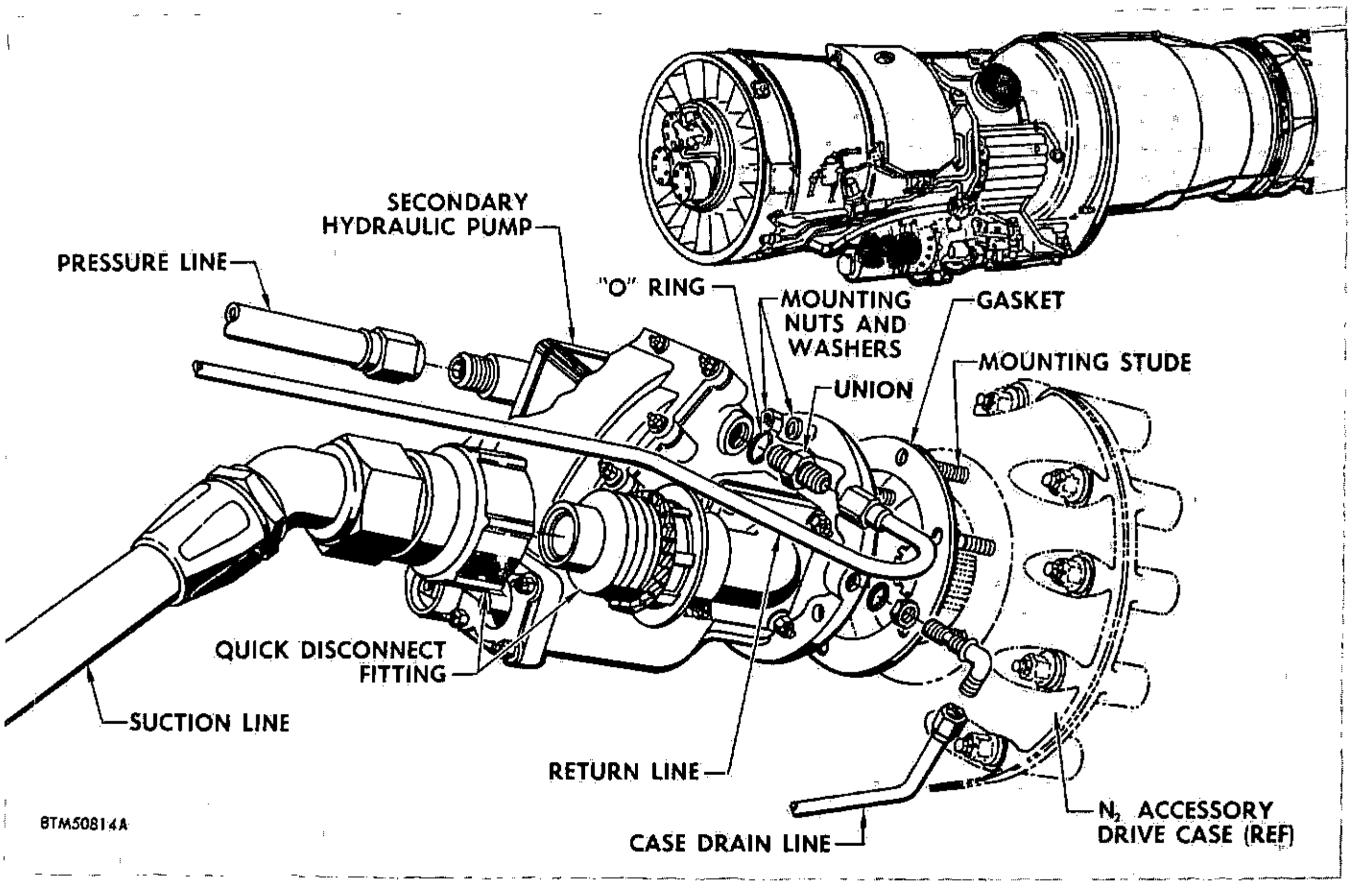

Figure 2-6. Secondary Hydraulic System Pump

within the cylinder block as they rotate—in other words, they make an effective stroke.

As the cylinder block rotates it picks up incoming (suction) fluid through the slots in the top of each cylinder. Then as this cylinder block continues its rotation, the pistons move up in their respective cylinders and pressurize the fluid. This pressurized fluid is forced out the pressure side of the pump and into the system. Part of this pressurized fluid is also routed to the control piston assembly shown at the bottom of the pump. The amount of pressure on this control piston assembly determines the angle of the yoke assembly through the connecting linkage. The net result of this pump operation is a smooth flow of pressurized fluid that keeps the system at a constant pressure.

How Pump Output Is Controlled.

You have learned that the main hydraulic pumps vary their output according to the amount of pressure demanded by the subsystems, and that they always maintain 3000 psi of pressure whether or not demanded by the subsystems. This is variable-displacement, and is governed by the angular position of the yoke and cylinder block. When the cylinder block is at its maximum angle, the stroke of the nine pistons is the longest; this results in maximum pump output. When the cylinder block is at its minimum angle, the stroke of the pistons is the shortest; this results in minimum pump output. The angular position of the cylinder block is controlled by the hydraulic pressure on the control piston assembly.

Figure 2-8 shows an operational cutaway of the pump with the cylinder block at its two extreme positions. Note in the upper view that the cylinder block is at its 0° position, and in the bottom view this cylinder block is at the 30° position. As you already know, part of the pump output pressure is routed to the control piston assembly. The hydraulic pressure that enters the control piston assembly is always the same as that in the system.

In both views of the illustration you can see the linkage that connects the control piston to the yoke assembly. In the upper view note that the 3000 psi system pressure (pump output) enters the control cylinder and pushes the piston back against the spring tension. With the piston completely back, the linkage positions the yoke near the 0° angle. At 3050 psi, the control piston positions the yoke at the 0° angle.

As you can see when the yoke and its cylinder block are at 0°, the piston rods are parallel to the axis of the spline shaft. Thus rotation of the spline will not cause

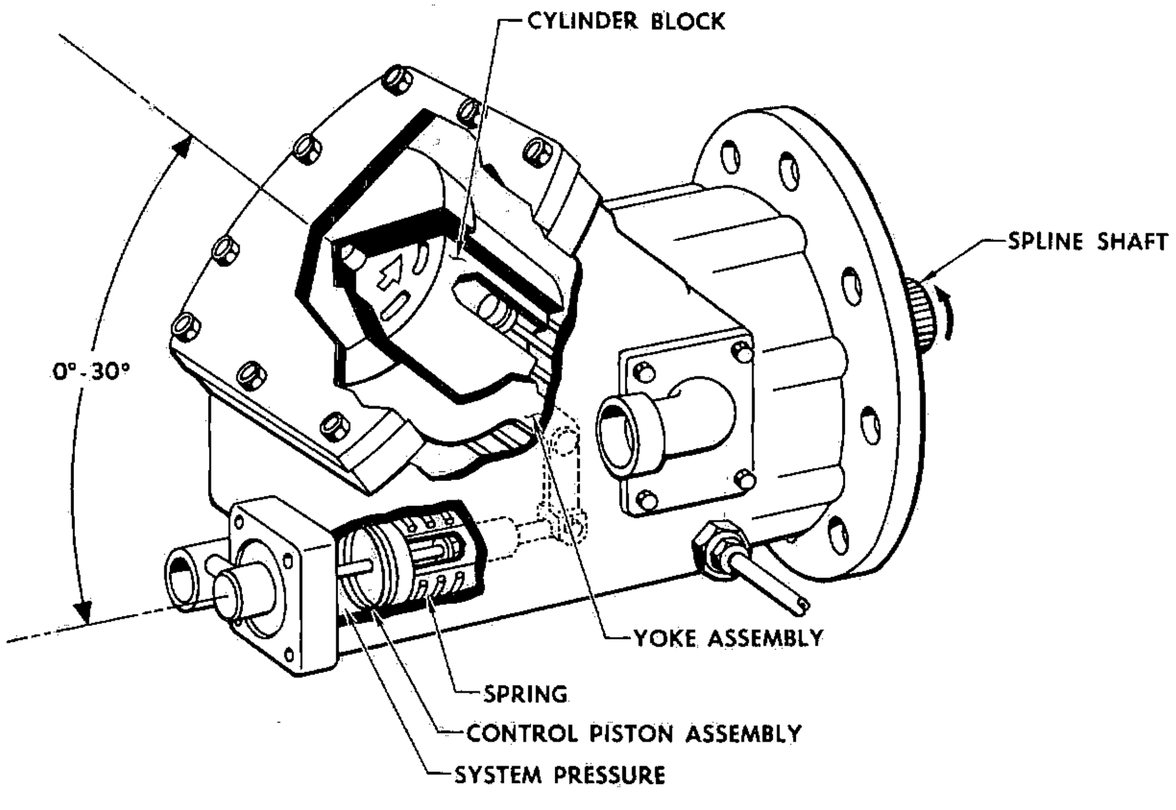

Figure 2-7. Main Hydraulic Pump Operational Cutaway

the pistons to move in and out of the cylinder block. However, if the angle is increased slightly (as at 3000 psi output), the piston rods will be at a slight angle to the spline shaft axis. The pistons will have an extremely short stroke resulting in just enough pump output to maintain system pressure when not demanded by a subsystem.

In the lower view pressure has decreased, caused by the pressure demand of subsystems. The spring tension is greater now than the pump output, or system pressure, and pushes the control piston to the left. This results in the yoke and cylinder being moved up toward the 30° angle. The angle between the piston rods and the spline shaft axis is greatest at 30°; thus the piston stroke is the longest, and the pump displacement is at its maximum output.

The angle of the cylinder block, and thus the stroke or displacement of the pistons, constantly varies from the 0° to the 30° angle depending on the system pressure requirements. This is why the pump is called the variable displacement type.

Maintenance.

The maintenance that you can perform in the hydraulic pumps is actually very little. About all you can do in line maintenance is change the pump if it becomes defective. The pump requires no lubrication, since it is self-lubricated by the hydraulic fluid it pressurizes. The most maintenance you will have to perform will be on pump leaks. Two types of leaks may be encountered. One is the external type leak in which the fluid drips from the pump or its connections, and the other is the type leak in which the pump draws in air.

The first type of leak, if it is not too much, will not affect the system operation materially except to deplete the system supply in the reservoir. The greatest problem from this type of leak is *fire*. This hydraulic fluid will *burn* if exposed to a spark or an excessive amount of heat.

The second type of leak can give you much more trouble. As you know, air drawn into the system will cause the actuation of a subsystem to be spongy, jerky, or slow. Air in the system will be easily detected by the subsystem operation by checking the bleed indicator on the system reservoir, by low system pressure, and by the oscillating needle on the pressure gage. Air in the system will also cause a noise at the pump due to cavitation (starving because of air mixing with the fluid). This noise will also exist in the system lines—much like water pipes in your home when they "hammer" due to air in the water.

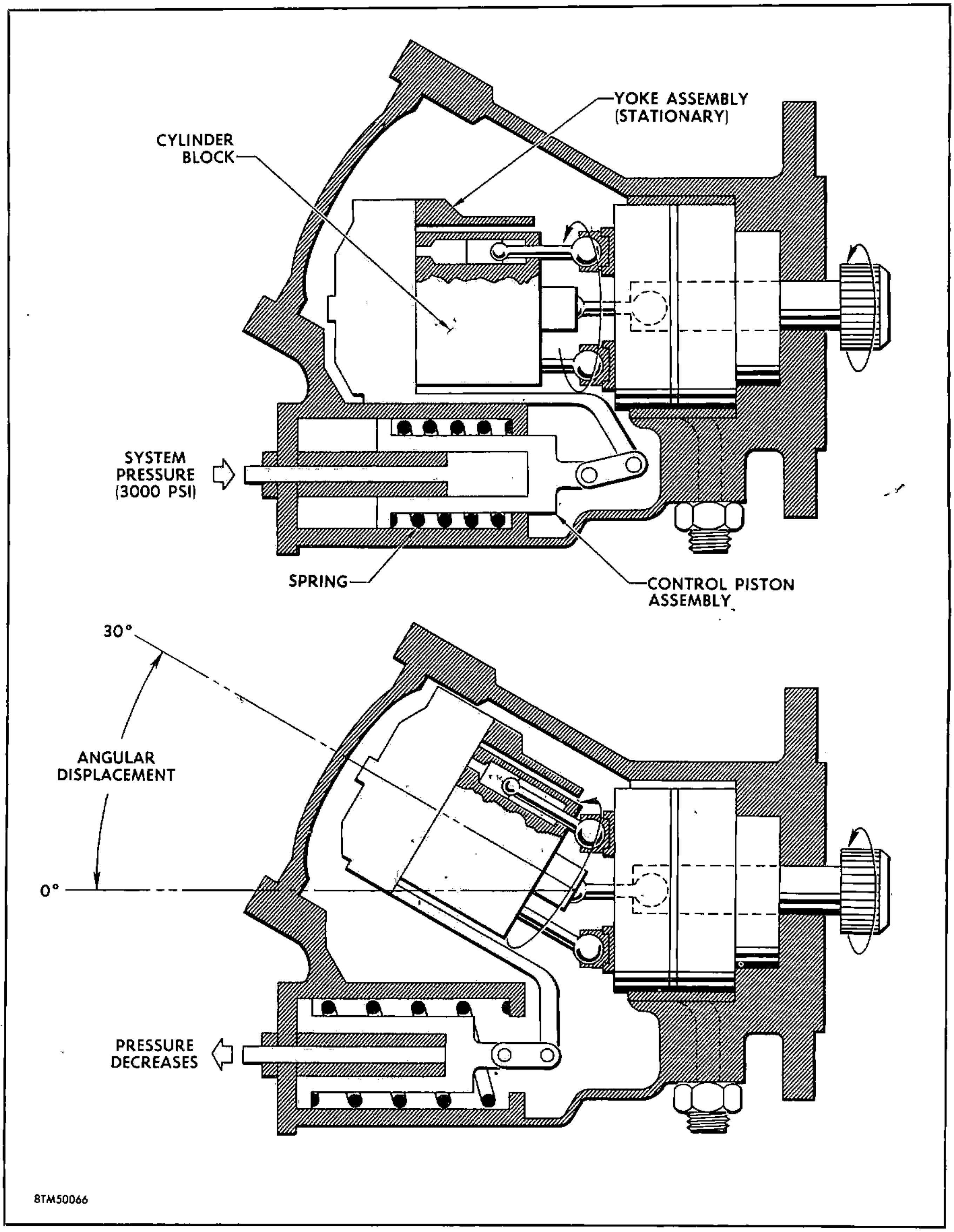

Figure 2-8. Main Hydraulic Pump Cutaway

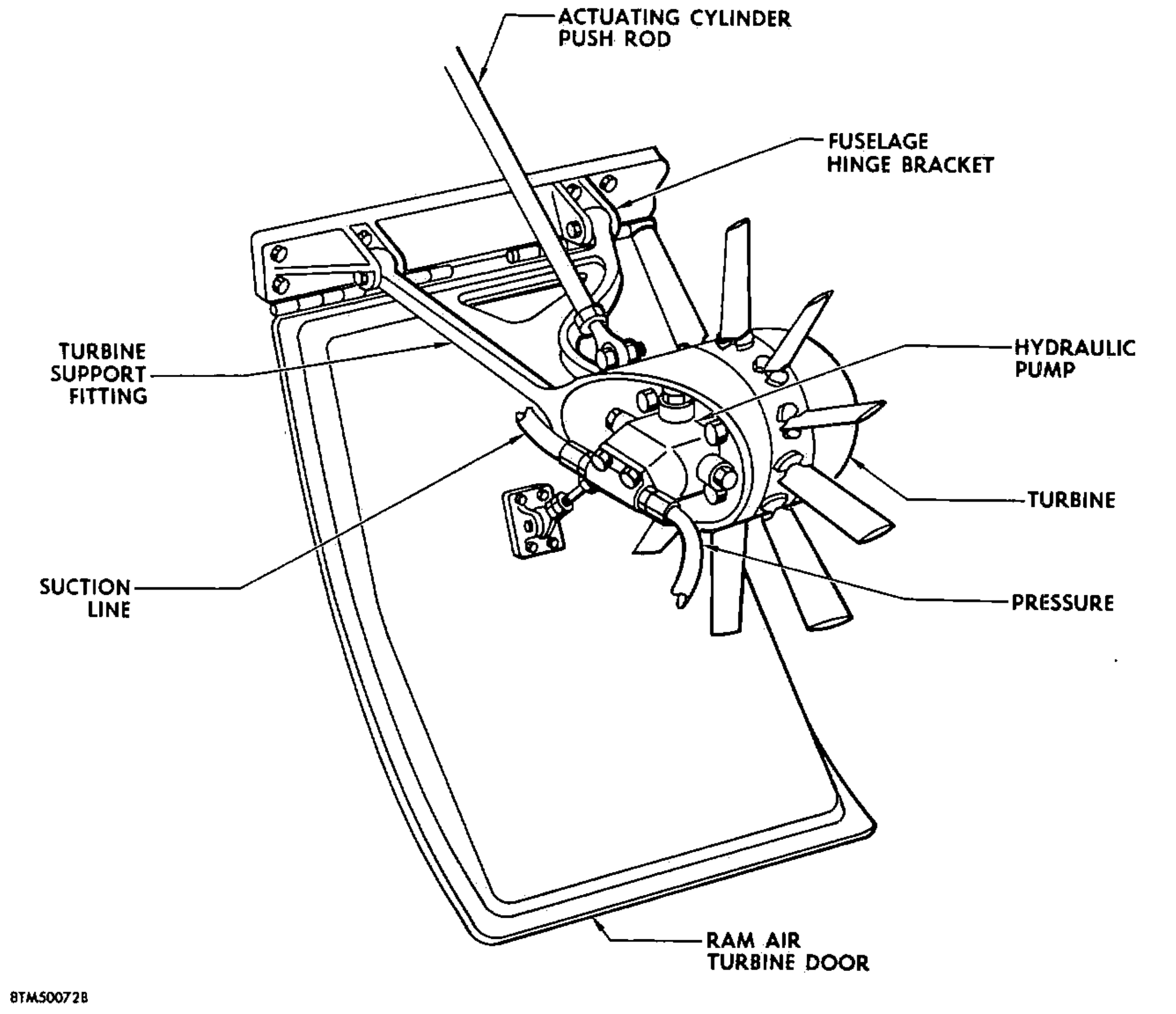

Figure 2-9. Emergency Hydraulic Pump and Ram Air Driven Turbine

However, due to the noise of the jet engine you probably will not hear the cavitation of the pump or hammering of the lines. Instead, you must rely on the other methods to determine if air is in the pump or in the system. If all line fittings between the reservoir and the pump are satisfactory, then you can suspect the pump of causing the trouble. However, before you change the pump, be sure all fittings at the pump are satisfactory. Remember, you can very seldom tell where an internal leak exists by visual observation.

If any leaks, either external or internal, exist in the pump itself, then replace the pump. Other than replacing the mounting gasket, never attempt to take the pump apart to stop a leak. There's a trick to putting the nine pistons in their cylinders and this should be left for overhaul maintenance.

Although shearing of the pump shaft is rare, you may encounter this malfunction occasionally. This trouble would be indicated by a complete loss of system pressure. Again you would replace the pump if this malfunction occurred.

When changing a pump, fill the pump housing with hydraulic fluid to prevent cavitation when the pump first starts up. Remember, both air and dirt are harmful to the operation of hydraulic pumps, and every effort should be made to keep the system free of these two damaging elements.

EMERGENCY HYDRAULIC PUMP.

The emergency hydraulic pump is a constant displacement, piston-type pump—its displacement is constant and does not vary as in the main hydraulic pumps. The pump is attached to, and driven by, the ram-air

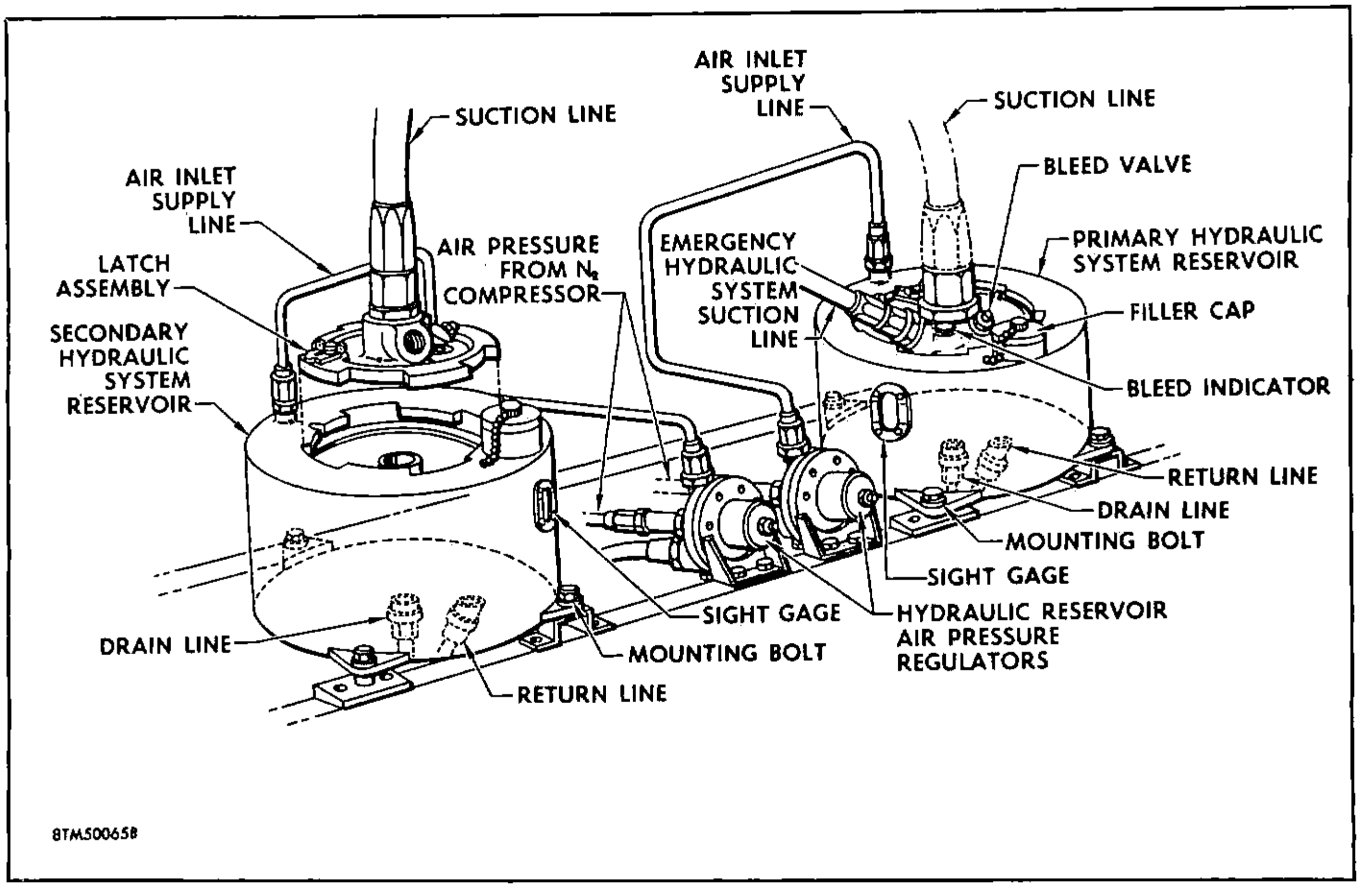

Figure 2-10. Primary and Secondary Hydraulic System Reservoir

turbine when the assembly is extended into the airstream. This pump supplies emergency hydraulic pressure to the primary power system when the pump in the primary system fails. The emergency pump produces hydraulic pressure in flight to control the rudder and elevons. Figure 2-9 shows the emergency pump and turbine in the deployed or operating position.

Note how these two units are attached to the hydraulic compartment door. The support fitting holds the turbine and pump assembly and attaches to the fuselage hinge bracket. The small adjustable rod below the assembly positions the turbine correctly so it will not hit other units in the compartment when the door is closed. At the rear of the turbine you can see the pump and its two hydraulic lines. The suction line connects to the primary reservoir; the pressure line connects to the primary system through the flow control valve. The actuating cylinder push rod shown is part of the pneumatically operated cylinder that opens the door into the airstream. This actuating cylinder is controlled from the cockpit.

How the Emergency Pump Produces Pressure.

The rotation of the ram-air driven turbine turns a geared shaft that connects to the rotor of the pump. This rotor turns the cylinder block in the pump in the same manner as the main hydraulic pump. This rotation causes the nine pistons to move up and down and produce the hydraulic pressure. Since this pump is the fixed displacement type, its cylinder block and piston angle is pre-set to allow maximum output at all times. This means that the strokes of the pistons do not vary as they do in the main pumps, but continually operate at maximum displacement.

Effect of Airspeed on Turbine and Pump Operation.

The turbine that drives the emergency pump consists of ten variable pitch blades and a governor assembly which is attached to the turbine main shaft. The governor assembly is a centrifugally-operated device composed of flyweights, torsion bars, and a gear train which connects the flyweights to the turbine blades. This governor assembly allows for changes in airspeed by changing the pitch of the ten blades. When the ram-air turbine is extended into the airstream, the velocity of the airstream rotates the turbine and main shaft. Since the hydraulic pump is connected to the main shaft, rotation of the turbine drives the pump to produce emergency hydraulic pressure.

An increase in airspeed causes an increase in the rotational speed of the turbine with a resultant increase in the centrifugal force acting on the flyweights in the speed governor. An outward radial movement of the flyweights results and this movement is transmitted to the turbine blades. Outward radial movement of the flyweights rotates the turbine blades to decrease their angle of attack to the airstream. The blade angle of attack decreases until a reduced turbine speed is reached.

Decreased velocity of the airstream results in a decrease in the rotational speed of the turbine. The flyweights in the speed governor then rotate the turbine blades to increase their angle of attack to the airstream, resulting in an increase in turbine speed.

Although the ram-air turbine has a speed governor which maintains its rotation at a near constant speed, the emergency pump output will vary slightly with changes in speed. Fluid pressurized by the emergency pump first passes through an emergency flow control valve before it enters the primary system. This valve is set so that a minimum of 6.3 gallons-per-minute (gpm) is delivered into the primary system at a minimum indicated airspeed of 125 knots.

Immediately upon deployment into the airstream the pump begins to operate, but the flow control valve bypasses its fluid output back to the reservoir until the pump develops a minimum flow of 6.3 gpm. This protects the turbine unit and allows the pump time to build up its operating pressure to a minimum of 300 psi. With an increase in airspeed the pressure output of the pump increases. The following chart shows the approximate pressure produced by the pump at various airspeeds.

INDICATED AIRSPEED	PUMP PRESSURE OUTPUT
125 knots	1000 psi
140 knots	1450 psi
190 knots	2500 psi
220 knots	2600 psi

Maintenance.

The maintenance on the emergency pump is the same as that given on the main hydraulic pumps. If the turbine becomes defective or malfunctions, you should replace the unit just as you would with a malfunctioning pump. Malfunctioning of the turbine would be caused by the speed governor not adjusting the pump speed for changes in airspeed. There is no provision for governor adjustment. Damaged turbine blades would also cause the unit to malfunction so always exercise extreme caution when performing maintenance on the turbine.

PRESSURIZED HYDRAULIC RESERVOIRS.

Each of the primary and secondary hydraulic systems has a pressurized fluid reservoir located in the hydraulic accessory compartment. In figure 2-10 you can see that the two reservoirs are basically identical. The reservoir on the left is the secondary system reservoir and has a slightly larger fluid capacity than the primary reservoir (shown on the right). However, other than size and the emergency system suction line fitting on the primary reservoir, there is practically no difference in the two reservoirs. These two reservoirs are arranged in the hydraulic accessory compartment just as they appear in figure 2-10. Both are accessible for servicing through the hydraulic accessory compartment door.

We are aware that the reservoirs are storage places for the hydraulic fluid not at work in the system. However, in high-pressure, closed-center hydraulic systems like those in the F-102A, the reservoirs must perform several other important tasks. The reservoirs must filter the system fluid and aid the hydraulic pumps in drawing fluid through the suction line so as to prevent cavitation at the pumps. To accomplish these requirements properly, both reservoirs are provided with filters and are pressurized with air.

In the F-102A this air pressure is supplied by the N_2 compressor section of the engine and is regulated to about 50 psi by the two regulators shown between the reservoirs. This pressurization applies a forward pressure head to the fluid leaving the reservoir through the suction line to the pump. With this pressure applied on the fluid in the reservoir, the pump operates more efficiently and cavitation is less likely to occur.

Referring again to figure 2-10, note the filler cap on the top of each reservoir. This is the filler used for servicing the reservoir with hydraulic fluid. A small hole drilled through the internal portion of the filler cap permits the pressurized air in the reservoir to escape when the filler cap is unscrewed approximately one complete turn. Therefore, when the filler cap is unscrewed just slightly, the hole becomes a safety device which protects against a sudden release of high pressure air. Inside the filler opening is a screen that prevents any foreign matter from entering the reservoir during servicing. This screen is accessible for cleaning through the filler opening.

The bleed indicator on top of the reservoir is a transparent plastic disc which has the word BLEED stamped on it. Normally you cannot read this word when the hydraulic fluid in the reservoir is free of air. However, if the fluid in the reservoir collects air and bubbles form, then you will be able to read the BLEED indication as you look down at the small plastic disc. Therefore, whenever you can read the lettering on the indicator disc, the reservoir needs bleeding. To bleed the reservoir of air, unscrew the bleed fitting adjacent to the bleed indicator on top of the reservoir cover.

The reservoir drain and system return lines are shown on the bottom of each reservoir. The drain line from the bottom of the primary system reservoir goes to a capped drain fitting on the aft bulkhead in the center armament bay. The drain line from the secondary system reservoir goes to a capped drain fitting on the forward bulkhead in the right main wheel well. It is through the suction lines on the top of each reservoir that the fluid leaves the reservoirs and flows to the respective pumps.

The primary reservoir on the right also has a suction line to the emergency hydraulic system, but the secondary reservoir on the left does not. This is because the secondary system is not connnected to the emergency system. Other than this emergency system suction line on the primary reservoir and the size of the two reservoirs, there is no difference between the primary reservoir and the secondary reservoir.

Note the sight gage on the side of each reservoir. Although not shown in figure 2-10, two lines labeled FULL and REFILL are inscribed as reference lines for servicing the reservoir with fluid. When the fluid level drops to the REFILL line, you must service the tanks with hydraulic fluid until the level again reaches the line marked FULL.

The cover of the secondary hydraulic system reservoir is shown separated from the reservoir. The cover on both reservoirs can be removed in the same manner. The latch assembly at the left edge of the cover locks the cover to the reservoir. This latch assembly is composed of a movable latch and wing nut. The latch fits into the small recess at the top of the reservoir inner circumference, opposite the air inlet supply line fitting.

When installing the cover, the cover is aligned with the reservoir, then given approximately ⅛ turn to engage the cover in reservoir. The latch then seats in the reservoir recess and is secured by the wing stop nut. Removal of the cover is the reverse of the installation. O-ring seals between the cover and the reservoir seal the reservoir air pressure in and keep foreign matter out.

A large circular low pressure filter element fits down into the reservoir where it completely fills the inner chamber of the reservoir. The cylindrical pipelike object you see in the center of the opened reservoir on the left is the standpipe. The low pressure filter unit fits around this standpipe. The standpipe is a permanent part of the reservoir. The two reservoirs use identical filter elements despite the fact that the secondary system reservoir is slightly larger.

How Reservoirs Are Pressurized.

Bleed air from the N_2 area of the engine is routed through air lines into the top of the reservoirs where it maintains pressure on the hydraulic fluid. The air from the N_2 engine area first passes through air pressure regulators regulating devices which control the pressure in the reservoirs. (See figure 2-10.) Since the pressure of the bleed air from the engine varies in both extremes, you can readily see the need for a regulator to control the air pressure entering the reservoir.

The two regulators are shown between the reservoirs in figure 2-10. Each reservoir has its own air pressure regulator; the regulators are connected to their respective reservoirs by an air inlet supply line. The regulators are pre-adjusted to maintain 50 ± 5 psig of air inside the reservoirs. These regulators are the common diaphragm type regulators and are provided with adjustable screws for variable settings. It is rare that an air regulator will get out of adjustment but should such a case occur, the adjustments must be made with a bench test set-up. Whenever a regulator malfunctions, the operation of the hydraulic pump in the respective system will be affected. So, if the malfunctioning persists, remove the regulator and replace with a new or serviceable unit. Before removing one of these low-pressure regulators the engine must be shut down and the air pressure bled from the reservoir.

Fluid Flow Through The Reservoirs.

The reservoirs have inner and outer chambers which permit pressurization of the reservoir without the danger of air entering the hydraulic system. The entire inner chamber of the reservoir (figure 2-11) is a low-pressure fluid filter while the outer chamber contains the bulk of the hydraulic fluid as well as the pressurized air. Pressurized fluid passes from the outer chamber and enters the inner chamber at the bottom of the tank through a full length standpipe.

A close look at the schematic view of the reservoir shows that this standpipe has two separate fluid passages—one inside the other. The outer tube of the standpipe routes fluid from the outer chamber to the filter area in the inner chamber of the reservoir. The inner tube routes the fluid from the inner chamber to the pump. The standpipe has the same characteristics as a waterfall; the fluid flows in through the outer tube, overflows the top of the standpipe and into the filter. Fluid flows out of the reservoir through the inner tube feed line of the standpipe. Any foreign material in the fluid is trapped and retained in the inner chamber filtering element.

In the lower half of figure 2-11 you can see the flow pattern of the standpipe. The fluid from the outer chamber enters the standpipe at (1) then flows up and around the inner tube (2 and 3) and overflows at (4). By comparing this standpipe schematic with the schematic of the reservoir you see that point (5) is actually not in the standpipe, but in the filter area of the reservoir. This means that only filtered fluid which has overflowed at point (4) and circulated through area (5) can reach the outlets at (6). Through these outlets the fluid enters the inner tube (7) and flows out of the reservoir to the pump, and on through the system for usage.

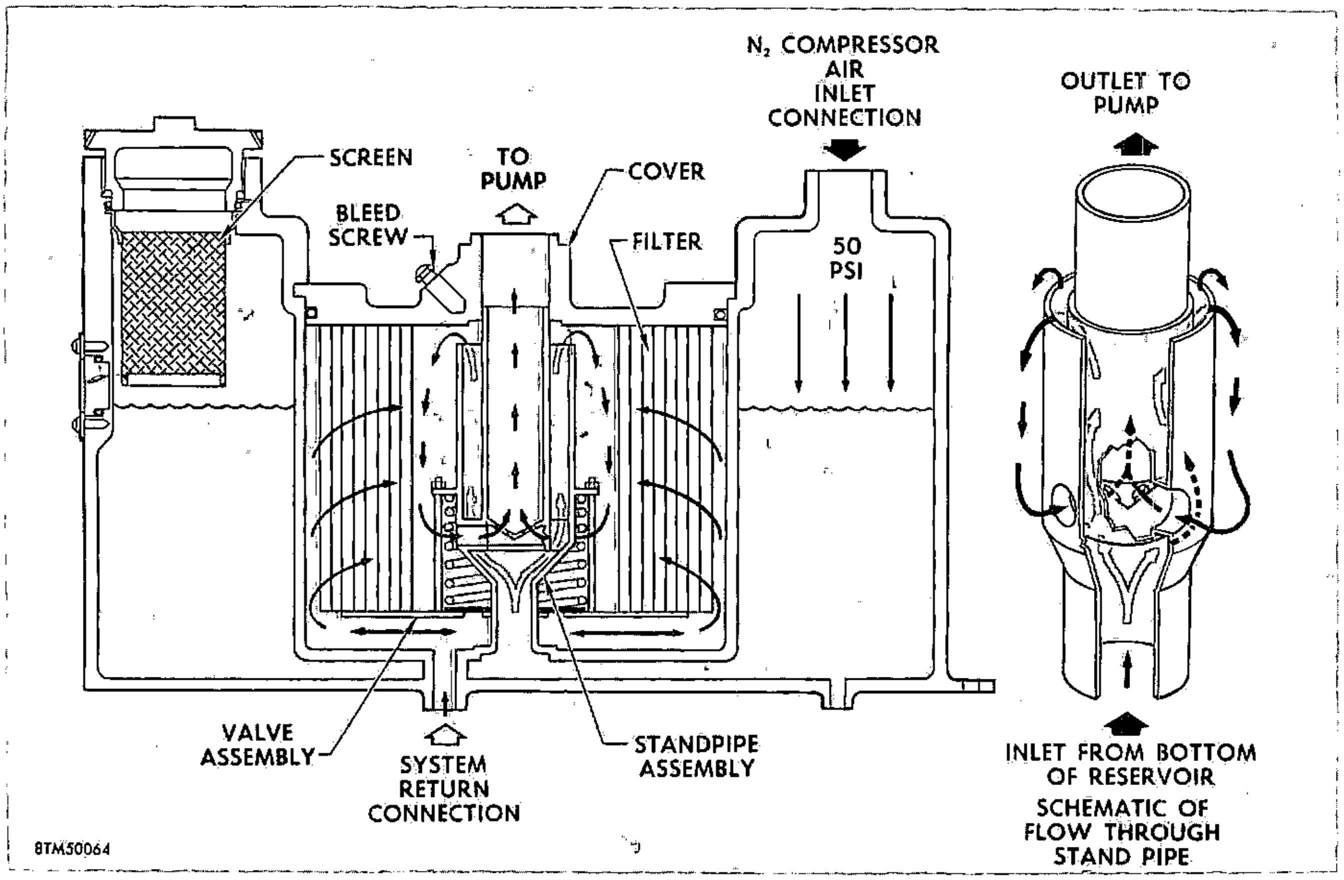

Figure 2-11. Hydraulic Reservoir Cutaway

HIGH-PRESSURE FILTERS.

Two high-pressure filters collect any foreign matter that may be in the fluid after it leaves the main engine pumps. The action of these filters protect the many delicate mechanisms that are a part of the hydraulic system. These two filters are installed in the hydraulic system downstream from the main engine pumps and upstream of the system accumulators. The primary system filter unit is accessible through the right engine accessory compartment door, while the secondary system filter is accessible through the left engine accessory compartment door.

Figure 2-12 shows the components of a filter unit on the left and two schematics of the flow path through the filter on the right. Note that the all-metal filter housing acts as the container for the filter element. This filtering element is made of a specially treated parchment-type cellulose paper. This paper is so made that solid impurities greater than 20 microns (about 0.000394 inch in diameter) are unable to pass through it. The filter housing screws into the filter head and is hermetically sealed by the two seals. The filter head contains both an inlet and an outlet port marked IN and OUT on the filter head casing.

High-pressure hydraulic fluid enters the inlet port of the filter head and leaves through the outlet port. The two fluid ports are marked along with a directional arrow to avoid the mistake of installing the filter unit backwards in the system. The filter will not operate properly when installed backwards, and could possibly cause severe damage to the system. If the filter is installed backwards the emergency bypass will not be able to function if the filter should become clogged.

In the schematic showing the normal flow, fluid enters through the filtering element from the outside to the center, and then out the outlet port on the left. The spring-loaded bypass valve in the filter head is held in the closed position because the free flowing fluid has no back pressure.

The high-pressure filter has an integral relief valve within the head of the casing. This pre-adjusted valve, shown in the two schematics, (figure 2-12) opens and allows fluid to bypass the filtering element when the element becomes clogged. The fluid is bypassed when the difference in pressure between the filter inlet port and outlet port exceeds 50 psi; this avoids a fluid stoppage to the remainder of the system. In the right-hand view of the illustration, you can see the bypass flow in operation.

Here a clogged filter is stopping or restricting normal flow through the filtering element to the outlet port.

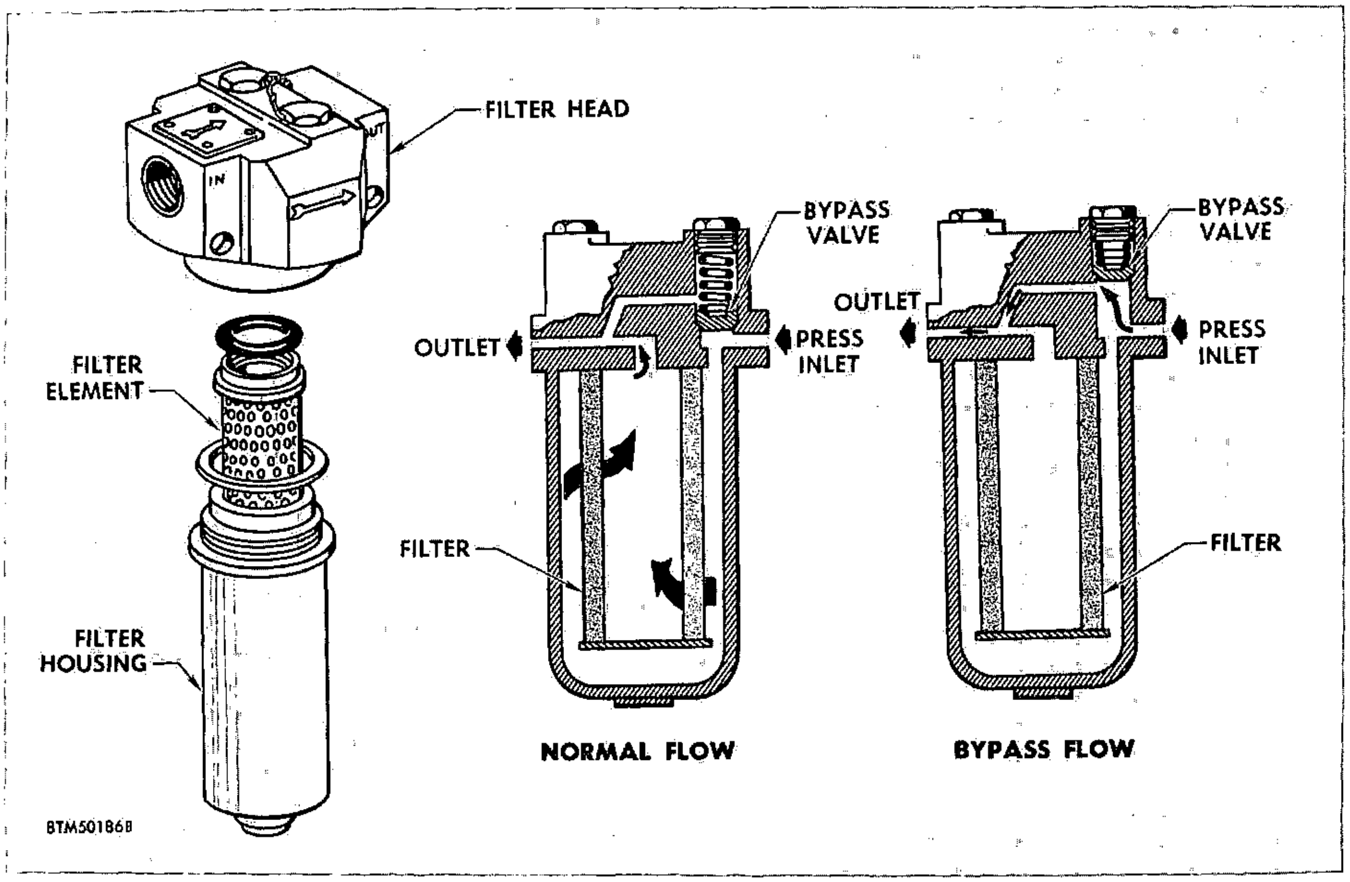

Figure 2-12. High Pressure Filter

The pressure backs up until it overcomes the spring-loaded bypass valve thus forcing the valve to open and allow fluid to bypass. The fluid which bypasses through the filter to the outlet is not filtered but instead permits the system to continue its operation.

How The Filter Element Is Replaced.

Before attempting to remove a filtering element from the housing of a high-pressure filter, be sure that all hydraulic system pressure is released. After removing the safety wire that secures the filter housing to the head, the housing can be unscrewed for access to the element as shown in figure 2-12. After the filter element is removed, the housing should be cleaned with a dry cleaning solvent as specified in your F-102A Handbook of Maintenance Instructions, 1F-102A-2-3.

Since the filter element cannot be cleaned, you must install a new element. New seals are also necessary when replacing the filter housing. When installing the element and housing in the filter head always check to insure that the seals remain in their retainer. Be sure to replace the safety lockwire on the filter head; then with full system pressure, check for leaks to determine that your installation is satisfactory.

ACCUMULATORS.

Because of the use of two piston-type accumulators, the F-102A hydraulic systems have a very quick response to fluid pressure demands by the subsystems. Figure 2-13 shows the installation of these accumulators as they appear in the hydraulic accessory compartment. There is an accumulator for each hydraulic system, and each accumulator operates independently of the other. The line at the top of each accumulator connects that accumulator with its respective hydraulic system—the left accumulator to the secondary system and the right accumulator to the primary system. The lines at the bottom of the accumulaors connect to their respective pressure gages and air charging valves (see figure 2-1).

The pressure gages are pneumatic instruments that measure pressure from 750 to 3000 psi. The gages are mounted on a bracket with the charging valves and are readily accessible from the ground for checking accumulator pressure. The charging valves positioned behind the gages make it convenient for you to charge and check the accumulators in one position. The 750 psi indication shows when the engine hydraulic pumps are operating. Whenever you charge or discharge the accumulators you must follow the special instructions in your F-102A Handbook of Maintenance Instructions, T. O. 1F-102A-2-3. Pushing on

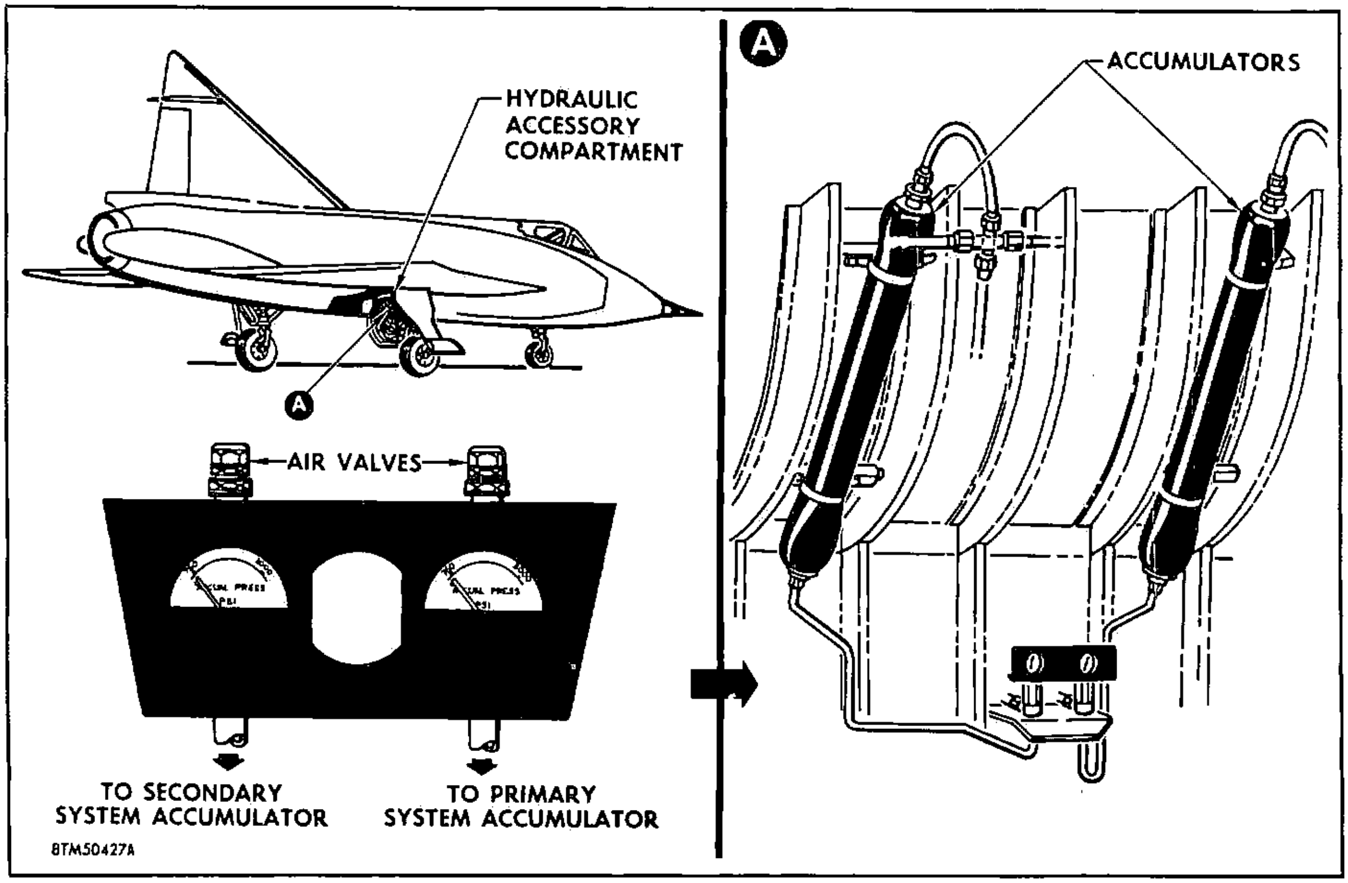

Figure 2-13. F-102A Hydraulic Accumulators

the valve core or loosening the valve assembly in its boss (fitting) must NEVER be done to discharge air. Follow your Maintenance Handbook instructions.

Accumulator Operation.

The operational schematic (figure 2-14) shows a typical accumulator of the F-102A. The lower portion of the accumulator is the nitrogen or air chamber, while the upper portion is the fluid chamber. A sealed but movable piston divides the gas and fluid chambers. When the hydraulic system pressure is at 0 psi, the 750 psi air precharge pushes the floating piston against the top of the accumulator. However, when the hydraulic pump is operating, fluid pressure pushes the piston down. Since air is compressible and fluid is not, the piston will move down until the pressure of the air in the nitrogen chamber equals that in the fluid chamber. When the system demands pressure, it takes it from the fluid chamber. This causes the higher pressure in the nitrogen chamber to push the piston up. Thus you can see that any pressure changes, such as pump surging or sudden system demands, are absorbed in the accumulator. The accumulator therefore serves as a shock absorber and a source of reserve pressure.

HYDRAULIC PRESSURE-RELIEF VALVE.

A high-pressure-relief valve is installed downstream of each accumulator. These hydraulic pressure-relief valves are the conventional, spring-loaded type that operate in the same manner as the relief valve described in Chapter I. It is often called a safety valve since it has no function in the system except to protect the system against excessive pressure. The valve begins to crack open when system pressure reaches 3250 psi, and continues opening until it is fully open at 3700 psi. All excess pressure is released from the system by this relief valve and is routed back to the reservoir through the system return lines. When normal pressure is again restored, the valve closes and thereafter it has no function until another over-pressure condition occurs in the system.

The pressure-relief valve has an adjustment screw which is used to vary the cracking pressure in the valve. The hazards of having an improperly adjusted relief valve make it important that only valves which have been previously adjusted on a bench test setup should be used. Do not attempt to change the setting of a relief valve in the airplane. When you detect a relief valve that opens at either too low or too high a pressure, replace it with a new valve that checks out satisfactorily on a hydraulic test stand.

FLOW CONTROL VALVE.

A flow control valve, which operates as a variable pressure relief valve, is installed in the pressure line between the emergency ram-air driven hydraulic pump and the primary system (as shown in the system installation at the beginning of this chapter). The valve has no action that makes it a "low-pressure" relief valve. Pressure from the emergency pump that is below a certain pre-set minimum is relieved by the valve and routed back to the primary reservoir. The reason for this valve arrangement is to protect the pump from overloads and the dangers of excessively *high starting torque.*

NOTE

Torque is the force required to twist or turn an object around an axis. By that we see that the starting torque of a rotating pump is the force required to actually get the pump rotating mechanism to turn. If the hydraulic pressure line from the emergency pump were connected into the primary hydraulic system directly, the initial force required to produce pressure would overcome the efforts of the ram-air turbine and stall the emergency pump. The flow control valve keeps the emergency pump pressure from entering the system until this pressure builds up sufficiently to work in the system without the danger of a stall-out. When the ram-air turbine is deployed into the airstream, it takes a few moments for the emergency pump pressure output to build up. It is in this short time that the flow control valve functions to relieve the early build-up of fluid pressure back to the reservoir.

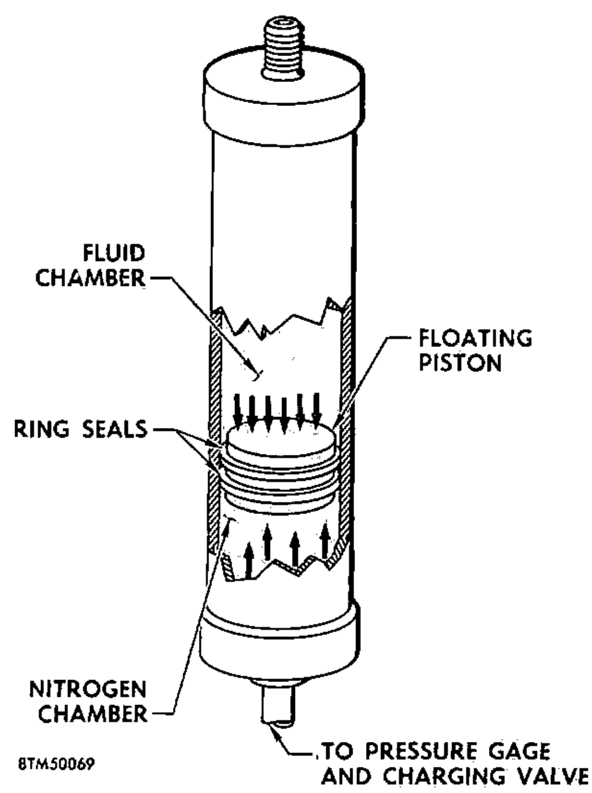

Figure 2-14. Cutway of Hydraulic Accumulator

Flow Control Valve Operation.

Figure 2-15 shows an operational schematic of the flow control valve in two positions—one in which initial pump pressure returns to the reservoir, and one in which pump operating pressure is diverted to the primary system. Note that this valve consists of a housing, sleeve, spool, two springs, and three fluid ports. Before any flow from the emergency pump enters the inlet port of the valve, both sleeve and spool are at the extreme right as shown in the upper view of the illustration. In this position the relief orifices in the sleeve and spool are aligned.

The initial flow from the pump enters the inlet port at the right end and passes through the holes in the sleeve. Since static pressure between this valve and the check valve in the primary system is at the lower port marked FROM SYSTEM, inlet flow follows the path of least resistance and flows through the aligned relief orifices. From the inside of the spool the fluid can flow out the line to return.

While initial flow is passing through the valve, some of this pressure is applied to the right end of the spool. This pressure tends to misalign the relief orifices thereby reducing the outlet flow to return. However, as you remember the starting torque for the emergency pump must be as low as possible; so as pump pressure builds up, the flow sensitive orifice restricts this pressure causing the sleeve to move to the left along with the spool. When the inlet flow reaches 6.3 gallons per minute (gpm), the sleeve and spool are positioned as in the lower view of figure 2-15.

In this view you can see that the sleeve and spool have moved to the left—the sleeve is against its housing stops, while the spool is unseated from its valve housing stops. At this position the relief orifices are completely misaligned and no fluid can pass through to return. All inlet fluid pressure is now directed through the lower port and into the primary system. The sleeve and spool relief orifices will remain misaligned as long as the emergency pump output is 6.3 gpm or more. If inlet flow drops below 6.3 gpm, the springs will expand and return the sleeve and spool toward their original (no flow) position.

As is the case in most other units in the hydraulic system, you should replace this valve if it goes bad.

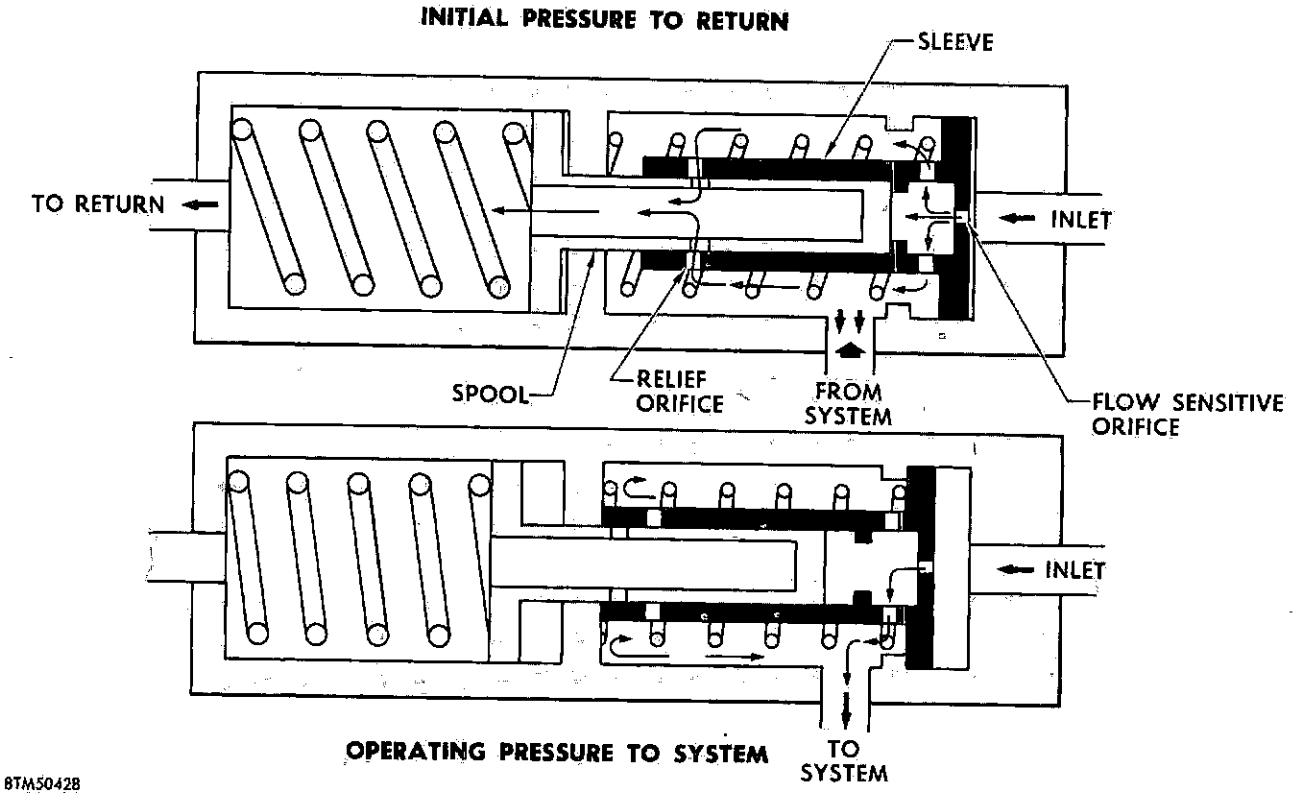

Figure 2-15. Flow Control Valve Operational Schematic

Actually this valve should give very little trouble; but if it should, malfunctioning will be confined either to restricted openings within the valve or weak spring tension. Restricted openings, probably caused by foreign material, will reduce emergency system operating pressure. Weak spring tension, although not likely to occur, would permit pump pressure to enter the primary system before the flow reached 6.3 gpm. This may cause the emergency pump to stall out before it reaches operating pressure.

QUICK-DISCONNECT COUPLINGS.

Quick-disconnect couplings are provided in the F-102A hydraulic system fluid and pneumatic lines as means for connecting the hydraulic test stand. These couplings are of the self-sealing type. They consist of two self-sealing halves—one which is permanently installed in the aircraft structure and the other which is a detachable screw-in section attached to a flexible hose. A fluid or pneumatic connection is made when the detachable section is coupled to the fixed section on the airplane. However, when uncoupled both sections automatically seal themselves. This self-sealing action prevents leakage of system fluid that ordinarily would pour out through an opening.

There are five quick-disconnect fittings to which an outside source can be connected in the F-102A systems. There are two couplings at each of the two main engine hydraulic pumps, and one at the low pressure pneumatic coupling through which the hydraulic reservoir pressurization system receives air.

Figure 2-16 shows a typical quick-disconnect coupling, such as those used on the F-102A. In the upper section are the two self-sealing sections as they appear when disconnected, while the lower view shows these sections coupled correctly. No tools are required to connect a coupling—they are merely hand-tightened by turning the union nut until the lock teeth engage the lock spring assembly. When correctly assembled you will have the 1/16 inch minimum gap shown in figure 2-16 and the back side of the union nut will be flush with the flange on the valve.

HYDRAULIC PRESSURE INDICATING SYSTEM.

The hydraulic pressure indicating system for the F-102A is a simple, but highly precision-operated system. It consists of a pressure gage, two pressure transmitters with snubbers, and an indicator selector switch. Electrical power for the system is supplied by the 26-volt, single-phase, 400-cycle a-c bus. The system indicates pressure for either of the two hydraulic power systems by the pilot manually actuating the selector switch. The switch completes a circuit that permits indication of primary system pressure when selected in one direction, and indication of secondary system pressure when selected in the opposite direction. One pressure gage indicates pressure for both systems even though each system has individual transmitters and snubber units.

Each transmitter has a hydraulic pressure connection and a vent opening. These openings are differentiated by the lettering P and V on the transmitter case. The diagram of the indicating system (figure 2-18) shows that the electrical indicating circuit is common for both hydraulic systems, although no connection or interchange of system pressure is made. (Also see figure 2-1 for details.)

PRESSURE TRANSMITTER AND SNUBBER OPERATION.

The hydraulic pressure transmitters relay pressure signals to the pressure indicating instrument on the instrument panel. Each transmitter is a hydraulically-actuated mechanical device that generates an electrical signal when actuated by fluid pressure. The transmitter unit consists basically of a Bourdon-tube, an electrical transducer, and an adjusting arm mechanism.

The Bourdon-tube operates in such a way that a change in system pressure opens or closes the arc of the tube in much the same manner as the basic one you learned about in Chapter I. In this transmitter the change in the Bourdon-tube arc is the mechanical motion that is transformed into electrical energy and sent as a signal to the remote controlled indicator gage on the instrument panel. In this way a pressure change in the system is made known by the pointer movement on the dial of the indicator gage.

In figure 2-17 is a cutaway view of one of the hydraulic pressure transmitters. Hydraulic pressure enters the transmitter through the snubber and continues on through the capillary tube to the Bourdon-tube. In figure 2-17 you can easily visualize the adjustment arm moving as a result of the expanding and contracting motion of the Bourdon-tube. This is exactly what happens. The change in pressure inside the Bourdon-tube creates the movements the tube makes. All movements of the Bourdon-tube are relayed through the adjustment arm to the electrical transducer. This electrical transducer transmits electrical signals to the pressure gage in a *direct ratio* to the magnitude of the Bourdon-tube movements that are sensed and relayed by the adjustment arm.

The snubber valve shown in figure 2-17 connects the transmitter to the hydraulic pressure source. This snubber valve contains a flow-limiting fluid opening that has two filtering screens—one on each side of the orifice. The use of a snubber protects the transmitter from pressure surges. You can see how surges would cause a sensitive Bourdon-tube type transmitter to operate in an oscillating manner and make reading of the pressure gage difficult. If the filtering screens in the snubber valves become restricted, they must be removed and cleaned. A suitable dry cleaning solvent is the recommended agent for cleaning the snubber orifice and screens. On the other hand, no flight line maintenance should be performed on the pressure transmitters other than inspecting, removing, and replacing of the entire assembly.

THE HYDRAULIC PRESSURE GAGE.

The hydraulic pressure gage provides indication for both the primary and secondary hydraulic system pressure. Pressure gage readings are obtained by placing the gage selector switch in the desired system position, then observing the movement of the pointer on the gage. The pressure gage is adjacent to the pressure selector switch and both are on the right console in the cockpit.

The operation of the pressure gage is similar to the one shown in the simple indicating system (figure 1-23). Pressure induced electrical signals are remoted from the transmitters through the selector switch to the gage. When the switch is moved to one of the two indicating positions, the circuit is completed and the remoted electrical signals are then able to energize the gage. The magnitude of the electrical signals determines the strength of the induced electro magnetic power which moves the pointer that you see in this gage.

HOW SYSTEM INDICATES PRIMARY AND SECONDARY HYDRAULIC PRESSURE.

Figure 2-18 shows the location of the hydraulic pressure indicating system units in the F-102A airplane and a wiring schematic of the system. In the schematic, note that the hydraulic fluid itself does not come in contact with the indicating unit. However, the electrical circuit is controlled by the accurate transformation of hydraulic pressure into a corresponding electrical potential through the use of the pressure transmitter. Figure 2-18 also shows the gage and indicator selector switch as they appear on the cockpit console.

In the wiring diagram, (figure 2-18) locate the 26-volt a-c essential bus. Note that the power from this bus is directed to terminal B of all three components, and that each unit goes to ground by way of terminal A. You can see that actuation of the selector switch to either the *primary* or *secondary* position connects the transmitter on that particular side of the switch to the pressure gage and completes the indicating circuit.

With the switch actuated to either position, transmitter pressure signals are sent to the D and C terminals at the pressure gage. Now that the circuit is completed the changes in system pressure that are sensed and transmitted by the transmitter are sent to the pressure gage where these signals move the pointer in the gage. Thus, you can see that the pressure indication gage serves both systems, but is able to indicate the pressure of only one system at a time.

HYDRAULIC LOW-PRESSURE WARNING SYSTEM.

A warning system is included in the hydraulic system to warn the pilot of critically low hydraulic pressure. If the hydraulic pressure in either system drops to 800 psi or lower, a light on the instrument panel flashes a warning to the pilot. He then checks his pressure gage to determine which system is low. If both systems go out, the light glows steadily; thus, this warning system monitors both hydraulic systems at the same time. The warning light goes out when the pressure again increases to about 1000 psi, or when the pilot *resets* the warning light.

The warning system utilizes two hydraulic pressure switches, a flasher unit, a pressure test switch, and a warning light. The pressure switches, which are located in the hydraulic accessory compartment, complete the circuit from the two hydraulic systems to the instrument panel warning lights. The two switches are hydraulically actuated and work in conjunction with each other to illuminate the warning light when low-pressure causes them to close the electrical circuit to the lights.

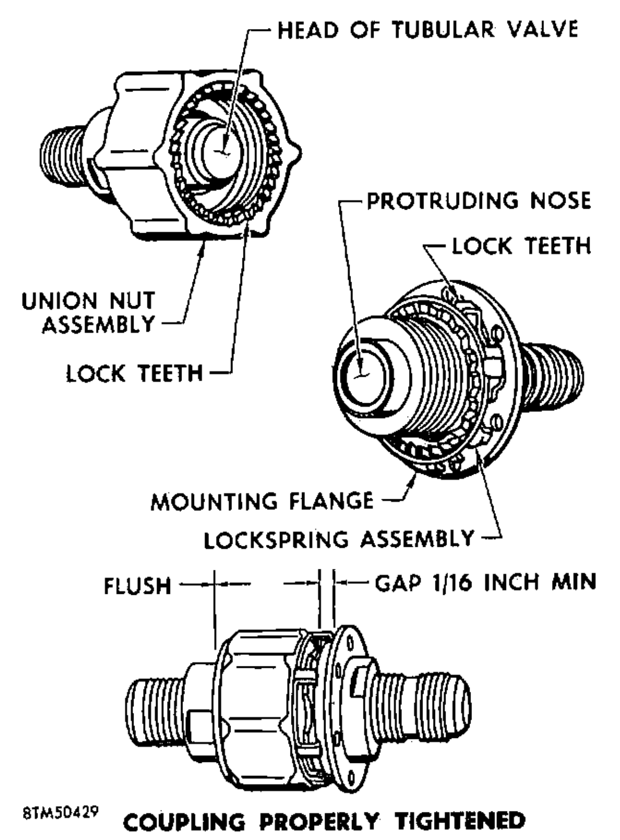

Figure 2-16. Quick-Disconnect Couplings

Each hydraulic pressure switch has contacts that when closed complete a 28-volt, d-c electrical circuit to the hydraulic pressure warning light. The contacts in either switch close whenever the pressure in that system drops to 800 psi.

No attempt should be made to disassemble or adjust these pressure switches since they are factory sealed and adjusted. However, if either switch malfunctions and gives false indication, then replace the entire switch. When the portable hydraulic test stand is used during ground checkout operations, you can check the warning light system to see that it operates properly.

HOW THE WARNING SYSTEM OPERATES.

Figure 2-19 shows the complete low-pressure warning circuit for both primary and secondary hydraulic systems. Note the two hydraulic pressure switches—one for the primary system and one for the secondary system. Either switch can complete the warning circuit to illuminate the warning lights. Although two warning lights are shown in figure 2-19, they are behind one reflector on the instrument panel. You will note that the lights are wired in parallel so that if one burns out, the other will continue to give warning indication. This is a safety feature in the warning system. Now let's see how this warning system works.

Both pressure switches are shown actuated. This is the condition they would be in if the pressure in both hydraulic systems were 800 psi or less. Note the spring in each switch. These springs are compressed by system pressure which holds the electrical contacts *open* when system pressure is above 800 psi; thus the electrical contacts are open. However, the spring tension is strong enough to overcome hydraulic pressure of 800 psi or less; thus they move the contacts to the *closed* position.

First let's see how the warning system functions when the pressure in one system is low and the pressure in the other is satisfactory. Assume that the primary system is below 800 psi and the secondary system is at 3000 psi. In this case the primary pressure switch would be as shown. The two switches D and A in the secondary pressure switch would be at terminals F and C respectively instead of the position in which they are shown. Power from the 28-volt d-c bus enters the primary pressure switch at terminal A, leaves at terminal B and goes to terminal B at the flasher unit.

This same power goes to terminal A on the secondary switch; but since this switch is open at terminal C as we assumed above, the circuit is incomplete. From the

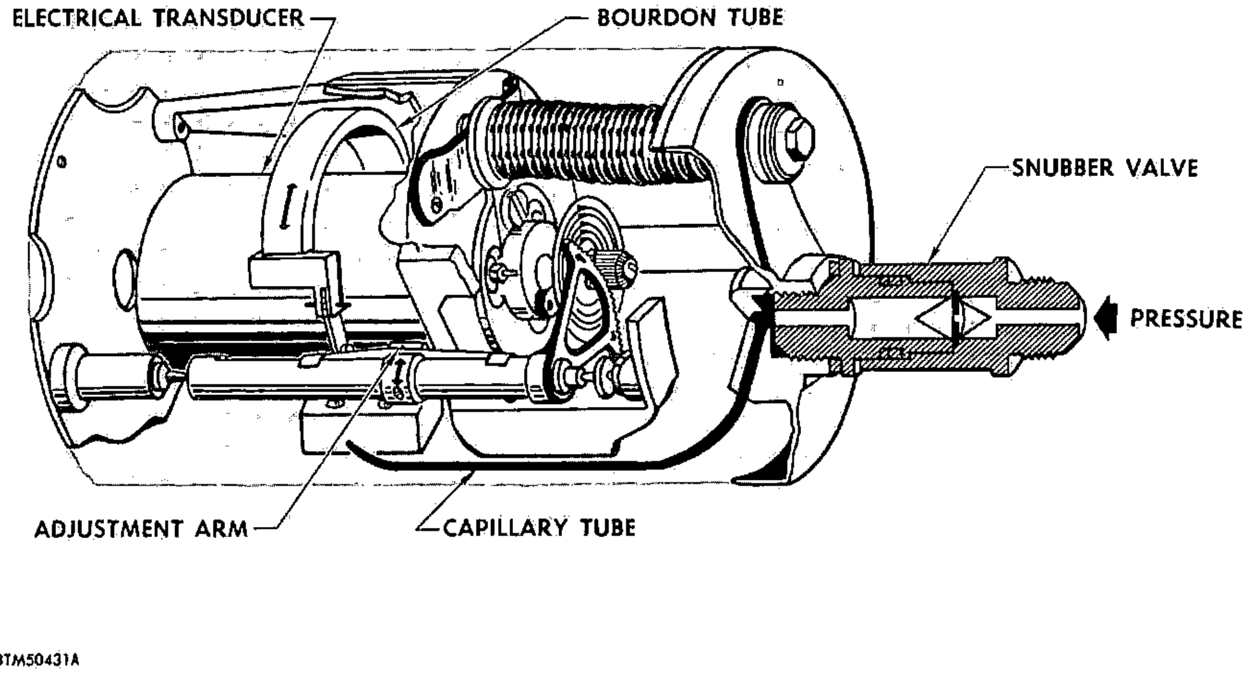

Figure 2-17. Hydraulic Pressure Transmitter and Snubber

flasher unit the circuit is completed through the resistor to the lights causing them to flash on and off. At the resistor you will note that the circuit is connected to the master warning system. We will discuss this phase later.

With the flashing of the warning light, the pilot checks his hydraulic pressure gage to determine which system is low. Then, so he won't have the flashing light to distract his attention, he presses the pressure switch to RESET. This completes a circuit from the 28-volt d-c bus to the relay in the lower left corner of the schematic (figure 2-19) and energizes it, moving the contact to the other post. This breaks the circuit to the flasher unit, the flashing lights go off, and the circuit through the pressure switch now energizes the relay even after the pilot releases the RESET switch. The lights will not illuminate again until a low-pressure indication is received from the secondary system.

You will also notice that power entered the primary pressure switch at terminal D and went through terminal E to terminal D in the secondary pressure switch; but since this switch was open, power does not leave the switch.

Now let's assume that pressure in both systems is below 880 psi as shown in the schematic. The circuit through the primary pressure switch will remain the same, but now power from terminal E in the primary switch will pass through terminals D and E in the secondary switch. This power from terminal E will bypass the flasher unit and the circuit will be complete direct to the lights, causing them to illuminate in a steady condition.

Referring back to the resistor and its connection to the master warning system, three conditions can occur. If the instrument panel lights are on, the warning circuit will be completed through the resistor shown, and illuminate dimly. If the panel lights are off, as in daylight hours, or the thunderstorm lights are on, the warning circuit will bypass the resistor and be completed through the master warning system. This will cause the lights to illuminate brightly. All this does is control the brightness of the warning lights to agree with light conditions in the cockpit.

The test position on the pressure switch provides a means of checking the system to determine that the lights and flasher unit are satisfactory. Placing the switch in TEST causes the lights to flash on and off.

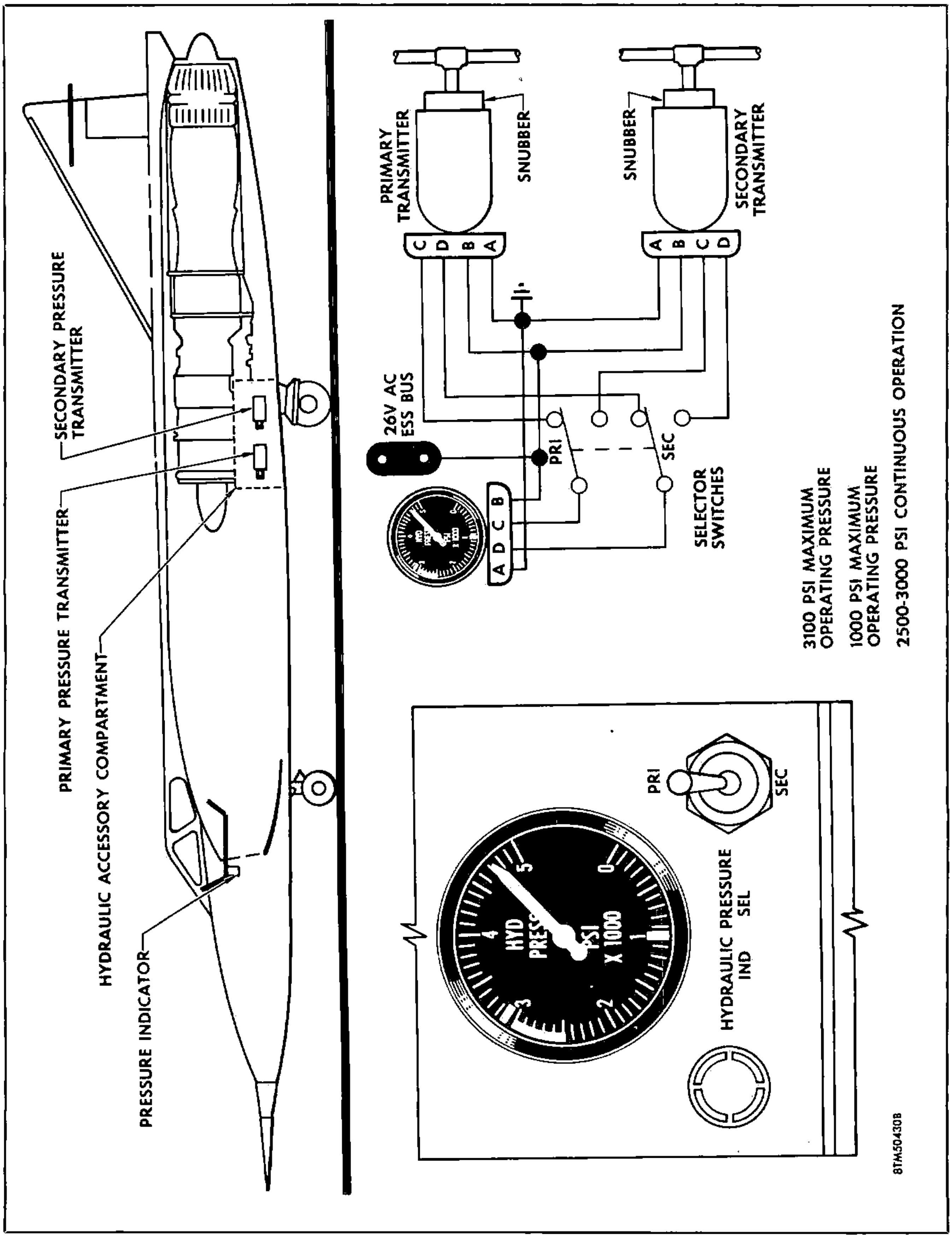

Figure 2-18. Pressure Indicating System Schematic

SERVICING THE F-102A HYDRAULIC POWER SYSTEMS.

The flight line service requirements of the F-102A hydraulic system are considerably few when compared to the large number of operating components in the entire hydraulic system. Hydraulic cylinders, valves, and gages need very little servicing, but they must be replaced after they have been in use for a prescribed number of operating hours.

As the hydraulic fluid flows through the system, all of the moving parts in the pumps, selector valves, actuators, etc., are kept lubricated by the hydraulic fluid. For this reason it is only necessary to check the hydraulic fluid level in the reservoirs periodically to insure that sufficient fluid is in the system at all times. However, it is not necessary to bleed the system unless you find that it contains air. For details and specific inspection periods refer to T.O. 1F-102A-2-3 and T.O. 1F-102A-6.

SERVICING THE RESERVOIR.

To you, the ground checkout and maintenance man, the most common component in the system is the reservoir because you will service it more often than any other component of the hydraulic system. At the reservoir you bleed and refill the system, check fluid levels, and change the filter. The reservoir filler and cap assembly, the bleed indicator and bleed valve, and the fluid outlets are on top of the reservoir. A wire mesh strainer in the filler neck prevents solid foreign matter from entering the reservoir during servicing. The return lines are connected to the bottom of the reservoirs near the point where welds hold the reservoir mounting bracket to the frame of the airplane.

How Reservoirs Are Replenished.

The most common servicing item on the airplane is the replenishing of the hydraulic reservoir. Under normal operating conditions the reservoir is pressurized with air from the low-pressure pneumatic system. This means that the engine must be off and the residual air pressure bled from the reservoir. The filler cap is constructed in such a way that unscrewing this cap about one full turn exposes an air release opening in the side of the cap sufficiently to allow reservoir air pressure to vent. The hissing sound of the air escaping is clearly heard as long as pressurized air vents from the reservoir. After all the air is released, it is safe for you to remove the reservoir filler cap and add fluid to the reservoir to replenish the fluid. The sight gage shows the fluid level in the reservoir. The level of the red hydraulic fluid, Specification MIL-O-5606, is easily seen through this transparent sight gage window on the side of the reservoirs.

Draining The Reservoirs.

Whenever you must drain the reservoirs, either to work on the system or to change the system fluid, then drain the fluid through the drain lines at the bottom of the reservoirs. Since the drain lines at the bottom of the reservoirs are not accessible, a drain fitting is provided for each reservoir. The drain fitting for the primary reservoir is on the bottom of the aft bulkhead in the center armament bay, and the drain fitting for the secondary reservoir is at the bottom of the forward bulkhead in the right main wheel well. However, before you remove the drain fitting caps you must first vent the air pressure from the reservoirs. This prevents the fluid from draining under pressure. To drain the reservoir, remove the drain fitting cap, permit the fluid to drain into a container of at least two gallon capacity, then replace and tighten the fitting cap.

If any hydraulic fluid spills in or near the airplane, always *remove it immediately* and wipe the area clean. Spilled hydraulic fluid, in addition to damaging the airplane equipment, is a fire hazard. A spark or an arc from electrical equipment can ignite hydraulic fluid. While the reservoirs are drained of fluid, you may wish to replace the low-pressure filter in each reservoir. The covers of the reservoirs are removable and permit easy replacement of these filter units. To replace the filter, loosen the small wing nut near the edge of the circular reservoir cover, and remove the latch from the recess in the reservoir.

To take the cover off, turn it about one-eighth turn and raise it up and off of the reservoir. Once the cover is taken off, the filter unit is merely removed and replaced. Replace the cover by inserting it in place, turning it until locked; then position the latch in the recess and secure it with the wing nut.

SERVICING THE ACCUMULATORS.

The accumulators have very important functions in the overall efficiency and operation of the hydraulic systems. Next to the reservoirs the accumulators require more attention and servicing than any other component in the system. Each system has an identical accumulator arrangement; therefore their servicing problems are also the same. Both accumulators must be precharged and discharged as required.

Charging an Accumulator.

The precharge pressure of the nitrogen gas in the pneumatic end of the accumulators is 750($\pm$25) psi when the hydraulic system is not pressurized. Although the system may not be in operation, some hydraulic pressure may remain in the accumulator. Therefore, to assure yourself that the accumulators are exhausted of hydraulic pressure, operate the controls until all pressure is dissipated. You can then charge the accumulators with nitrogen. Nitrogen is the only gas that may be used in the accumulators.

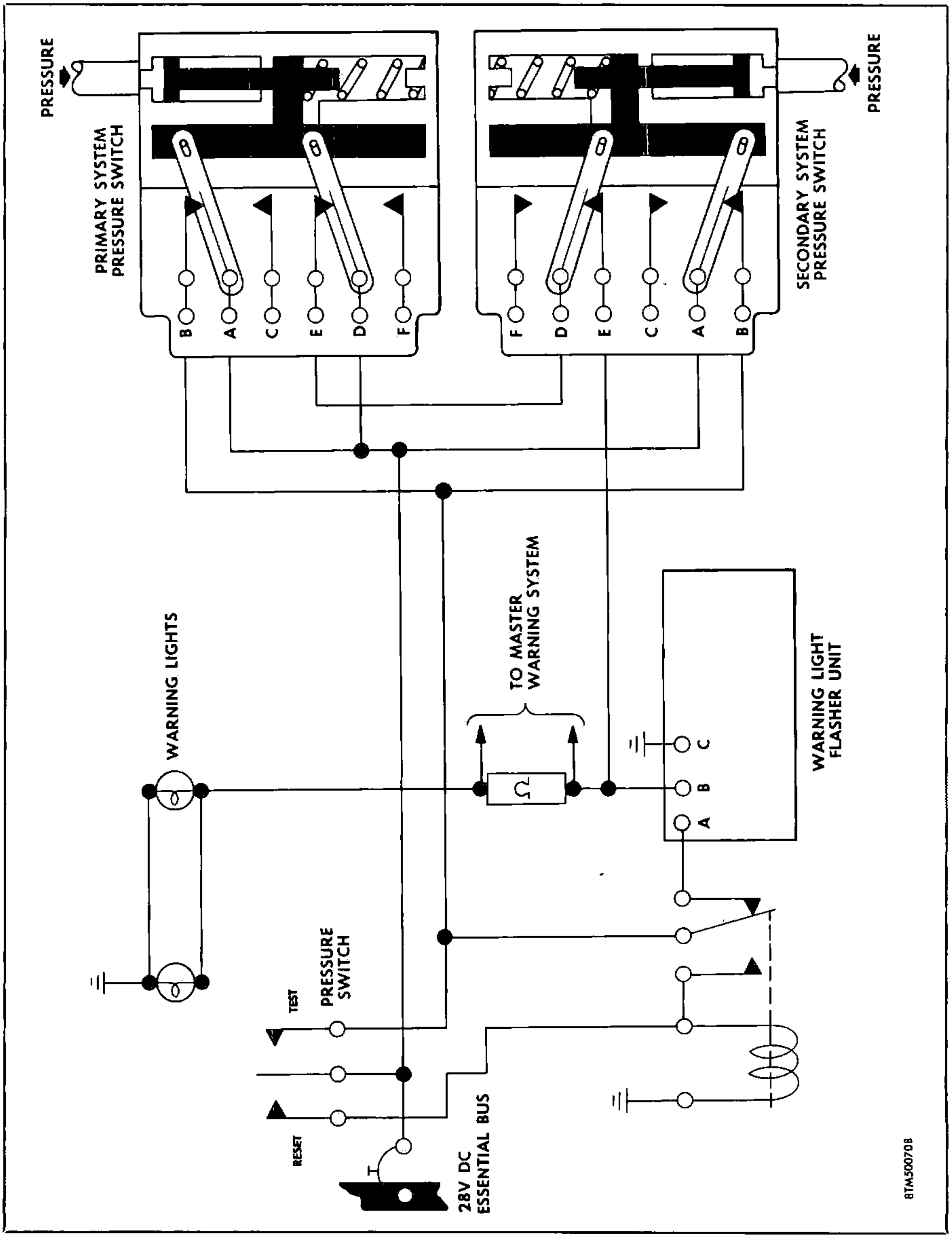

Figure 2-19. Low-Pressure Warning System Schematic

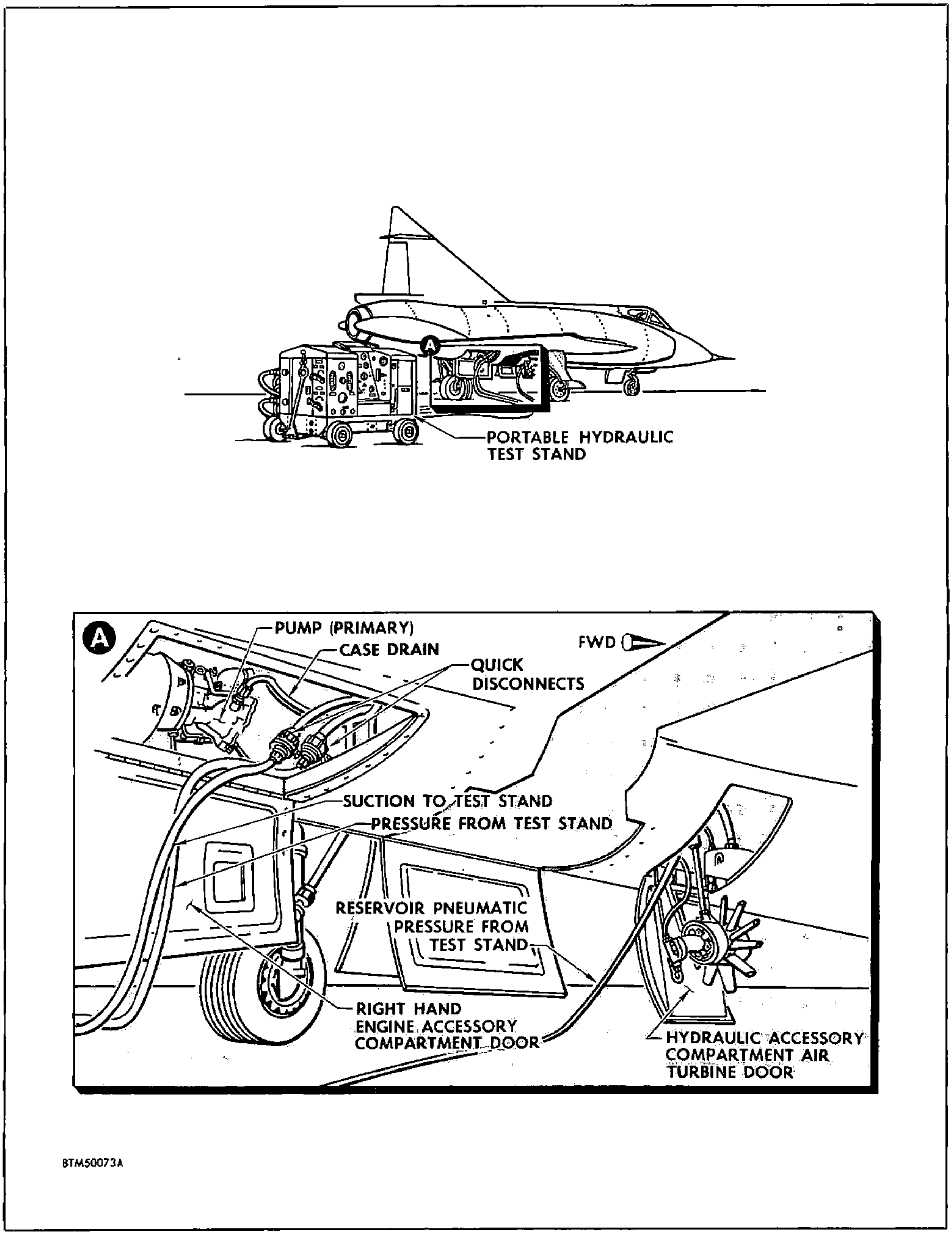

Figure 2-20. F-102A Hydraulic Test Stand Provisions

WARNING

Be sure that another gas is not mistakenly used in place of nitrogen. Other gases, such as oxygen and acetylene, will cause an explosion.

Extreme ambient temperature changes, such as those encountered in desert and arctic regions, cause changes in the required amount of initial accumulator precharge. Temperatures around 150°F increase the required initial charge to a maximum of 870 psi so that 750 psi is in the accumulator when it cools to normal temperatures. Extreme low temperatures, such as —65°F, require that the accumulator be precharged to only about 570 psi.

When charging the accumulator, remove the dust cap on the air filler valve and attach the filler hose. Then loosen the ⅝ inch hexagonal swivel nut on the valve a maximum of ¾ of a turn. After charging the accumulator, tighten this swivel nut from 50 to 70 inch-pounds torque and replace the dust cap.

Discharging Air From an Accumulator.

To discharge an accumulator perform the procedure opposite from that of charging the accumulators. Again the hydraulic pressure in the accumulator is dissipated to allow the charge of gas to escape slowly when released and to obtain an accumulator reading. Loosen the ⅝ inch hexagonal swivel nut a maximum of ¾ of a turn, and depress the small valve core in the center of the filter valve. A suitable tool comparable to the size of a match stick is ideal to depress the valve core and permit the nitrogen gas to discharge. After the operation is completed torque the ⅝ inch nut on the valve to 50 to 70 inch-pounds and replace the dust cap.

GROUND OPERATION OF HYDRAULIC POWER SYSTEMS.

To perform a complete check and adjustment many of the operating parts of the hydraulic system, a means of ground operation is necessary. On many occasions the various operating systems require hydraulic pressure for test operations when the airplane is on the ground and the engine is not operating. To perform these checks, several powered hydraulic test stand connections are on the airplane to allow use of such outside facilities. A portable test stand is standard organizational equipment in F-102A maintenance sections.

HYDRAULIC TEST STAND.

The hydraulic test stand, shown connected to the F-102A in figure 2-20, is the portable hydraulic test stand used by F-102A maintenance crews. The test stand is a steel cabinet which houses the test stand components and is mounted on a four-wheeled heavy-duty trailer. This unit contains an air compressor and two separate hydraulic systems complete with pumps, reservoirs, valves, gages, controls, and hoses for connection to the airplane hydraulic systems. An industrial gasoline engine installed on the test stand supplies power for producing pressure. The test stand power system is capable of producing 3000 psi pressure for use in both systems at the same time.

Figure 2-21 shows the connections between the test stand and the primary system. You can also see the pneumatic pressure hose (8) entering the hydraulic accessory compartment. The test stand suction (3) and pressure hoses (4) connect to the corresponding primary system lines at the pump quick-disconnect fittings (2). The test stand thereby bypasses the engine pump (6) and furnishes hydraulic power to the airplane system. Although not shown, a second set of test stand suction and pressure lines connect to the secondary system at its engine pump quick-disconnect fittings. The test stand pneumatic pressure line (4) connects to the reservoir air pressure line at its quick-disconnect fitting. This line supplies air pressure to the two reservoirs when the test stand is operated using the airplane reservoirs.

The illustration of the test stand unit (figure 2-21) shows its many controls and connections which may seem quite complicated. However, after a second look you see that the stand is merely a dual hydraulic power supply unit. This portable test stand has all of the necessary equipment that allows it to power the airplane hydraulic systems without the use of other external aids. It is a good idea to learn everything about the test stand and all of its gages, valves, and connections so that you can use it to the best advantage. You will utilize nearly every control each time you operate it. You should check that the test stand hydraulic reservoirs are full before each operation, and that the water, oil, and battery levels in the test stand engine compartment are satisfactory.

Study figure 2-21 to become familiar with these controls and connections. Details A, B, and C are close-ups of these controls and connections. In detail A note the two levers used to select either the airplane reservoirs or stand reservoirs. Then on the front you can see the primary and secondary system flowmeters, volume indicators, air pressure gage, and the air shut-off valve. In detail B are the hydraulic and air connections and the test stand reservoir filler provision.

In detail C note that there are two instrument panels which can be covered with a folding head. The larger panel provides the controls for operating the primary and secondary hydraulic systems, while the side panel contains the test stand engine instruments and controls. To the right and on the front of the stand, note

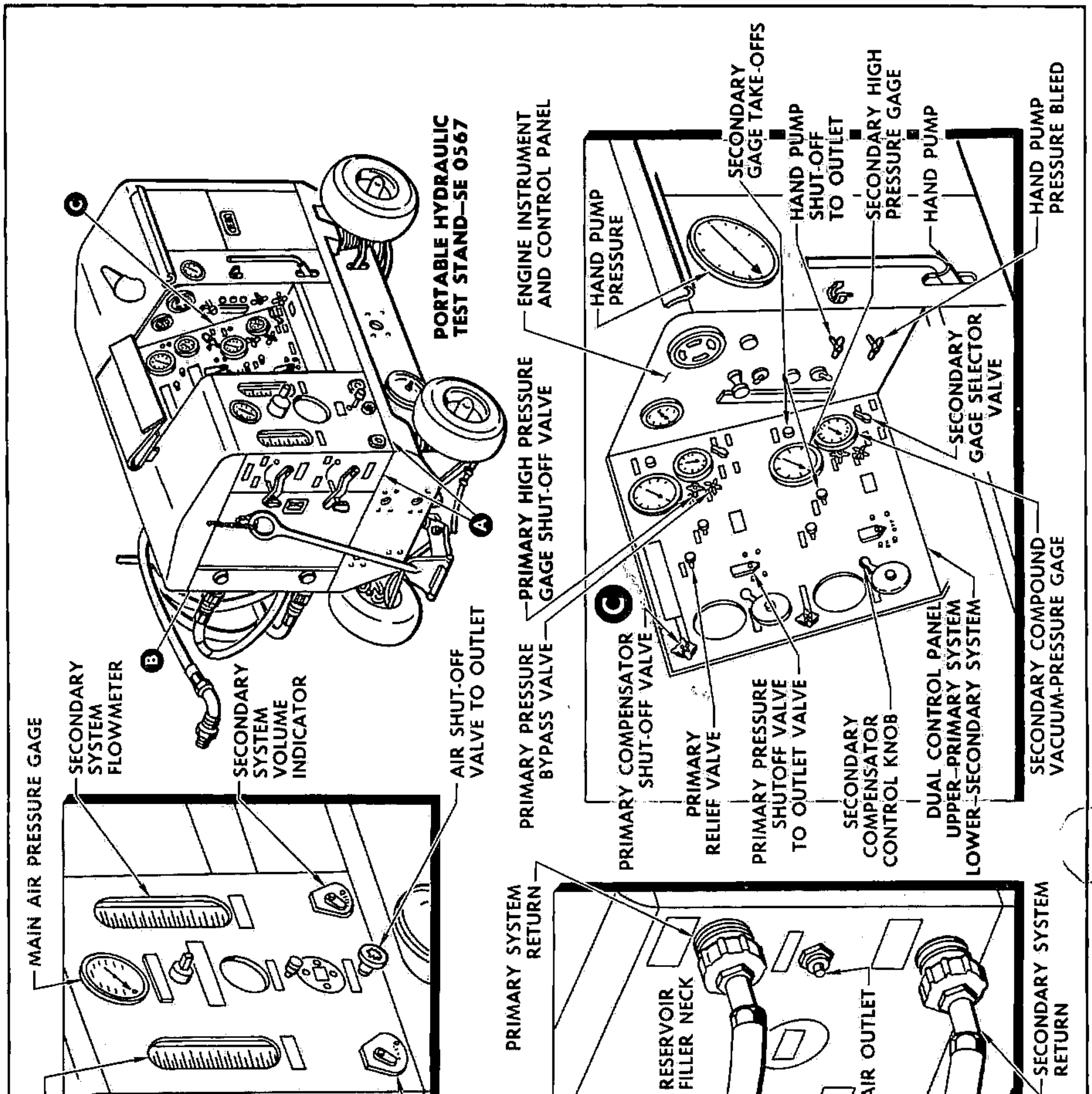

Figure 2-21. Hydraulic Test Stand

a hand pump and its pressure gage. This hand pump is used when you need a little pressure but do not want the full system pressure.

When operating the test stand always wipe the hose connections dry and keep stand clean of grit and dirt. In all cases before starting the stand, place the airplane controls in neutral and clear all personnel from the danger areas of operating units around the airplane. Complete instructions on the operation of the test stand are given on the instruction plates on the test stand and in the Handbook of Maintenance Instructions, T.O. 1F-102A-2-3. Before you actually operate the test stand alone, however, it is advisable to have an experienced test stand user demonstrate the operations to you.

Operating Hydraulic Systems With Test Stand Reservoir.

The STAND RESERVOIR position of the test stand selector valve allows the hydraulic fluid in the test stand reservoir to operate the systems in the airplane. Connect the test stand hoses to the quick-disconnect fittings of the desired system and release the air from that system's pressurized reservoir. Pressurize the test stand reservoir to 15 psi before you adjust the air pressure regulator and place the reservoir pneumatic selector valve in PRESSURE position.

Before actually applying pressure to the hydraulic system start the stand engine and allow it to heat to normal temperature.

Now, several precautions are necessary to assure safe and correct operation. A safety check before applying pressure includes an accumulator check for the 750 psi preload, pressure bypass valve opened, and the airplane controls placed in neutral. After this check has been made, open the pressure shutoff valve to the OUTLET position and slowly close the pressure bypass valve. A compensator control knob is on the test stand panel and is used to adjust the pressure as dictated by the requirement of the system in the airplane. After you are through using the test stand, open the pressure bypass valve, close the pressure shutoff valve to the outlet position, and then turn the test stand power unit off.

Operating Hydraulic Systems With Airplane Reservoirs and Test Stand Pumps.

Only minor adjustments are necessary to change the test stand operation of the system from using the test stand reservoir fluid to using the airplane reservoir fluid. Before starting the test stand power unit, all of the previously performed preliminary steps are re-enacted except three. First, instead of disconnecting the airplane reservoir air pressure connection, allow it to remain and adjust the air pressure regulators to 40 psi. Second, place the reservoir selector valve on the test stand in SHIP RESERVOIR position. And lastly, after the stand engine is started and warmed up, slowly close the pressure bypass valve.

Although this description will familiarize you with the hydraulic test stand and its purpose, refer to T.O. 1F-102A-2-3 for complete operating instructions from time to time.

MAINTENANCE PROBLEMS OF HYDRAULIC POWER SYSTEMS.

In a complex hydraulic system a number of malfunctions are possible. In almost every instance the cause of most malfunctions will be due to air getting into the system, to dirty fluid, or to leaky seals and connections. These three conditions can cause damage to the pumps, reservoirs, valves, and other operating components throughout the system. Therefore, it is the aim of every squadron maintenance section to keep these three troublemakers to a minimum.

AIR IN HYDRAULIC FLUID.

The hydraulic power system is hermetically sealed to keep air from the fluid in the system and thus avoid many undesirable complications. The hydraulic fluid used in the F-102A hydraulic system loses its efficiency when it is mixed with air. Low efficiency caused by air mixture leads to erratic performance by the using subsystem.

Air cavitates the rotating parts in hydraulic pumps by forming "dry spots" or small fluid vacuums on the metal surface. This causes bumpy, unstable fluctuations in the pump output. So, since air is a principal enemy to the hydraulic systems, the importance of having properly sealed and connected parts cannot be emphasized too much.

DIRT AND GRIT HARMFUL TO PRECISION PARTS.

To avoid foreign solid particles from endangering the components in the system, low pressure and high pressure filter elements are strategically placed throughout the system. Also several filtering screens and snubbers are used to sift out any foreign matter. However, the problem of dirty, gritty hydraulic systems continues to exist. The best way to eliminate the trouble of dirt and solid particles in the system is to KEEP IT OUT. However, here again is a big problem that seems unsolvable. Dirt always seems to get into the system.

Dirty hydraulic fluid clogs up orifices, causes pumps and other precision parts to malfunction, and creates faulty gage indications. Bleeding dirty fluid and recirculating clean fluid in a system periodically, in addition to frequent filter element changes, usually eliminates the bigger portion of these headaches. When you replace component parts, be sure these parts are both clean and tightly sealed—this is one of the principal ways of keeping dirt out of the system.

HOW LEAKS ARE PREVENTED.

Properly sealed joints, unions, pistons, and valves prevent leaks in hydraulic systems. It is very important that the proper seal is selected as well as to make sure the seal is properly installed. Every seal used in the F-102A hydraulic system has a *blue* dot on it. This is a color code identification. Never use O-ring seals with red or yellow dots unless a blue dot is also on the seal. Seals are listed in T.O. 1F-102A-4, and this Illustrated Parts Breakdown must be consulted to obtain the proper seals for replacement. Seals are replaced after a connection is broken or a sealed unit is removed. In almost every instance the removal of a sealed unit either damages the O-ring seal, or the seal needed replacing beforehand.

SUMMARY.

In this chapter you learned about the operation and servicing of the F-102A hydraulic power supply system. Each major component was described and its function in the overall operation of the system was explained. Now you are prepared to go into a more detailed discussion of the five subsystems that use this hydraulic power as their means of operation. The constant high pressure produced by the two power supply systems keeps fluid pressure at the control valves and selector valves of these subsystems. Upon demand, this pressure is available for use in the operation of the subsystem actuators. Therefore, if you thoroughly understand this chapter, you have half of the system operation under your belt. However, an equally important part is yet to come, since it is in the subsystems where the hydraulic fluid must fulfill precisioned actuations.

Chapter III is a complete description and operation of the dual-powered flight control system. Chapter IV consists of the complete hydraulic operation of the other subsystems that use this power—the landing gears, the nose wheel steer damper unit, the speed brakes, and the emergency a-c generator.

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Chapter III

FLIGHT CONTROL HYDRAULIC SYSTEM

Contents *Page*

In the preceding chapter you learned how the primary and secondary hydraulic power supply systems function to produce power for the operating subsystems. You also learned when the emergency hydraulic power supply system is used and how it operates. The material in this chapter will acquaint you with the elevon and rudder hydraulic actuating systems on the airplane.

This chapter gives you a short review of the basic theory of flight controls on conventional aircraft as compared to those on the F-102A so that you will have a better understanding of the flight control systems. Following the review is a description of the elevon hydraulic system and how it operates. This operational description includes three methods of controlling the flight of the F-102A. Next you will become acquainted with the rudder hydraulic system which is similar to that of the elevons.

The other systems that make up the complete elevon and rudder systems, such as the damping system, trim system, mechanical linkage, feel system, and the like, are mentioned or described only as necessary to explain the operation of the control hydraulic systems. All of these systems are described fully in another manual of this series covering the Flight Control System.

BASIC THEORY OF FLIGHT CONTROLS.

Before discussing the F-102A flight control hydraulic system, let us review some of the basic theory of flight controls. We will discuss these, and the importance of the use of hydraulic power for their actuation. This review of conventional airplane flight controls is on a comparative basis with the F-102A flight controls.

THE AXES OF AN AIRPLANE.

For the purpose of studying or discussing stability, flight maneuvers, or any other motion of an airplane, we must consider that the airplane has three axes. On figure 3-1 note the three axes, lettered XX, YY, and ZZ. You may think of these axes as rods extending through the center of gravity of the airplane. Each axis is perpendicular to the other two. As you will recall from basic theory, when the airplane's fuselage and wings are level with respect to the ground, the position of their axes is as follows: the vertical axis (ZZ) is perpendicular to the ground, while the lateral axis (YY) and the longitudinal axis (XX) are parallel to the ground.

The motion of an airplane about these axes is shown in figure 3-2. Now, referring to both illustrations, assume that the airplane is rotating around the lateral axis (YY). Note that the other two axes (XX and ZZ) move with it. This motion is called *pitch*. Now imagine that the airplane is rotating about either of the other two axes, and note that the alternate two axes also rotate. When the airplane rotates about the longitudinal axis, or "banks," the movement is called roll. Movement about the vertical axis is called *yaw*.

METHODS OF CONTROL.

In earlier studies, you probably learned that the pitch movement about the lateral axis (YY) is controlled by elevator surfaces, that the roll movement about the longitudinal axis (XX) is controlled by aileron surfaces, and that yaw movement about the vertical axis (ZZ) is controlled by the rudder surface. In the F-102A with its delta-type wing, the *elevator* and

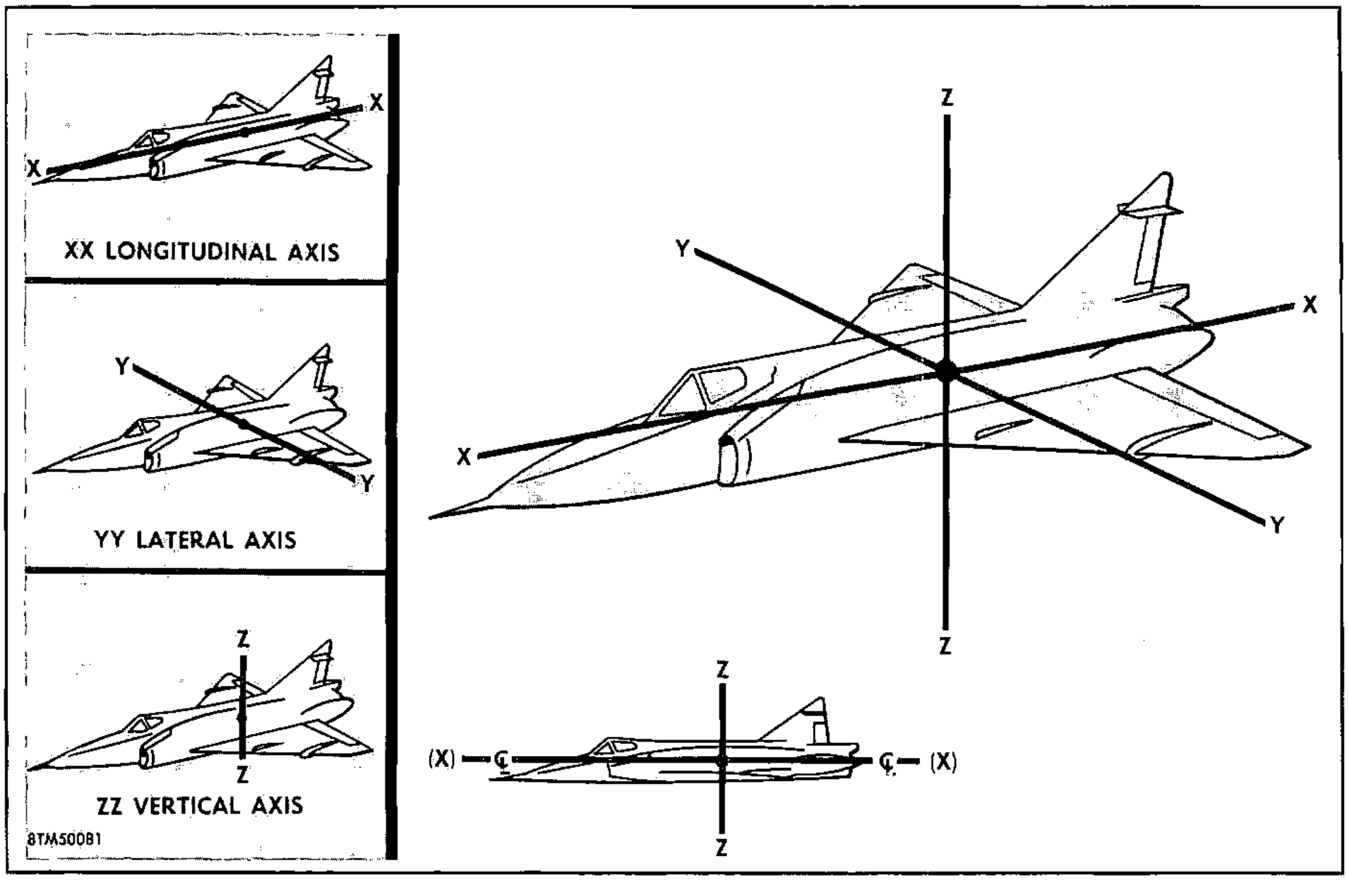

Figure 3-1. The Three Axes of an Airplane

aileron surfaces are combined into control surfaces called *elevons.* These elevons are hinged to the wing trailing edges. So, where an airplane of conventional design has a pair of elevators attached to the horizontal stabilizer and ailerons attached to the wings, the delta-wing F-102A combines the elevators with the ailerons. The elevon control system combines elevator and aileron motion in the single pair of elevons. During straight and level flight, each of these control surfaces is in a streamlined position—in line with the immovable wing surface to which it is attached. This is also true of the rudder which is attached to a conventional vertical stabilizers on the upper aft end of the fuselage. When a control surface is moved out of line with its attached surface—in one direction or the other—air pressure against it deflects the airplane to a new course.

TYPES OF CONTROL SURFACE ACTION.

Control surfaces are often balanced, or arranged, so that a part of the movable surfaces extend forward of the hinge line. In such designs, air pressure is on the forward surface and also on the surface aft of the hinge line. Air pressure on the section forward of the hinge line, therefore, helps counteract pressure on the aft section of the surface.

Several types of surface balancing, counterweights, and horns are used on airplanes to work against the air pressure on control surfaces. On some airplanes, adjustable tabs are hinged to the trailing edges of control surfaces to assist in moving these surfaces. The moving of these tabs in a direction opposite to the desired control surface movement helps force the control surface in the desired direction. Some large airplanes use a combination of the above plus hydraulic actuation to move their control surfaces. This combination helps the pilot move the large surfaces which would be impossible to do without these or similar aids.

Of the aids just discussed, the F-102A uses only hydraulic actuation. The reason for this selection is that on an airplane which operates at extremely high, and sometimes supersonic speeds, *cleanness* of all surfaces is most important. In addition, more structural strain exists on a high-speed airplane than on the slower ones. For instance, an airplane traveling at high speed strikes turbulent or rough air much harder than one traveling slowly. And too, a change of direction at high speed causes a much greater increase in weight on the wings and the control surfaces. The action that causes this increase in weight is called *centrifugal force.* The increased loads require greater pilot effort to move the control surfaces. Because of

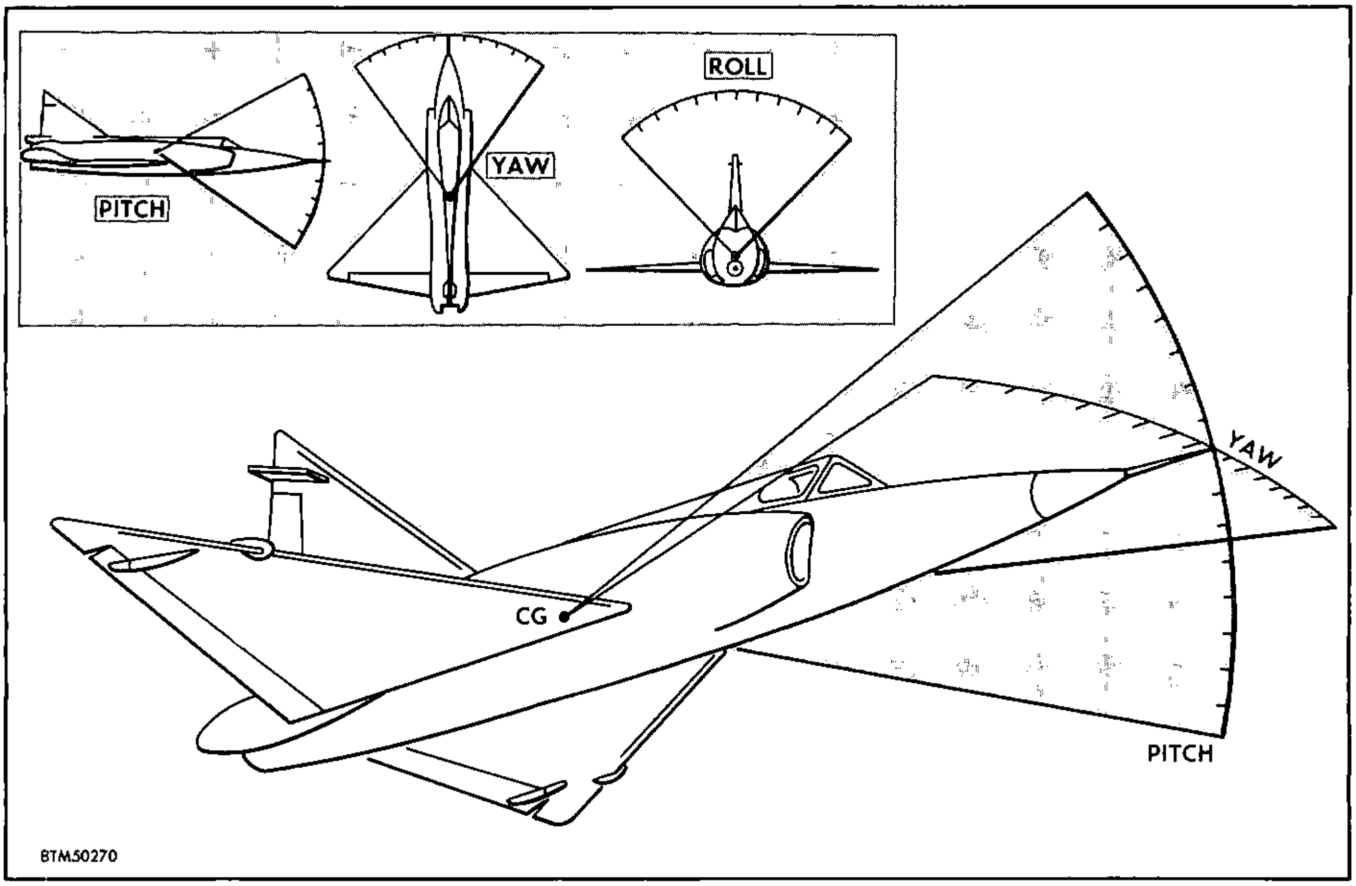

Figure 3-2. Rotation About the Three Axes

this, modern airplane control surfaces are designed to be moved by hydraulic power with the pilot moving only the hydraulic system control valves instead of the entire flight control surfaces. On the F-102A, all control surfaces are hinged at their leading edges, they have no tabs, and they are powered by hydraulic pressure. You will learn about this hydraulic actuation of the F-102A control surfaces in this chapter.

WHY CONTROL SURFACES ARE TRIMMED.

Tab trimming of control surfaces is used to do some of the work the pilot would otherwise have to do. Tabs maintain the control surfaces in particular positions away from neutral during straight and level flight so as to trim the airplane. For example, if you were a pilot and your airplane tended to turn constantly to the left, you would have to hold the rudder continuously to the right. However if the rudder were equipped with a trim tab, you could move the tab so that its trailing edge is toward the left. This trimming would force the entire rudder toward the right so that it would not be necessary for you to exert a constant pressure on the right rudder pedal.

On the F-102A the rudder does not have a tab, so the entire surface is trimmed to the right to do the same trimming action. The same type of trim corrections are applied to the ailerons (elevons on the F-102A) if one wing is "heavy" because of unbalanced loads, and to the elevators (elevons again on the F-102A) if the airplane loads are unbalanced forward or aft of the lateral axis.

We have discussed why directional controls are needed, the methods of control used on different airplanes, and the types of control surface action. We have also seen why and how control surfaces are trimmed. In the following paragraphs we will discuss the F-102A flight control hydraulic systems, and how hydraulic power moves the control surfaces. To understand these systems better, we will discuss the elevon and the rudder hydraulic systems separately.

F-102A FLIGHT CONTROL SYSTEM.

The flight controls on the F-102A consist of only two systems instead of the usual three systems found on other aircraft with conventional wings and empennage. These two systems are the elevon and rudder systems. As stated earlier, this chapter deals with the hydraulic portion of the two flight control systems. However, a brief description of the flight control systems is included at this time so that you will have a better understanding of the hydraulic system operation. No attempt will be made to give you a comprehensive description of why or how some control signals that affect the hydraulic system are generated. If you want a complete description of the flight control system and all its phases, refer to the Flight Control Systems Training Supplement in this series on the F-102A.

ELEVON CONTROL SYSTEM.

The elevon control system combines mechanical movement and hydraulic power to move the two elevons on the wing trailing edges. A diagram of the elevon control system is shown in figure 3-3. Conventional stick movement is retained in the cockpit to attain elevator and aileron action. That is, fore and aft movement of the control stick give elevator action for pitch control of the airplane, and movement of the stick from side to side produces aileron action for roll or bank control of the airplane. Movement of the control stick in either direction is transmitted by mechanical linkage to a "mixing" assembly which combines elevator and aileron stick action. Movement of the mixing assembly is again transmitted by mechanical action in proportional amounts to the left and right elevon hydraulic components.

Since the elevons are moved by full hydraulic power, the pilot cannot feel airloads imposed on the surfaces when he moves them. To overcome this condition an artificial feel system is included to give the pilot *feel* of the controls and prevent him from overloading the elevon surfaces.

In addition to the pilot's direct control of the elevons, there are two automatic systems that control elevon surface movement. One system known as the *pitch damper system* automatically corrects for minor pitch deviations of the aircraft, regardless of the course. This system controls elevon movement when the pilot selects the *manual* mode of operation. The other automatic system is known as the *pilot assist system* and is engaged when the *pilot assist* mode is selected. It operates similar to the damper system, but controls elevon movement over a wider range of deviation. This system maintains the airplane on a pilot-selected course. Both of these systems are electronic in nature and operate on the principle of a servo-type system.

Both the mechanical movement of the elevon system and the signals from the electronic systems are further described when we discuss the elevon hydraulic system and its components, later in this chapter.

RUDDER CONTROL SYSTEM.

The rudder control system also combines mechanical movement and hydraulic power to move the rudder surface. A diagram of the rudder control system is shown in figure 3-4. Movement from the rudder pedals is transmitted by mechanical linkage to the hydraulic components at the rudder. These hydraulic components in turn move the rudder surface in a direction corresponding to pedal movement. Later in this chapter you will learn just how these hydraulic components funcitton to move the rudder. Just as in the elevon system, the rudder system also has an artificial feel system to give the pilot simulated *feel* of the rudder.

The two automatic electronic systems (*manual* and *pilot assist* modes) that control the elevons also control the rudder movement in addition to the pilot's direct control of the surface. When engaged in *manual* mode, electronic signals control rudder movement through the hydraulic components to keep the airplane on a stable course. In the *pilot assist* mode, rudder movement is coordinated with aileron movement in a pilot initiated turn.

The mechanical movement of the rudder system and the signals from the electronic systems are further described when the rudder hydraulic system and its components are discussed later in this chapter.

ELEVON HYDRAULIC SYSTEM.

Full hydraulic power is supplied to the right and left elevon flight control systems from the primary and secondary hydraulic power supply systems. In figure 3-5 you can see the hydraulic system for the right elevon. (Refer also to figure 1-4 in Chapter I.) The hydraulic system for the left elevon is exactly the same as the right. Mechanical actuation from the control stick and from the actuating cylinders is represented by dashed lines. Hydraulic actuation from the primary and secondary hydraulic systems is represented by solid lines. Note that the primary hydraulic system supplies power by way of the control valve to the inboard and outboard actuating cylinders at the elevons. As just mentioned, the secondary hydraulic system also supplies power through the control valve to the same actuating cylinders. However, note that the secondary system also supplies power to other hydraulic components in each elevon control system. These components are the servo shutoff valve, the servo actuator, and the lockout valve, which operate when the automatic flight control systems are engaged.

Pilot control of the elevons is by means of the conventional control stick. Control stick motion mechanically moves the hydraulic control valves, which in turn meter primary and secondary hydraulic fluid to the control surface actuating hydraulic cylinders (rams). The surfaces move only as long as the pilot moves the control stick. When the pilot stops moving the stick, a mechanical follow-up mechanism which works through the bellcrank shown on the diagram neutralizes control forces in the hydraulic control valve. The valve then equalizes fluid pressure in the actuating cylinders, causing the control surfaces to stop moving. The position in which the control surfaces stop is determined by the position at which the

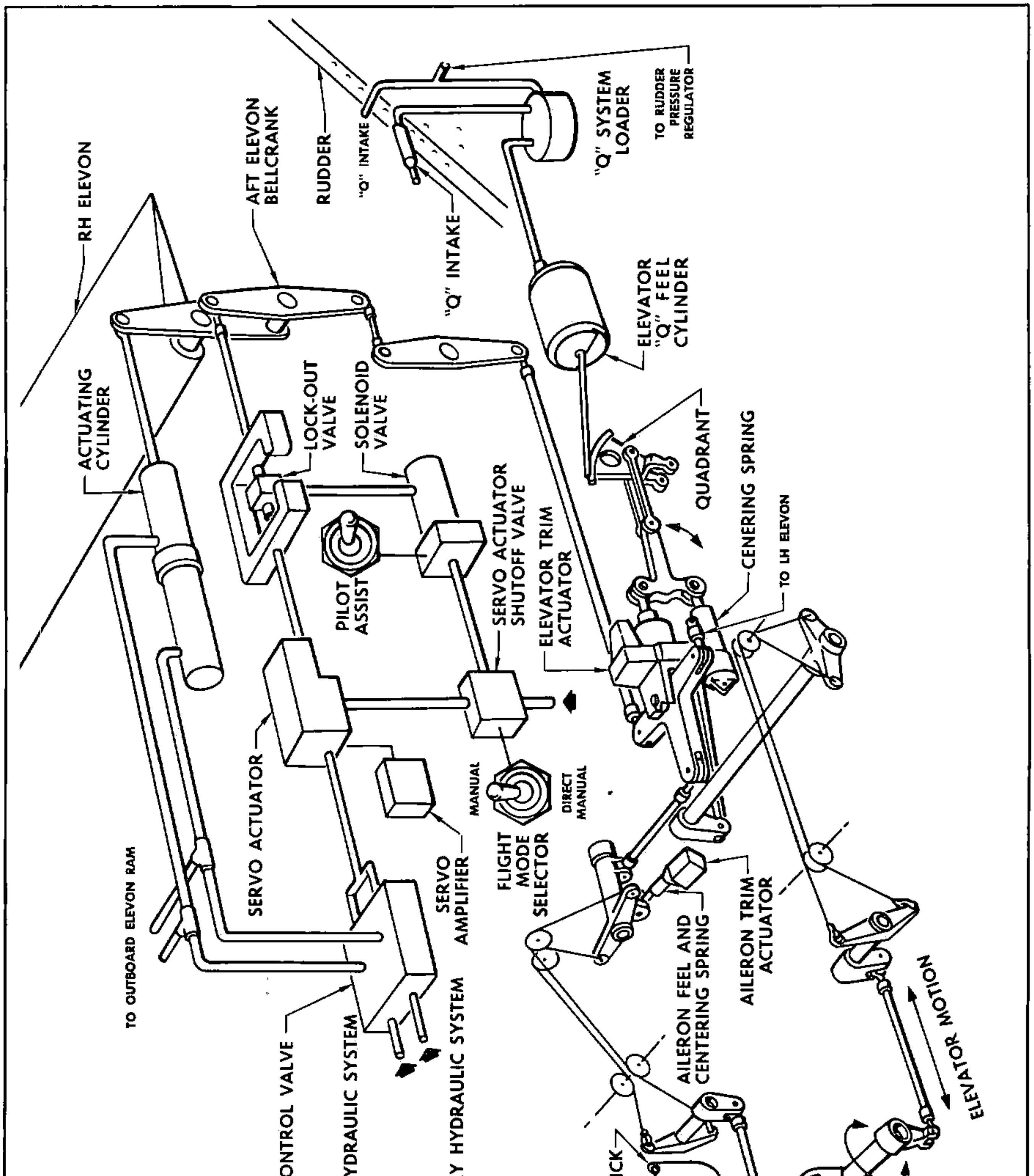

Figure 3-3. Elevon Control System Diagram

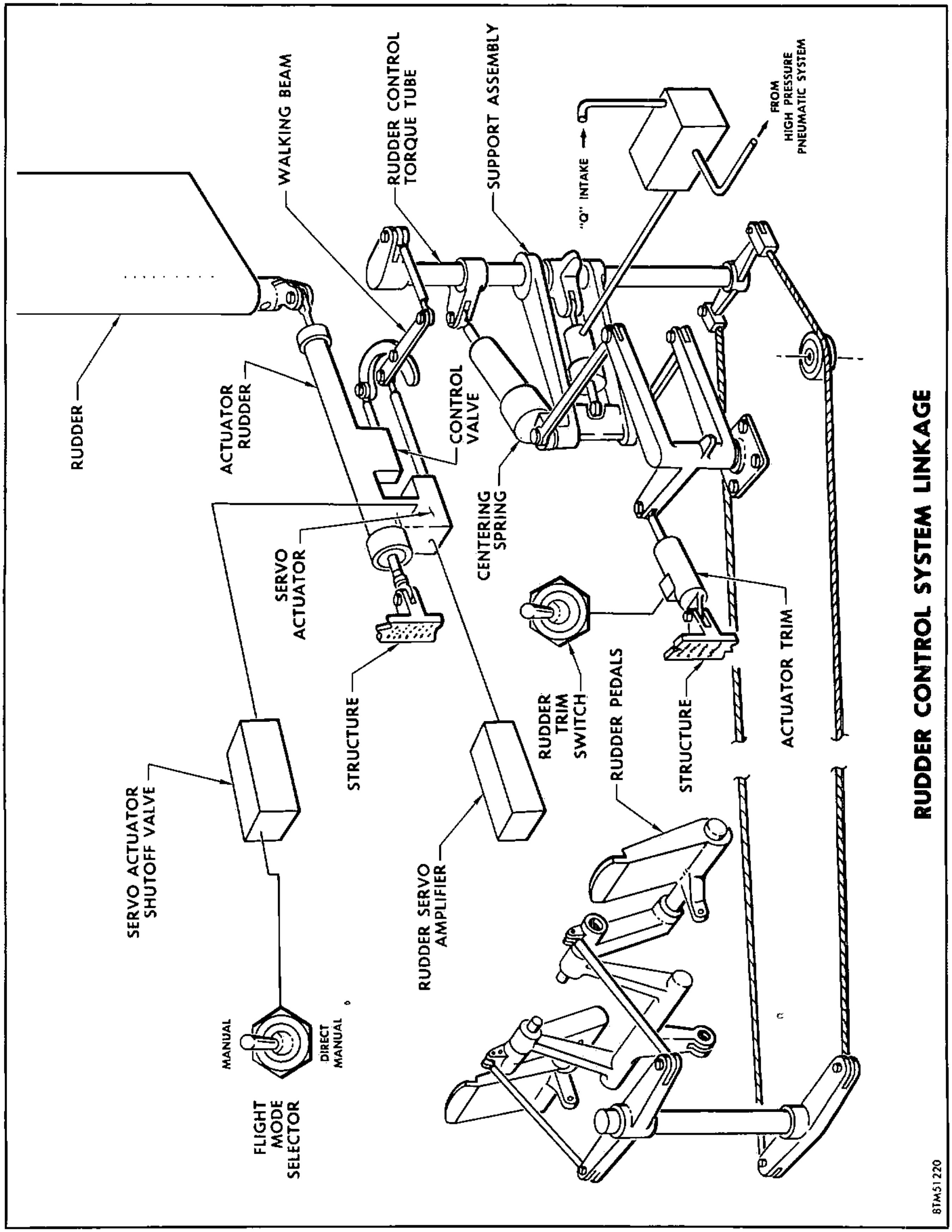

Figure 3-4. Rudder Control System Diagram

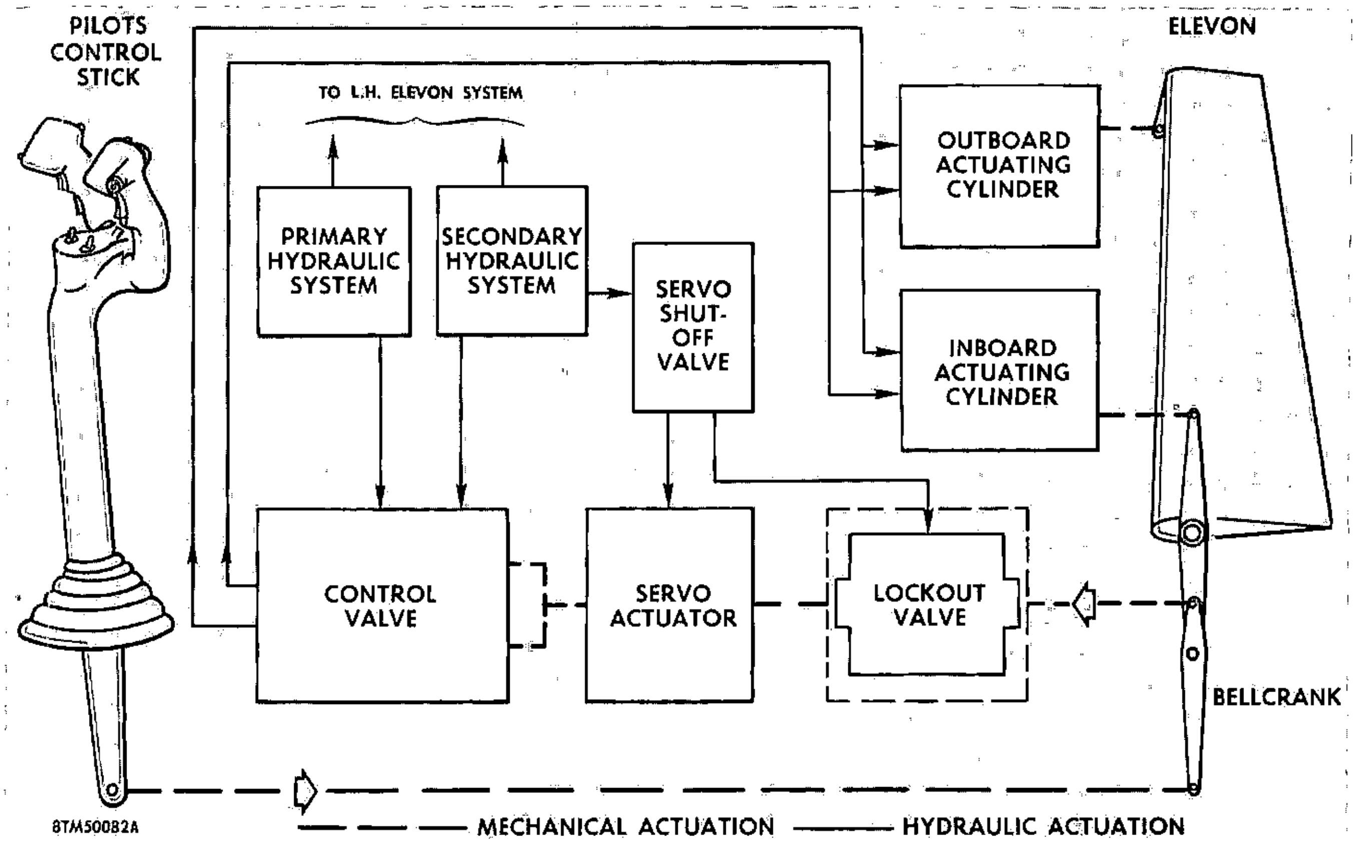

Figure 3-5. Elevon System Diagram

pilot stops movement of the control stick. Operation of the pilot's control stick is the same as on conventional airplanes.

SYSTEM OPERATION.

As you know, the elevon systems on both sides of the airplane are identical, so only the hydraulic operation of one system will be described. Figure 3-6 shows the complete hydraulic system for one elevon system. Note that pressurized fluid is routed from the primary system at the bottom of the schematic to one section of the control valve. The fluid passes through the primary section of the control valve when it is displaced, and then to the primary section of both inboard and outboard actuating cylinders.

Now, note how fluid from the secondary system entering at the top of the schematic passes through its side of the control valve, and on to its section of both actuating cylinders. Note that hydraulic pressure from both systems does not enter corresponding chambers in the two actuating cylinders. Hydraulic pressure in the outboard cylinder extends the piston rod while pressure in the inboard cylinder retracts the piston rod. The outboard cylinder piston rod connects to a horn below the centerline of the elevon while the inboard cylinder piston rod connects to a horn above the elevon centerline. (Also refer to figure 3-7.) Thus, retraction of one cylinder rod and extension of the other moves the elevon in the same direction. The details of how these cylinders function and why this method of operation is used is explained when the actuating cylinders are discussed.

In addition to supplying power to the actuating cylinders, the secondary hydraulic system also supplies power to other components in the control system. Note at the top of figure 3-6 how another line branches off the secondary system pressure line and is routed through the shutoff valve. This shutoff valve controls fluid to the elevon servo actuator and the lockout valve.

The shutoff valve is located in the line so that fluid cannot be admitted to the lockout valve unless it is also admitted to the servo actuator. Also, the servo actuator may be hydraulically energized, but the lockout valve solenoid can at the same time prevent fluid from hydraulically energizing the lockout valve. These combinations of control are used in varying ways during different phases of operation which are discussed later in this section.

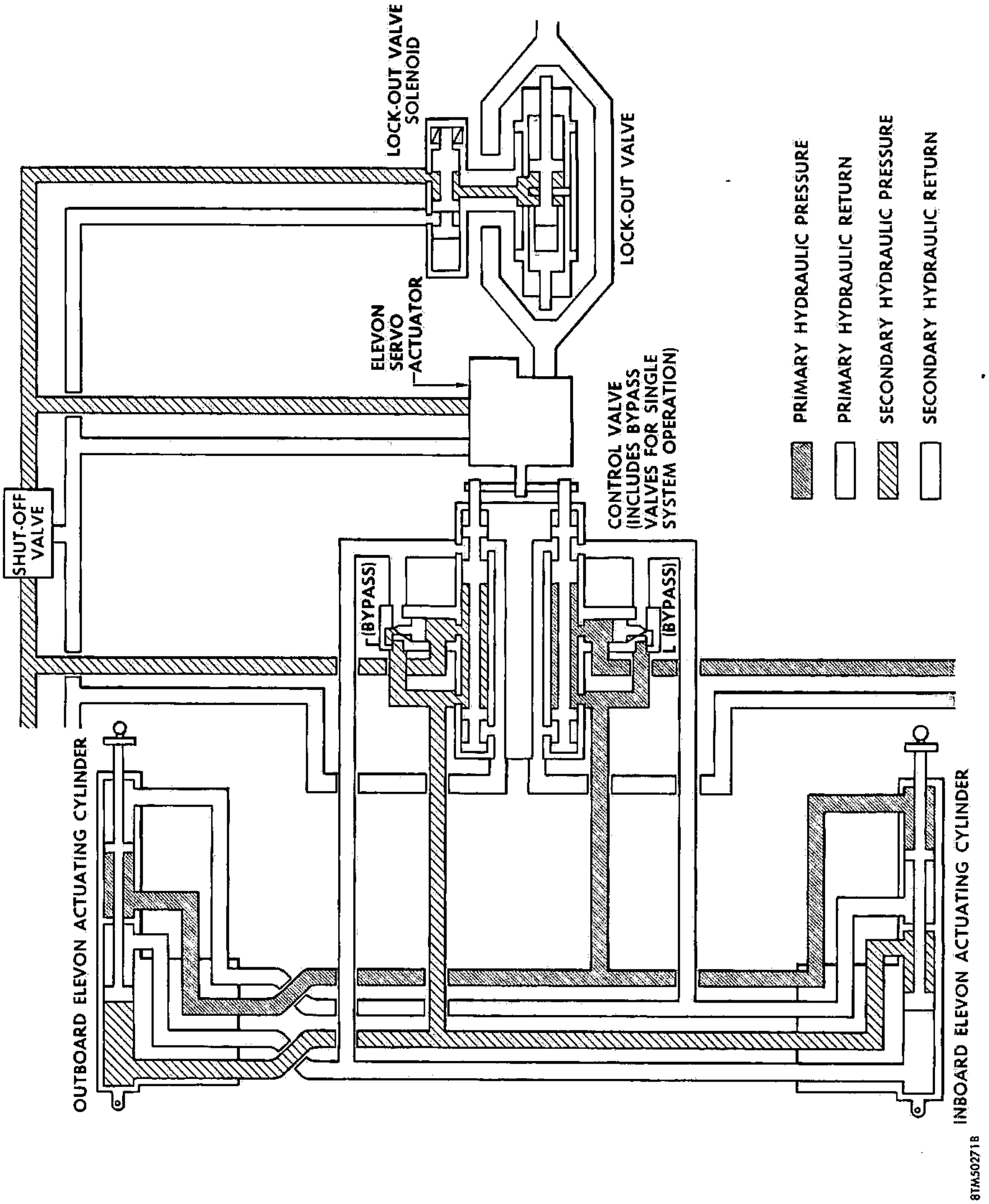

Figure 3-6. Elevon Hydraulic System Schematic

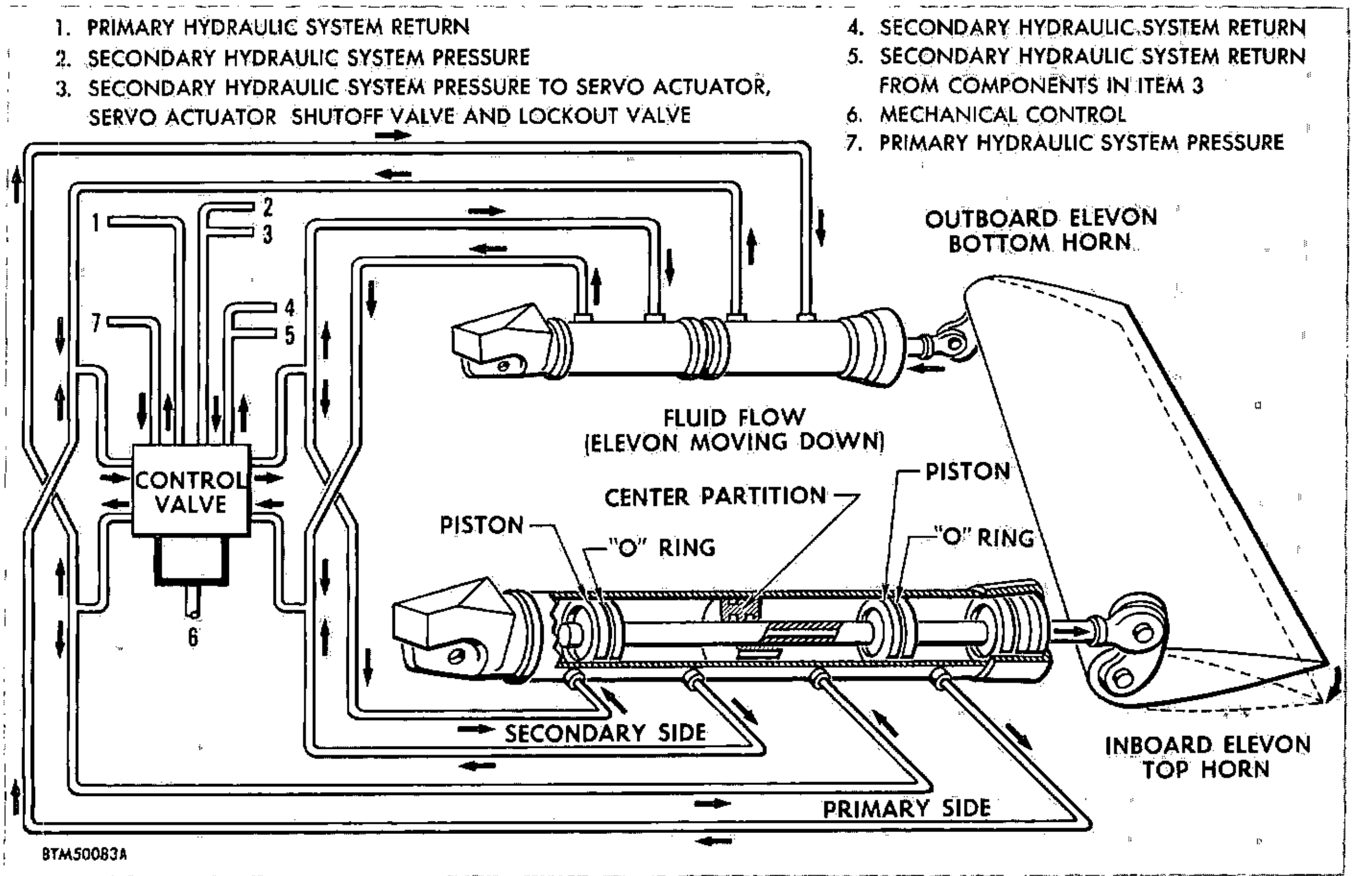

Figure 3-7. Elevon Hydraulic Flow Diagram

ELEVON HYDRAULIC ACTUATING CYLINDERS.

As you may recall, dual hydraulic actuating cylinders and full hydraulic power are used to move the control surfaces. The hydraulic power is supplied simultaneously from both the primary and the secondary hydraulic systems. This is accomplished by the internal arrangement of the actuating cylinders.

Inside the cylinders, two power pistons are mounted on a single piston rod and act within two separate sections of the cylinder bore. See figure 3-7. The piston rod is sealed with two O-rings at the point where it passes through the center partition of the cylinder bore. This prevents interflow of fluid from one hydraulic system to the other when both sides are pressurized. Although not shown in the illustration a small drain hole drilled in the housing between the two O-rings permits any leakage due to a defective O-ring to drain off, thus preventing build-up of back pressure on the other O-ring.

Remember, an inboard and an outboard cylinder are provided for each elevon, and the rod ends of these cylinders are attached to the inboard top and outboard bottom elevon horns. The inboard and outboard cylinders are hydraulically cross-connected as shown in figure 3-7. When positioning the elevon, one cylinder rod extends while the other retracts.

Unbalanced Design of The Elevon Cylinders.

In figure 3-7 note that in the secondary (forward) side of the cylinder shown in the cutaway view, the piston rod attaches to only one side of the piston. Now note that the piston in the primary (aft) side of the cylinder has the rod passing completely through it and into the secondary side of the cylinder. The piston rod attaches to both sides of the primary piston and to one side of the secondary piston. It occupies a percentage of the total piston area that would otherwise be presented to fluid pressure.

The forward side of the secondary system piston has no rod attached to it so its whole area on that side is presented to fluid forces. Therefore, there is an unequal area presented to the fluid forces. This unequal area presented to the fluid forces causes a greater force to be exerted in the *piston rod extend* direction than in the *rod retract* direction by an amount equal to the operating pressure times the piston rod area of the retracting piston. In other words, the fluid has more area to work against if the rod is not there, and therefore can push harder—do more work.

Both the outboard and inboard cylinders operate on the same principle. Therefore, since the piston rods of the inboard and outboard cylinders move in opposite directions to produce the same elevon movement, one cylinder is always providing more work thrust than the other. In one direction of elevon movement the outboard cylinder does more work; in the other direction, it is the inboard cylinder. As you can see on figure 3-7, when the elevon is moving *down,* the inboard cylinder is doing more work than the outboard cylinder; when the elevon is moving up, the outboard cylinder will do more work. Remember, this unequal force is found only in the secondary section of the cylinders; not the primary section.

Since one actuating cylinder is attached to the outboard bottom horn of the elevon and the other to the inboard top horn of the elevon, the elevon is forced slightly out of alignment at its trailing edge by the unequal thrust. The thrust is just enough to present a slightly twisted control surface to the airstream. One purpose of this relatively small twisting movement, or windup force, is to assist in overcoming any tendencies toward flutter or other unstable characteristics. The main purpose of this inequality of forces, however, is to take up the accumulation of slack in the various mechanical linkages to each actuating cylinder. This helps to prevent excessive wear of the linkage and prevents lag and jerkiness in the system.

Fluid Flow to The Cylinders.

Referring again to figure 3-7 you can see the directional arrows indicating fluid flow when the elevon is moving down, also the arrows on the cylinder piston rods near the elevon horns denoting mechanical movement. Note that there are four hydraulic lines connected to each cylinder. These lines connect to both cylinders and to the control valve. Looking at the top center of the control valve, you can also see a cluster of lines numbered 1, 2, 3, 4, 5, and 7. These lines are the pressure and return lines from and to the main primary and secondary systems, which are the sources of fluid pressure for the elevon system. Note also the mechanical actuator of the control valve, numbered 6.

When the valve is mechanically operated, two spools inside move a small amount and allow the pressure from the main hydraulic systems to actuate the cylinders. As you can determine from the crossed over lines between the cylinders and from the flow direction arrows, when fluid pressure is ported to the extend side of one cylinder, the same pressure is also ported to the retract side of the other cylinder. The return lines are also crossed over so that the return fluid flows in the same way. To give you an example: on figure 3-7, note the line coming out of the control valve at the upper right side. This line extends to a tee which branches in two directions—to the right side of the secondary piston of the upper cylinder, and to the left side of the secondary piston of the bottom cylinder. Tracing the flow arrows in this part of the system, you can see that when pressure is admitted into these lines by the control valve they will route pressurized fluid to the sides of the pistons described above.

The primary side of the cylinders is pressurized in the same manner, and at the same time. Since the inboard cylinder is attached to the top horn and the outboard cylinder to the bottom horn, and since the cylinders move in opposite directions from each other because of the crossed lines, you can see that they will both push the elevon in the same direction. Remembering the unbalanced feature of the cylinders though, you should realize that the inboard end of the elevon will be pushed down harder than the outboard end. The elevon will retain the established twist in its trailing edge when it is returned to neutral or moved up, because the outboard cylinder will then be pushing harder than the inboard cylinder.

You can trace the arrows in figure 3-7 and see that when the inboard cylinder is pushing its piston and rod toward the elevon and the outboard cylinder is pulling its piston and rod away from the elevon, the return fluid from the return side of the pistons will be pushed through the control valve and to the main hydraulic systems again. When the cylinders reverse the direction of the elevon, the return sides of the pistons will become the pressure sides, and the sides that are now pressurized will become the return sides.

ELEVON HYDRAULIC CONTROL VALVE.

The elevon hydraulic control valve is actually two valves in one; it controls both primary and secondary hydraulic system fluid to its respective actuating cylinders. This complete valve assembly consists mainly of two selector spools, two bypass valves, and four fluid ports for each hydraulic system. Figure 3-8 shows a cutaway view of one side of a control valve; the other side is identical. The valve spools are linked together mechanically at one end so that both valves stroke uniformly and meter an equal amount of primary and secondary system fluid to each side of the actuating cylinders.

In figure 3-8 you can see that the valve selector spool shown incorporates a bypass valve which is built into the control valve body. The other spool, which is not shown, also incorporates a bypass valve and is identical to the one shown. These bypass valves are incorporated to provide a low-resistance flow path through the valve if one side becomes inoperative. That is, when one hydraulic system—say the primary—becomes inoperative, the other system will still move the actuating cylinder pistons, thus causing the piston in the inoperative side of each cylinder to move its existing static fluid through the inoperative side of the control valve. If the bypass valve were not incorporated, the static fluid on the inoperative side would be forced across the lands of the spool. This is a relatively high resistance flow path and would cause damage to the system.

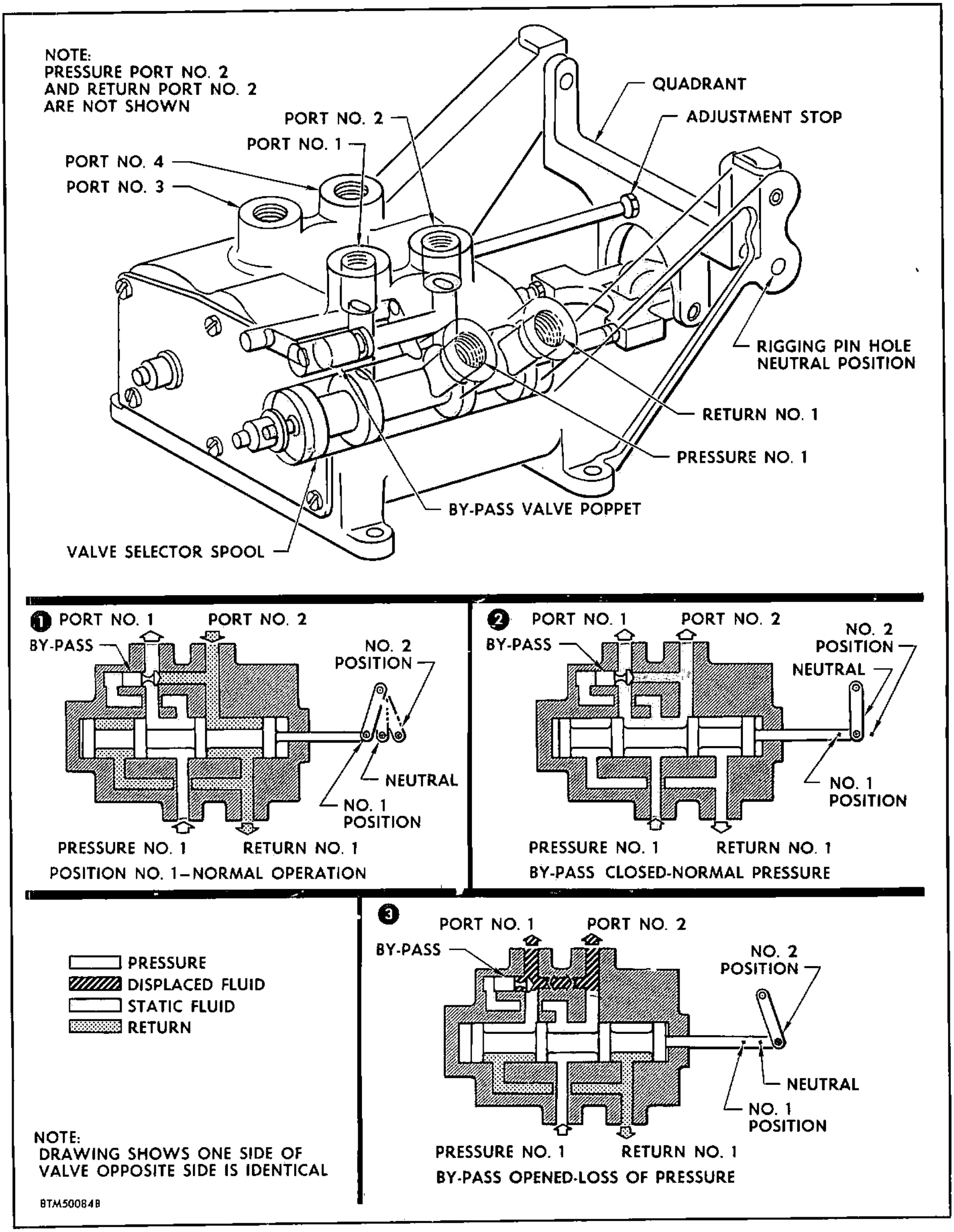

Figure 3-8. Elevon Control Valve Assembly

Control Valve Operation.

When both systems are in their normal operating condition (and they would seldom if ever be otherwise), both hydraulic systems are supplying equal pressure to their respective sides of the actuating cylinders. Mechanical actuation of the valve is instigated either by direct pilot action of his controls, or by the servo actuator when it receives an electrical impulse from the damping system. The normal flow and action of one of the valve spools is shown in the upper schematic of figure 3-8.

You can see that normal pressure is passing through the spool area in an equal amount to both sides of the cylinder piston. Now imagine the spool moved to position No. 1. This would move the spool to your left and allow full pressure to leave port No. 1. In this position return fluid from the actuating cylinder can flow through the control valve to return No. 1. If the spool is moved in the opposite direction to position No. 2, the pressure and return flows would be reversed. Note that when the spool is in the neutral position, the spool lands allow equal pressure past their inside edges to both No. 1 and No. 2 ports in equal amounts. This locks the cylinder pistons in the position they have been placed in at the moment. The only thing that will move them from this position is movement of the spools as described above.

In the upper schematic (figure 3-8) note that system pressure entering the valve is routed to the back of the bypass valve holding it to the right. In this position the valve closes the bypass passage between port No. 1 and port No. 2. Now in the lower schematic a condition is set up which shows how the bypass valve functions. Here system pressure is inoperative and the control valve spool is displaced for elevon movement. Note that pressure No. 1 is now static (no pressure). Movement of the actuating cylinder by the remaining operating system causes fluid in the inoperative side of the cylinder to be displaced from one side of the piston to the other. This displaced fluid entering the control valve moves the bypass valve to the left, thus allowing pressure to bypass through the valve between port No. 1 and port No. 2. The bypass valve remains in this position until system pressure again enters pressure No. 1.

Maintenance.

You will find that the valves are made to very close tolerances, and that any clogging of screen elements in the valve body or any particles of foreign matter around the spools will affect performance of the actuating cylinders. During maintenance, these conditions may be detected by sluggish response to motion of the control stick. Also, defective O-rings may cause leaking, which of course will affect control performance too. If control is sluggish and hydraulic pressure is up to standard in the system, the control valve is the most likely unit to be suspected. Replacement of the entire unit is the remedy.

To remove a faulty valve, you should first be sure that there is no pressure in the two hydraulic systems, then you can disconnect the eight hydraulic lines from the valve. The four center lines which connnect to the top of the valve will have to be removed completely to allow the valve to be lifted out of position. Therefore, you should mark or tag the lines in some manner so that you will know where they go when you reinstall them. You should also cap or plug all openings in disconnected tubing and components when working on any part of a hydraulic system. This is to insure that no foreign matter or moisture enters the system. After removing or disconnnecting the tubing, you then disconnnect the push-pull shaft of the servo actuator from the control valve by removing its connecting bolt and nut. *Do not disturb the shaft adjustment on the servo actuator.* All that remains to be done then is to remove the mounting bolts from the valve and lift the valve out of its position.

Installing a new valve is the reverse of the removal procedure; that is, the new valve is bolted in position, the push-pull shaft is connected to the servo actuator shaft, and the tubing is reinstalled. The new valve is pre-adjusted for *neutral* position and for throw of the spools; otherwise, it is identical to the old valve in adjustments. It should not be necessary for you to disturb the adjustment stop on the quadrant. However, you should install a rigging pin in the rigging hole shown on figure 3-8 so that the valve mechanism is held rigid when connecting the push-pull shaft to the servo actuator. Always be sure that you remove this pin when the installation is completed.

The preceding description has pointed out the highlights of the valve replacement. Refer to your F-102A maintenance manual, T.O. 1F-102A-2-3, for detailed step-by-step instructions and check-out of the system after valve replacement.

DIRECT PILOT CONTROL OF THE ELEVONS.

There are three ways of controlling the F-102A elevons in flight. These methods are called modes. They are direct pilot control (*direct manual* mode), and the two electronic systems (*manual* and *pilot assist* modes). Only the elevon control valve and the elevon actuating cylinders are required for control of the elevons while in the *direct manual* mode of control. This type of control is the condition of flight when the pilot has direct control of the airplane, and only his motions at the control stick result in a direct change of course or attitude.

Figure 3-9 shows the right-hand elevon control assembly in the fuselage, inboard of the elevon. The same type of assembly is on the other side of the fuselage for the left elevon. This view shows the tie-in of the mechanical system with the elevon hydraulic components. Note how the upper rod at the aft elevon bell crank connects through two units to the elevon

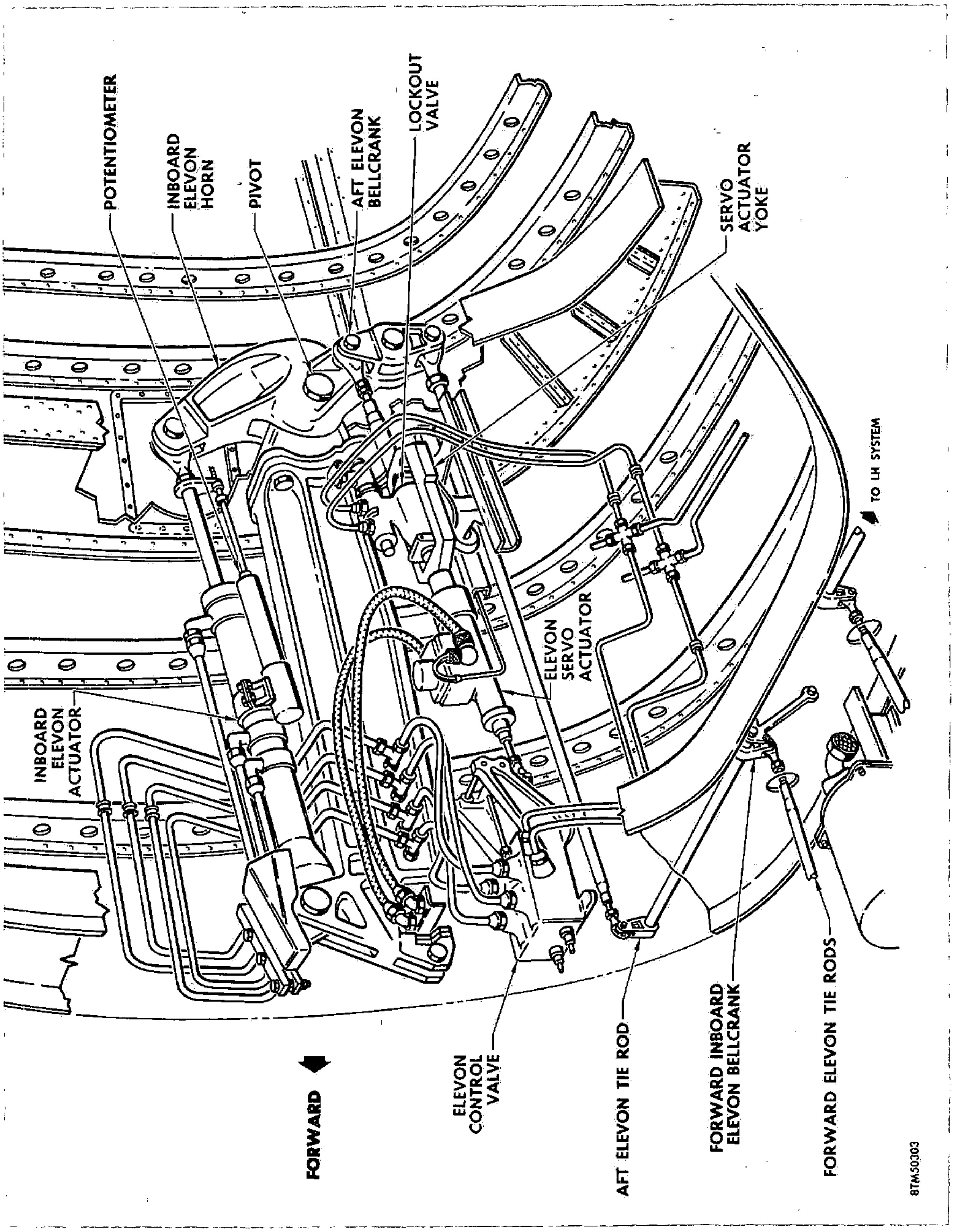

Figure 3-9. Right Hand Elevon Control System

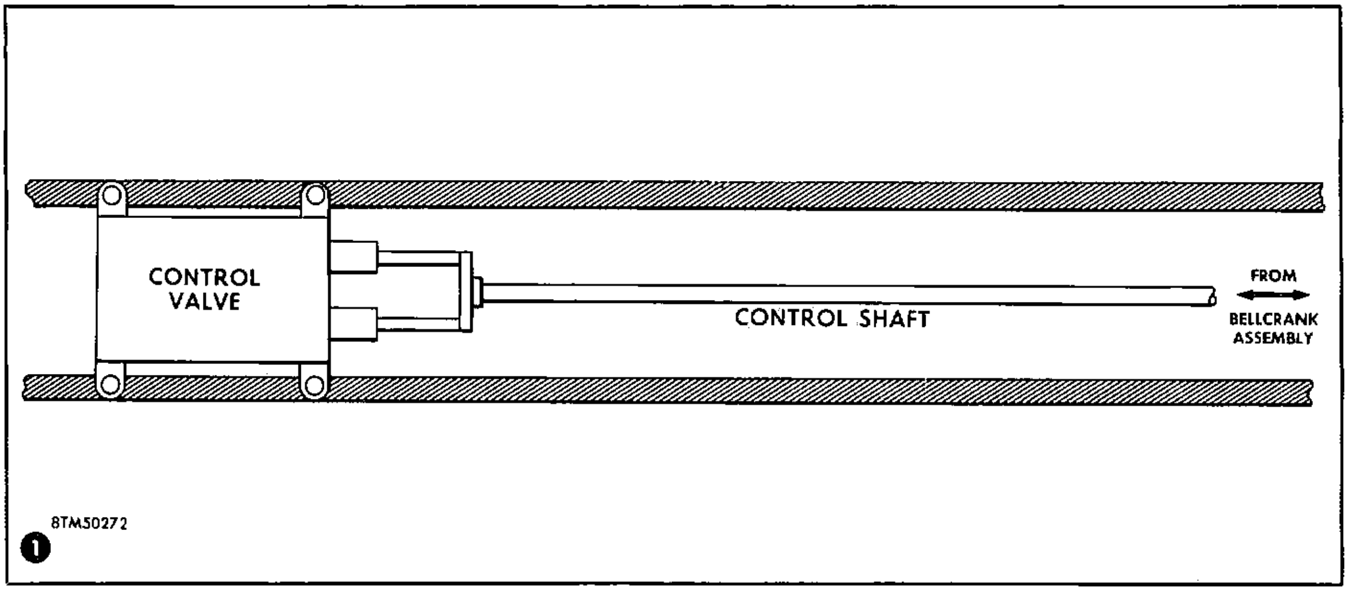

Figure 3-10. Diagram of Direct Pilot Control of Elevon Control Valve

control valve. You can see how this rod forms into a yoke around the lockout valve, and is attached to the elevon servo actuator. The servo actuator is then mechanically connected to the elevon control valve. Above this rod and the hydraulic components are the inboard elevon actuator and potentiometer. The potentiometer is part of the automatic control system which you will learn about later.

Note how the cylinder rod attaches to the inboard (upper) elevon horn. This horn pivots on the bolt just above the *aft* elevon bell crank. Pivoting of the bell crank is explained later when *feedback* is discussed. Below the hydraulic control components you can see the elevon tie rods that connect with the pilot's control stick.

Since the servo actuator and lockout valve do not function in *direct manual mode,* consider the rod between the aft elevon bell crank and the elevon control valve as being a solid shaft without the servo actuator or the lockout valve attached to it, as shown in figure 3-10. When the pilot moves the control stick, there is a corresponding amount of motion transmitted through this shaft to the valve. The amount of motion depends on the amount he has moved the stick. If the pilot moves the stick to either side (aileron motion), a mixer T-crank shown in figure 3-11 *rotates* to move the elevon control rods fore and aft. If he pushes the stick forward or pulls it back (elevator motion), the same mixer tee travels fore or aft to move the control rods in the same manner.

This mixer T-crank moves elevon control rods through mechanical linkage, in the fore and aft directions regardless of the particular control that moves the mixer tee; that is, aileron or elevator control initiated by the pilot. All the mixer tee does is answer thrust from these different pilot controls to move the control shaft and valve of each elevon (shown in figures 3-3 and 3-10) through the elevon control rods and aft bellcrank. These movements are in coordinated directions, or in directions opposite to each other.

In aileron action, the elevons move in directions opposite to each other. In elevator action, the surfaces move in the same directions as elevators. Then in a climbing or diving turn, they combine the movement of aileron and elevator together. At that time, as shown on figure 3-11, the mixer tee travels fore and aft as a complete unit along the airplane's longitudinal axis to move both elevons in the same direction as elevators; and at the same time this tee rotates to the right or left around a vertical axis to change the position of the two elevons in relation to each other as ailerons. The point made here is that regardless of *why* the elevons are moving or what flight control they are serving as, the function of the control shaft, control valve, and actuating cylinders is the same.

This point has been discussed here merely to show that coordinated motions of all controls are "mixed" at the tee to move both elevons in the degree and relationship necessary to their function as both elevators and ailerons. Keep in mind that these motions are transferred from the mixer tee through push-pull rods and a bell crank assembly to the one shaft on each elevon assembly; and that regardless of the purpose of the movement of the elevon (for elevator or aileron motion) its hydraulic actuation is the same.

Now, note on the diagram of the shaft on figure 3-10 that movement of the shaft toward or away from the control valve to which it is attached also moves the spools of the control valve. As the spools move, they

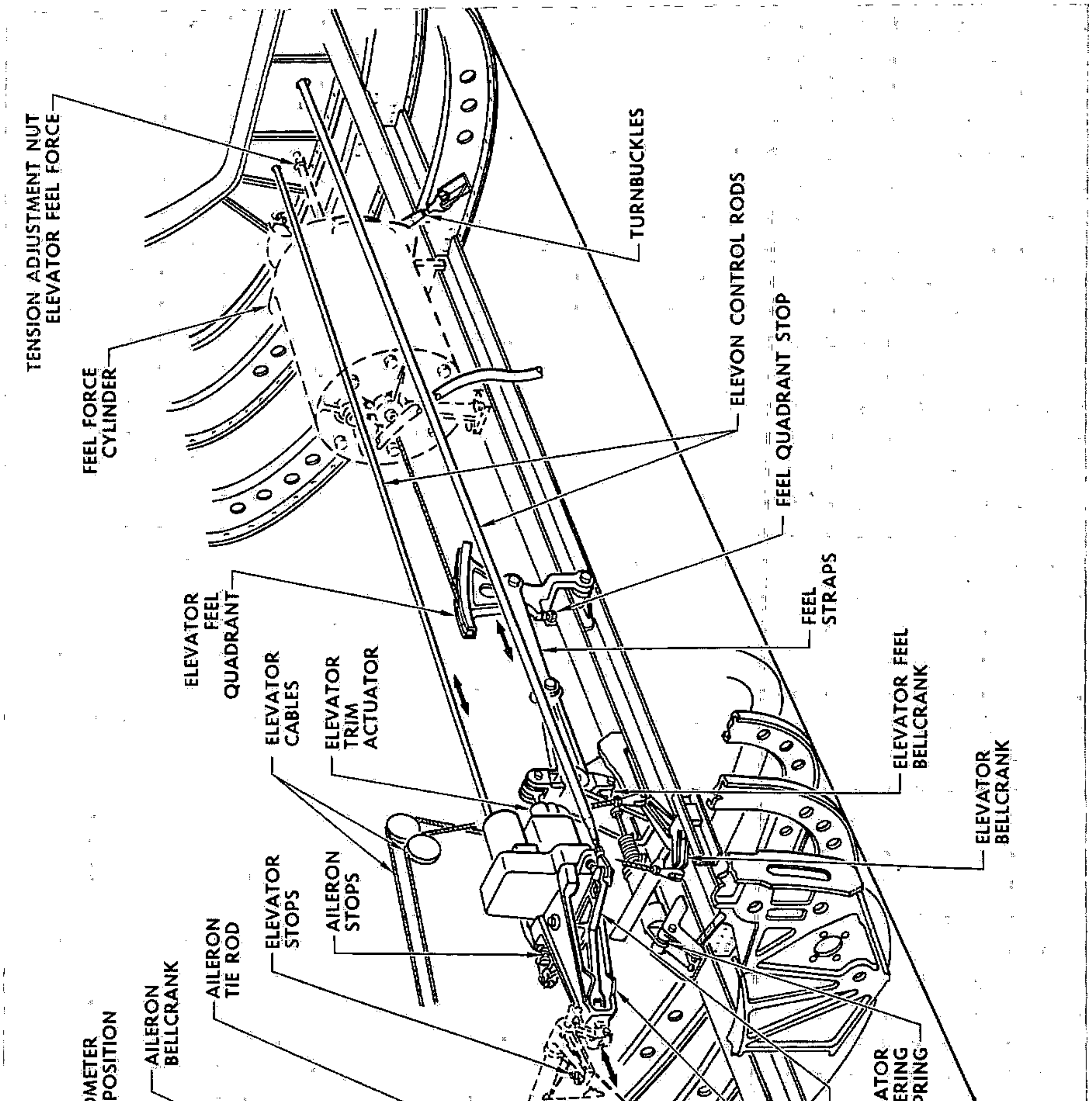

Figure 3-11. Action of the Elevator—Aileron Mixer Tee

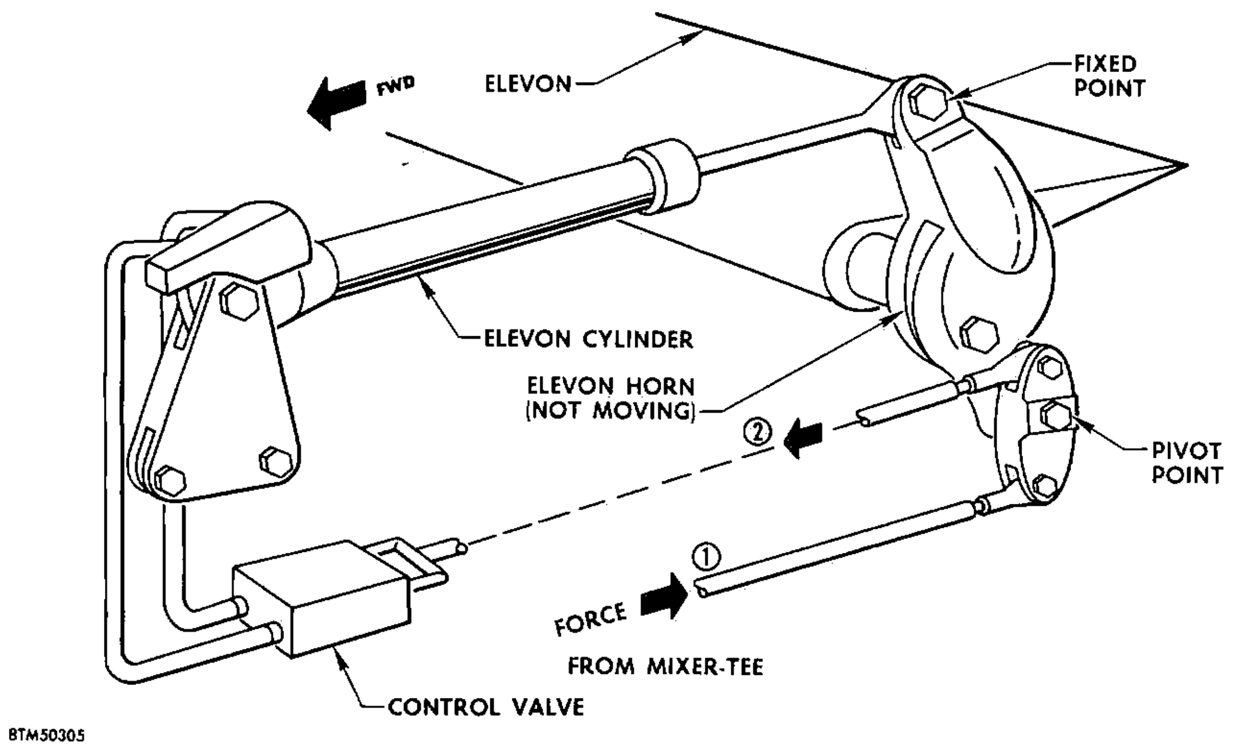

Figure 3-12. Elevon Mechanical Linkage Action—Initial Motion

port pressurized fluid to the actuating cylinders of the elevon, and also open ports to the return side of the power supply systems. The direction away from *neutral* that the spools move determines the direction in which the elevon actuating cylinders move the elevon.

Figures 3-12 and 3-13 show how the mechanical linkage directs the control valve to move from neutral, then return to neutral. Figure 3-12 shows the direction the bell crank moves in answer to a push from the mixer tee. The arrows show the direction the crank moves, and show its rotation around its pivot point. Note that the pivot point of the crank is attached to the elevon horn. The rotation of the bell crank moves the shaft that displaces the control valve spools. The spools admit fluid to the elevon actuating cylinder. The cylinder (inboard) retracts its shaft to move the elevon *up*. The outboard cylinder, as you know, extends at the same time for elevon up movement.

Now look at figure 3-13. The bottom of the bell crank is being held rigid in position by the pilot linkage from the tee and serves as the pivot point. As the top of the elevon horn moves forward the bottom part, to which the bell crank is attached, moves backward. With the lower end of the bell crank being held, the top half is carried back with the horn, and therefore pivots on its lower pivot to pull the control valve spool back to *neutral*. The spools automatically arrive at *neutral* at the same time the elevon arrives at the new position for which the pilot moved his controls. The reason this happens is that the mechanical linkage attached to the shaft incorporates the mechanical feedback, or follow-up linkage action we have just discussed.

Action of the elevon cylinder reverses the initial movement of the control valve by the pilot through this feedback action. The elevon remains in the new position until the pilot again moves the control stick, at which time the cycle of motion and stabilization at another position is again accomplished. When the control valve spools return to neutral, they port an equal though restricted flow to both sides of the cylinder pistons, thus locking both pistons in the actuating cylinders so that they are immovable and hold the control surface rigidly in position.

DAMPING CONTROL OF THE ELEVONS.

As you know, damping control of the elevons occurs when the *manual* mode is engaged. In this mode of control, the pilot does not have to correct for slight pitch variations to maintain stability. Control movements are made to some degree by signals from the damping system. This damping system moves the elevons through the servo actuator to correct stabilization before the pilot knows correction is needed.

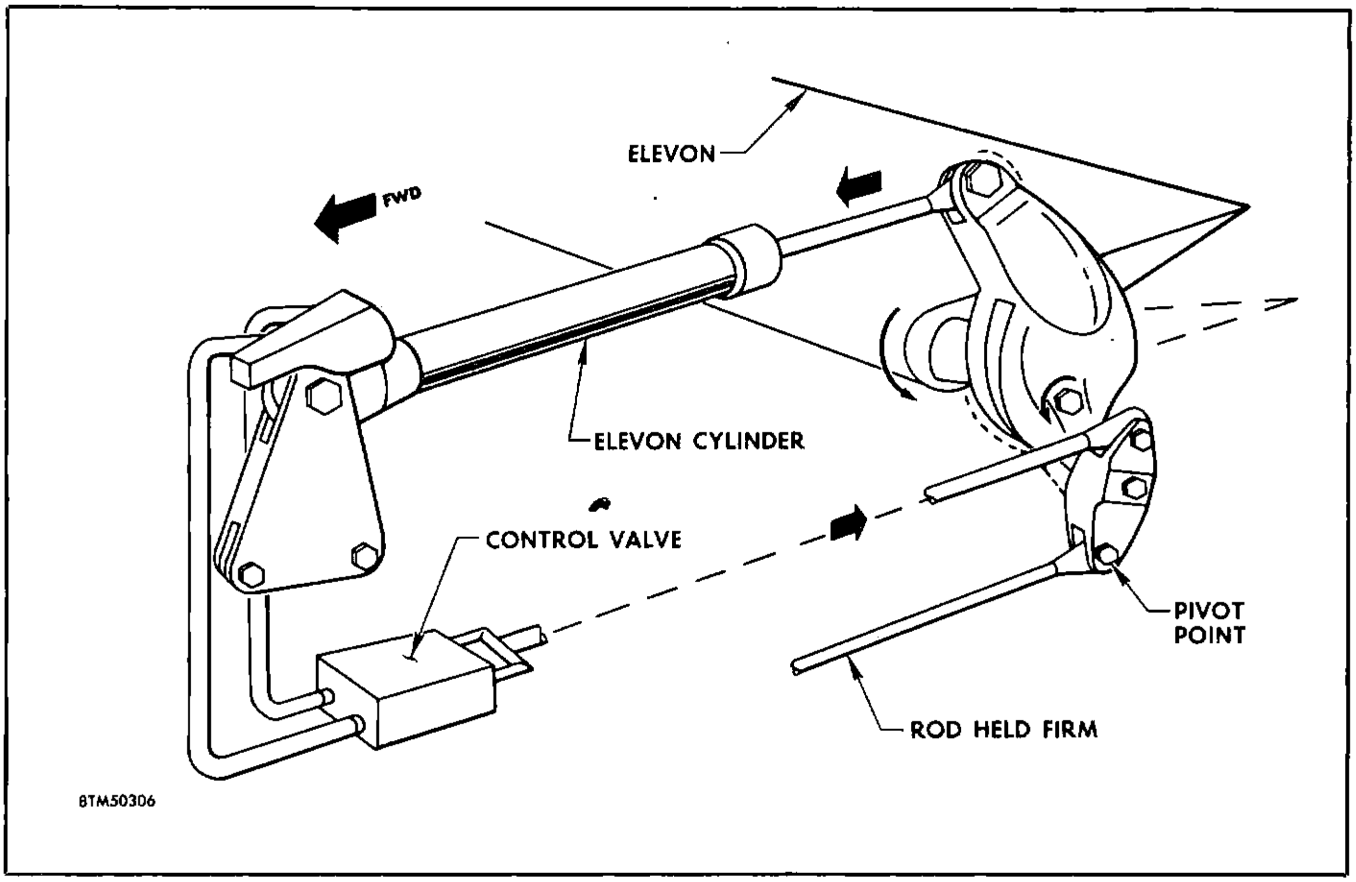

Figure 3-13. Elevon Mechanical Linkage Action—Feedback Motion

As you can see in figure 3-14, our control shaft now has another component. This is the servo actuator which is attached to the control valve. The servo actuator receives signals from the damping system and converts them into control motion which counteracts pitch motion to some degree. When only the pilot has control of the airplane, the servo actuator is rigid because the servo actuator shutoff valve is closed.

As you remember, when the servo shutoff valve is closed, the servo actuator is hydraulically locked in one position. At that time, since it is not functioning hydraulically, it acts merely as part of the straight shaft, or rigid link, between the mechanical linkage and the control valve. But now in the manual mode, it is serving to dampen pitch motion by moving the control surfaces. To do this, it is hydraulically energized by the opening of the servo actuator shutoff valve. When hydraulically energized, it is free to move and has hydraulic power to answer damping signals from the damping system.

These damping signals cause the servo actuator to displace the control valve partially and thus move the elevons for any corrective stabilization. At high speeds it is very easy for the pilot to overcontrol too much in the opposite direction when he corrects flight attitude. So the damping system counteracts slight pitch motion by moving the control valve through the movement of the servo actuator.

This damping control action of the servo actuator, however, does not mean that the pilot does not fly the airplane. He does, as in *direct manual* mode; but instead of having to correct for minor stabilization variations, the damping system relieves him of this task by sending signals to the servo actuator.

Servo Actuator.

Figure 3-15 shows the servo actuator as it appears in the control shaft. This actuator is comprised of a spring-loaded pilot spool, a spring-loaded slave piston, an electrical torque motor, and flow passage restrictors. In figure 3-15 note how the shaft connects to the elevon control valve. This shaft is part of the slave piston shown in the lower schematic view. When signals come from the damping system, this shaft can move either in or about 0.045 inch maximum depending on the corrective signal received. This 0.045 inch movement can displace the control valve spools about 1/8 inch, or 1/16 inch on either side of *neutral*. Thus you can see that this valve will affect elevon movement only to a very small degree (about one degree on each side of neutral).

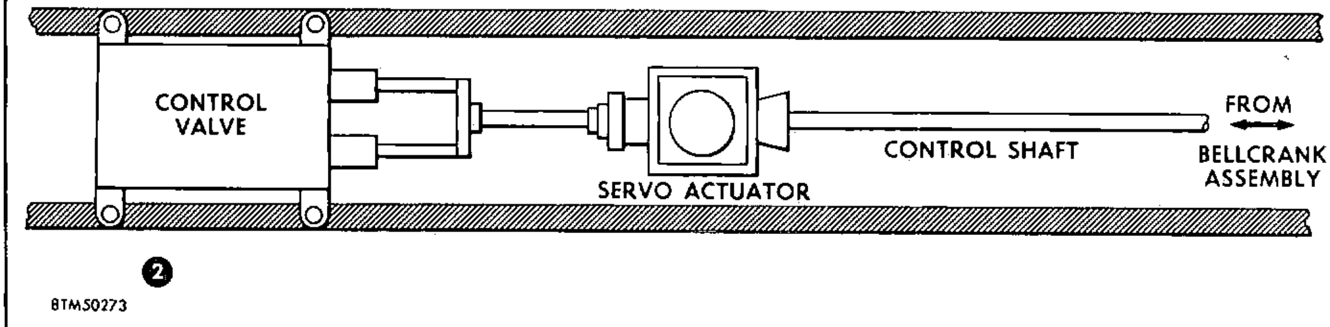

Figure 3-14. Elevon Hydraulic Damping Control Diagram

The other (aft) end of the servo actuator is rigidly connected to the control shaft so that pilot initiated motion of the elevons still displaces the control valve as in the *direct manual* mode. On top of the actuator is the torque motor that receives the corrective signals from the damper system.

SERVO ACTUATOR OPERATION. You learned earlier that the servo actuator cannot be energized hydraulically unless the shutoff valve is energized. So now in this *manual* mode the shutoff valve is open, and the servo actuator is energized hydraulically and is free to answer electrical signals from the damping system to control flight.

The servo actuator is shown in the *neutral* position in the schematic diagram on figure 3-15. Note that hydraulic pressure entering the actuator bleeds in around the center portion of the pilot spool and goes to the two nozzles at the flapper. When the torque motor is de-energized (no signal from the damping system), the flapper is an equal distance from the two nozzles. Pressure bleeds through the two nozzles then passes on out the return outlet.

However, when the damper system sends an electrical signal to the torque motor, the torque motor moves the flapper toward one of the two nozzles in an amount equal to the intensity (voltage) of the signal. As the flapper moves toward a nozzle, a fluid pressure unbalance is set up at the pilot spool, thus causing the pilot spool to move to one side. The pressure unbalance is set up because a back pressure is created at the restricted nozzle. As an example of this pressure unbalance at the pilot spool, let's suppose that the torque motor moved the flapper against the right nozzle. This flapper then restricts flow through the right nozzle, and at the same time in moving away from the left nozzle allows more flow through it. Pressure in the passageway between the restricted right nozzle and the right end of the pilot spool is increased because of this restriction.

This pressure increase causes the pilot spool to move to the left against the spring. Pressure which in *neutral* bled past both sides of the pilot spool, now is diverted exclusively to the right end of the slave piston causing it to move to the left. The resultant motion of the slave piston which is part of the servo actuator shaft moves the control valve spools away from *neutral*. This control valve repositioning admits fluid pressure to, and ports return fluid from, the appropriate sides of the elevon actuating cylinders. The elevon is thus moved to its new position to correct flight attitude.

As the elevon actuating cylinders move, the feedback potentiometer connected to the cylinder (figure 3-11) sends another electrical signal to the damping system. The signal intensity depends upon the distance the cylinder moves. This second signal is of opposite electrical sign to the first signal and tends to null it out. The second signal thereby rapidly reduces the differential current (opposite current) to the torque motor to zero. As the two signals balance they move the servo actuator flapper towards neutral.

When the two signals are completely balanced, the flapper has returned to neutral, again equalizing the fluid pressure on both sides of the pilot spool, thus causing the servo actuator slave piston to return to neutral. Consequently, the control valve spools have also been returned to neutral, but the elevon actuators have been locked in the new position called for by the original input signal to the torque motor. The elevon actuators

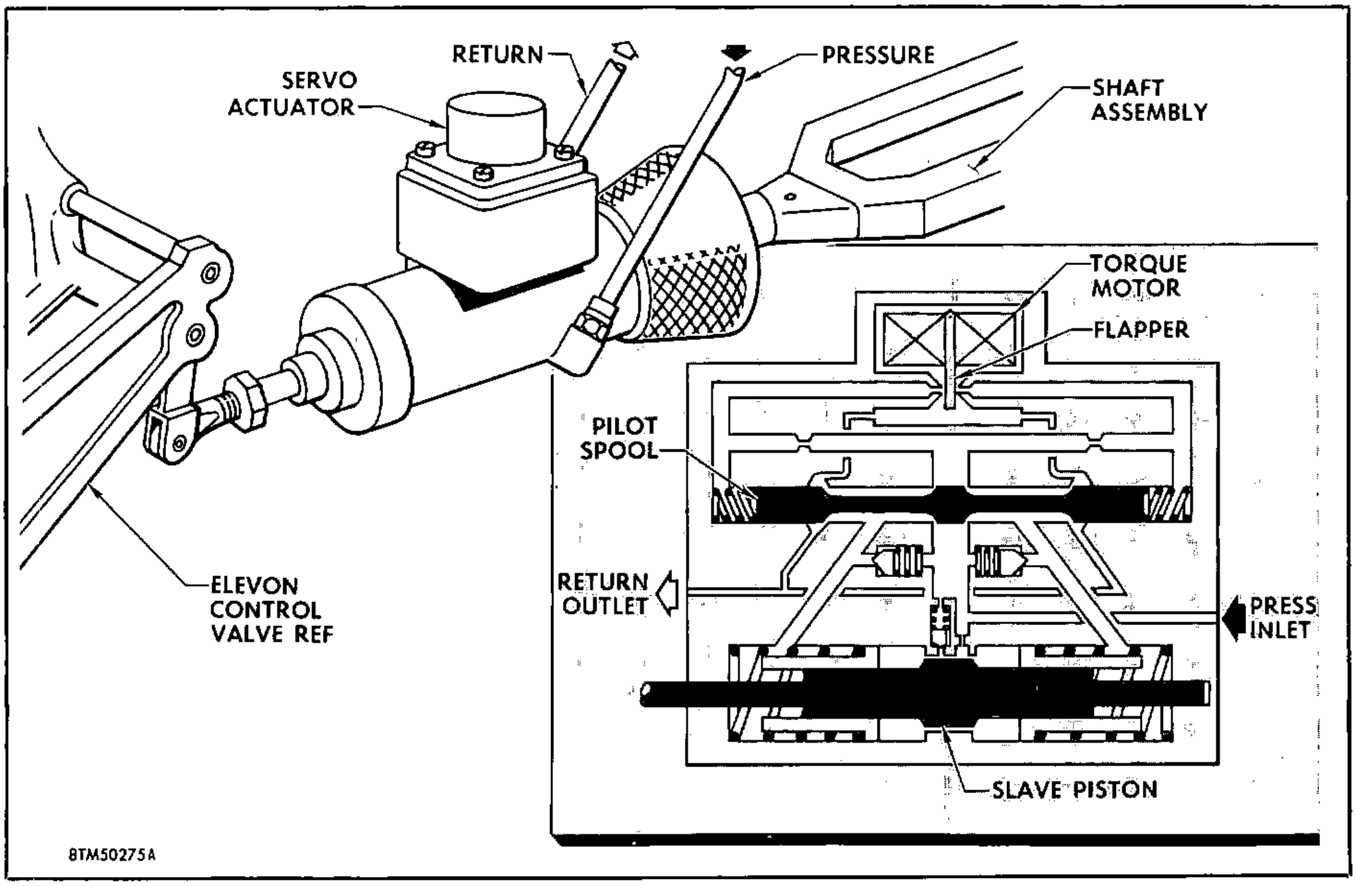

Figure 3-15. Elevon Servo Actuator

stay in the new position because the return side of the control valve spools are at neutral, thus locking the fluid on each side of the actuating cylinder pistons.

The only way the actuating cylinders can move further out, or back to neutral, is for the control valve to move again, so as to direct fluid pressure to the appropriate sides of their pistons. This control valve action, as you know, can either be originated by damping signals through the servo actuator or by pilot initiated motion.

In several instances, this chapter of the hydraulic manual has mentioned the electrical signal which control the hydraulic system for the control surface damping action. Because of the complexity of the electrical networks which control the pitch and yaw damper system and the pilot assist system, the circuits are not introduced in this manual. For a complete and comprehensive coverage of these circuits, refer to chapters 4 and 5 of the Flight Control Training Supplement.

MAINTENANCE. If the servo actuator does not function properly, or not at all, first check to see if signals are being received from the damping system. To do this, you should check with the maintenance personnel in charge of the automatic flight control equipment. They can tell you if signals are arriving at the servo actuator. If signals are being received, then the trouble is in the actuator. In this case you would replace the entire servo actuator assembly. Troubles in the actuator would either be electrical or hydraulic.

If the electrical circuitry and torque motor are satisfactory, and the secondary hydraulic system is delivering proper fluid pressure to the servo, then leaks or restrictions within the servo would cause the trouble. If exterior leaks are causing actuator malfunction you may be able to correct the difficulty without replacing the actuator. If a leak is discovered in the fittings, remember to return the hydraulic pressure to zero before attempting to tighten or replace the fittings. A fitting could crack while being tightened and cause personal injury if the line were pressurized.

To remove a faulty servo actuator, you must first return the hydraulic system pressure to zero. Then disconnect the two flex hoses from the servo, disconnect the electrical fitting, and remove the bolt connecting the push-pull shaft to the control valve mechanism. After everything has been disconnected, remove the servo by unscrewing the knurled collar nut from the back end of the assembly.

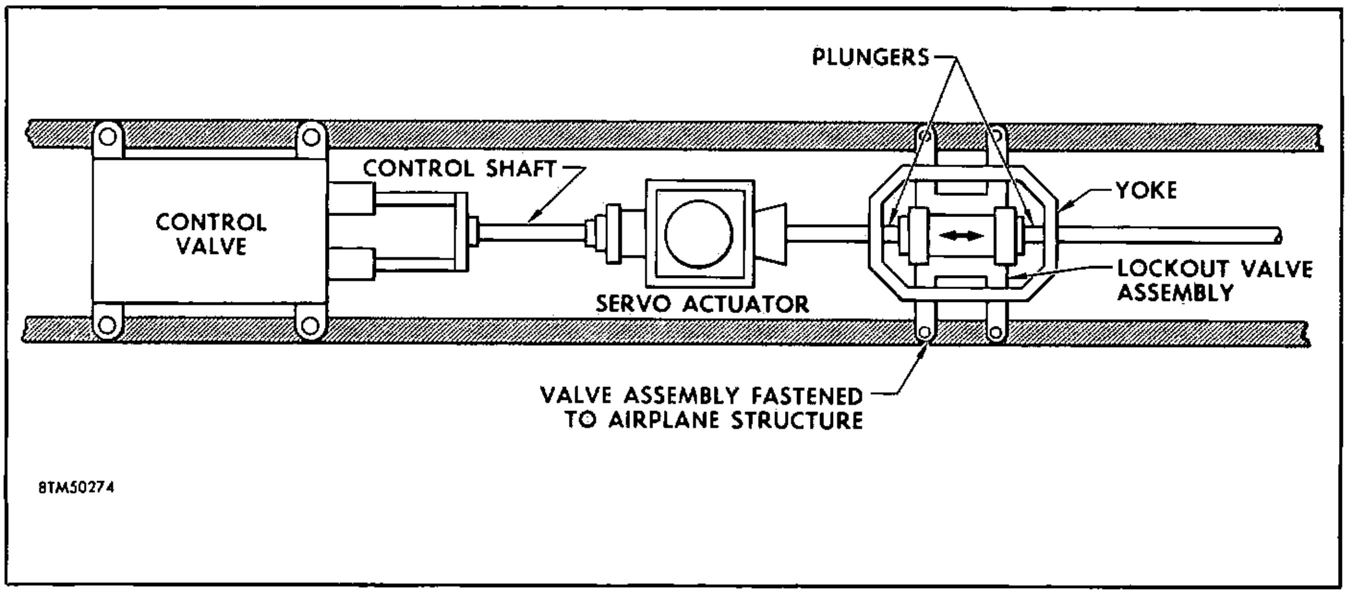

Figure 3-16. Elevon Hydraulic Pilot Assist Diagram

Before installing the new unit, you must first install two rigging pins in the system linkage. The forward pin inserts in the rigging pin holes in the control valve quadrant. These pin holes are located in the forks immediately above the point where the servo push-pull shaft connects to the elevon control valve, as shown in figure 3-8. To install the aft rigging pin you must first remove the upper push-pull rod connecting bolt from the aft elevon bellcrank assembly attached to the elevon horn.

With the bolt removed and the elevons in the *neutral* position, you can push the rigging pin through the bellcrank and rod until it enters the hole in the end of the elevon horn pivot bolt outboard of the bellcrank. Now you can install the servo assembly on the end of the push-pull control shaft and safety wire the knurled collar. With the rigging pins still in place, you then connect the actuator shaft to the control valve quadrant. If the actuator shaft connecting point is too far forward or too far aft, making it difficult to install the bolt, loosen the lock nut on the shaft and adjust its length to suit so that the connecting bolt can be installed. Don't forget to tighten and re-safety the lock nut. Then attach the flex hoses and the electrical connector and remove the rigging pins. Again be sure to refer to your F-102A Maintenance Manual (T.O. 1F-102A-2-3) for the step-by-step instructions on actuator replacement and check-out procedures for the system.

PILOT ASSIST CONTROL OF THE ELEVONS.

In the *pilot assist* mode of flight control we have another unit added to the elevon control system—the lockout valve. You will recall in the discussion of the pilot's direct control of the elevons that we considered the shaft as a solid link between the aft bellcrank and control valve, and showed it as such in figures 3-3 and 3-10. Then in *manual* mode another hydraulic unit was added—the servo actuator. In figure 3-16 you can see this last component that is being added—the lockout valve. Note that the control shaft has been changed to include a yoke. This yoke surrounds the lockout valve which is fastened to the airplane structure, and does not touch the valve when the airplane is being flown directly by the pilot or when being damped in the *manual* mode.

When the pilot assist system is engaged, however, this valve is energized hydraulically and extends two plungers that lock the control shaft so that pilot initiated motion of the control stick cannot move the shaft (figure 3-17). Control of the elevons then comes from the pilot assist system and corrections the pilot makes by the use of an electrical switch on the control stick. This *pilot assist* mode is similar in operation to that described in *manual* mode, except that it covers a wider range of control.

Elevon Lockout Valve and Yoke.

Figure 3-17 shows the lockout valve installed. Note how it extends up from the airplane structure and through the yoke of the control shaft. This valve cannot be energized hydraulically unless the pilot assist system is engaged. The shutoff valve which hydraulically energized the servo actuator also routes hydraulic pressure to the lockout valve (figure 3-10). The lockout valve consists of a lockout valve solenoid and two plungers, illustrated in figures 3-3, 3-6 and 3-10.

When the pilot assist system is engaged, the solenoid is energized and allows hydraulic fluid to enter the lower part of the lockout valve. This hydraulic pressure extends the two plungers that lock the control

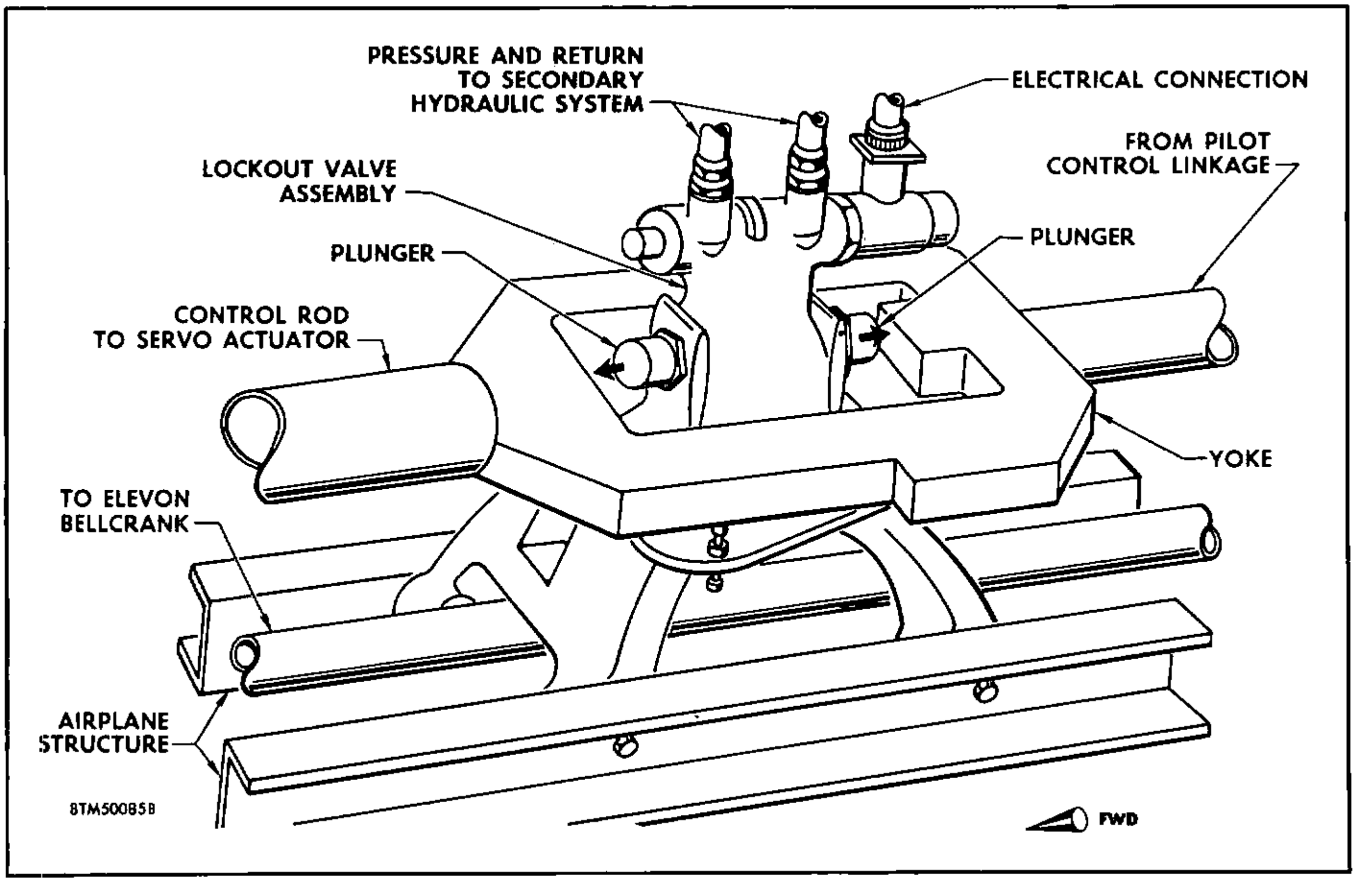

Figure 3-17. Elevon Lockout Valve and Yoke

shaft so that motion is not transmitted from the control stick to the elevon control valves. The pilot can, however, overpower this valve when necessary by exerting approximately 20 pounds overcontrol to the stick.

The maintenance of this valve is comparatively simple. If the plungers do not extend when the pilot assist system is engaged, or do not retract when the system is disengaged, you should first determine that power is reaching the solenoid for its operation. If power is at the solenoid, and if secondary hydraulic system pressure is up to standard, then the lockout valve is malfunctioning and will require replacement. Do not try to disassemble this valve to correct for malfunctioning; instead replace it.

This discussion of the lockout valve completes the units found in the elevon system. Now let's take up the hydraulic section of the rudder control system.

RUDDER HYDRAULIC SYSTEM.

The rudder control hydraulic system is basically the same as the elevon control system. However, there is one main difference. This difference is that the rudder system does not incorporate a lockout valve; therefore the rudder is always at the command of the pilot regardless of the *mode* of flight. Figure 3-18 shows all the components of the rudder hydraulic control system, in addition to other components of the rudder system, in the area at the base of the vertical stabilizer (fin). These rudder hydraulic system components include the rudder actuating cylinder, control valve, servo actuator, and servo actuator shutoff valve. For a more detailed view of the complete rudder system from the rudder pedals back, don't forget to refer to figure 3-4. The components operate on hydraulic power furnished by both the primary and the secondary hydraulic power supply systems.

In figure 3-18, note that the control valve and rudder actuator are incorporated within the same casting. These two units have interconnecting fluid passages between them. Note, too, that the servo actuator and shutoff valve are also one unit, and this unit is attached to the actuating cylinder-control valve assembly so that the rudder hydraulic system components are all contained within one package. Note in figure 3-4 how the rudder actuator attaches to the rudder horn. This rod is part of the actuator case. The piston rod, which moves in and out of the cylinder, is at the forward end of the actuator. (Refer to figure 3-4 for a better view.) The end of this rod attaches to the airplane structure so that when the cylinder actuates, the entire rudder package moves to displace the rudder. The four

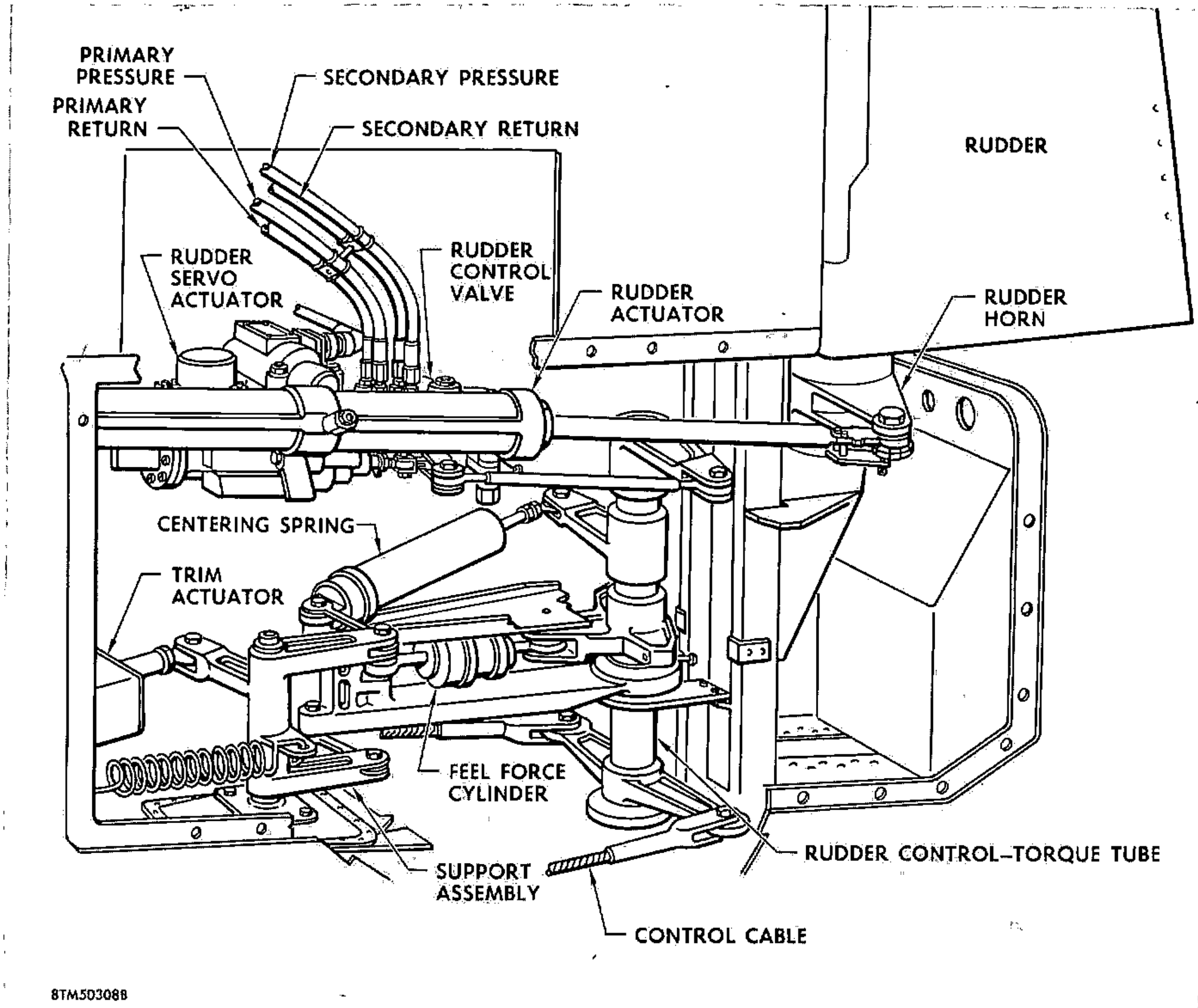

Figure 3-18. Rudder Hydraulic Control System Components

hydraulic lines (primary and secondary system) shown at the top of the actuating package are flex lines to permit the package to move.

HYDRAULIC SYSTEM OPERATION.

The hydraulic operation of the rudder system is directly affected by only two of the three modes of flight control—*direct manual* and *manual*—that you learned about in the elevon hydraulic system section earlier in this chapter. This system is indirectly affected by the third mode—*pilot assist*. When in *direct manual* mode, the control valve and rudder actuator move the rudder surface. When in the *manual* mode, the servo actuator moves the control valve and actuator to effect slight movement of the rudder surface in correcting for yaw stabilization.

Engagement of the pilot assist system does not directly affect the rudder hydraulic system operation; instead the pilot assist system sends signals only to the pitch damper system for either pitch (elevator) or roll (aileron) control of the elevon surfaces. The turn coordination feature of the yaw damper system (manual mode) is still in effect in pilot assist. Therefore, aileron movement of the elevons, as a result of pilot assist signals, causes an aileron position potentiometer to send signals to the yaw damper system. Thus the rudder moves indirectly as a result of pilot assist signals being applied to aileron control.

The schematic diagram in figure 3-19 shows the entire rudder hydraulic system. This system is contained in the rudder package previously described. Note the passages that route the flow of primary and secondary hydraulic system fluid from the control valve to the

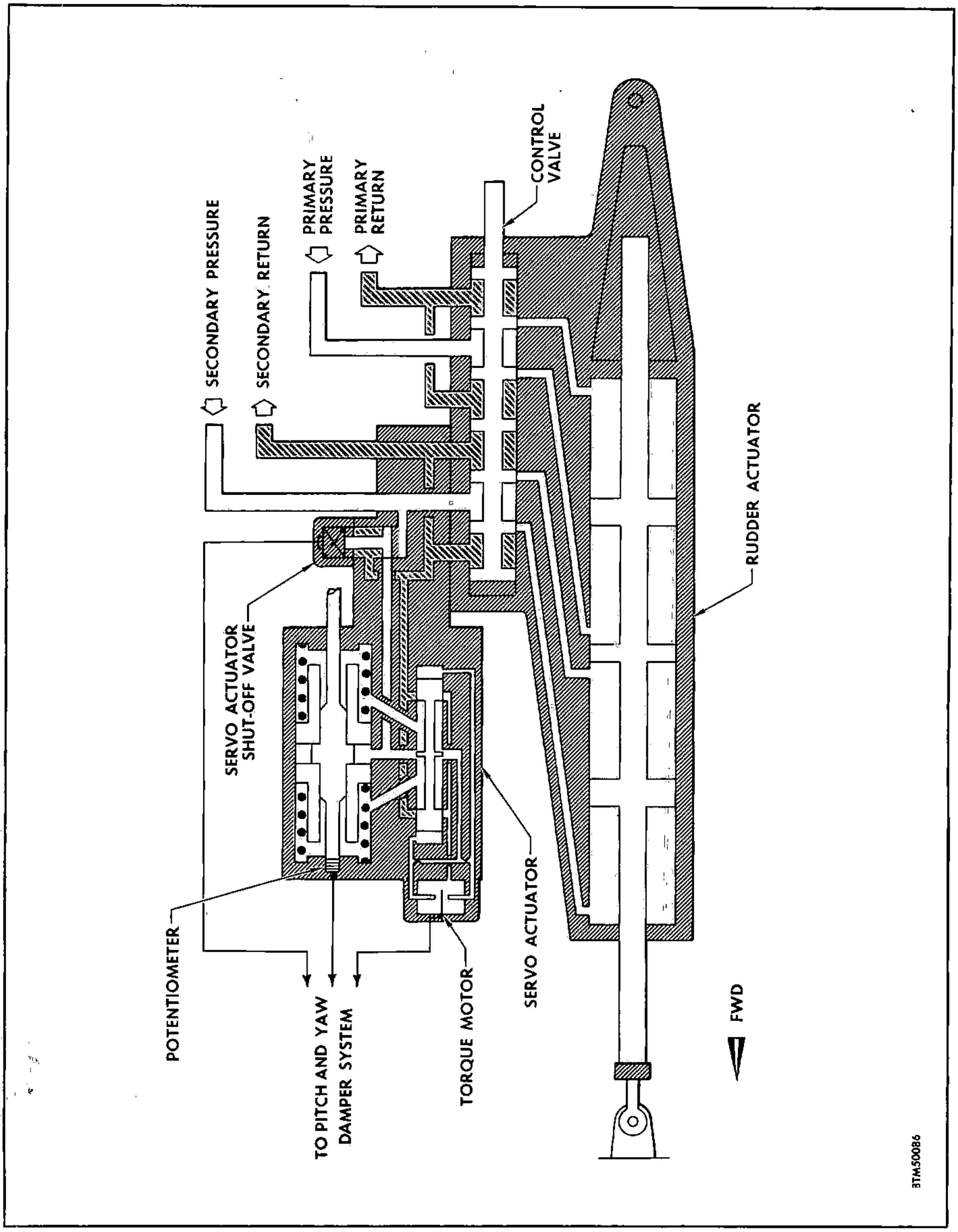

Figure 3-19. Rudder Hydraulic System Schematic

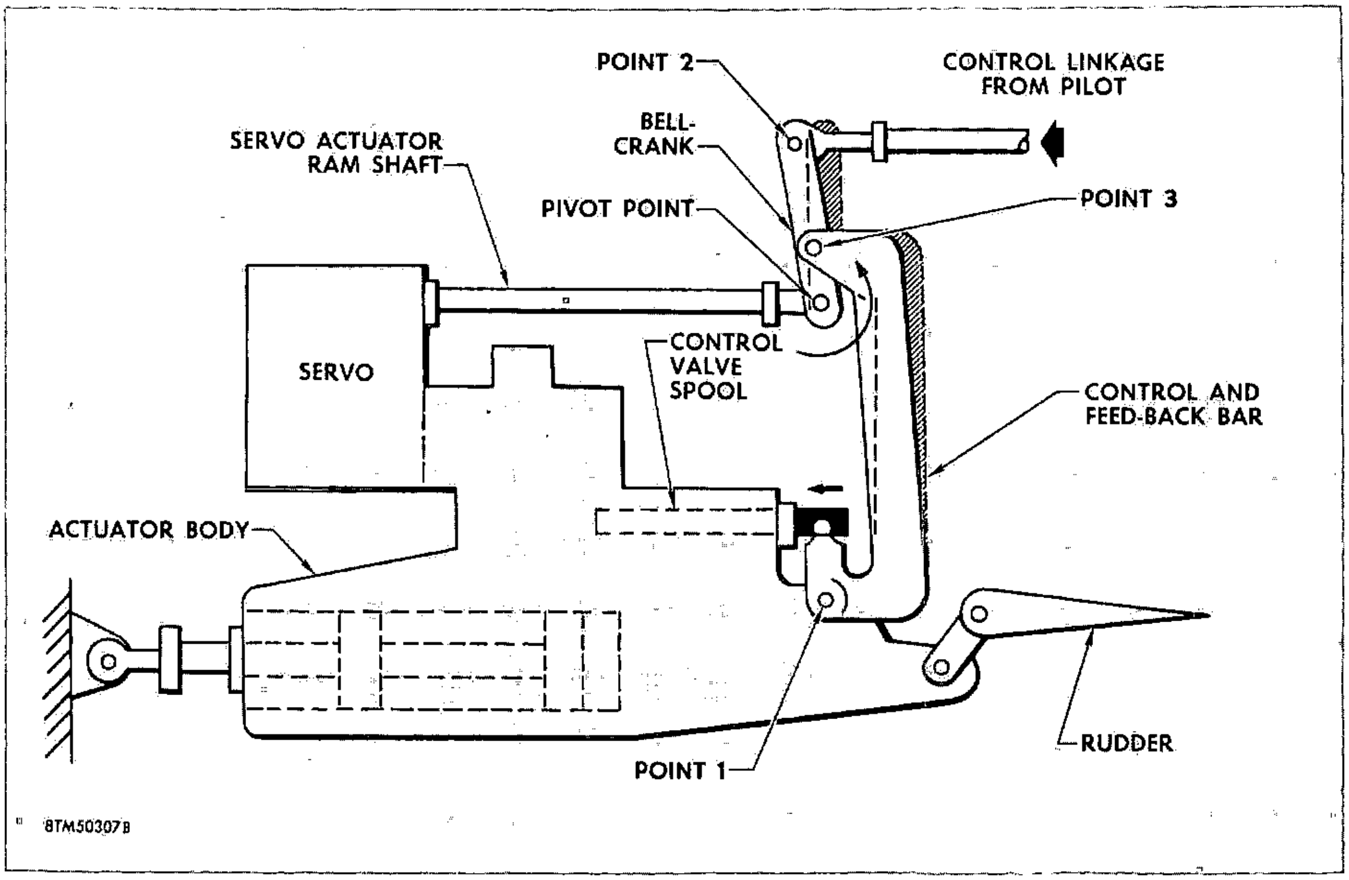

Figure 3-20. Rudder Mechanical Linkage Action—Initial Motion

appropriate sides of the rudder actuator pistons. Following the flow of fluid in figure 3-19 you can see that, like the elevon system, the primary system fluid powers—via one side of the control valve—only the aft side of the rudder actuator. The secondary system fluid powers the other (forward) side of the rudder actuator cylinder, also by way of the control valve. The secondary hydraulic system also supplies the hydraulic power needed for the operation of the servo actuator.

The side of each rudder actuator piston to which the primary and secondary system fluid is delivered depends on the direction in which the control valve spool is moved. Unlike the elevon actuators, the rudder actuator is of a balanced design so that an equal piston area exists on both sides of both pistons, as shown in figure 3-19. Therefore, equal reaction to hydraulic control forces is obtained in either direction. Since the piston rod of the rudder actuator is attached to the airplane structure and the cylinder body is attached to the rudder horn, hydraulic force applied to the actuator extends or retracts the actuator on its own shaft, causing the actuator cylinder itself to move the rudder. For clarity of rudder hydraulic system operation, we will discuss the control valve, actuating cylinder, and servo actuator as separate units.

The Control Valve and Actuating Cylinder.

The rudder control valve cylinder is divided into two sections, one for control of primary system fluid and one for control of secondary system fluid to their corresponding sections in the rudder actuator. On the rudder hydraulic system schematic (figure 3-19) note that the arrangement of the two control valves is the same as the actuator; that is, they are in tandem. This arrangement allows the movement of one double spool through both sections of the valve cylinder. Fore and aft movement of the spool in the control valve cylinder routes the fluid of both systems to the rudder actuating cylinder by way of drilled passages in the actuator casting.

Figures 3-20 and 3-21 show how the mechanical linkage connected to the rudder control valve and actuator operates. Figure 3-20 shows how initial motion, originated at the rudder pedals, moves the control valve spool which in turn moves the rudder. Figure 3-21 shows how *feedback* returns the control valve to neutral. When the pilot moves his rudder pedals in one direction or the other, motion is transmitted through the control linkage to the bellcrank shown in figure 3-20. This bellcrank pivots on the aft end of the servo actuator shaft (point 1). The bellcrank pivoting action

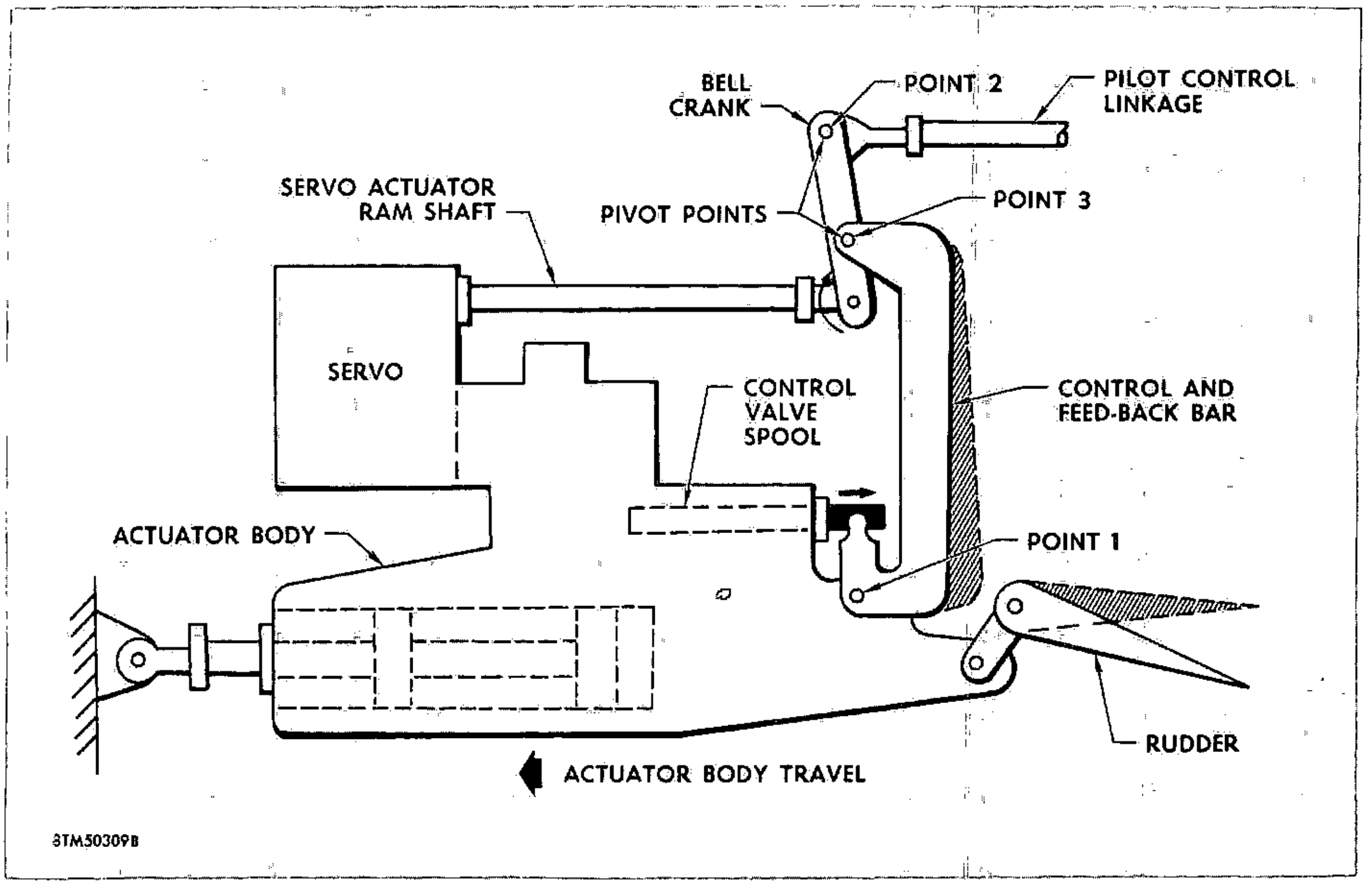

Figure 3-21. Rudder Mechanical Linkage Action—Feedback Motion Indirect Manual Mode

moves the upper end of the control and feedback linkage bar attached to it.

The control and feedback linkage bar pivots about its attach point (1) at the rudder actuator cylinder body. This pivoting action moves the short arm of the bar that you see attached to the control valve spool, thus displacing the control valve spool to the right. Depending on which direction the pilot moves his pedals, the control valve spool either moves in or out of the control valve. This control valve movement opens the internal passages to permit fluid pressure from both systems to enter the rudder actuator and move the rudder in a direction corresponding to the rudder pedal movement.

Now note in figure 3-21 the point where the control linkage to the pilot's pedals attaches to the bellcrank of the mechanical linkage (point 2). The pilot has stopped his pedals somewhere away from their neutral position, and is holding them there. So now, as the control valve is carried forward with the rudder actuating cylinder and body, the control and feedback linkage bar is pivoting at its attach point on the rigid bellcrank (point 3). This feedback bar is also pivoting at the point where it is attached to the body of the actuating cylinder (point 1).

As the actuating cylinder and body moves forward, the short arm on the bar gradually moves the control valve spool back to neutral. When the rudder finally reaches the new position that corresponds to the position being held by the rudder pedals, the control valve spool has completely returned to neutral and has closed off hydraulic pressure supply to the rudder actuating cylinder. When neutralized, the control valve also provides a hydraulic lock to hold the actuating cylinder in the new position. This position can be changed only by repeating the cycle of operation just described.

Since the control valve is incorporated within the casting of the actuator cylinder, malfunctioning of either section will necessitate replacement of the entire unit. If control surface motion becomes sluggish, and test stand hydraulic pressure is correct, the control valve is the most likely unit to be at fault. An inspection of the valve for evidence of clogged element screens or particles of foreign matter is indicated if the above malfunction occurs. There is also the possibility that an O-ring is faulty. The latter is true also if leakage is noted around the rudder actuator cylinder. Air in the system will also cause malfunctioning. If air is present in the system, it must be bled off by repeated cycling of the rudder system.

Servo Actuator.

As in the elevon system, the servo actuator is used for modes of control other than direct manual. However, there are actually only two modes of control that the rudder is directly operated in—the *direct manual* mode and the *manual* mode. In the third mode—*pilot assist*—the rudder is moved indirectly by signals sent from the aileron position potentiometer through the pilot assist system. In both manual and pilot assist modes signals from these systems go to the servo actuator. Since the rudder system does not incorporate a lockout valve, the hydraulic operation is the same for both modes.

We have just discussed the direct manual mode of operation in which the servo actuator is locked in a rigid position, and serves as a pivot point for part of the mechanical linkage. Now we will see in the following illustrations and discussion how the linkage moves in the manual and pilot assist modes of operation. In these modes of operation the pilot does not move his rudder pedals. Instead, the yaw damper system (in manual and pilot assist modes) sends signals to the servo actuator which in turn controls the displacement of the rudder control valve.

As previously mentioned, a comprehensive coverage of the electrical control circuits for the pitch and yaw damper system and the pilot assist system is given in the Training Supplement on the F-102A Flight Control System. Since the rudder linkage is spring-loaded to neutral, as shown in figure 3-18, the point where the pilot control push-pull linkage connects to the bellcrank of the rudder linkage (point 2) serves in these modes as a pivot point, except when the pilot overrides or boosts rudder motion while the servo actuator is in service. In that instance, the pivot point would vary between the top and bottom of the bellcrank (either point 2 or point 3).

The servo actuator is bolted to the rudder actuating cylinder casting and is controlled by a solenoid-operated, three-way servo actuator shutoff valve. This shutoff valve is located between the servo actuator and the secondary hydraulic system pressure line to the control valve. As you can see, the servo actuator shown in figure 3-22 is the same internally as the elevon servo actuators shown in figure 3-15. It is comprised of a spring-loaded pilot spool, a spring-loaded slave piston or spool, flow passage restrictors, and an electrical torque motor. The servo actuator is energized hydraulically through its shutoff valve when the pilot selects the manual or pilot assist mode of operation. After it is energized hydraulically it can then respond to electrical signals from the yaw damper system.

The yaw damper system signals cause the servo actuator to operate hydraulically in the same manner as the elevon servo actuator. When these signals move the servo actuator piston, the piston ram moves the rudder bellcrank to pivot it off the pilot-held push rod (point 2). You can see in figure 3-23 how the bellcrank then moves the attached control and feedback bar to operate the control valve spool as described in direct manual mode. The servo actuator accomplishes this action as follows: signals from the damper system energize the torque motor which in turn moves the flapper, shown on figure 3-22, toward one nozzle or the other in an amount depending on the intensity (voltage) of the signal.

As the flapper moves toward a nozzle, a fluid pressure unbalance is set up at the pilot spool. This pressure unbalance causes the servo actuator pilot spool to move in the desired direction to port fluid pressure to the correct side of the servo actuator slave piston or spool, and to port fluid from the other side of the piston to return. Consequent motion of the servo actuator slave piston pivots the system bellcrank and feedback linkage bar to position the control valve spool away from neutral. The control valve repositioning admits fluid pressure to, and ports return fluid from, the appropriate sides of the rudder actuating cylinder piston. The rudder actuator then moves the rudder to the new position.

As the rudder actuator moves, a feedback potentiometer connected to it sends another electrical signal, the voltage of which is determined by the amount of actuator displacement, to the yaw damper system. This second signal is of opposite electrical sign to the first signal and tends to null it out. The second signal thereby rapidly reduces the first signal to the torque motor to zero. As the two signals balance, they move the flapper towards neutral. When the two signals are completely balanced, the flapper is again at neutral and equalizing the fluid pressure, thus causing the servo actuator pilot spool and slave piston to also return to neutral.

The control valve has also been returned to neutral by reverse action of the linkage, but the rudder actuator has been locked in the new position called for by the original input signal to the torque motor. The effect on the mechanical linkage is indicated by arrows in figure 3-24. As you can see, the push rod from the pilot's controls has not moved, so the bellcrank pivots at that connection (point 2). The servo actuator ram retracts to draw the lower end of the bellcrank back to its original position. This action makes the control and feedback bar also travel back to its original position, pivoting at both of its ends (points 1 and 3).

Of course, as the actuator assembly moves in response to the hydraulic actuation force, you can see that it will also contribute to the feedback action of the control and feedback bar, as described in the manual mode condition of control. The rudder actuator stays in the new position that has been established, because return

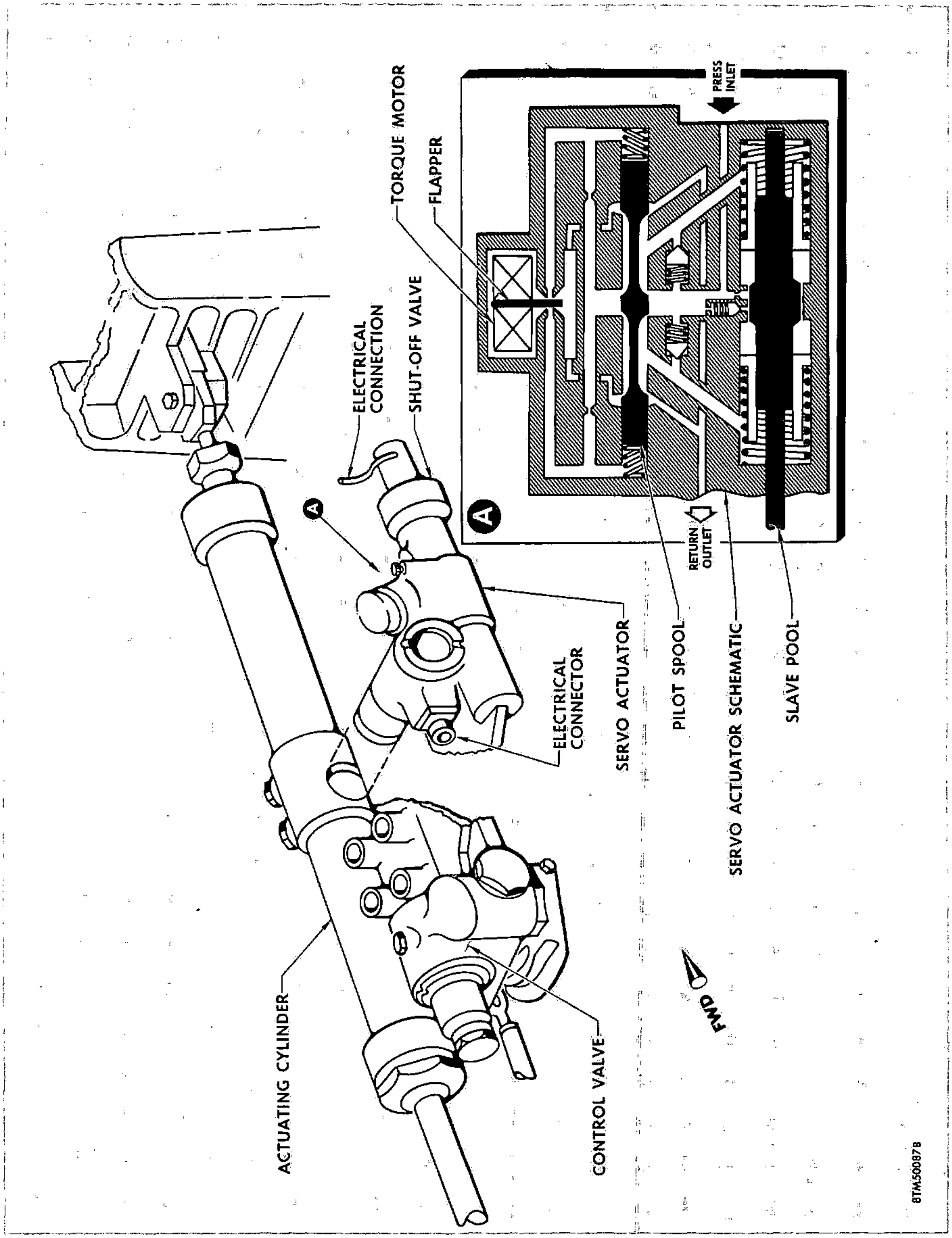

Figure 3-22. Rudder Servo Actuator

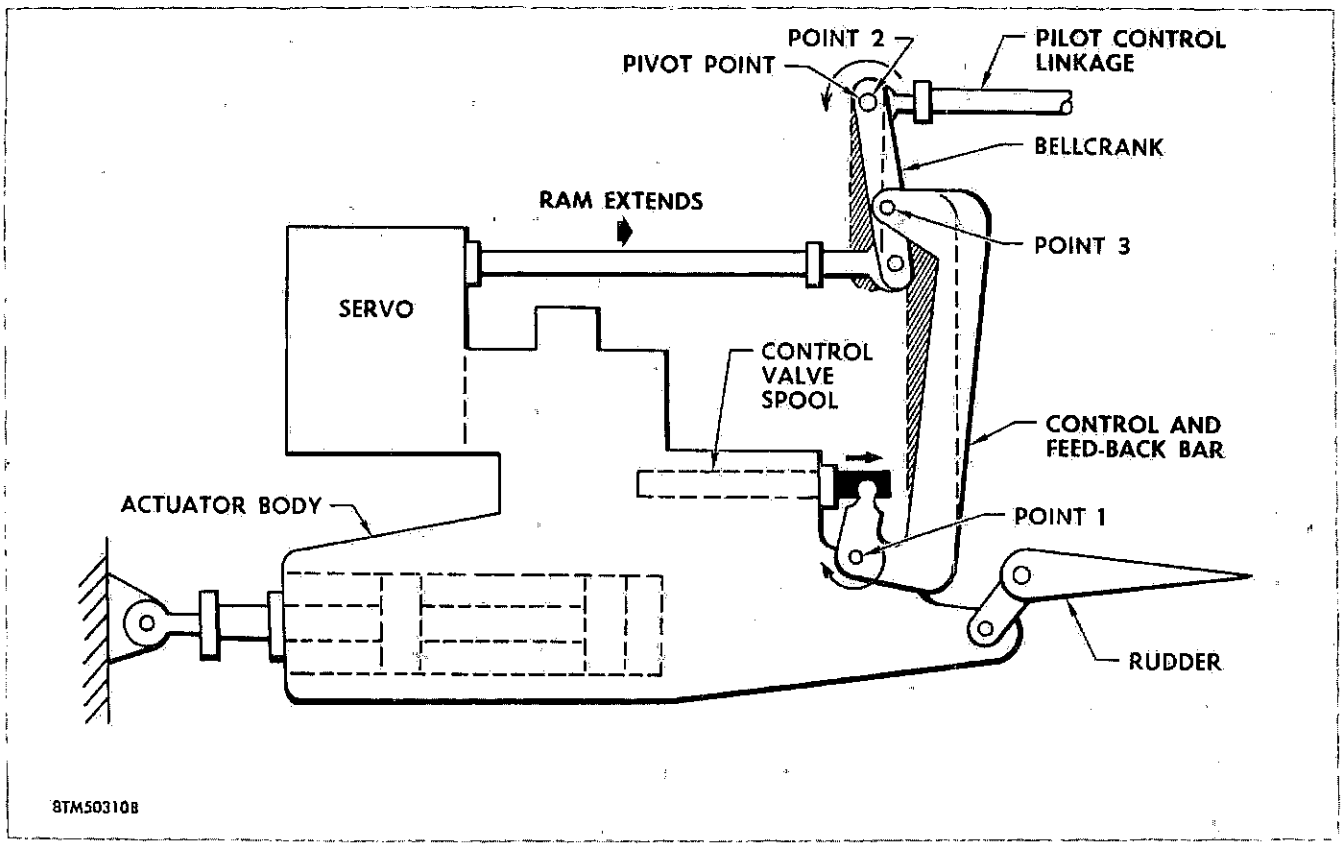

Figure 3-23. Rudder Mechanical Linkage Action—Initial Motion in Manual and Pilot Assist Modes

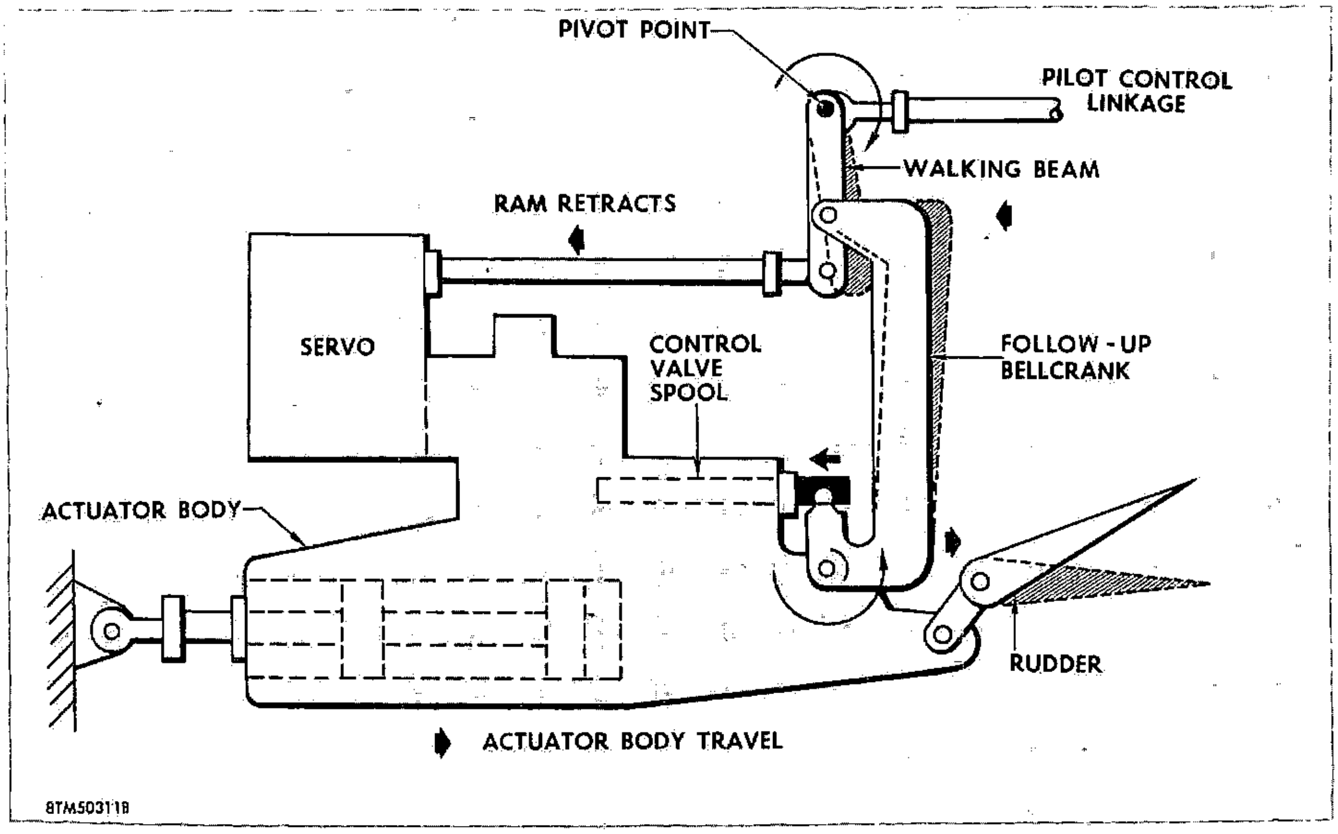

Figure 3-24. Rudder Mechanical Linkage Action—Follow-up Motion in Manual Mode

of the control valve to neutral releases and then maintains a restricted but equal amount of pressurized fluid to each side of the actuator pistons. This locks the actuator immovably in its new position and the only way it can be repositioned is to again move the control valve spools.

MAINTENANCE.

During maintenance of the rudder hydraulic system, here are a few points to bear in mind. A faulty servo actuator will cause the control surface to jump instead of moving in a smooth motion. This condition can be caused by improper rigging, or mechanical interference. After checking for the two latter conditions, check for air in the servo actuator by cycling the rudder actuating cylinder through its limits until any air that exists in the system is bled off. If this does not correct the condition, the servo actuator is probably faulty and must be replaced. You must remember that when any hydraulic lines are disconnected for component replacement or for any other reason, the rudder hydraulic system *must* be bled of air.

Faulty electrical circuits to the servo actuator shutoff valve, or a faulty shutoff valve, may prevent release of hydraulic power to the servo actuator. A continuity check of the electrical system will help to determine if this is the trouble. An indication of a trouble of this type can be detected when the system engages, and then disengages, before it is supposed to. The *least likely reason* for this particular malfunction would be air in the hydraulic fluid in the servo actuator.

Always refer to your Maintenance Manual, T.O. 1F-102A-2-3, for instructions on trouble shooting and checking out the rudder hydraulic system. You should also refer to the Maintenance Manual for specific instructions on the replacement and adjustment of all components in this rudder system and the elevon hydraulic system.

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Chapter IV

HYDRAULIC SUBSYSTEMS

Contents *Page*

In Chapter II you learned about the primary and secondary hydraulic power systems and how they produce fluid pressure for their operating subsystems. Then in Chapter III you saw how the pressure from these two power sources operated the elevon and rudder hydraulic subsystems. Now, in this chapter you will learn about the four subsystems and their components which are operated solely by the secondary hydraulic power system.

These four subsystems consist of the main and nose landing gear hydraulic system, the nose wheel steer-damper hydraulic system, the speed brake hydraulic system, and the emergency a-c generator hydraulic system. All mechanical and electrical phases that are associated with the above subsystems are mentioned or discussed as necessary so that you will better understand how the hydraulic systems function.

As you know, this Maintenance Supplement is designed to give you the hydraulic systems and their operation, so if you wish to learn the mechanical and electrical aspects of these systems you can refer to other Maintenance Supplements in this F-102A training series. This final chapter is concluded with a discussion of operational check-outs and what they have to offer you, the maintenance technician.

LANDING GEAR HYDRAULIC SYSTEM.

The landing gear hydraulic system, which is powered by the secondary hydraulic system, raises (retracts) and lowers (extends) the F-102A landing gear for normal operation. This system is controlled electrically by the corresponding up and down movement of a control handle on the extreme left side of the instrument panel and just above the left console. The high pressure pneumatic system furnishes air pressure for emergency extension of the landing gear. Air pressure will not retract the gear, it must be done hydraulically. Emergency extension of the landing gear is controlled by a handle at the lower left of the instrument panel and inboard of the normal operating handle. Pulling this handle pneumatically extends the gears in an emergency when the secondary hydraulic system malfunctions.

The F-102A landing gears are much the same as the gears on all modern aircraft. This airplane uses the retractable tricycle-type of gear. The main gears retract inboard and up into wheel wells in the wing and fuselage; the nose gear retracts forward and up into the wheel well in the fuselage nose. The wheel on the nose gear is hydraulically turned for steering on the ground and is controlled by rudder pedal action. This steering system is discussed later in this chapter.

The landing gear hydraulic system—one of the five hydraulic subsystems on the F-102A—consists of the main landing gear (MLG) hydraulic system and the nose landing gear (NLG) hydraulic system. Both of these systems function simultaneously and in an identical manner. You may think of these two systems as being separated into three basic sections, or units; the actuating cylinders that raise and lower the gears, the actuating cylinders that open and close the gear doors, and the electrically-controlled selector valves that control the hydraulic pressure to the gear and the door actuating cylinders.

Four selector valves are used in the landing gear hydraulic system, two in the main gear system and two in the nose gear system. One of the selector valves in each gear system controls fluid pressure to the gear actuating cylinders, while the other selector valve in each gear system controls pressure to the gear door actuating cylinders.

ELECTRICAL CONTROL OF THE LANDING GEAR HYDRAULIC SYSTEM.

Mechanically actuated electrical limit switches control the selector valve operation and *sequence* the operation of the gears and gear doors during extension and retraction cycles. These limit switches are mounted in the main and nose wheel wells in such positions that they will be actuated by contact of the landing gear doors or the landing gear itself. As a result, when the landing gear control handle is moved to the gear-down position, electrical circuits are completed first to the door selector valve solenoids. The doors move down to the open position and actuate certain limit switches. These limit switches then complete electrical circuits to the landing gear selector valve solenoids. These selector valves then admit hydraulic pressure to the landing gear actuating cylinders to extend the gear. In other words, landing gear extension is delayed until the doors are open and a free path has been made for the gear. With the gears down and locked, electrical power remains on the selector valves so that the gear actuating cylinders are pressurized to hold the gears down and locked.

The retraction sequence of the landing gear doors is the reverse of extension, the gears retract, then the doors close. However, other limit switches open the electrical circuits to the gear and gear door actuating cylinder selector valves when the doors are closed. Therefore, hydraulic pressure is not maintained in either the gear or gear door cylinders when the gears are up.

GEAR AND GEAR DOOR UP-LOCK CONTROL.

You are probably wondering by now what holds the landing gear and doors in the closed position. This lock control is accomplished in three different ways. The nose landing gear strut is held in the up position by an up-lock in the knuckle of its drag brace assembly. The lock engages and disengages due to initial motion of the actuating cylinder. This locking action is accomplished through the action of a bell crank mechanism on the gear trunnion which locks the gear in both the up and down position.

The nose gear door actuating cylinder incorporates an internal mechanical (ball) lock within its cylinder head end; and when the door closes completely the lock within the door cylinder engages and supports the door in the closed position. Mechanical overcenter locks in the main landing gear door mechanisms lock these doors in the closed position and hold the landing gear itself in the retracted position.

Hydraulic pressure to the main landing gear actuating cylinders maintains the gears in the up position until their doors have completely closed and locked. Limit switches are then actuated by the doors to neutralize both the gear and gear door selector valves. Neutralizing of the selector valves shuts off hydraulic pressure to the door and gear actuating cylinders.

The bearing point of support for the main gears in the retracted position is a Teflon (inert plastic) block attached to the wheel axle. This block bears on a plate attached to the inside center of each main gear door when hydraulic pressure is removed from the landing gear and gear door actuating cylinders.

The preceding discussion has familiarized you generally with the F-102A landing gear system, and will help you in understanding the hydraulic system as we go through it in the following pages. This manual, as you know, is devoted to the hydraulic systems in the F-102A. If you wish to learn all the aspects of the landing gear system, refer to the Airplane General Maintenance Supplement of this training series.

HYDRAULIC SYSTEM OPERATION.

You learned in the preceding discussion that the landing gear hydraulic system is powered by the secondary hydraulic systems and is separated into three basic units; the selector valves, the gear door actuating cylinders, and gear actuating cylinders. Figure 4-1 shows the entire landing gear hydraulic system and these basic units together with their restrictor valves. In the schematic, note that the secondary hydraulic system pressure and return lines connect to each of the four selector valves. Since all four selector valves are identical, only one valve is shown schematically to show flow through the valve. You will also note that each actuating cylinder has a restrictor valve in its two fluid lines, except the nose gear door cylinder. This cylinder has a restrictor valve, a restrictor check valve, and a relief valve. The peculiarities of this particular section of the gear hydraulic system will be covered separately.

Gear Down Operation.

Secondary hydraulic system is always present at the four selector valves with the engine running. When the landing gear control handle is placed in the *gear down* position, the No. 2 solenoids on the MLG and NLG door selector valves are energized electrically and route pressurized fluid through the valves, as shown in the typical MLG selector valve schematic (figure 4-1). When the No. 2 solenoid is energized, it pulls the pilot spool to the right and allows hydraulic pressure to go to the left side of slave piston. This pressure displaces the slave piston so that pressure can then pass through the valve to the actuating cylinder. Energizing No. 1 solenoid reverses the pilot spool position which in turn reverses the slave piston position so that pressure is routed to the other side of the actuating cylinder. With both solenoids de-energized, the pilot spool and slave piston return to their spring-loaded neutral position.

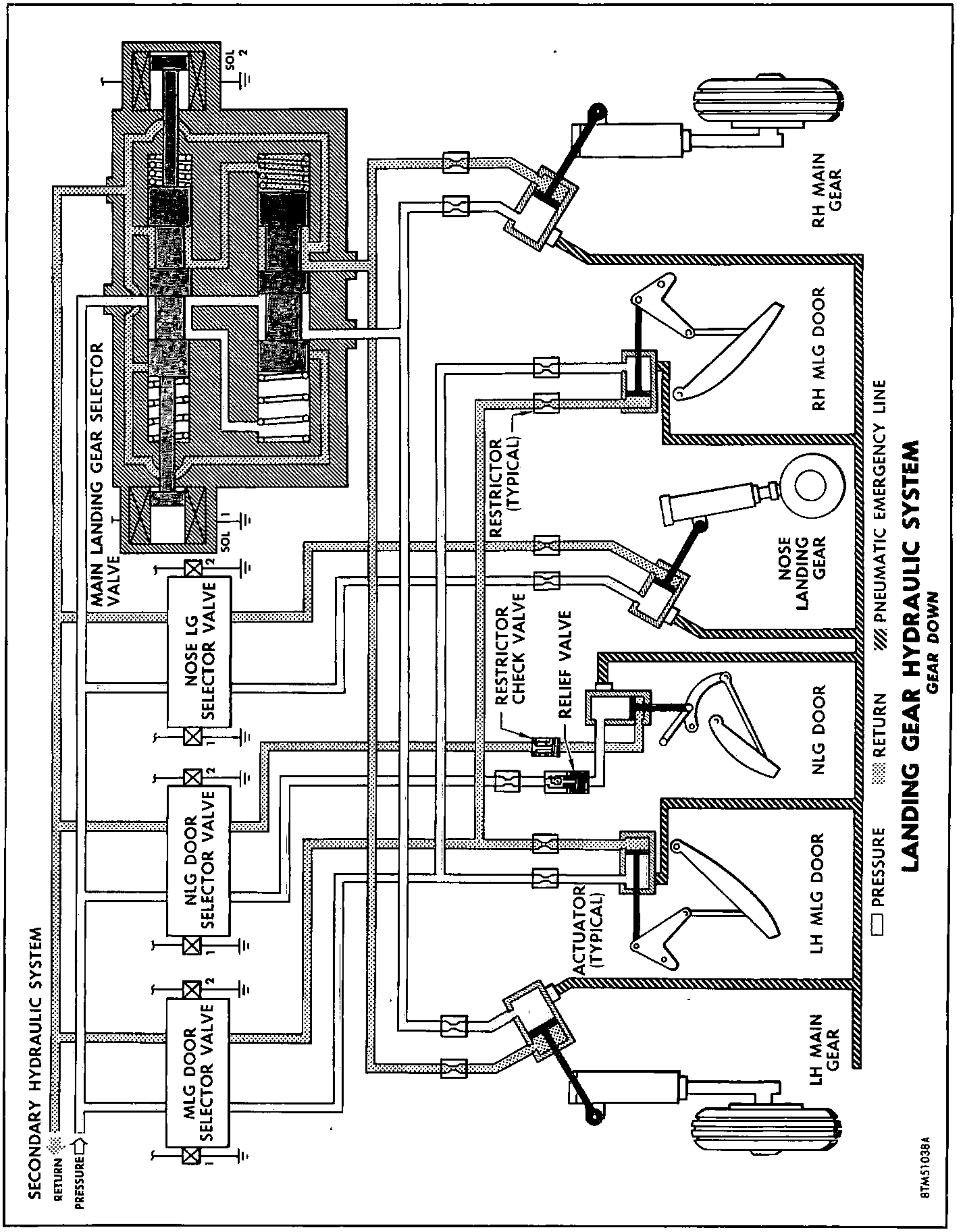


Figure 4-1. Landing Gear Hydraulic System Schematic

The pressure from the MLG door selector valve then goes to the MLG door actuating cylinders (two on each door) to retract the cylinders and open the doors. Pressure from the NLG door selector valve extends the nose gear door cylinder to open this door. The restrictors in the NLG and MLG door *open* and *close* lines limit the flow of fluid pressure to the actuating cylinders to control the speed of the door operation so that they will not "slam" open.

When the gear doors reach their full open position, they mechanically actuate limit switches that energize their respective gear selector valves. Now, with the No. 2 solenoids in the NLG and MLG selector valves energized, fluid pressure flows throw the valves to extend the actuating cylinders and lower the three landing gears. Final action of the extending cylinders moves bell cranks on each gear to actuate the gear down locks. Restrictors in gear *up* and *down* lines also control the speed of the gear operation. Hydraulic pressure remains on the gears in the *down* position as long as pressure remains in the secondary system (engine running).

Gear Up Operation.

When the landing gear control handle is placed in the *gear up* position, the No. 1 solenoids in the MLG and NLG selector valves are energized. These solenoids reverse the position of the pilot spool in each gear selector valve with the resultant repositioning of the slave pistons. The pilot spools and slave pistons are now to the extreme left (opposite the position in the schematic in figure 4-1). Pressure leaving the two gear selector valves retracts the gear actuating cylinders to raise the landing gear. (The initial movement of the gear cylinders unlocks the gear down-locks.) As the gears reach their full up position, they actuate limit switches which energize the No. 1 solenoids on the NLG and MLG door selector valves.

The energizing of these solenoids permits fluid pressure to retract the nose gear door cylinder and close its door, and to extend the main gear door cylinders and close their doors. When the doors reach their fully closed and locked position, they actuate limit switches that de-energize all gear and gear door selector valve solenoids. De-energized, the selector valves return to neutral and shut off pressure to the gear and gear door cylinders.

Nose Landing Gear Door Operation.

As mentioned before, the nose gear door hydraulic circuit is different in that it contains a restrictor check valve and an inverted relief valve which the other gear and gear door circuits do not. Figure 4-7 shows this nose gear door hydraulic circuit. Note the positions of these two additional units, one in the *down* line and one in the *up* line.

When the left solenoid is energized for door open action, fluid pressure passes through the selector valve and into the *down* line. In the down line this pressure passes through the conventional restrictor, which limits the speed of door operation, and up to the inverted relief valve. This inverted relief valve opens when *down* line pressure reaches 1500 psi. The 1500 psi of pressure is of sufficient force to move the "locking" balls in the door actuating cylinder to the unlocked position. With the piston unlocked, fluid pressure extends the piston and rod to open the gear door. Displaced return fluid in the cylinder passes through the restrictor check valve in the restricted direction, as noted on the schematic, then through the selector valve and into the secondary system return.

Going back to the cylinder, note the spring-loaded pin at the "locking" balls and the shuttle valve. As *door open* pressure enters the cylinder it unseats the poppet above the pin and releases pressure on the switch attached to the left side of the cylinder. Actuation of this switch completes an electrical circuit to the gear selector valve.

The shuttle valve is normally in the position shown in the schematic. When the landing gear is lowered during emergency operation (hydraulic system inoperative), high pressure air displaces the shuttle valve so that it blocks the hydraulic *down* line and allows air pressure to open the nose gear door in the same manner as hydraulic pressure. When hydraulic pressure is again used to open the door, the shuttle valve is displaced back to its normal position.

In closing the nose gear door, the right solenoid on the selector valve is energized and fluid pressure enters the *up* line. Here is where the restrictor check valve comes into action. Pressure entering this valve in the *free flow* direction unseats the tapered jet and allows full pressure to enter the cylinders. The free flow of full pressure is sufficient to force the piston into its locked position (locking balls moved to "locked" position). The retracting of the cylinder piston also pushes the spring-loaded pin down to actuate the attached switch and open the circuits to the gear and gear door selector valve solenoids. With the solenoids de-energized, fluid pressure is removed from both the gear and gear door cylinders. The gear and gear door remain in the up and locked position until the gear door is opened.

Emergency Operation.

As you learned, emergency operation of the gears is controlled by pneumatic pressure. The pneumatic pressure line you see at each actuating cylinder in figure 4-1, displaces shuttle valves at each of the cylinders and moves the cylinders to open the gear doors and lower the gears when the *emergency gear down* handle is pulled. Displaced fluid in each actuating cylinder passes through its respective selector valve, which is in neutral, and into the secondary system return line.

The nose landing gear door and gear are automatically sequenced by a pneumatic priority valve that directs initial pneumatic pressure to the door actuating cylinder. Then when the door is open and pneumatic pressure has built up to about 850 psi, the priority valve unseats and pressure enters the nose gear strut actuating cylinder to extend the nose gear.

The main landing gear doors and gears are also automatically sequenced for emergency pneumatic operation. A restrictor check valve in the pneumatic *gear down* line delays the gear actuating cylinder movement until the main gear doors have unlocked and opened. When the doors are opened air pressure then overcomes the restrictor check valve and lowers the main gear.

Bleeding After Emergency Extension.

You cannot retract the gear after an emergency extension until you have placed the emergency gear control handle in the full forward position. This vents pneumatic pressure from the actuating cylinders and emergency gear lines. You must then place the airplane on jacks and operate the gear hydraulically through several cycles, using a hydraulic test stand. This cycling repositions the shuttle valve, and bleeds all air from the actuating cylinders and the landing gear system hydraulic lines.

Hydraulic System Components.

In the preceding paragraphs of this chapter we discussed *what* the various components in the gear hydraulic system do, but not the details of *how* they function. The reason is that we all know the general function of the components and *what* they do in a system. For example, we know that in an automobile cylinder or a steam pump cylinder, the cylinder provides a housing and guide for the piston; the piston is attached to a shaft that transfers work thrust to the object to be moved; and that a force—steam expansion, burning gas-air mixture, fluid pressure, and the like—is introduced to the cylinder to move the piston.

However, the actuating cylinders and other components in this hydraulic subsystem have unique features which you need to know to understand better *how* the system operates. By knowing their details you will be better able to trouble shoot and maintain the system. All of the components in the landing gear hydraulic subsystems are discussed in the following paragraphs.

Line Restrictors.

Line restrictors are installed in each *up* and *down* line to control fluid flow to all landing gear door and gear actuating cylinders. By controlling fluid flow, these restrictors regulate the speed of gear doors in opening and closing and gears in extending and retracting. The restrictors in the main landing gear door and gear lines restrict fluid flow in both directions, and are in both lines to each actuating cylinder. Both nose landing gear line restrictors and the nose landing gear door *open* line restrictor function in the same manner as those in the main landing gear lines. One of these restrictors is shown in figure 4-2 in the *down* line below the selector valve. Note that it has a narrower passageway than the diameter of the lines on each end of it; it causes a restriction of fluid flow in both directions.

Restrictor Check Valve.

Only one restrictor check valve is used in the landing gear hydraulic system. This valve is shown in the nose gear door *up* line in the schematic (figure 4-2). It is different from the other restrictor valves in that it provides *free flow* in the line when the door is actuated to the closed position. Note the free flow direction in the schematic. The reason for this free flow is to provide full pressure in the actuating cylinder when the door is *closing,* thus assuring that the locking mechanism will engage properly. This precludes the possibility of the lock in the cylinder not engaging fully.

This cylinder locking action will be covered more fully when we discuss the nose gear door cylinder. The restrictor check valve restricts in the same manner as the other restrictors when the door is being actuated to the *open* position. This is due to the action of the spring-loaded jet assembly in the valve. Referring to figure 4-2 again, you can see that when fluid is passing through the restrictor check valve in the free flow direction, it will force the jet assembly from its seat—at the right hand side of the valve—and allow a free passage of fluid. When fluid is traveling in the other (restricted) direction, it forces the jet assembly against its seat, and leaves only a restricted passage for fluid flow.

Inverted Relief Valve.

The inverted relief valve, shown on figure 4-2, serves two purposes. One purpose is to withhold fluid pressure from the nose gear door cylinder until a minimum of 1500 psi of pressure has built up in the *down* (door open) line. The other purpose is to prevent fluid backpressure from unlocking the actuating cylinder lock mechanism. The inverted relief valve is the same as an ordinary relief valve in that it stops fluid until a certain pressure has built up, then it opens and allows fluid to pass. However, when this valve does allow the fluid to flow, it routes it to the pressure side of the nose gear door cylinder instead of to a return line as in a conventional installation.

If you will recall, there is no hydraulic pressure in the cylinder when the door is in the closed position; so

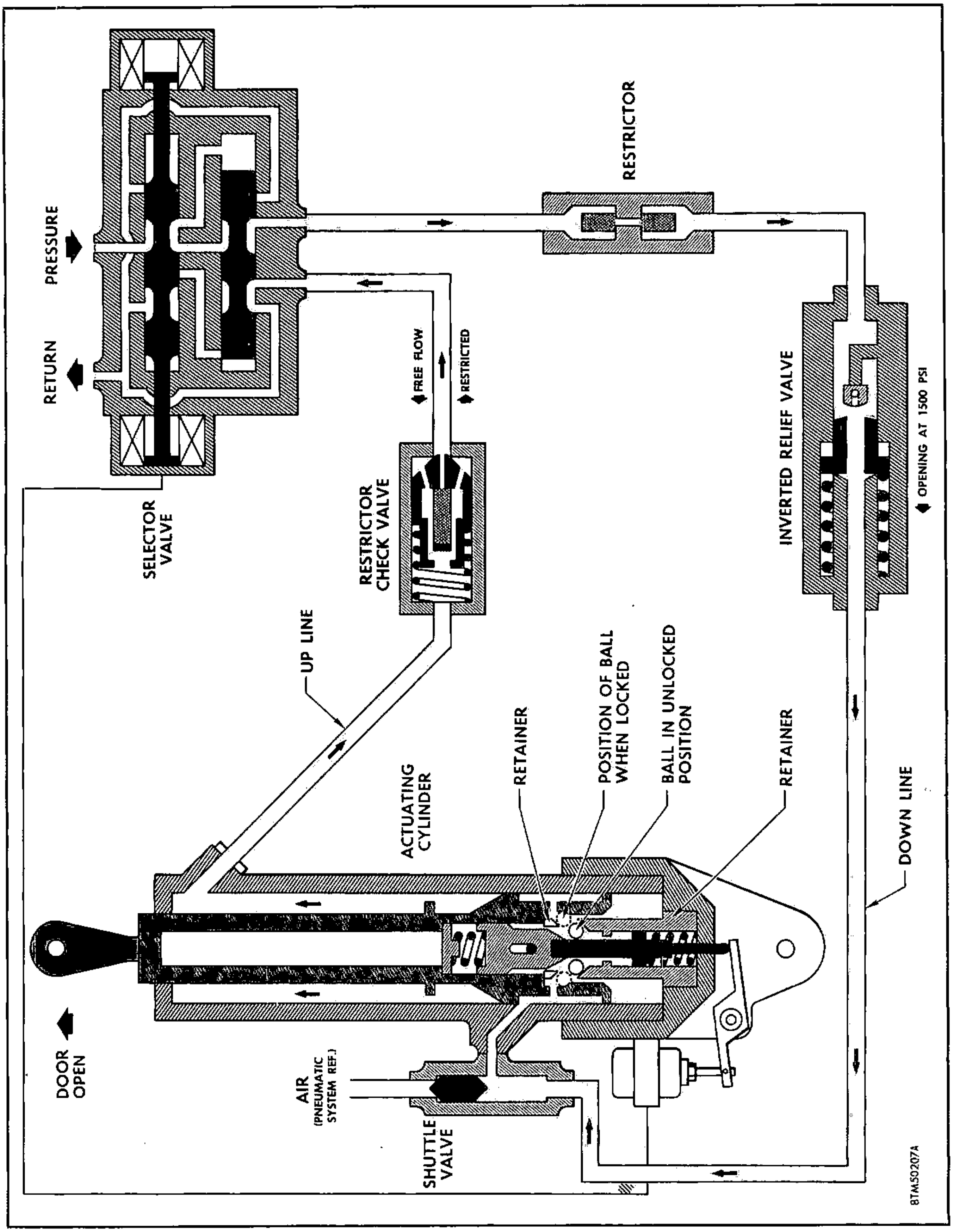


Figure 4-2. Nose Landing Gear Door Hydraulic System

when the selector valve is energized, pressure is routed to the cylinder to open the door. However, the inverted relief valve in the line is spring-loaded to stop the fluid until the pressure has built up to 1500 psi at the valve. At 1500 psi the pressure forces the relief valve to open, allowing full system pressure to enter the cylinder that up to now has been unpressurized. This provides sufficient shock to force the poppet valve (a lift valve) in the lock mechanism to lift and allow the cylinder to open the nose gear door.

Air-Hydraulic Shuttle Valves.

Each actuating cylinder in the landing gear system has a shuttle valve attached to its door *open* or gear *down* side. This valve, with its internal plug, is shown in figure 4-2. The normal position of the valve, as shown, provides for free passage of hydraulic fluid to the cylinder. At the same time, it blocks the passage of hydraulic fluid to the high pressure pneumatic line attached to it, which is normally unpressurized. If hydraulic pressure is lost and the pilot is ready to land, he lowers the landing gear by actuating the emergency gear extend handle in the cockpit. This releases high pressure pneumatic system air pressure to the shuttle valves. The air pressure forces the line plugs in the shuttle valves over to the other end and then enters the actuating cylinders to open the gear doors and lower the gears. These plugs, when displaced to the other end of the shuttle valves, serve in their new position to prevent air contamination of the rest of the hydraulic system.

Referring again to figure 4-2, imagine the line plug at the lower end of the shuttle valve; in that position, you can see that it will stop air from entering the cylinder hydraulic *down* line, and at the same time allow air to enter the cylinder. As the selector valves of the landing gear system are normally in their neutral position in flight, there is no fluid resistance on the return side of the actuating cylinder pistons when the air pressure actuates them. To insure that the selector valves will be in neutral to allow passage of return hydraulic fluid, provision is made for the emergency gear extend handle to open an emergency *down* switch. Opening of the switch removes electrical power from the entire landing gear circuit beyond the switch so that all selector valves are de-energized.

Selector Valves.

Four solenoid-operated selector valves are employed in the landing gear hydraulic subsystem to control the flow of hydraulic pressure to either raise or lower the gear and to either close or open the gear doors. Of these four selector valves, one controls main gear operation, one controls main gear door operation, one controls nose gear operation, and the remaining valve controls the nose gear door operation.

The main gear and gear door selector valves are adjacent to each other on the aft bulkhead in the hydraulic accessory compartment. The door selector valve is below and outboard of the gear valve. The nose gear and gear door selector valves are also adjacent to each other on the nose wheel well bulkhead at the forward end of the nose gear drag strut. The door valve is outboard and to the left of the gear valve.

Figure 4-3 shows a cutaway and operational schematic of one of these "four-way" selector valves. In the cutaway view you can see the solenoid at each end of the valve, the pressure and return ports, the two ports to the actuating cylinders, the pilot spool, and the slave piston. Note that the three schematic views are actually flat patterns of the cutaway. In details A, B, and C we will energize one solenoid and see how fluid pressure passes through the valve to pressurize one side of an actuating cylinder, and how displaced fluid from the cylinder is routed through the valve to system return. Energizing the other solenoid reverses the action of the valve and the flow of hydraulic fluid.

Detail A shows the neutral position of the valve pilot spool and slave piston when neither solenoid is energized. Note that system pressure is blocked at the pilot spool and cannot enter the valve. Note also that static fluid from both sides of the cylinders is connected to system return. This arrangement provides a *return* for the displaced fluid in the actuating cylinders when the valve is de-energized and the landing gear is lowered with emergency pneumatic pressure. When the right solenoid is energized, it pulls the pilot spool to the right as shown in detail B in figure 4-3.

Now you can see how movement of the pilot spool removes the "block" from the system pressure line at the top of the valve. The pressurized fluid is then free to move around and past the narrow throat of the spool and to enter the left end of the slave piston chamber. This pressure overpowers the right spring and pushes the piston to the right, as shown in detail C. Now, system pressure that was blocked at the center of the slave piston in detail B can pass around the narrow part of the slave piston, as it did around the pilot spool, and enter the line to the actuating cylinder. Note that displacement of the slave piston also allows return fluid from the actuating cylinder to pass through the valve to the system return line at the top of the valve.

The pilot spool and slave piston will stay in the positions shown in detail C as long as the right solenoid remains energized. When it is de-energized, the pilot spool and slave piston returns to the neutral position shown in detail A. If the left solenoid is energized instead of the right one which we have just discussed, the pilot spool and slave piston move in the opposite direction (to the left). This valve action routes system pressure to the opposite side of the actuating cylinder and routes cylinder return fluid through the left side of the valve to system *return*.

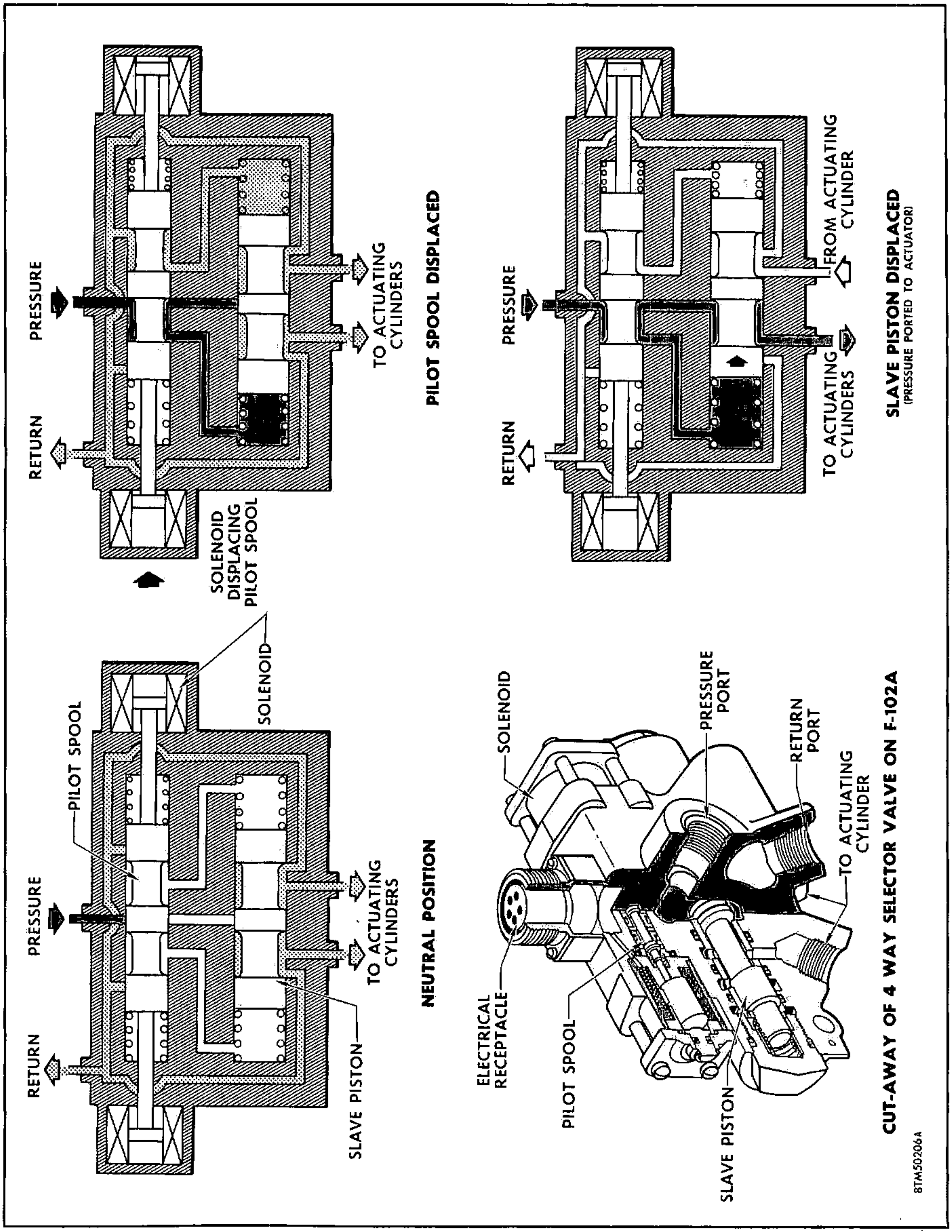


Figure 4-3. Landing Gear Selector Valve

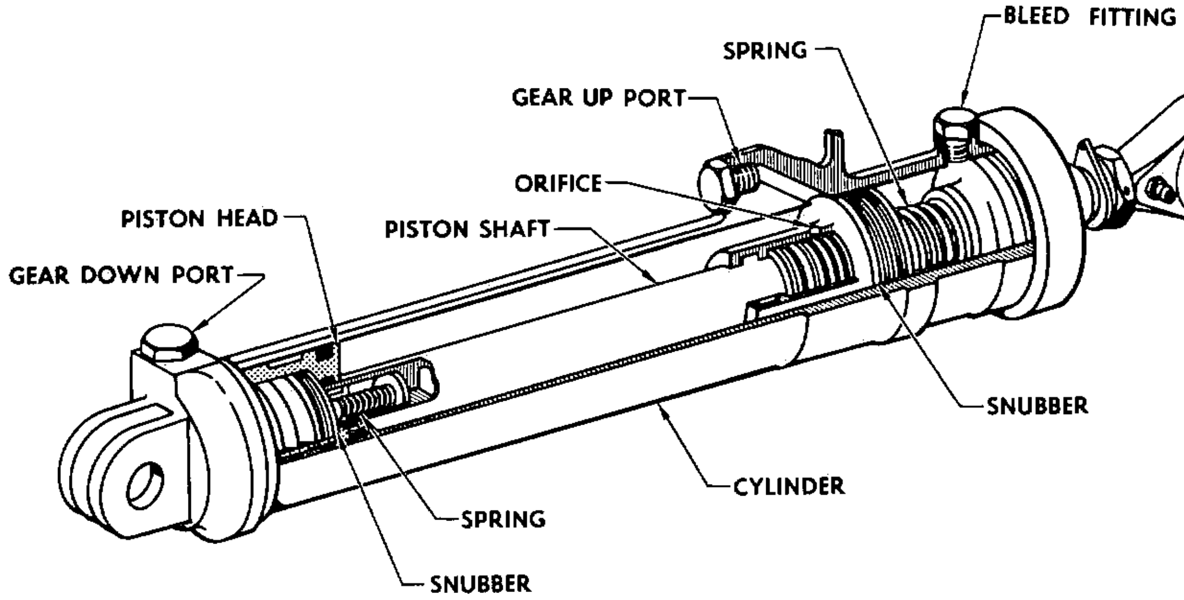


Figure 4-4. Main Landing Gear Actuating Cylinder

Main Landing Gear Actuating Cylinders.

The main landing gear actuating cylinders retract to raise the landing gear and extend to lower the gear. As you will note on figure 4-4, the cylinder has an internal snubber (shock absorber) at each end. The purpose of these snubbers is to relieve the shock of piston travel during extension or retraction of the piston shaft. Note on the illustration that the snubber on the right end shows an orifice (hole) drilled through its head to permit fluid to flow into the area between the snubber and the cylinder end cap. Also note the springs installed between the snubbers and the cylinder heads (caps). The snubber on the other end is similar in construction, but its orifice is not shown. The combined restraint of the fluid on the return side of the piston and the springs between the snubbers and the cylinder caps act against the hydraulic pressure on the piston. This causes the piston travel to be "snubbed" or slowed to an easy stop at each end of the stroke.

In extending the main gear, pressure enters the shuttle valve, which is attached to the *gear down* port, and pushes the piston head to the right end to lower the gear. At the end of the piston shaft extend stroke the piston head contacts the left end of the snubber and compresses the springs. Fluid trapped on the right side of the snubber piston bleeds through the orifice and adds its snubbing action.

Snubbing action is the same on piston shaft retraction (gear-up), except that a spring inside the piston shaft contacts a boss to start the snubbing action. The bleed fitting on the right end provides a means of bleeding air from the area between the snubber piston and the cylinder cap. Since *gear-down* pressure enters the head end cap, no bleed fitting is necessary. Any air in this side is bled when the system is cycled through several operations.

Main Landing Gear Door Actuating Cylinders.

The main landing gear door cylinders are simple actuating cylinders that operate in a conventional manner. They retract to open the main gear doors and extend to close the doors. These cylinders consist of *door open* and *door close* ports, a piston and attached piston rod, and a housing or cylinder case. A shuttle valve is attached to the *door open* port so that either hydraulic pressure can open the doors in normal operation or air pressure can open the doors in emergency operation. There are two cylinders provided to actuate each main landing gear door. They operate in the same manner as the simple cylinder described in Chapter I. Mechanical overcenter locks in the door actuating mechanism, which we discussed earlier, lock each main landing

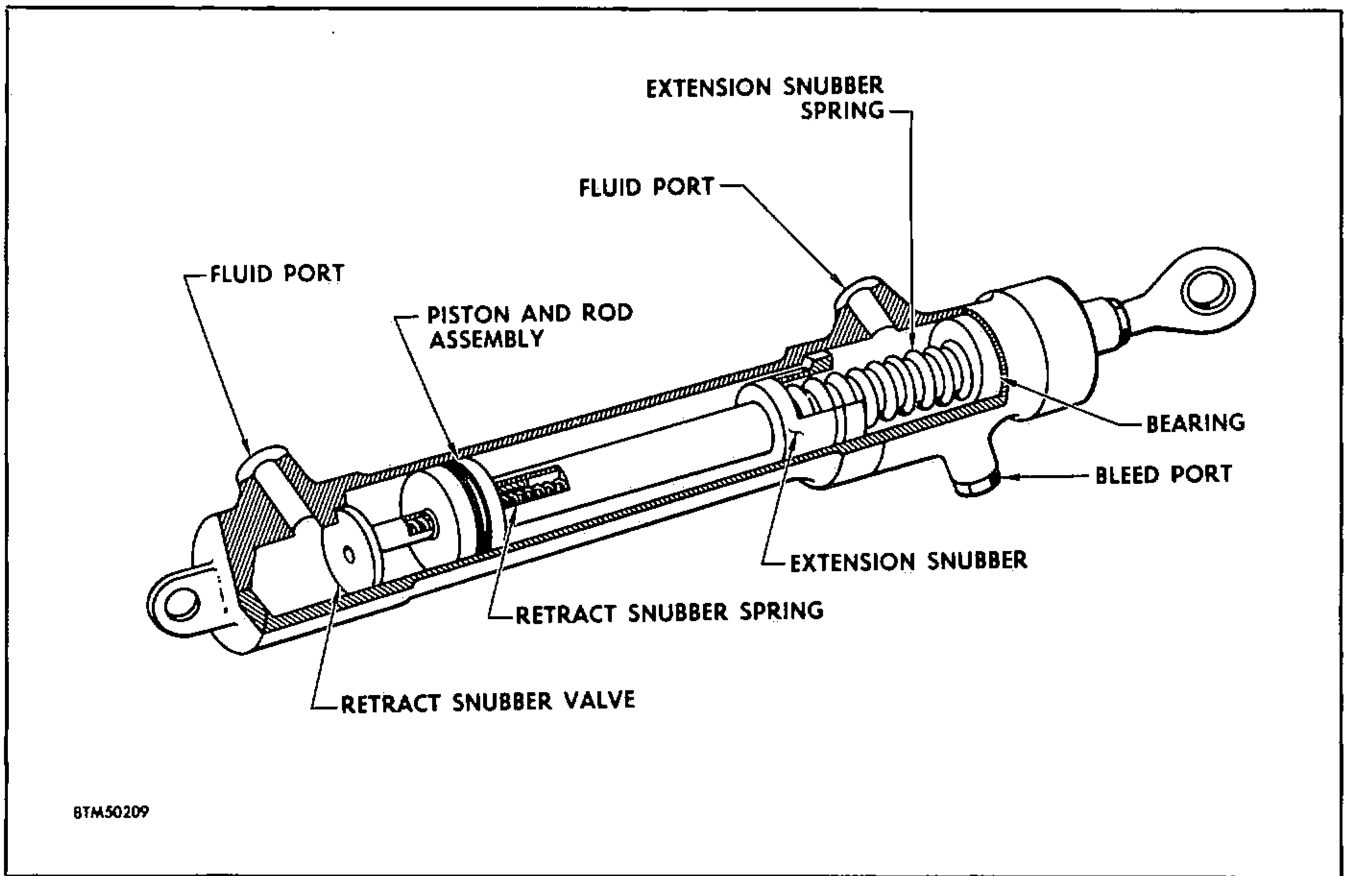


Figure 4-5. Nose Landing Gear Actuating Cylinder

gear door in its closed position. The overcenter locking mechanism of the doors also serves to hold the main landing gears in the retracted position.

Nose Landing Gear Actuating Cylinder.

The nose landing gear actuating cylinder, shown in figure 4-5, extends its piston rod to raise the nose gear, and retracts to lower the gear. This cylinder is similar in construction to the main gear cylinder and offers snubbing action both in raising and lowering the gear.

In figure 4-5, note the extension snubber and its spring, and the retraction snubber and its spring. When the piston and rod assembly extends to raise the gear, pressure entering the left fluid port unseats the retract snubber valve and pushes the piston in its extend direction. Trapped fluid between the piston and extension snubber pushes this snubber to the right past the right (gear down) fluid port so that the displaced fluid behind the piston can leave the cylinder through the right port. The piston contacts the extension snubber in its final extension movement and compresses the spring, thus producing a snubbing action for gear up operation.

When the nose gear is lowered, pressure enters the right fluid port and retracts the cylinder piston and rod assembly. Pressure entering this port holds the extension snubber to the right until pressure is removed from this port (engine stopped or gear control handle is put in the *up* position). As the piston approaches its fully retracted position, the retract snubber contacts the boss inside the cylinder (as shown) and compresses the spring inside the piston rod to produce snubbing action. Since the piston itself has not reached its fully retracted position, it forces fluid through the orifice until it is fully retracted. Fluid passing through the orifice provides additional snubbing action.

The bleed port on the right end is provided to bleed the area between the extension snubber and the bearing. The air-hydraulic shuttle valve connects to the right (gear down) fluid port, so that the gear can be normally lowered with hydraulic pressure and lowered in an emergency with pneumatic pressure. Pneumatic action is the same as hydraulic in retracting the piston and rod assembly.

Nose Landing Gear Door Actuating Cylinder.

The nose landing gear door actuating cylinder is also a double acting unit which *retracts to close* the nose wheel door and *extends to open* the door. As mentioned in the earlier discussion on nose gear door operation, an

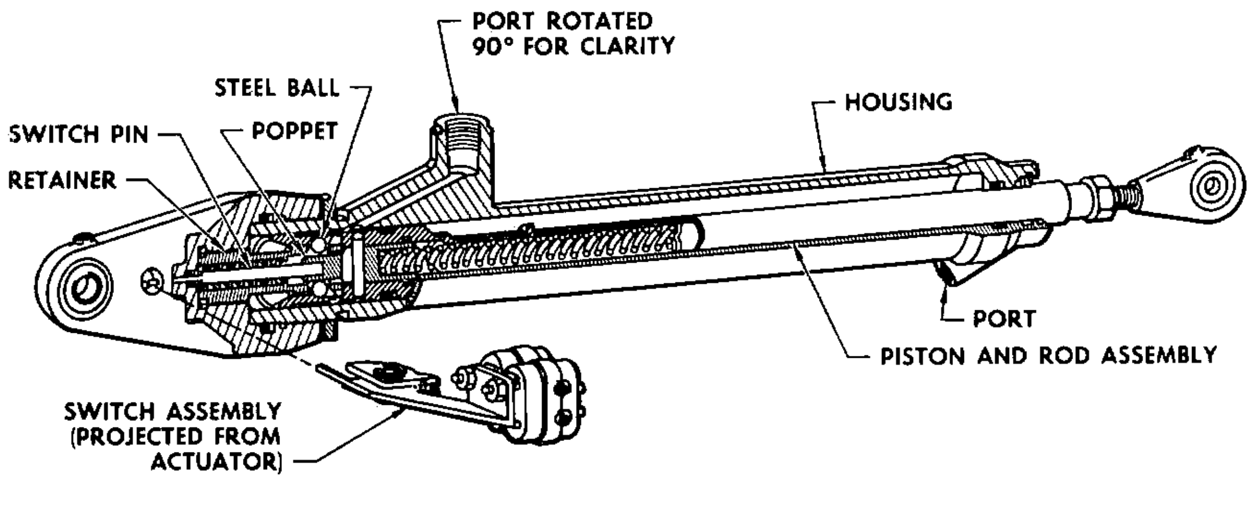


Figure 4-6. Nose Landing Gear Door Actuating Cylinder

integral ball-locking device is incorporated in the cylinder to lock it in the retracted (door closed) position. As you will note on figure 4-6, the lock consists essentially of a ball retainer sleeve, mounted in the cylinder head end cap, and six steel balls which are retained in holes drilled at an angle around the circumference of the right end of the retainer. A spring-loaded switch pin within the retainer extends the full length of the retainer and through the cylinder end cap to actuate the limit switch assembly.

When the poppet (lift) valve on the end of the piston is actuated by door open fluid pressure, the balls are free to travel inwards towards the switch pin. This action releases the lock, and therefore the piston. As the locking device releases, the switch pin serves to retain the steel balls in position for relocking.

The one-piece piston and rod assembly is machined internally to slide over the ball retainer sleeve, while the spring-loaded poppet slides inside the piston head. The head of the poppet is machined to permit it to slide inside the ball retainer sleeve and over the end of the spring-loaded switch pin which actuates the attached switch assembly for electrical sequencing action of the nose gear.

When *door open* pressure enters the left (head end) port, it enters with sufficient force to unseat the spring-loaded poppet and move it to the right. This allows the steel locking balls to drop into the retainer toward the switch pin and out of the annular groove on the inside of the piston head. With the poppet pushed against the spring inside the piston and rod assembly and the locking steel balls out of the piston head groove, the cylinder is now unlocked and pressure extends the piston rod to open the nose gear door. At the same time that the poppet unseats, the spring-loaded switch pin retracts so that the switch assembly can complete a circuit to the NLG selector valve.

In retracting to close the nose gear door, the piston head first slides over the retainer; then the poppet pushes the steel balls up and into the piston head annular groove. The steel balls are now partly in the retainer and partly in the piston head so that the cylinder piston and consequently the nose gear door are locked. As before, simultaneous action happens at the switch pin. The poppet pushes this pin to *open* the circuit to the NLG selector valve. Although not included on this cylinder cutaway, the air-hydraulic shuttle valve attachment is shown schematically on figure 4-2.

MAINTENANCE.

Maintenance of the landing gear hydraulic system is much the same as any other hydraulic subsystem. It involves periodic inspection to determine that all units are secure, clean, free of leaks, and are lubricated where required. Maintenance of the system also requires that you trouble shoot for any malfunction the pilot may report or you detect in your inspections, and for the correction of these malfunctions. Preventative maintenance—the detecting and correcting of a malfunction before it occurs—is the best maintenance.

The F-102A maintenance manual, T.O. 1F-102A-2-8, provides you with trouble shooting charts which include troubles, possible causes, and the correction. This manual also gives you step-by-step procedures for replacing, adjusting, and testing the various components in the system. However, your proficiency in maintaining the landing gear hydraulic system is by "doing."

The following paragraphs give you some of the general troubles that could be encountered; then with your knowledge of the system and its components, the maintenance manual instructions, and your experience by "doing," you soon recognize a malfunction and know just what to do to remedy the situation. Whenever you have a reported malfunction, it is best to jack the airplane and operate the landing gear. In this way you can see just what is happening.

Hydraulic system malfunctions are usually confined to lack of system pressure; leaks (both external and internal); air in the system; an inoperative unit; or some connecting mechanical or electrical system malfunction. In trouble shooting this system always check the easiest and most obvious cause first, then by the process of elimination you can put your finger right at the cause. Whenever you have a malfunction, first analyze what could cause the trouble, then proceed.

External leaks are the most obvious of any malfunction. Usually they can be corrected by tightening a fitting. However, if external leaks exist at a unit where tightening is not the corrective measure, then replace the unit. Do not try to overhaul it. In correcting external leaks at fittings always use two wrenches, one on the unit fitting and one on the line fitting. If two wrenches are not used, you will put a *twist* into the line which can cause an early failure or additional leaks.

Internal leaks in the operating units are not always so easy to detect. They usually are indicated either by a slow or spongy actuation of the system or by "creeping." However, since air in the hydraulic system can give similar indications, first check for air before replacing any units. Air in the system is removed by cycling the landing gear several times until all fluid has completely circulated through the pumps and reservoir, and then bleeding the air from the reservoir.

If you encounter a malfunction where one or all of the landing gears will not operate, first analyze the cause. If all gears are inoperative, then check for a central cause. Check that you have full system pressure, that control handles are positioned correctly, that electrical power is reaching the selector valve solenoids, and that the landing gear ground lock safety pins are removed. Usually one of the above is the cause for complete system malfunctioning. Should only one gear or gear door malfunction, then you know approximately where the trouble lies. First check that there are no mechanical malfunctions, such as bent, damaged, or binding components. Then check that electrical power is reaching the selector valve.

Assuming that the mechanical and electrical systems are satisfactory, you then know the trouble is in the hydraulic system. A completely inoperative gear or gear door is usually caused by a defective selector valve. A slow or creeping component is usually due to leaks—internal or external—or to restricted passages. Replace the defective component. Sometimes you may replace several components in a system before you replace the defective unit. However, as you get to know the system better and have corrected any malfunction, you will soon pin point the difficulty by the manner in which the system malfunctions.

Remember, any time that you replace a landing gear hydraulic system component, you must place the airplane on jacks, bleed the hydraulic system of air, and then check the gear for proper operation.

THE LANDING GEAR CONTROL CIRCUITS.

The operation of the landing gear and gear doors is controlled electrically by the gear control handle in the cockpit which the pilot positions to either the GEAR UP or GEAR DOWN position. (The rest of the system control is exercised by the landing gear and landing gear door position switches which sequence the retraction and extension operations after the pilot has positioned the control handle.)

In this portion of Chapter IV, you will learn how the electrical circuits control the normal operation of the gear cycles and how the emergency control circuit allows the gear to be retracted while the airplane is on the ground. First, however, let us take a look at the entire control circuit which sequences the retracting and extension cycles.

In the schematic of the control circuit (figure 4-7), locate the power source. Note that all power for the landing gear electrical circuit is taken from the 28-volt d-c essential bus. As you have already learned, the F-102A has both an essential and a non-essential d-c bus. The essential bus supplies current only to those circuits that are considered essential to the safety and continued flight of the airplane, while the non-essential bus supplies current to the other d-c circuits. Operation of the landing gear is necessary, of course, for the safety of the airplane.

The landing gear control circuit breaker (5-amp) is of the push-pull variety and is located on the forward auxiliary circuit breaker panel. This auxiliary panel is situated in the left side of the cockpit slightly above the cockpit floor at the forward end of the left console.

Referring to figure 4-7, locate the four hydraulic selector valves that control the fluid flow to the landing gear and gear door actuating cylinders. Note that each selector valve has two leads. Each lead takes care of one of the two solenoids on the valve. After locating the valves and their leads, locate each switch on the schematic and note its name. These are the limit switches which sequence the gear and gear door operations that were mentioned earlier in the hydraulic section.

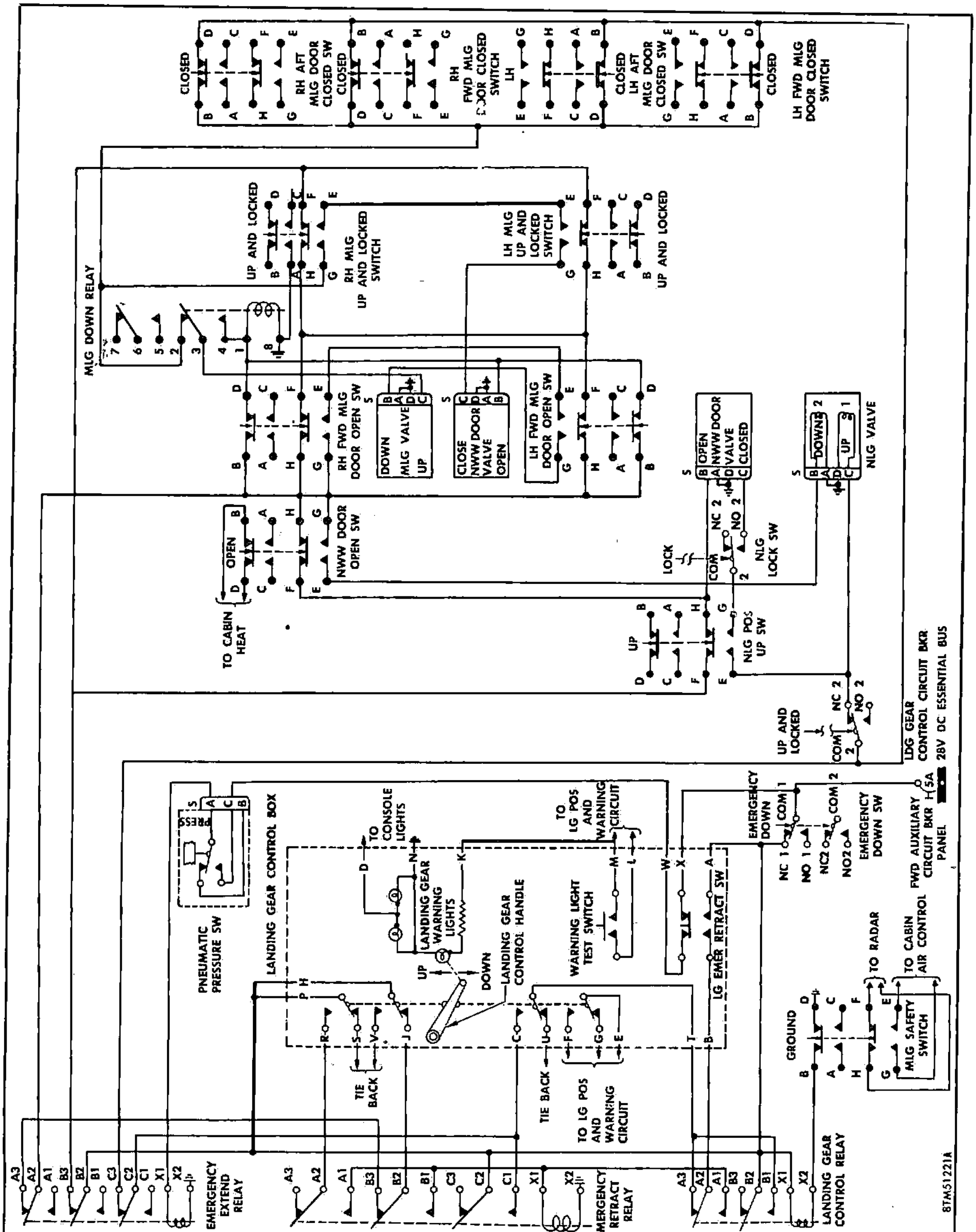
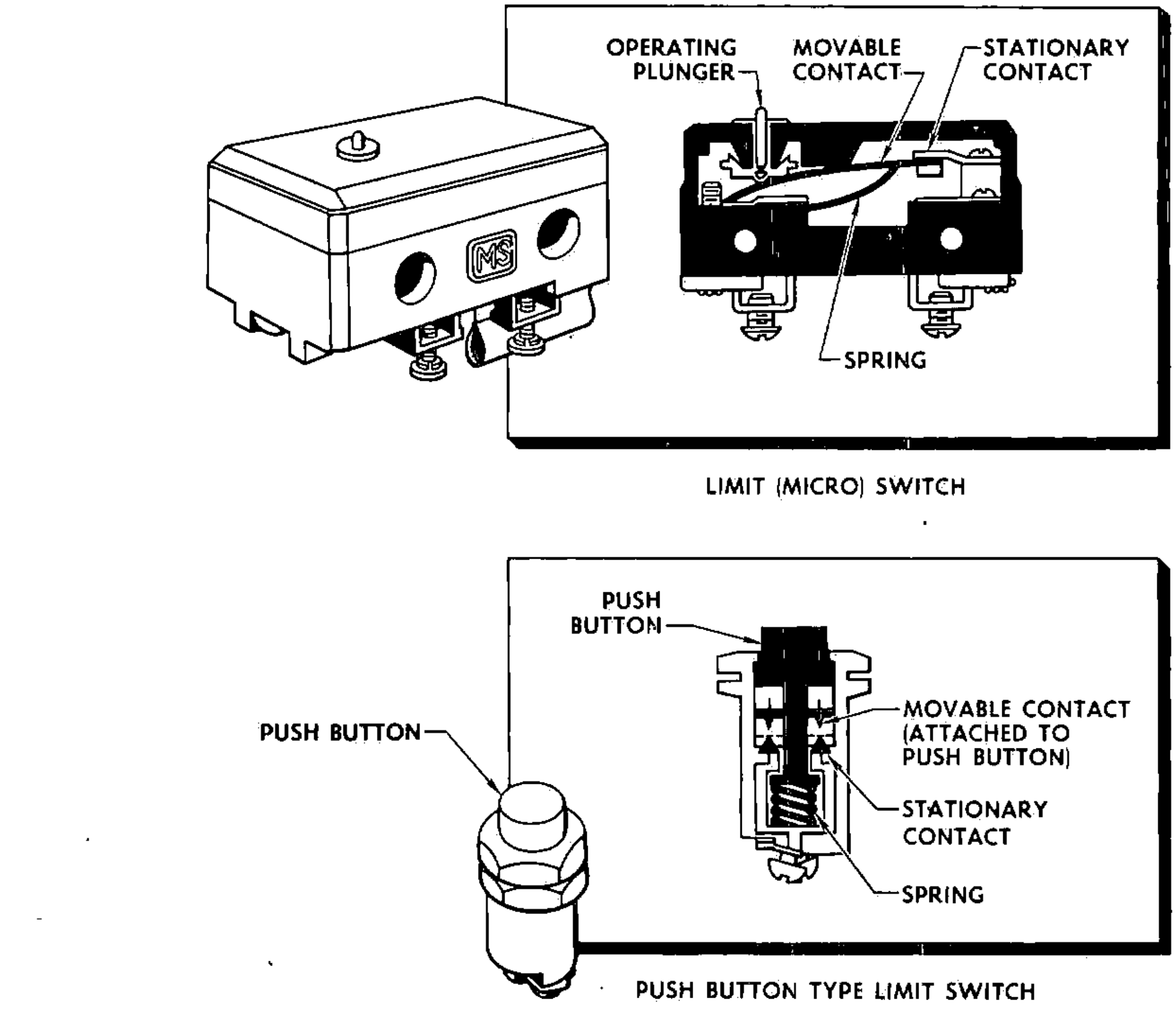


Figure 4-7. Landing Gear Electrical Control Circuit



LIMIT (MICRO) SWITCH

PUSH BUTTON TYPE LIMIT SWITCH

Figure 4-8. Landing Gear System Limit Switches

The landing gear control handle is shown at the left side of figure 4-7. Note that control handle movement positions four switches inside the handle. Also locate the four relays in the schematic—the landing gear control relay, the emergency retract relay, the emergency extend relay, and the MLG down relay. Each of these system components will be discussed in the following paragraphs as we trace the current path during normal and emergency gear and gear door operations. Before we start on the different circuits, however, let's take a detailed look at the limit switches which act as the sensing elements for the circuits.

Limit Switches.

All aircraft electrical circuits are equipped with switches to provide a quick and efficient way of starting and stopping current flow. Although there are many different types of switches in the F-102A, the two types used throughout the landing gear circuits are the pushbutton and the limit variety. Both types of switches are shown in figure 4-8. The limit switch shown in the upper view is more commonly referred to as a microswitch. When the operating plunger is pushed in, the three-bladed spring is pushed down and the contact point attached to the spring is separated from the fixed contact, this opens the switch and breaks the circuit. As the operating plunger is released, the movable contact springs back and positions against the fixed contact, this closes the switch.

The push button type of switch, shown in the lower view, is more-or-less self-explanatory. Note that the movable contact in the switch is attached directly to the pushbutton. As the button is pushed in, it compresses the spring and forces the movable contact away from the stationary contact, and this action breaks the circuit. Releasing the pushbutton allows the spring pressure to reposition the movable contact against the fixed contact and close the circuit.

MLG DOOR CLOSED SWITCHES. Each MLG wheel well has a forward and aft MLG door closed switch located respectively on the forward and aft wheel well bulkheads as shown in figure 4-9. The MLG door closed switches are actuated by the door actuator cylinder bell crank arms in the up position. When actuated, the switches open the circuit to the up solenoids. This closes off hydraulic pressure to the gear when the doors are locked in the up position. The door closed switches are also in the gear position indicating and warning circuit. In the door open position, the switches provide a current path to the landing gear warning light. In the door closed position, they provide a path to ground for the UP indicator current.

MLG UP SWITCHES. The MLG up switches are located just aft of wing spar No. 3 in each wing wheel well and are actuated by the landing gear as it is retracted to the full-up position. The MLG up switches sequence the operation of the MLG and the MLG door selector valves by allowing power to the close side of the MLG door selector valve only after the gear is completely retracted. The switch also provides a path for power to the landing gear warning light until the gear is completely up.

MLG DOOR OPEN SWITCHES. The MLG door open switches are located on the forward bulkhead of the main wheel wells and are actuated by the bell crank on the forward door actuating cylinders, as shown in view E of figure 4-9. In the door open position, these switches provide a path for power to the down side of the MLG selector valve. This action sequences the MLG selector valve to open only after the doors are fully open.

MLG DOWN-AND-LOCKED SWITCHES. The MLG down-and-locked switches are located on the downlatch mechanism of the side brace of each MLG. These switches are actuated by the downlatch when it is in the latched position. *The down-and-locked switches are in the position and warning circuit only.* In the gear unlocked position, these switches provide a path for power to the gear warning light. In the gear locked position, the switches provide a ground for the DN (down) indicator in the MLG position indicating systems.

NLG POSITION UP SWITCH. The NLG position up switch is located in the nose wheel well (NWW) on the lower side of the cockpit floor and is actuated by the nose gear when it is in the fully retracted position. The switch sequences the retract operation of the NLG and NLG door selector valves by providing a current path to the *close* side of the NWW door selector valve when the nose gear is fully retracted. In the gear up position, it also provides a path to ground for the NLG position indicator up circuit.

NLG POSITION DOWN SWITCH. The NLG position down switch is located in the wheel well on the canted bulkhead and is actuated by the steer-damper unit linkage when the gear is fully extended. *The down position switch is in the position and warning circuit only.* In the gear down position it provides a path to ground for the NLG down position indicator current. In the gear up position, the switch provides a current path to the landing gear warning light through the landing gear control handle switch.

NLG LOCK SWITCH. The NLG lock switch is located on the nose landing gear up and down latch mechanism and is actuated when the latch mechanism is locked in either gear up or gear down position. When the switch is in the gear locked position, it provides a path for current to the close solenoid of the NLG door selector valve so that the NLG door will close only when the nose gear is fully retracted and locked. In the gear locked position, the switch also provides a path for the 28-volt, d-c power to the terminal of the NLG position indicator. When the switch is in the gear unlocked position (gear in transit), the switch also provides a current path to the warning light.

NLG DOOR CYLINDER SWITCH. The NLG door cylinder switch is located on the upper end of the door cylinder and is actuated by a plunger from the internal lock within the cylinder. The switch is thus actuated when the door is locked in the *up* position. In the door unlocked position, the switch provides a path for current to both the NLG selector valve and the NLG door selector valve. When the door is locked, the switch contacts are opened thereby de-energizing the solenoids in the two selector valves. This closes off hydraulic power to the nose gear actuators. The door cylinder switch is also in the position and warning circuit. In the door unlocked position, the switch provides a path for current to the warning light. In the locked position, the switch provides a current path to the solenoid of the NLG door position indicator relay. When this relay is energized through the cylinder switch, the *up* terminal of the indicator is connected to ground and will thus cause an *up* indication to appear on the indicator.

NLG DOOR OPEN SWITCH. The NLG door open switch is located on the left side of the NLG beneath the door actuating arm. The switch is actuated by movement of the arm to the full open position. During the extend cycle the switch sequences the operation of the NLG selector valve to actuate after the NLG door is open.

How The Electrical Circuit Controls The Gear Normal Retraction Operation.

Now that you have a general idea of the location of the limit switches which are actuated by the landing gear and the landing gear doors, let's analyze the electrical circuit as the landing gear is being retracted.

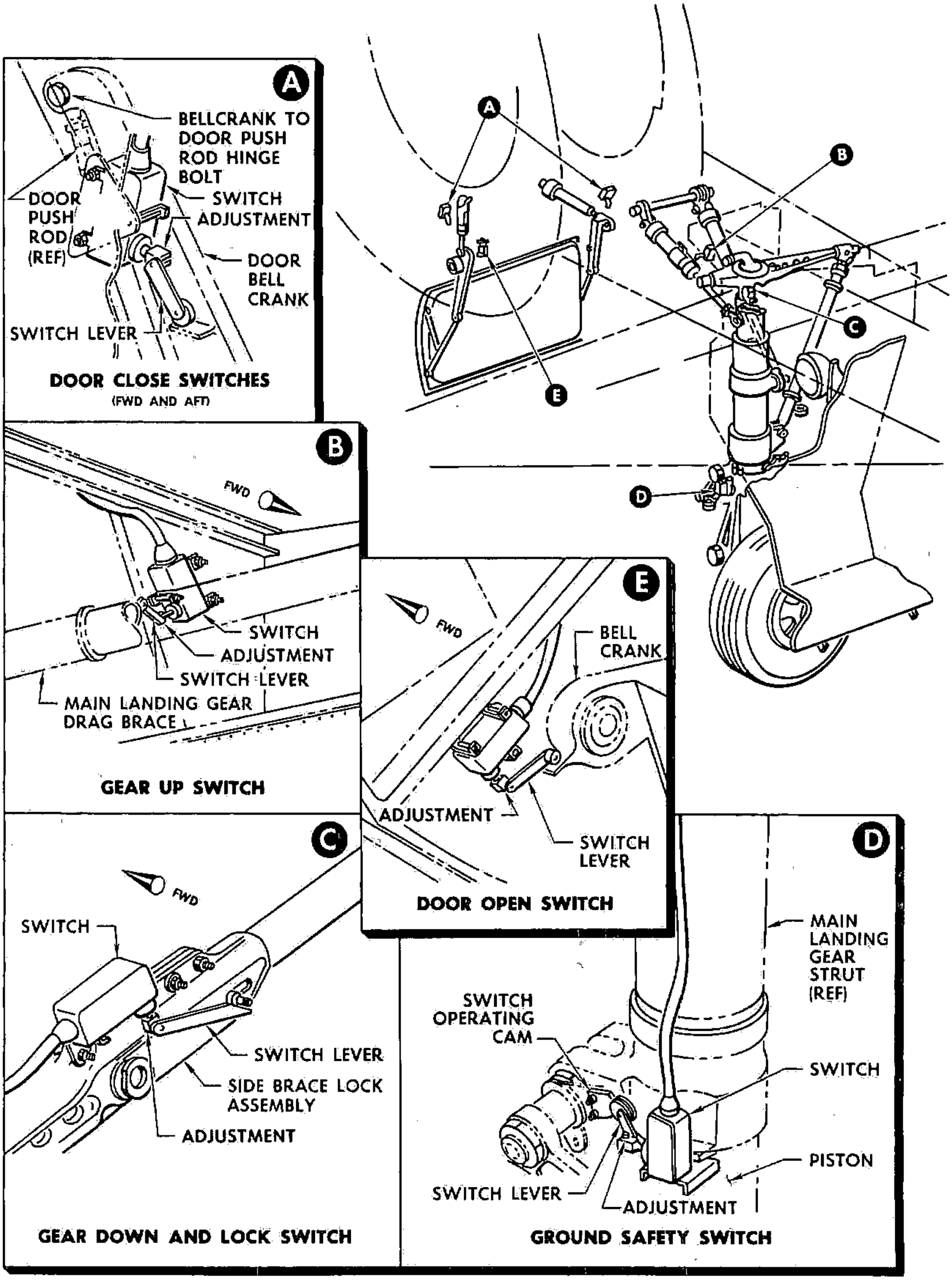


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Figure 4-9. MLG and MLG Door Limit Switches

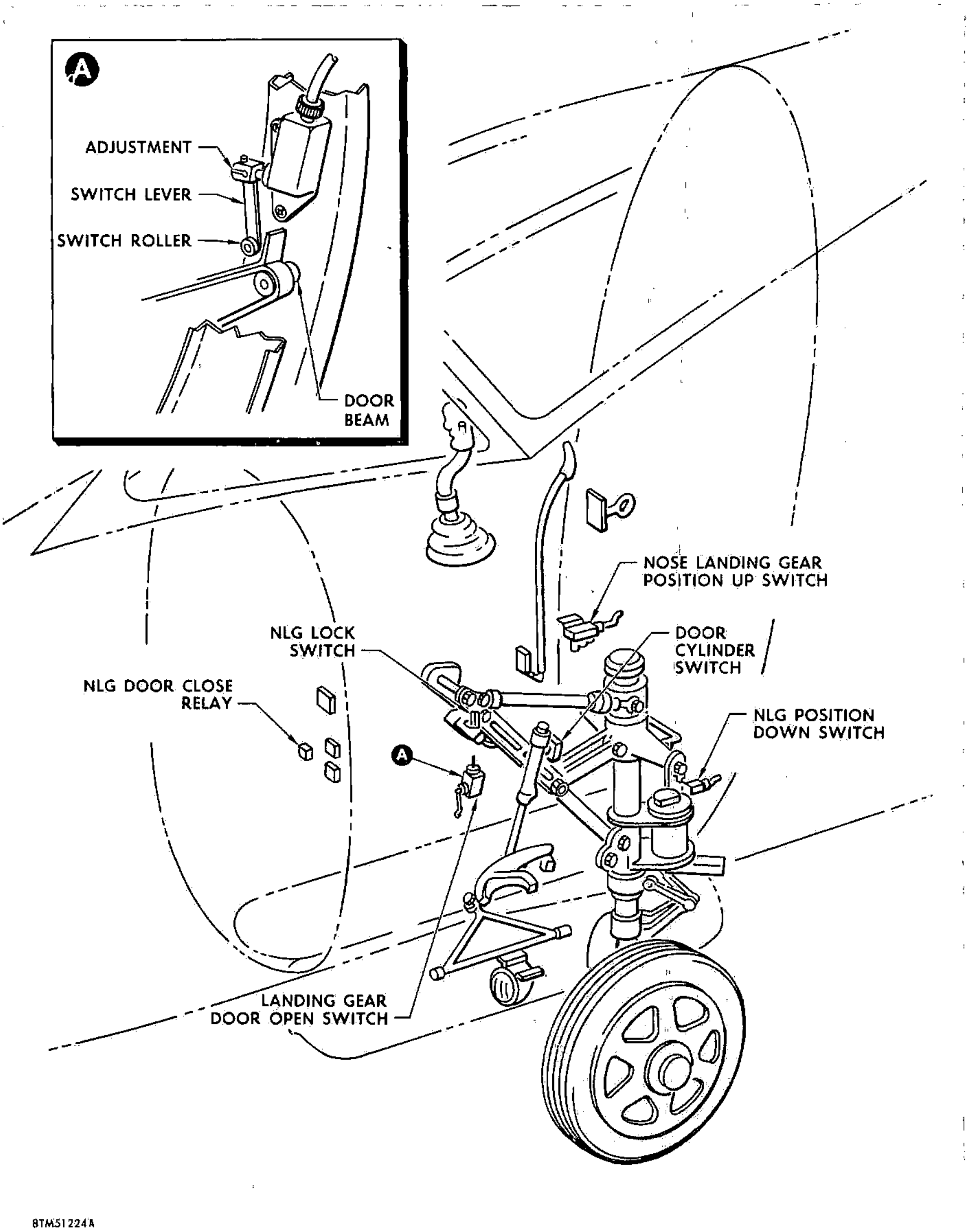

Figure 4-10. NLG and NLG Door Limit Switches

You should keep in mind that there are more switches in the landing gear control circuits than those just discussed, but the rest of the switches will be described as we trace current paths through the circuit.

The electrical schematic in figure 4-11 shows the position of the switches, relays, and controls when the landing gear handle is in the *gear up* position, the landing gear doors are open, and the gears are retracting. The energized portion of the circuit is shown in dark lines.

First, locate the 28-volt d-c essential bus and the 5-amp circuit breaker. In tracing the current path from the essential bus, note that it travels through the circuit breaker and then through the emergency control valve switch. This control valve switch must be closed for all operations of the landing system except gear emergency extension. The function of this switch will be discussed in greater detail later.

Following the energized portion of the circuit from the circuit breaker, note that the current takes two different paths after it passes through the emergency control valve switch. One path leads to terminal B2 of the emergency extend relay; the other path leads to the landing gear control relay. The ground safety switch, shown directly beneath the landing gear control relay, provides a path to ground for the relay.

The ground safety switch, sometimes called the "scissors" switch, is located just above the torque arm assembly on the left main landing gear shock strut cylinder. The switch is *closed* when the weight of the airplane is on the strut, and *open* when the airplane is either in the air or on jacks. As you will note in figure 4-11, this ground safety switch must be closed for the control relay to be grounded. This type of arrangement prevents retracting the landing gear while the airplane is on the ground except by depressing the emergency retract button. This emergency phase of retraction will be discussed later in this section.

With the landing gear control relay energized, current passes through the control relay, up to the switch in the landing gear control handle, and down to terminal C2 of the emergency extend relay. As shown in the schematic (figure 4-11), two leads from the emergency extend relay supply current to the rest of the circuit. First, trace the current path for the nose landing gear action. Following the current from B3, note that it travels to the LH MLG up switch and the NLG up switch. Note that the NLG up switch is in the de-actuated position such as would be the case when the gear is in any position other than up-and-locked. With the switch in its de-actuated position, current then passes through the switch and on to the No. 1 solenoid on the NLG door selector valve.

As you will recall from the discussion of the hydraulic system, the selector valves direct pressure to the gear and gear door cylinders when they are in their down-and-locked position.

Current from C3 of the emergency extend relay passes through the NLG door-closed switch and on to the NLG selector valve where it energizes the No. 1 solenoid. The NLG selector valve then directs hydraulic pressure to the actuating cylinder which retracts the nose landing gear.

The current for the main landing gear and door selector valves follows a somewhat different path, but the results are just about the same. Note that current from B3 of the emergency extend relay travels to both the LH and the RH MLG up switches. The output leads of these two switches join together on the opposite side of the switches. In other words, the MLG up switches are connected in parallel. The output lead travels to the MLG door selector valve.

Current also travels through the four parallel MLG door-closed switches shown at the right side of figure 4-11. The output lead from these switches travels through the MLG down relay and to the MLG selector valve. This selector valve directs hydraulic pressure to the MLG actuating cylinders to retract the ML gear.

In summarizing the schematic, you should remember that all four selector valves are energized while the gears are retracting and the doors are still open. The door selector valves assure that the doors are held in their down-and-locked position, while the gear selector valves direct fluid to retract the gear.

How The Electrical Circuit Controls The Gear Door Closing Operation.

The landing gear doors close immediately after the landing gears reach the up-and-locked position. In the schematic in figure 4-12 the control circuit is shown with the gear control handle in the GEAR UP position, the landing gears in their up-and-locked position, and the landing gear doors closing. The energized portion of the circuit is again shown in dark lines.

By comparing this schematic with the one which showed gear retraction, you will note that three switches have changed position. These switches are the NLG up position switch, and the LH MLG up position switch. Recalling the description of these three switches, you will remember that each of these switches is actuated only when its respective gear reaches the full up-and-locked position. Note in figure 4-12 that current is still being delivered to the MLG and NLG selector valves. These selector valves must direct hydraulic pressure to the retract side of the gear actuating cylinders *until* the doors reach their full up-and-locked positions.

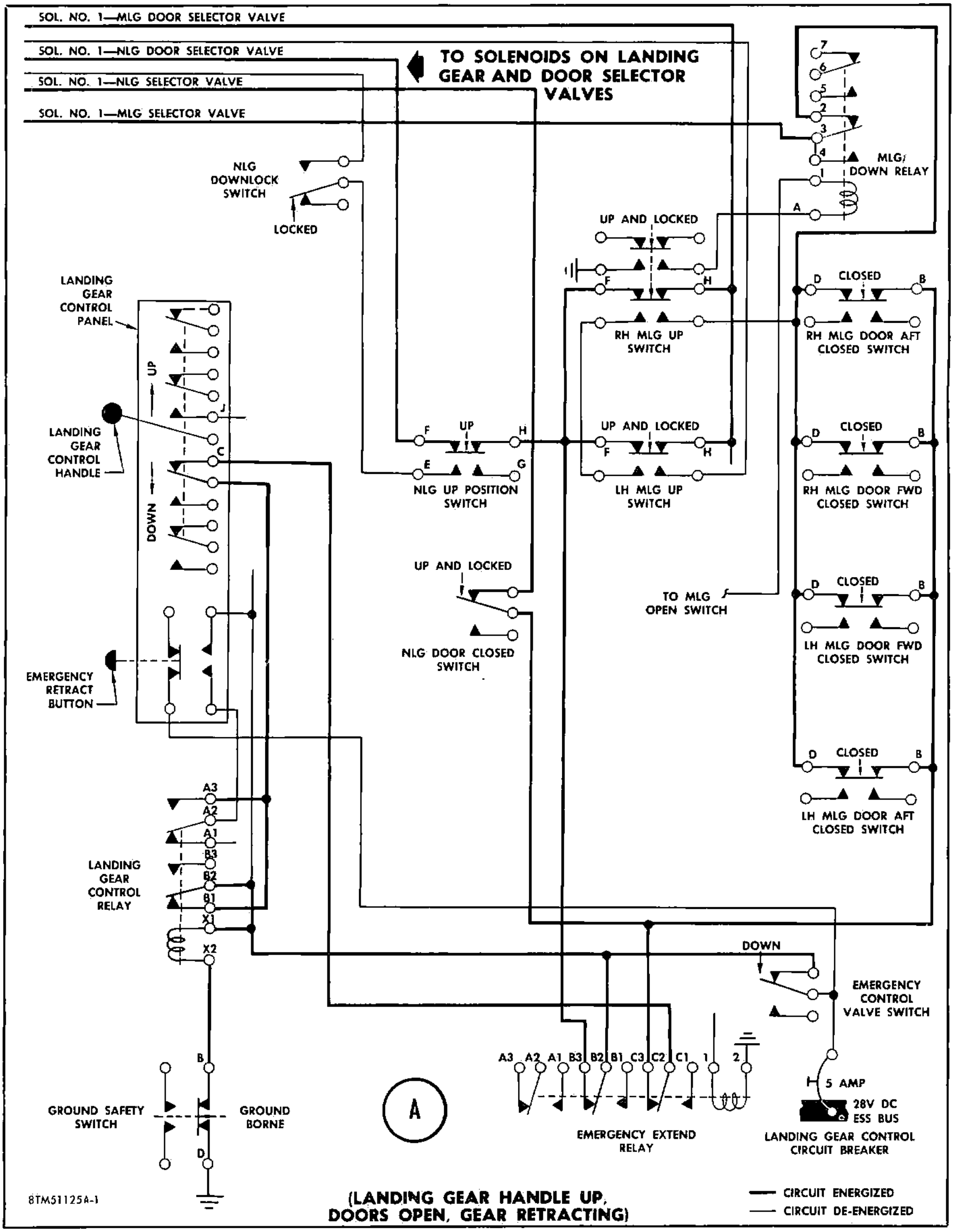

Figure 4-11. Landing Gear Circuit for Gear Normal Retraction

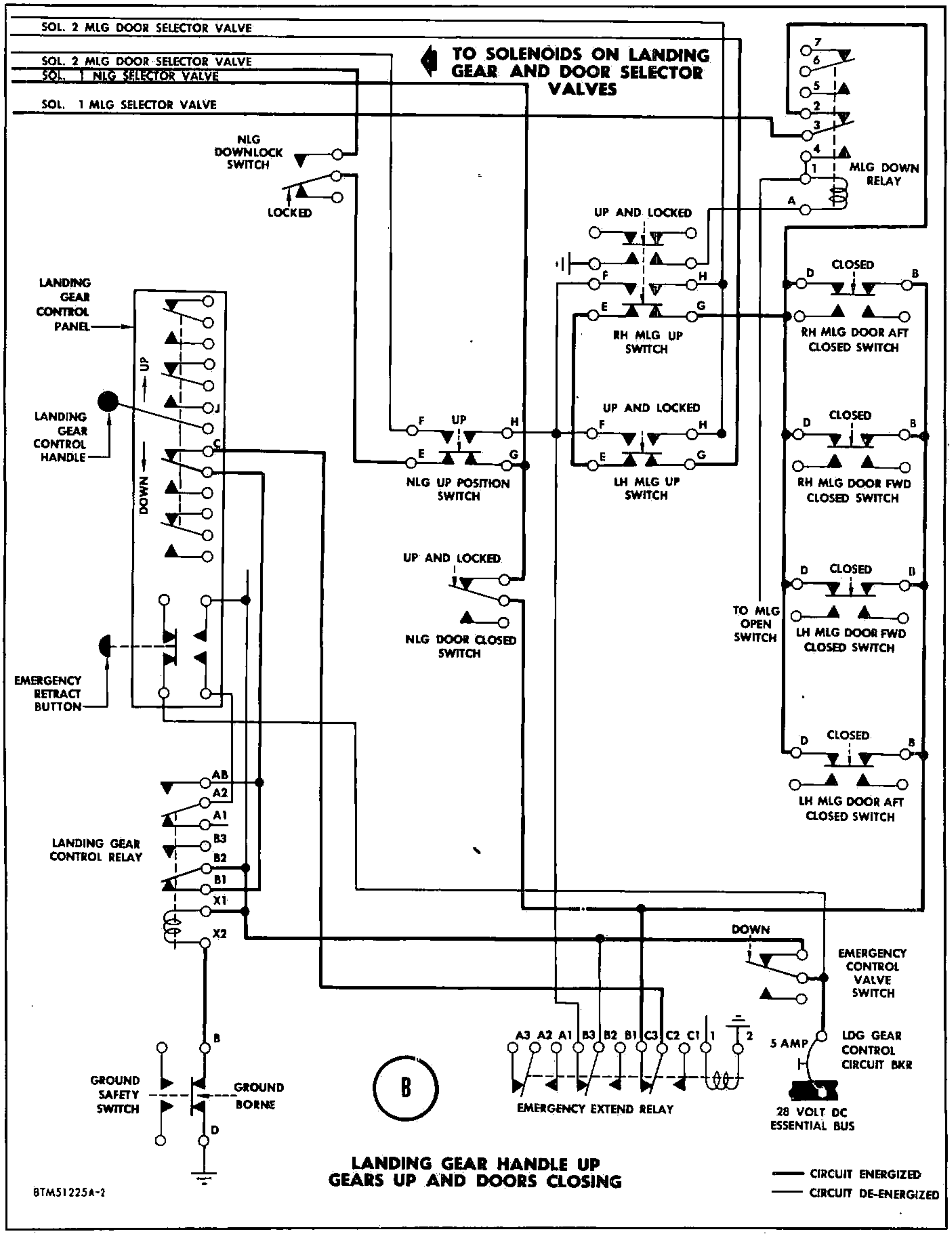

Figure 4-12. Landing Gear Circuit for Door Closing

When the NLG up position switch is actuated by the nose gear, this switch routes current to the NLG door selector valve. The valve then directs hydraulic pressure to the door actuating cylinder and the door closes.

The switch arrangement for the main landing gear is slightly different. Note that the current output from the MLG door-closed switches is still passing through the MLG down relay. This current keeps the MLG selector valve solenoid energized so that hydraulic pressure holds the main landing gears in their up-and-locked position until the MLG doors are fully closed and locked. Current for the MLG door selector valve solenoid is supplied from terminal G of the LH MLG *up* position switch. Tracing current flow backwards from this terminal, note that the LH and RH MLG *up* position switches are connected in series so that no current can travel to the MLG door selector valve until both main landing gears have reached their full up-and-locked positions. Current is supplied to the MLG *up* position switches from the parallel-connected MLG door-closed switches. By having the door-closed switches connected in parallel, current is supplied to the MLG door selector valve as long as any one of the four switches is not actuated by its door closing.

Figure 4-12 has shown how the solenoids are energized so that the landing gear selector valves direct hydraulic pressure to the landing gear actuating cylinders to keep the gears retracted while the gear doors are being closed. As the gear doors reach their full up-and-locked positions, they actuate the door-closed switches and the current to all four selector valves is terminated. The gears are then locked in their up position in the manner that you learned about earlier.

How The Electrical Circuit Controls The Door Opening Operation.

Figure 4-13 shows the electrical circuits as the normal gear extension cycle begins. As noted at the bottom of the schematic, the landing gear control handle is in the GEAR DOWN position, the landing gear is in the up-and-locked position, and the landing gear doors are opening.

In the upper portion of the schematic, you will note that only three leads to the solenoids on the gear and gear door selector valves are receiving current. As you learned earlier in this chapter, the nose landing gear is held in its up-and-locked position by means of the mechanical locking device, but the main landing gear has no locking device other than the doors. Because of this arrangement, the MLG selector valves must direct hydraulic pressure to the MLG actuating cylinders during the *door-open* operation to hold the gear in their up-and-locked position. In figure 4-13, then, one of the power leads to the selector valves supplies current to the gear-up solenoid of the MLG selector valve while the other two leads supply current to the landing gear door selector valve solenoids.

By comparing figure 4-13 with figures 4-11 and 4-12, you will note that the emergency extend relay, emergency retract relay, and the landing gear control relay perform the same functions. Current passes through the landing gear control handle, down through the emergency retract and emergency extend relays, and then through the gear extend circuits. Note that the main gear door-open switches are connected in parallel. This type of arrangement assures that current will be supplied to the MLG selector valve as long as any one of the door-open switches has not been actuated, and that the main gear will not extend until all doors have reached their full down-and-locked positions.

Current from terminal D of the MLG door-open switches passes through the MLG down relay. Since this relay is grounded through the RH MLG up-and-locked switch, the relay is energized, and current travels to the No. 1 solenoid on the MLG selector valve. This selector valve then directs hydraulic pressure to the *retract* side of the MLG actuating cylinder. Tracing the current path from terminal F of the MLG door-open switches, note that this current travels to the No. 1 solenoid on the MLG door selector valve which then directs hydraulic pressure to the *open* side of the MLG door actuating cylinder.

How The Electrical Circuit Controls The Gear Normal Extension Operation.

As the landing gear doors reach their down-and-locked positions, the electrical circuit is shifted by the limit switches and the landing gears start to extend. Figure 4-14 shows the electrical circuit for the gear normal extension operation. Note that the landing gear control handle is still in the GEAR DOWN position, the gear doors are *open*, and the gears have started to extend. As in the other schematics, the energized portion of the circuit is shown in dark lines.

In the upper portion of the schematic, note that all four selector valves are receiving current. The landing gear door selector valves are directing hydraulic pressure to hold the doors in their down-and-locked position, and the landing gear selector valves are directing hydraulic pressure to the extend side of the gear actuating cylinders.

When the landing gear doors originally reached their down-and-locked position, they actuated their door-open switches. Tracing the current path from terminal A3 of the emergency extend relay, note that it still passes to the NLG and MLG door-open switches. At this time, however, these door-open switches are in their *open* position and current now passes through the E terminals of the MLG door-open switches and the G terminal of the NLG door-open switch. This current travels to the No. 2 solenoids of the NLG and MLG selector valves. The selector valves then direct hydraulic pressure to the *extend* side of the gear actuating cylinders. Since the MLG door-open switches are

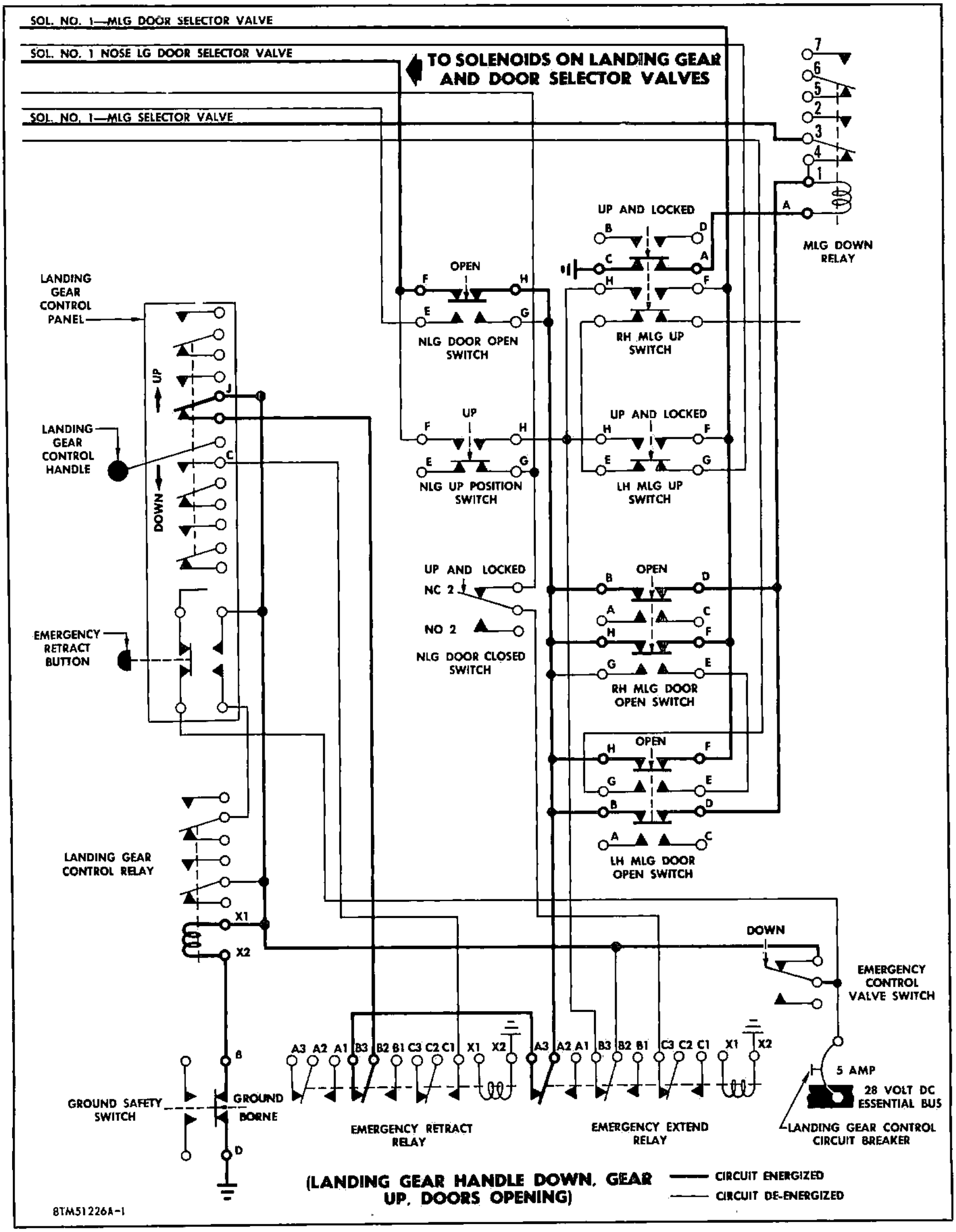

Figure 4-13. Landing Gear Circuit for Door Opening

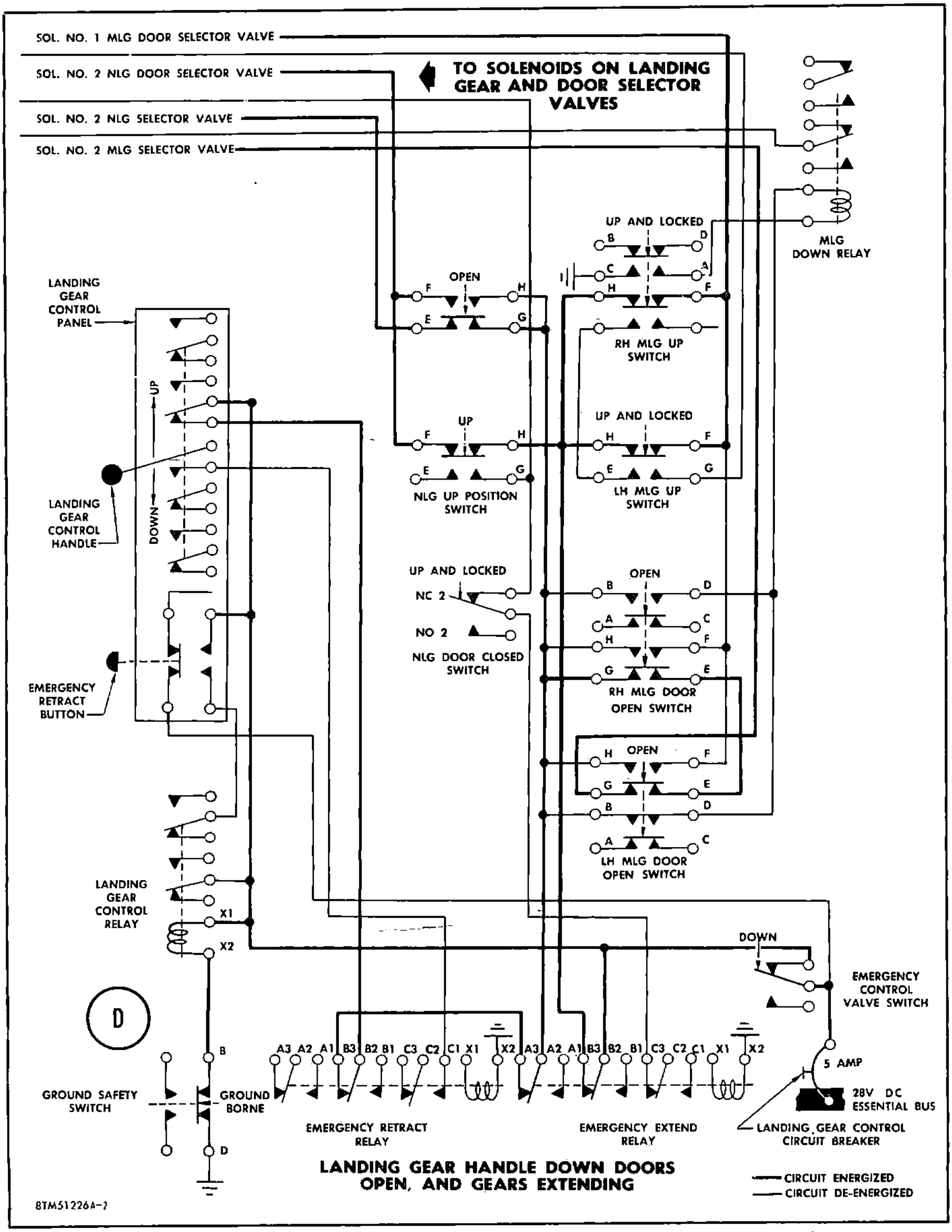

Figure 4-14. Landing Gear Circuit for Gear Normal Extension

connected in series, the main gear will not begin to extend until both of the MLG doors have reached their full down-and-locked positions.

Note in figure 4-14 that the current path to the selector valves is not broken after the gears reach their down-and-locked positions. After the gears have extended, hydraulic pressure remains on the gear and gear door actuating cylinders. This arrangement maintains the gears in the down-and-locked position and prevents the landing gear doors from buffeting. As mentioned earlier in this chapter, there are no mechanical locking devices provided for the gear doors when they are in the open position.

How The Electrical Circuit Controls The Gear Emergency Retraction Operation.

The landing gear may be retracted while the aircraft is either in the air or on the ground if an emergency makes such action advisable. The control for emergency retraction is the sheathed pushbutton placarded EMER GEAR UP on the panel and located just above the warning light test switch. The electrical circuit control for emergency gear retraction is almost the same as that which we have just analyzed with the exception of the ground safety switch and the emergency retract button itself. The circuit is shown in figure 4-15. As the legend at the bottom of the schematic indicates, the regular landing gear control handle is in the GEAR UP position, the gear doors are open, the emergency retraction button has been pushed, and the gears are retracting.

When the airplane leaves the ground, the ground safety switch is deactuated. Note in the lower left section of the schematic that deactuating of the ground safety switch breaks the ground circuit for the landing gear control relay. The current from the bus then travels to terminal of the emergency retract button. When this emergency retract button is pushed in, the current has a path through the emergency switch, through the landing gear control relay, through the landing gear control handle, and to the emergency retract relay.

Note that the emergency retract relay is de-energized. The only time that the emergency retract relay is energized is during an emergency gear retraction when the airplane is on the ground. With this condition, the ground safety switch will be at the groundborne position and ground the landing gear control relay. The control relay, in turn, supplies a path for current to the emergency retract relay. In figure 4-15, though, the circuit is shown when the emergency gear retraction is taking place while the airplane is in the air.

Tracing the current path from terminal C3 of the emergency extend relay, note that the current takes two different paths. One path leads through the NLG door closed switch and to the No. 1 solenoid of the NLG selector valve. This selector valve then directs hydraulic pressure to the retract side of the actuating cylinder. The other current path goes to the MLG door switches, through the MLG down relay, and then to the No. 1 solenoid of the MLG selector valve. The MLG selector valve directs hydraulic pressure to the retract side of the MLG actuating cylinders. Note that the MLG selector valve will receive current as long as any one of the four MLG door-closed switches are not actuated.

Tracing the current path from B3 of the emergency extend relay, note that it passes through the NLG up position switch and then to the No. 1 solenoid of the NLG door selector valve. This keeps the NLG door open until the nose landing gear has fully retracted.

In the emergency retraction circuit, just as in the gear normal retraction circuit, current is continued to the landing gear selector valves so that they will direct hydraulic pressure to the main landing gear actuating cylinders until the MLG doors have reached their full up-and-locked positions.

The Electrical Circuit During Gear Emergency Extension.

As you have already learned in this chapter, the landing gear is extended during emergencies by pneumatic pressure. No electrical controls are needed for emergency gear extension. Instead, however, there must be an absence of electrical control. Removal of all electrical power from the landing gear control system is accomplished by the emergency control valve switch. This emergency control valve switch is a limit switch mounted on the pneumatic control valve in the landing gear emergency extension system. As the control handle in the cockpit is pulled out to the full-out position, a small lever on the control valve actuates the limit switch. The switch then breaks the circuit just beyond the landing gear control circuit breaker. This effectively removes all electrical power from the landing gear system.

MAINTENANCE REQUIREMENTS ON THE LANDING GEAR CONTROL CIRCUITS.

The maintenance responsibilities for the landing gear control circuit will be relatively simple if you have the proper tools and the required knowledge. As mentioned earlier in the discussion of the hydraulic system maintenance requirements, the landing gear malfunctions will usually consist of "squawks" which the pilot has written in the airplane Form 1A. In a very general sense, the pilot remarks will consist of "something the landing gear system did that it *wasn't* supposed to do", and "something the landing gear system didn't do that it *was* supposed to do."

To effectively determine just what is wrong with the landing gear system, you should always place the airplane on jacks and cycle the landing gear so that you

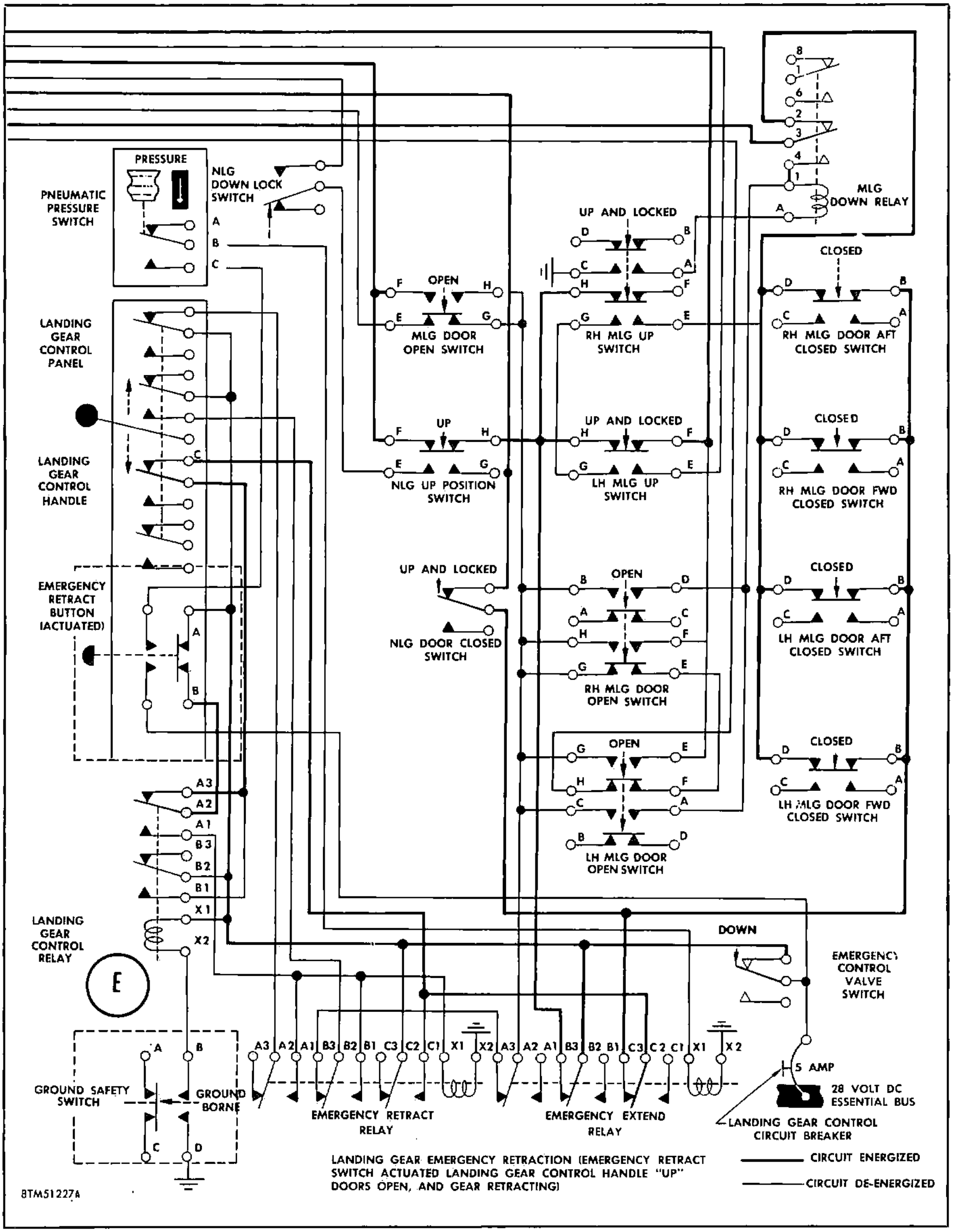

Figure 4-15. Landing Gear Emergency Retraction Circuit

can analyze the malfunction symptoms. After determining which cycle is malfunctioning (extension or retraction, and doors or gears) refer to the electrical schematic and refresh your memory on which components take place in that particular phase of the operation. Then with electrical power *on,* use a voltmeter to check for 28-volt d-c at each switch, connection, solenoid, and relay where current should be passing. This method will allow you to determine just where the circuit is not receiving any current.

In some instances you may find that the circuit is energized at places where it should not be. It is quite possible for a switch to become defective and pass current through when it should not. In this case, you will have to check the solenoids on both sides of the selector valve to determine if both solenoids are being energized at the same time. As you can well understand, having both solenoids on one selector valve energized is just the same as having neither solenoid energized.

Defective components must be replaced with serviceable items. The location of the selector valves, the circuit breaker, and the cockpit controls have already been mentioned. The MLG down relay is situated on the forward bulkhead in the right main wheel well. The other three relays discussed—emergency extend, landing gear control, and emergency retract—are all located on the forward bulkhead in the nose wheel well. All of the components in the landing gear control circuit have conventional attaching devices and electrical connections.

NOSE WHEEL STEER-DAMPER HYDRAULIC SYSTEM.

Directional control of the F-102A while taxiing on the ground, and nose wheel shimmy damping during the takeoff and landing run are accomplished by means of the nose wheel steering-damping system. There are only two hydraulic components, in this nose wheel steering-damping system, an electrically-actuated hydraulic shutoff valve and a hydraulic steer-damper unit. Control of this system is by means of a switch on the control stick and movement of the rudder pedals. Hydraulic pressure for this system is supplied by the secondary hydraulic power supply system.

You will recall from the preceding discussion of the landing gear hydraulic system that the steer-damper system receives its hydraulic pressure through the nose landing gear selector valve. You will also recall that pressure can only go to the steer-damper system when the gear selector valve is in the *gear extend* position. Thus, you can see that this steer-damper system can only be used when the gear is *down.* In addition, the steering phase can only be used when the weight of the airplane is on the gear and its "oleo" strut is compressed. You will learn just how steering and shimmy damping is accomplished as we go through the nose wheel steering and damping hydraulic system.

NOSE WHEEL STEER-DAMPER OPERATION.

Whenever the pilot wishes to turn the airplane while taxiing on the ground, he presses an electrical switch on his control stick and pushes either the left or right rudder pedal. Actuating the switch completes an electrical circuit to the steering hydraulic shutoff valve. The valve opens and admits hydraulic pressure to the steer-damper unit. With operating pressure in the unit, the nose wheel turns left or right when the pilot pushes on either the left or right rudder pedal. This turning action is accomplished by the rotation of a control shaft in the steer-damper unit, which is rotated by mechanical linkage from the rudder pedals.

Figure 4-16 shows the mechanical linkage between the rudder pedals and the steer-damper unit on the nose gear strut. Note how this linkage connects to the steer-damper rotating shaft at the left hand bell crank. Rotation of the shaft ports fluid pressure to either one side or the other of a vane in the steer-damper unit. The vane in turn moves an external steering arm at the bottom of the unit, and a collar assembly on the gear strut, to steer the wheel to either the left or right. When the wheel reaches the desired turning angle, the pilot releases the electrical button and holds the wheel turning angle with his rudder pedals. To realign the wheel or reverse the steering direction, the pilot again presses the electrical button and pushes on the pedal opposite the one originally used to deflect the nose wheel.

When the rudder pedals are in the neutral position, no fluid flows and the unit dampens shimmy (oscillations) of the nose wheel. This is true even though the electrical button may be actuated to pressurize the unit. Also, when the pilot releases his control stick steering switch to the OFF position, the hydraulic shutoff valve closes and stops fluid pressure to the steer-damper unit. The unit then functions solely as a damping control.

The steering electrical circuit incorporates a steering limit switch mounted on the gear strut. This switch is mechanically actuated by a roller operating on a cam track attached to the gear collar assembly. When the nose gear wheel rotates farther than 50° in either direction, the roller leaves the cam track and opens the electrical circuit. This action causes the solenoid in the hydraulic shutoff valve to be de-energized, thus closing off the hydraulic power supply to the steer-damper unit. The limit switch also opens the steering electrical circuit when the nose wheel is not firmly on the ground during takeoff and landing.

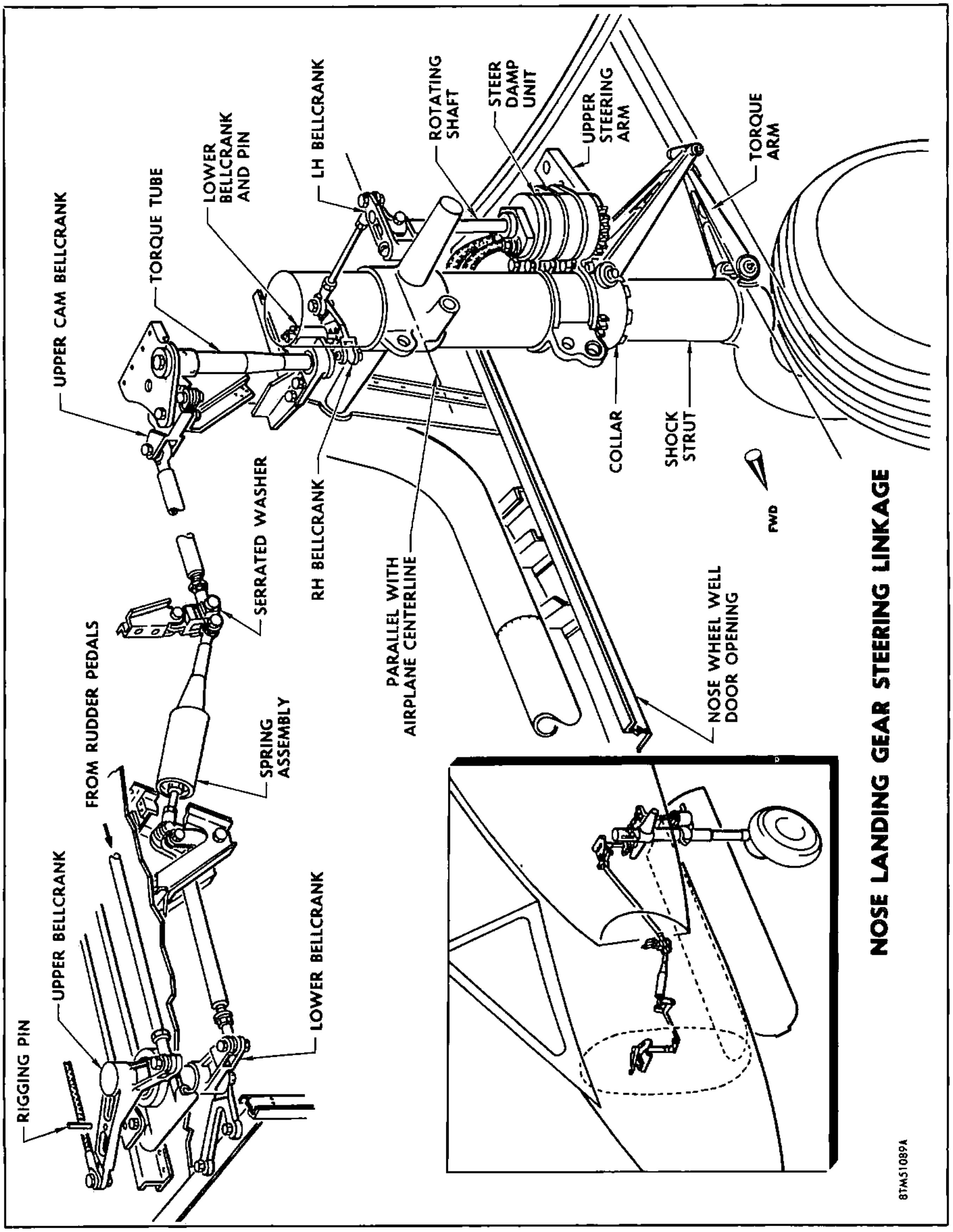

Figure 4-16. Nose Landing Gear Steering Linkage

How The Steering Arm Turns The Nose Wheel.

In figure 4-16 you can see how the horizontal steering arm attached to the shaft is similar in function to the "Pitman" arm that extends from an automobile steering worm-gear box to the front wheel steering linkage. A channel in this arm houses a roller that is attached to the end of another arm which is connected to the collar assembly and torque arm (scissors) on the lower end of the gear strut cylinder. When the steer-damper vane moves to the right or left, it turns the vane shaft and therefore the steering arm attached to it. As the steering arm rotates about the vane shaft, it in turn moves the roller attached to the lower arm assembly in a fore and aft direction. The lower arm and collar then swivels in an arc about the oleo strut, using the strut as a pivot point. The torque arm (scissors) which connects the lower section of the strut (the piston) with the strut mounted collar on the upper section, turns the wheel to follow the rotation of the collar. The roller and channel connecting the two turning arms is necessary because the steer damper unit turning-center is off-set from the turning-center of the gear strut collar.

In other words, the arm attached to the steer-damper unit rotates in an arc about the center of the steer-damper *vane shaft* and the arm attached to the strut mounted collar rotates in a circle around the center of the *gear strut.* The two arcs prescribed by these arms must bisect each other at two points of the arcs they describe. You can see, therefore, that if the two arms were *rigidly* attached to each other and trying to rotate around two different centers, they would bind each other. So the roller slides or rolls inside the channel of the upper arm to allow constant change of the positions of the two arms in relation to each other. The roller must, therefore, pass beyond the outer end of the slot if the wheel is turned too far. This point is established at 50° of wheel turn in either direction from dead-center or neutral.

Disconnection at 50° of Turn.

At the end of 50° of wheel turn, the roller passes completely out of the aft end of the channel in the upper arm, placing the wheel then in a free-castering position. To prevent the wheel from being dislocated in relation to the steer-damper when this happens, gear teeth are built into the outside perimeter of the strut collar and onto the forward end of the upper steering arm. At 50° of turn these two gears automatically engage each other, to maintain the wheel in the correct position relative to the steer-damper vane shaft. As the wheel—and therefore the lower steering arm—moves back toward center, the gears cause the roller to re-engage with the channel. As the wheel continues turning toward the center, the gears separate or disengage.

As mentioned before, a cam track actuates the limit switch roller to open the electrical circuit to the hydraulic shutoff valve of the system. This renders the steer-damper unit hydraulically inoperable until the wheel is manually rotated back to less than 50° from dead-center. Also, the rudder pedals must be deflected to an angle corresponding to wheel deflection from center, before the steer-damper can again be used for steering.

Mechanical System Tie-in.

There is one other control point that isolates steer-damper actuation and immobilizes the nose wheel steering while the airplane is in flight. As the airplane weight lifts off the nose wheel during take off, the lower section of the strut and the attached nose wheel drops to the lower end of the gear strut travel. One centering cam in the lower section of the strut is carried down against a matching centering cam inside the oleo strut to align the nose wheel in a centered position. This prevents the wheel from turning left or right as it retracts into the wheel well.

The strut mounted steer-control linkage is connected to the fuselage mounted portion of the control linkage by a spring-loaded pin as shown in figure 4-16. With the gear in the down position, this pin is held in the slotted left hand bell crank of the strut mounted linkage by the spring. When the pilot retracts the gear, the strut mounted left hand bell crank travels with the gear away from the pin, thus breaking the steering control connection. Of course, this separation makes the steer-damper unit inoperable when the gear is retracted and the rudder pedal motion during flight cannot affect it.

The preceding discussion has shown you how the nose steering system functions. This information will help you to understand the hydraulic operation of the steering and damping system which follows.

NOSE WHEEL STEERING SHUTOFF VALVE.

The nose wheel steering system shutoff valve, which controls the flow of secondary hydraulic system fluid from the nose gear selector valve to the steer-damper unit, is a solenoid operated, spring-loaded, poppet-spool type valve. When the solenoid is energized by the control stick switch, it unseats the spring-loaded poppet. Opening the poppet valve allows pressurized hydraulic fluid that has been trapped at the poppet spool to flow past the seat and through the outlet to the steer-damper unit. When the solenoid is de-energized by releasing the control switch, the spring returns the poppet to its seated or shutoff position. This shutoff valve is shown schematically in the two views of the steer-damper unit (figure 4-17). In the *damping condition* it is closed, while in the *steering condition* it is open.

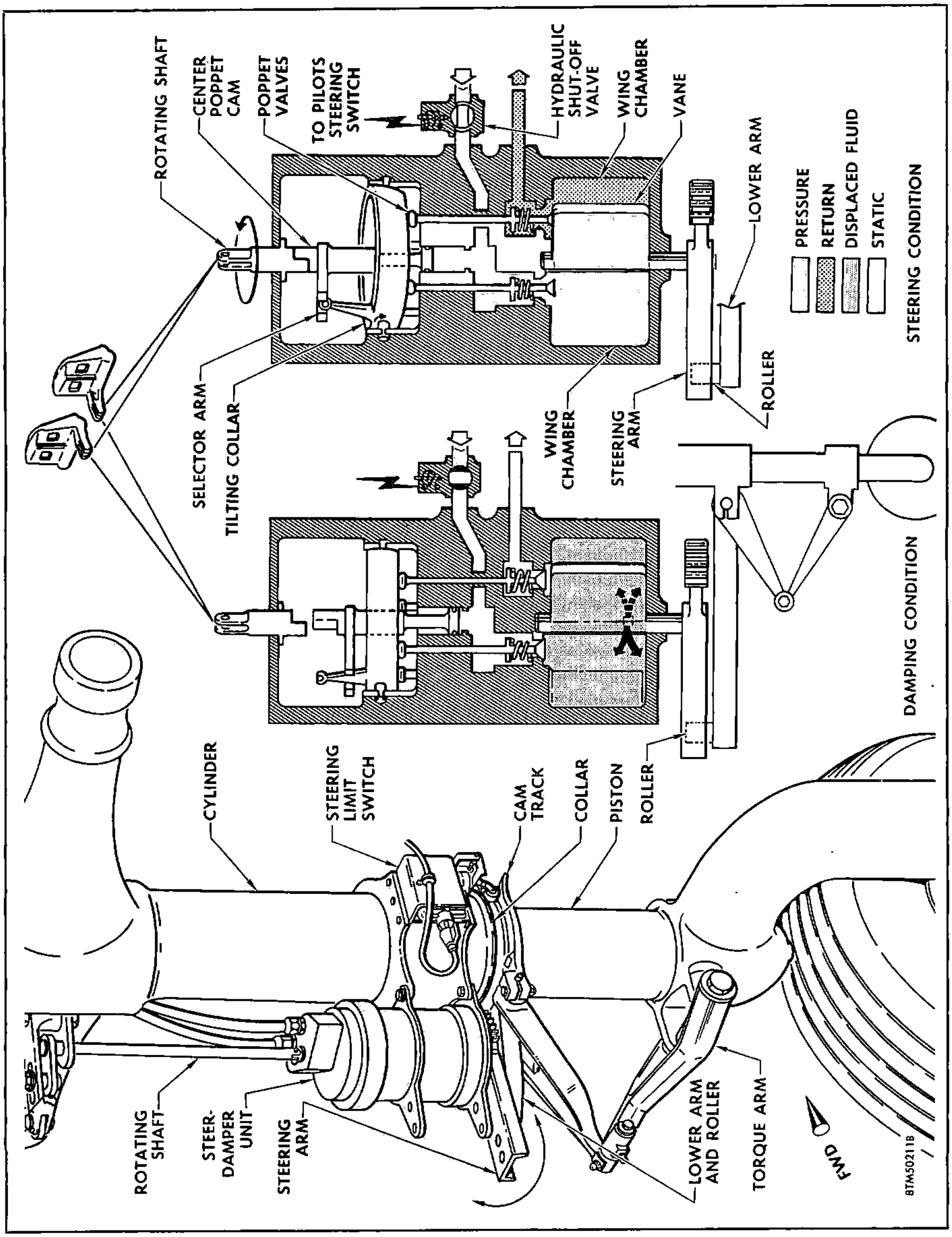

Figure 4-17. Steer-Damper Unit

This spring assembly in the valve incorporates an adjustment screw for adjustment of the opening tension. Normally, adjustment of this spring tension should not be necessary since the valve is pre-adjusted prior to installation. However, when a valve is found to pass fluid in its closed position, the spring tension must be set higher. If the valve will not pass sufficient fluid when opened, the tension must be lightened. This condition is corrected by removing the retainer cap and turning the adjustment screw until the proper tension is acquired.

NOSE WHEEL STEER-DAMPER UNIT.

Figure 4-17 shows the steer-damper unit mounted on the nose gear strut and schematic views of the two functions it serves, steering condition and damping condition. Basically, the unit is comprised of four spring-loaded poppet valves, a *wing* chamber and vane, and a selector arm and tilting collar. The poppet valves (two of which are shown) control the admission of hydraulic pressure into the wing chamber and also open the wing chamber to system return. Since the steer-damper unit serves two functions, we will discuss each function separately.

Steering Function.

In figure 4-17, you will note two schematic views of the steer-damper unit. For this discussion on steering, follow the flow of pressurized fluid through the right view marked *steering condition.* When the weight of the airplane is on the nose gear and the actuating switch on the control stick is pressed, the shutoff valve will be energized to the open position as shown in the right schematic. The first action of hydraulic pressure entering the unit raises the center poppet cam to engage it with the mechanical rotating shaft. (You can see this pin and shaft disengaged in the *damping condition* schematic.)

Turning of the rotating shaft, discussed earlier in this section, causes cam action of the tilting collar in the unit to unseat two of the poppet valves. Whenever a pressure inlet poppet in one side of the wing chamber is unseated, a return outlet poppet in the other side is also unseated. This is accomplished by pilot action of the rudder pedals rotating the selector arm shown on the right hand schematic.

Comparing the two schematic views, note how this action tilts the tilting collar and depresses (opens) one pressure and one return poppet valve. As you can see, this action will admit pressure to one side of the vane in the wing chamber and provide a return outlet for the trapped fluid on the other side of the vane. Hydraulic pressure moves the vane which in turn moves the steering arm to turn the nose wheel. When the vane is centered in the wing chamber, the nose wheel is also centered; and when the vane is forced to move, it forces the nose wheel to turn.

The other pair of poppets, not shown in figure 4-17, actuate in the same manner to move the vane in the opposite direction, thus turning the wheel in the opposite direction. In one respect, the vane and wing chamber can be compared with an ordinary actuating cylinder and piston; fluid under pressure is admitted to one side of the piston (vane) and as it forces the piston to move, a path must be made for the fluid on the other side to escape. If the path or outlet is not provided, the piston moves just enough to compress the trapped fluid to a pressure equal to that being introduced on the opposite side. This principle is employed to some extent in the steer-damper unit when it is used for nose wheel shimmy damping.

Shimmy Damping Function.

When the steering function of the unit is not being utilized, hydraulic pressure is isolated by the closed shutoff valve. However, even though there is no pressure in the unit, it remains full of fluid as shown in the *damping condition* schematic (figure 4-17). When the steer-damper shutoff valve is closed, the poppet valves in the steer-damper are inoperable, fluid is trapped in the wing chamber and the center cam is disengaged from the rotating shaft. If the wheel starts to shimmy from side to side, the vane in the wing chamber is forced to move with it. So, you can readily see that since the fluid has no escape path it will be rapidly compressed and act as a snubber, or shock absorber, when the vane is forced against it.

A bleed passageway through the vane allows enough interflow of fluid to each side of the wing chamber to avoid a "hardness" of action; that is, too abrupt a halt of the vane as it compresses the fluid on either side of the chamber. This interflow is shown by the arrows in the wing chamber. The snubbing effect damps the shimmy motion in one direction, and as the vane and wheel move back in the other direction, the same thing occurs in the opposite side of the chamber. Subsequent oscillations of the wheel are, therefore, progressively damped to back and forth motions of smaller and smaller intensity.

The channel passageway through the body of the vane also serves another purpose. If sustained external pressure—rocks, ruts, and the like—is maintained on the wheel from one direction, the fluid on one side of the vane forces through this bypass passage to reduce excessive pressure in the chamber and, therefore, on the wheel. Because the wheel is designed to track dead-center, the fluid volume on each side of the vane is again equalized by the centering of the wheel when external pressure is removed (wheel has rolled past the obstruction).

Cam Disconnect.

We mentioned that the poppet valves are inoperable when the steer-damper is being used for shimmy damping. This is because hydraulic pressure is removed from the steer-damper unit when the shutoff valve is closed. Referring to the left-hand schematic on figure 4-17, you can see that with the pressure removed, the spring-loaded poppet cam in the upper end springs back to disengage from the steer-damper rotating control shaft. When the unit is being used for steering, fluid pressure overrides the spring and forces the cam to connect with the control shaft as shown in the right-hand schematic.

With the cam disconnected, the fluid is trapped in the wing chamber by the inoperable poppets, and the unit functions as a shimmy damper. Note that the connect point of the cam and shaft is slotted, or off-set, so that the wheel has to be pointing straight ahead, or the rudder pedals aligned to the wheel position, before mechanical re-engagement of the two parts can be accomplished. You can see on the right-hand schematic in figure 4-17 that when the shutoff valve is opened, fluid pressure re-engages the cam with the control shaft for steering control.

Maintenance of Steer-Damper Hydraulic System.

The following is a general discussion of some of the hydraulic maintenance problems of the steer-damper system that you may encounter and their remedies. Since only the hydraulic phase of this system was discussed in detail in this section, you should refer to the Airplane General Supplement in this training series for complete coverage of the landing gear and nose steering system.

Whenever the steering system malfunctions, you should first check that the secondary hydraulic system has sufficient operating pressure. Following this check, determine that electrical power is reaching the solenoid shutoff valve and that the mechanical linkage between the rudder pedals and the steer-damper unit is correctly adjusted. Assuming that all the above are satisfactory, then you can isolate any troubles to the shutoff valve and steer-damper unit.

The shutoff valve can cause two types of malfunctioning. If malfunctioning it will either admit hydraulic fluid to the steer-damper unit when de-energized, or will not open when energized. If the first condition exists, the nose wheel will turn when the rudder pedals are moved even though the control stick switch is not actuated. If the second condition exists, the nose wheel will not turn when the control switch is actuated and the pedals are moved. In either condition, remove and replace the valve.

After determining that the shutoff valve is satisfactory, then check the operation of the steer-damper unit. Before replacing a steer-damper unit that you feel is malfunctioning, you should again check other possibilities that would appear as though this unit were at fault.

If the steering is erratic and the nose wheel has a tendency to hunt, it may be due to air in the hydraulic system, steering linkage binding, or the steer-damper unit rotating shaft binding. For correction of the above you should bleed the air from the system and check the linkage for binding. If the above corrective measures do not remedy the malfunction, the steer-damper unit is probably at fault and should be replaced. In addition, if you cannot get a full 50° turning of the nose wheel and the steering arms are not binding, then the steer-damper unit is probably again at fault and should be replaced.

In the event the nose wheel shimmies during the take off or landing runs, you should first suspect the nose wheel and tire assembly of being out-of-balance. If replacement of the nose wheel and tire assembly does not correct a shimmy condition, then the steer-damper unit is probably bypassing or leaking fluid internally and must be replaced.

You should refer to your F-102A Maintenance Manual, T.O. 1F-102A-2-8, from time to time for the latest procedures in replacing or adjusting the nose wheel steering and damping system components.

SPEED BRAKE HYDRAULIC SYSTEM.

The speed brakes on the F-102A are two hydraulically operated door-type surfaces located at the aft end of the vertical fin island. These brakes are contoured to form the aft part of the fin island and are hinged at the forward end to swing sideways into the airstream, as shown in figure 4-18. The main purpose of the brakes is to slow the airplane while in flight and when making a landing approach. In addition to functioning as speed brakes for the airplane, these doors also provide a housing for the drag chute. The drag chute is stowed in the compartment just forward of the point where you see the actuating cylinders attached to the airplane structure. When extended into the airstream, these brakes provide a free opening through which the drag chute is deployed.

Four hydraulically actuated cylinders extend and retract the speed brakes. These cylinders, two on each brake, are attached to the airframe structure at the forward end through fittings on the drag chute housing. The piston ends of the cylinders hinge to the speed brake structure. Two cylinders are furnished for each brake so that the brakes will be stable and not flutter when extended into the airstream. When the doors are extended at excessive airspeeds, a relief valve in the speed brake hydraulic system bypasses pressure fluid around the actuating cylinders so that the brakes cannot be forced to the full open position.

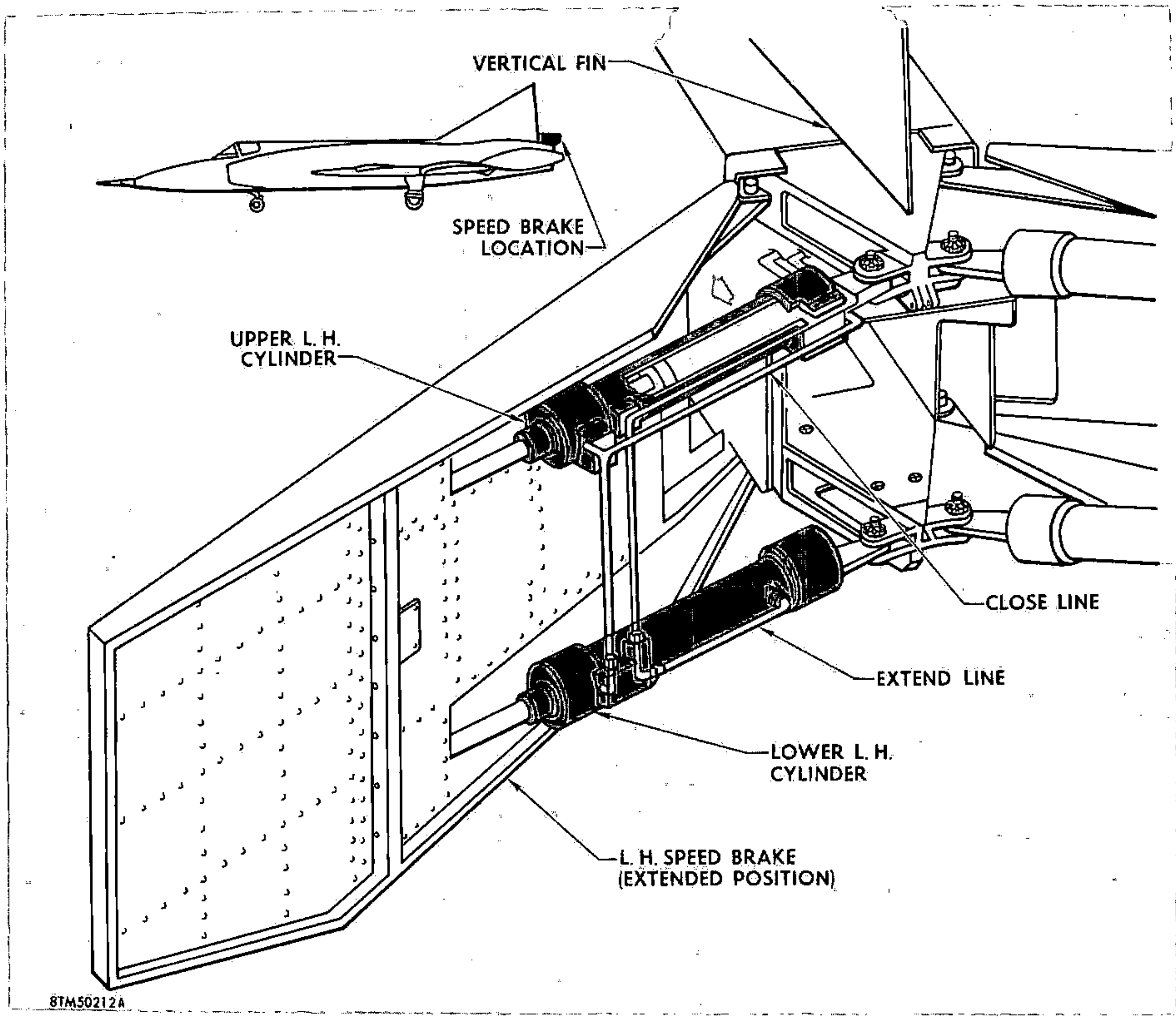

Figure 4-18. F-102A Speed Brakes

HOW THE SPEED BRAKES OPERATE.

As mentioned above, there are two conditions under which the speed brakes are actuated to their open position. One of these conditions is when the pilot desires braking action to slow the airplane while in flight. To accomplish this braking action the pilot pushes the speed brake switch on the power control lever to the OUT position. Actuation of the switch to the OUT position completes an electrical circuit to the *extend* solenoid of the speed brake hydraulic selector valve. With the selector valve solenoid energized, pressurized hydraulic fluid then passes through the selector valve to the *extend* side of the four speed brake actuating cylinders. These cylinders open the brakes to a full 45° of travel to slow the airplane's speed. The speed brakes will remain extended until the pilot moves the switch on the power control lever to the IN position. The speed brakes will partially retract if opened at excessive speeds. This automatic retraction feature is accomplished by the opening of a relief valve in the speed brake hydraulic system.

When the drag chute is to be deployed to slow the landing roll, the pilot pulls a T-shaped handle on the left side of the instrument panel. This action not only operates a pneumatic valve which deploys the chute, but also actuates an auxiliary speed brake electrical switch. The switch completes a circuit to the speed brake selector valve to extend the brakes. An interlocking device in the chute deployment assembly prevents the deployment of the chute until the brakes have opened a minimum of 30°, thus insuring that the chute will not jam against the brakes and fail to open properly. The interlocking device is comprised of two pins that hold the drag chute release lever in a down position.

As the speed brakes open, flat spots on the head ends of both top speed brake cylinders contact two pivot assemblies. Rotation of the pivot assemblies by the cylinders pulls the two pins from contact with the drag chute release lever when the doors are opened a minimum of 30°. This frees the lever and allows it to release the drag chute ripcord mechanism.

A safety switch incorporating a time-delay feature is mounted just aft of the drag chute release mechanism to prevent a premature closing of the speed brakes before the chute is jettisoned. Jettisoning of the drag chute by the pilot at the end of a landing roll automatically pulls a pin from the safety switch; this switch then energizes the speed brake retract circuit to close the speed brakes.

SPEED BRAKE HYDRAULIC SYSTEM OPERATION.

The speed brake hydraulic system, shown schematically in figure 4-19, is a comparatively simple operating system. This system consists of a dual solenoid-operated selector valve, four actuating cylinders, two restrictors, a relief valve, and a check valve. The secondary hydraulic power system, as you know, furnishes the operating pressure for the speed brakes and is always present at the solenoid valve inlet port when the engine is running.

You learned earlier in this section that the speed brakes are operated either by the switch on the power control lever or by an auxiliary switch that is actuated by the drag chute control. Since the hydraulic system operation is the same regardless of the method of control, let's consider the hydraulic system as being controlled by the power control lever switch.

Figure 4-19 also shows the speed brake system as it is when the control switch is in neutral and the brakes are closed. Note that secondary hydraulic system pressure is at the selector valve, but is stopped by the centered position of the pilot spool. The pressure in the *open* and *close* lines then would be in a static condition.

Now, let's open the brakes by moving the power control lever switch to the OUT position. This action completes an electrical circuit that energizes solenoid No. 1 in the selector valve. Energizing this solenoid pulls the pilot spool up to allow hydraulic pressure to pass the pilot spool and push the slave piston up. With the slave piston up, hydraulic pressure at the pilot spool then passes around the slave piston and into the *open line* where it is routed to the right and left speed brake cylinders.

The restrictors in each open line are merely fittings with restricted (smaller) passages that reduce the flow of fluid to prevent the brakes from "slamming" open or closed. Note that pressurized fluid is first routed to both upper cylinders, then to the lower cylinders. Hydraulic pressure entering the cylinders forces the plunger housing back, which releases the latching finger. (The latching finger holds the cylinder in the retracted position in the speed brake *closed* position.) With the latching finger disengaged, pressure extends the piston and piston rod to open the brakes.

Return fluid from each lower cylinder passes to its respective upper cylinder then to the selector valve. At the selector valve return fluid flows past the pilot spool and into the *return line*. The check valve in the return line prevents any reverse flow of fluid into the speed brake system.

When the control switch is moved to the IN position, solenoid No. 2 is energized causing the pilot spool and slave piston to reverse the positions they were in when the brakes were opened. Pressurized fluid then enters the *close line* and retracts the actuating cylinders. Return fluid must pass through the restrictors in the *open line* and out the selector valve to the return line. Again these restrictors limit the flow of fluid so that the brakes will not "slam" closed.

When the actuating cylinders reach their fully retracted position they engage the latching fingers and actuate the limit switch plunger. Actuation of the cylinder limit switch *opens* the circuit to de-energize the No. 2 solenoid and neutralize the selector valve. With the selector valve in neutral, no pressure exists on the retract side of the cylinders. The latching fingers which engaged when the cylinders are retracted hold the brakes in the closed position.

As mentioned before, the speed brakes will automatically retract or partially retract at excessive airspeeds. This automatic retraction is due to the opening of the relief valve shown in the schematic. This relief valve opens at a predetermined pressure setting to bleed off excess pressure in the *open line,* thus allowing the brakes to retract. Excess pressure in the open line is caused by the excessive airloads creating a back pressure.

SPEED BRAKE HYDRAULIC SYSTEM COMPONENTS.

The speed brake hydraulic system consists of those components shown in the schematic of the system in figure 4-19. These components are the solenoid operated selector valve, the four brake actuating cylinders, the two restrictor valves, the system relief valve, and the system check valve. Of all these components, only the actuating cylinders are different from other components you have learned about in this manual. The selector valve is similar to the type used in the landing gear system, except that the slave piston blocks the system *open* and *close lines* when the valve is de-energized.

The restrictors, relief valve, and check valve operate like those you learned about in Chapters I and II. So, we will just discuss the actuating cylinders and how they operate in opening and closing the speed brakes.

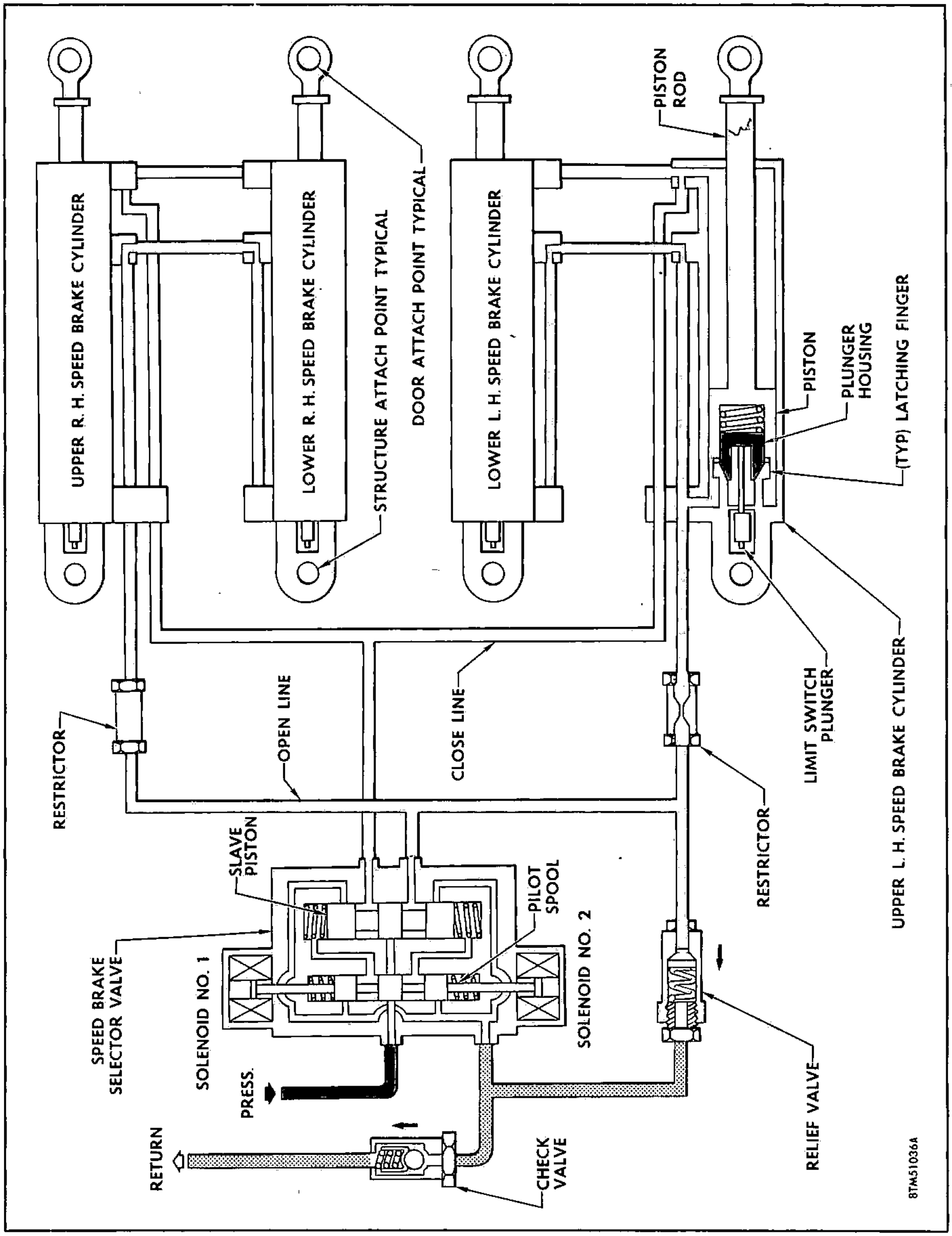

Figure 4-19. Speed Brake Hydraulic System Schematic

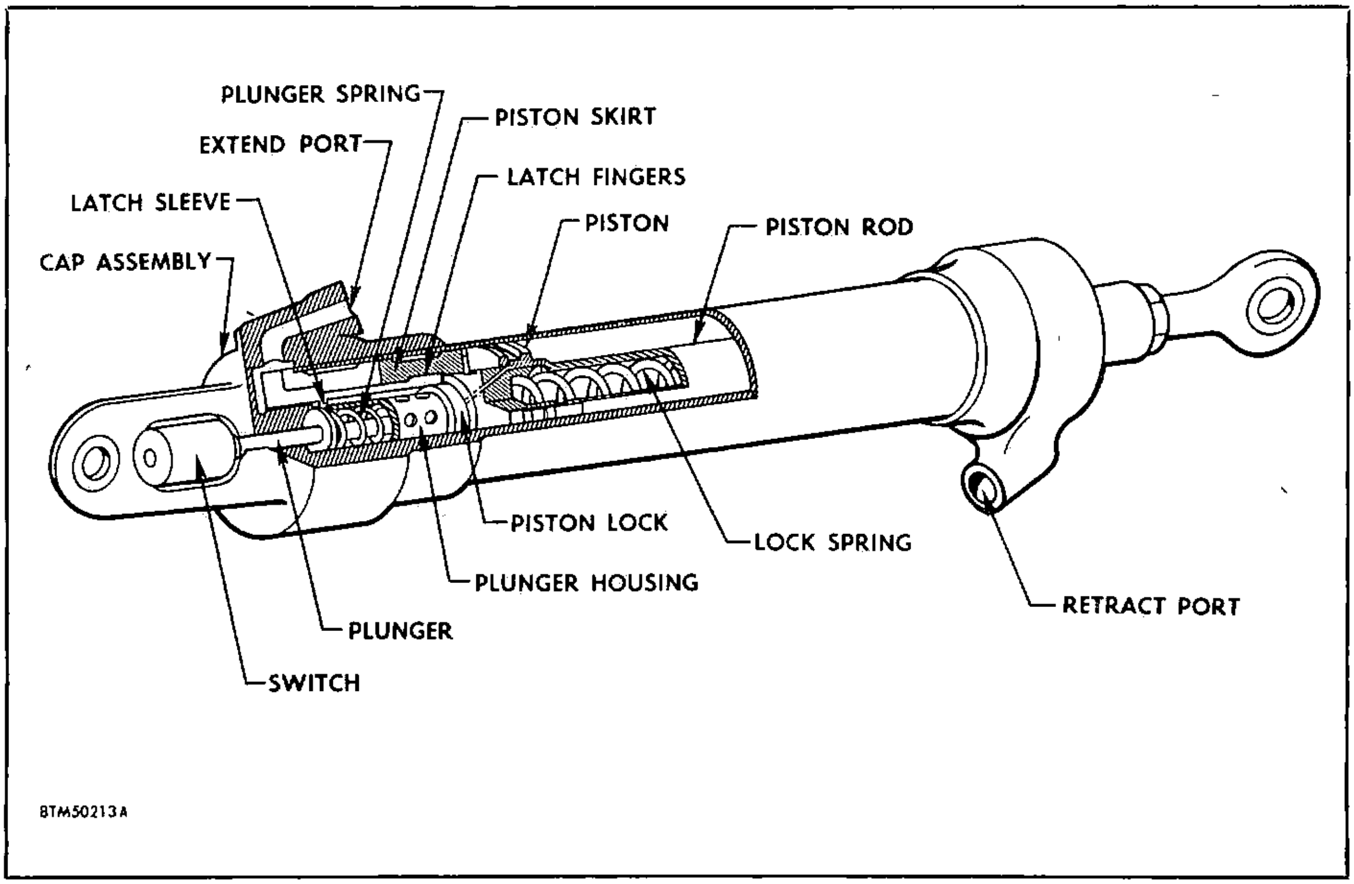

Figure 4-20. Speed Brake Actuating Cylinder

Speed Brake Actuating Cylinders.

All four speed brake actuating cylinders are similar in construction. These cylinders are conventional in design, except for several unique features. One of these features is an integral lock that holds the brakes closed without the aid of hydraulic pressure. Another feature is the method of routing fluid to the cylinders. As you can see in figure 4-18, the cylinders incorporate side fittings in the caps on each end. These fittings have passageways that route pressure and return fluid between the two cylinders on each brake. With this fluid routing method, only one pressure line and one return line is required to furnish hydraulic power to the cylinders on each brake.

Actuating Cylinder Operation.

The cutaway illustration (figure 4-20) shows the internal operating components of one of the speed brake actuating cylinders. Note the integral piston lock and limit switch that was discussed in the system operation above. When the piston has moved to its retracted position in closing the speed brakes, as shown in the illustration, the inside groove of the piston skirt is aligned with the latch fingers. (There are six of these latch fingers on the latch sleeve and they surround the plunger housing and piston lock.) With the piston and piston skirt in the fully retracted position, the lock spring within the piston rod then thrusts the piston lock inside the perimeter of the six latch fingers as shown. This action expands the latch fingers causing them to engage in the inside groove of the piston skirt and lock the cylinder assembly in the retracted position.

A second function of the piston lock is to actuate the limit switch plunger shown on figure 4-20. When the limit switch is actuated, the electrical circuit to the *close* side of the system selector valve is opened. This opened circuit allows the spring-loaded selector valve to neutralize and shut off fluid pressure to the actuating cylinders. By locking the cylinders in the retract position and shutting off fluid pressure to them, the brakes are mechanically locked and do not have to rely on hydraulic pressure to hold them closed.

Note that the limit switch plunger is spring-loaded to return to its normal OFF position when the piston lock releases it by moving to the extend position. As hydraulic pressure is admitted to the *extend* side of the cylinder to extend the piston, the pressure overrides the plunger spring in the piston lock. The same fluid pressure then forces the piston lock to retract into the piston rod and allow the latch fingers to spring in and release the piston skirt so that the piston and its piston rod can move to the extend position and open the speed brakes.

MAINTENANCE.

The hydraulic maintenance problems of this speed brake system are much the same as those that you would encounter in any other hydraulic system. The problems would result in a sluggish or inoperable system. The causes again could be leaks (either internal or external), a malfunctioning unit, an *open* electrical control circuit, or no system pressure. Servicing of this system is comparatively simple—periodic lubrication of the actuating cylinders (other than the regular inspections you will make of the system for leaks, security of attachment, and cleanliness), and system check-outs are all that are required.

In trouble shooting the hydraulic system, first determine the most logical thing that could cause the type of malfunctioning at hand. If the speed brakes will not operate either open or closed you would first determine that there was sufficient secondary hydraulic system operating pressure. Assuming that sufficient operating pressure existed, the next most logical item to check would be the controlling unit, the selector valve. Since this unit controls the entire speed brake hydraulic system, its malfunctioning would affect the brake operation as a whole. If an electrical check proved that current was reaching the selector valve, then the valve would be replaced.

If the speed brake hydraulic system operates, but not satisfactorily, then you would suspect some unit or units downstream from the selector valve as causing the malfunction. Since there are only a few units in this system, analyze the problem and then tackle it from the most logical cause. By referring to the schematic of the system (figure 4-15) and to the trouble shooting charts in your F-102A maintenance manual, 1F-102A-2-8, you will soon determine the cause of a malfunction. Experience is the best teacher and you can get this experience only by "doing." So, analyze any problem that comes up and tackle it from the most logical cause.

EMERGENCY A-C GENERATOR HYDRAULIC SYSTEM.

The emergency a-c generator (sometimes referred to as an a-c alternator) is driven by a hydraulic motor which receives its power from the secondary hydraulic system. The emergency a-c generator system is placed in operation only when the main a-c generator fails. Should the main a-c generator become inoperative, a warning light on the main instrument panel will come on, thus notifying the pilot that he does not have a-c power. The pilot then actuates his A-C GEN switch to *reset*. If this action does not correct the situation, he moves the A-C BUS switch from *normal* to *emergency*. Placing this switch in *emergency* completes an electrical circuit which energizes the a-c generator motor shutoff valve solenoid. With the solenoid energized, secondary hydraulic system power then drives the emergency a-c generator hydraulic motor to produce emergency a-c electrical power. The A-C POWER FAIL lights will continue to burn even though the emergency a-c power system is operating. However, the pilot can check the output of this emergency system by referring to his a-c voltmeter.

HYDRAULIC SYSTEM OPERATION.

Figure 4-21 shows the emergency a-c generator hydraulic system in operation. Note that the shutoff valve solenoid is in the energized condition. In this condition the valve spool is positioned so that fluid pressure from the hydraulic system can pass through the valve to the a-c generator drive motor. Hydraulic pressure entering the motor pushes the nine drive pistons up, causing the cylinder block to rotate and thus drive the a-c generator through a reduction gear box that turns the generator at the required speed to produce emergency a-c electrical power. When each piston reaches the top of its stroke, it is aligned with the motor outlet line which routes fluid back to the system return line through the check valve. This check valve prevents reverse flow of fluid through the hydraulic drive motor.

As mentioned above, the shutoff valve is energized only when the a-c bus switch is in *emergency* position. When this switch is in *normal*, the shutoff valve is de-energized and fluid cannot pass through the valve to the a-c generator drive motor.

A-C GENERATOR MOTOR SHUTOFF VALVE.

The emergency a-c generator motor shutoff valve is located in the top of the hydraulic accessory compartment directly above the access door. This valve is the same type of shutoff valve that you learned about in the nose wheel steer-damper system. It consists mainly of a solenoid, a sliding spool, a spring, and a housing as shown in figure 4-21. When de-energized, the spring positions the sliding spool so that it closes the inlet port and prevents secondary hydraulic system fluid from passing through the valve. Energizing the valve solenoid causes the sliding spool to overpower the spring and move to a position so that hydraulic pressure can pass through the valve.

Although the spring tension is adjustable on this shutoff valve, it is advisable to replace the valve if it malfunctions. This spring tension should be bench adjusted using special test equipment. Adjusting spring tension while the valve is installed in the system is only a hit-or-miss method and should not be attempted. Maintenance on the shutoff valve is comparatively simple—either it operates or it doesn't. If the emergency generator does not operate when the A-C BUS

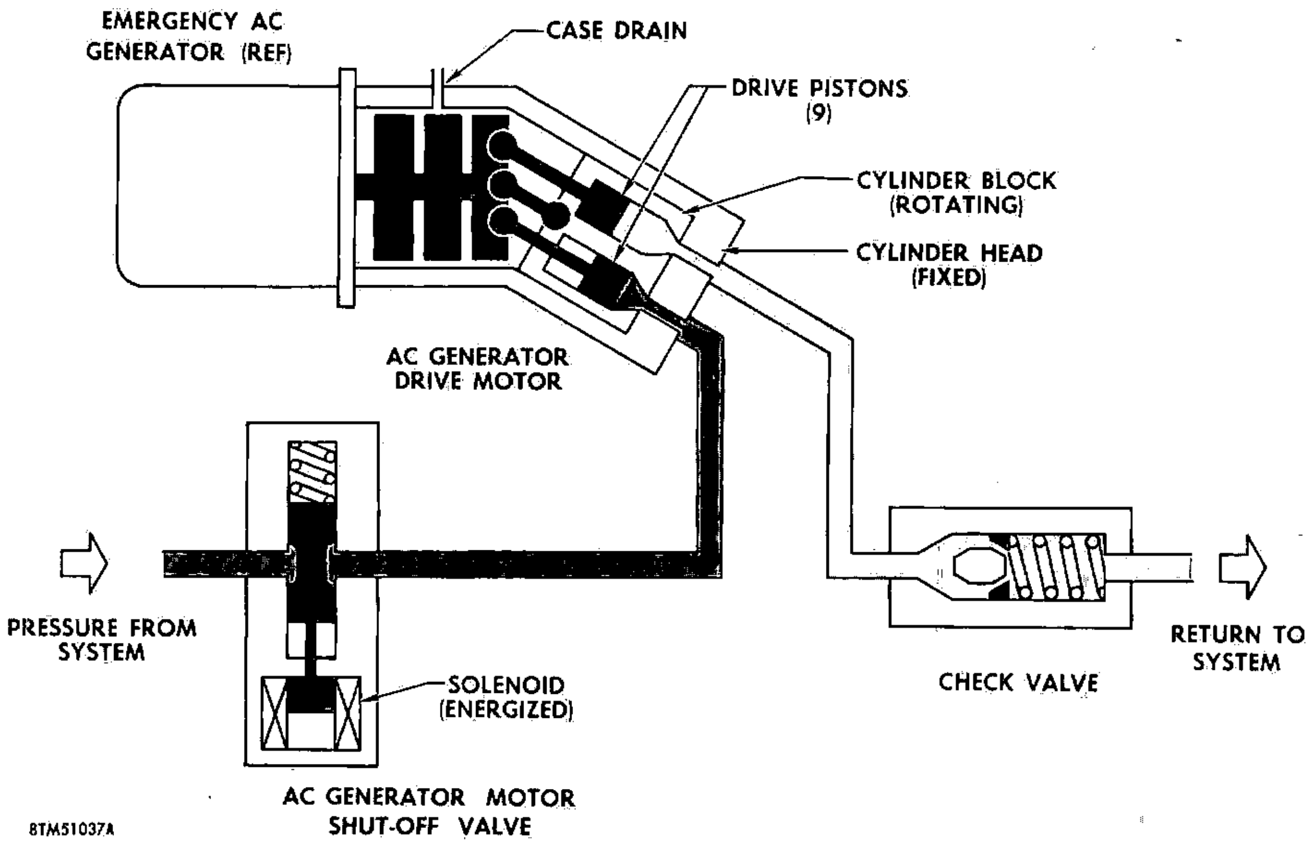

Figure 4-21. Emergency A-C Generator Hydraulic System Schematic

switch is placed in *emergency* position and secondary hydraulic system pressure is sufficient, the first thing to suspect would be a malfunctioning shutoff valve.

A-C GENERATOR HYDRAULIC-DRIVE MOTOR.

The a-c generator hydraulic drive motor is a constant-speed, nine-cylinder, fixed-stroke unit. Basically, this motor is the same as the emergency hydraulic pump you learned about in Chapter II, except that instead of producing hydraulic pressure, this motor converts hydraulic pressure into torque force to drive the emergency a-c generator.

Figure 4-22 shows the hydraulic motor and a-c generator unit that is installed in the forward outboard corner of the right main wheel well. The plate with two curved slots, one for pressure inlet and one for return outlet, is attached in a fixed position against the end of the rotating cylinder block. This cylinder block, which houses nine pistons, rotates in a clockwise direction as the curved arrow indicates. As hydraulic pressure enters the *inlet* port (black arrows) it pushes up on the piston aligned with its curved slot. Pressure applied against the piston rotates the drum which is at a 30° angle to the piston rods. Rotation of the drum turns the cylinder block through the universal link so that the next piston is aligned with the pressure slot.

When the pistons reach the upper end of their stroke, they align with the *return* slot (white arrows). Then as the pistons start their down stroke they force fluid out the curved return slot. Each piston is at the bottom of its stroke when it reaches the end of the *return* slot and is again ready for a pressure stroke. Once pressure is applied against the pistons, rotation of the drum and cylinder block is continuous until hydraulic pressure is stopped. Rotation of the drum drives the a-c generator in the direction of the arrow shown on the spline connection.

About the only maintenance you will have on this unit will be leaks or an inoperative unit. If any leaks occur at the pressure and return port fittings, you correct this trouble by tightening or replacing the fittings. Don't forget to use two wrenches when tightening a line nut on a fitting. By using two wrenches—one on the line nut and one on the fitting—you will prevent a twisting strain on the line or additional leaks at the fitting. If any leaks occur around the hydraulic motor, replace the entire motor, do not attempt to overhaul or repair it.

If you encounter a malfunctioning emergency a-c generator unit, first determine whether the hydraulic motor or the generator is at fault; then replace the malfunctioning unit. It is possible for a generator not

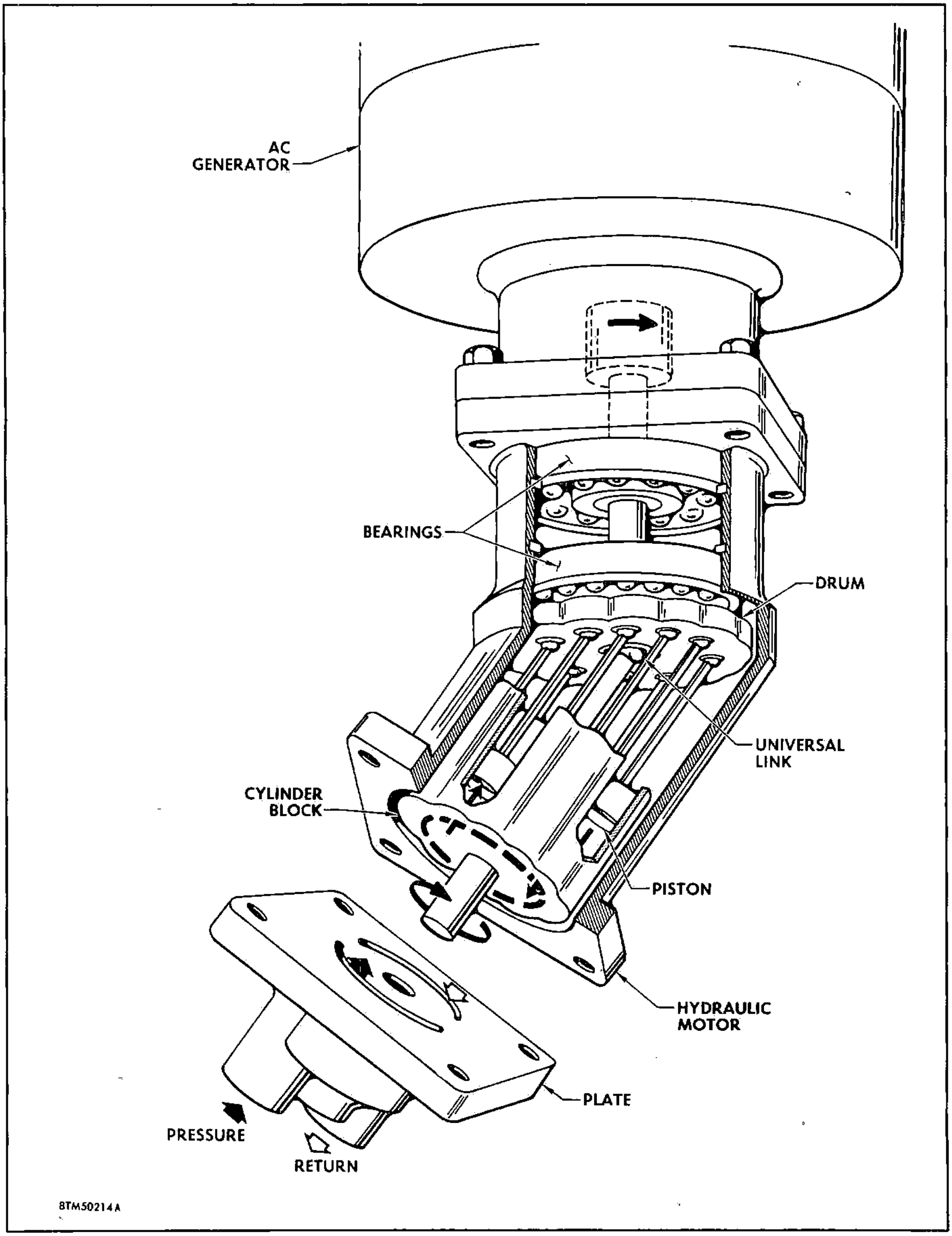

Figure 4-22. Emergency A-C Generator Hydraulic Motor

to produce a-c electrical power and still have its drive motor functioning properly. In this case you would replace the generator. Then, if the generator were satisfactory and the drive motor malfunctioned due either to a sheared shaft or a binding condition, you would replace the drive motor.

HYDRAULIC SUBSYSTEM OPERATION CHECK-OUTS.

Operational checks are checks accomplished on the ground to prove that any equipment worked on has been installed, adjusted, reassembled, or repaired satisfactorily. The checks are made in such a manner as to duplicate as closely as possible those conditions which will be encountered during actual use of the airplane. The operational checks are made on equipment that has been worked on during inspection, maintenance, or repair operations. The equipment being checked is operated through at least five cycles unless its particular design makes that rule impractical. In the latter instance, the duration or frequency of an operation is listed in your F-102A maintenance manual, T.O. 1F-102A-2-3, or will be determined by the maintenance officer.

HYDRAULIC POWER FOR OPERATIONAL CHECK-OUTS.

We know that all subsystems discussed in this chapter depend upon secondary hydraulic power for their operation. Therefore, hydraulic power (from a portable test stand) is used during operational check-out operations. We are primarily interested, in this chapter, in the general procedure for conducting checks to insure the correct functioning of the hydraulic portion of these subsystems. Consequently, we will not discuss other parts of the operational checks more than is necessary to illustrate a point. When a test stand is used, you *must* hook it up to both the primary and the secondary hydraulic systems. Otherwise, pressure in the secondary system alone will cause the hydraulic fluid to bleed through interconnecting components to the primary system.

PRE-CHECK PREPARATION.

You must make preparations and observe all safety precautions before making a check of any system. Remove power from any electrical circuits adjacent to hydraulic equipment unless a particular circuit is to be used for the check to be accomplished. Clear all areas around doors, struts, elevons, or any other moving parts. Before you apply electrical or hydraulic power to the airplane, station personnel at switches and levers that will be powered. This precaution will avoid inadvertent actuation that would endanger personnel. Visually check all parts and components for obstructions and leakage.

Before conducting tests of those subsystems that either combine air or hydraulic power for their operation or use air for emergency operation, make sure that sufficient pressure is in the high pressure pneumatic system. If sufficient pressure does not exist, connect the high-pressure air compressor (3000 psi) to the high-pressure pneumatic system filler connection and charge the system until air pressure stabilizes at approximately 3000 psi.

Prior to conducting the actual checks, set the various switches and levers of the system to the proper position to begin the test as outlined in your F-102A maintenance manual, T.O. 1F-102A-2-3. Whenever you check the landing gear systems, you must jack the airplane to suspend the gear. This will permit the gear to swing free of the ground and simulate actual in-flight operation. NEVER operate the gear control on the ground, unless the airplane is completely jacked.

PREVENTING LANDING GEAR DOOR CLOSURE.

The first part of the landing gear operation check includes inspection and observance of the gear when it is retracted. This is not possible if the doors close when the gear retracts. To overcome this problem disconnect the main landing gear doors and the nose landing gear doors from the actuators. Make sure that you tie or support the cylinders securely in a position that will prevent their jamming against any object when they are disconnected. Of course, after the rest of the landing gear system is checked, you will reconnect the doors and proceed with their test.

BLEEDING AIR FROM SYSTEMS.

As the check of any system begins, observe the flowmeter on the test stand to detect air bubbles in the fluid. If air is present, the fluid will be cloudy and full of tiny air bubbles instead of being a clear transparent red. When you observe this condition, cycle the system until the condition is corrected, since air in the system will cause faulty operation and false gage indication.

OPERATIONAL CHECK-OUTS AND TROUBLE SHOOTING RELATIONSHIP.

The close relationship between the maintenance problems encountered in trouble shooting the airplane and in performing operational check-outs cannot be stressed too strongly. Always bear in mind that the best and quickest way to solve a maintenance problem when it is encountered in an operational check is to refer to the trouble shooting lists found in the maintenance manuals, or what you have learned from previous trouble shooting.

PERFORMING OPERATIONAL CHECKS.

During the actual operational checks observe all parts and components to verify their correct actuation, check for proper extremes of travel, check for leakage during and at end of component travel, and finally check for proper sequencing of the various components in a system. Troubles that could be encountered during the checks and their causes and corrections are outlined in the F-102A Maintenance Manual, T.O. 1F-102A-2-3.

SUMMARY.

This training supplement has presented you with a general review of hydraulic principles, and it has shown how these principles are adapted to a modern hydraulic system such as the one in the F-102A. After reviewing basic hydraulic theory, you learned about the primary, secondary, and emergency hydraulic systems in the F-102A and how these three systems furnish fluid pressure for the five operating subsystems. This information provided you with a complete descriptive and operational analysis of the entire F-102A hydraulic power system. Chapter III described how the primary and secondary systems assist the flight control system in performing its duties. The last chapter has explained the hydraulic operation of the landing gear, nose wheel steering, speed brakes, and the emergency a-c generator systems.

In addition to the operational and maintenance information which you retain from this manual, you should also remember several other important facts. These facts pertain to the use of this manual. In many cases, the descriptions of the various hydraulic systems contained specific values and pressures. These values and pressures were used for explanatory purposes only, and they should not necessarily be used on the flight line. The latest information of this nature can be found in your F-102A Maintenance Manual, T.O. 1F-102A-2-3. In some instances, references have been made to the fact that not all of the airplanes have the same types of components in their hydraulic systems. No attempt has been made to point out the particular airplanes which have certain components and/or those airplanes that have other types. This information, too, should be obtained from your F-102A maintenance manual, T.O. 1F-102A-2-3. You should keep in mind that this training supplement does not replace the maintenance manual—it is for your on-the-job training for this new airplane. From this point on you will learn the fine points of maintenance by actual experience.

ALPHABETICAL INDEX

*The Symbol * Indicates An Illustration*

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F-102A

MAINTENANCE TRAINING SUPPLEMENT

(EXPERIMENTAL)

To Be Used in Conjunction With

T.O. 1F-102A-2-4

POWER PLANT INSTALLATION

CONVAIR
F-102A

MAINTENANCE TRAINING SUPPLEMENT

POWER PLANT INSTALLATION

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE

AIR FORCE—AFPS—SAN ANTONIO, TEXAS—5-57-3600

1 JULY 1956

Foreword

The F-102A Training Supplements have been prepared by Convair, A Division of General Dynamics Corporation, to provide you, the system mechanic or maintenance technician, with basic information concerning the F-102A airplane and its operating systems. There are ten of these supplements, each covering an airplane operating system or major component. The text is direct and informal and is supplemented with illustrations and diagrams to aid you in understanding how the systems operate. The following is a list of the Training Supplements on the F-102A airplane:

Title

Flight Control System
Hydraulic System
High-Pressure Pneumatic System
Low-Pressure Pneumatic System
Airplane General
Airframe Fuel System
Power Plant Installation
Airframe Armament System
Electrical System
Instruments

Each supplement describes an operating system or major component, how and why it operates, and maintenance problems that you may encounter. The entire major system is further simplified by describing each of its subsystems separately. Thus, when you understand how all of the subsystems operate, you will be familiar with the functions of the major system. This knowledge of the airplane systems will enable you to use other operational, service, and maintenance instructions.

Since the purpose of these publications is to familiarize and acquaint you with the airplane systems and components, specific values, measurements, and tolerances are not given. Any values that appear in these supplements are approximate only and are used to emphasize more clearly a certain operation or condition. You must refer to your 1F-102A-2-4. Technical Order and other pertinent handbooks for specific values when adjusting or checking a system and its components. These Technical Orders are revised periodically to include the latest maintenance procedures and data on the equipment installed in the F-102A.

The F-102A Maintenance Technical Orders consist of the following handbooks:

T.O. 1F-102A-2-1	**General Airplane**
T.O. 1F-102A-2-2	**Ground Handling, Servicing, and Airframe Group Maintenance**
T.O. 1F-102A-2-3	**Hydraulic and Pneumatic Power Systems**
T.O. 1F-102A-2-4	**Power Plant**
T.O. 1F-102A-2-5	**Fuel Supply System**
T.O. 1F-102A-2-6	**Air Conditioning, Pressurization, and Anti-Icing Systems**
T.O. 1F-102A-2-7	**Flight Control Systems**
T.O. 1F-102A-2-8	**Landing Gear**
T.O. 1F-102A-2-9	**Instruments**
T.O. 1F-102A-2-10	**Electrical Systems**
T.O. 1F-102A-2-11	**Radio-Communication and Navigation Systems**
T.O. 1F-102A-2-12	**Armament and Armament Electronics**
T.O. 1F-102A-2-13	**Wiring Data (F-102A)**
T.O. 1F-102A-2-13A	**Wiring Data (TF-102A)**

The Air Force is vitally interested in seeing that its equipment functions properly and is maintained as efficiently as possible. The Air Force, like a manufacturer of an automobile or commercial appliance, provides printed instructions for use with its equipment. The printed instructions you will use most frequently in maintaining the F-102A are the dash-2 Technical Orders listed above. They are available for your reference at the Maintenance Office on your base. You must refer to them frequently so that you will always have the latest maintenance information. The Training Supplements will help you to use these Technical Orders by answering many of the "hows" and "whys" that may puzzle you from time to time.

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Chapter I

J57 POWER PLANT

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Many of us, watching a present day high speed jet aircraft streak across the sky, find it hard to appreciate the tremendous strides taken in the past 50-odd years of development. Improvements in military aircraft ability to fly faster and higher and carry more armament have demanded improvements in powerplant engineering and fuel/oil chemistry to the point that the turbo-jet engine has become a precision instrument. It definitely was not always this way.

Inspection and maintenance procedures have been radically changed and have become more and more complicated. Therefore, the knowledge of the mechanic must be expanded. To properly inspect and maintain the high speed aircraft of today with all their advanced systems, YOU HAVE TO KNOW WHAT YOU ARE DOING. When you view the F-102A airplane, it is certain to impress you as an intricate and highly advanced mechanism. The airplane loses much of its complexity, though, when you break it down into parts and study the operation of each part.

The purpose of this training supplement is to acquaint you with the J57 engine as installed in the F-102A airplane. Much time has been spent in anticipating your questions about the engine and its related systems. Every attempt has been made to make this an interesting training supplement which can be used, not only for familiarization by the student mechanic, but also as a reference guide for the top mechanic in your squadron. The amount of information you digest from this supplement will determine your effectiveness as a member of the ground support team of the F-102A airplane.

DEVELOPMENT OF THE GAS TURBINE.

Many of us may think of the theory of the gas turbine as being relatively new. Actually, the gas turbine prin-

Figure 1-1. Hero's Aeolipile

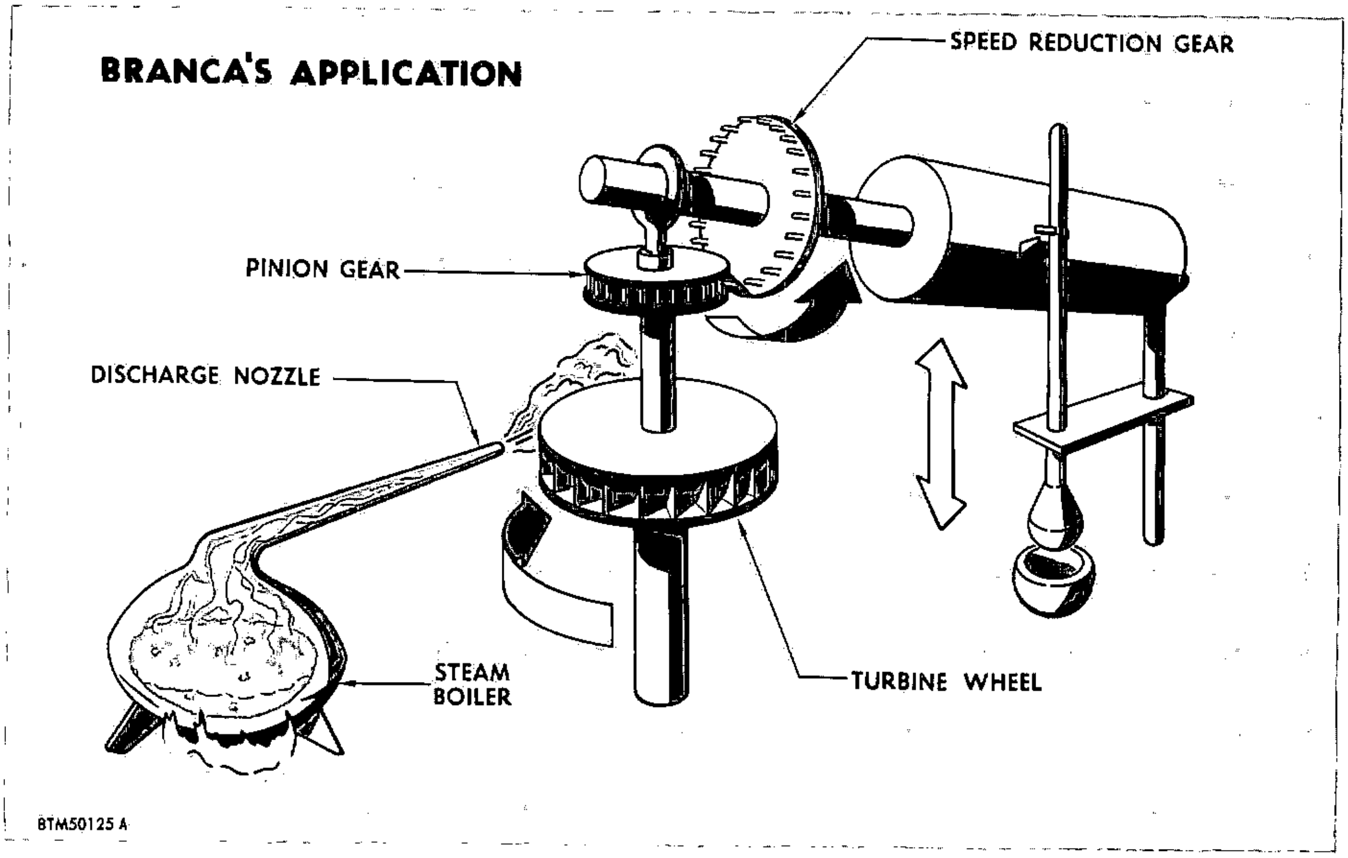


Figure 1-2. Branca's Turbine Application

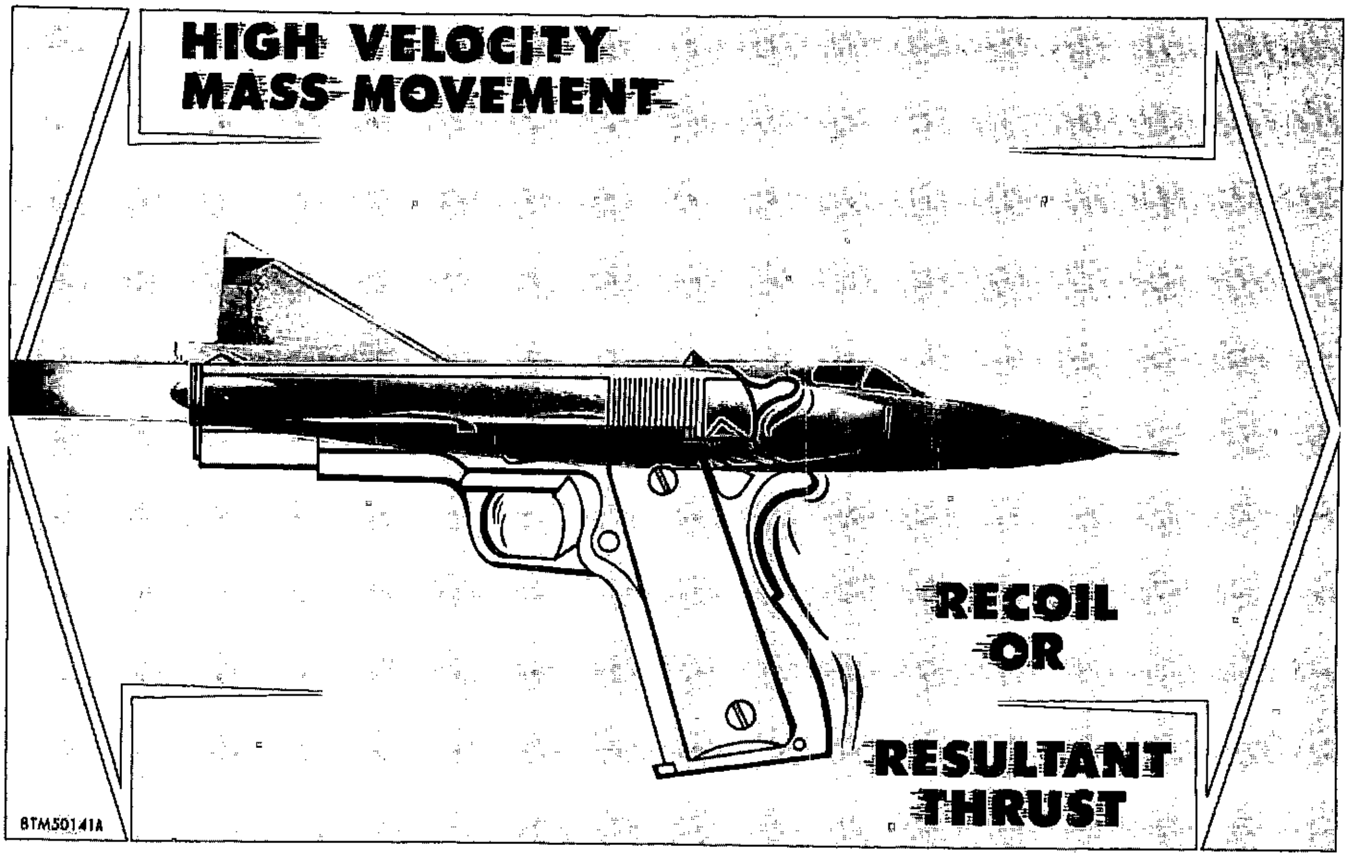


Figure 1-3. Recoil and Thrust Reaction Forces

ciple was used as far back as about 130 B.C. At that time, Hero designed and constructed what was known as "Hero's Aeolipile". You will notice in figure 1-1 that the Aeolipile was nothing more than a revolving boiler. Two nozzles discharged steam at right angles to the axis of rotation, thus imparting a spinning action to the boiler. The Aeolipile was nothing but a toy; however, it laid the ground work for later experiments which accomplished useful work.

Possibly the earliest date at which a machine accomplished useful work was in 1629, when an Italian engineer, Giovanni Branca, designed and built the first actual turbine. This machine, called "Branca's Turbine Application," is illustrated in figure 1-2. As you can see, it was an open-air combustion chamber with a spherical boiler and a discharge nozzle directed against a turbine wheel. Note also how the turbine wheel drives the pinions which in turn, through a speed reducer, drive a stamp mill. This was the first reduction gear. It is interesting to note that Hero's Aeolipile, and other machines since, were of the pure reaction type; while in Branca's Turbine Application we have the first concept of the impulse wheel.

The reaction of a mass when its velocity is changed may be considered jet propulsion. Newton's Laws of Motion state: (1) "every action produces a reaction which is equal in force and opposite in direction," (2) "the force exerted equals the mass times the rate of change in velocity." As an example of Newton's Laws of Motion, let us consider a pistol when fired, as illustrated in figure 1-3. To some, it may seem that a pistol recoils because of the expelled gases pushing on the air. This is not true—a pistol recoils in accordance with Newton's Laws of Motion. Whether a pistol is fired through water, ordinary atmosphere, or a vacuum, the recoil in all cases is exactly the same. This, of course, disproves the commonly accepted idea that the expelled gases push on the surrounding air. The high velocity jet blast of an airplane engine may be considered a continuous recoil, imparting an opposite reaction (thrust) to the airplane.

The automobile is commonly accepted as the first horseless carriage, but in 1680 Isaac Newton designed a model steam carriage of the pure reaction type. As you can see in figure 1-4, it housed a jet reaction unit on a four-wheel chassis. In this model, combustion took place in a closed combustion chamber—steam being generated in a spherical boiler and ejected rearward through a discharge nozzle. This action propelled the carriage forward. The driver controlled a steam cock in the discharge nozzle to control the speed. This device aptly illustrated Newton's Law of Motion, "for every action there is a reaction which is equal and opposite in direction," as mentioned above.

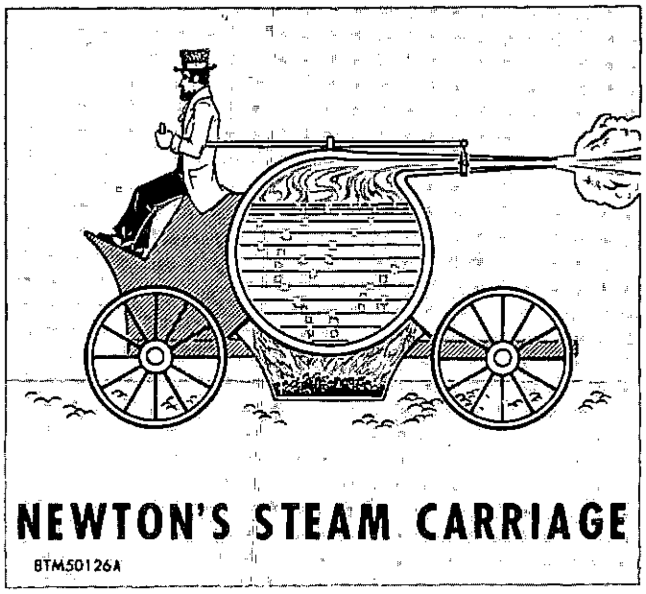


Figure 1-4. Newton's Carriage

RECENT DEVELOPMENTS IN JET ENGINES.

The patent which outlined the basic form for all modern gas turbines was applied for on January 16, 1930, in Great Britain, by Air Commodore Frank Whittle. Studying Whittle's patent sketch (see figure 1-5), you will notice its similarity to present day turbojet engines. Commodore Whittle first conceived his idea in 1928 while in his fourth term as a flight cadet at the R.A.F. College, Cranwell. However, it was not until May 14, 1941, that his dream became a reality and a turbine engine was installed in an airframe and flight tested.

While Whittle was busy conducting tests on his engine, the Germans brought out several interesting designs, among which was the Jumo-004. This engine was interesting in that it had an eight stage axial-flow compressor, six individual can-type combustion chambers, a single-stage turbine wheel, and a tail cone with an adjustable bullet. The compressor construction was interesting because of its disc design and method of assembly. The burners were also interesting because of the method of flame propagation.

During World War II, there was a constant interchange of ideas between England and the United States to hasten the development and production of jet engines. Under this agreement a Whittle engine was brought to this country for study. The General Electric Company was awarded a military contract to develop an engine for flight-test purposes. Bell Aircraft Company was selected to build the airframe. The Westinghouse Electric Company and Allison Engine

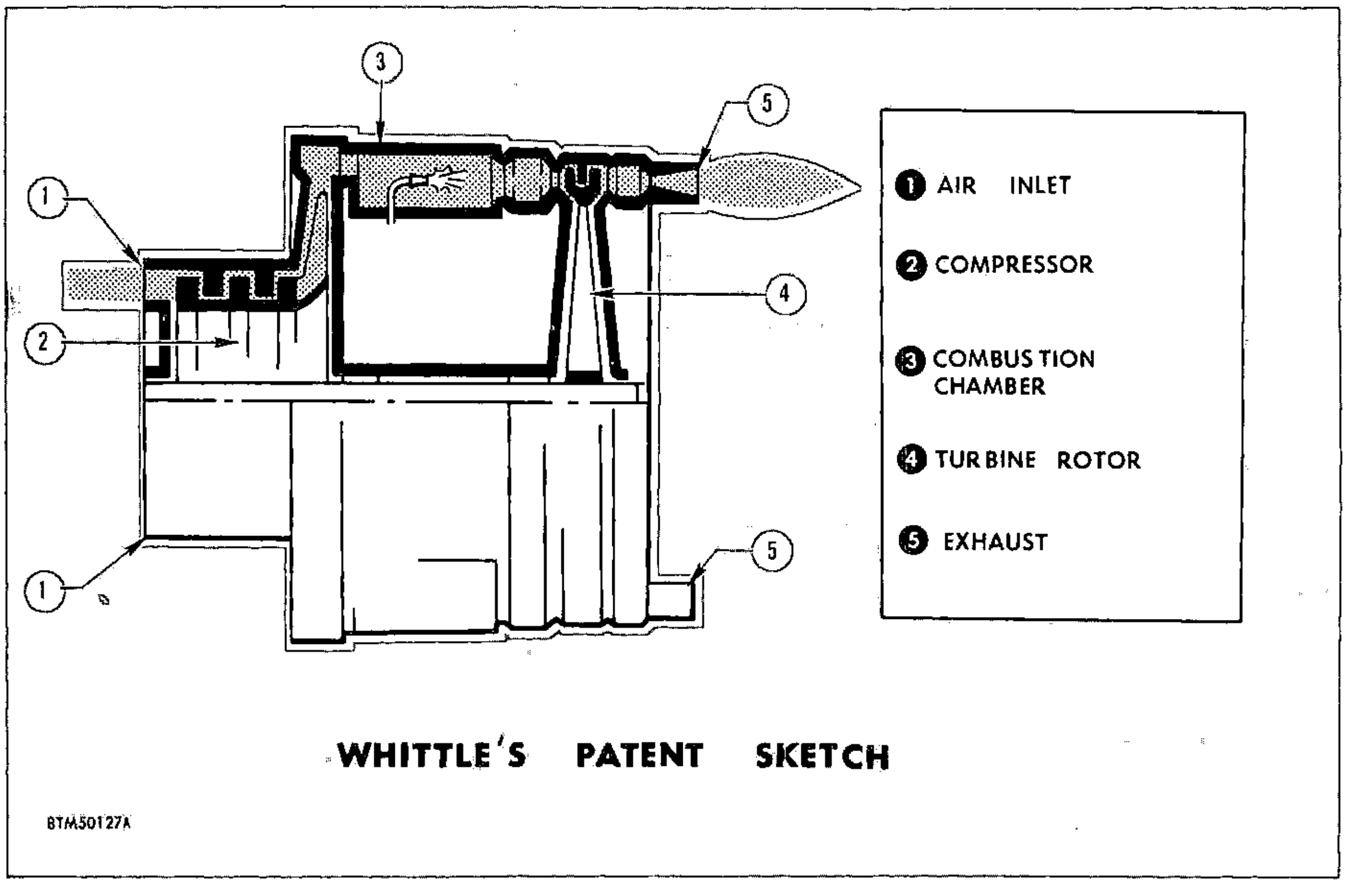


Figure 1-5. Whittle's Engine

Company were both awarded military contracts for research and development in the turbojet engine field. At the same time, the Rolls-Royce Company of England was given the original Whittle Engine to develop. After a series of modifications and improvements, the Rolls-Royce "Nene" engine was produced. This engine was tested in the United States and the rights to produce it were purchased by the Pratt & Whitney Aircraft Engine Company. Many improvements and conformances to American standards have subsequently been made on the engine. Many organizations in the United States are currently working on numerous experimental engines. In view of this and the vast amount of engineering hours devoted to research and development, it is safe to conclude that the jet engine is here to stay.

TYPES OF JET ENGINES.

At this point, it becomes obvious that the term "jet engine" in itself means very little. Many different types of jet engines are being developed and produced. As a result of all this research and development we now have aviation gas turbines, athodyds, and rockets —all of which will be explained in the following paragraphs.

AVIATION GAS TURBINES.

Turbojet engines fall into two categories. They are either of the axial-flow or the centrifugal-flow type. The turbo-prop engine also falls into either of these categories; however, it is usually an axial-flow design incorporating a propeller. From figure 1-6, you can see the primary design differences between these types. Each has its advantages and disadvantages. For instance, the axial-flow design is more efficient because of its high pressure rise; but its manufacturing costs are greater, it is less stable, and its over-all length is greater. The centrifugal-flow design is more stable and is cheaper to manufacture; but it has a limited pressure rise. Also, its diameter is greater, and, as a result, the airframe can not be as streamlined as with the other design.

ATHODYDS.

Athodyds fall into two separate and distinct types—the pulse-jet and the ram-jet. The pulse-jet, like other jet engines, creates its propulsive force by accelerating the airstream passing through the engine. The outstanding characteristic of the pulse-jet is that the air is admitted into the combustion chamber through shutters or flapper valves in gusts, thereby causing the jet to be intermittent (pulse). The ability of the ram-jet

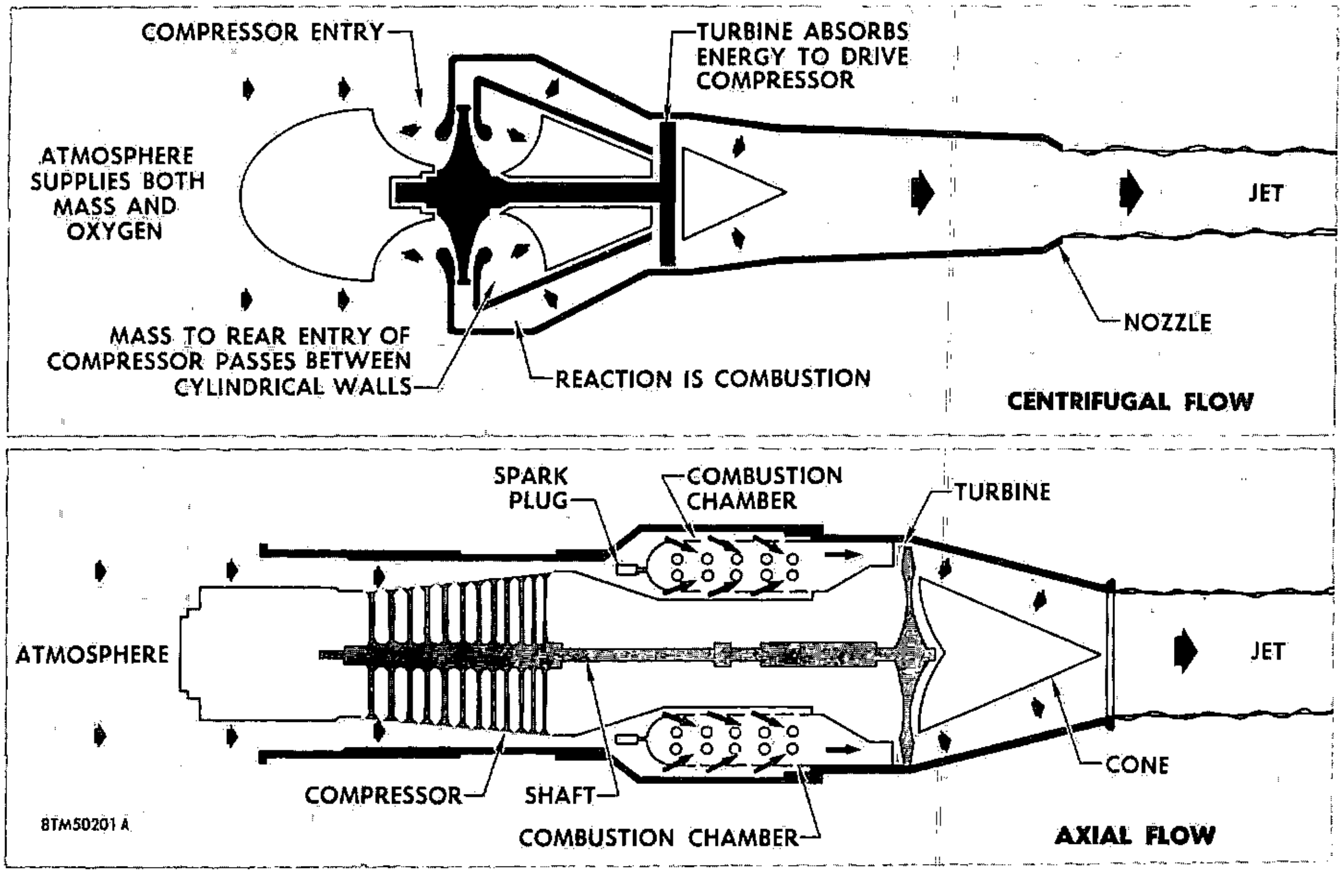


Figure 1-6. Aviation Gas Turbines

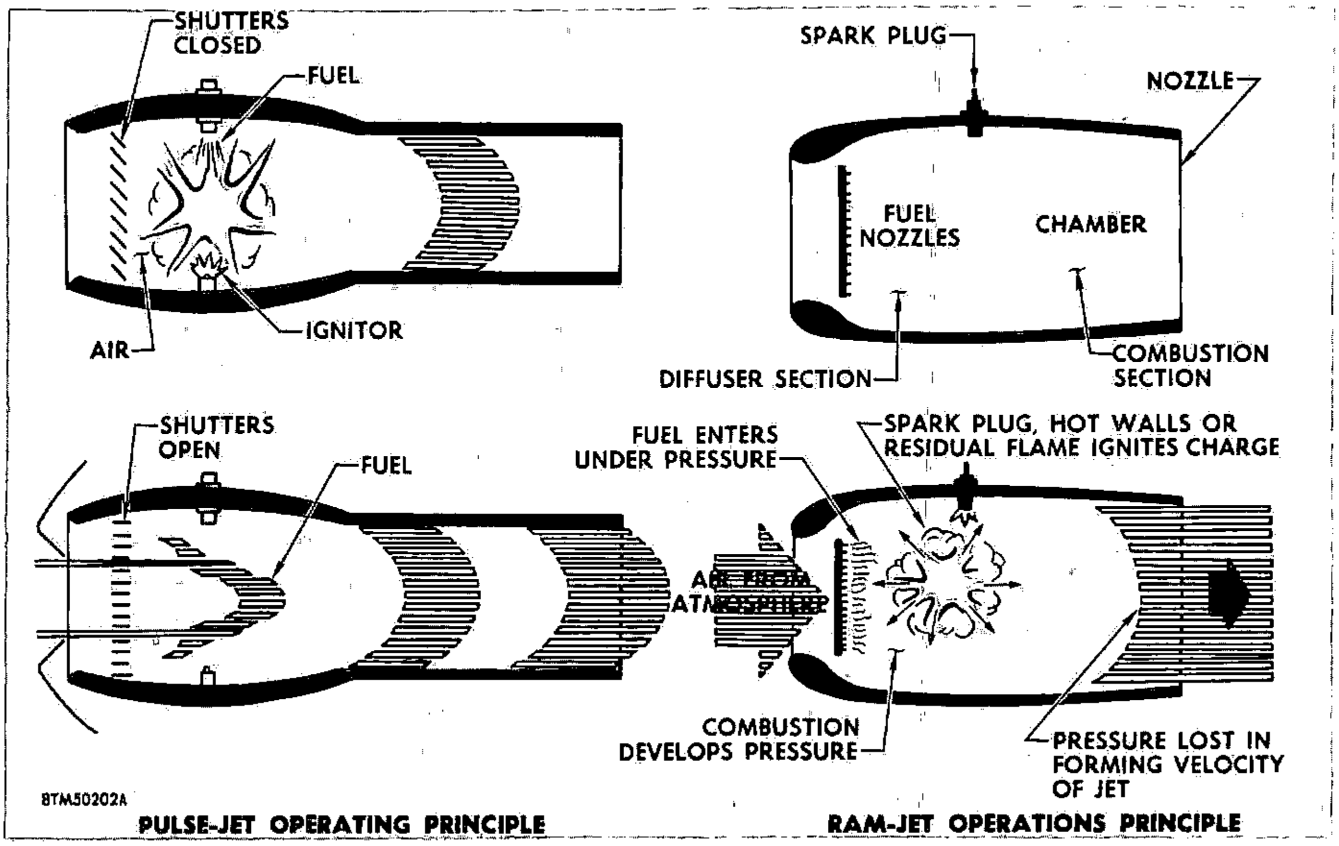


Figure 1-7. Athodyds

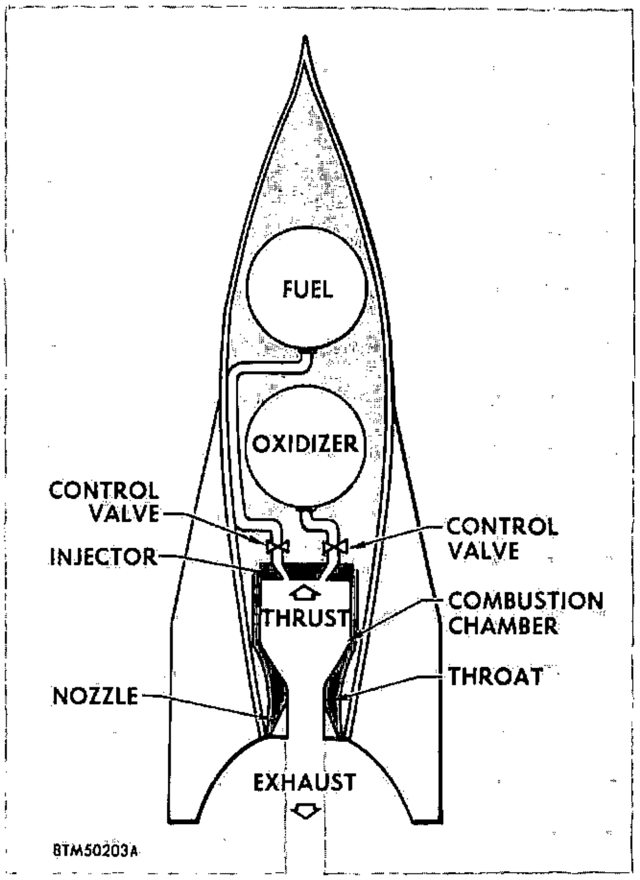


Figure 1-8. Rockets

to produce thrust depends on the velocity change in the same manner as any other reaction engine. It can be defined as a compressorless engine which depends upon the conversion of energy of the incoming air for its compression pressure. Figure 1-7 shows a cross-sectional schematic of both types of athodyds.

ROCKETS.

The rocket develops thrust by accelerating large quantities of gases generated by the chemical reaction of self-contained propellants. The chemical reaction is produced by mixing two propellants together inside the rocket. Figure 1-8 shows the principal elements of a bi-propellant liquid-fueled rocket. Notice, it has nothing more than a combustion chamber and a converging exhaust nozzle. The propellant gases are produced in the combustion chamber at pressures governed by the chemical characteristics of the propellants, their rate of consumption, and the cross-sectional area of the nozzle throat. The gases are exhausted into the atmosphere through the nozzle at supersonic speed. The nozzle converts the pressure of the propellant gases into active energy. The reaction to the discharged propellant gases is the thrust developed by the rocket.

The propellants employed in a rocket may be a solid, two liquids (fuel plus oxidizer), or materials containing an adequate supply of available oxygen in their chemical composition.

THE J57 ENGINE AND ITS COMPONENTS.

Whether you are an experienced or inexperienced mechanic, or whether you have had a wide variety of jet engine maintenance experience or have only recently graduated from Air Force Technical School, this engine should prove a challenge.

There are two engine models installed in the F-102A airplane. The earlier airplanes have a Pratt & Whitney J57-P-41 series engine which develops 14,800 pounds static thrust at sea level with afterburning. On later airplanes, a J57-P-23 series engine rated at 16,000 pounds static thrust at sea level with afterburner is installed. Figure 1-9 shows the main physical differences in the two engines.

Both engine models are of a unique design in that each incorporates a dual-spool compressor section (two separate compressors) working in conjunction with a three-stage turbine section (three turbine wheels). The compressors have an inter-compressor bleed system for matching compressor outputs throughout the engine operating range. Burner "can" construction is interesting because of the method of flame propagation and control of the flame within the "can." These two factors will be discussed later in Chapter II when you will learn about the engine fuel system. On earlier engines, the afterburner has a two-position iris-type (opens like a flower) discharge nozzle with 24 actuating cylinders. The later engine has a two-position flap-type (eight segments) afterburner discharge nozzle with only eight actuating cylinders. In figure 1-10 the two types of afterburner discharge nozzles are shown.

The engine is installed in the aft section of the fuselage. Because of the engine location, a split-type (dual) air intake duct, converging into a single duct just forward of the inlet guide vanes, is necessary. Removing and installing the engine is accomplished from the aft end of the airplane. This practice is necessary because of the "delta wing" design which prevents the more conventional method of "breaking" the fuselage and handling the engine from the fuselage "break point." The lubrication system is an engine-contained system with the exception of the air/oil cooler which is supplied by the airframe manufacturer. Fuel is supplied to the engine from six wing fuel tanks with a total usable fuel capacity of approximately 1070 US. gallons. There are also provisions for the installation of jettisonable external wing fuel tanks. As you can

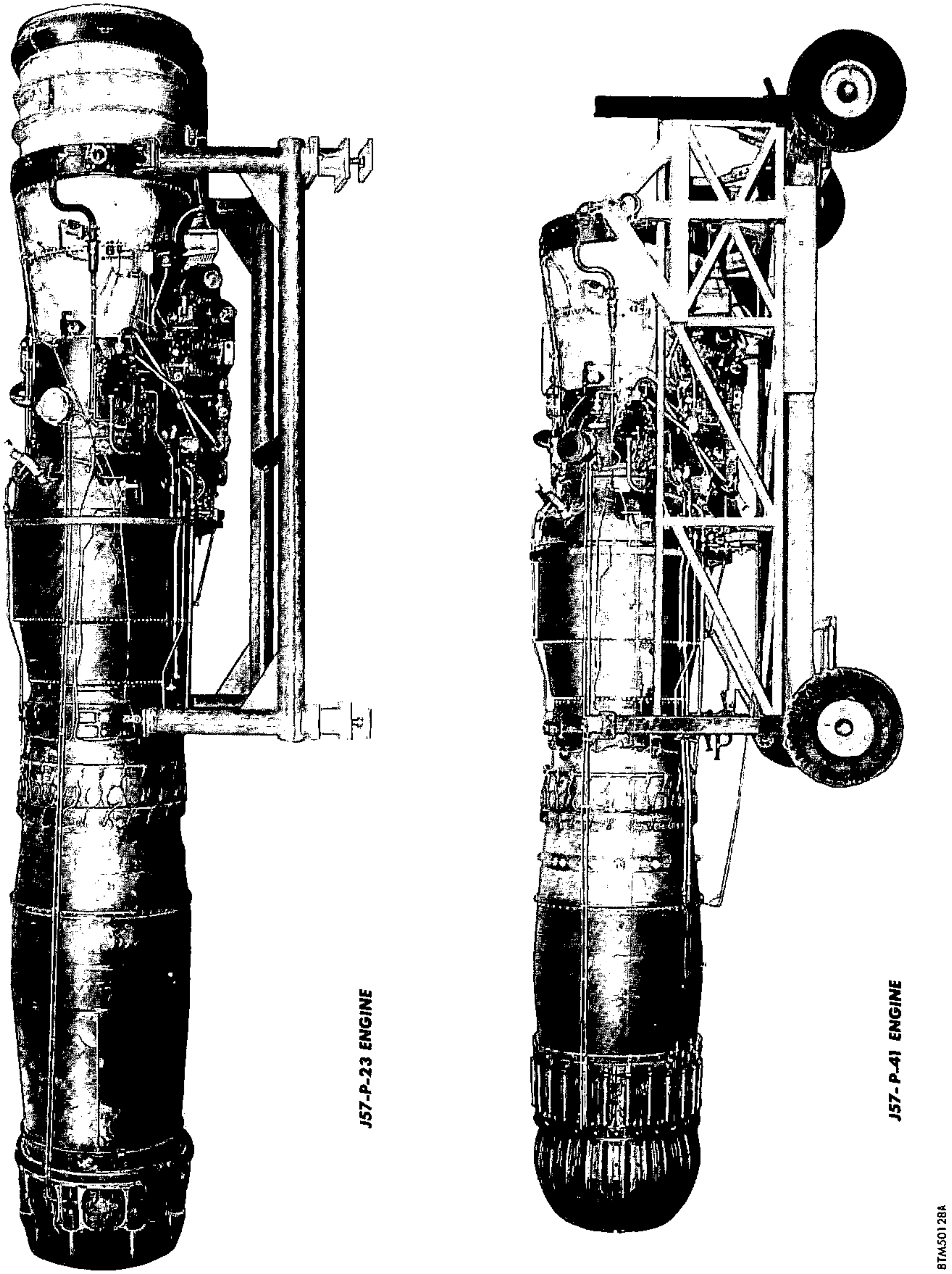
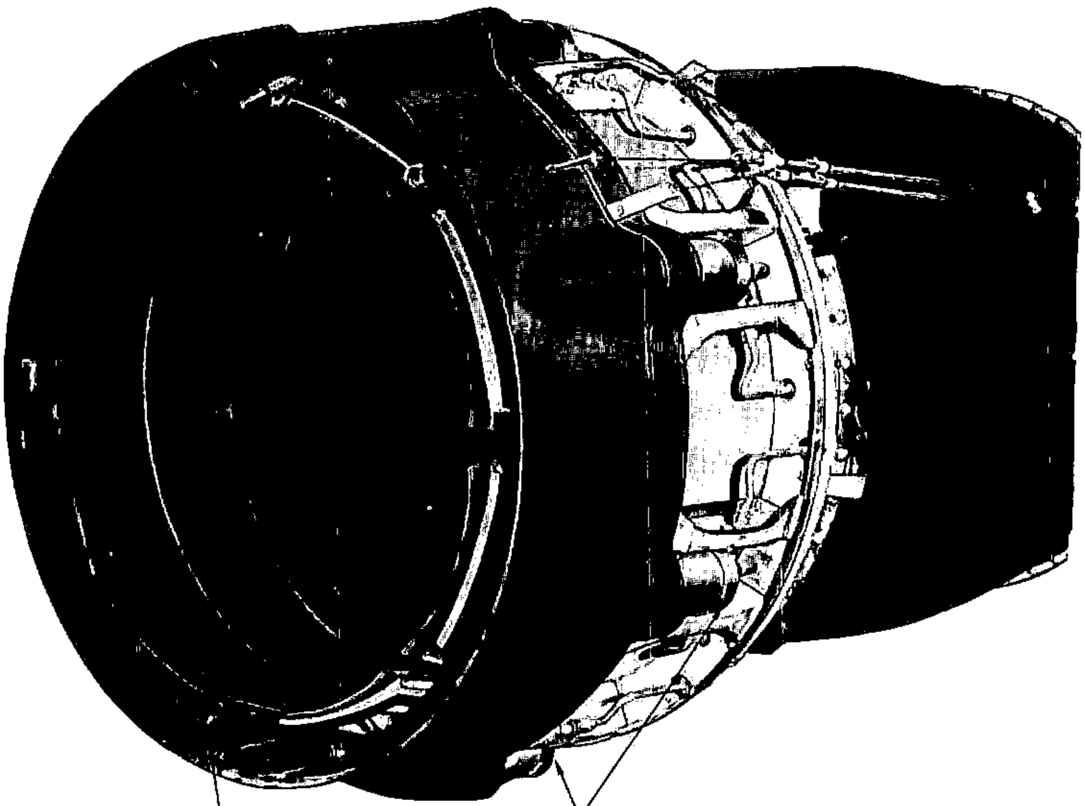
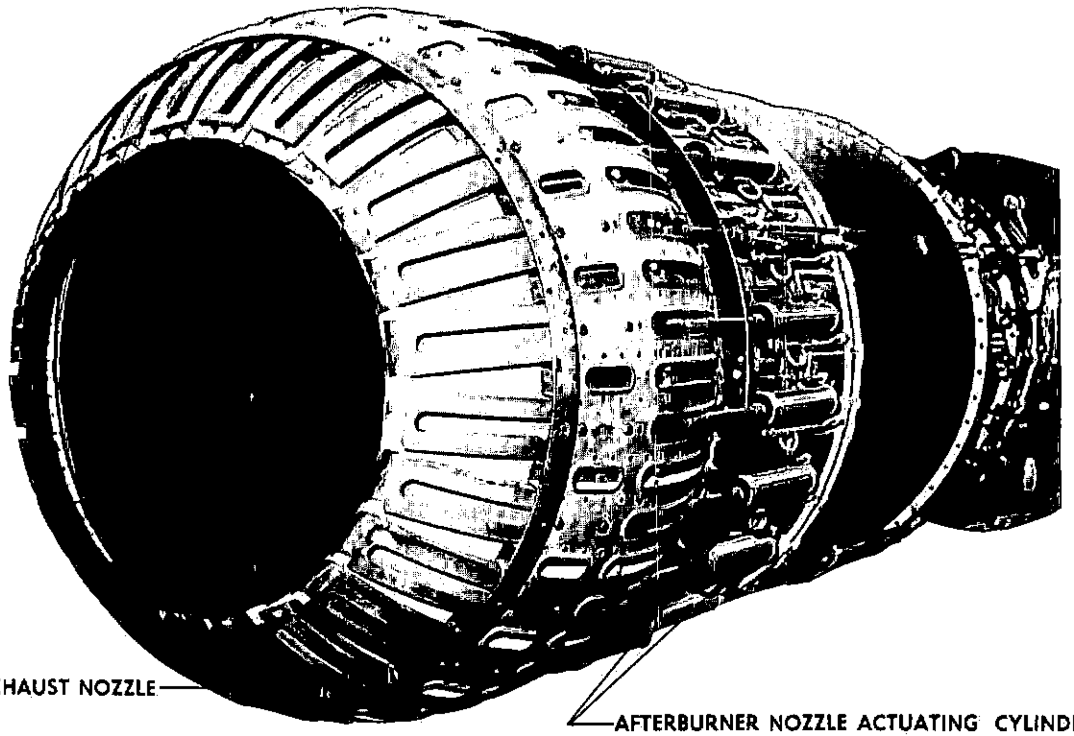


Figure 1-9. J57 Engine



FLAT TYPE EXHAUST NOZZLE

AFTERBURNER NOZZLE ACTUATING CYLINDERS



IRIS TYPE EXHAUST NOZZLE

AFTERBURNER NOZZLE ACTUATING CYLINDERS

8TM50313A

Figure 1-10. Afterburners

see, we have come a long way since the 16-horsepower machine the Wright Brothers flew at Kittyhawk in 1903.

COMPRESSOR SECTION.

From the sectional view of the illustration in figure 1-11, you can see that the forward N_1 (low pressure) compressor is made up of nine rotor stages, eight disk spacers, and a front and rear hub. The stator, or fixed, blades, are located radially between the rotor blades and are attached to the compressor case. The aft N_2 (high pressure) compressor is made up of seven rotor stages and six fixed blade stages. All rotating blades are relieved or have "thinned" tips to allow for closer tip tolerances. They are fitted to their rotor disks by contoured slots and are secured with clips. The blades are not rigidly attached, rather, each blade actually has a small amount of "play" or freedom of movement.

COMBUSTION SECTION (BURNER SECTION).

Compressor sixteenth-stage air pressure passes through the diffuser section that directs the airflow into the combustion area. The fuel manifold and nozzles are mounted on the burner cans and discharge fuel into the cans. The burner section housing is attached to the diffuser rear flange and contains the eight horizontal, radially-mounted and interconnected combustion chambers. As you can see in figure 1-12, the combustion chamber consists of perforated outer and inner liners. It has six openings on the forward face that align with the fuel nozzles. The cans can be removed individually and are basically interchangeable. The odd-numbered cans have female interconnect flame tubes, and the even-numbered ones have male interconnects. Because of this the odd-numbered cans are only interchangeable with other odd-numbered cans and even-numbered cans with other even-numbered cans. No. 4 and No. 5 burner cans have openings for igniter plugs (spark igniter guide). There is no pressure rise in the burner "can" despite the fact that combustion takes place within the can, producing both a temperature and a velocity increase. Actually, pressures at the turbine inlet are lower than those in the diffuser section. The turbine inlet collects the combustion section gases and adapts the gas flow to the turbine. The expansion of gases takes place across the turbines, furnishing the power required to drive the turbines. It is also interesting to note that the combustion is contained within the can; there is no flame passing through the turbine stages except during afterburner ignition.

TURBINE SECTION.

As noted previously, the turbine section is made up of three stages. As you can see from figure 1-11, the first stage turbine drives the aft or high-pressure compressor rotor. Note also how the final two interconnected stages drive the forward or low-pressure compressor rotor. Fixed vanes are incorporated in the turbine inlet, with stator blades located between the turbine stages. The shrouded turbine blade tips provide a good seal and permit thinner blade sections. The rotating blades are attached to the turbine disk by "fir tree" serrations and are secured by rivets lying fore and aft at the top of each "fir tree." The shrouded blade tips are scarf cut (cut on an angle) and interlock with adjacent tip shrouds; however, a small gap exists between adjacent tip shroud edges on the first and second turbine stages, permitting a small amount of "play." The third stage turbine blades are attached to the disk in the same manner, but are without the shroud edge gap in the static condition. During engine operation, however, the forces on the blades result in a small edge gap and blade freedom—similar to that found in stages 1 and 2. Seals, shaped like a knife-edge, are used to prevent the gases from entering the rotor base areas, and fixed turbine exit vanes straighten this air flow as it leaves the turbine. The turbine rear bearing support is secured by eight rods projecting radially through the turbine exhaust cast struts.

AFTERBURNER SECTION.

Normal engine thrust is augmented by afterburning such as needed for take-off, climb, or for any other flight condition requiring additional thrust. Afterburner operation usually increases normal engine thrust by approximately 50 per cent. This thrust augmentation results from the ignition of fuel introduced into the afterburner for this purpose. The oxygen that is necessary for combustion is furnished by the surplus air that is not required during the normal engine combustion process. Although afterburning is far from efficient, fuel-wise, it provides a relatively simple means of augmenting normal engine thrust.

The afterburner is merely a ram-jet type engine attached to the engine turbine section. It is composed of a diffuser section, burner section, and a variable nozzle. Looking at figure 1-13, you will see that fuel is introduced through 24 spraybars in the diffuser section. This fuel is ignited by the "hot streak" ignition method. In this "hot streak" type of ignition, the igniter valve introduces a small amount of fuel into No. 3 burner can. This momentary enrichment results in a flame—extending through the turbine blades—which ignites the fuel at the afterburner spraybars. Three circular flame holders, located in the diffuser section, retain the flame within the afterburner. A two-position pneumatically operated exhaust nozzle is automatically opened (during afterburning) and closed (for normal engine operation). During afterburner operation, the opening of the exhaust nozzle insures normal tailpipe temperatures and pressures.

The afterburner is a cantilever structure that features a double-wall construction in the aft portion for rigidity and cooling purposes. Afterburner fuel pressure

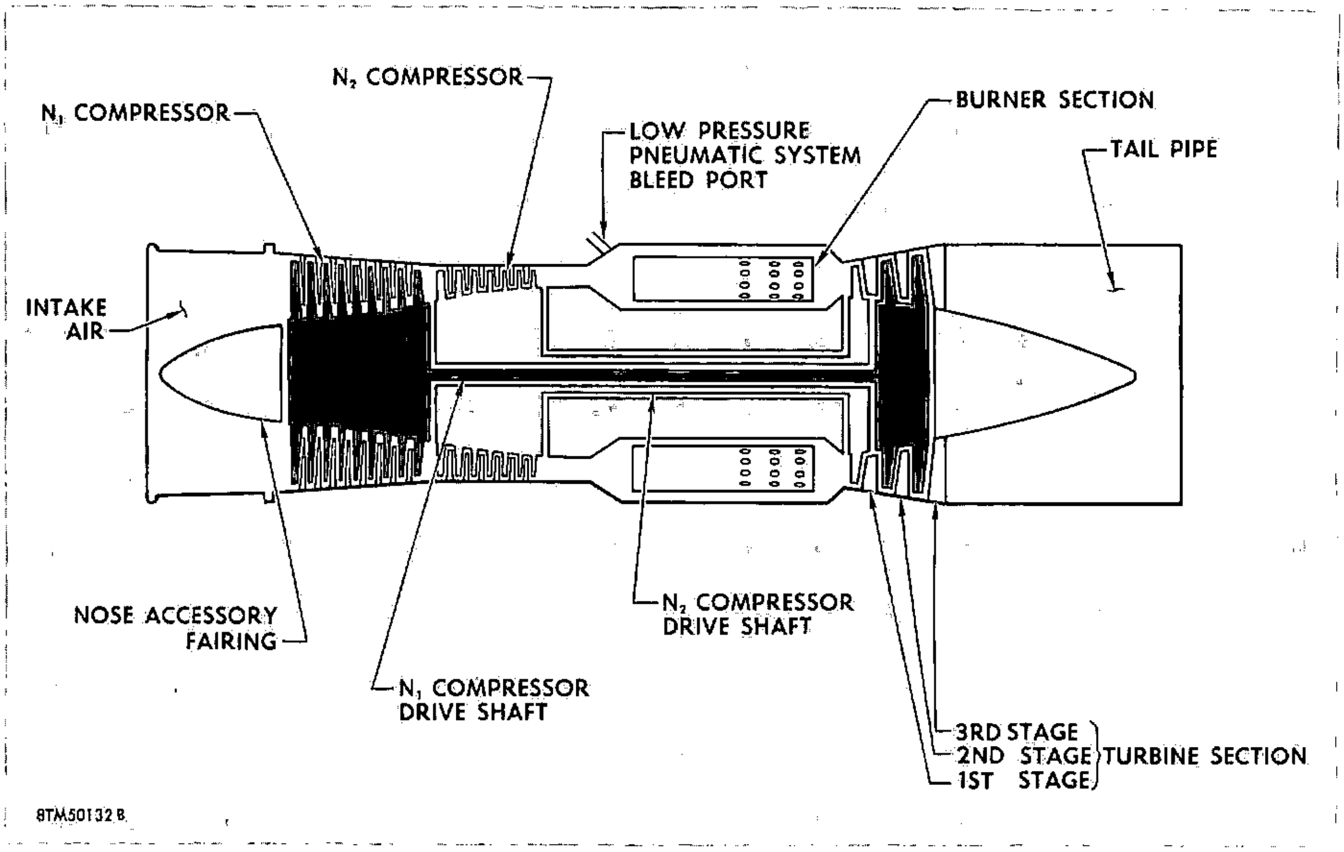


Figure 1-11. Compressor and Turbine Sections

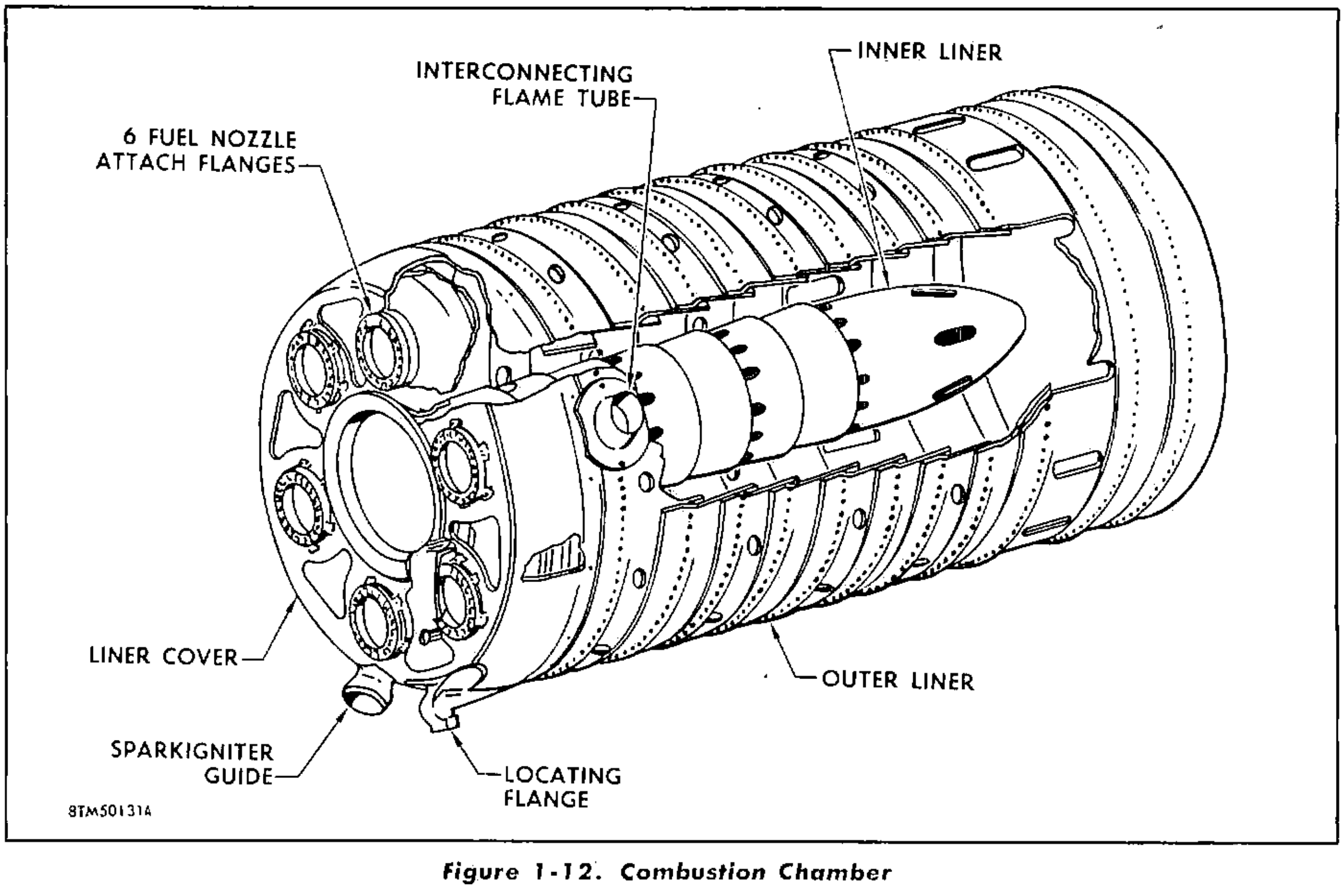


Figure 1-12. Combustion Chamber

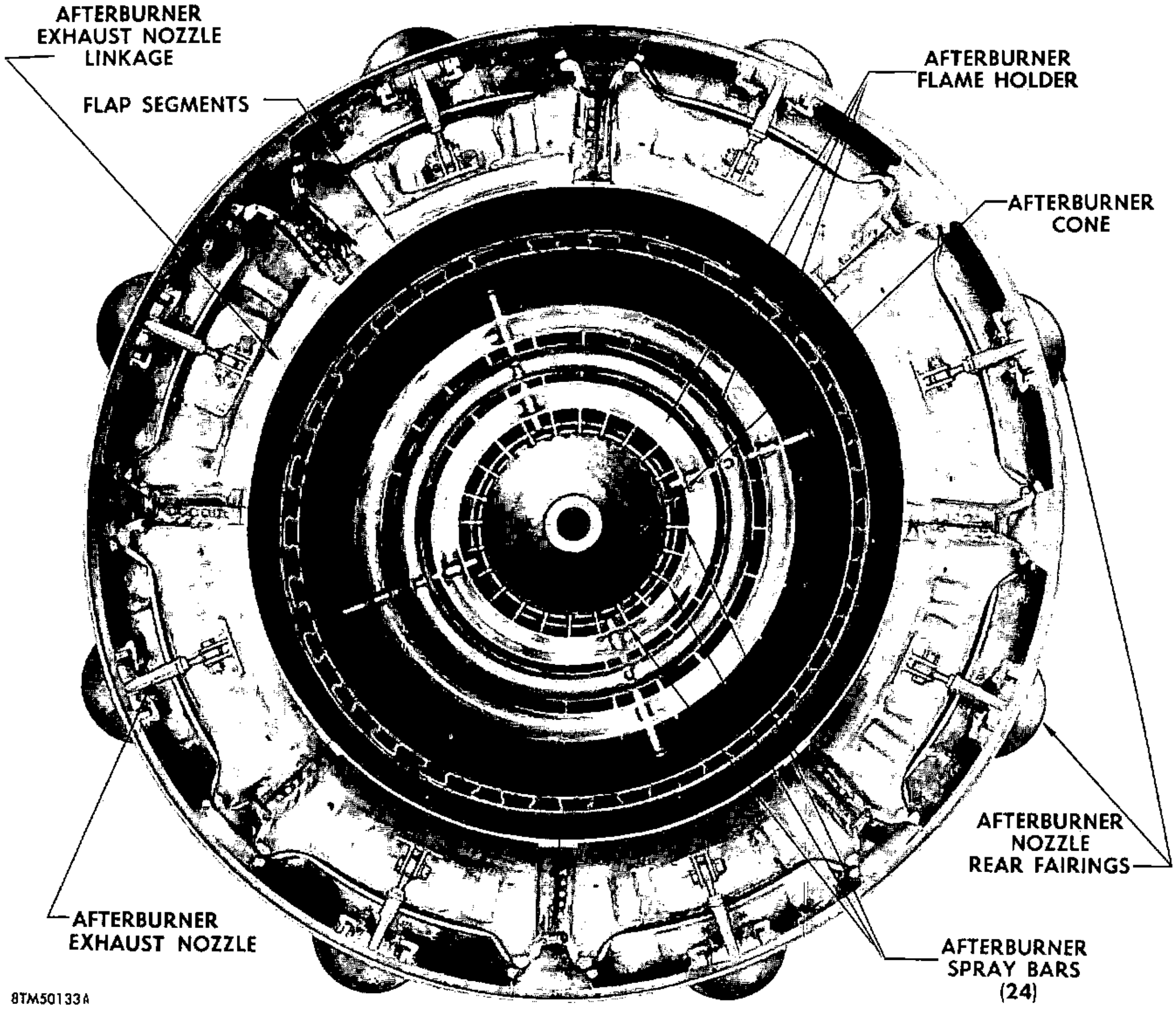


Figure 1-13. Afterburner Components

actuates the exhaust nozzle control valve, which ports sixteenth-stage air pressure to the OPEN side of the nozzle actuating cylinders. When afterburner fuel pressure falls off, spring pressure in the exhaust nozzle control valve ports sixteenth-stage air pressure to the CLOSE side of the nozzle actuating cylinders.

ACCESSORY SECTION.

The accessory section is located at the bottom of the engine in the "wasp waist" or smallest cross sectional area, as shown in figure 1-14. Power to drive this accessory section is taken from the high-pressure compressor rotor's rear hub—transmitted through matching gears on a canted shaft and forward through the horizontal shaft to the accessory drive case—at a ratio of 7 to 1. Driven units are the two hydraulic pumps (17) and (40), the pneumatic starter (16), the tachometer-generator (mounted on the forward face of the fuel pump/transfer valve (21), and the engine fuel control (35), located on the right and left aft face of the accessory drive housing, respectively). Other components located in this area are the fuel pressurizing and dump valve (28) and the two ignition transformers (26) and (33). The afterburner (A/B) igniter valve (2), A/B fuel regulator (23) and the exhaust nozzle control valve (27) are located on the right side of the engine in the "wasp waist" area. The Sundstrand constant-speed drive unit (39) is mounted between the starter and the mounting pad. Notice that some fuel and oil system components are located in this area too —these will be discussed later in this supplement.

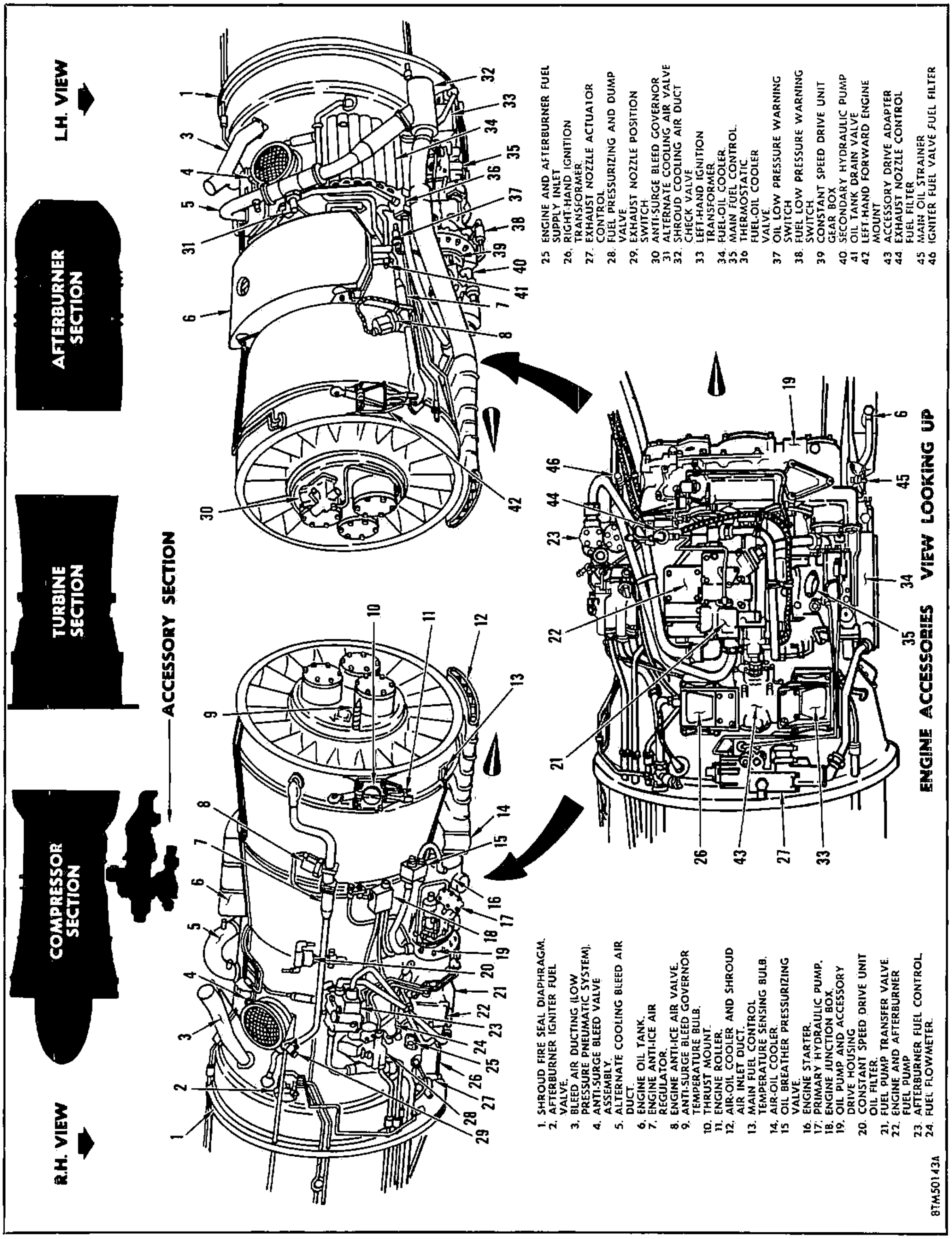


Figure 1-14. Accessory Section

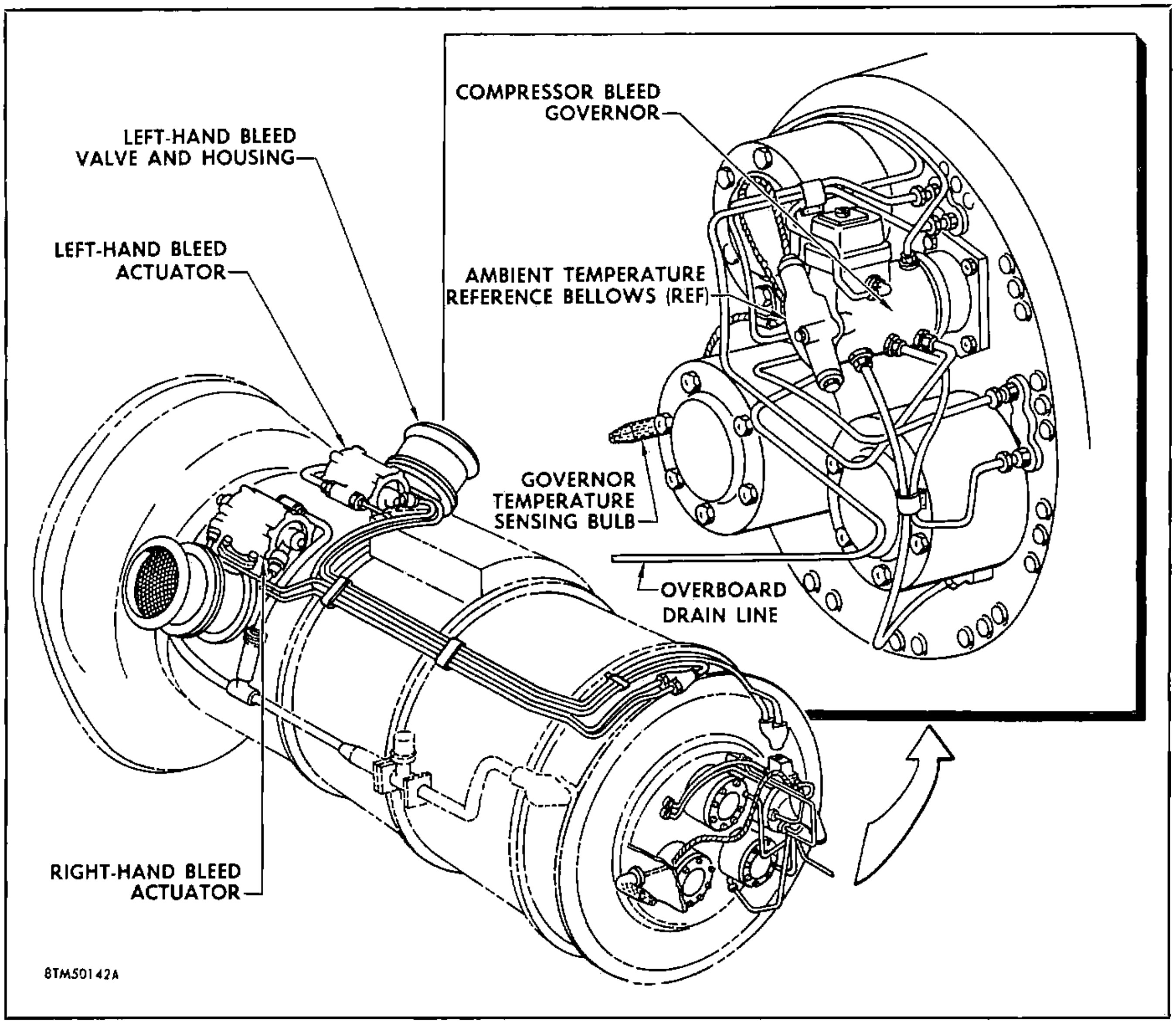


Figure 1-15. Compressor Bleed System

COMPRESSOR BLEED SYSTEM.

The compressor bleed system is used to maintain the best rpm and pressure relationship between the N_1 and N_2 compressors throughout the engine operating range. Its primary function is to minimize the possibility of compressor stall during acceleration and deceleration. The J57-P-41 engine incorporates two bleed valves, while the J57-P-23 engine features a single bleed valve. The function of both engine bleed systems is identical, but for clarity we shall study the dual bleed valve system. The two bleed valves and their actuators are located on either side of the top center line of the "wasp waist" section. The compressor bleed governor is mounted on the N_1 accessory section to control the bleed valve operation. At low power settings both bleed valves are open and a portion of N_1 compressor outlet is vented overboard. These two valves do not operate simultaneously; instead, during engine acceleration the right valve closes first and the left valve closes shortly thereafter. During engine deceleration the left valve opens first and the right valve opens later as engine speed is reduced. This operation is entirely automatic. The bleed valves are controlled by the compressor bleed governor, which senses N_1 rpm, along with temperature and pressure, at the inlet guide vanes. Varying low pressure compressor speeds, biased by inlet air temperature and pressure, position the pilot valves which direct sixteenth-stage (air pressure bled from the last stage of N_2 compressor) air pressure to either the OPEN or CLOSE side of the two bleed valve actuators.

ENGINE ANTI-ICING SYSTEM.

To prevent ice accumulation on the inlet guide vanes, heated sixteenth-stage air pressure is ducted through

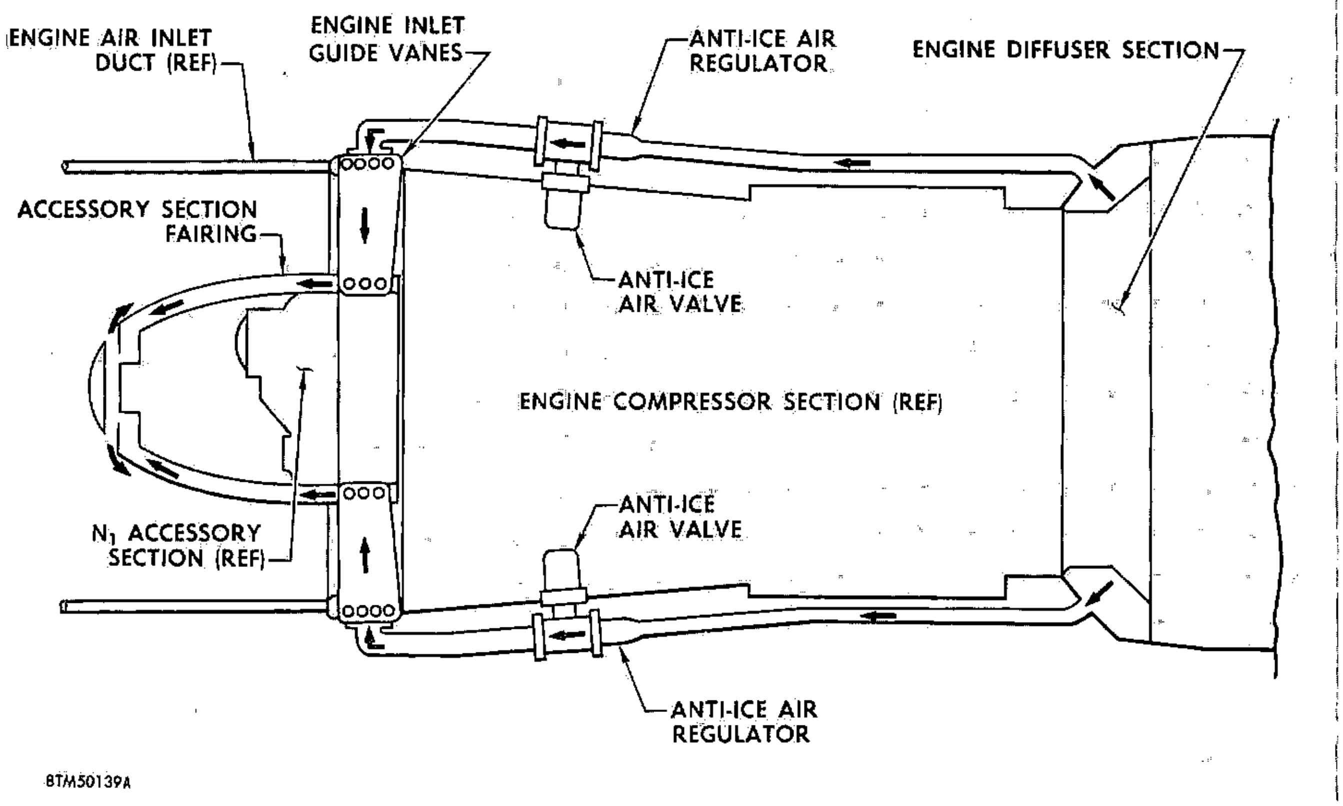


Figure 1-16. Anti-Icing System

hollow vanes. Figure 1-16 shows the two separate air lines leading forward from the diffuser section to the chamber surrounding the inlet guide vanes and through the vanes. Airflow is controlled directly by the two electrically-operated anti-ice air valves and indirectly by the two anti-ice air regulators.

The two air valves open when the anti-icing system is initiated by the engine anti-ice detector system. When the valves are open, air from the compressor diffuser section is routed to the inlet guide vanes. The anti-ice air regulators—located directly upstream from the shutoff valves—automatically control the air flow. Each regulator contains a bi-metallic coil spring that moves its valve toward the CLOSED position as the air temperature increases. Accordingly, airflow to the inlet guide vanes proportionately reduces as the anti-icing air temperature rises.

Notice that anti-icing air flows from the inlet guide vane outer chamber—between the outer case of the inlet guide vane and the outer shroud — inward through the hollow vanes to the inner chamber—between the inlet guide vane inner shroud and the inlet guide vane air passage cone. You can see that air flows forward from the guide vane roots through the double wall accessory section fairing and exits aft of the nose cap.

ENGINE MAIN BEARINGS.

Seven main bearings support and maintain alignment of the compressors, drive shafts, and turbines. You can find the exact location of the seven main bearings in figure 1-17. Bearing No. 1—a roller bearing located in the center of the forward compressor guide vane housing—supports the low pressure rotor front hub. A matched pair of ball bearings—attached to the compressor intermediate bearing support housing—make up the No. 2 thrust bearing that supports the low-pressure rotor rear hub. The high-pressure rotor front hub is supported by No. 3 roller bearing whose inner race is secured to the N_1 compressor rear hub. The second pair of matched ball bearings make up the No. 4 thrust bearing that supports the aft compressor rear hub. The bearing housing is secured to the diffuser case center section, and the inner race bears on the high pressure compressor drive shaft. Bearing No. 4.5 is the intershaft bearing and is located about halfway between bearing No. 4 and No. 5. The inner race of this roller bearing is secured to the forward rotor drive shaft, and the outer race bears against the inner surface of the aft rotor drive shaft. The first stage turbine is supported by another roller bearing, No. 5, whose outer race is attached to the aft section of the turbine front bearing support structure. Bearing No. 6 supports the two rear turbine stages. The

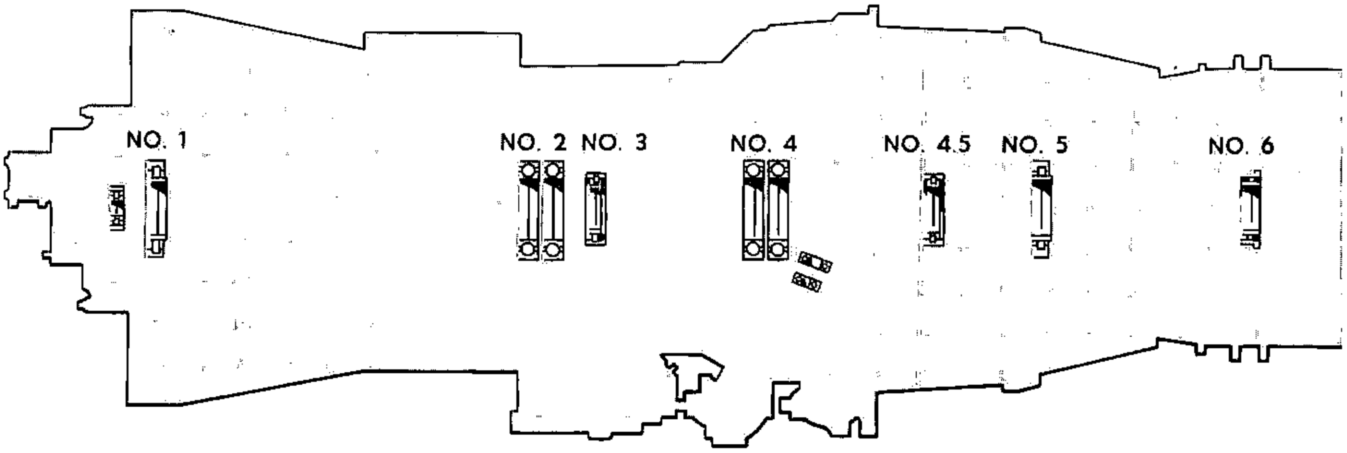


Figure 1-17. Engine Main Bearings

inner race of this roller bearing bears on the turbine rear hub; the outer race is supported by the turbine rear support. This support is secured by eight support rods projecting radially through the struts and ending in support journals on the outside of the case.

These bearings incorporate spring-loaded hard carbon seals to control oil leakage. Two of these seal rings are located aft of No. 1 bearing. No. 2 bearing has two seals located ahead of the bearing installation. Two smaller ring seals for No. 3 bearing are located aft of the bearing in the seal housing. Two carbon rings, aided by knife-edged seals, are located ahead of No. 4 bearing assembly. Oil escaping from No. 4.5 bearing is prevented from flowing aft between the two shafts by a pair of ring seals. Two more ring seals are located aft of No. 5 bearing to prevent oil leakage into the turbine area. Bearing No. 6 is sealed by two carbon ring seals located ahead of the bearing.

SUNDSTRAND CONSTANT-SPEED DRIVE UNIT.

The constant-speed drive unit is incorporated in the F-102A airplane engine installation for the purpose of driving the a-c and d-c generators, and also as a mounting pad for the engine starter. You can see that the drive consists of an engine mounted gear box, an interconnecting drive shaft, and an airframe mounted transmission and gear box assembly. The engine mounted gear box has a mounting pad for the engine starter. This gear box is directly connected to the engine compressor shaft by bevel gears.

The airframe mounted transmission and gear box assembly has two mounting pads, one for the a-c generator and one for the d-c generator. When the engine is operating, its rotation is transmitted from the engine mounted gear box, to the interconnecting drive shaft, to the airframe mounted transmission and gear box assembly, and finally drives both the a-c and the d-c generators.

As has been previously mentioned, the airframe-mounted transmission and gear box assembly has a mounting pad for the a-c and d-c generators. The a-c generator mounting pad provides rotation for the a-c generator in a counterclockwise direction (facing the pad) at a constant speed of 6000 ($\pm$60) rpm, when the input (engine rotation speed) to the engine mounted gear box varies between 3585 to 7020 rpm. The d-c generator rotational speed is not as critical as that of the a-c generator—as a result, it is driven at a direct ratio to engine rpm.

OPERATION OF THE J57 ENGINE.

Air entering the intake ducts of the airplane is directed into the forward N_1 low pressure compressor by the intake section inlet guide vanes. The action of the rotor blades against the stator blades increases the pressure of the incoming air. From the N_1 compressor, the air enters the aft N_2 high pressure compressor and is further compressed through seven stages of rotor and stator blades. At this time, there is a relatively high pressure increase accompanied by a slight velocity decrease. This velocity decrease is necessary to prevent the flame from blowing out in the combustion chambers.

As the air pressure leaves the N_2 compressor, it enters the burner section combustion chambers where it is mixed with injected fuel and ignited by two igniter plugs. All burner "cans" are interconnected, which permits simultaneous combustion in all eight "cans." Six spray nozzles in each "can" control the fuel spray

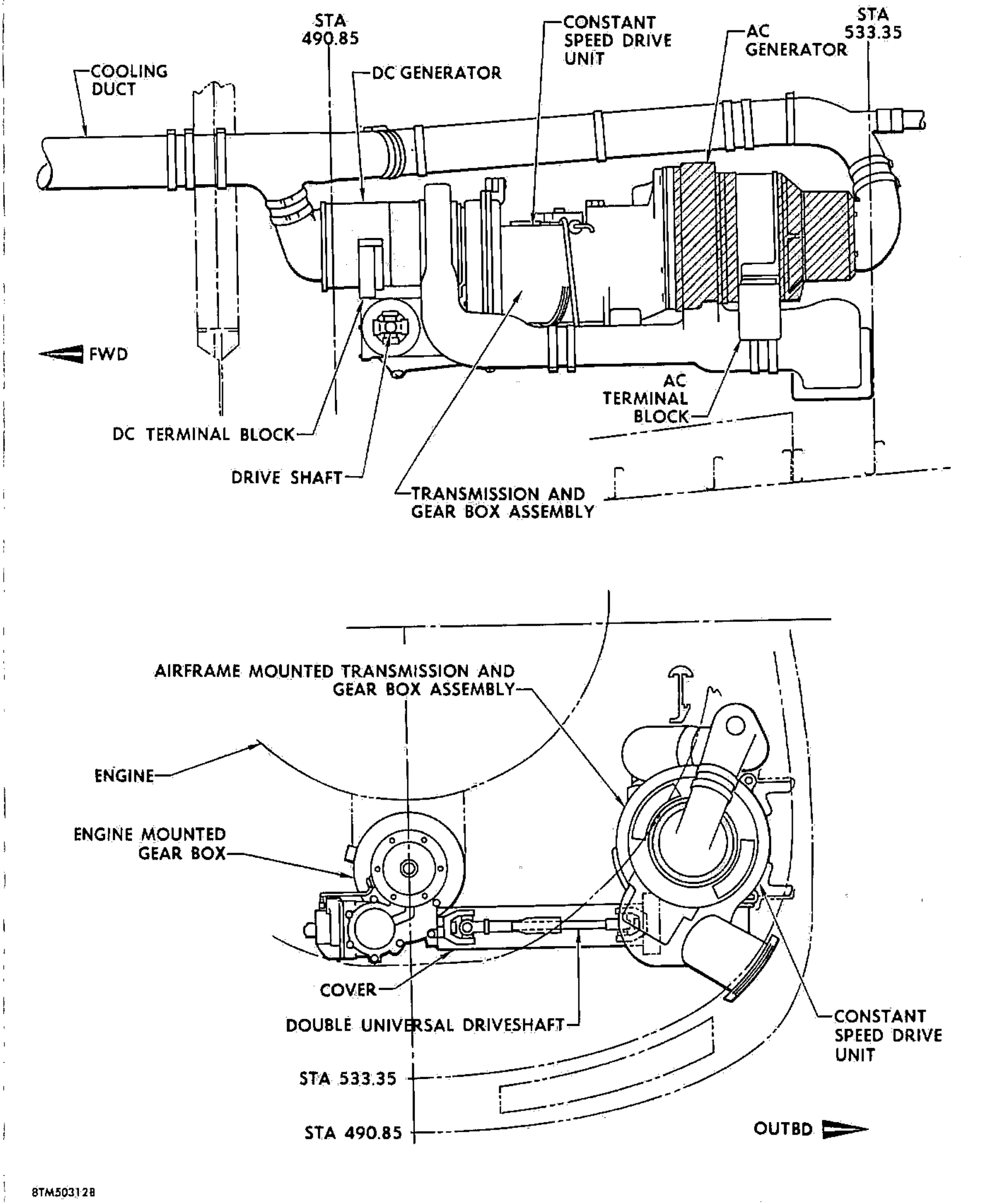


Figure 1-18. Sundstrand Constant-Drive Units

pattern, which in turn controls the flame pattern within the "cans." Rapidly expanding gases move aft through a three-stage turbine to drive the turbine wheels. The second and third stages of the turbine are directly connected to the N_1 compressor by a common shaft, and the first stage turbine is directly connected to the N_2 compressor shaft. Both drive shafts operate independently and rotate freely at their own speeds. The expanding gases force or drive the turbine wheels to turn at a high rpm. They in turn rotate the compressors at the same rmp. This causes air to enter the engine air intakes to provide the mass airflow to produce thrust. There is an intercompressor bleed system which will balance the two compressor outputs automatically should they become unbalanced. This increases engine efficiency and stability. Each of the earlier J57 engines has a dual intercompressor bleed valve, while later engines have single valves.

As these rapidly expanding gases speed rearward out of the tailpipe, they create a reaction in the opposite direction. This reaction produces the forward thrust necessary to move the airplane.

When the power lever in the cockpit is positioned into the afterburner detent, fuel is injected into the tailpipe by 24 spraybars and ignited by a flame from the No. 3 burner "can." There are three "V" shaped circular flame holders in the tailpipe which control the flame pattern within the afterburner. The two-position discharge nozzle opens, afterburner ignition takes place, and steady afterburning begins. As the throttle is retarded, fuel is shut off to the afterburner, the discharge nozzle is closed, and normal engine operation continues.

SUMMARY.

In this chapter you have learned about the J57 engine, its operation, its design features, and the physical location of components. By this time you should feel well acquainted with the engine and its closely related systems. If not, you should review this chapter before starting the next.

In the next chapter you will learn about the J57 engine systems—this will aid you in understanding the complete engine and its operation.

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Chapter II

THE POWER PLANT SYSTEMS

Contents *Page*

In a turbojet engine, such as the J57, the compressor and turbine wheels are mounted on a single shaft at opposite ends of the combustion area. When combustion takes place, the rapidly expanding gases flow through the turbine wheel vanes and cause the wheel to rotate. When the turbine wheel rotates, it drives the compressor, which packs air into the combustion areas. The increased combustion pressure imparts additional force to the expanding gases. This is the combustion cycle in a turbojet engine; and if there is some means of starting the rotation and combustion, the engine will produce a continuous flow of power.

In addition to the basic part, which accomplishes the combustion cycle, every turbojet engine must also have several associate systems. These associate systems function to assure that the engine will produce its continuous flow of power in the best operating conditions possible. In this chapter you will learn about the major associate systems in the J57 engine. These include the fuel, oil, starting and ignition, induction and cooling, and anti-icing systems.

THE J57 ENGINE AND AFTERBURNER FUEL SYSTEM.

The acceleration and deceleration control of a piston-type engine is usually accomplished by the throttle through the mechanical linkage between the throttle and the carburetor, and a similar arrangement for controlling the pitch of the propeller. With this type of installation, the rate of engine acceleration and deceleration is directly proportional to the *rate of movement* of these two controls in the cockpit. This condition, however, does not exist on jet engines.

During the very early stages of jet engine development, it was learned that rapid movement of the throttle in either direction usually resulted in a rich blow-out or a lean die-out of the flame in the engine combustion chambers. Blow-out merely means that if the engine operator moves the throttle forward too rapidly, he injects too much fuel into the combustion chambers and the flame is blown out. Conversely, if he retards the throttle too rapidly, the engine flame dies out from lack of fuel. Since most pilots were accustomed to the piston-engine type of fuel controls, blow-out and die-out in early jet operation were quite common. Power plant design engineers decided that the best solution was to relieve the pilot of all direct control over the engine acceleration and deceleration rate. This decision led to the development of the automatic fuel control unit, which is more commonly called the main fuel control.

The main fuel control allows the jet engine to accelerate and decelerate at a predetermined safe rate, thereby reducing the possibility of a flameout. This rate of acceleration and deceleration is kept constant by the main fuel control even though the power lever in the cockpit is moved rapidly. As you learn about the J57 engine fuel system in this chapter, you will

Chapter II

THE POWER PLANT SYSTEMS

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In a turbojet engine, such as the J57, the compressor and turbine wheels are mounted on a single shaft at opposite ends of the combustion area. When combustion takes place, the rapidly expanding gases flow through the turbine wheel vanes and cause the wheel to rotate. When the turbine wheel rotates, it drives the compressor, which packs air into the combustion areas. The increased combustion pressure imparts additional force to the expanding gases. This is the combustion cycle in a turbojet engine; and if there is some means of starting the rotation and combustion, the engine will produce a continuous flow of power.

In addition to the basic part, which accomplishes the combustion cycle, every turbojet engine must also have several associate systems. These associate systems function to assure that the engine will produce its continuous flow of power in the best operating conditions possible. In this chapter you will learn about the major associate systems in the J57 engine. These include the fuel, oil, starting and ignition, induction and cooling, and anti-icing systems.

THE J57 ENGINE AND AFTERBURNER FUEL SYSTEM.

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The main fuel control allows the jet engine to accelerate and decelerate at a predetermined safe rate, thereby reducing the possibility of a flameout. This rate of acceleration and deceleration is kept constant by the main fuel control even though the power lever in the cockpit is moved rapidly. As you learn about the J57 engine fuel system in this chapter, you will

discover how the main fuel control unit governs both of these functions and relieves the pilot of almost all of the responsibilities of monitoring the fuel to the engine.

The J57 engine and afterburner fuel system is designed to schedule fuel in the proper amounts to the engine under all conditions of flight. This fuel system will supply the correct amount of fuel for fixed-power settings at varying temperatures and altitudes. The system will also permit proper engine acceleration and deceleration—without exceeding engine limitations, experiencing lean or rich blow-out, or experiencing compressor surge.

Although the engine and afterburner fuel system is designed to function as one individual fuel system, it will be treated in this manual as if it were composed of two subsystems—the engine main fuel system and the afterburner fuel system. The engine main fuel system meters fuel according to the requirements of the engine for any given power lever setting. The afterburner fuel system schedules the fuel for afterburner operation, actuates the exhaust nozzle control valve, and initiates afterburner ignition.

The engine-driven fuel pump/transfer valve is the only component that is common to both the engine and afterburner fuel systems. It is a combination unit with one part of the unit functioning as the fuel pump for both fuel systems and the other part of the unit scheduling fuel to the afterburner fuel system whenever required. The transfer valve part of this combination component also includes a safety device which automatically switches the fuel flow from the afterburner fuel system to the engine main fuel system in the event that the fuel pump for the engine main fuel system fails.

The components in the engine main fuel system include the fuel pump/transfer valve, the main fuel control unit, the fuel flowmeter, the fuel/oil cooler, the pressurization and dump valve, and the burner nozzles and manifolds.

The afterburner fuel system components are the fuel pump/transfer valve, the afterburner fuel regulator, the igniter valve, the exhaust nozzle control valve, the manual afterburner shutoff valve, and the fuel spraybars and manifolds.

THE ENGINE POWER CONTROL SYSTEM.

The pilot's power lever control quadrant is located on the left side of the F-102A cockpit. The movement of the power lever is transmitted mechanically to the engine main fuel control unit and to the mechanically-operated afterburner shutoff valve. As you will note in figure 2-1, a Teleflex cable installation connects the power lever quadrant bellcrank to the control mechanism in the left side of the engine compartment. This aft control mechanism is attached to a bellcrank on the engine control cross-over shaft. Solid linkage connects the bellcrank to the fuel control unit and to the afterburner shutoff valve.

The slot in the power lever control quadrant permits fore and aft movement of the power lever. It also permits the power lever to be moved outboard and inboard to a limited degree. The fore and aft movement of the power lever is transmitted mechanically to the fuel control unit by means of the Teleflex cable installation. The inboard and outboard motion of the power lever actuates microswitches in the quadrant which control the engine starting and afterburner initiation.

In figure 2-2, note that the fore and aft motion of the power lever is affected by both internal and external detents and a trigger-operated mechanical stop (power lever trigger). The detents serve a two-fold function; they prevent the pilot from inadvertently moving the power lever to some undesirable position, such as retarding the lever to the OFF position, and the like; they also assure that the power lever must be moved in such a manner that it will actuate the microswitches inside the quadrant.

A flat pattern of the quadrant slot is shown in the right hand portion of figure 2-2. Note the START detent which permits the lower lever to be moved outboard from the OFF position. This detent also limits the forward movement of the power lever when the lever is positioned to the extreme outboard side of the lever slot. If the power lever is to be moved forward beyond the START detent, the lever must be returned to the inboard side of the lever slot so that it will clear the IDLE detent. From the IDLE position, the power lever can be advanced forward still further until it reaches the FULL MIL POWER position. At this point, a mechanical stop prevents advancing the power lever any farther until the trigger on the power lever is depressed. When the operator pushes this trigger down, he can move the power lever from the FULL MIL POWER position to the TAKE OFF position. Movement of the power lever from the FULL MIL POWER position to TAKE OFF does not increase engine power; that is, the movement does not increase the amount of fuel that is delivered to the engine—the power lever movement merely positions the mechanical takeoff locks in the fuel control unit. These fail-safe takeoff locks prevent loss of engine power in the event of a malfunction in the burner pressure or inlet pressure sensing systems. The mechanical takeoff locks in the fuel control unit will be explained in greater detail later in this chapter.

When the power lever has been advanced to about 80% power, the AFTERBURNER detent in the quadrant cover allows the power lever to be positioned

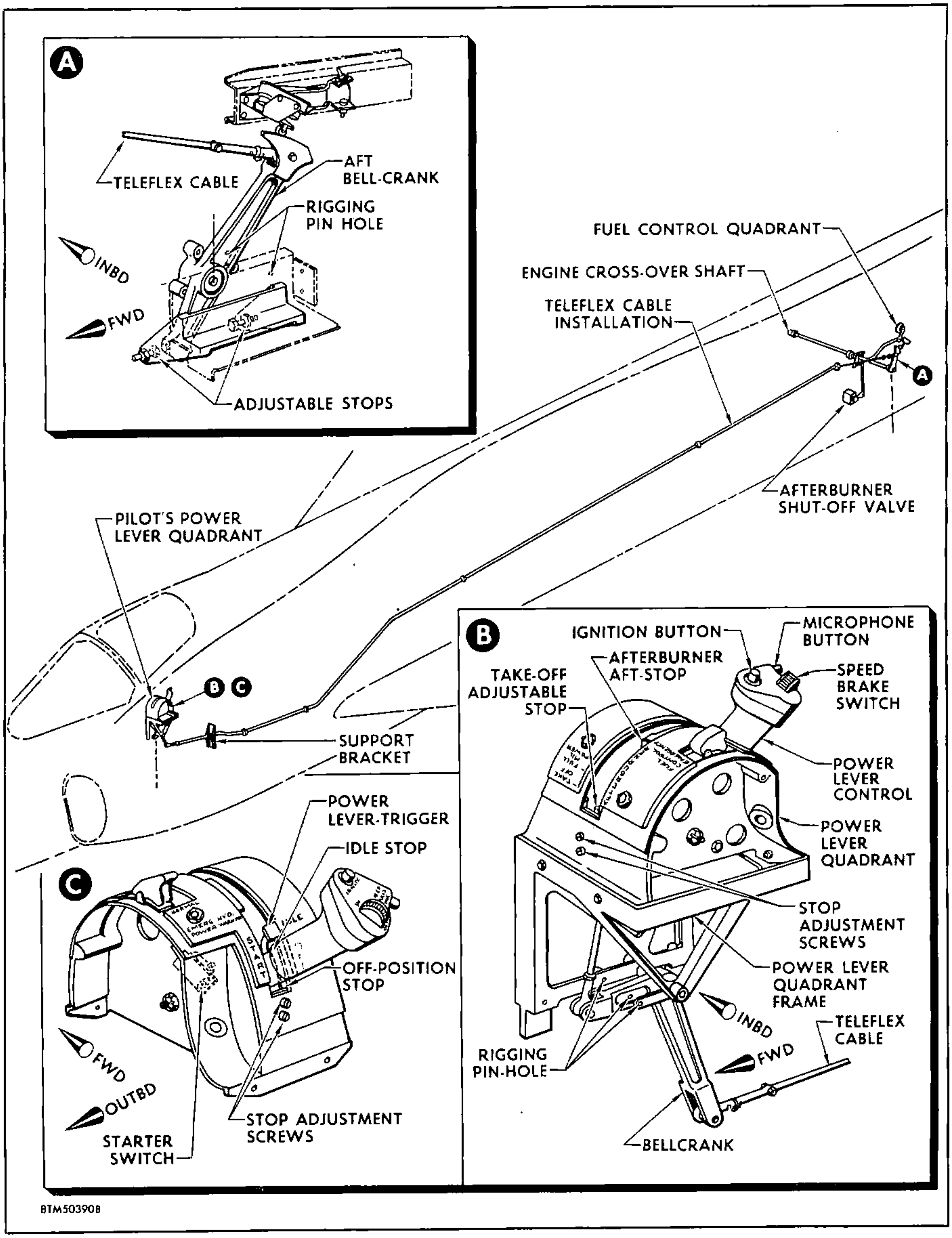


Figure 2-1. Power Lever Control System

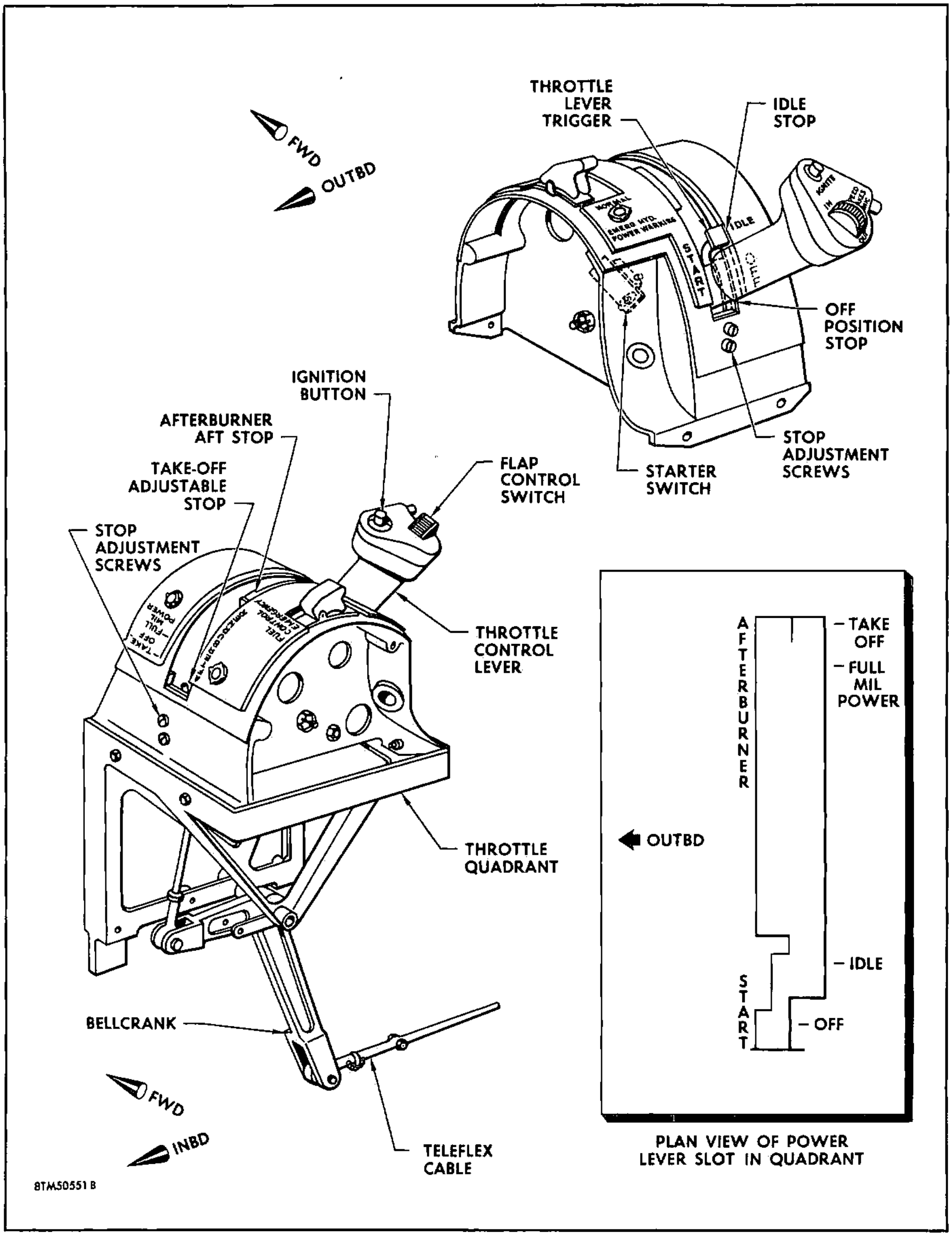


Figure 2-2. Pilot's Power Lever Positions

outboard. As the power lever is pushed outboard in this detent, it actuates microswitches inside the quadrant to initiate afterburning. A spring-loaded ball detent holds the power lever in the AFTERBURNER detent. The afterburning operation is normally stopped by bringing the power lever back to the inboard side of the quadrant, thus de-activating the microswitches in the quadrant. The power lever has to be retarded about five degrees from the full forward position before it can be brought out of the AFTERBURNER detent. This feature insures against the possibility of inadvertently stopping afterburning during take off. A mechanically-operated afterburner shutoff valve (controlled by the power lever movement) stops afterburning when the power lever is retarded below 80% power.

During power reduction the power lever contacts the IDLE detent in the quadrant cover. A slight outboard movement of the lever allows the lever to clear the IDLE detent and be retarded still farther back into the OFF position.

Referring back to figure 2-1, note that both the power lever quadrant and the engine compartment aft bellcrank have adjustable mechanical stops which limit power lever movement. These mechanical stops are preset on all airplanes. Normally you will not have to adjust these stops unless you are installing a new Teleflex cable assembly. Even when the engine is changed, the power control rigging is not disturbed, and adjustment of the mechanical stops is not required.

Rigging the power lever linkage after bellcrank or Teleflex cable replacement, however, should not present any complex problems. Before attempting to secure any part of the linkage, you must insert rig pins in both the power lever and engine aft bellcranks. The rig pins will keep the two bellcranks in their OFF positions while you install the Teleflex cable assembly. Both of the adjustable mechanical stops on the bellcranks should be backed off so that they will not interfere with the rigging process. When installing a new Teleflex cable assembly, you must adjust the rod end at each end of the cable until the rod end linkage bolts can be inserted. Do not remove the rigging pins until all bolts have been tightened and all safetying completed.

THE ENGINE FUEL SYSTEM.

The engine fuel system is usually considered to be the heart of the engine. This fuel system performs two essential functions: first, it provides a means for controlling the engine power output; and, secondly, it schedules the fuel to the engine as required for varing engine operation conditions. Power regulation could possibly be accomplished by controlling the air flow at the engine air intake. However, the mechanical problems imposed by such a control system make this type of installation very impractical. Accordingly, the fuel flow rather than the air flow is regulated in the J57 engine.

Modern jet engine design permits the pilot to select a power setting by merely positioning the power lever. The fuel control unit will then schedule sufficient fuel to prevent rich mixture blow-out during acceleration, lean mixture die-out during deceleration, and provide correct fuel flows for varying engine operating conditions during fixed power lever settings.

As mentioned earlier, the engine fuel system consists of an engine-driven fuel pump/transfer valve, the fuel control, fuel flowmeter, fuel/oil cooler, pressurizing and dump valve, and burner nozzles and manifolds. Figure 2-3, shows the entire engine fuel system. In these first few paragraphs you will be given a general description of the fuel system. Then, each component in the system will be discussed in detail.

Note on figure 2-3 that the fuel flows from the fuel pump/transfer valve to the main fuel control unit. This main fuel control unit has both a primary (normal) and an emergency metering system. The primary fuel system meters fuel automatically to the engine during normal operation. This automatic fuel metering is accomplished by the fuel control governor flyweight reaction against the power lever position. The amount of flyweight reaction is dependent upon the N_2 compressor pressure and the pressure in the burner "cans." As a safety feature in the primary fuel system, mechanical takeoff locks are incorporated in the fuel control unit to assure an adequate fuel flow during takeoff in the event that either of the pressure sensing systems fail. The emergency fuel system of the fuel control unit is entirely separate from the primary fuel system. Fuel metering during emergency operations is directly dependent upon the position of the power lever. There is, however, a device for limited altitude compensation in the emergency fuel system.

Referring to figure 2-3 note that the fuel flows from the fuel control unit through the fuel flowmeter, then passes through the fuel/oil cooler into the pressurization and dump valve. Fuel from this valve flows to the fuel nozzle manifolds located in the engine burner section. Each of the eight burner cans incorporate six dual fuel nozzles with primary and secondary openings.

How the Fuel Control Primary Fuel System Operates.

The main fuel control unit is the only component in the engine fuel system which the pilot can directly control. As a result, the fuel control unit is the most

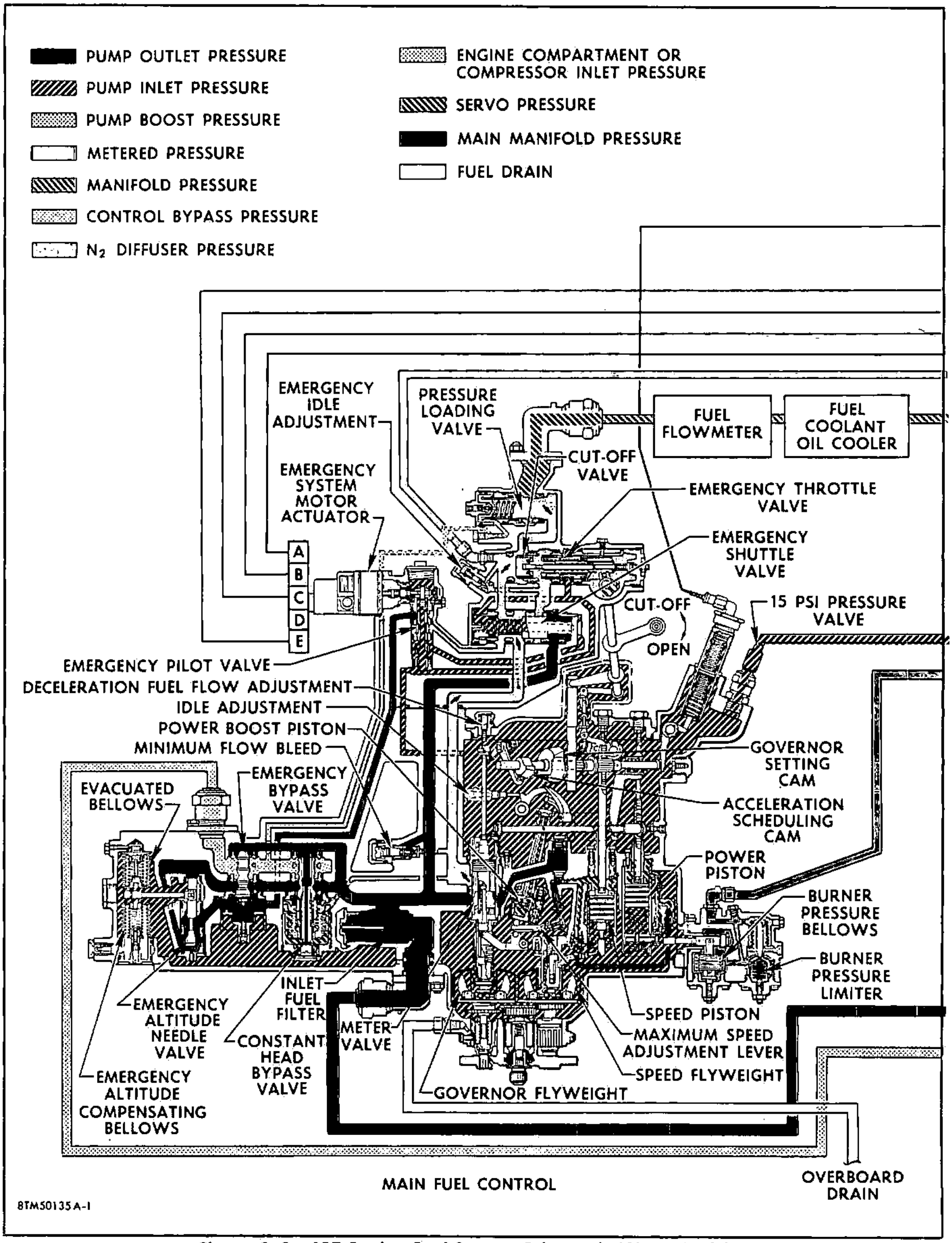


Figure 2-3. J57 Engine Fuel System Schematic (Sheet 1 of 2)

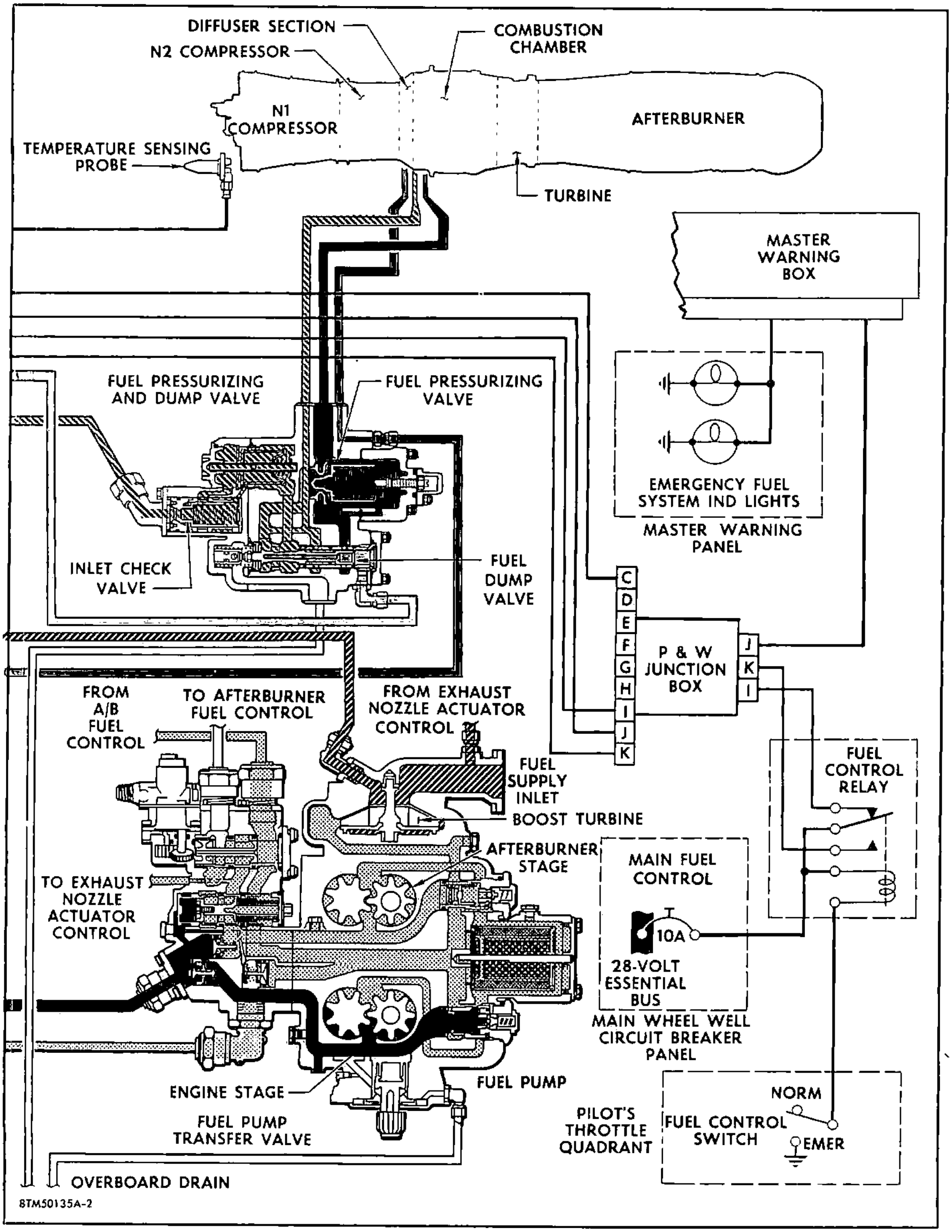


Figure 2-3. J57 Engine Fuel System Schematic (Sheet 2 of 2)

important single unit in the entire engine fuel system. This component provides fuel to the engine burner section at the proper volume and pressure to maintain the engine performance as demanded by the power lever setting.

To perform these functions properly, the fuel control must be able to: provide acceleration control so that turbine temperatures are not exceeded and a flame-out condition or compressor stall is not experienced; provide deceleration control without lean mixture die-out; maintain a constant power setting regardless of altitude, air temperature, or airspeed; and provide emergency fuel without engine failure in the event the normal engine fuel system fails.

Fuel control units vary with different engine installations; however, by studying one particular type, the basic function of all fuel control units may be understood. The type discussed in this manual is the hydromechanical fuel control unit used on the J57 engine, which is shown in figure 2-4.

The main fuel control unit is mounted on the lower left side of the engine accessory drive housing. This unit is driven at a ratio of 0.344 to 1. Its basic function, as a speed-density fuel control, is to schedule fuel to the burner cans for normal engine operation without exceeding the engine limits. Fuel requirements are determined by the engine operating conditions and the position of the power lever. Thus, the fuel control unit is basically a speed governor that is biased by burner pressure and inlet air temperature.

The following paragraphs explain how the fuel control unit operates. By frequently referring to the schematic of the engine fuel system, (figure 2-3), as you read these paragraphs, you will obtain a better understanding of the main fuel control unit and its functions.

The primary fuel system supplies fuel for normal engine operation. Following the fuel flow from the pressure outlet of the fuel pump/transfer valve, see figure 2-3, you will note that fuel enters the fuel control unit and immediately passes through the spring-loaded fuel inlet filter. If the filter should become clogged, the incoming fuel will force the filter from its seat and bypass unfiltered fuel into the fuel control unit. Downstream from the filter, the fuel flow is divided—one fuel passage leads to the fuel metering valve, the second passage leads to the anti-spring side of the emergency shuttle valve in the upper part of the fuel control unit, and the third passage leads to the constant-head bypass valve. The constant-head bypass valve gets its name from the description of its job—it maintains a constant fuel head (pressure) across the fuel metering valve. The fuel in excess of that required by the metering valve is routed by the bypass valve back to the fuel pump/transfer valve. In the schematic of the engine fuel system, figure 2-3, this return flow is shown in yellow.

Metered fuel enters the hydraulically-balanced emergency shuttle valve (which is held open by spring pressure) and then flows to a cutoff valve. As shown in figure 2-3 this cutoff valve is mechanically operated by the power lever and the valve is always open when the engine is operating. As the fuel flows from the cutoff valve, it passes through a pressure loading valve and then out of the fuel control unit discharge port.

THE MAIN METERING VALVE AND BYPASS VALVE. The metering valve and the constant-head bypass valve combination can be considered the heart of the primary fuel system in the fuel control unit. These two valves determine how much fuel is metered to the engine combustion chambers. The fuel from the engine-driven fuel pump is routed to the metering valve where it flows through a single orifice, or hole, in the metering valve. The size of the metering valve orifice is variable and it is automatically controlled. A constant pressure drop across this orifice is maintained by the constant-head bypass valve.

The metering valve is a sleeve-type valve with two stepped slots (180° apart) and two deceleration slots (also 180° apart) which align with fixed slots in the valve sleeve. The valve sleeve is attached to the fuel control unit body and the valve moves inside the sleeve. The metering valve moves both radially and axially to establish the required metering orifice. In other words, the metering valve turns as well as moving fore and aft. The valve axial movement is obtained by the fore and aft movement of the power lever in the cockpit. This axial movement of the valve tends to open or close the metering valve orifice, depending upon whether the power lever is being advanced or retarded. This *axial* (fore and aft) movement is accomplished through the governor setting cam and attendant linkage by servo action to the governor spring. The power lever motion either increases or decreases the governor flyweight's spring tension and the resultant metering valve position depends on governor flyweight reaction and the governor spring pressure. The metering valve movement is also biased by two other factors: burner pressure acting through the burner pressure servo mechanism causes a *rotary* motion of the metering valve; a restraining action to axial movement is imposed during acceleration by the acceleration governor, cam, and linkage. In addition to these biasing factors, compressor inlet temperatures also control the metering valve position. The varying inlet temperatures result in movement of both the governor setting cam and the acceleration cam. Both of these cams, by means of cam followers and linkage, *axially* position the metering valve.

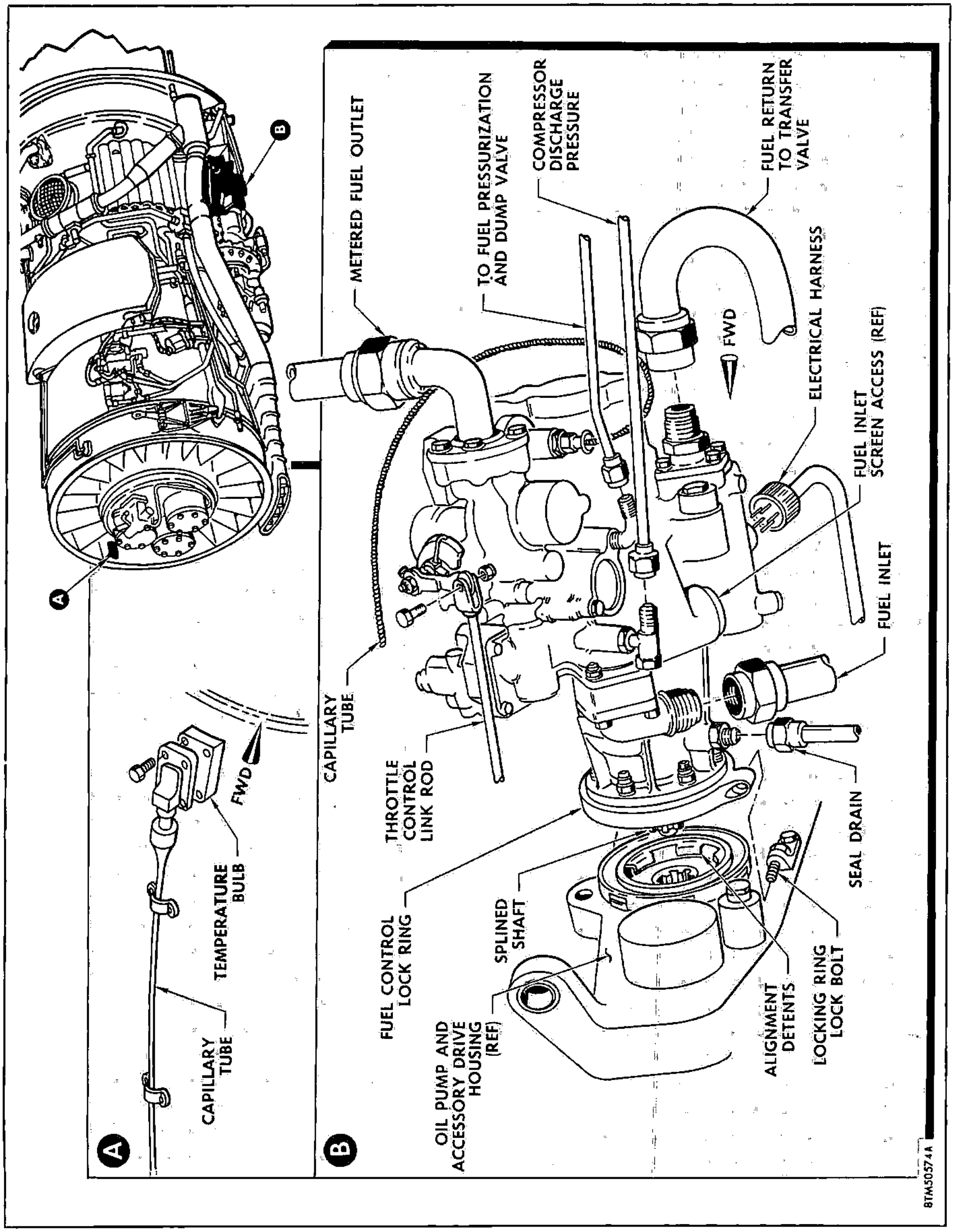


Figure 2-4. Main Fuel Control Unit

As mentioned before, the constant pressure drop across the metering valve is maintained by the constant-head bypass valve assembly. This keeps a constant pressure differential between the unmetered and the metered fuel. If the pressure drop across the metering valve starts to vary from its proper value, the bypass valve either opens or closes to regulate the amount of unmetered fuel that is to bypass the metering valve. The bypass valve accomplishes its job by means of spring pressure. A bellows and hollow-stemmed servo valve maintain the spring at a constant length by varying the bellows length to follow the movement of the bypass valve. This type of arrangement always keeps the same spring pressure on the bypass valve regardless of where the valve is positioned. Referring to figure 2-3, you will note that metered fuel is also routed to the bypass valve. The pressure of the metered fuel combines with the spring pressure to oppose the pressure of the incoming unmetered fuel. The resulting balancing action of the pressures position the bypass valve.

TEMPERATURE COMPENSATION IN THE FUEL CONTROL UNIT. The temperature of the air at the compressor inlet also affects fuel scheduling in the primary fuel system in the fuel control unit. This air temperature compensation is accomplished by mechanical linkage inside the fuel control unit which axially positions the governor setting and acceleration cams. Note in the engine fuel system schematic, figure 2-3, that a temperature sensing bulb is installed in the compressor inlet case. This sensing bulb is connected by a capillary tube to a bellows and spring-loaded piston in the fuel control case. The temperature sensing bulb is actually a liquid-filled bulb. A better view of the sensing bulb and its connecting capillary tube is shown in figure 2-4. The temperature variations of the engine compressor inlet air cause the liquid to expand and contract. This increasing and decreasing pressure of the liquid is transmitted to the fuel control unit by means of the capillary tube. The resulting motion of the bellows and piston caused by the liquid pressure variations moves the cam shaft axially. This axial cam shaft motion causes the cam followers to reposition in respect to the axial contour of the cams. The cam followers are mechanically connected to the metering valve, and their repositioning movement caused by the temperature variations affects the meter valve position.

As mentioned earlier in the description of the fuel control unit, you should never disconnect the capillary tube from the fuel control unit. Each fuel control unit has a temperature bulb and capillary tube attached to it at the factory and the units are matched for proper operation; consequently, the units should never be changed individually.

BURNER PRESSURE COMPENSATION IN THE FUEL CONTROL UNIT. The efficiency of the combustion of the fuel-air mixture in the jet engine combustion chambers is directly related to the pressure inside the combustion chambers (burner cans). The combustion efficiency is highest when the chamber pressure is high, and the efficiency steadily decreases as the chamber pressure drops. The pressure in the turbojet burner cans depends on engine speed, aircraft speed, and altitude. Since the fuel-air mixture depends on the combustion chamber pressure, the fuel control unit uses a sensing system so that the combustion chambers receive the correct amount of fuel for their existing pressures. This sensing system is known as the burner pressure compensation system.

A pressure probe in the No. 4 burner can acts as the sensing element for the burner pressure compensation system. The pressure line from the No. 4 can passes through the pressurization and dump valve housing and enters the burner pressure limiter inside the fuel control unit. In the engine fuel system schematic, figure 2-3, the burner pressure is labeled in the index as N_2 Diffuser Pressure, and this pressure is shown in a blue-green color.

As the pressure in the burner can changes, because of altitude and engine operating conditions, the pressure on the burner pressure bellows also changes. These bellows contract or expand (depending upon whether the pressure is increasing or decreasing) and activate the burner pressure servo. This servo action is transmitted through a rack-and-pinion, a shaft, and bevel gears to produce a *rotary* motion of the fuel metering valve. The result of this action is to either narrow or widen the fuel-metering slots in the fuel meter valve and to affect the primary fuel flow accordingly.

THE INTERNAL CUTOFF VALVE AND FUEL CONTROL BODY PRESSURIZATION. The mechanically-operated cutoff valve stops the fuel flow when the engine is shut down. In figure 2-3, the cutoff valve is shown in the upper portion of the fuel control unit. This valve is connected to the power lever linkage and the valve is open whenever the engine is operating. A return line, connecting the fuel control body to the fuel pump inlet, prevents fuel pressure buildup within the fuel control unit body when the engine is shut down. This line is shown in yellow on figure 2-3. A resultant pressure drop within the fuel control unit body is transmitted to the pressurizing and dump valve by a sensing line, and this pressure causes the dump valve to drain the fuel nozzle manifolds. A 15-pound, spring-loaded valve is incorporated to control the body/pump inlet return line. The resultant fuel control unit body pressure insures proper operation with warm fuel by preventing over-speeding of the internal flyweight speed governor.

INTERNAL PRESSURE LOADING VALVE. Metered fuel from the emergency shuttle valve enters the pressure loading valve chamber directly above the cutoff valve. Passage of the fuel to the control body discharge outlet is restricted by a spring-loaded piston that is backed up with fuel bypass pressure. The pressure exerted by the metered fuel to overcome the pressure on the loading valve insures sufficient pressure for normal servo action.

FUEL CONTROL INTERNAL SERVO SYSTEMS. Servo assist is utilized in the power lever actuating linkage, acceleration cam/speed piston linkage, and burner pressure metering valve linkage. Force reduction of these three linkage systems is essential for proper fuel scheduling. Translation of temperature variation from the governor setting cam to the governor flyweight spring is made more effective through servo action. Servo action also reduces the cockpit power lever load. Translation of burner pressure to the metering valve, and the speed flyweight reaction to the acceleration cam is made possible by the burner pressure servo and the speed servo assemblies, respectively.

The source of pressure for these servo systems is unmetered fuel taken from the metering valve area. The servo pressures are shown in yellow-green color in the schematic of the engine fuel system, figure 2-3. After passing through a filter, the unmetered fuel is ported to the three servo systems and, unless the servo system exit is restricted, the fuel passes out into the regulator case without causing servo action. If, however, a servo system exit is restricted, pressure immediately builds up within the respective system, and the servo piston moves until the exit restriction is removed. Removal of the restriction decreases internal pressure and stops servo piston motion. Actually, the restrictions of the servo exits are caused by half-balls being positioned over the exit. This positioning of the half-balls in each of the three servo systems determines the degree of servo action. When a half-ball restricts a servo pressure exit, the resultant servo action will unseat the half-ball (through mechanical linkage) after traveling a predetermined distance. This action may be traced by careful examination of the servo systems in the fuel system schematic, figure 2-3.

TAKEOFF LOCKS. A pair of mechanical locks is incorporated in the fuel control to insure adequate fuel flow in the event of either burner pressure or inlet temperature sensing system failure. During takeoff these ratchet-type locks are engaged mechanically when the power lever is advanced from FULL MIL POWER position to the TAKE OFF position. Theoretically, no power increase results during this power lever advancement, because the governor setting cam follower is following the cam "flat" during this operation. A rise in power does, however, occur from 54° power lever setting to the takeoff locks, or approximately 1½% military power increase. One locking lever is positioned adjacent to a collar on the cam shaft to prevent excessive shaft travel in the event of temperature compensating system failure. The second lock is positioned to prevent the burner pressure servo rack from closing the metering valve orifice below the predetermined position. Excessive engine overspeeding is prevented by governor flyweight action, which limits metering valve axial motion.

After the takeoff has been accomplished, it is mandatory that the power lever be returned to the FULL MIL POWER position to remove the takeoff locks. If the locks are permitted to remain in the takeoff position, they will severely limit both the inlet temperature and the burner pressure compensation that is necessary for proper fuel scheduling at high altitudes. Should either or both of the "locked" systems fail during takeoff, a slight power reduction may be noted. The amount of power fluctuation noted will depend on ambient air and altitude. If a component had failed, a more noticeable and possibly severe power reduction would be noted when the power lever releases the locks as it is brought back to the FULL MIL POWER position. Under this condition it will be impossible to re-engage the locks by repositioning the power lever back to the TAKE OFF position. Under normal operating conditions the power lever should be moved back from the TAKE OFF position before reaching 6000 feet.

How the Emergency Fuel Control System Operates.

The emergency fuel control system operates independently of the primary (normal) fuel control system. It is important to note that there is, for the present at least, no provision for automatically switching from the primary operating system to the emergency system. Instead, the emergency system can be activated only by actuating the emergency switch in the cockpit. A motor-operated rack-and-pinion drive positions the emergency pilot valve. In the normal position, the emergency pilot valve insures that the primary fuel control system output passes the emergency shuttle valve and is discharged in the conventional manner to the fuel pressurizing and dump valve. The emergency shuttle valve is hydraulically balanced during primary (normal) fuel scheduling. Spring pressure moves the shuttle valve to shut off the fuel passage to the emergency throttling valve, and the metered fuel is ported to the cutoff valve. When the cockpit emergency switch is actuated, the emergency system motor repositions the emergency pilot valve so that fuel pressure is ported to the anti-spring side of the emergency shuttle valve. The emergency shuttle valve then moves against the spring pressure, opens the passage for unmetered fuel to enter the emergency throttling valve, and also shuts off the metered fuel passage to the cutoff valve.

Since emergency fuel operation completely isolates the primary or normal fuel regulating components, an emergency throttle valve, emergency bypass valve, emergency altitude needle valve, and an emergency altitude compensating bellows (working in conjunction with an evacuated bellows) take over the fuel scheduling duties. Emergency fuel metering is achieved by varying the emergency throttle valve. The linear travel and flow through this valve are in turn varied by moving the valve's fixed orifice in relation to a fixed, contoured needle and an idle bleed, which operates to maintain minimum flow. A constant pressure drop is maintained by the emergency bypass valve. This valve is similar to that used in the primary or normal system, except that there is no compensation for varying spring length. Instead, the emergency bypass valve operates in conjunction with the emergency altitude needle valve to establish the desired metering load across the emergency throttle valve. The emergency altitude needle valve is operated by linkage from the emergency altitude compensating bellows. Accordingly, altitude variations will either increase or decrease the fuel flow to the emergency throttle valve.

During emergency operation, fuel scheduling is a *direct function* of power lever movement and no provisions are made to alter the fuel flow during acceleration, deceleration, or for variations of temperature and burner pressure. Because of this, the pilot must observe rpm, tailpipe temperature, and pressures very closely during emergency operation. Limited altitude (approximately 30,000 feet) compensation, however, is provided by action of the emergency altitude compensating bellows and the emergency altitude needle valve.

Flight Line Adjustments on the Fuel Control Unit.

Almost all of the adjustments that can be made on the fuel control unit must be done on a flow bench by fuel control specialists. The only adjustments which you will be allowed to make on the flight line are the IDLE and MAXIMUM flow adjustments. These two adjustment points are shown in the upper portion of figure 2-5. Each of the adjustment screws are designed with spring-loaded detent balls to provide click-lock adjustment. Fourteen clicks equal one full turn of the adjustment screws.

Turning the IDLE adjustment screw in a clockwise direction increases the preload on the governor spring through the cam follower, increasing IDLE fuel flow. Turning the screw in the counterclockwise direction, of course, decreases IDLE fuel flow.

The MAXIMUM adjustment screw repostions the maximum rpm lever inside the fuel control unit. This varies the effective length of the governor cam follower-rod. Turning the MAXIMUM adjustment screw clockwise increases the fuel flow. Because of the rather complex mechanical linkage between the adjustment screw and the maximum rpm adjustment lever, it is advisable to make four turns beyond the desired setting point when increasing the trim; then back off to the desired point. This method of first turning past the point and then back will eliminate the built-in backlash in the complex mechanical linkage. When reducing the MAXIMUM trim, however, it is necessary only to turn directly to the desired setting. The later model fuel control units will have a revised, spring-loaded, mechanical linkage that will eliminate the need for turning beyond the desired setting point and then backing off.

Fuel Pump/Transfer Valve.

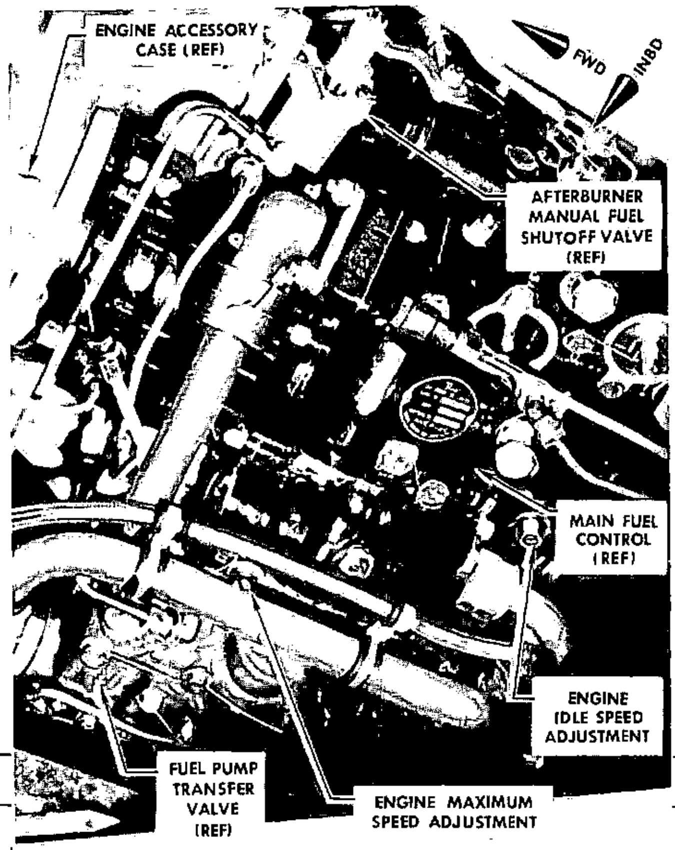
A three-stage engine-driven fuel pump, located on the right side of the engine wasp-waist section, provides fuel to both engine and afterburner (A/B). The fuel pump is shown schematically in figure 2-3. The first stage in the pump is a centrifugal boost turbine at the fuel inlet. Fuel from this turbine flows through a spring-loaded screen to the inlet side of the two gear stages. Both gear stages and the turbine stage are driven from a common shaft at a ratio of 0.344 to 1.0 and incorporate individual shear sections. Basically, one gear stage supplies fuel to the engine fuel control, the other supplies fuel to the A/B fuel regulator and associated units. Each gear stage has a pressure relief valve which limits the pressure rise to approximately 750 to 775 psi. A low-pressure warning light (in the cockpit) is connected to the engine gear stage outlet passage. This warning light system will be discussed later in Chapter IV.

As you will note in figure 2-6, a transfer valve is attached to the engine-driven fuel pump body. The primary purpose of the transfer valve is to provide fuel passage for engine operation and to supply fuel to A/B fuel components when afterburning is desired. During non-afterburning operations the normal A/B outlet is blocked by a motor-operated fuel shuttle valve. The fuel output from the A/B gear stage is then bypassed back to the gear inlet. When A/B operation is initiated, the motor-driven fuel shuttle valve opens both the fuel supply line to, and the return line from, the A/B fuel regulator.

An automatic fuel-regulating transfer valve, located between the A/B gear stage and the motor-driven shuttle valve, insures fuel flow to the engine fuel control in the event of engine gear stage failure. This two-position regulating transfer valve has engine gear stage fuel pressure bled to one side of the valve; this pressure opposes A/B fuel regulator bypass pressure and spring pressure on the other side. During normal operation the engine gear stage fuel pressure raises the valve against spring pressure and A/B fuel regulator bypass pressure, this permits the A/B gear stage

ENGINE TRIM PREPARATION

A. NO INLET DUCT SCREENS INSTALLED.
B. REFRIGERATION AIR "ON" (N_2) BLEED AIR).
C. CONSTANT SPEED DRIVE UNIT LOAD MAY BE EITHER ON OR OFF.



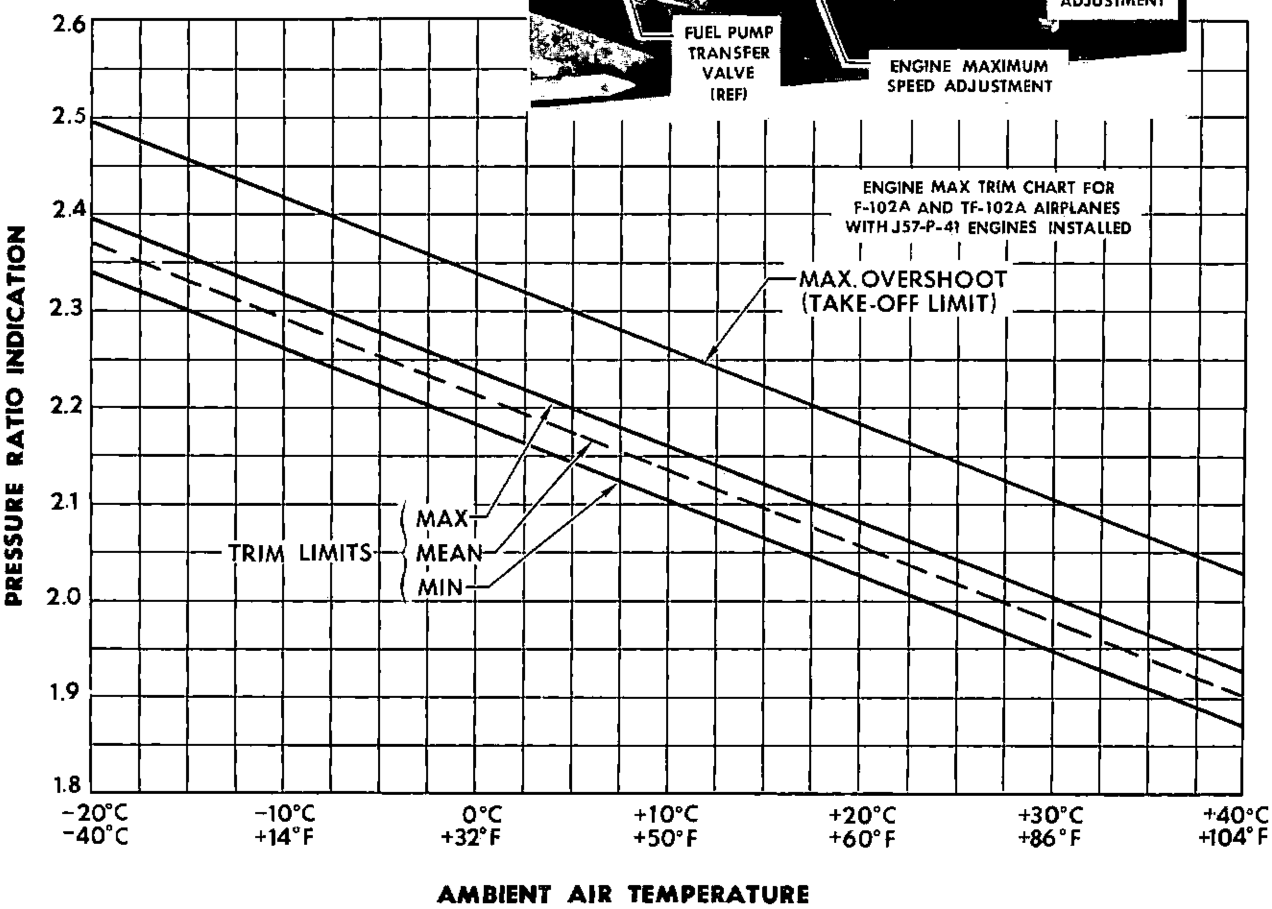


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Figure 2-5. Adjustment Points on the Main Fuel Control

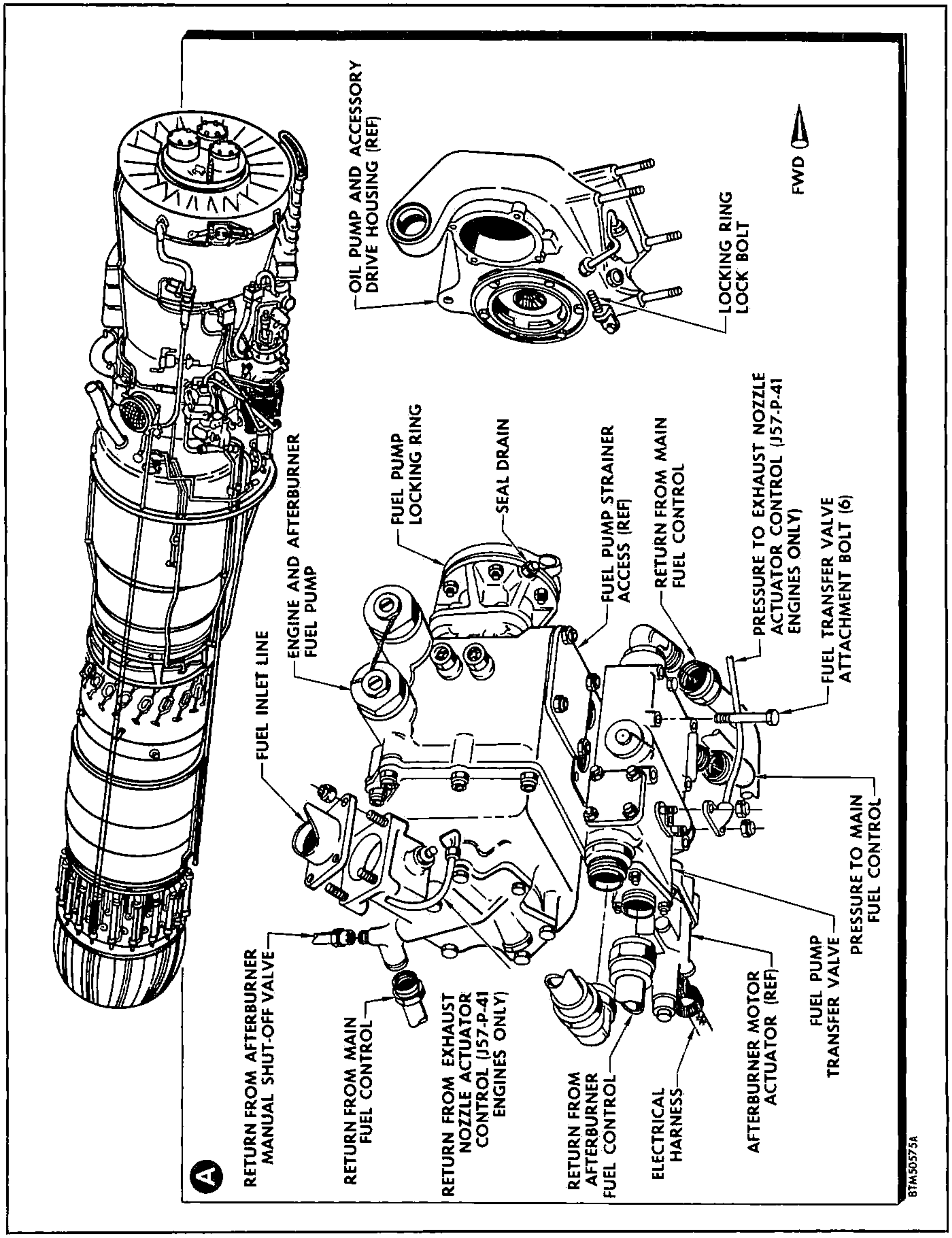


Figure 2-6. Engine-Driven Fuel Pump/Transfer Valve

fuel to flow to the motor-driven fuel shuttle valve. Should the engine gear stage fail, however, the resultant fuel pressure drop will cause the spring-loaded fuel regulating valve to move to block off the A/B gear stage outlet. Pressure building up in this line will unseat the spring-loaded check valve and permit fuel flow into the engine fuel regulator supply line, this prevents engine fuel starvation.

If the afterburner is operating and the engine gear stage fails, the A/B fuel supply will be limited or shut off entirely. Earlier airplanes did not incorporate provisions to supply fuel to the A/B during the emergency action noted. Later airplanes, however, will pass sufficient fuel through drilled passages in the transfer valve to permit limited afterburning.

The power lever quadrant in the cockpit has an A/B detent which parallels the regulator power setting slot through a range of 80% to 100% thrust for engines with A/B shutoff valves. Placing the power lever in this A/B detent actuates a microswitch in the A/B actuator motor circuit, and the afterburning cycle is initiated. When the power lever is taken out of the A/B detent, the A/B actuator motor is reversed and fuel flow to the A/B fuel regulator is stopped. Fuel scheduling for engine operation and for afterburning utilizes two separate sets of components downstream from the pump/transfer valve. The discussion immediately following will concern only those units used during engine (non-afterburning) operations.

Fuel Pressurizing and Dump Valve.

The metered fuel from the fuel control unit passes through the flowmeter and the fuel/oil cooler to the pressurizing and dump valve. As you will note in figure 2-7, this pressurizing and dump valve is mounted on the engine fuel manifold just aft of the fuel control unit.

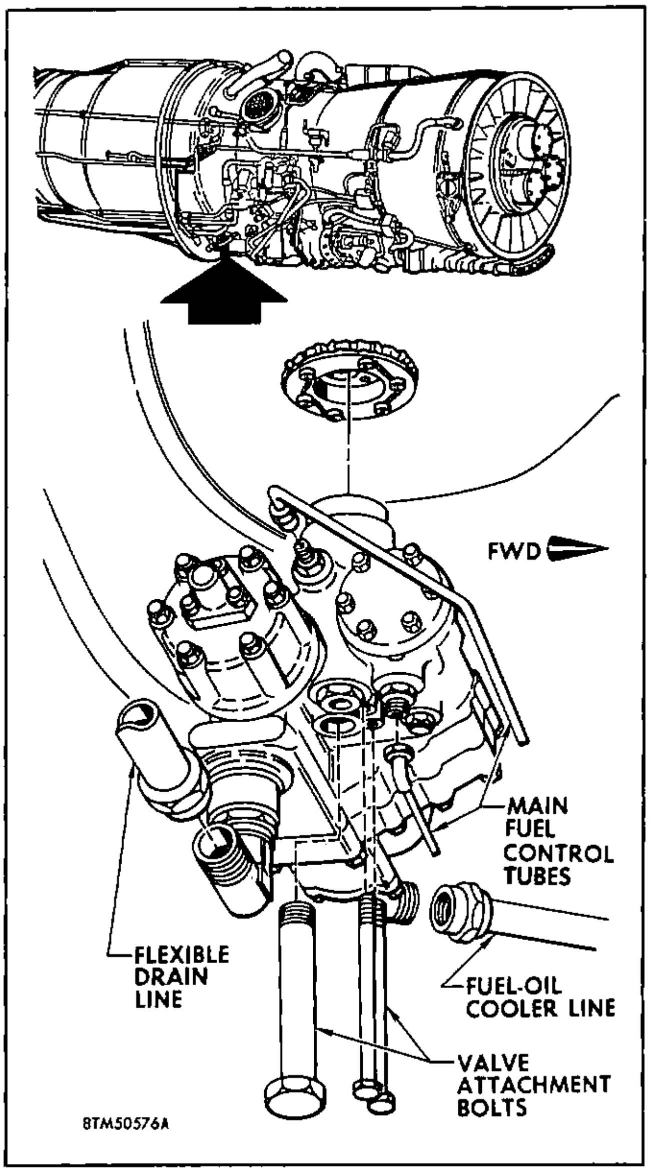


Figure 2-7. Fuel Pressurizing and Dump Valve

The pressurizing and dump valve accomplishes two functions—it controls the fuel flow to the primary and secondary injector nozzles in the engine burners, and it also dumps the nozzle manifold fuel when the engine has been shut down. When the engine is operating, fuel enters the unit through the inlet spring-loaded check valve and passes through a 200-mesh screen. Tracing the fuel flow on the schematic, figure 2-3, note that the fuel flows through the screen to the pressurizing valve. Fuel pressures from the fuel control unit body are routed by means of a sensing line (shown in brown) to the anti-spring side of the dump valve—this sensing pressure keeps the dump valve closed while the fuel passes on to the pressurizing valve in the unit.

During engine starting and low-power operation, all of the fuel (shown in green) flows directly to the pilot (primary) fuel manifold. But as the fuel pressure increases, the pressurizing (flow dividing) valve is unseated. This causes a part of the fuel (shown in blue) to flow to the main (secondary) fuel manifold. As the engine power increases, more fuel flows to the main fuel manifold.

When the power lever is brought back to the OFF position, the flow of fuel to the pressurization and dump valve is stopped. As the fuel pressure inside the fuel control unit body drops, the pressure in the sensing line also drops. The spring pressure on the dump valve overrides the decreased fuel control unit body pressure and forces the dump valve open. This drains the residual fuel from the manifolds and prevents any more fuel from entering. The spring-loaded inlet check valve in the pressurizing and dump valve prevents the fuel in the fuel/oil cooler from "boiling off" and dumping when the engine is shut down.

Engine Fuel Manifold.

The shrouded dual fuel manifold inter-connects the 48 dual fuel nozzles. This manifold is built up as one complete assembly. Fuel metering for the engine fuel system terminates at the burner cans—each of the eight cans has six dual nozzles. The pattern of fuel discharge in the burner cans is accomplished by the nozzle design, by air flow within the nozzle heads, and by swirl vane openings in the nozzles.

Maintenance on the Engine Main Fuel System.

The maintenance and trouble shooting requirements for the J57 engine fuel system are somewhat more complex than the requirements for piston-type engine fuel systems. As you have already learned in the preceding discussion of the engine main fuel system, each component in the fuel system does several jobs. As a member of the F-102A ground maintenance team, it will be your responsibility to assure that each component adequately performs its jobs, and that all components function together to supply fuel to the engine as demanded.

In all probability, your most difficult task in maintaining the engine main fuel system will consist in diagnosing engine malfunctions and than trouble shooting the fuel system to determine which component is causing the trouble. Although a trouble shooting guide is included in the engine maintenance technical order, some of the more common difficulties and their most probable causes are discussed in the following paragraphs.

About 90% of the engine fuel system troubles will be traceable to the fuel control unit. This should be understandable since the fuel control is a very complex system component. As you will recall, the fuel control unit should provide acceleration and deceleration control; maintain a constant power setting regardless of altitude, temperature, or air speed; and provide emergency fuel without engine failure in the event the regular engine fuel system should fail. Whenever one of these functions is not performed, you can almost be assured that the fuel control unit is malfunctioning. Since you are not allowed to perform any maintenance other than the IDLE and MAXIMUM flow adjustments on the fuel control, you must replace it with a serviceable control unit and send the defective control unit to the overhaul area for a bench check.

Referring to figure 2-4, you will note that the installation and removal of this engine fuel system component will be relatively easy. Before attempting to remove the unit from the engine accessory drive housing, you should disconnect all of the fuel, sensing, and electrical lines, and then provide some type of support for the unit. The capillary tube which connects the fuel control unit to the temperature bulb in the engine inlet must not be disconnected at the fuel control unit. The bulb and the connecting capillary tube must be removed with the fuel control unit. The reasons for this will be discussed later in this section. Note the locking ring and lock bolt directly under the attach point on the accessory drive housing. You will normally find it best to loosen the lock bolt until it begins to snug up, and then tap the bolt head until the bolt becomes loose. Keep doing this until the fuel control lock ring is free of the accessory drive housing flange. If the fuel control unit is to be left off for any time longer than a few minutes, be sure to cap all of the disconnected lines.

The installation procedure for the fuel control unit is essentially the reverse of the removal procedure; however, you should always coat all of the seals and gaskets with engine lubricating oil prior to installing the unit.

Other fuel system malfunctions which you might encounter will possibly occur in the emergency fuel system. The fuel control unit, as you will recall, also has a complete emergency metering system. If there is no emergency fuel flow at IDLE, replace the fuel control unit. Although there is no specifially recommended procedure for checking the emergency fuel flow, some helpful suggestions for determining whether the emergency fuel system is functioning properly are as follows: (1) carefully watch the tachometer for rpm change as you actuate the emergency switch, (2) watch the tailpipe temperature indicator for any change in temperature, and (3) watch how soon the red light that indicates emergency fuel comes on after you actuate the switch. In those cases where the engine will not change over to the emergency fuel system or the fuel control system selects the emergency system with the control switch in the NORM position, you will probably find the difficulty in the electrical circuit to the emergency system actuator. By performing a continuity check on the circuit from the 28-volt d-c bus to the actuator, you will be able to determine whether the malfunction is in the actuator itself. The most logical sources of difficulties of this type are the fuel control relay on the forward side of the right main wheel well or the emergency system actuator on the fuel control unit. When a malfunction is traced to either of these units, the defective unit must be replaced with a serviceable item.

FILTERS IN THE ENGINE MAIN FUEL SYSTEM. In the discussion of the engine main fuel system, it was mentioned that several of the components incorporated filters to assure clean fuel to the burner chambers. As you will recall, each of these filters is spring-loaded to bypass fuel in the event the filter becomes clogged. The filters in the system are of the

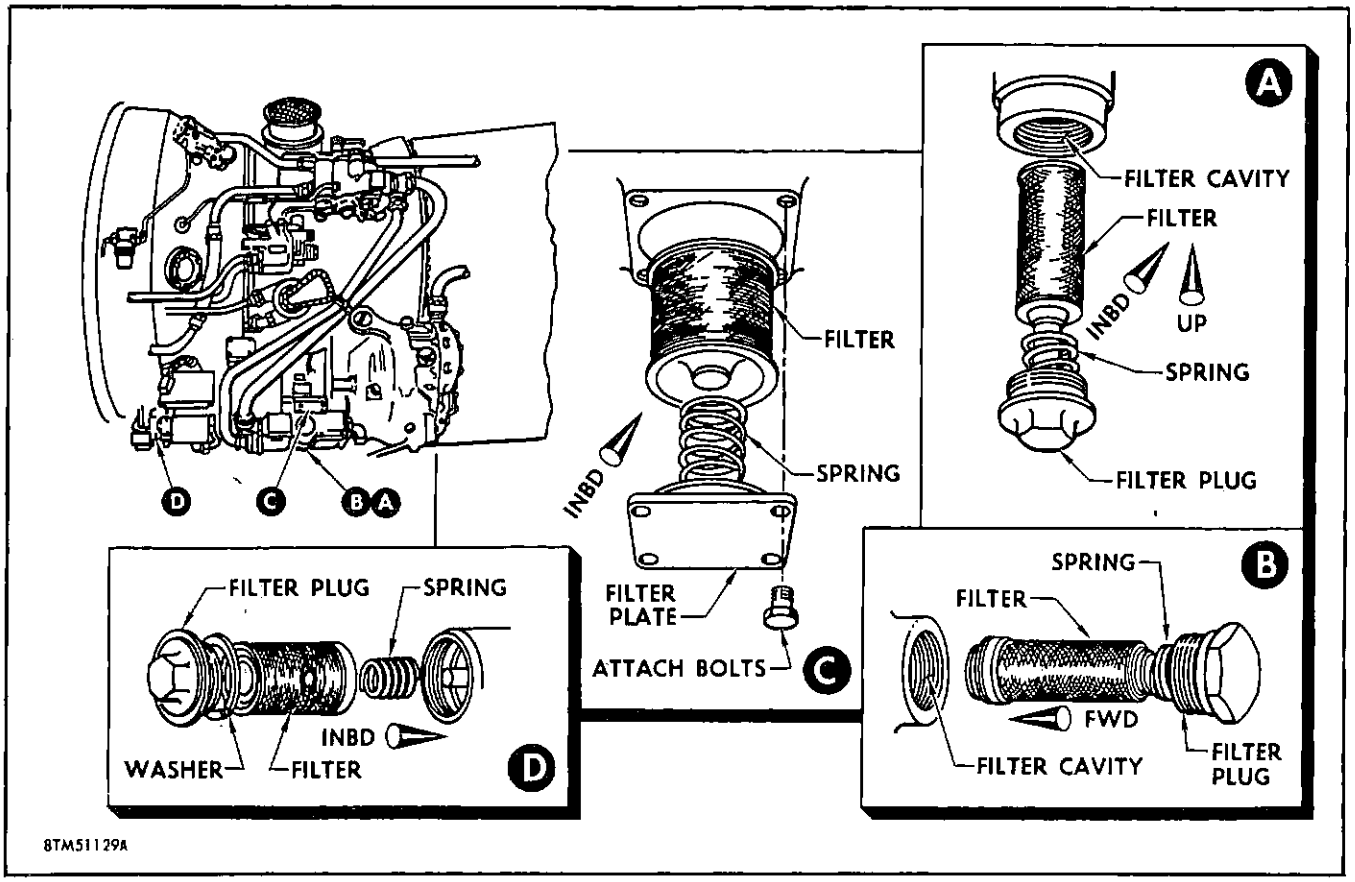


Figure 2-8. Fuel Filter in the Engine Fuel System—Schematic

200-mesh variety; that is, they are designed to restrict the passage of those particles which are larger than normally found in fuel. As you may already know, military specifications allow for some contaminants in jet fuels. The components in the engine fuel system are designed to function properly while handling fuels which do have contaminants in them; however, the components will not function properly if the size of the contaminants exceeds the size of the openings in the filters.

Cleaning the filters will be a part of the normal engine inspection activities. There are four filters in the engine main fuel system. As you will note in figure 2-8, they are located in the fuel control, fuel pump, and the pressurizing and dump valve. Although the filters do not all look alike, they perform the same function; and they are removed, cleaned, and replaced in the same manner. Note that each filter is held in place by a retaining plug.

To remove the filter, cut the safety wire and then unscrew the plug. All of the plugs have conventional right-hand threads. If the filters are not damaged in any manner, they can be cleaned by using either JP-4 fuel (MIL-F-5624), or naphtha, and then blowing the filter dry with clean, dry air. Installation of the filters for the engine fuel system is essentially the reverse of removal. Small holes in the heads of the plugs allow them to be safetied after they have been installed and tightened. You should always assure that the filter retaining plugs are tightened and safetied to eliminate the danger of fuel leaks and subsequent fire hazards.

THE AFTERBURNER FUEL SYSTEM.

The design of gas turbine engines limits the acceleration of jet aircraft during ground run, takeoff, and climb. This is due to the gas turbine engine's nearly constant thrust at all airspeeds at a given altitude and engine speed. In a turbojet engine, for example, the takeoff thrust may not be more than 15 or 20% above the normal rated thrust. This inherent design deficiency led to the development of thrust augmentation; thrust augmentation is merely a means of increasing the engine thrust to provide more power for takeoffs, climbs, or any other time when a rapid rate of acceleration is needed.

The J57 engine utilizes an afterburner as its means of thrust augmentation. As you learn about the afterburner fuel system in this section, you will see that the afterburner can be described as a ram-jet engine that is attached to the turbine section of the regular engine. The afterburner fuel system and the system

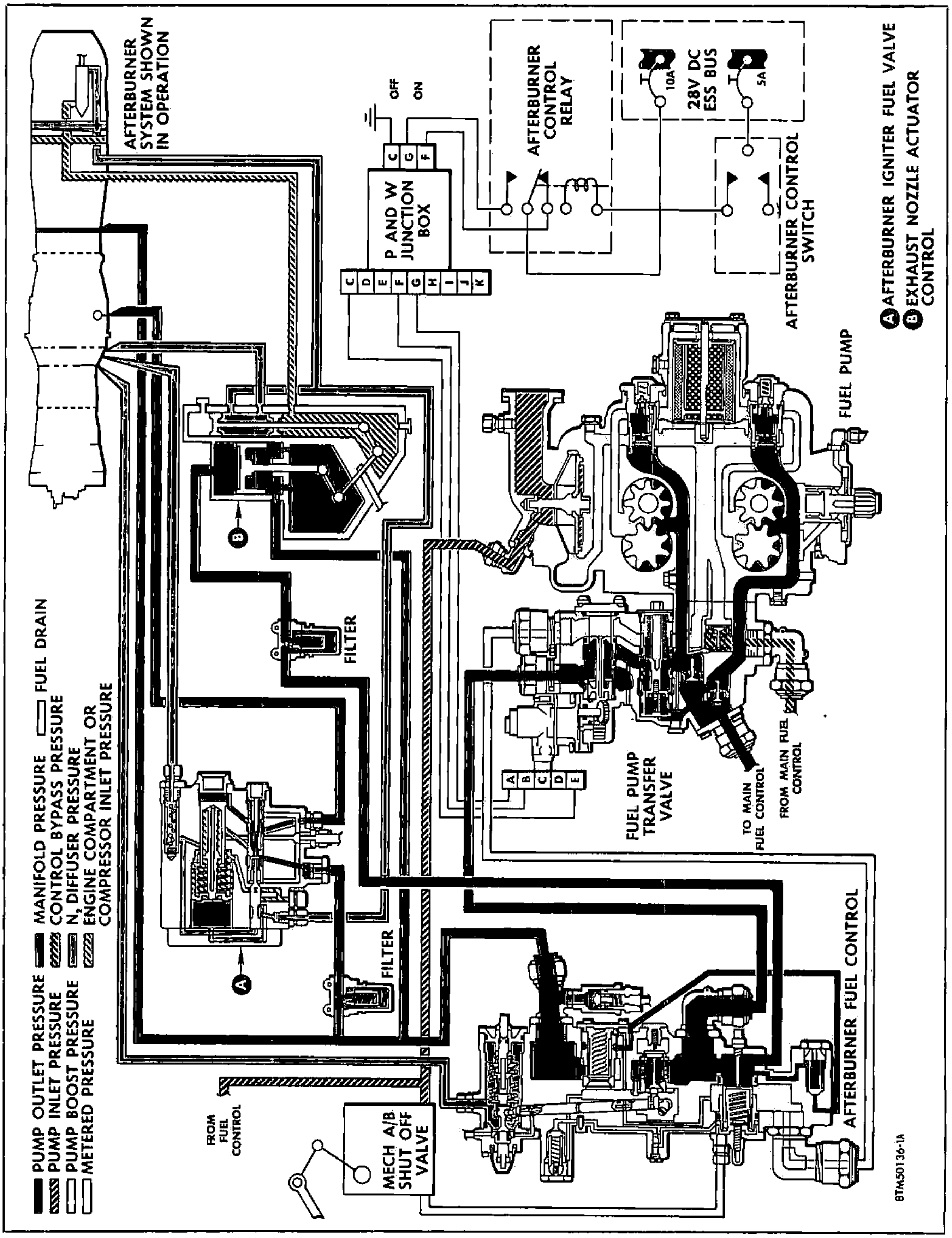


Figure 2-9. J57 Engine Afterburner Fuel System—Schematic (Sheet 1 of 2)

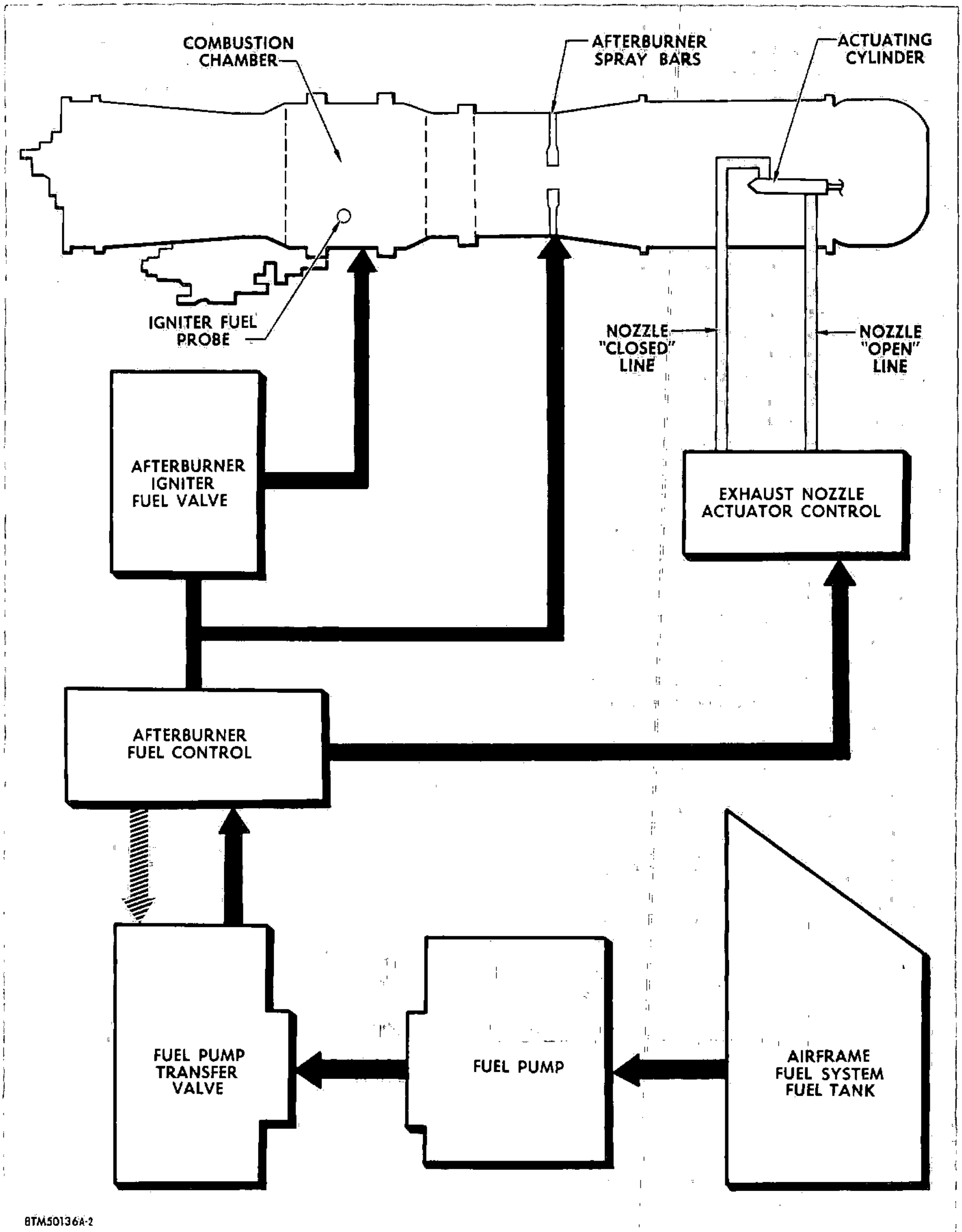


Figure 2-9. J57 Engine Afterburner Fuel System—Schematic (Sheet 2 of 2)

components are covered in the following paragraphs. By referring to the schematic of the system in figure 2-9, while reading the text, you will acquire a more comprehensive understanding of the afterburner fuel system.

Operation of the Afterburner Fuel System.

As mentioned earlier, in this section, the afterburner fuel system consists of the fuel pump/transfer valve, an afterburner fuel regulator, afterburner igniter valve, exhaust nozzle control valve, a manual shutoff valve, and the fuel spray bars and manifolds.

Afterburner fuel scheduling utilizes one gear stage of the engine-driven fuel pump. The pump outfit is either routed back to the gear stage input or flows to the afterburner fuel control. Note in the schematic that afterburner fuel scheduling depends on the position of the fuel regulating transfer valve and/or the motor operated fuel shuttle valve. During afterburning, fuel flows to the afterburner fuel control which then schedules fuel to the A/B spray bars according to the engine burner pressure. Fuel in excess of that required by the A/B system returns to the pump/transfer valve through the control pressure regulator return line.

The output of the afterburner fuel regulator flows to the 24 afterburner spray bars. A separate line from the afterburner igniter valve also connects into the spraybar supply line to supply fuel for "hot streak" ignition and to "trigger" the igniter valve. Fuel from the igniter valve is routed to the No. 3 burner can and the resulting momentary enrichment causes the flame to extend through the turbine wheel and back to the A/B spray bars. This flame ignites the fuel at the afterburner spray bars.

A small pressure sensing line connects the unmetered fuel side of the A/B fuel regulator to the anti-spring side of the exhaust nozzle control valve (this sensing line is shown in red). When A/B is selected, fuel pressure actuates the exhaust nozzle control valve which then directs sixteenth-stage compressor air to the *open* side of the nozzle actuating cylinders. When afterburner operation is stopped, the sensing line pressure is reduced and spring tension in the exhaust nozzle control valve repositions the shuttle valve. Then sixteenth-stage air pressure is ported to the nozzle *close* side of the actuating cylinders.

Normally, afterburner operation is initiated and terminated electrically. However, a mechanically-operated valve—attached to the power lever linkage—stops afterburner operation whenever the power lever setting is reduced below 80%. This provides a positive means of stopping afterburning in the event of an electrical failure in the afterburner circuit.

AFTERBURNER FUEL REGULATOR. The A/B fuel regulator, shown in figure 2-10, is located on the right side of the engine in the "wasp waist" section. Its purpose is to schedule fuel to the A/B spraybar manifold. This A/B fuel regulator control unit incorporates: a variable area-metering valve; a pressure regulator; an automatic shutoff valve; a burner can pressure sensing bellows; a pair of filters; and a drain valve.

Figure 2-9 shows the fuel flow through the regulator. Note that fuel entering the A/B fuel regulator control unit is routed to the metering valve—the excessive fuel is bypassed through the pressure regulator and routed back to the transfer valve. The metering valve orifice area varies in relation to the engine burner pressure. The bypass valve (pressure regulator) maintains a constant pressure drop across the metering valve by using spring and metered fuel pressure to resist unmetered fuel pressure. This arrangement is similar to the bypass valve in the engine fuel control unit. A check valve, having spring and bypass pressure resisting metered fuel, is located downstream from the metering valve. This valve serves a dual purpose: it keeps the A/B fuel regulator full; and prevents burner gases from entering the fuel regulator when the A/B is shut down. The burner can pressure-sensing bellows assembly in the A/B fuel control unit consists of a spring-loaded evacuated bellows linked to the metering poppet valve. Accordingly, burner pressure variation is reflected in the metering valve, and the metering valve then varies the A/B fuel with respect to engine operation.

Other components incorporated in the A/B fuel regulator are two filters and a manifold drain. The two filters provide filtered fuel to the sliding surfaces of the pressure regulator and shutoff valves. This filtered fuel provides a "washing action" which insures free movement. The A/B manifold drain valve is located in the regulator fuel outlet fitting and is slightly spring-loaded to the open position. During A/B operation, fuel pressure closes the valve; but when the A/B is shut down, the valve opens and drains the A/B fuel manifold. Normal engine operating pressures will also close the valve to prevent escape of hot gases.

AFTERBURNER FUEL IGNITER VALVE. As mentioned earlier, the ignition of A/B fuel is accomplished by the "hot streak" ignition method. The igniter valve is located high on the right side of the wasp waist section, as shown in figure 2-11. When A/B is initiated, the igniter valve directs a small charge of fuel into the No. 3 burner can. This enrichment results in a flame pattern that extends into the A/B area and ignites the fuel that is being discharged from the A/B spray bars. As shown in the schematic of figure 2-9, the igniter unit is made up of: an air pilot valve, a fuel pilot valve, and a spring-loaded air piston working in conjunction with a fuel piston.

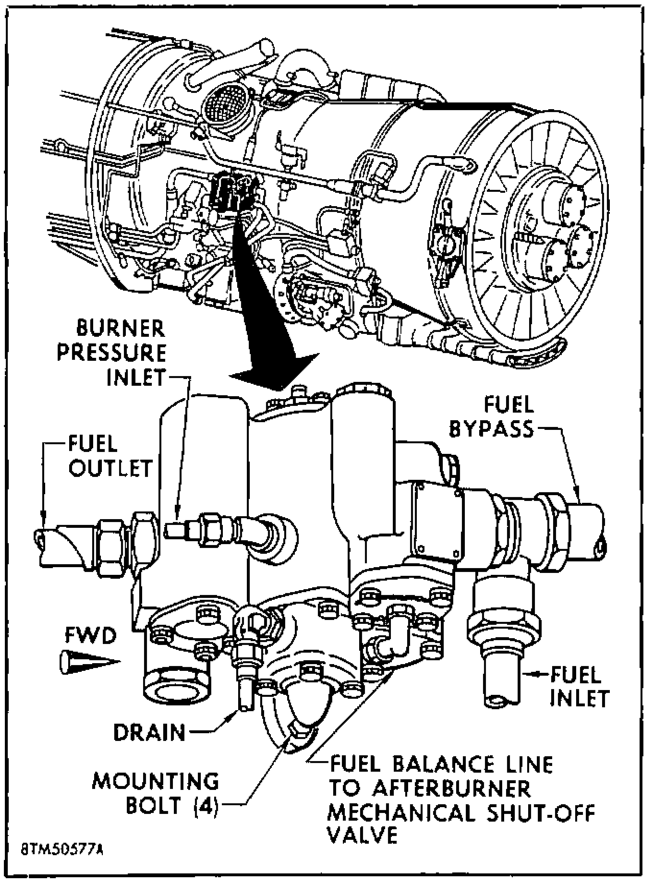


Figure 2-10. Afterburner Fuel Regulator

A line leading to the A/B igniter valve is connected into the A/B fuel manifold line downstream from the A/B fuel regulator. Fuel from this source passes through a filter and enters the igniter. During this charging action, fuel fills the fuel chamber, and at the same time, fuel is bleeding through a restricted passage into the fuel pilot valve. When the pressure becomes great enough, the fuel pilot valve moves the air pilot valve against burner pressure. When the fuel pilot valve moves, it shuts off the supply of incoming fuel and opens the fuel chamber outlet. The same movement causes the air pilot to open the burner pressure passage to the air piston. Burner pressure acting on the air piston compresses the air piston spring and forces the fuel behind the fuel piston into No. 3 burner can. The total afterburner ignition action takes approximately one-half second to complete and is a non-repeating action. Each component remains in the discharged position until the A/B operation is stopped. When A/B is discontinued, the fuel pressure—which was resisting engine burner pressure—drops off, and the burner pressure then moves the air pilot valve and fuel pilot valve back to the charge postion. This action shuts off burner pressure to the air piston, and the piston spring pressure returns the air piston back to its charged position. A check valve is installed in the afterburner igniter discharge line to prevent any hot gases from No. 3 burner can from entering the igniter valve during engine operation.

EXHAUST NOZZLE CONTROL VALVE. The A/B exhaust nozzle control valve, located just below and aft of the A/B fuel regulator, is shown in figure 2-12. Its function is to divert sixteenth-stage air pressure to either the OPEN or CLOSE side of the nozzle actuating cylinders. The afterburner nozzle is opened for afterburner operation and closed during non-afterburning operations. One of the key devices in the A/B nozzle actuating system is the air relay valve inside the exhaust nozzle control. A fuel pressure-sensing line connects the A/B fuel outlet passage in the pump/transfer valve to the anti-spring side of the air relay valve in the exhaust nozzle control valve. This pressure-sensing line is shown in red on the schematic of figure 2-9. On later airplanes, the sensing line source is from the A/B fuel regulator. This later configuration is not shown in figure 2-9. The air relay valve is positioned either by A/B fuel pressure or by spring pressure. During A/B operation the A/B fuel pressure moves the air relay valve against the spring pressure, and sixteenth-stage air pressure is then ported to the nozzle actuating piston open side of the nozzle actuating cylinders. Two spring-loaded flapper-type vent valves exhaust the discharge air from the relay valve. Normal fuel leakage

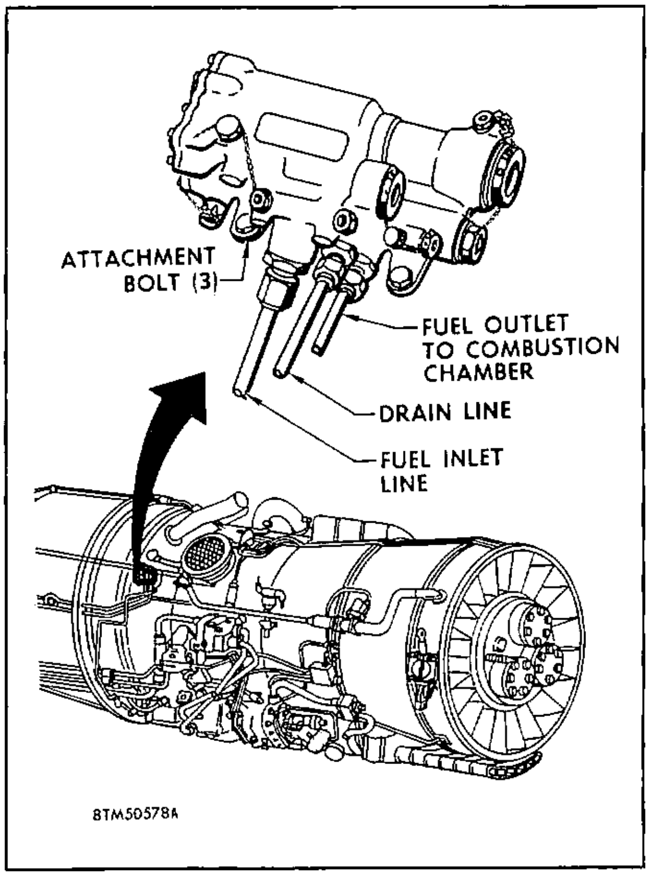


Figure 2-11. Afterburner Fuel Igniter Valve

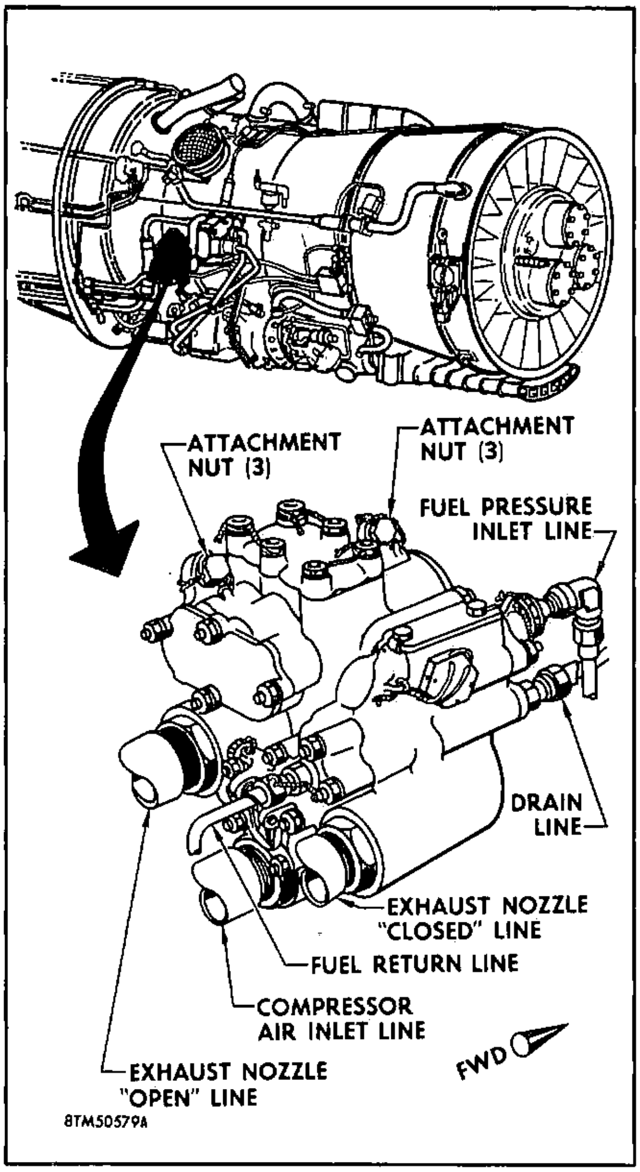


Figure 2-12. Exhaust Nozzle Control Valve

is drained overboard. The later engines are equipped with a differential-type exhaust nozzle control valve—the fuel section is separate from the air section. Refer to the maintenance technical order for information on these later-type control valves.

AFTERBURNER MECHANICAL SHUTOFF VALVE. Normally the A/B operation is stopped when the power lever in the cockpit is taken out of the A/B detent. This action energizes the A/B motor actuator which in turn closes the A/B fuel shuttle valve.

A fuel line that incorporates a mechanically-operated shutoff valve is installed in the A/B fuel system. This line (shown in yellow) at the left of the schematic, connects the spring side of the pressure regulator in the A/B fuel regulator to the fuel pump inlet. The shutoff valve is opened and closed mechanically by power lever movement. When the power lever is positioned for engine power outputs of 80% or more, the shutoff valve is closed and the A/B system functions normally. However, when the power lever setting is below 80%, the mechanical inter-connect between the shutoff valve and the power lever linkage opens the shutoff valve and A/B operation is then stopped. When the mechanically-operated shutoff valve is opened, the metered fuel that normally assists the pressure regulator spring to resist unmetered fuel pressure is routed back to the fuel pump inlet. This reduces the total pressure on the spring side of the pressure regulator and routes the fuel back to the pump/transfer valve. This stops A/B operation.

Maintenance on the Afterburner Fuel System.

Performing the necessary maintenance and trouble shooting on the afterburner fuel system will be quite similar to what you must do on the engine main fuel system. With the exception of the system filters, no regular maintenance is required. Trouble shooting will consist of determining why the exhaust nozzles fail to open or close or why afterburning does not occur.

Malfunctions in the afterburner nozzle system should be traced from the nozzle actuator control unit back to the nozzle actuators. If replacing the actuator control unit does not remedy the situation, the difficulty will lie in the associate air lines, in the relay valve, or in the nozzle actuators and their linkage. If the trouble is in the actuators and their linkage, you will find it best to lubricate all of the components using a commercially available graphite in a volatile liquid. This type of lubrication will reduce the drag and allow the actuators to move more freely. In case the air lines or the air relay valve are at fault, you will have to replace the components with serviceable items.

Cleaning or replacing the filters in the afterburner fuel system is just about the same as for the engine fuel system. Note in figure 2-13 that there are five filters in the afterburner fuel system. With the exception of the filter in the fuel control sensing line, all of the filter cavities must be cleaned with naphtha after the filters have been removed—P-S-661 is quite satisfactory for this purpose. The filters for the igniter control, the nozzle actuator control, and the afterburner fuel control units are not reusable—*new filters must be installed.*

THE J57 ENGINE OIL SYSTEM.

The J57 engine oil system is a dry-sump, self-contained, high-pressure system which supplies lubrication to the engine main bearings and accessory drives. This system is designed to use a synthetic lubricating oil — Specification MIL-L-7808.

Figure 2-14 shows the engine oil system in schematic form. Note how the oil from the oil tank is supplied

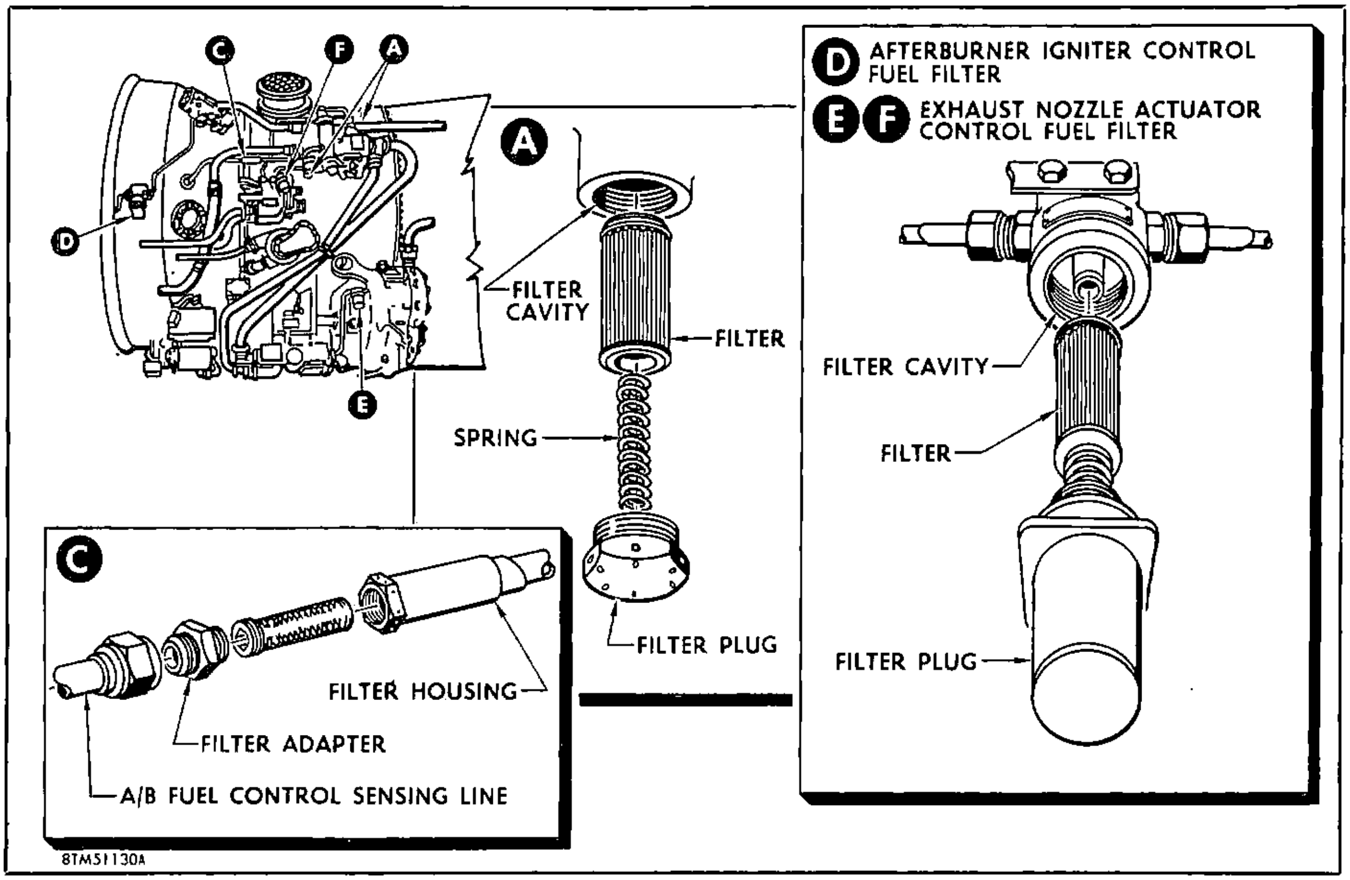


Figure 2-13. Fuel Filters in the Afterburner Fuel System

to the engine oil pressure pump by gravity. The oil forced from the pump is routed through the engine main pressure oil filter. This oil filter is equipped with a bypass valve which permits the oil to bypass in the event of filter clogging. From the filter, the oil is routed to the combination oil pressure regulator and relief valve. This component regulates the oil pressure at about 45 psi, which is the proper pressure differential for the oil metering jets at the engine bearings. From the regulator-relief valve the oil is routed to the bearings by means of external tubing and internal oil passages.

The oil from the bearings is picked up from the engine oil sumps by six scavenge pumps. Following the oil flow in the schematic, figure 2-14, note that the scavenge pumps force the oil through the air/oil cooler, then through the fuel/oil cooler. If the temperature of the scavenged oil is below 160°F, a thermostatic valve mounted on the fuel/oil cooler opens and routes the oil around the fuel/oil cooler and back to the engine oil tank. If the temperature of the oil is above 160°F, however, the thermostatic valve begins to close. At 186°F the valve is in the fully-closed position and all scavenged oil is routed through the fuel/oil cooler.

In the top of the oil tank, the returning oil passes through a de-aerator which removes the trapped air from the engine oil. This air is removed from the tank by means of the oil breather system. The engine oil system also provides lubrication for the engine-mounted Sunstrand constant-speed drive unit. This drive unit uses its own pressure pump, but the oil is taken from the engine oil tank and returned to the oil tank through a separate filter.

The oil tank is contoured to the engine diameter, has a 5.5 US gallon capacity, and is mounted on the upper left "wasp waist" area. The tank has a conventional filler cap for servicing and a cable-type dip stick for measuring oil quantity. Although the tank holds 5.5 gallons, only about 3 gallons are usable. One gallon is for reserve and about 1.5 gallons are contained within the flow lines in the oil system. The tank also has a 1.6-gallon area to allow for expansion during high engine temperatures.

HOW THE ENGINE OIL BREATHER SYSTEM IS PRESSURIZED.

The oil breather pressurization system is provided to insure proper oil flow away from the engine main bearings when the engine is operating at high altitudes. Note in the schematic of the engine oil pressure system, figure 2-14, that breather pressure *is* shown in dark color. This pressurizing air is supplied

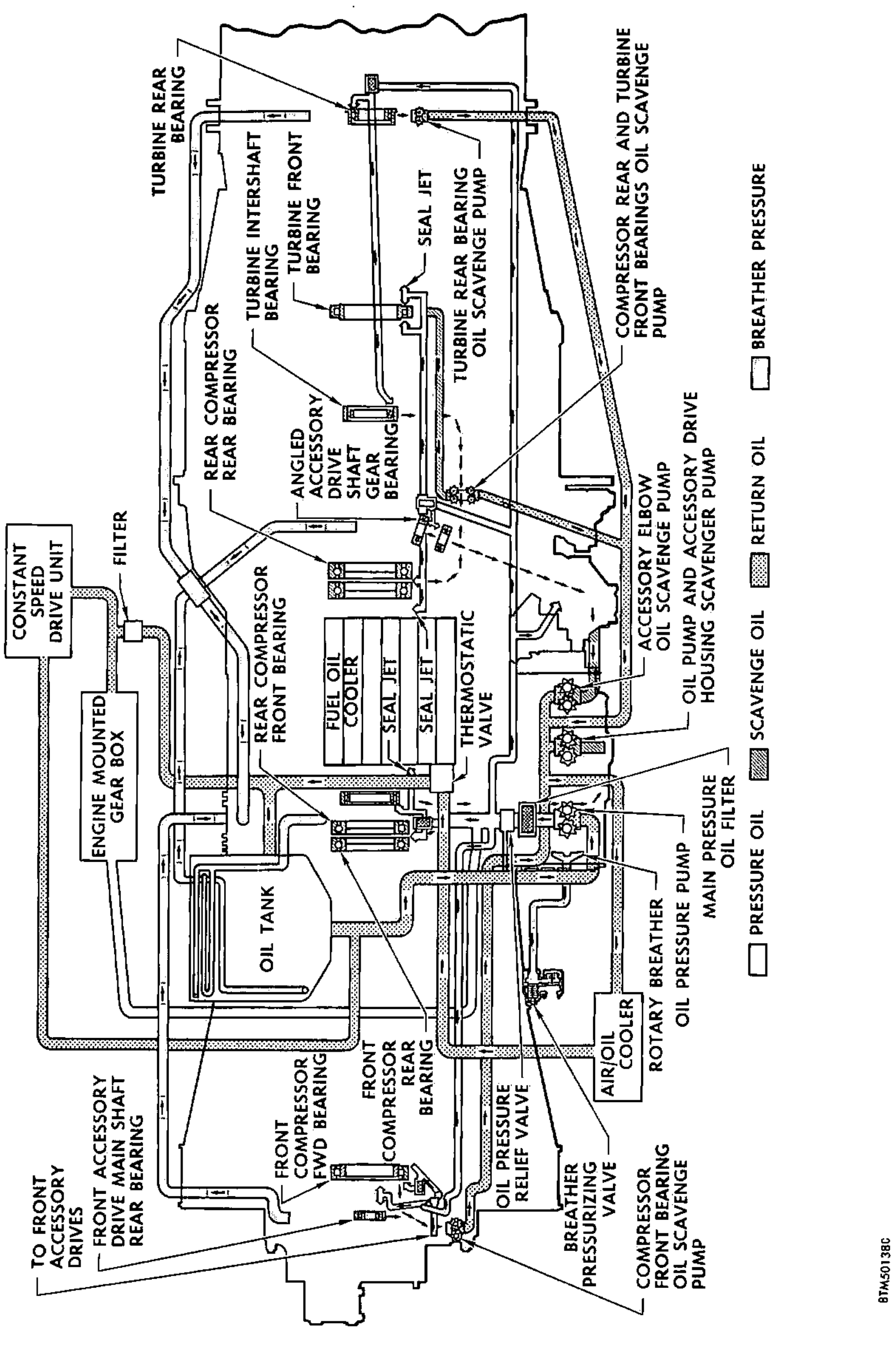


Figure 2-14. The J57 Engine Oil System Schematic

by engine compressor leakage, the compressor air leaks past the compressor seal into the inner case of the engine. Internal passages and tubing on top of the engine connect all of the bearing compartments, the oil tank, and the annular passage around the compressor into the breather system. This assures that all of the oil system will be supplied with the same breather pressure.

The rotary breather, shown in schematic adjacent to the oil pressure pump, removes oil from the pressurizing air by means of centrifugal action. As the pressurizing air passes over the impeller of the rotary breather, the oil is thrown out radially and the relatively oil-free air passes on to the hub of the breather. As the air reaches the hub, it is routed to the breather pressurizing valve which determines how much pressure will be maintained in the breather system. You should understand that the engine compressor air is leaking into the engine case at all times. The breather pressurization valve vents this compressor air to the atmosphere. At sea level operation the pressurization valve is open; however, with increasing altitude the valve gradually closes and regulates the escape of the engine compressor air so that the breather pressure is similar to that at sea level.

HOW THE OIL SCAVENGING SYSTEM OPERATES.

Six gear-type scavenge pumps are used to scavenge the main bearing compartments and the accessory drives. The pumps then force the oil to the oil coolers before it is returned to the tank. One scavenge pump is located in the lower part of the front accessory section; two are located in the right side of the main accessory section; two are located in the lower part of the No. 4 bearing area (rear compressor-rear bearing); and the sixth scavenge pump is located in the No. 6 bearing area (turbine rear bearing). Referring again to the schematic of the engine oil system, figure 2-14, you will note that all of these pumps discharge into one outlet line that runs to the air/oil cooler.

THE OIL LOW-PRESSURE WARNING SYSTEM.

The oil low-pressure warning system notifies the pilot whenever the oil pressure drops below a safe operating level. The warning system consists of a pressure switch, warning lights, and the associate electrical wiring. The pressure switch is mounted on the left side of the engine just aft of the oil tank. This switch is set to actuate, or close, whenever the oil pressure drops to 36 psi or below; if the fuel pressure rises to about 40 psi or more, the switch will deactivate, or open. The pressure switch measures the discharge pressure of the oil pump. The inside of the pressure switch is vented to the accessory drive housing. By comparing the oil pump discharge pressure with the oil system breather pressure, the pressure switch senses differential pressure regardless of altitude.

The warning light for this system is located on the pilot's right hand auxiliary instrument panel. The warning light and the electrical circuit for the oil low-pressure warning system will be discussed in detail in Chapter IV of this supplement.

SUNDSTRAND CONSTANT-SPEED DRIVE OIL SYSTEM.

All oil used for operation and lubrication of the Sundstrand constant-speed drive unit is supplied by the engine oil system. As you will recall from the discussion in Chapter I, the complete drive consists of an engine-mounted gear box and an airframe-mounted transmission and gear box assembly. The engine-mounted gear box receives its oil supply from a line connected into the engine oil pressure switch line. The fuselage-mounted gear box/transmission incorporates a pressure pump and receives its oil directly from the engine oil tank. Scavenging oil lines from both units come togther and return to the engine oil tank through a filter. There are breather lines which connect the constant-speed drive gear box/transmission with the engine breather system. By following the oil flow, as shown in figure 2-15, you will better understand the following description of the system. Oil from the engine oil tank flows to the two charge pumps. This flow is shown by the red diagonals. The smaller of the two pumps is directly driven by the airframe-mounted transmission and gear box assembly input, pumping oil at a rate varying with this input to about 2.75 gallons per minute. Oil from these charge pumps moves to the transmission cylinders by way of a filter which has a bypass valve that opens at a pressure differential of approximately 50 psi, should the filter become clogged. The oil moves into the cylinder block by way of a drilled passage in the manifold and eccentric shaft (6). Inside the transmission itself, oil is pumped from the pump cylinders (1) to the motor cylinders (2) (or from motor to pump depending upon the phase of transmission operation). The oil then moves out of the cylinder block by way of the manifold (6). This oil is maintained at a pressure of approximately 250 to 350 psi by the charge relief valve whi ports surplus oil to the lubrication lines. Another valve—the lubrication relief valve—maintains this surplus oil pressure at approximately 15 to 30 psi. The excess oil from the lubrication relief valve is ported to the transmission.

Appropriately-located jets direct this oil pressure on moving parts. Leakage or blow-by oil from the pistons also drains into the sump, where it is scavenged by another gear-type pump which returns the oil to the aircraft engine oil system.

The oil system for the engine-mounted gear box is quite simple in respect to that of the other component of the Sundstrand constant-speed drive. As shown in the upper right portion of figure 2-15, the oil for

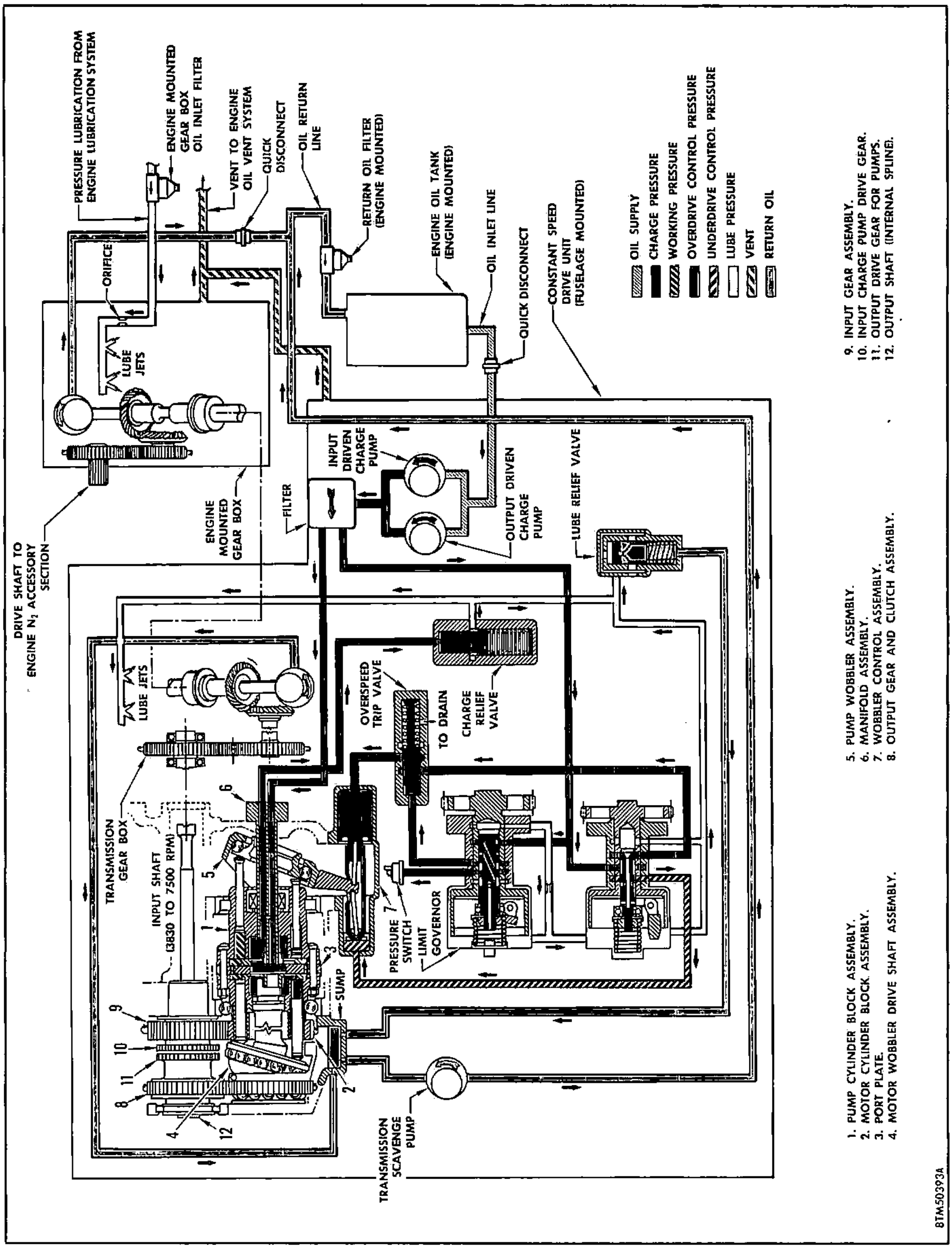


Figure 2-15. Constant-Speed Drive Unit Oil System

pressure lubrication of the engine-mounted unit is obtained from the engine lubrication system. The oil then passes out the outlet line where it joins the return oil from the transmission and gear box assembly oil to flow back to the engine oil system.

THE ENGINE STARTING AND IGNITION SYSTEM.

The engine starting and ignition systems are closely related to each other. Each system is dependent upon the other for completion of electrical circuits and for operation of the system. For this reason, the two systems will be described together in this section.

THE ENGINE STARTER.

The J57 engine uses a pneumatic starter. This starter, which weighs about 35 pounds, is mounted on the forward face of the engine N_2 accessory section. The starter is similar to an axial flow turbine and is actuated by compressed air from a gas turbine compressor (GTC). The flexible hose from the GTC unit is connected to the starter at the quick-disconnect unit in the right wheel well. The air from the GTC unit does not enter the starter until the pilot positions the power lever in the START position. Movement of the power lever actuates a microswitch in the quadrant, and the microswitch completes the circuit to the starter air regulator valve. As this starter valve opens, the air from the GTC unit strikes the turbine blades and exhausts through the starter air exit duct. The force of the air against the turbine blades causes the starter turbine wheel to turn, and the turbine, by means of a gear reduction train and output shaft, drives the engine. When engine starting speed is attained, the engine-mounted *starter* gear reduction train is disengaged from the engine by means of an integral clutch. A centrifugal action switch in the starter causes the air regulator valve to close when engine speed reaches 3400 to 3700 rpm. This action shuts off the air supply to the starter.

The lubrication of the engine starter is self-contained —the lubricating capacity is 12 fluid ounces. Normally, the fluid level should not have to be checked except when there is indication of oil leakage; however, the system has to be drained and refilled at intervals of 25-hours engine operating time.

ENGINE IGNITION.

As mentioned earlier, the ignition and starter circuits are dependent upon each other for successful engine ignition and starting. This can be seen more clearly by referring to the schematic of the engine starting and ignition system, (see figure 2-16). Let us trace the sequence of actions and current flow through the schematic for a normal engine start.

First, locate each of the eleven components in the schematic. The starting action is controlled through the power lever movement and the ignition button. The starter switch (5) (microswitch) is shown in the de-activated postion such as would be the case when the power lever is in the OFF position. As you will recall, the first step in starting the engine is to move the power lever outboard to the START position—this closes the starter switch. Current then flows from the 28-volt d-c essential bus, numbered 4 in the illustration, to the ignition switch and also to the engine starter, number 7. As current reaches the engine starter, the starter air regulator valve opens and compressed air from the ground GTC drives the starter.

As the engine speed reaches 12 to 16%, the operator depresses the ignition button. This action closes the ignition switch and energizes the starter relay. As already mentioned, depressing the ignition button does not cause ignition—it just arms the starter relay. This type of arrangement allows the operator to move the power lever forward without stopping the starter. When the power lever is moved forward towards the IDLE position, the starter switch is deactuated and fuel starts metering into the engine combustion chambers. Note in figure 2-16 that as the starter switch is deactuated, current is fed to the ignition power relay, numbered 1. With the engine turning at 12 to 16%, and with fuel being metered to the chambers, the engine starting operation is ready for ignition. As the ignition power relay is energized, current is fed through the engine junction box (11) to the ignition transformer (8). There are two of these transformers, however, only the No. 2 transformer is shown in the schematic. The other transformer, No. 1, is exactly like the one shown except that it supplies ignition to the No. 5 combustion chamber instead of the No. 4 chamber.

When the current first enters the ignition transformer, it passes through several input filters (choke coils and capacitors). Then it is led to a 6000 rpm motor. Note that the same current that operates the motor is tapped off to a cam-type switch. The cam, geared to the motor, turns at about 1200 rpm. This cam has the effect of "chopping up" the direct current input and turning it into pulsating d-c. This pulsating d-c then flows to the step-up transformer where the 24 volts are stepped-up to about 2000 volts. You will note that this high voltage then flows through the selenium rectifier which converts the pulsating current back to d-c. This rectification is necessary so that the ignition capacitor can be charged. In the figure 2-16 this ignition capacitor is shown directly below the second motor-driven cam. At the same time the ignition capacitor is being charged, the second cam is turning at 240 rpm. Each time this cam closes its ignition point, a pulse of direct current will flow through the "triggering" transformer to the ignition capacitor. This pulse of direct current is sufficient to spark across the igniter plug gap. As it sparks across the gap, it provides a "path" for the charge in the ignition

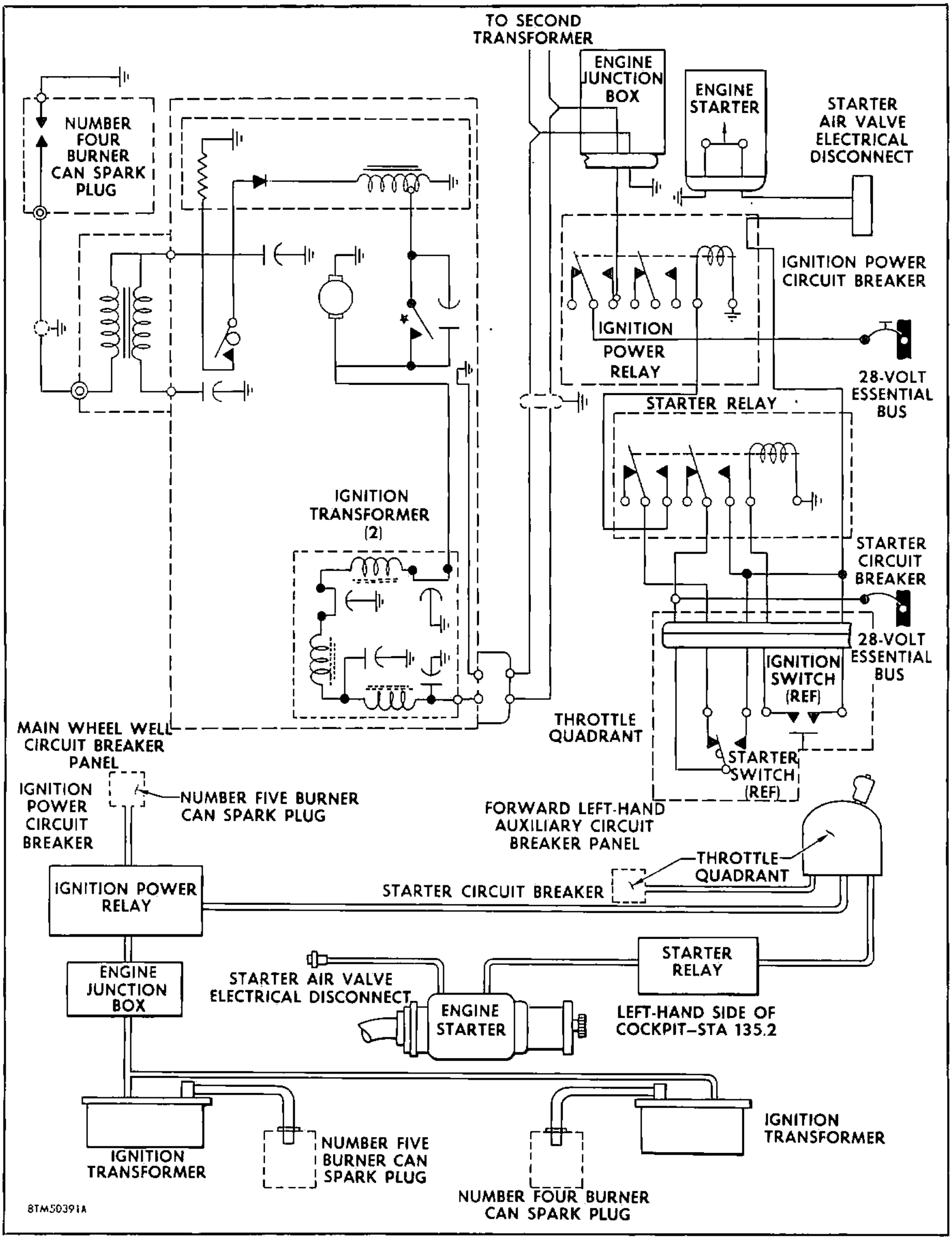


Figure 2-16. Engine Starter and Ignition Electrical System Schematic

capacitor. With an open "path" of small resistance, the charge on the capacitor flashes through the "triggering" transformer and across the igniter plug gap.

Summarizing the engine ignition circuit, you should retain several important facts. The power lever movement and the ignition button provide the only pilot control for the engine ignition system. By means of two relays—the starter relay and ignition power relay—and the two ignition transformers, this ignition system provides "hot" spark ignition for the No. 4 and 5 combustion chambers. You should also keep in mind that there is no timing device in the ignition circuit. Because of this fact, you should never keep the ignition button depressed longer than 30 seconds after the power lever has been advanced to IDLE. This 30-second time limit is necessary to prevent over-heating of the ignition transformers.

This electrical system, like the rest of the engine electrical systems, is covered completely in the F-102A maintenance technical order for the airplane electrical systems, T.O. 1F-102A-2-10.

HOW THE ENGINE IS STARTED AND OPERATED.

During the engine starting procedure, care must be exercised to correctly operate the starter, ignition, and power lever controls to successfully complete the engine start. Actuation of the engine starter is initiated by the outboard movement of the power lever to the START position. The power lever is spring-loaded to the OFF position and must be held over in the START position. The power lever movement operates a microswitch in the quadrant which opens the starter air valve. Opening the air valve permits compressed air from the gas turbine compressor to actuate the starter. The power lever is held in the START position until the starter is turning the engine from 12 to 16% rpm. Depressing and holding the ignition button energizes the starter holding relay. Ignition does not occur at this time, but it will permit the starter to continue cranking the engine while you move the power lever—with the ignition button still depressed—forward from the START position. This movement of the lever initiates ignition through the actuation of the microswitch in the quadrant. Operation of the engine starter will continue as long as the ignition button is depressed, or until the starter centrifugal switches actuate due to starter overspeed. Moving the power lever forward towards the IDLE position actuates the main fuel control. This permits fuel to be injected into the engine burner section. Ignition is supplied to the igniter plugs in the No. 4 and 5 burner cans. As the fuel is ignited, the engine rpm increases. The ignition button should be held down until the tachometer and temperature gage indicate a positive start, then the button should be released. If you should momentarily release the ignition button when the power lever is between OFF and IDLE and the engine does not start, return the power lever to the OFF position. This action will prevent additional fuel from being injected into the engine burner section. You must then repeat the complete starting cycle from the beginning. After returning the power lever to the OFF position, do not attempt another start until fuel drainage from the engine combustion chambers drain has stopped. Never operate the starter more than 90 seconds during any two minute period. After a normal engine start has been completed, the following procedure is recommended for checking the engine operation: set the pressure ratio indicator for the correct ambient air temperature, and then advance the power lever to the MIL POWER postion and allow all instrument readings to stabilize.

CAUTION

Do not operate the engine at MIL POWER any longer than five minutes.

While the power lever is in the MIL POWER range, check the tailpipe temperature for a maximum of 610°C. The indicator needle on the pressure ratio gage should fall within the maximum power range marked on the gage. None of the engine warning lights should be burning. Then move the power lever to AFTERBURNER.

CAUTION

Do not operate the afterburner longer than one minute.

A rapid increase in tailpipe temperature and an rpm reduction of about 4% usually mean that the exhaust nozzle has failed to open. When this occurs, terminate the afterburning operation immediately. Then, opening the afterburner power circuit breaker, move the power lever to FULL MIL POWER and then retard it—without hesitation—to some point below 80% power. Afterburning should cease at this 80% power. There should be no indication of engine roughness as the power lever is being retarded before afterburning is terminated. Return the power lever to IDLE, and allow an afterburner drainage period of two minutes.

When the preceding operations have been accomplished, actuate the emergency fuel control switch from NORMAL to EMERGENCY. The emergency fuel warning light should illuminate; then advance the power lever to MIL POWER. Fuel flow should be 6350 to 6950 pounds per hour at sea level. Returning the power lever to IDLE, note that normal engine operation continues throughout power lever range. Then, actuate the emergency fuel switch to NORMAL—the warning light should go out.

HOW TO STOP THE ENGINE.

Prior to stopping the engine, you should allow it to idle for approximately 5 minutes. In an emergency, however, the engine may be shut down immediately. Have a ground crew assistant connect a ground compressor, which is operating at maximum output, to the airplane ground receptacle. Connect a-c and d-c power to the airplane ground receptacles. Turn all fuel boost pump switches OFF, and retard the power lever to the OFF position. In the event that exhaust temperature rises above 225°C after engine shut down, actuate the power lever to the START position and crank the engine for about 20 seconds. Then, return the power lever to the OFF position.

ENGINE AIR INDUCTION AND COOLING.

Although the cooling system for the gas turbine engine is not as complex as that required for piston-type engines, considerably more effort must be spent in protecting the surrounding aircraft structure from the heat. This is usually accomplished by a shroud installed around the outside of the engine with cooling air directed between the shroud and the outside of the engine. Some engines require cooling for accessory drives, turbine wheels, bearing housings, and fuel pumps. Other engine installations require a method of cooling the oil. Gas turbine engines usually use any of three cooling mediums: air, oil, or fuel. Air is by far the most commonly used method of the three.

Air cooling must be accomplished by creating high air flow and pressure with a minimum loss from drag, turbulence, and other detrimental factors. There are four methods by which air cooling can be accomplished: induced air, ram air, auxiliary air (fan), and engine compressor bleed. The induced air method is the most commonly used, while the compressor bleed system is rarely used. The turbine wheels, bearing housing, oil coolers, and tailpipe usually use one of the above cooling methods. The tailpipe shroud is sometimes kept cool by an insulating blanket installed between the engine and the shroud. The burner cans are usually kept cool by internal air flow which is actually a part of the combustion system. Fuel pumps are almost always cooled by the fuel being pumped through them. Fuel as a cooling agent has been used in fuel/oil heat exchangers, but not to any great extent. Oil is used for cooling such components as accessory drives, bearings, and reduction gears—the oil itself being cooled by an oil cooler similar to that used on reciprocating engine installations. The cooling system, as a rule, requires no pilot attention or operation. The oil pressure gage in the cockpit is the only connection between the pilot and the cooling system. The oil pressure gage could indicate any serious over-temperature condition in a part being cooled or lubricated by oil.

Some of the earlier gas turbine engine main rotor bearing temperatures were very critical. As a result, bearing temperature gages were required. Since that time, engineering developments have solved the bearing problem. In all probability, bearing temperature gages will not be encountered in late model and future gas turbine engines.

IN-FLIGHT COOLING OF THE ENGINE COMPARTMENT.

In the F-102A, the engine compartment cooling airflow pattern varies between engine ground operation and flight operation conditions. Air flow during flight is based on a positive pressure condition existing in the engine intake ducts; while engine cooling during ground operation is dependent upon air bled from the N_1 rotor and reverse air flow through the air/oil cooler and generator cooling ducts.

Note in figure 2-17 that there are three sources of air and that each source is located in the engine intake duct. One source is bled from a scroll, numbered 2, located on the forward end of the engine stub duct. Part of the air taken from this scroll is passed through the air/oil cooler (13) and the remainder is ducted aft under the shroud (10) that surrounds the engine hot section. That part of the air which passes through the air/oil cooler is exhausted into the accessory section and flows aft between the shroud and the fuselage. The cooling air, which is ducted aft, flows upstream from the air/oil cooler and runs aft along the left side of the engine to the forward section of the engine shroud. Cooling air entering the shroud is diffused circumferentially by the distribution section and flows aft between the shroud and the engine exhaust nozzle.

GROUND COOLING OF THE ENGINE COMPARTMENT.

First of all, in-flight cooling below 150 knots IAS with the landing gear extended and ground cooling are the same thing; in other words, the air flow about the engine is identical in both cases. This type of cooling will be referred to as ground cooling in the text.

Cooling during ground operation varies substantially from that described in the preceding paragraphs. J57 engines installed in the F-102A airplanes incorporate a top center-line bleed which furnishes ninth stage N_1 air to cool the hot section of the engine. An electrically operated shutoff valve, a check valve, and two series-mounted flapper valves permit rescheduling of air flow for ground cooling and cooling at airspeeds below 150 knots with the gear extended.

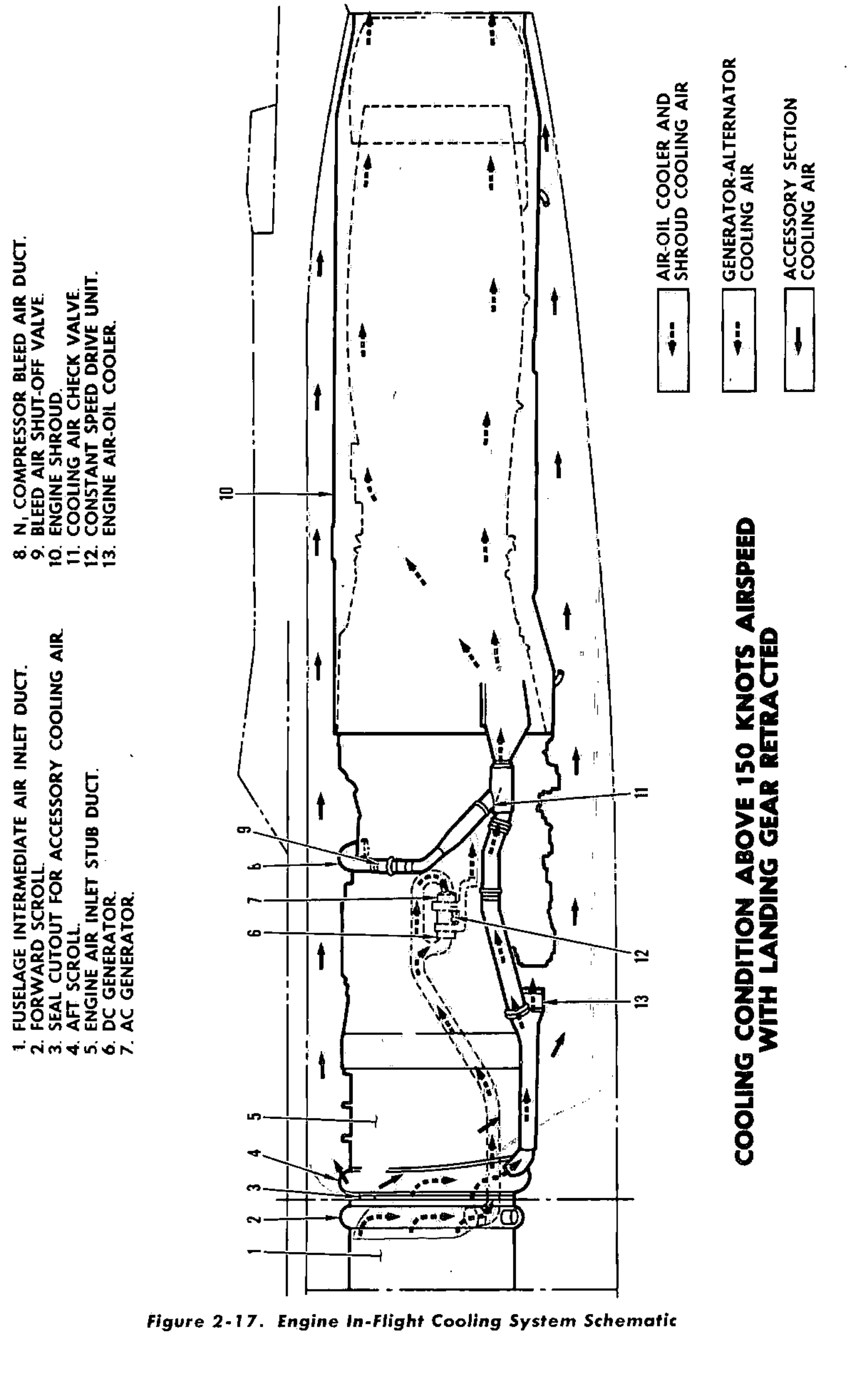


Figure 2-17. Engine In-Flight Cooling System Schematic

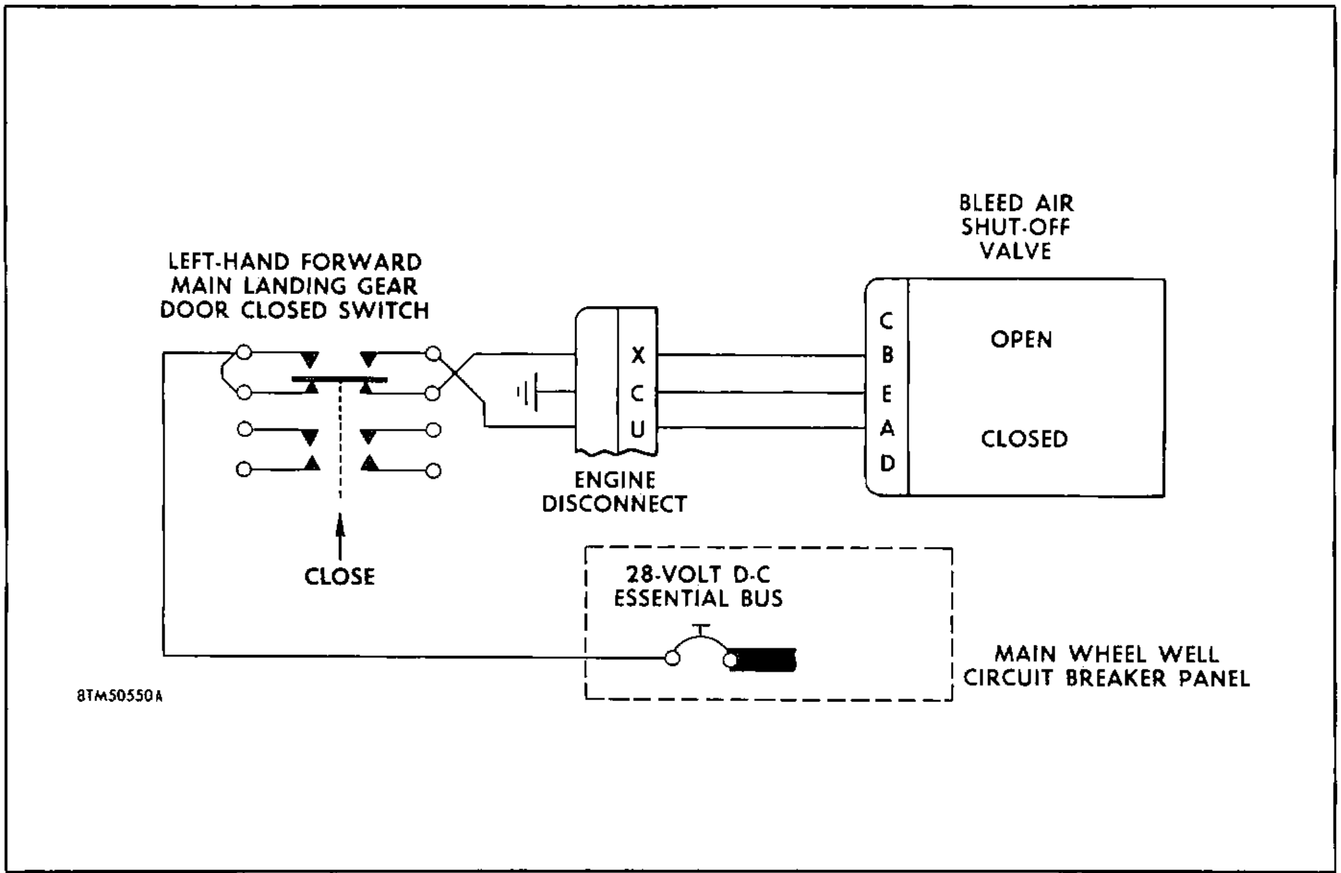


Figure 2-18. Engine Cooling Electrical Schematic

Notice in figure 2-18 that the electrical circuit for the shutoff valve is wired in series with the left landing gear door—thereby opening the valve when the gear is retracted. This is understandable, since higher airspeeds are associated with a clean airplane (gear up and locked); at these high airspeeds no help in cooling is needed from the top center N_1 compressor bleed system. The check valve (11) located in the cooling duct and connecting the stub duct scroll to the shroud, prevents N_1 compressor air from flowing forward into the scroll.

During ground run a partial vacuum exists in the engine air intake ducts, this vacuum causes a flow reversal in tthe cooling ducts. Cooling for the engine oil and the generators is also provided by this reverse flow during ground operation. The reverse air flow through the air/oil cooler is easily accomplished without check or flapper valves; however, the generator cooling operation is more complex. Generator cooling, both in the air and on the ground, is covered in the next section of this chapter.

When an airplane is on the ground and the engine is running, the cooling air bleed shutoff valve (9) is open. Note in figure 2-19 that this allows ninth-stage air pressure to flow down the duct on the left side of the engine and force the check valve (11) to open. This compressor air then flows under the shroud surrounding the engine. The air is exhausted around the exhaust nozzle between the engine and the shroud.

As mentioned, air flow reversal also takes place when the airplane is on the ground. This air flow reversal is caused by the low pressure condition in the intake ducts. Air is actually drawn in the aft end of the airplane between the engine shroud and the aircraft structure. This air is drawn forward through the air/oil cooler, and through the two previously mentioned slots between the scrolls. The air is then pulled back through the engine for the combustion process.

IN-FLIGHT COOLING OF THE A-C AND D-C GENERATORS.

The intermediate duct scroll takeoff is located in approximately the 4 o'clock postion. Note in figure 2-20 that the cooling duct from this takeoff runs aft down the right side of the engine to the a-c and d-c generators. Cooling air simultaneously enters the aft end of the a-c generator and the forward end of the d-c generator. Both ducts come together downstream of the generators and at this point there are two opposite-acting flapper valves. Exit air closes the overboard flapper valve and opens the other flapper valve to the accessory section. Under these conditions exit air from

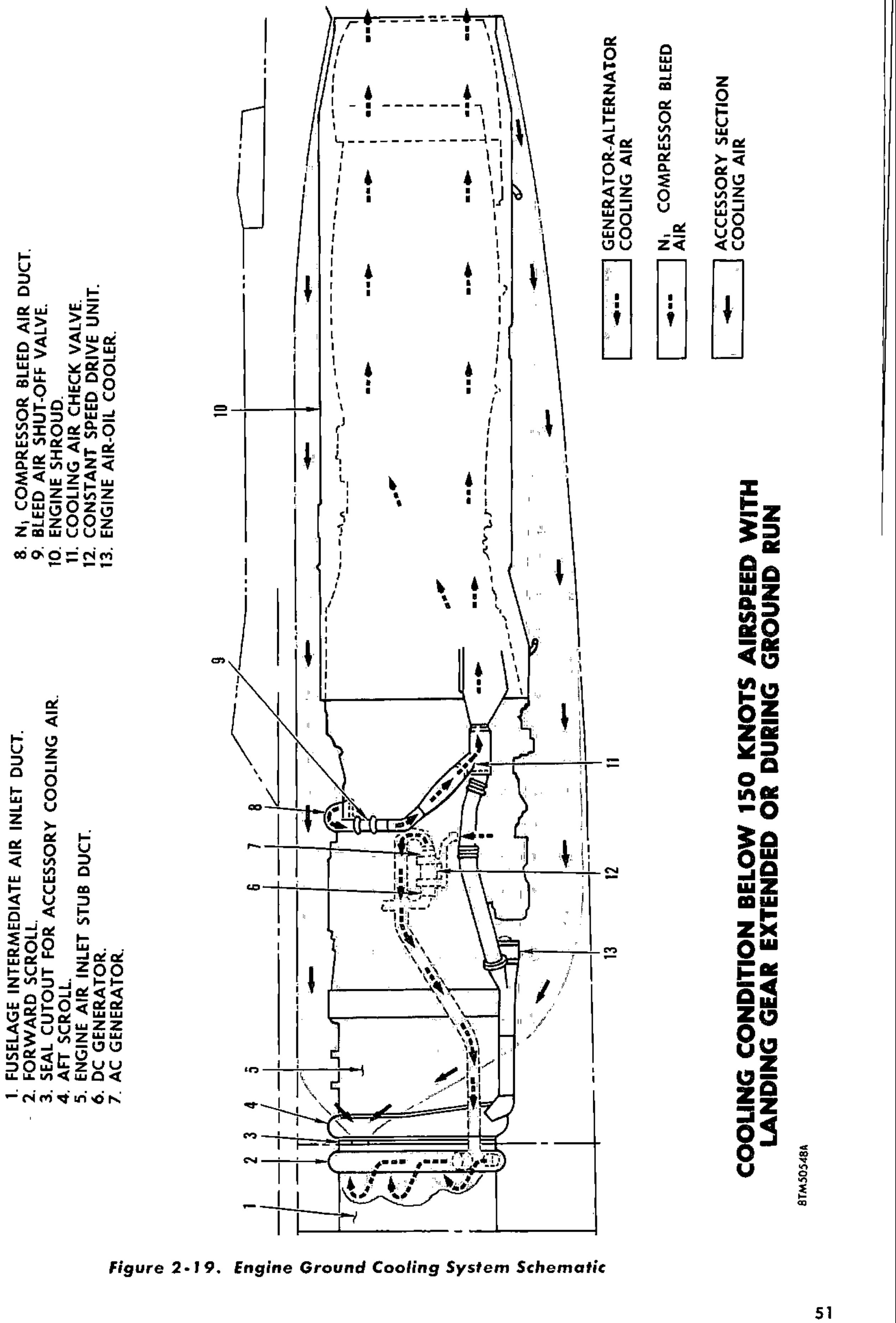
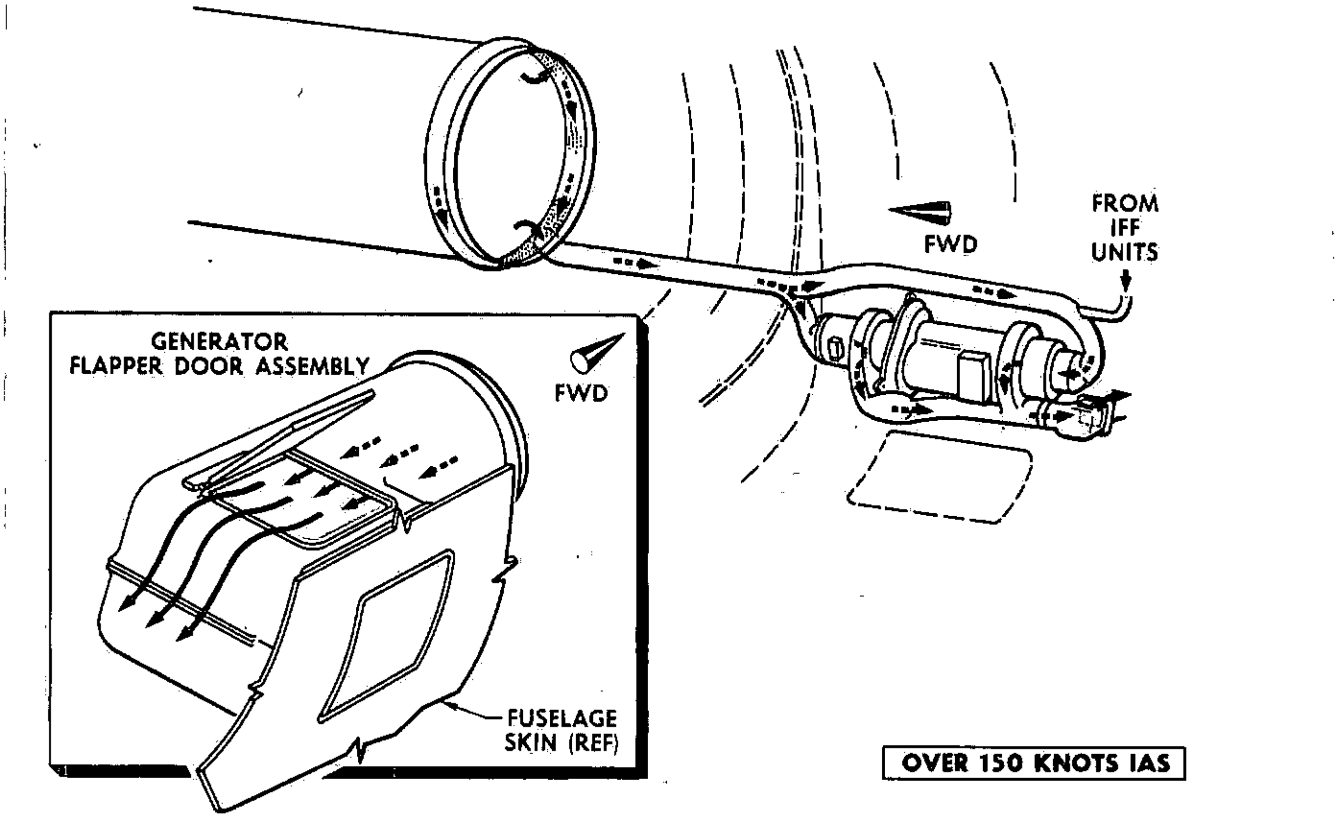


Figure 2-19. Engine Ground Cooling System Schematic



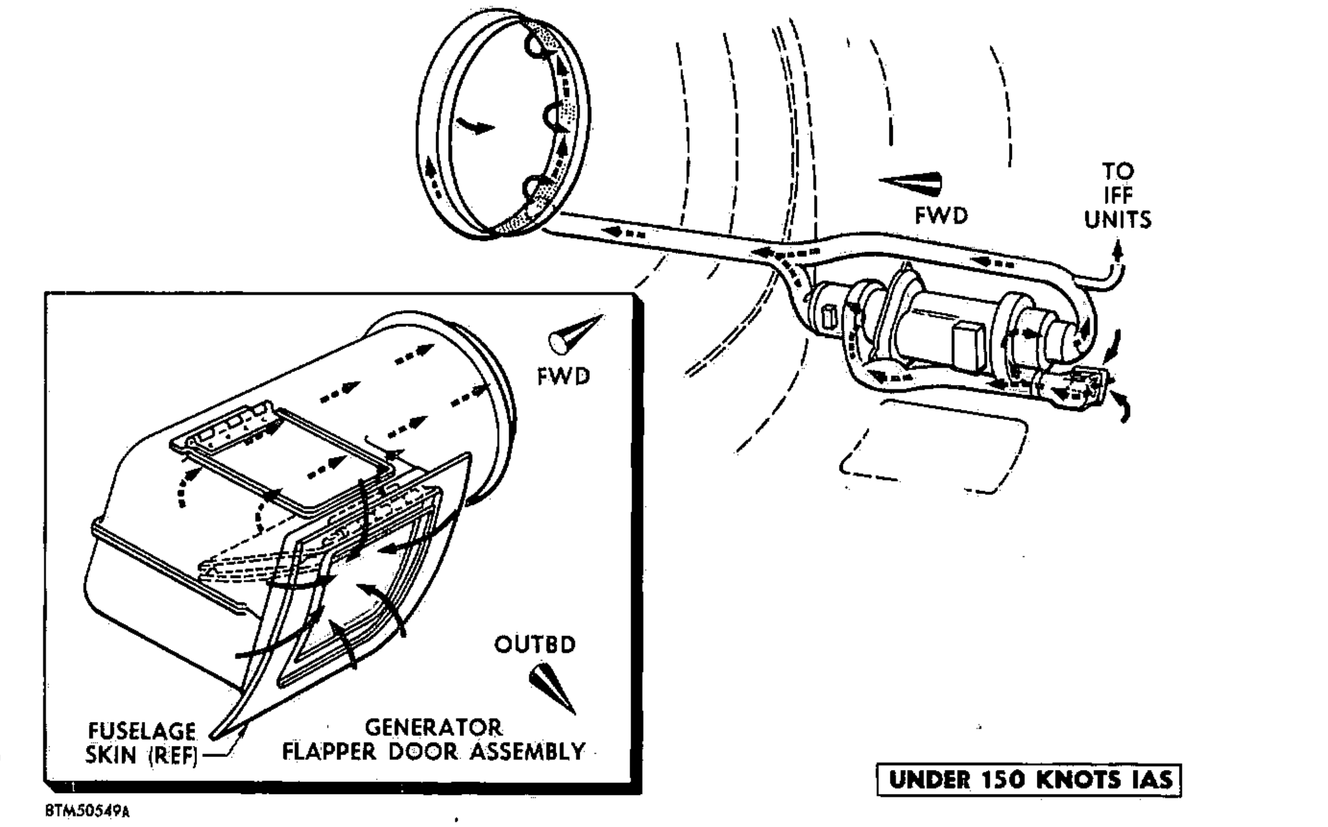


Figure 2-20. Generator Cooling System Schematic

both of the generators is further utilized for cooling the accessory section. The air continues aft and exits between the afterburner shroud and the tailpipe structure.

GROUND COOLING OF THE GENERATORS.

Generator cooling under low airspeed conditions (which is the same as ground operation) is accomplished on the reverse flow principle, as shown below. This reverse air flow opens the ground run door on the airplane skin and closes the accessory section flapper valve. In this manner, cooling air is drawn in from the outside and flows forward through the duct scroll. At this point the air is drawn into the compressor section and used for combustion purposes.

ENGINE ANTI-ICING SYSTEM.

Whenever icing conditions exist, ice will form not only on the airframe structure, but also in the engine inlet area. If ice formation is allowed to continue, it will seriously reduce the engine power output. In the F-102A airplane, provisions are made to prevent the ice accumulation on the inlet guide vanes and surrounding area by directing sixteenth-stage heated air through the hollow vanes and forward around the nose accessory fairing. This hot air is exhausted behind the nose accessory fairing cap and allowed to enter the engine intake. Airflow is controlled directly by two electrically-operated shutoff valves, and indirectly by two flow-control valves.

In figure 1-16 of Chapter I, the complete airflow pattern is shown. The electrical controls and the components for the engine anti-icing system will be discussed more fully in Chapter IV.

SUMMARY.

In Chapter II you have learned how the engine associate systems function to assure that the basic engine combustion cycle will produce its continual flow of power. These systems have included the fuel, oil, starting and ignition, induction and cooling, and the anti-icing systems. The knowledge of how these systems function and how each component plays its part in the overall function of each system will enable you to intelligently perform the required maintenace and trouble shooting on the J57 engine. Although, in some cases, specific values for pressures or temperatures were given, you should keep in mind that these were for explanatory purposes only. For the exact values, you should always refer to the maintenance technical order which covers the power plant installation in the F-102A, T.O. 1F-102A-2-4.

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Chapter III

ENGINE BUILD-UP AND HANDLING

Contents *Page*

The first two chapters in this supplement have acquainted you with the J57 engine, its components and major operating systems, and how the engine operates. These two chapters have explained *what* the engine does while it is installed in the airplane. As a power plant maintenance man, you should understand that you may have to accomplish the handling and maintenance requirements of the engine when it is *not* installed in the airplane. These requirements will consist of: preserving and depreserving the engine; building up the engine when it is first received from the manufacturer; and installing the engine in (and removing the engine from) the F-102A.

All of these requirements are covered in this chapter. After first learning some of the basic principles involved in the prevention of corrosion on engines, you will be familiarized with the preservation and depreservation requirements of the J57 engine. The major portion of the chapter is devoted to the engine build-up operation. In this discussion you will learn just what you must do to a new or overhauled engine before it can be installed in the F-102A. The last part of this chapter covers the engine installation and removal operations and the special tools which you will need to accomplish them.

Before proceeding with this chapter, there is one point which you should thoroughly understand. As mentioned earlier, both the J57-23 and the -41 engines are used in the F-102A. In some instances the maintenance requirements and techniques are not the same. In most of these cases the differences are mentioned in the text or shown in the illustrations. Before attempting any maintenance on either model engine, however, always determine the type of engine first and then check the maintenance technical order (T.O. 1F-102A-2-4), for the exact procedures and instructions for that particular model.

ENGINE PRESERVATION AND DEPRESERVATION.

The combat against corrosion of aircraft engines is primarily a fight against moisture. In the light of present-day technical knowledge, the chemical principles involved in metal corrosion are fairly well understood. It is generally accepted that there are two types of surface corrosion—the direct chemical attack on metals by corrosive liquids, and the electro-chemical attack in which the metal being corroded becomes a part of an electrolytic cell in the presence of moisture. Both types of corrosive action are effectively retarded by the *absence* of moisture. This fact has led to the use of sealed, metal containers to protect engines during shipment and extended storage periods.

THE USE OF JET ENGINE SHIPPING AND STORAGE CONTAINERS.

The metal shipping and storage container for a jet engine serves four important functions. First, it protects the engine from vibrations encountered in normal transit. Second, it protects the engine from shock encountered in abnormal or rough handling. Third, it protects the engine from the various atmospheric conditions encountered in world-wide distribution and storage. Fourth, the container permits the storage of an engine for greater lengths of time, without frequent inspection and represervation operations.

Because metal containers are sealed from the atmosphere, they require different sealing and dehydrating techniques from those used with wooden containers. For example, the use of a protective envelope over the engine inside the shipping container is unnecessary. It is also unnecessary to seal some of the engine openings, to prevent air from entering, or to install dehydrating plugs in other engine openings, to reduce the moisture content.

By using the sealed metal containers, the engine openings need only be covered with ventilation plugs which are designed to keep foreign matter out of the engine interior. Dehydration of the engine and the metal container is accomplished by placing a predetermined amount of bagged dehydrating agent in baskets inside the container and then pressurizing the engine container wth dehydrated air.

One of the most beneficial results of the use of metal containers is the limited attention which the stored engines demand. Adequate protection from the elements is afforded the engine by the container itself. The only maintenance requirements are the maintaining of the air pressure within the container, and the observing of the prescribed inspection and represervation periods.

EFFECTIVENESS OF ENGINE PRESERVATION.

The effectiveness of engine corrosion preventive procedures depends upon two factors: prompt and complete application of preservation procedures, and faithful compliance with the instructions for maintaining the preserved state of the engine during storage. As you can well understand, each of these factors is dependent upon the other. If you have not satisfactorily completed engine preservation in the first place, maintaining the preserved state of the engine will not do very much good. Conversely, if engine preservation was accomplished satisfactorily, improper maintenance of the stored engine, will also make the engine preservation ineffective.

You should understand that engine preservation maintenance is *not* represervation; but it is the regular inspection, and if required, replacement of the dehydrating agent. If the integrity of the moisture vapor barrier inside the container has not been maintained, or if the dehydrating agent has become saturated, an internal inspection of the engine is required. This inspection should reveal whether corrosion exists on the interior of the engine; whether overhaul or complete represervation is necessary; or whether renewal of the dehydrating agent will be sufficient.

Reports show that most of the corrosion of engines in storage is directly traceable to delay (permitting the onset of corrosion prior to preserving the engine); poor application of the required preservation procedures; or a lack of proper maintenance of preservation during extended periods of storage.

When the entire preservation process is properly carried out and adequately maintained, however, it has been proven that corrosion is not likely to occur while the engine is in storage.

Although the engine preservation process is completely covered in military specifications and technical orders, some of the more important points are mentioned in the following paragraphs.

Cleaning the engines and engine parts with chlorinated solvents, such as trichlorethylene and carbon tetrachloride, should be avoided whenever possible. Although chlorinated solvents are excellent cleaners, they tend to encourage corrosion on metal surfaces. Cleaning solvent, Federal Specification P-S-661, is preferred for hand cleaning the exterior engine surfaces to which corrosion preventative compounds are to be applied.

The proper installation of the dehydrating agent requires that this material be handled with considerable care and speed to allow a minimum of exposure time between removal of the agent from its package and installing the agent in the engine metal container. Partial saturation of the dehydrating agent (which may occur rapidly with exposure to outside air), reduces the effective life of the agent. Installation of the dehydrating agent requires reasonable care in timing, but is not so difficult that satisfactory preservation cannot be accomplished.

HOW TO PRESERVE THE J57 ENGINE.

Preservation of the J57 engine is quite simple if it is done while the engine is installed in the airplane. Basically, the preservation process consists of replacing the engine oil and fuel with preservative mixture and then spraying the engine compressor inlet with preservative mixture. The preservation operation is accomplished by running the engine and substituting preservative mixture for the fuel and oil. Although the process listed below is fairly detailed, you should never start to preserve a J57 engine until you have first read the controlling Military specifications and the engine maintenance technical order, (T.O. 1F-102A-2-4).

Preserving the Engine Oil System.

To preserve the engine oil system: first, place oil receptacles under the oil tank drain, the N_2 gear box drain, the N_2 rear gear box (elbow gear) drain, the drain on the left side of the N_2 gear box near the oil temperature bulb connection, and the fuel/oil drain. Remove the drain plugs and allow the oil to drain until the oil flow slows to a slow drip at all of the openings, then re-install and safety the drain plugs. The engine oil filter on the top of the main gear box has to be removed, completely disassembled, and cleaned with a suitable cleaner (JP-4 fuel is satisfactory for this purpose); after being cleaned the filter should be reinstalled. When you have accomplished the preceding operations, service the engine oil tank with 5½ gallons of preservative mixture. For the exact type of preservative mixture, you should consult the controlling specifications and the engine maintenance technical order, (T.O. 1F-102A-2-4).

Run the engine up before you start on the fuel system preservation operation. To assure that the preservative mixture thoroughly lubricates the interior of the engine, operate the engine at 75% power for a period of ten minutes.

Preserving the Engine Fuel System.

After shutting the engine down, following the oil preservation run-up, you are ready to begin preserving the engine fuel system. Disconnect the pressurizing and dump valve sensing line at the fuel control unit, cap the fitting on the fuel control unit but leave the sensing line open to atmosphere. Place a 5-gallon drainage receptacle under the dump valve drain discharge point (located at the left of the engine accessory compartment access door); then disconnect the afterburner fuel line at the forward side of the afterburner manifold drain valve (located forward of the firewall). Connect an overboard discharge hose to the end of the disconnected afterburner fuel line and route it to another 5-gallon receptacle under the airplane. Disconnect the flexible fuel line to the engine fuel pump inlet port and connect a 2-inch flexible hose to the pump inlet port. Place the other end of the hose in a 10-gallon container of preservative mixture. As a special note, be sure to keep this 10-gallon container at least four feet above the pump inlet port. This will assure sufficient inlet pressure to the pump. If the container should become empty at any time during the following operation, shut off the engine starter and refill the container.

Next, pull out the ignition circuit breaker on the breaker on the main wheel well circuit breaker panel, connect the external source of a-c and d-c power, and post a fire guard. Now connect the GTC unit air duct to the starter air duct receptacle in the main wheel well. Eight circuit breakers must be closed—FUEL SEL LH, FUEL SEL RH, FUEL CONT, AB CONT, AB POWER, EXT AC PWR, OIL PRESS, and MASTER. These circuit breakers are located on the forward left hand auxiliary and main wheel well circuit breaker panels.

The rest of the fuel system preservation operation has to be timed (either a timer or a stop watch is satisfactory). Let us assume that you use a stop watch. Advance the power lever to the TAKE OFF position; have the ground operator start the GTC unit and adjust the unit speed to 100% run; start the stop watch at the same time the GTC unit reaches 100%. After 15 seconds have elapsed, move the power lever outboard to the AFTERBURNER detent. When the watch reads 25 seconds, re-position the power lever to FULL MIL POWER. After 30 seconds total time, actuate the fuel control switch to EMERGENCY. Five seconds later, place the fuel control switch in the NORM position and move the power lever to the OFF position and then forward to the TAKE OFF position at a slow, steady rate. After 60-seconds total elapsed time, simultaneously shut off the starter and bring the power lever back to the OFF position.

After the engine has stopped turning, disconnect the 2-inch flexible hose from the engine fuel pump inlet. Then start the GTC again, adjust its speed to 100%, and allow it to turn the engine over. With the engine turning, spray about one-half pint of preservative mixture over the compressor inlet section. Be sure to keep the spray gun about 18 inches from the compressor inlet, and keep moving the gun constantly to assure that the entire compressor entrance area is sprayed evenly. When you have completed the spraying, disengage the starter valve, shut down the GTC unit, and disconnect the starter from the engine.

This completes the preservation operation of the engine while it is installed in the airplane. You are now ready to proceed with engine removal.

HOW TO DEPRESERVE THE J57 ENGINE.

All jet engines shipped from overhaul depots or engine manufacturers are preserved, to prevent engine corrosion. As a result, you will have to flush the engine fuel system and drain the preservative from the oil system. No special fluid is needed for the fuel system flushing operation—regular JP-4 fuel is entirely satisfactory. The basic idea of the engine depreservation operation is to drain the oil and then to crank the engine over (without starting it), so that fuel will be forced through the engine fuel system and wash away the preservative. Before starting the depreservation operation, fill the airplane fuel tanks.

Depreservation of the Engine Oil System.

In comparison to the depreservation operation for the fuel system, the engine oil system depreservation operation is relatively simple. Place oil drain receptacles under the oil tank drain, the N_2 gear box near the oil temperature bulb connection, and the drain on the fuel/oil cooler. These are the same drains that were used to drain the oil in the engine preservation operation which was discussed earlier in this section. Remove each of the drain plugs and allow the preservative to run from the drains.

Disconnect the afterburner sensing line from the afterburner igniter and provide a receptacle for the oil from this line. Leave the drain plugs out until the oil discharge slows to a slow drip. Then, remove and completely disassemble the engine oil filter at the top of the main gear box. Clean this filter with a suitable cleaning solvent or JP-4 fuel. Then, re-assemble the filter and install it on the engine. When the preservative has drained out, replace and safety the drain plugs, and connect the sensing line. Then fill the engine oil tank with 5.5 gallons of oil, specification MIL-L-7808.

Flushing the Fuel System.

As mentioned earlier, some J57 engines are equipped with pneumatic starters and other engines are equipped with combustion-type starters. The procedure given in this training manual for flushing the engine fuel system is for the pneumatic starter-equipped engines only. Information on flushing the engine fuel system on the combustion-type starter-equipped engines will be given in the engine maintenance technical order when the information is available.

Several preparations must be made before you are ready to flush the engine fuel system. Some of the preparations are safety precautions and others are maintenance duties. Never omit *any* of these preparations. First, disconnect the pressurizing and dump valve sensing line at the fuel control unit and cap the fitting on the fuel control. The sensing line can be left open to atmosphere during the flushing operation. Next, disconnect the afterburner fuel line at the forward side of the afterburner manifold drain valve and attach a drain hose to the disconnected line. Place the free end of the drain hose in a 5-gallon receptacle. Another 5-gallon receptacle should be placed under the drain discharge of the fuel pressurization and dump valve. Although receptacles of other sizes can be used, it is best to have at least the 5-gallon size in case more fuel is discharged than normally expected.

Although the engine is not going to be started for the fuel flushing operation, electrical power is needed. Pull the ignition circuit breaker in the main wheel well circuit breaker panel, connect the external source of a-c and d-c power, and post a fire guard.

Now, connect the GTC unit air duct to the starter air duct receptacle in the main wheel well. Eight circuit breakers must be closed—FUEL SEL LH, FUEL SEL RH, FUEL CONT, AB CONT, AB POWER, EXT AC PWR, OIL PRESS, MASTER. These circuit breakers are located on the forward left-hand auxiliary and main wheel well circuit breaker panels. You are now ready to proceed with the fuel system flushing operation.

With the GTC unit operating at 100% speed, advance the power lever to the TAKE OFF position. Notify the operator of the GTC unit to open the starter air valve and begin engine cranking. From this point, you should use either a timer or a stop watch to time the operation. Crank the engine over for 15 seconds, and then move the power lever to AFTERBURNER. After 10 seconds of this, retard the power lever back to non-afterburning. After 5 seconds of non-afterburning, actuate the fuel control switch to EMERGENCY for 5 seconds. Then move the power lever back to OFF and forward to TAKE OFF. This movement should not take longer than 5 seconds. One full minute after you start the flushing process, retard the power lever to the OFF position and release the starter switch.

At this time the ground-assisting crew man should check the amount of fuel in the 5-gallon receptacles. A minimum of three gallons of fuel-oil mixture should have drained from the dump valve drain and about two gallons from the afterburner fuel line. If less than this has drained, you should repeat the flushing procedure. If the minimum amounts are in the receptacles, remove the hose from the afterburner fuel line, then reconnect the pressurizing and dump valve line and the afterburner fuel line.

The fuel-filled fuel control unit must be allowed to soak for a minimum of eight hours. This soaking period is required to assure satisfactory sealing of the internal packings,and to condition the internal synthetic rubber diaphragms. After the eight hour soaking period has elapsed, the engine is ready for run-up.

ENGINE BUILD-UP.

Each jet engine received from the manufacturer for installation in an airframe must have certain components and accessories installed, removed, and, in some cases, replaced. This operation of preparing an engine for airframe installation is commonly referred to as *engine build-up.*

Almost every type of jet engine is designed and manufactured for use in more than one particular model and type of aircraft. Since all types of aircraft are not designed the same, as far as size and available space are concerned, aircraft engines must have their accessories and components located in such a manner as to compensate for the design peculiarities of each particular type of aircraft. Because of this fact, it has become an accepted practice for the engine manufacturer to supply the airframe manufacturer with the basic engine that has just a few of the necessary components mounted. The airframe manufacturer then installs those other accessories and components which the engine will need in his type of aircraft.

A typical example of the need for engine build-up is the power lever cross-over shaft that is installed on the J57 engine by the engine manufacturer. This cross-over shaft must be replaced with another cross-over shaft by the airframe manufacturer. The original cross-over shaft is not compatible with the F-102A power lever mechanical operating system. Another example of this nature is the outer exhaust nozzle leaves on the earlier J57 engines with the "iris" type of exhaust nozzles. These leaves, like the cross-over shaft, were designed for engine installation on types of aircraft other than the F-102A. The engine shroud configuration on the F-102A makes these exhaust nozzle leaves unnecessary and they must be removed before the engine is installed.

Now that you have a general idea of the need for engine build-up, let's go through a sample engine build-up operation and prepare a J57 engine for installation in the F-102A.

UNPACKING THE J57 ENGINE.

The J57 turbojet engine is shipped from the factory in a pressurized metal container. In figure 3-1, the engine is shown in the shipping container as the cover is being removed. The skids on which the metal container is mounted provide a means for keeping the engine and container upright while the engine is being shipped.

Before you start to remove the engine container cover, check the dehydrating agent inside the container. A small humidity window in the end of the engine container allows you to visually check the small bag of dehydrating agent and determine whether the agent is blue or pink. A dehydrating agent with a solid blue color indicates that moisture has not entered the container, while a pink-colored agent indicates that moisture has entered the container and engine corrosion has probably already started. If the dehydrating agent is pink, the engine should be handled in accordance with the existing technical orders. Normally, though, you will find that the agent is colored blue, and after determining this, you can proceed with the container cover removal operation.

Always release the air pressure from the container before attempting to loosen the hold-down bolts and remove the cover. The relief valve for this purpose is situated on the end of the container adjacent to the humidity inspection window. In most cases, engine containers are pressurized to only about 5 psi so no great length of time is required to relieve the pressure.

After removing all of the cover bolts, use a chain hoist or other suitable hoist or sling assembly to lift the container cover off. Note in figure 3-1 how the sling assembly is attached to the engine to lift the engine from the container. The sling attaches to one front and two rear attach fittings. A special sling (SE-0945) will be furnished as special equipment with the F-102A to provide a more efficient means for removing the engine from the shipping container and for lowering it on the engine build-up stand. The SE-0945 sling has provisions for lifting the engine, the engine and afterburner, or just the afterburner itself.

The J57 engine is mounted on shipping rails inside the metal container. These shipping rails are secured to the container. In figure 3-1, note that there are four bolts in each rail. Prior to lifting the engine from the container, the nuts from these eight bolts must be removed. Figure 3-1 shows how the shipping rails are secured to the engine mounts.

After you have lifted the engine from the shipping container, keep the engine on the sling and remove the shipping rails from the engine. As shown in figure 3-2, removal of the rails allows the right front thrust mount and its roller, and the left front mount bracket and its roller, to be installed on the front of the engine. These rollers are used only for engine build-up, engine removal, and engine installation. As you will see later in this section, the rollers assist in moving the engine into the F-102A. All of the rollers must be removed after the engine has been installed in the F-102A. The engine can now be lifted onto the rails of the engine removal and installation trailer SE-0635. The engine is then ready for engine build-up.

UNPACKING THE J57 AFTERBURNER.

The afterburner is shipped from the factory in a wooden container. In figure 3-3, the container is shown with the top and one of the sides removed. Note that this container also has a center of balance marker on the container. The bolts shown in the end of the box in the right hand portion of figure 3-3 are inserted through the afterburner mating flange and the box. These bolts hold the afterburner firmly in place during shipment. Although not shown in the illustration, SE-0945 sling assembly is used to lift the afterburner from the box bottom and hoist it onto an SE-0730-803 afterburner adapter stand. This stand is also furnished as special equipment with the F-102A. It was designed for use with the Lockheed truck 205226 which is widely used by the Air Force. By using the Lockheed truck with the afterburner adapter stand, you can easily move the afterburner around. As you will find out from experience, this movability aids greatly in the assembly of the afterburner to the engine.

AFTERBURNER TO ENGINE BUILD-UP.

The adapter stand, the Lockheed truck, and the afterburner are shown in figure 3-4. Note that the afterburner rests in the SE-0730-803 adapter stand which in turn rests on the rails of the Lockheed 205226 truck. The engine, shown in the right portion of figure 3-4, is suspended in the SE-0635 engine removal and installation trailer.

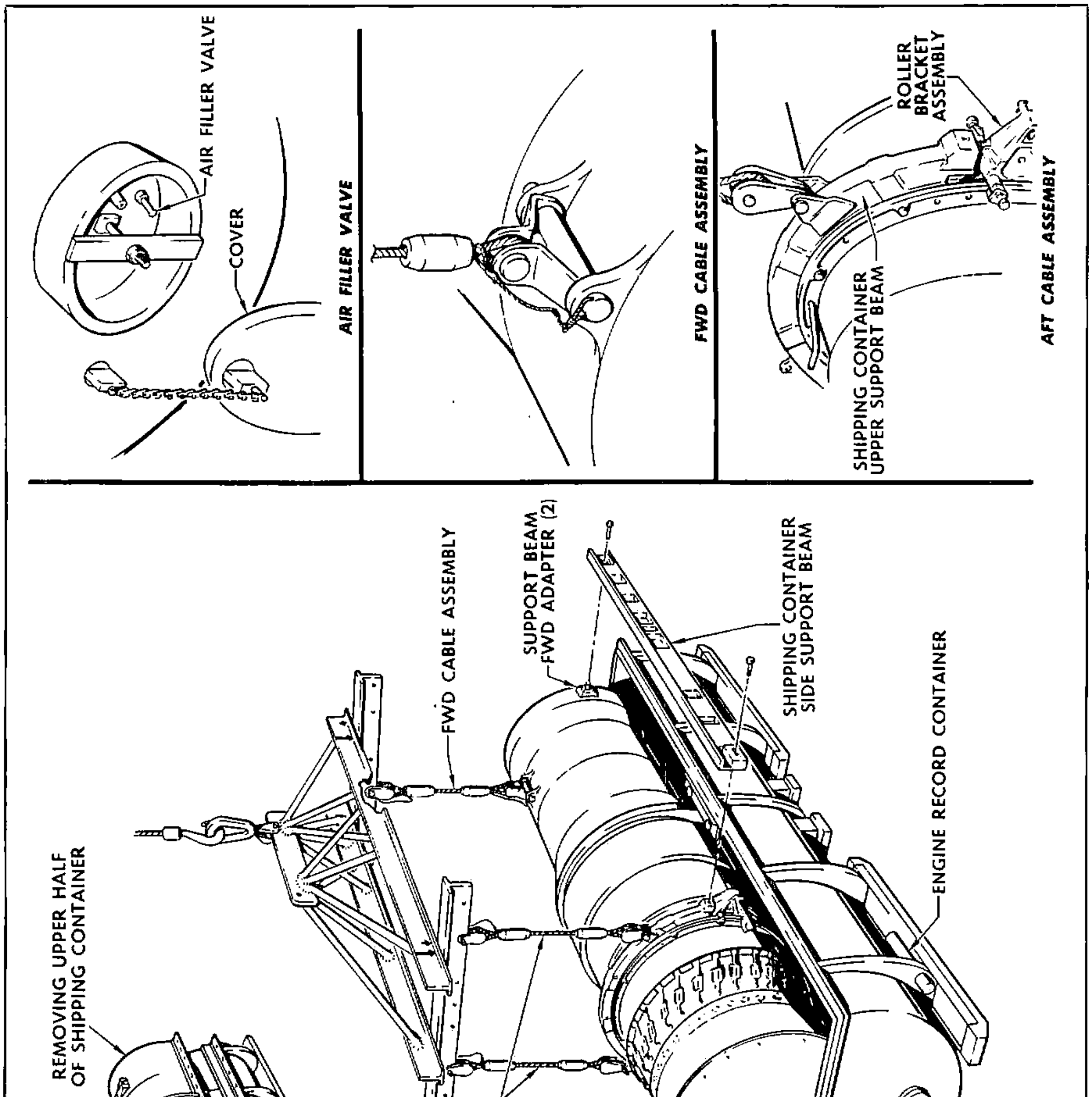


Figure 3-1. Unpacking the J57 Engine

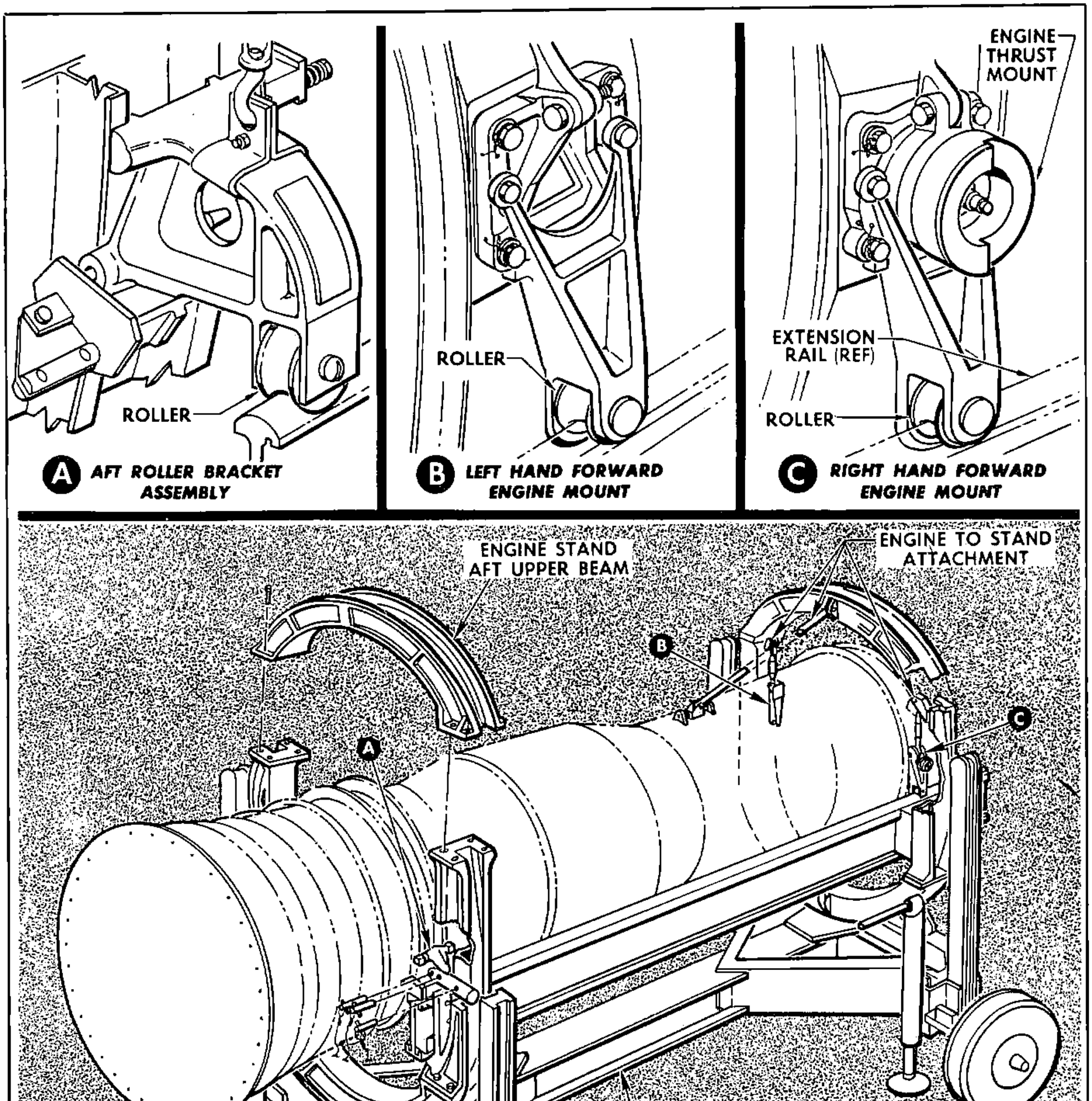


Figure 3-2. Preparing the J57 Engine for Engine Build-Up

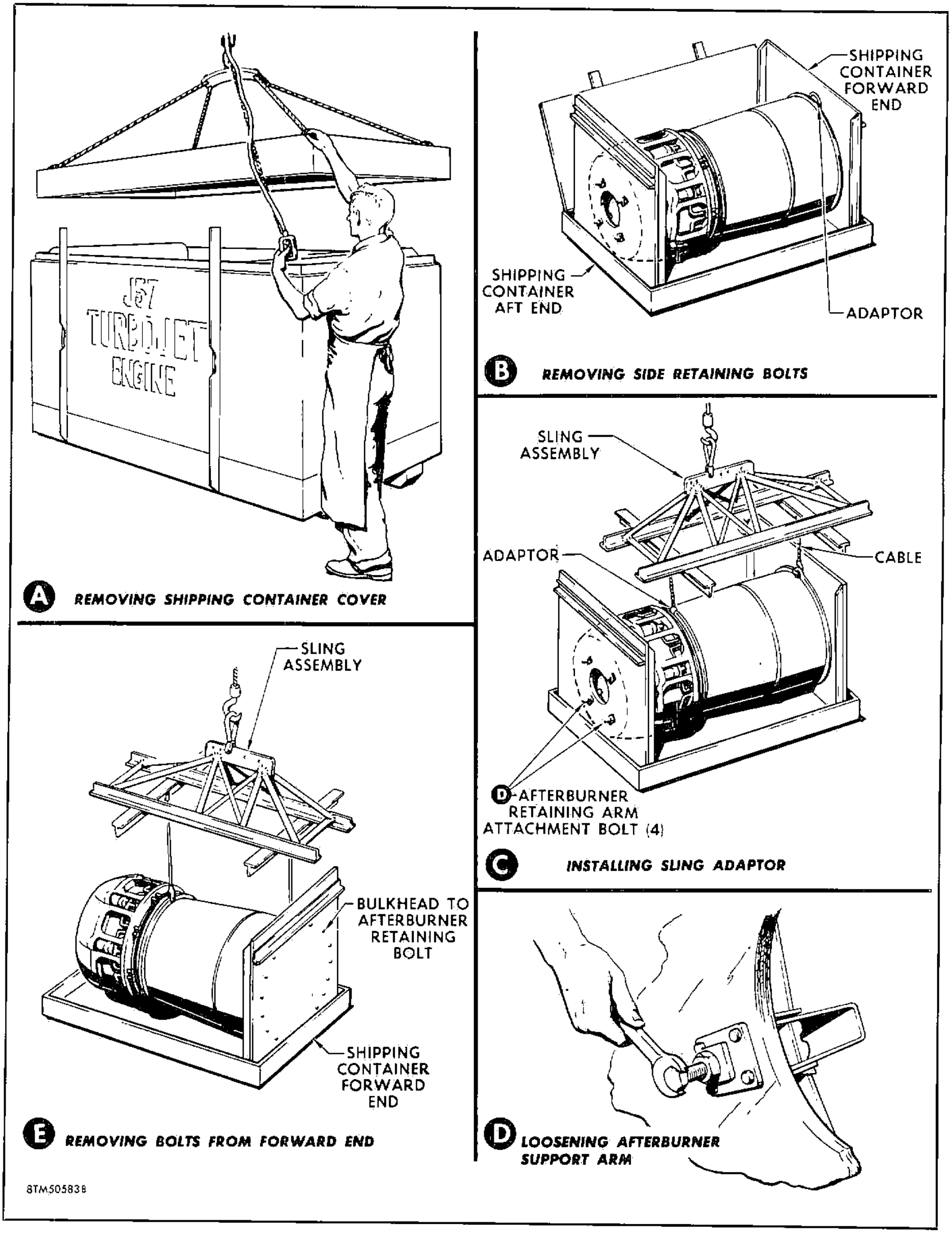


Figure 3-3. Unpacking the J57 Engine Afterburner

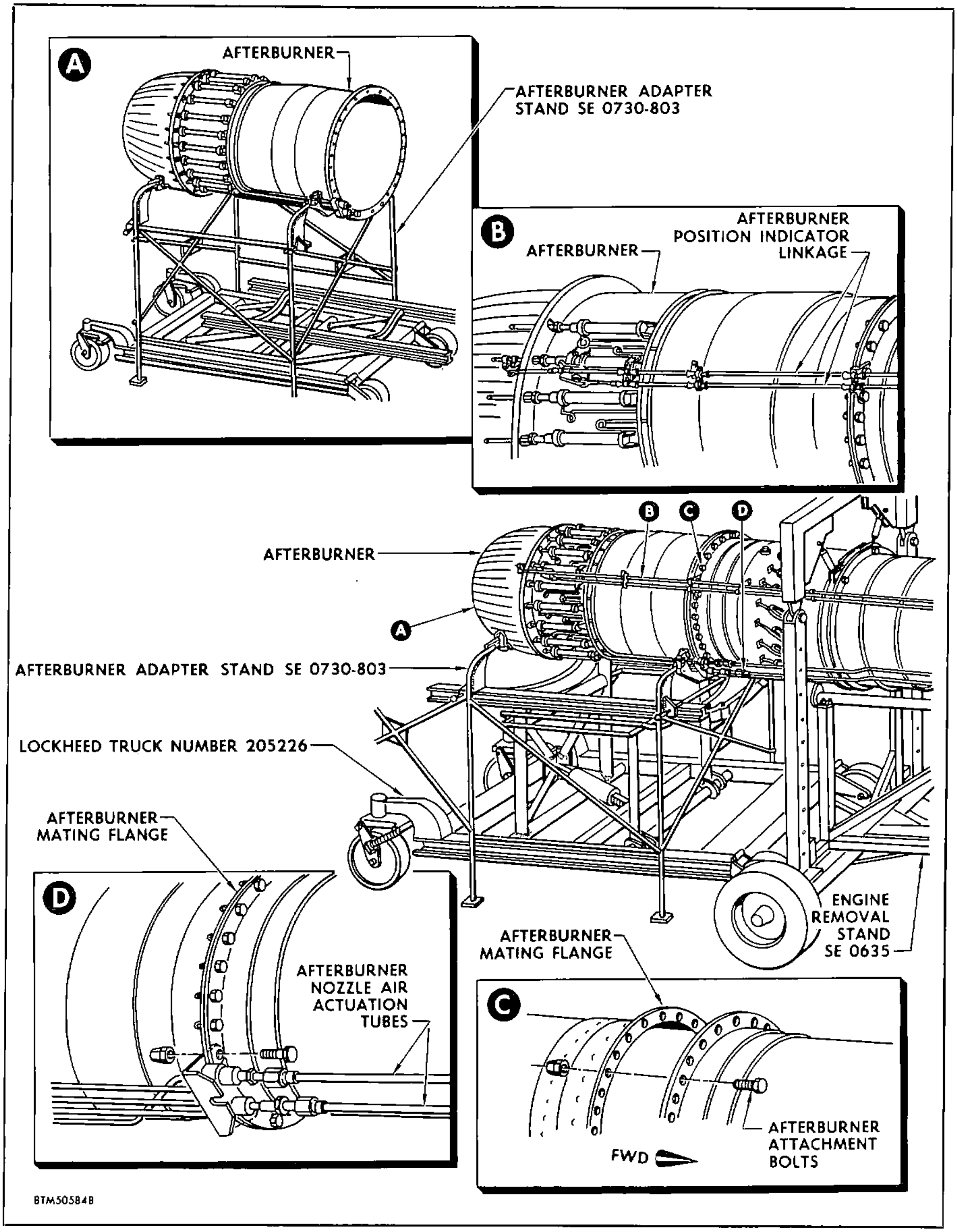


Figure 3-4. Installing the Afterburner

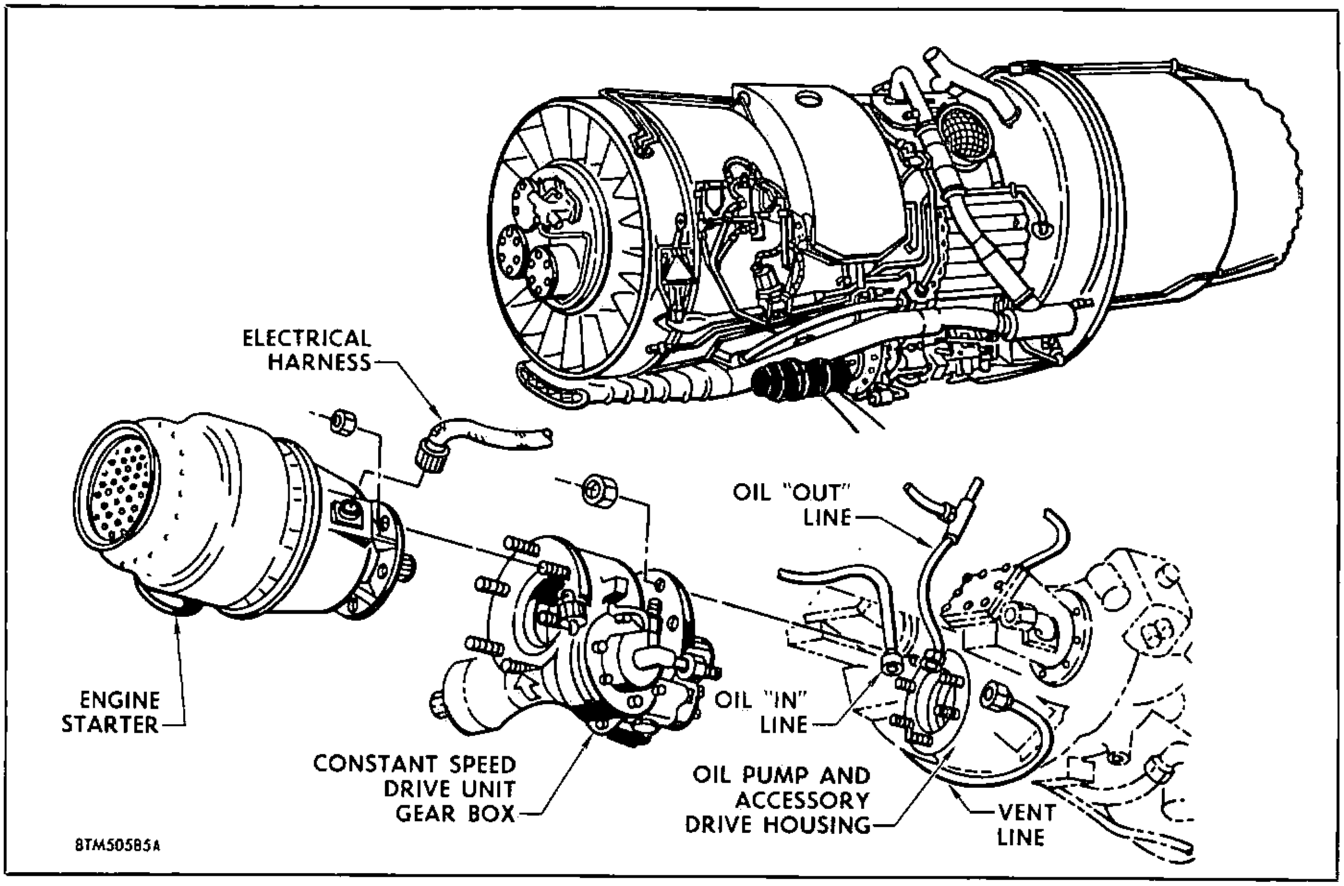


Figure 3-5. Sundstrand Gear Box and Engine Starter Installation

The afterburner installation is relatively simple. The adapter stand should be pushed up to the engine so that the afterburner mating flange aligns with the engine attach flange. Then install the attach bolts so that the bolt heads are towards the front of the engine as shown in view C of figure 3-4. When installing the attach bolts, you may find that a few bolts cannot be easily inserted in their proper holes. In this event, insert a drift pin in an adjacent hole and force the afterburner mating flange into the correct position—never hammer an attach bolt into its hole. You will probably find that there will always be a certain amount of "jockeying" involved in mating the afterburner with the engine attach flange; since there are 60 bolts to be installed, this is to be expected. Remember—never force the bolts into the holes.

In view B of figure 3-4, note how two of the attach bolts also hold the bracket for the afterburner nozzle air-actuation tubes. After these two bolts have been installed, the nozzle air-actuation tubes can be joined at their connection point just forward of the afterburner mating flange.

The afterburner attach bolts are torqued to approximately 70 inch-pounds. For the exact torque value, refer to the maintenance technical order, (T.O. 1F-102A-2-4). The attach bolts should be pattern-torqued; that is, you should start with the top bolt and work around in a counter-clockwise direction (looking from the front of the engine).

HOW TO INSTALL THE SUNDSTRAND ENGINE-MOUNTED GEAR BOX.

As you will recall from Chapter I, the Sundstrand constant-speed drive unit consists of two major components: an engine-mounted gear box; and an airframe-mounted transmission and gear box. For the purpose of engine build-up, we are interested in only the engine-mounted gear box. Figure 3-5 shows the location of the gear box and how it is attached to the engine accessory drive case. Note that the gear box shaft is splined to the accessory drive. The gear box is secured on its mounting pad by six mounting studs.

The installation of this unit should be relatively simple—place the oil seal gasket over the mounting studs on the gear case, align the gear box shaft with the gear case spline, and then slip the gear box over the six mounting studs. Tightening the six nuts and connecting the oil IN and OUT lines completes the gear box installation.

HOW TO INSTALL THE ENGINE PNEUMATIC STARTER.

The engine starter is mounted on the forward face of the Sundstrand gear box as shown in figure 3-5.

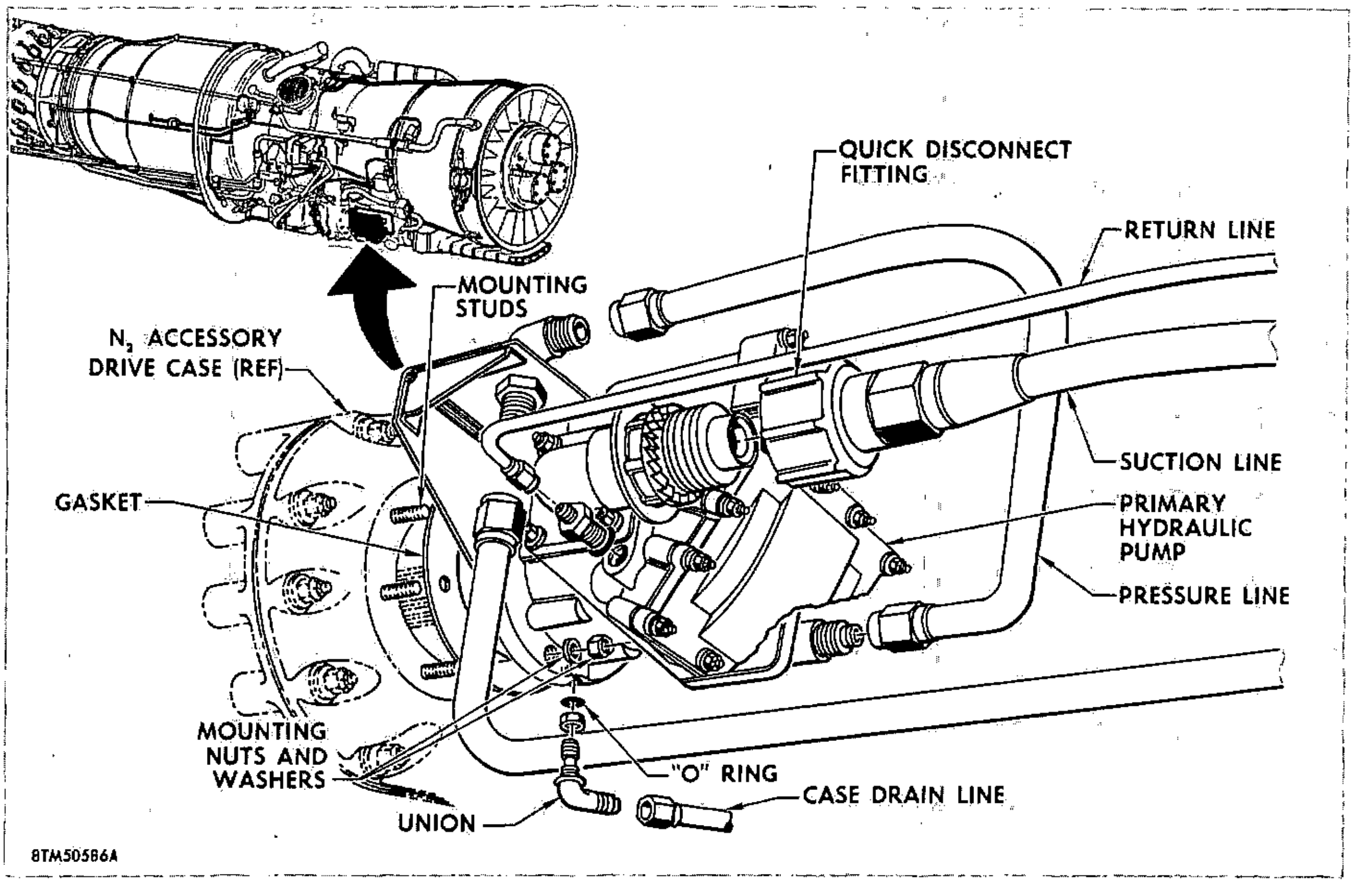


Figure 3-6. Primary Hydraulic Pump Installation

Note that the starter installation is just about the same as the gear box installation. After placing the gasket over the studs on the gear box, align the starter shaft spline with the gear box spline and slip the starter over the mounting studs. Then tighten the hold-down nuts and connect the electrical plug to the starter.

On some of the later model F-102A airplanes, a combustion-type starter is used. The installation of the combustion-type starter is quite similar to that of the pneumatic starter shown in figure 3-5. A starter mounting head (adapter) attaches to the constant-speed gear box and the combustion-type starter then attaches to the mounting head. In addition to the electrical connection, the combustion-type starter also has a starter air line and fuel line which must be attached. Complete information on the combustion-type starters will be furnished in the J57 engine maintenance technical order (T.O. 1F-102A-2-4), when the starters are furnished with the engine.

HOW TO INSTALL THE PRIMARY AND SECONDARY HYDRAULIC PUMPS.

Two hydraulic pumps must be installed on the J57 engine during engine build-up—the primary and secondary pumps. Each hydraulic pump supplies approximately 3000-psi hydraulic pressure to its respective hydraulic system. The two pumps are identical in appearance and function, and they are interchangeable. Note in figures 3-6 and 3-7, that the hydraulic pumps are mounted on the right and left side of the engine on the forward face of the engine accessory drive case. Each pump is splined individually to an accessory drive in the case.

A nebular spline must be used to install the pump. The nebular spline is a female spline adapter which fits over both the hydraulic pump spline and the engine spline. Usually, you will find it best to slip the spline adapter over the pump spline and then align the adapter with the engine spline. This will allow you to slip the pump over the six mounting studs. Tightening the bolts and connecting the hydraulic lines finishes the pump installation.

As a special note, always remember to fill the pump housing with hydraulic fluid, Specification MIL-0-5606, after you have installed the pump on the engine. The pump should be filled through the case drain line connection point.

Although the installation for only one of the pumps has been described, the installation procedure is just the same for the other. As mentioned, the primary and secondary hydraulic pumps are interchangeable.

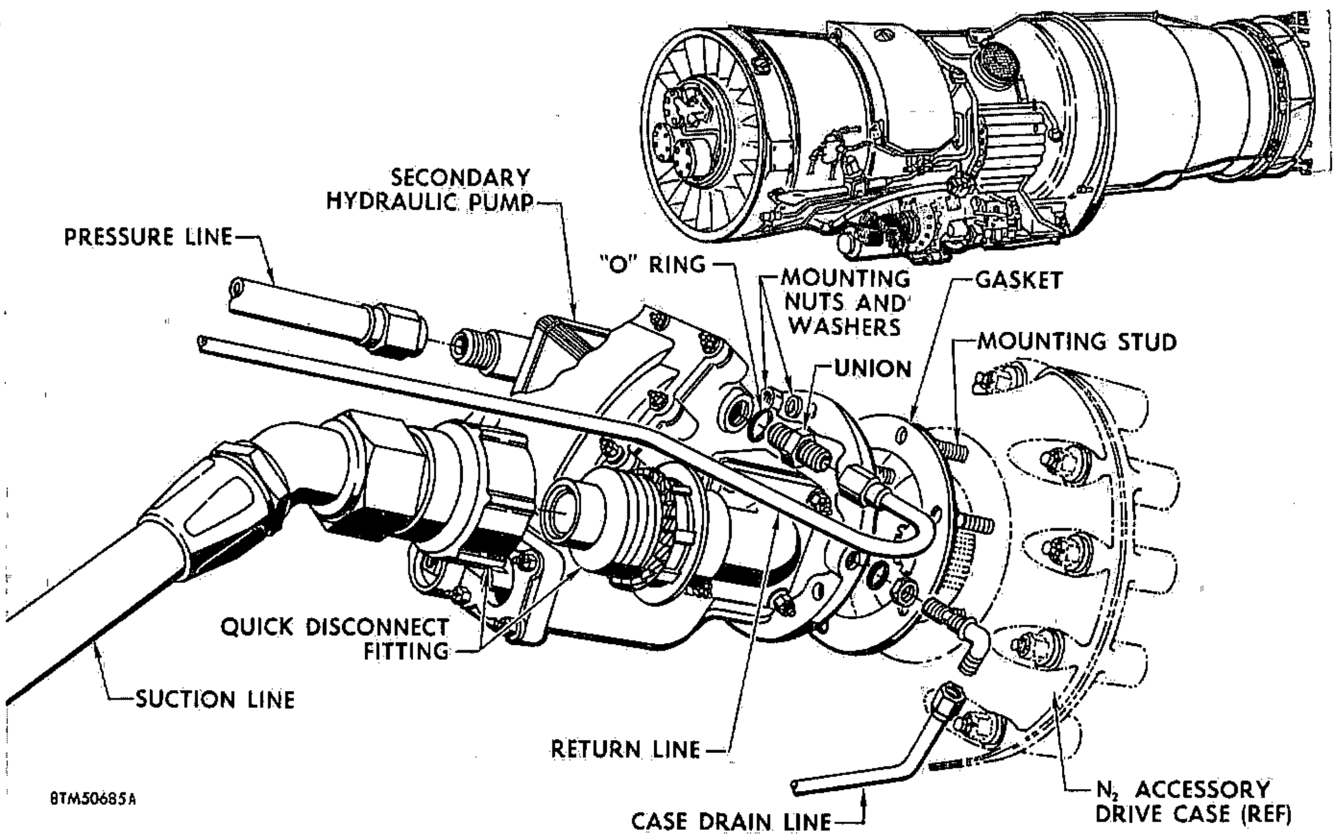


Figure 3-7. Secondary Hydraulic Pump Installation

The only difference is that either pump must be turned over before it can be installed on the other pump mounting pad. The primary pump, for example, must be rotated 180° when it is used as a secondary pump, to insure that the pump fittings align with the attaching hydraulic lines.

Comparing the installation of the two pumps, note that the case drain line for the primary pump connects to the bottom of the primary pump housing. The case drain line for the secondary pump connects to the side of the pump housing. The mounting face on each pump, primary and secondary, has four drain holes, but only one is used. The other three drains are plugged. This type of design permits the pumps to be installed on any type of engine and the most convenient drain opening can then be used.

HOW TO INSTALL THE TACHOMETER GENERATOR.

The tachometer generator is installed on the engine to furnish an indication of total engine rpm to the pilot on the tachometer instrument in the cockpit. The tachometer instrument is calibrated in percent of engine N_2 compressor rpm. The tachometer generator is installed on the accessory drive case immediately below the right hydraulic pump and the engine starter, as shown in figure 3-8.

The installation of the tachometer generator is a typical engine component installation and should not create any problem. Note that the generator is splined to the accessory drive in the case and is mounted on four studs. When installing the generator, place the gasket over the studs on the gear case and align the generator splined shaft with the spline in the accessory gear case. This will allow you to slip the generator down over the four mounting studs and tighten the hold-down nuts. Connecting the electrical connection on the generator and safetying the connector completes the installation.

HOW TO INSTALL THE POWER LEVER CROSS-OVER SHAFT.

The engine manufacturer's power lever cross-over shaft and bellcrank must be replaced with an assembly designed by the airframe manufacturer. This is necessary because the original cross-over shaft and bellcrank will not fit the F-102A's airframe-mounted power lever mechanical linkage system.

Figure 3-9 shows how the cross-over shaft and bellcrank ride in two bearings. Note how the shaft and bellcrank are secured to the bearings with collars and clevis pins. The bellcrank assembly slips over the end of the cross-over shaft on the left side and is clevis-pinned to the shaft. This makes the shaft and bellcrank one solid unit.

HOW TO INSTALL THE OIL LOW-PRESSURE WARNING SWITCH.

The oil low pressure warning system on the J57 engine notifies the pilot of dangerously low oil pressure conditions in the engine oil system. The oil low-pressure warning switch is actuated by a drop in the oil system pressure. When oil pressure drops occur, the switch closes and completes the electrical circuit to the oil low-pressure warning lights in the cockpit. When the pressure again rises, the switch opens and breaks the circuit.

The warning switch is installed on two mounting brackets between the N_1 and N_2 compressors on the left side of the engine as shown in figure 3-10. Two of the compressor flange bolts must be used to secure each pressure switch warning bracket. Tightening the switch bolts, and then connecting the electrical lead and the two one-fourth inch tubes, completes the installation of the switch. The operation of the warning systems and their components will be discussed in detail in Chapter IV.

HOW TO INSTALL THE FUEL LOW-PRESSURE WARNING SWITCH.

The fuel pressure warning switch actuates the warning light which notifies the pilot of a dangerously low

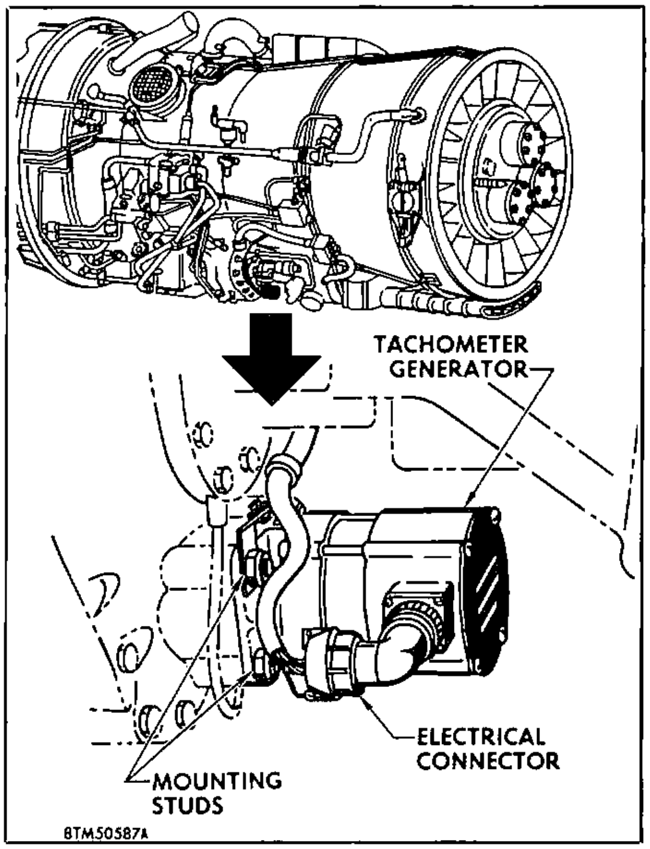


Figure 3-8. Tachometer Generator Installation

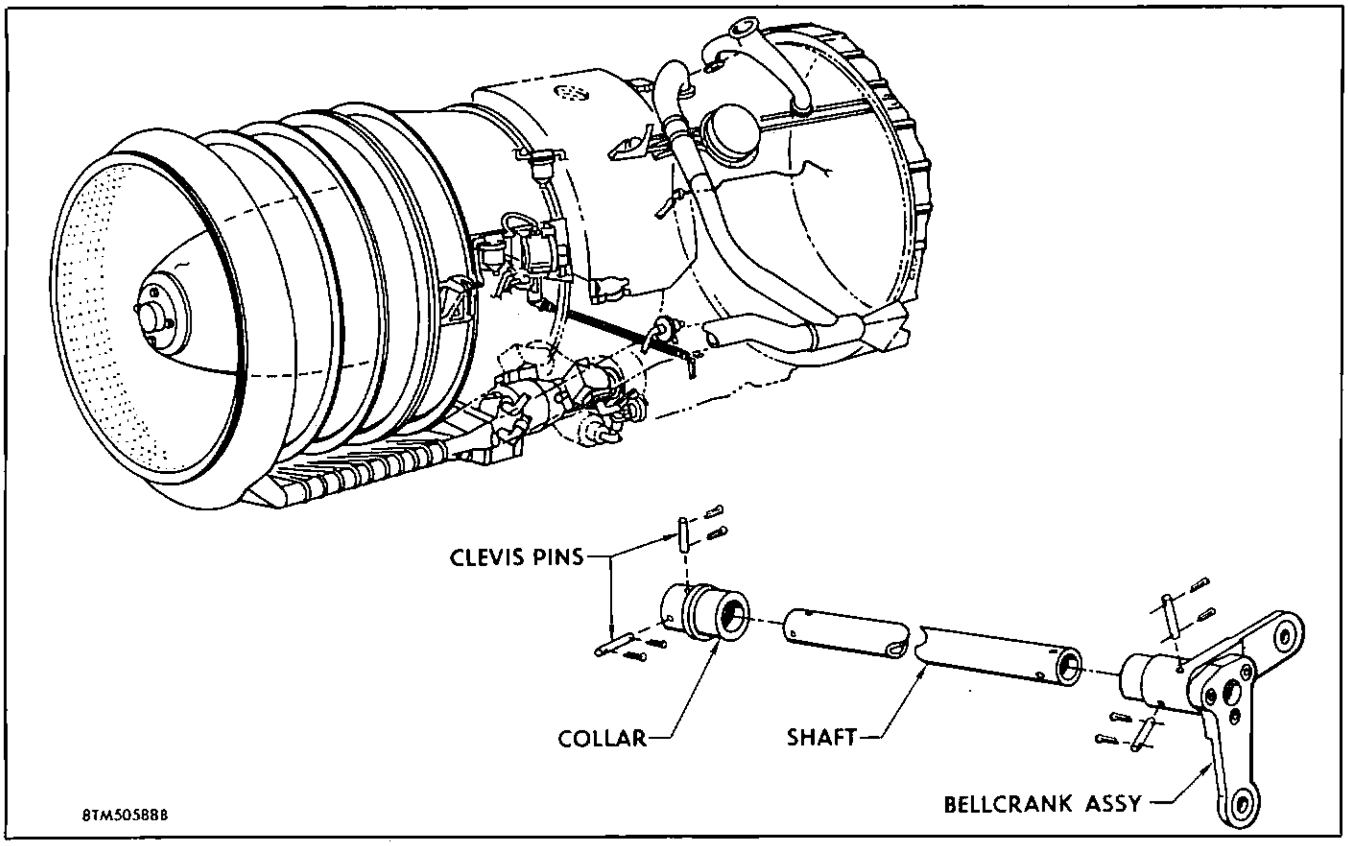


Figure 3-9. Engine Cross-Over

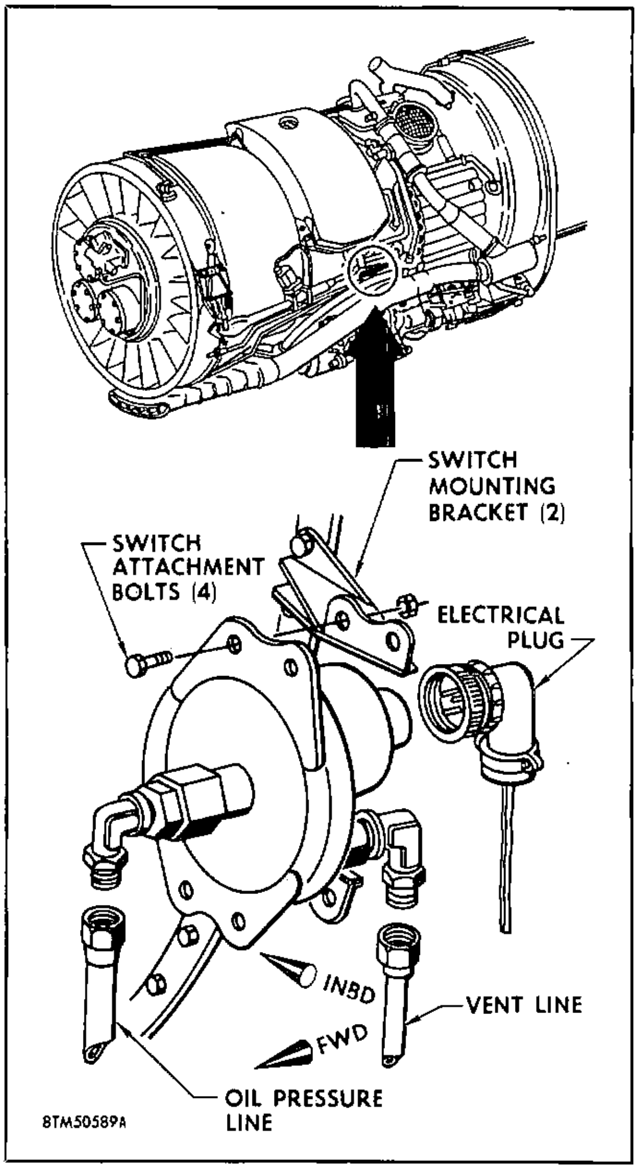


Figure 3-10. Oil Low-Pressure Warning Switch Installation

fuel pressure condition in the engine fuel system. When a drop in fuel pressure occurs, the switch closes and completes the electrical circluit to the warning light in the cockpit. When the fuel pressure rises the switch opens and the warning light goes out.

Note in figure 3-11 that the switch and mounting bracket are attached by bolts at three points on the lower left side of the accessory drive case. The fuel low-pressure switch has two tubing connections, the fuel pressure sensing line and the fuel drain line. Connecting these two lines and the electrical connection completes the low-pressure switch installation.

HOW TO INSTALL THE FUEL FLOWMETER.

The fuel flowmeter measures how much fuel the engine is consuming per hour. Note in figure 3-12 that the flowmeter is installed on the right side of the engine between the exhaust nozzle control valve and the main fuel control unit. The flowmeter is secured to the N_2 compressor case by three attach bolts. The arrows on the attach points on the flowmeter indicate the correct attachment of the fuel OUT and IN lines. The electrical lead that connects to the flowmeter runs to the fuel flow indicator in the cockpit. The internal details of this fuel flowmeter and the indicating instrument will be covered in Chapter IV.

HOW TO INSTALL THE AFTERBURNER NOZZLE POSITION SWITCHES.

The afterburner nozzle position switches detect the position of the afterburner nozzle. The switches then complete an electrical circuit to the indicating instrument in the cockpit. Note in figure 3-13 (detail B)

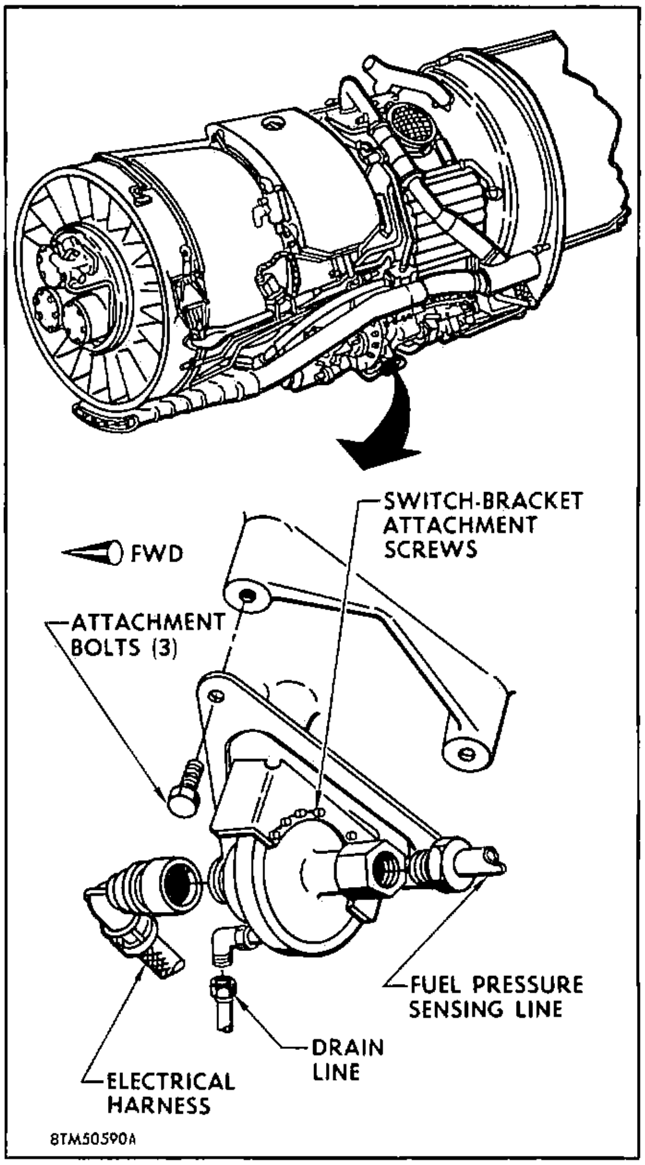


Figure 3-11. Fuel Low-Pressure Warning Switch Installation

that the switches are secured on the engine adjacent to the anti-surge bleed valve on the right side of the engine. In detail B of figure 3-13, you will not that a nozzle position cable connects to the position switch. The movement of the afterburner nozzle causes the switch cam to depress one of the switches. The indicator in the cockpit will then show that the afterburner nozzle is open or closed, depending upon whether the OPEN or CLOSED switch is actuated by the cam.

Although the switch installation is the same on both the -41 and the -23 engines, the position cable attachment at the exhaust nozzle is slightly different on the two engine models. This is shown in detail A of figure 3-13. On the -23 engine the cable bracket attaches directly to the nozzle door, while on the -41 engine the cable bracket attaches to the actuating cylinder. The operational and functional details of the afterburner nozzle position indicator switches and the indicating instrument will be covered in Chapter IV.

HOW TO INSTALL THE SUNDSTRAND CONSTANT-SPEED DRIVE OIL FILTER.

The oil filter for the constant-speed drive unit insures that no foreign material enters the engine main oil system through the drive unit oil system. This oil filter is mounted on the upper right side of the N_1 and N_2 compressor attach flange. Three bolts on the compressor flange must be "picked up" to secure the filter bracket. Note in figure 3-14 how the marman clamp holds the oil filter in place. The oil OUTLET and INLET lines attach to the filters by standard AN fittings.

HOW TO INSTALL THE ENGINE AIR INTAKE STUB DUCT.

The engine air intake stub duct is designed to direct air into the engine and also to bleed off cooling air for certain parts of the engine. Installation of the intake duct is shown in figure 3-15. Note that the duct attaches to the front frame of the engine. It is common practice to insert the attach bolts so that their heads are towards the front of the engine. The attach bolts must be torqued (as mentioned earlier in this chapter); all torque values for engine build-up should be obtained from the engine maintenance technical order, (T.O. 1F-102A-2-4).

HOW TO INSTALL THE AIR/OIL COOLER AND DUCTING.

The air/oil cooler cools the engine oil. As shown in figure 3-16, the cooler is mounted on the engine just forward of the engine starter. The cooler is secured to the engine with 12 bolts, six of which are engine flange attach bolts. The oil INLET and OUTLET lines connect to the left side of the cooler as shown. The other lines, shown in figure 3-16, routed through the

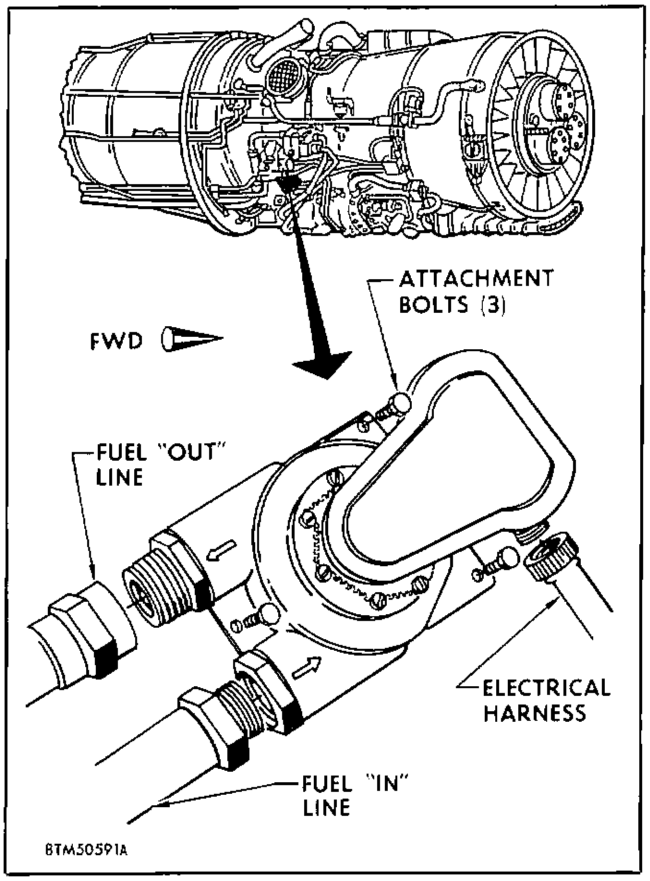


Figure 3-12. Fuel Flowmeter Installation

cooler bracket on the right side, belong in other systems. These lines use the cooler bracket as a stabilizer to prevent excessive vibration or chaffing of the lines Note in figure 3-16 how the air inlet duct attachs to the air/oil cooler. The duct flange must go inside the air/oil cooler flange.

HOW TO INSTALL THE ALTERNATE COOLING AIR VALVE.

The alternate cooling air valve assures that the engine receives cooling air at low air speeds or when the engine is operating on the ground. The air valve and its accompanying ducting are installed near the top of the engine on the N_2 compressor case. Note in figure 3-17 that the ducting for the air valve is made of flexible braided tubing. One end of the air valve ducting attaches to the engine shroud cooling duct; the other end of the ducting attaches to the air duct from the N_1 compressor case. Two flexible valve clamps hold the air valve securely in place between the two sections of braided ducting. Connecting the electrical lead and safetying it completes the installation of the alternate cooling air valve.

HOW TO INSTALL THE GROUND COOLING AIR DUCT CHECK VALVE.

As you will note in figure 3-18, this check valve has three openings—a ground cooling intake (from the

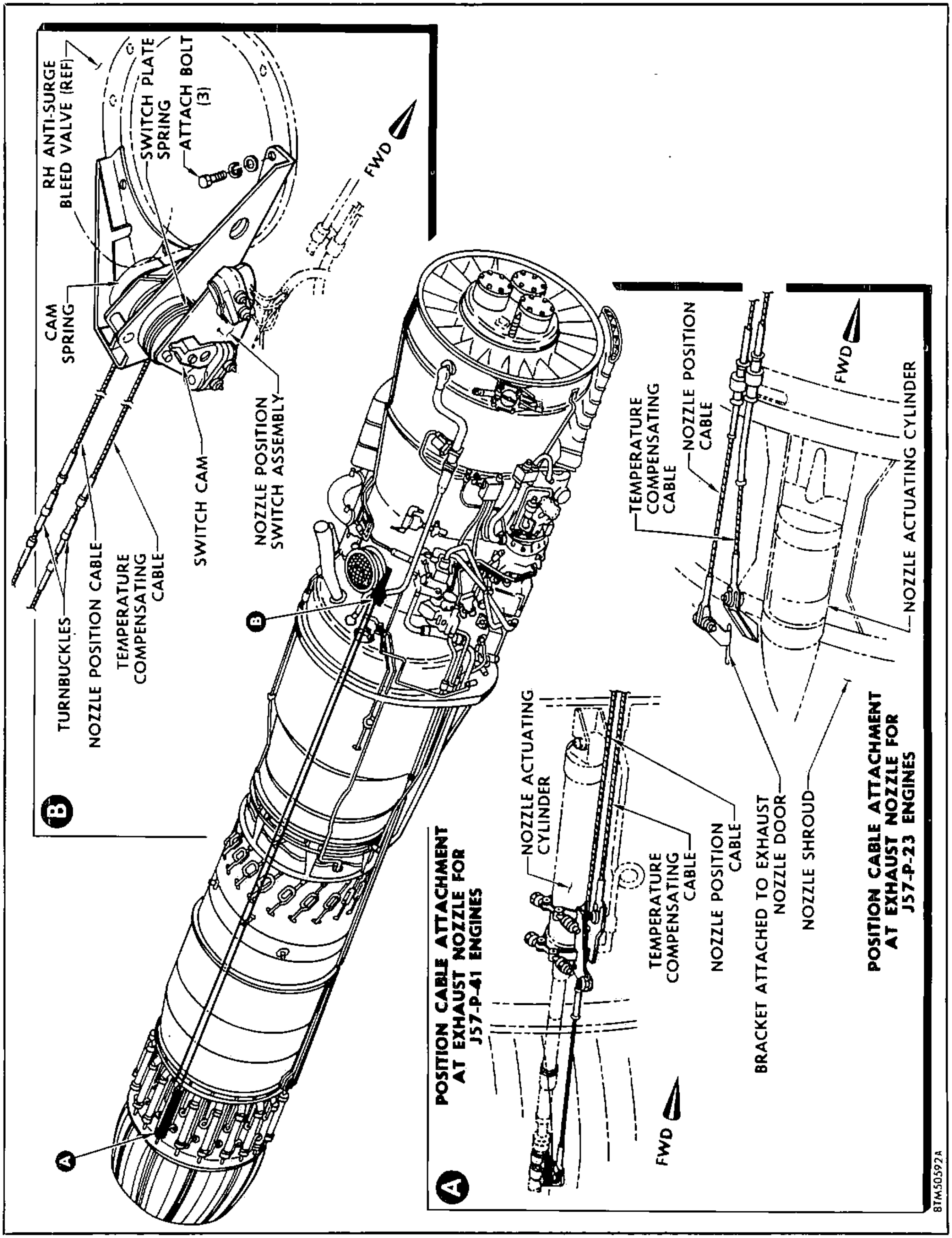


Figure 3-13. Afterburner Position Indicator Switch Installation

air valve just discussed), an in-flight cooling intake, and an outlet to the transition duct on the engine shroud. The check valve is mounted on the left side of the engine just below the fuel/oil cooler. The aft marman clamp secures the check valve to the engine. Note in figure 3-18, that the ground cooling air supply duct attaches to the check valves by means of two bolts. The smaller duct from the air supply ducting attaches to the check valve by means of a small marman clamp, while the outlet of the check valve attaches to the transition duct by means of a slip joint connection. This type of joint allows for duct expansion due to engine heat.

HOW TO INSTALL THE LOW-PRESSURE PNEUMATIC BLEED DUCT.

This low pressure pneumatic bleed duct is installed as a source of low-pressure pneumatic system pressure for operating various components in the airplane. The bleed duct is mounted on the top of the engine at the aft end of the N_2 compressor case as shown in figure 3-19. It is secured to the engine by three attach bolts at each connection point.

HOW TO INSTALL THE COMPRESSOR BLEED VALVE ADAPTER.

The compressor bleed valve adapters are installed directly under the bleed valves that are furnished with the engine. The bleed valves themselves are installed on the upper right and left sides of the engine. Although the J57-P-41 engine has two bleed valves—one on each side of the engine; the -23 model engine has only the left hand bleed valve installed.

Figure 3-20 shows a typical -23 engine. Note the adapter installed dircetly under the bleed valve. The airframe manufacturer has provided this adapter so that the bleed valve will reach the opening in the airframe. When you first receive an engine for engine build-up, you will have to remove the bleed valve from the engine and then secure the bleed valve

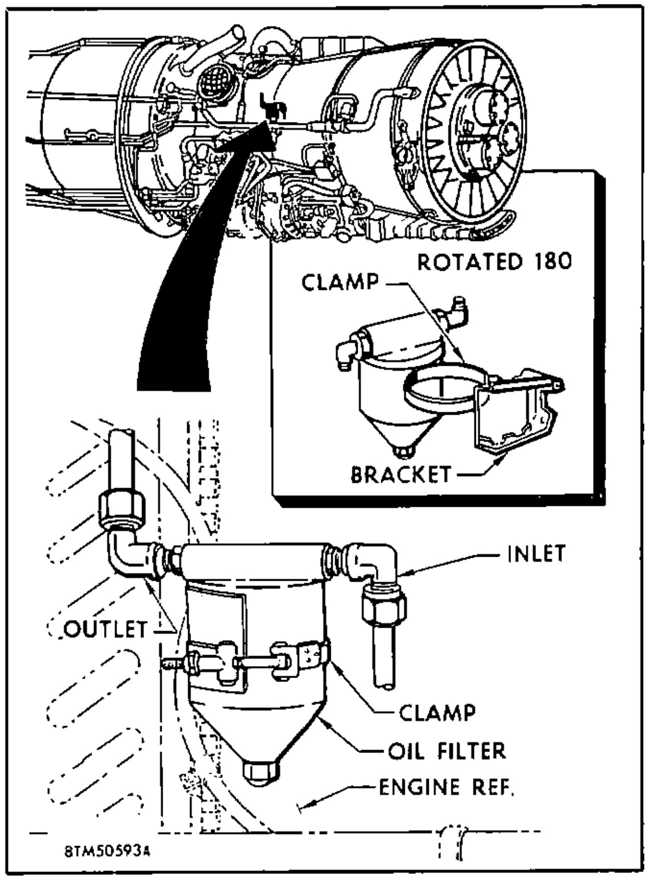


Figure 3-14. Sundstrand Drive Oil Filter Installation

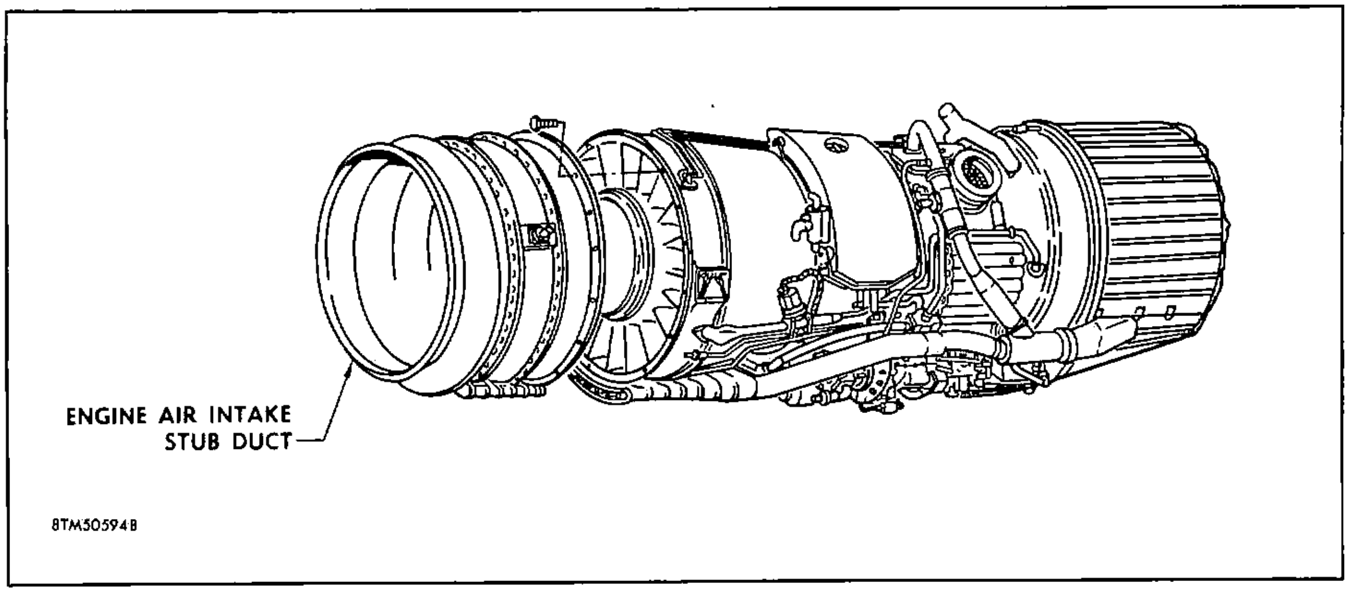


Figure 3-15. Engine Air Intake Stub Duct Installation

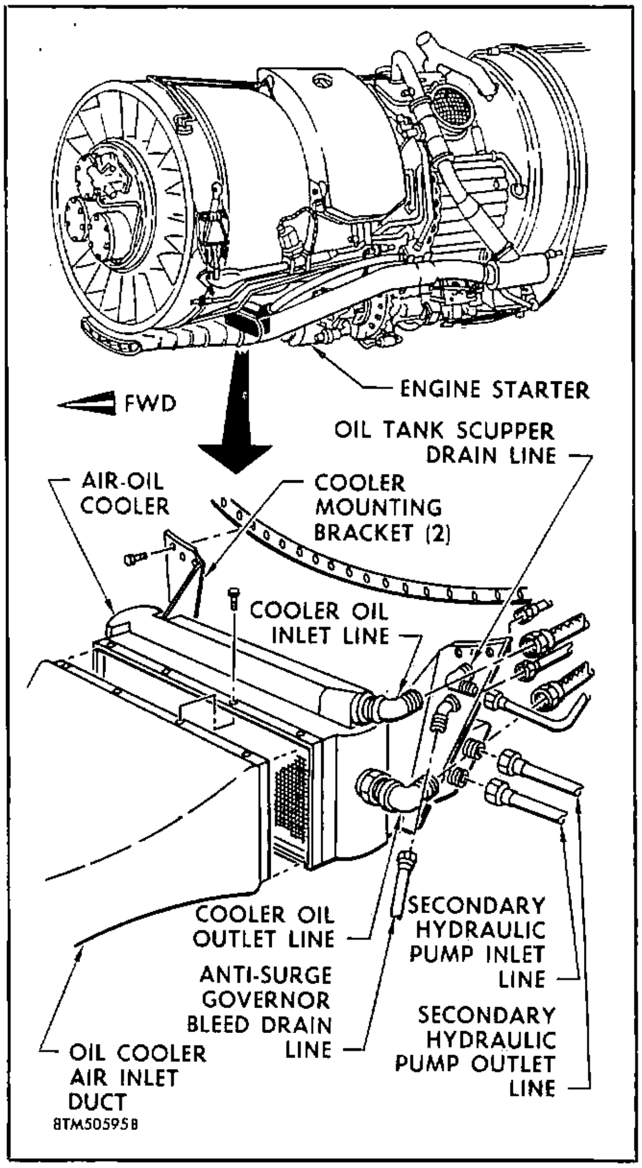


Figure 3-16. Air/Oil Cooler Installation

adapter with the same bolts that you took out when you removed the valve. The adapter then attaches to the bleed valve by means of the marman clamp as shown in figure 3-20. Three ¼-inch tubes connect to the fittings on the side of the bleed valve actuator.

HOW TO INSTALL THE EXHAUST NOZZLE ACTUATOR INSULATING BLANKET.

The exhaust nozzle insulating blanket is used only on J57 engines with the "iris" type afterburner exhaust nozzles. The blanket insulates the afterburner actuating components from the engine heat. The insulating blanket is fitted around the inside of the tailcone under the exhaust nozzle actuators, and it is laced in place with common safety wire. You will find that it is not necessary to remove the afterburner actuating mechanism to install the insulating blanket. Just loosen the forward attach bolts on the actuating cylinder so that the blanket can be slipped around the tailcone. For further information on the installation of the insulating blanket, refer to thhe engine maintenance technical order, T.O. 1F-102A-2-4.

HOW TO INSTALL THE ENGINE SHROUD.

The engine shroud provides efficient engine cooling and protects the airframe structure from excessive engine heat. As you can see from figure 3-21, the shroud is divided into two sections. The forward section of the shroud, with its transition duct, is positioned around the engine at the forward fire seal and is secured in eight places by eye bolts and nuts. The aft section is secured to the forward section by a large marman clamp. Note that the aft section is also supported in two places by two engine shroud support

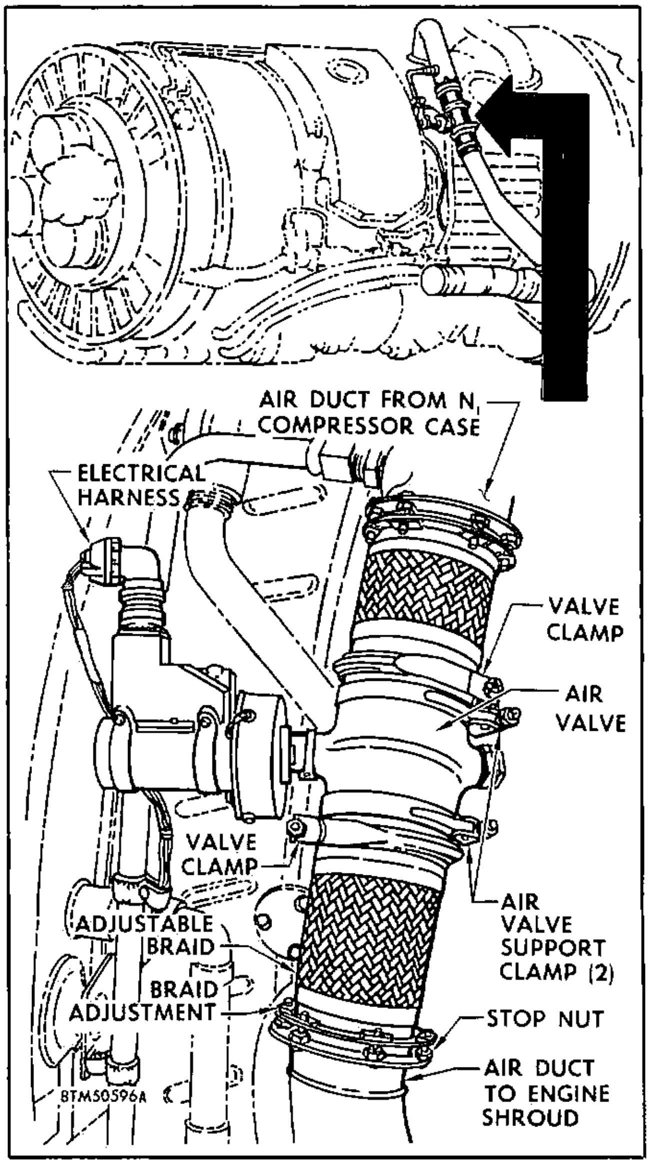


Figure 3-17. Alternate Cooling Air Valve Installation

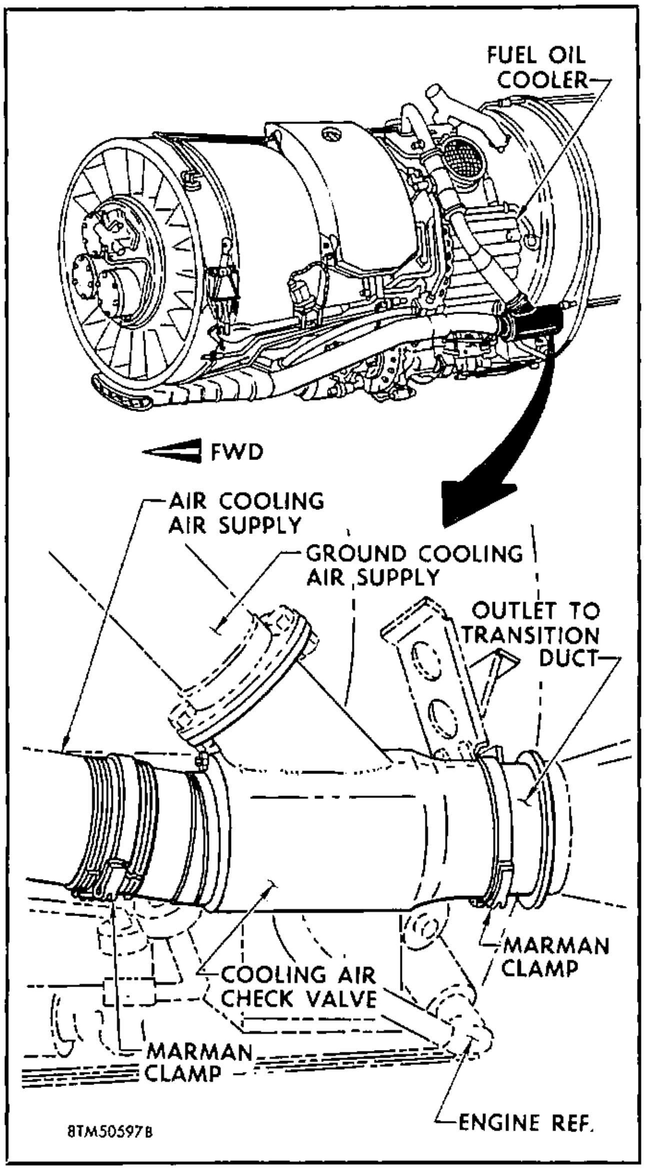


Figure 3-18. Cooling Air Duct Ground Check Valve Installation

brackets and turnbuckles. These two supports are accessible through sliding access doors in the shroud. The aft end of the engine shroud must be held away from the engine by support blocks during engine biuld-up. The four regular support turnbuckles hold the shroud away from the afterburner when the engine is installed in the aircraft.

HOW TO INSTALL THE N_1 ACCESSORY FAIRING.

The N_1 accessory fairing is mounted on the nose of the J57 engine at the air intake. The accessory fairing houses the bleed valve governor and the N_1 accessory case. Note in figure 3-22 that the fairing is secured in position by 12 studs in the accessory case. The hold-down nuts for the studs can be tightened only by reaching through the hole in the end of the fairing. The nose cap is then installed with four bolts to complete the fairing installation.

ENGINE INSTALLATION IN THE F-102A.

Installing the J57 engine in the F-102A airplane can be a compartively easy operation—providing you have and use the necessary ground support equipment. Following the procedures and instructions in the maintenance technical order (T.O. 1F-102A-2-4) will also make the operation as simple as possible. To acquaint you with the general procedure and requirements of the engine installation operation, let's go through a sample operation.

PREPARATION FOR ENGINE INSTALLATION.

The airplane must first be readied for the engine by stabilizing th airplane in a relatively level position. This leveling process can be accomplished quite easily by use of standard aircraft jacks. Then remove the tail cone (if it is not already removed) and either open or remove all of the fuselage access doors in the engine area. The tail cone removal is facilitated by the use of the Lockheed truck 205226 and the adapter stand SE-0731, which are shown in the lower right portion of figure 3-23.

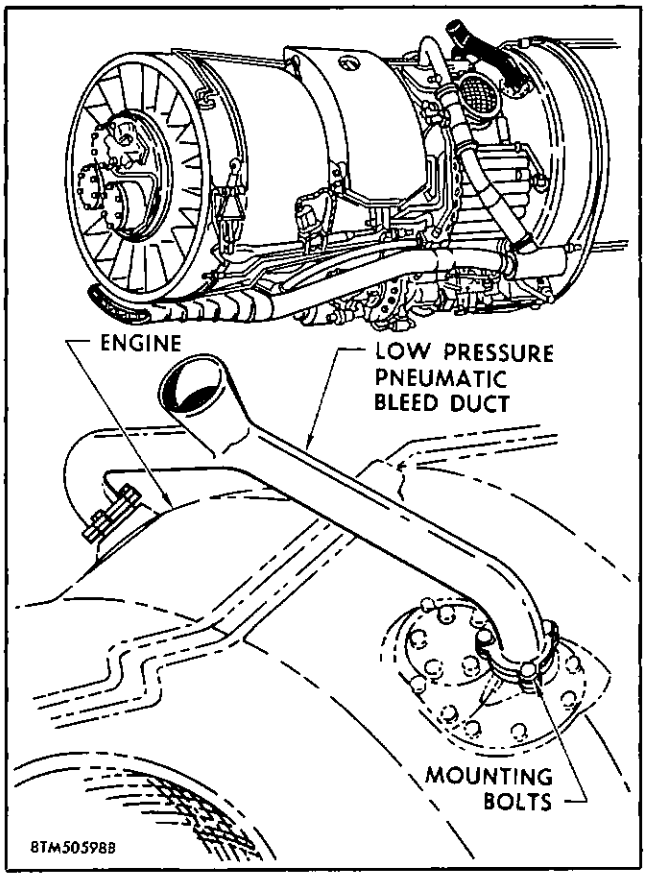


Figure 3-19. Low-Pressure Pneumatic Bleed Duct Installation

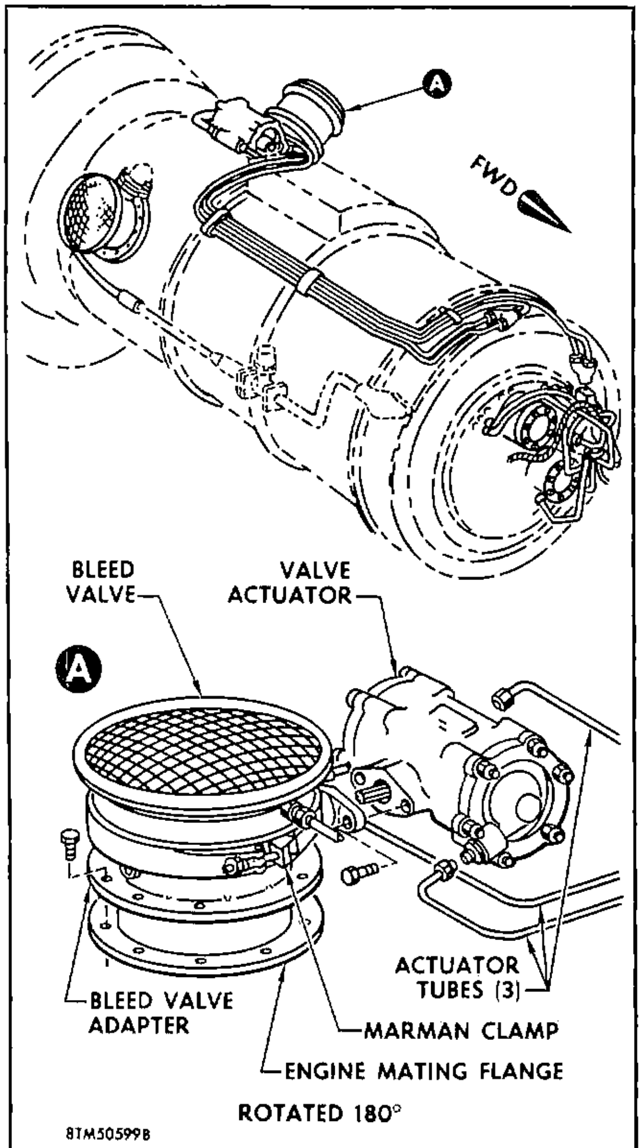


Figure 3-20. Compressor Bleed Valve Adapter Installation

Special installation rails and brackets must be installed inside the airplane to assist in rolling the engine into the airframe. These extension rails (SE-0857 -801 and -802) are furnished as special equipment with the F-102A. The extension rails and their brackets are shown in their installed position in the upper view of figure 3-23. Detail C in figure 3-23 shows the typical rail support bracket installation—note that the support bracket fits inside the slot in the side of the installation rail. You will have to install three of these rail support brackets on each side of the fuselage. The forward two support brackets, one on each side of the fuselage, stay in the aircraft permanently. These eight brackets support the total weight of the rails and the engine while the engine is rolled into the airframe.

After you have the engine installation rails firmly secured to the fuselage structure, you are ready to move the engine up to the airplane. It is a good practice to position the engine mount stand SE-0635 just aft of the fuselage so you can visually determine when the engine has been raised to about the right height. Then move the engine mount stand up close to the airplane. Using the jacks on the engine mount stand, raise the mount stand high enough that the rails on the stand align with the rails that you have installed in the airframe. When you have the two sets of rails aligned, insert the special rail splice bolts (provided with the stand) through both sets of rails so that the rails are locked together. Then attach the engine mount stand draw bar to the attachment bracket on the lower flange of the engine afterburner section. Keep in mind that this is just a general description of the operation. Detailed operating instructions for the engine mount are included on an instruction plate attached to the stand.

In detail C of figure 3-24 note the mount pad, the alignment ring, and the fuselage portion of the thrust mount. Place the alignment ring over the fuselage portion of the thrust mount. Then proceed with the next step—attaching the forward mount turnbuckles to the fuselage. It is a good practice to center the turnbuckle barrels between the turnbuckle eyes before you attach them to the fuselage. This will give you greater adjusting range with the turnbuckles after the engine is installed. At the same time you are attaching them to the fuselage, lubricate the turnbuckle threads wth grease—MIL-G-7187 or equivalent.

INSTALLING THE ENGINE.

Now you are ready to roll the engine into the airframe. Make sure that all lines and equipment will clear the airplane, then roll the engine forward on the engine installation rails until the right front engine thrust mount contacts the fuselage fitting. This is the fitting on which you have already placed the alignment ring. The assembled view of the forward mount fitting is shown in detail A of figure 3-23. In figure 3-24, note the two engine support link assemblies at the aft end of the engine. These assemblies indicated by the number 7 are more commonly called "dog bones." By adjusting the "dog bones" and the forward mount turnbuckle assemblies, you can lift the engine up from the engine installation rails. Raise the engine just far enough so that the engine support rollers turn freely. It is good practice to keep the number of turns equal for both of the forward mount turnbuckles.

Now, remove the engine installation rails, the rail support brackets, and the forward and aft engine rollers. As mentioned earlier, the two forward rail support brackets remain installed on the airframe. With the rails removed, adjust the forward right support

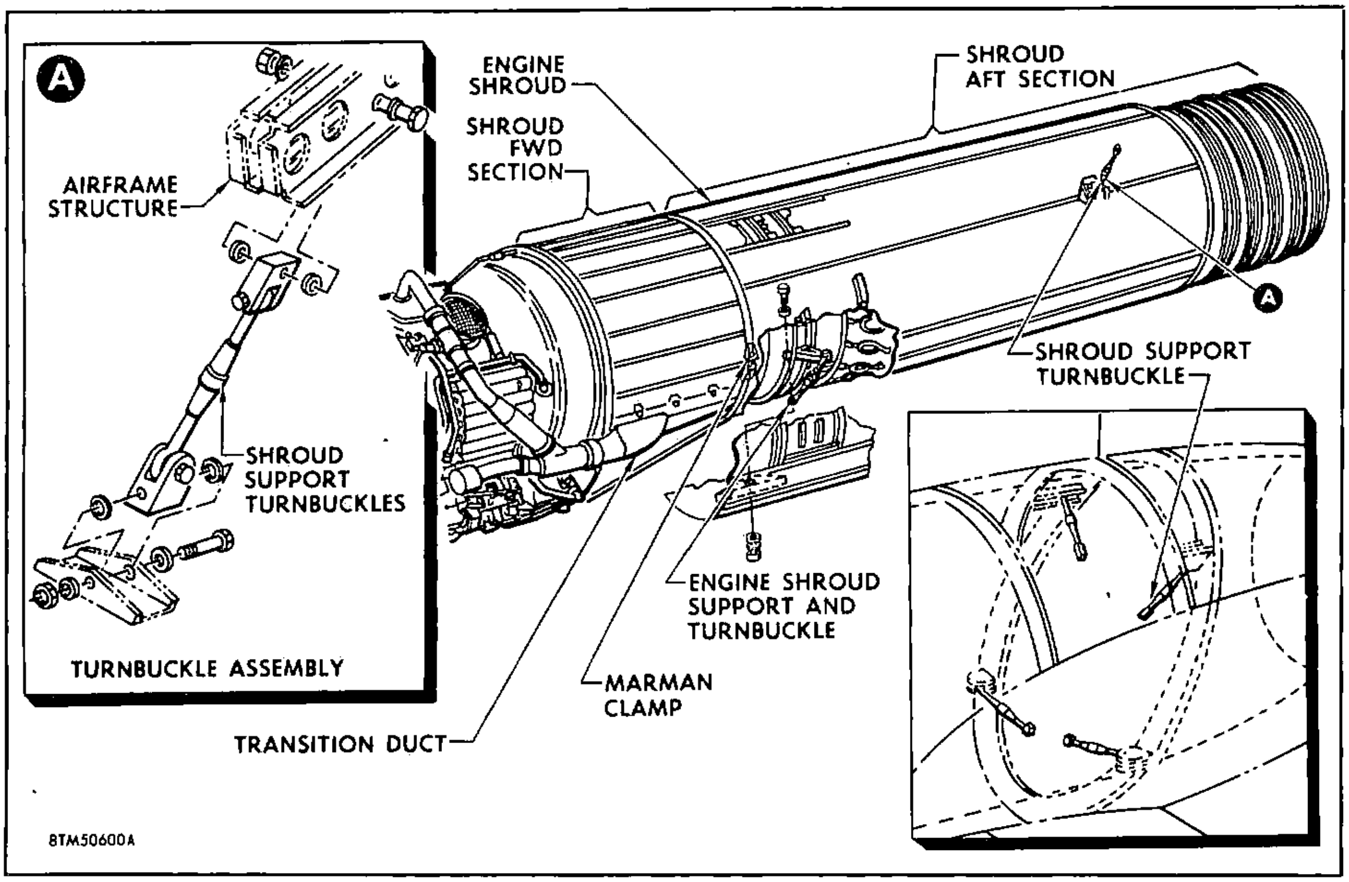


Figure 3-21. Engine Shroud Installation

linkage until the alignment ring can be positioned over the engine thrust mount. Keep track of the number of turns that the turnbuckle is adjusted. Then, install the clamp assembly over the alignment ring and the mounting connection, and secure the clamp. Adjust the forward left mount turnbuckle the same number of turns that you did on the right mount turnbuckle. This will suspend the engine in a level position.

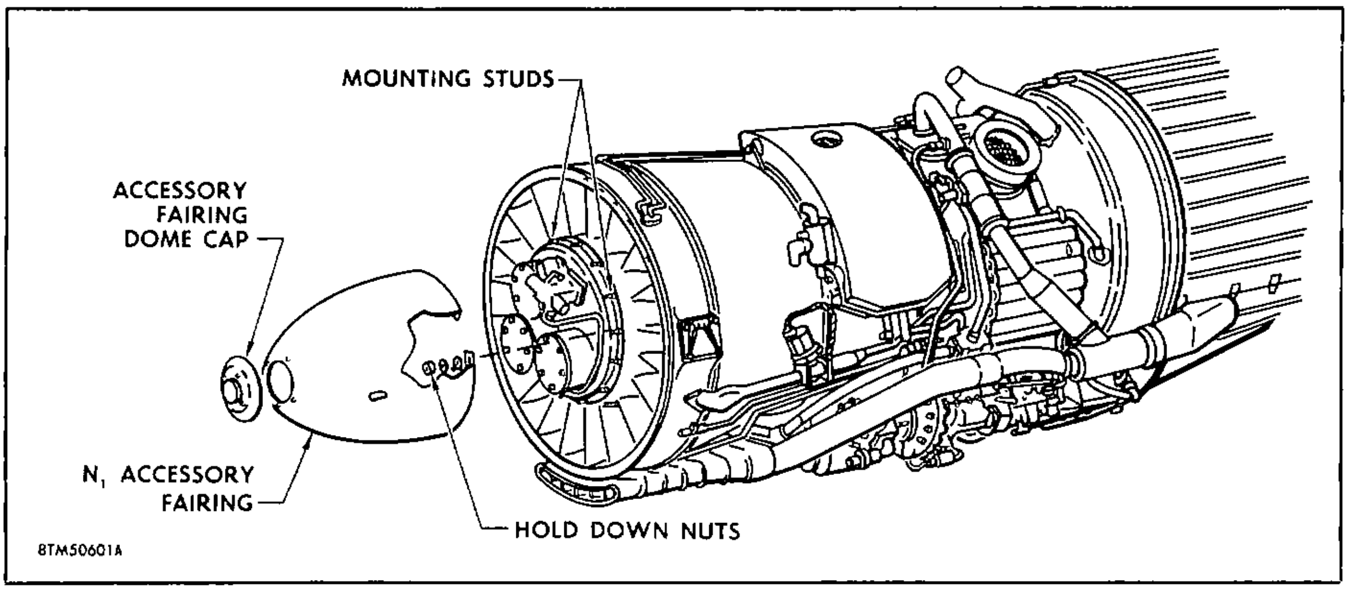


Figure 3-22. N_1 Accessory Fairing Installation

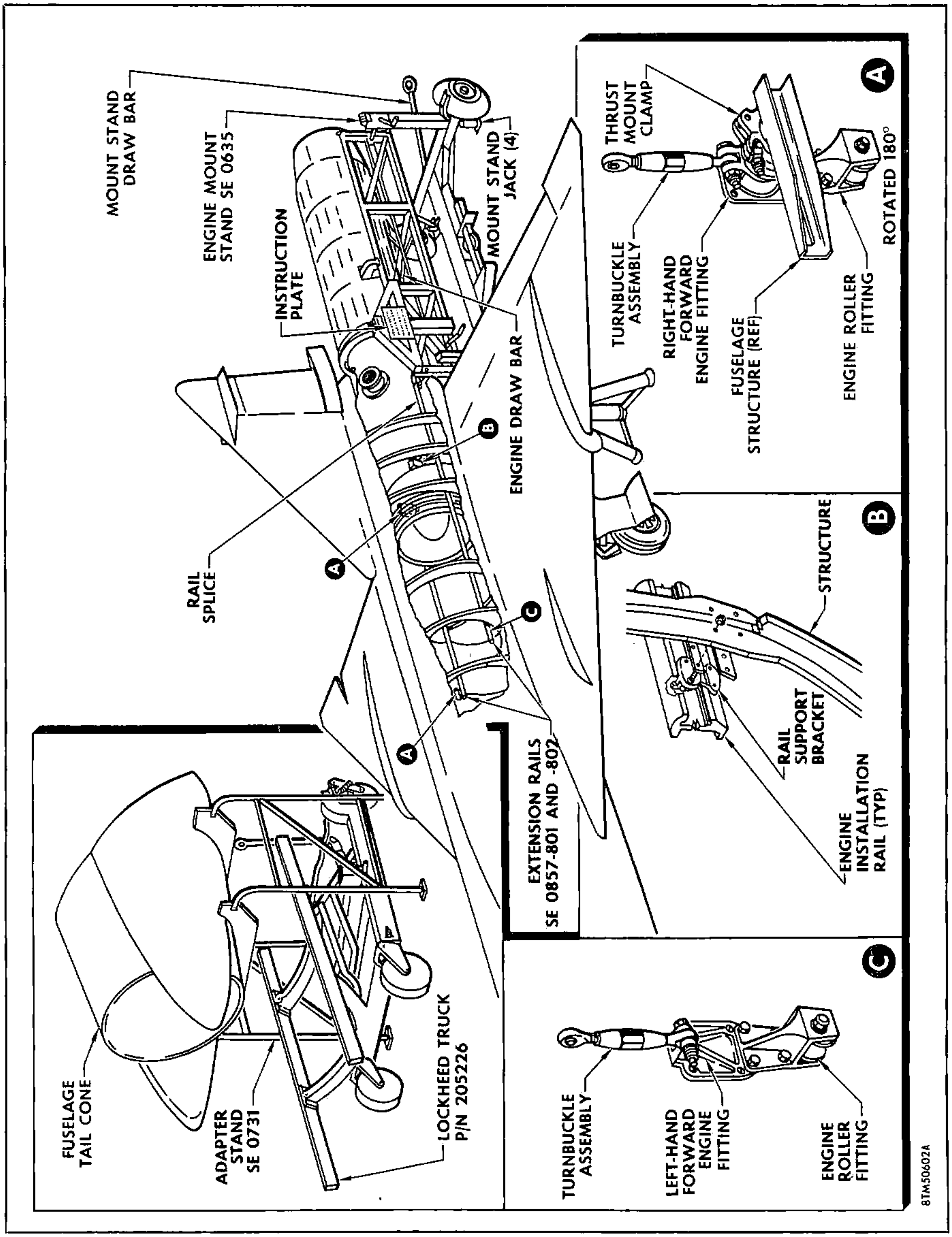


Figure 3-23. Engine Removal and Installation Equipment

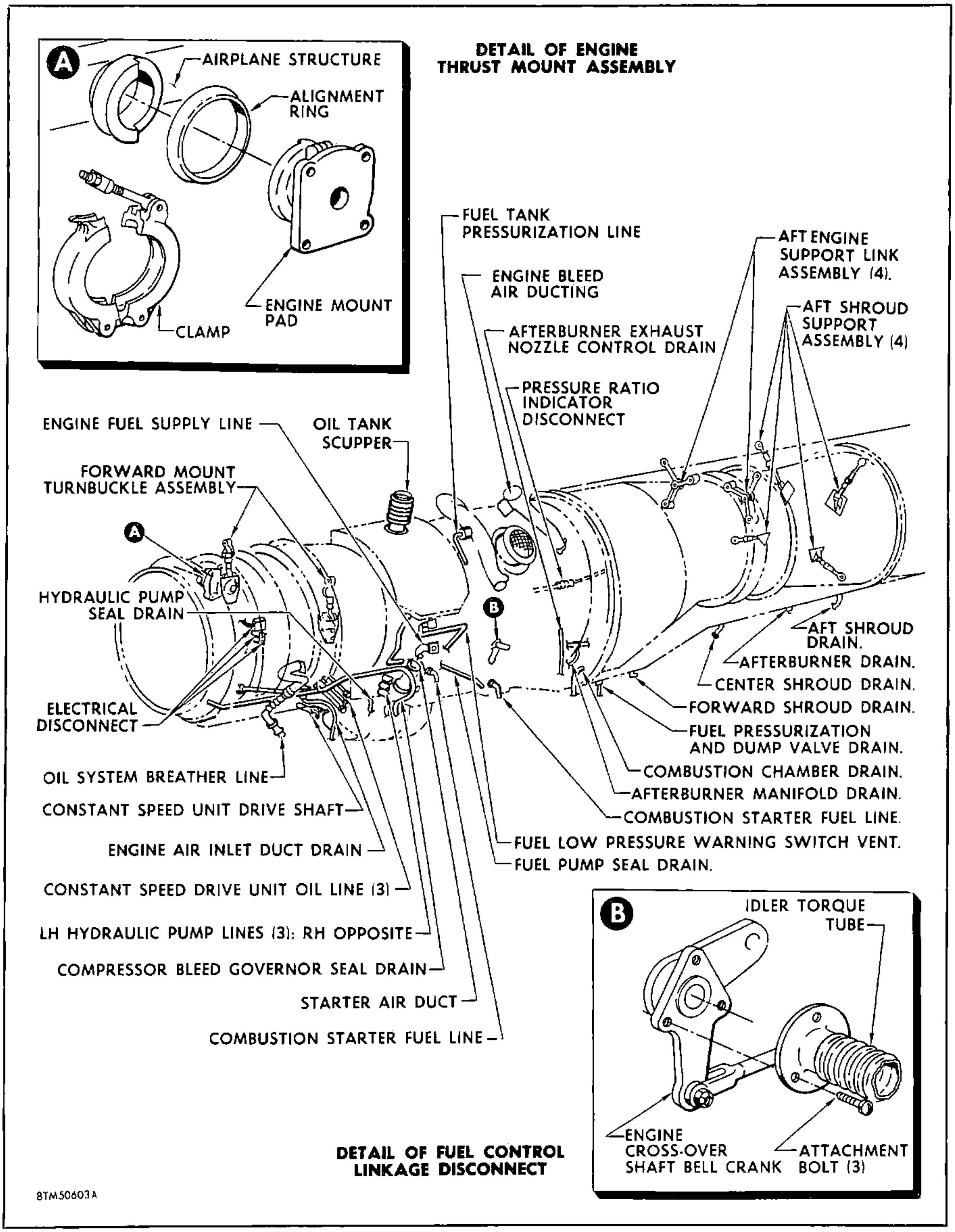


Figure 3-24. Engine Disconnect Points

To position the aft portion of the engine correctly, you must first install the tail cone assembly. All of the tail cone installation bolts need not be used since the cone installation is only temporary at this time. Having the tail cone assembly on the airplane will allow you to adjust the aft engine mounts so that the engine exhaust nozzle will center in the tail cone opening. Then remove the tail cone.

Earlier in this chapter it was mentioned that support blocks were necessary to support the engine shroud on the engine when the engine was not installed in the airplane. Your next step involves attaching the four shroud supports (8 in illustration) to the fuselage and then adjusting them until the shroud is centered around the engine. The support blocks can then be removed. Then safety wire the shroud supports, the forward mount support turnbuckles, and the aft engine mount adjustment bolts.

The J57 engine installation operation, from this point on, consists of making the connections at all of the engine disconnect points and installing the access doors. All connection points are accessible through the engine bay access doors. Once the engine has been "hung" in the airframe, there are about 28 separate steps to be accomplished to complete all engine connections. After completing all connections, the tail cone can be permanently installed. As the last step, install or close all of the access doors.

You will find that a wise and safe practice to follow during engine installation is the checking off of each item as you accomplish it. This practice will preclude any inadvertent ommissions. The preceding description of the engine installation process should give a general, overall view of the engine installation operation. For detailed information, refer to the power plant maintenance technical order, T.O. 1F-102A-2-4.

ENGINE REMOVAL.

Removing the J57 engine from the F-102A is just about the reverse of the engine installation operation discussed thus far. The preparation procedure, however, is exactly the same. As you will recall, this preparation procedure involved: stabilizing the airplane, removing the tail cone, installing the engine installation rails, and positioning the SE-0635 engine mount stand so that the rails could be spliced together. The rest of the engine removal operation is just the reverse of the installation procedure.

In some cases, the engine will be out of service for a very short time, and, in this event, it will not need to be preserved. Where the engine is going to be out of service for an extended period, however, it must be preserved to prevent corrosion within the engine. The engine preservation operation has already been outlined in this chapter.

SUMMARY.

In this chapter you have learned about the handling and maintenance requirements of the J57 engine when it is not installed in the F-102A. General practices and procedures for preserving and depreserving the engine were given to acquaint you with the need for engine preservation as well as with the operation itself. This chapter also outlined the engine build-up operation that is necessary to prepare a new or overhauled engine for installation in the F-102A. The last portion of the chapter was devoted to the engine installation and removal operations as they apply to the F-102A. In many cases, your maintenance duties will not include performing any of the engine maintenance, except for that required for installed engines on the flight line. Regardless of this, however, the knowledge of all engine requirements will prepare you to become a better J57 engine maintenance man.

Chapter IV

POWER PLANT INSTRUMENTS AND WARNING SYSTEMS

Contents *Page*

The engine instrument group, as the term implies, includes those instruments that make the pilot aware of the power plant operating conditions. Present-day jet engines, such as the J57, require some instruments that are not required by reciprocating engines. On the other hand, some of the instruments needed on reciprocating engines are not needed by jet engines. You will find that the engine in the F-102A (the J57) has only five instruments, but each of these instruments is vitally important to the successful operation of the engine.

The F-102A engine instruments and their positions on the instrument panel are shown in figure 4-1. These instruments include the pressure ratio indicator, tachometer, exhaust temperature indicator, exhaust nozzle position indicator, and the fuel flowmeter. The alternate pressure ratio indicator which you will find on some F-102A airplanes is also shown at the bottom of figure 4-1. All of these instruments and their associate systems will be discussed in this chapter.

THE TACHOMETER.

One of the instruments which tells the pilot how the engine is performing is the tachometer. Its function is to determine the revolutions per minute at which the engine N_2 compressor is turning and to provide an indication of that speed on the instrument dial.

Several types of tachometers are used in present-day aircraft. In a single-engine airplane, which has the engine mounted directly ahead of the cockpit, a mechanical tachometer is frequently used. This kind of tachometer is driven directly from the engine by a flexible shaft and operates much like an automobile speedometer. However, most modern aircraft use electrical tachometers, which eliminate the need of the direct drive shaft from the engine. The F-102A tachometer consists of an engine-mounted generator which is electrically connected to an indicator in the cockpit, as shown in figure 4-2. The tachometer generator is mounted on the J57 engine accessory drive case and is splined to the accessory drive gearing. When the engine is running, the generator also rotates. Note that the generator is secured to the engine accessory drive case by four mounting studs. As the generator turns, it produces an electrical signal which is transmitted to the indicator. The indicator, in turn, converts this electrical signal into the correct amount of N_2 compressor rpm and then shows this rpm on the instrument dial.

THE TACHOMETER GENERATOR.

The tachometer generator used on the F-102A is attached to the front of the accessory drive housing on the right side of the engine. As you learned in Chapter III, it is pad mounted; that is, the generator is

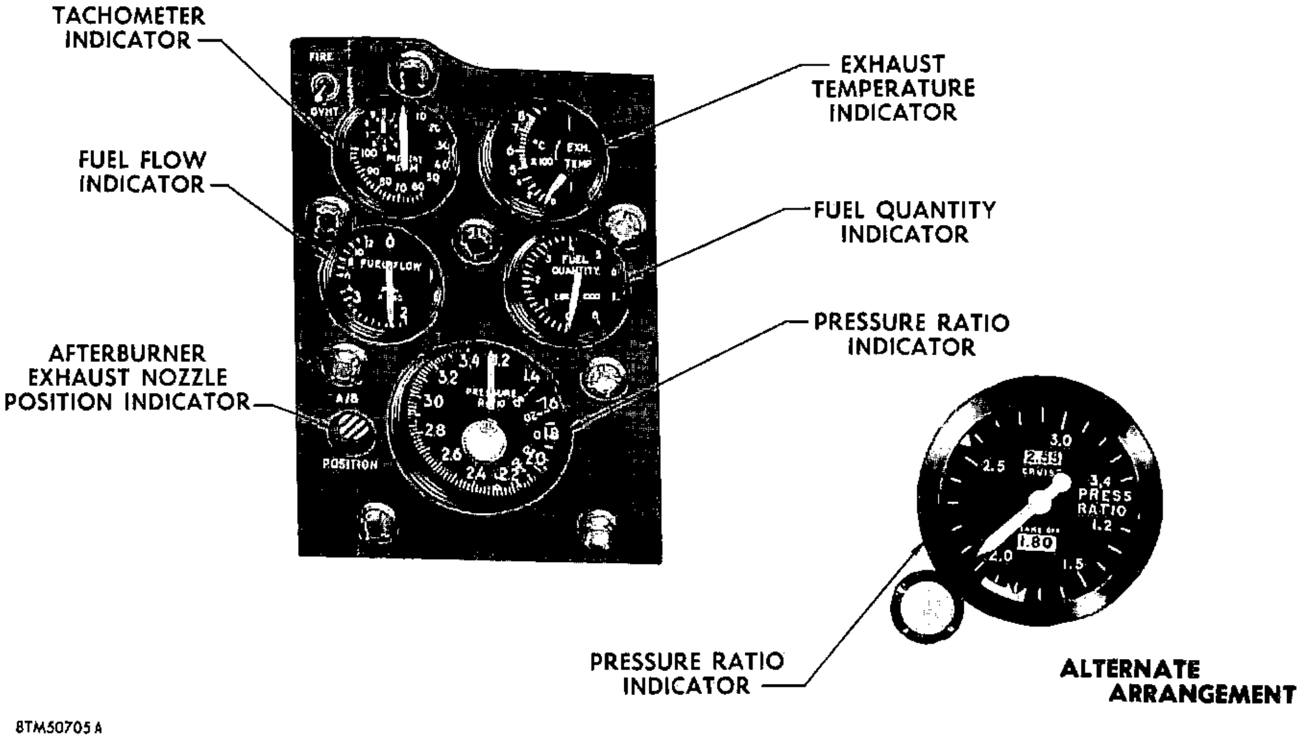


Figure 4-1. Engine Instruments

bolted to a flat, machined surface on the engine accessory housing. The square-end shaft shown on the end of the generator in figure 4-4 projects from the generator into the engine accessory drive section where it is turned by the accessory drive gears. When the engine is operating at maximum rated rpm, the tachometer generator turns at 4200 rpm.

The tachometer generator shaft runs through the center of the rotor (armature) shaft. A pin connects the two shafts at the end opposite the mounting pad. This type of construction permits the generator drive shaft to absorb any vibrations that might result from slight wear and misalignment at the shaft connecting points. Thus, the rotor shaft—and consequently the rotor—will spin at the same rpm as the generator drive shaft. A bearing at each end of the rotor shaft supports the shaft within the generator.

The rotor of the generator is a two-pole permanent magnet, made of very hard steel. Surrounding the rotor is the stator, made of a laminated, soft iron core around a three-phase winding. When the rotor is turned by the drive shaft, current is induced in the windings of the stator. Since the rotor has two poles (one north and one south), one complete revolution of the rotor will result in one cycle of alternating current in each of the stator windings. Three wires come from the stator and carry this induced current to the generator receptacle, and from there to the tachometer indicator in the cockpit.

THE TACHOMETER INDICATOR.

There is considerable similarity between the tachometer generator and the basic mechanism of the tachometer indicator. This should be understandable since the indicator incorporates a synchronous motor which has many of the same characteristics of the tachometer generator. The N_2 compressor rpm in the F-102A is indicated by the tachometer indicator in the percent of the engine's maximum rated speed. The dial of the indicator is calibrated from 0% to 100% in 2% increments, beginning at the top of the dial and extending around for 270°. The large pointer on the indicator rotates around the large dial face. In figure 4-2, you will note that the indicator also has a small pointer and dial in the upper left portion of the large dial. This small pointer rotates around its dial once for every 10% change in engine rpm. Using both the large and the small scales, the indicator can actually show up to 110% engine rpm.

As you might already know, the J57 engine rotates at approximately 10,000 rpm when it is developing maximum thrust. Some J57 engines, however, will develop

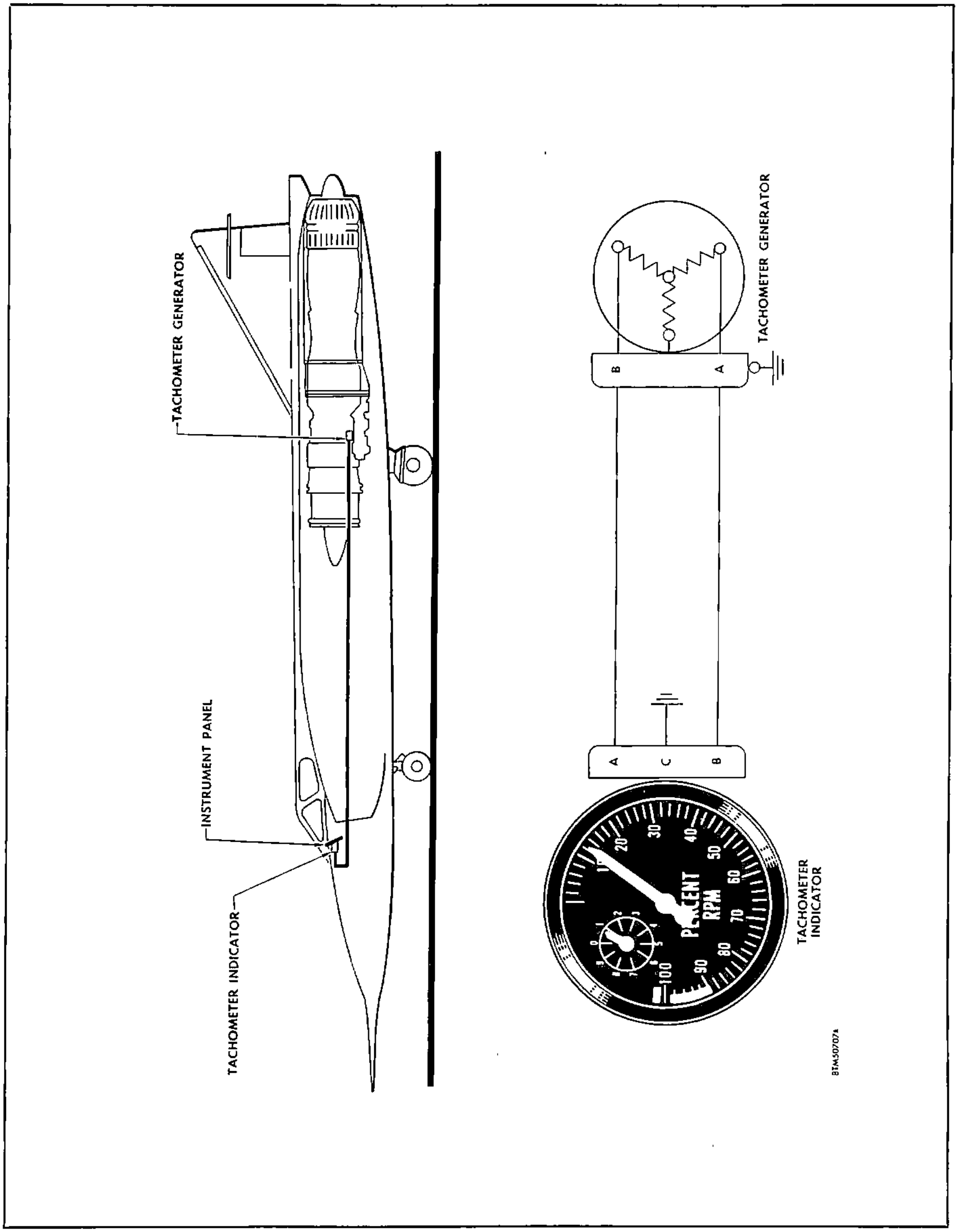


Figure 4-2. The Tachometer Indicating System

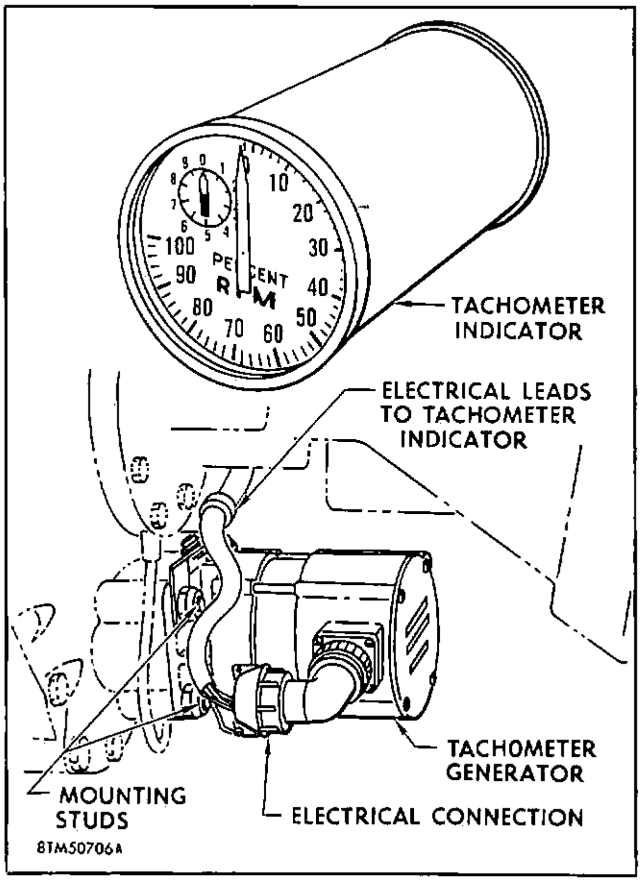


Figure 4-3. The Tachometer and Tachometer Generator

their maximum thrust at an rpm percentage figure that is less than 100%. This variation in rpm percentages in different engines is a natural outgrowth of modern jet engine design and cannot be eliminated. The rpm at which a jet engine develops its maximum thrust is always stamped on the engine data plate. If you have a J57 engine, for example, that has 9600 rpm stamped on its data plate, that engine will develop its maximum thrust when the tachometer indicator is registering only 96%.

You should realize then, from the above paragraph, that a slight variation in engine rpm is not sufficient cause to trouble shoot the tachometer indicating system. The important thing to remember is that the tachometer indicator should register about the same rpm that is stamped on the engine data plate when the engine is developing its maximum thrust.

As just mentioned, the tachometer indicator incorporates a synchronous motor which is quite similar to the one in the tachometer generator. The electrical power that operates the indicator motor comes from the tachometer generator. Since the indicator stator is wound in the same manner as the indicator stator in the generator, a rotating magnetic field is set up in the stator as it receives the induced current from the generator. The rotor in the indicator motor is a permanent magnet. As you probably know, a permanent magnet will line up in the magnetic field around an electrical conductor. Because of this fact, the indicator rotor is dragged around by the rotating magnetic field set up by the generator current. Since the tachometer is geared directly to the engine accessory drive section, the rpm of the indicator rotor is an accurate indication of the rpm of the N_2 compressor.

The exploded view of the tachometer indicator (figure 4-5) will give you a good idea of how the indicator rotor action is turned into a dial indication of engine rpm. Note that the shaft which turns the indicator rotor also turns another permanent magnet called the drag magnet. Around the drag magnet is the drum, which—because of proximity to the drag magnet—is dragged around with the drag magnet. The hairspring, however, lets the drum rotate just so far. The amount that the drum can rotate depends upon the amount of current from the tachometer generator, and the strength of the holding hairspring inside the indicator. Since the hairspring strength is always constant, the drum movement will be directly proportional to the current from the tachometer generator and the engine rpm. From the preceding description you should remember that the rpm of the engine is transmitted to the tachometer generator by gears, then to the tachometer indicator electrically, and finally to the indicator pointer by the magnetic force of the drag magnet.

MAINTENANCE OF THE TACHOMETER INDICATOR SYSTEM.

One of the most frequent troubles with tachometer systems—and especially electrical tachometers—is that

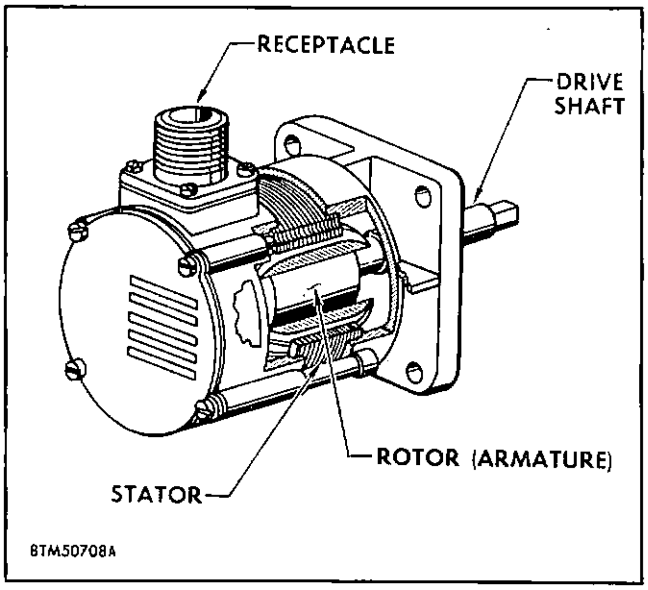


Figure 4-4. Cutaway View of the Tachometer Generator

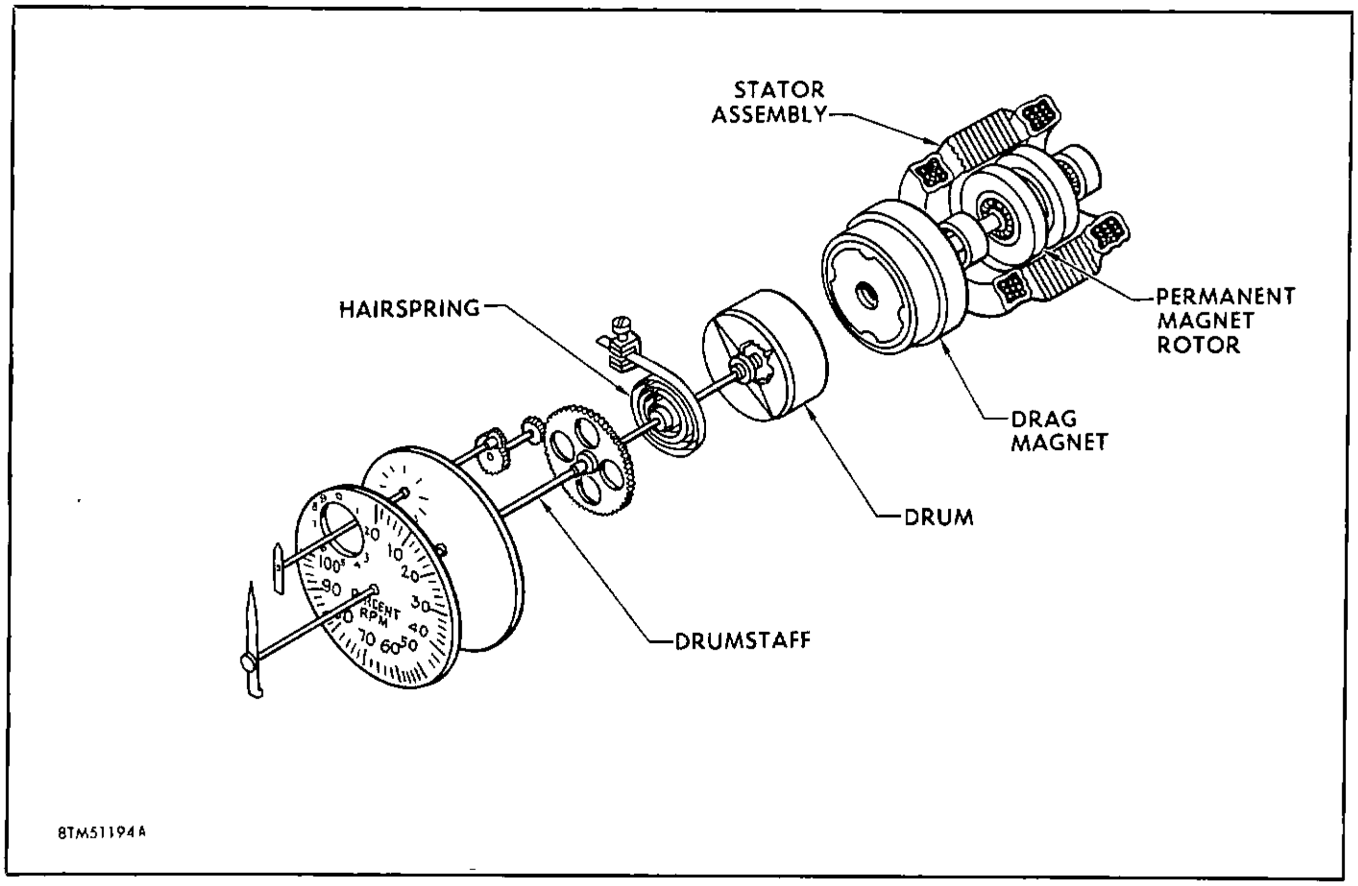


Figure 4-5. Exploded View of the Tachometer Indicator

the indicator pointers have a tendency to stick or "hang up" during engine starting. This type of trouble, however, is not a malfunction of any part of the system, and it can be remedied by lightly tapping the instrument panel in the immediate vicinity of the tachometer indicator. If the indicator needle still sticks after the engine is started and you have advanced the power lever into the IDLE range, then something is definitely wrong and you will have to shut down the engine and trouble shoot the tachometer indicating system.

An inoperative tachometer system can be traced to the indicator, generator, or the connecting electrical harness. Since you are not allowed to perform any internal maintenance or repairs on either the generator or the indicator, the easiest way to trouble shoot the system is to replace the components. An accepted method for line maintenance personnel is to first replace the indicator with an instrument that is known to be functioning satisfactorily. This will tell you whether the trouble is in the indicator or in the generator and wiring. If the condition persists after the indicator has been replaced, replace the tachometer generator. If the condition still has not been cured, then you will have to either replace the electrical harness or run a continuity check to determine whether there are any breaks or shorts.

THE PRESSURE RATIO INDICATOR SYSTEM.

On reciprocating engines, the pilot needs two basic indications to determine how much power the engine is developing. These two indications are engine rpm and engine manifold pressure. On jet engines the pilot must also have two basic indications to know how much power the jet engine is developing. One of these indications, the N_2 compressor rpm, was discussed in the preceding section. The other indication is the ratio of the engine exhaust pressure to the intake air pressure.

This pressure ratio is very important on all jet engines; especially on dual spool (two-compressor) type turbojet engines, such as the J57. In the dual-spool type of engine, it is possible for one of the compressors to operate at near-stall conditions, without the tachometer showing any appreciable change in rpm and without the pilot being aware of it. To really know how the engine is performing, the pilot needs an accurate measurement of the thrust that the engine is producing. The pressure ratio indicator provides this information.

Two different types of pressure ratio systems are used in the F-102A airplane. Early airplanes use the *direct*

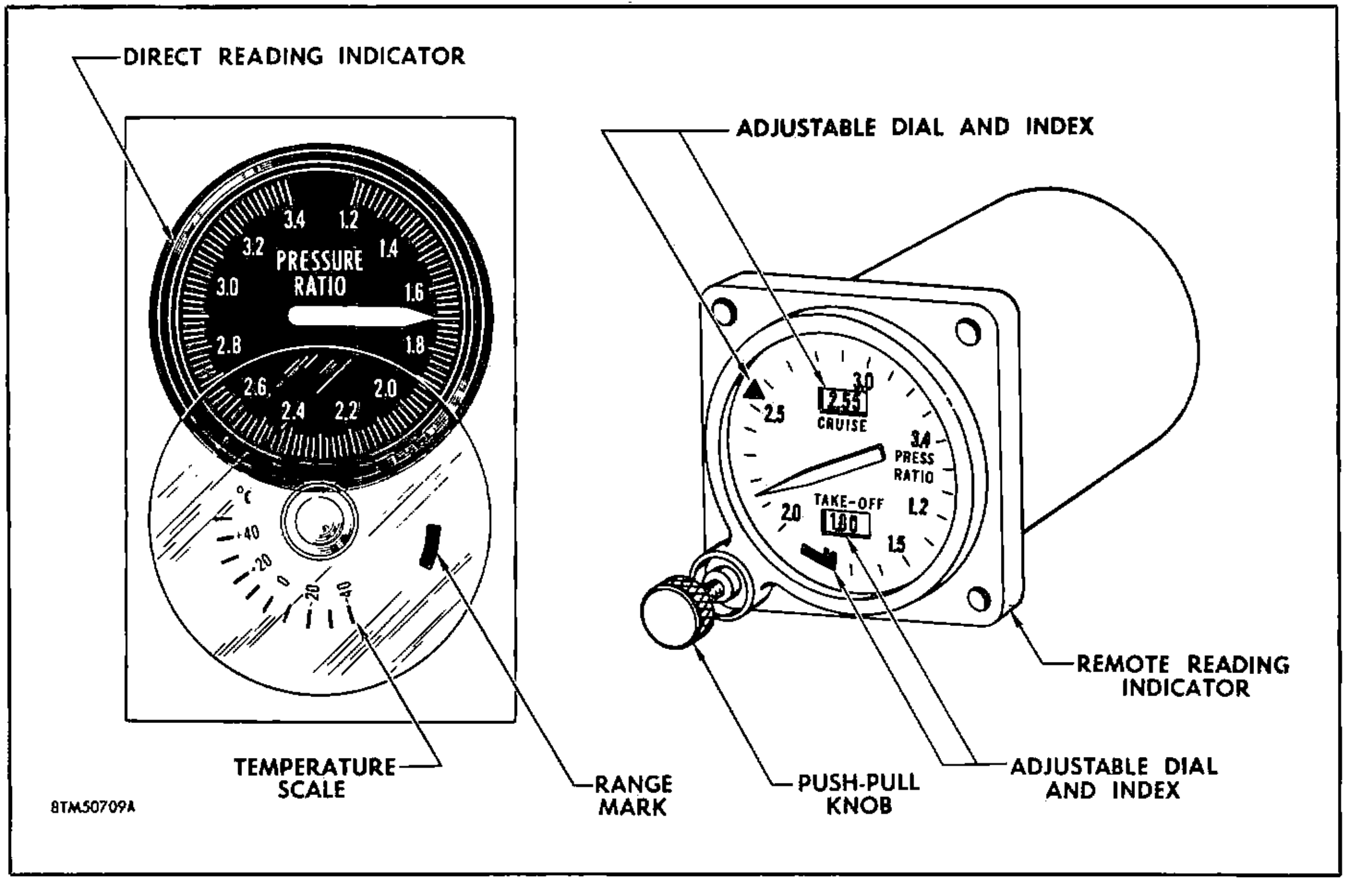


Figure 4-6. The Direct and Remote Pressure Ratio Indicators

reading system, while the later models use the *remote reading system.* In the direct reading system the pressures which are to be compared (intake and exhaust), are piped from the sensing areas and introduced directly into the indicator instrument case. In the remote reading systems, however, a remote transmitter is used to measure the two pressures and then send electrical signals to the indicator in the cockpit. Both the direct and the remote reading indicators are shown in figure 4-6. Each type of indicator is discussed individually in the following paragraphs.

THE DIRECT READING INDICATOR.

The direct reading indicator has a fixed dial that is calibrated from 1.2 to 3.4 to show the ratio of the tailpipe exhaust pressure to the engine intake pressure. The indicator also has a transparent dial which, in figure 4-7, has been detached so that you might see its dial markings better. This transparent dial can be rotated by manually turning the knob in the center of the dial.

To use the pressure ratio indicator correctly, the pilot must first know the temperature of the outside air. This information is provided by the outside air temperature indicator. By turning the center knob on the pressure ratio indicator, the pilot can set the transparent dial so that the outside temperature will coincide with the temperature index mark on the fixed dial.

If the outside temperature were 0°C, for example, the pilot would turn the transparent dial until the 0° mark was lined up with the temperature index mark. The range mark on the transparent dial would then indicate the most efficient power lever settings. The pilot would move the power lever until the pressure ratio indicator pointer was in the dial area covered by the range mark.

How the Direct Reading System Indicates Pressure Ratio.

Figure 4-8 shows an operational schematic view of the direct reading type of pressure ratio indicator. The diaphragm within the frame assembly is the tailpipe pressure diaphragm. When exhaust pressures enter this diaphragm, the diaphragm expands and raises the entire frame assembly. The amount of expansion depends upon the difference between the exhaust pressure inside the diaphragm and the pressure on the outer diaphragm surface.

The frame assembly can also rotate, or pivot, on its two axis points. This frame assembly movement is

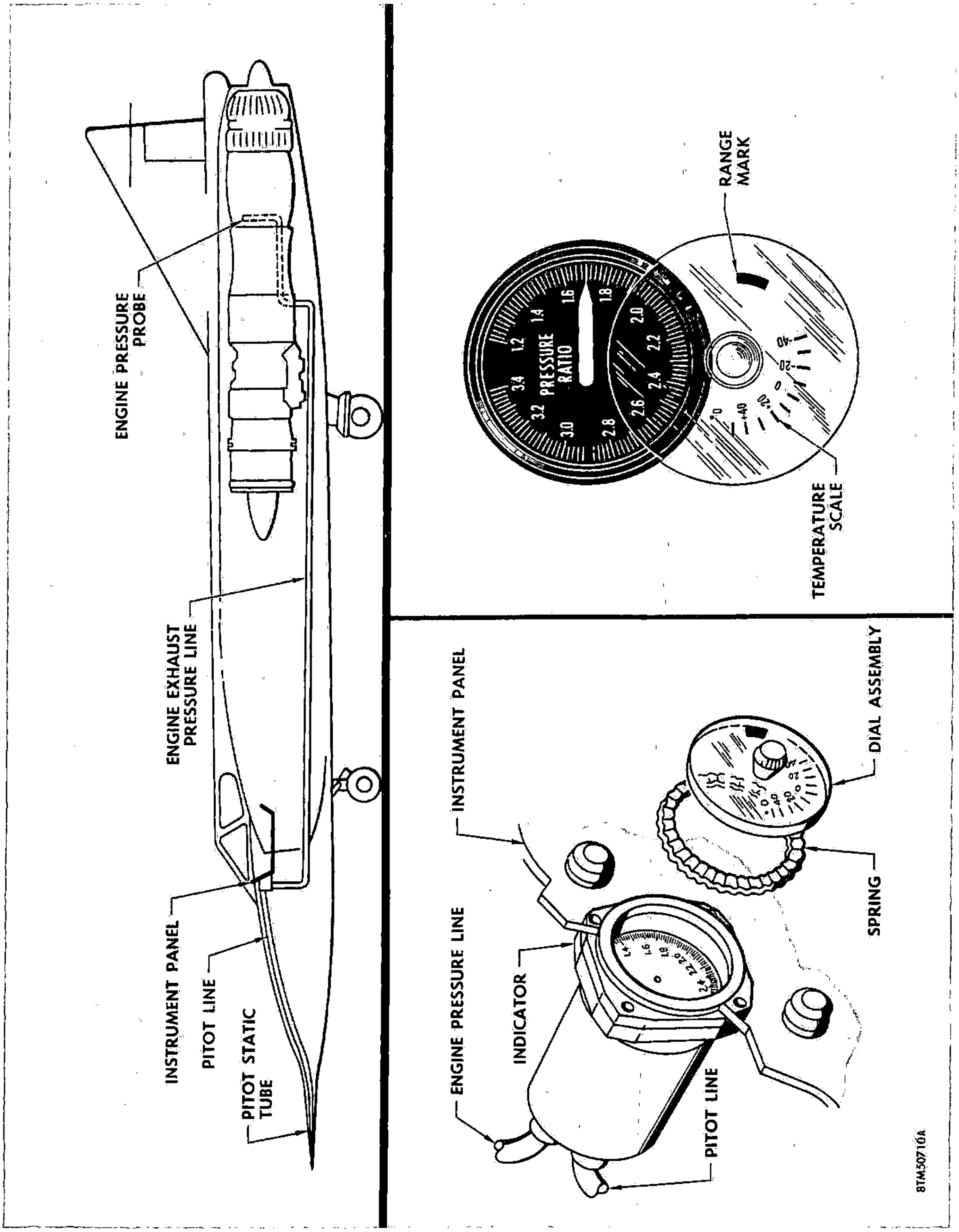


Figure 4-7. The Direct Reading Pressure Ratio Indicator System

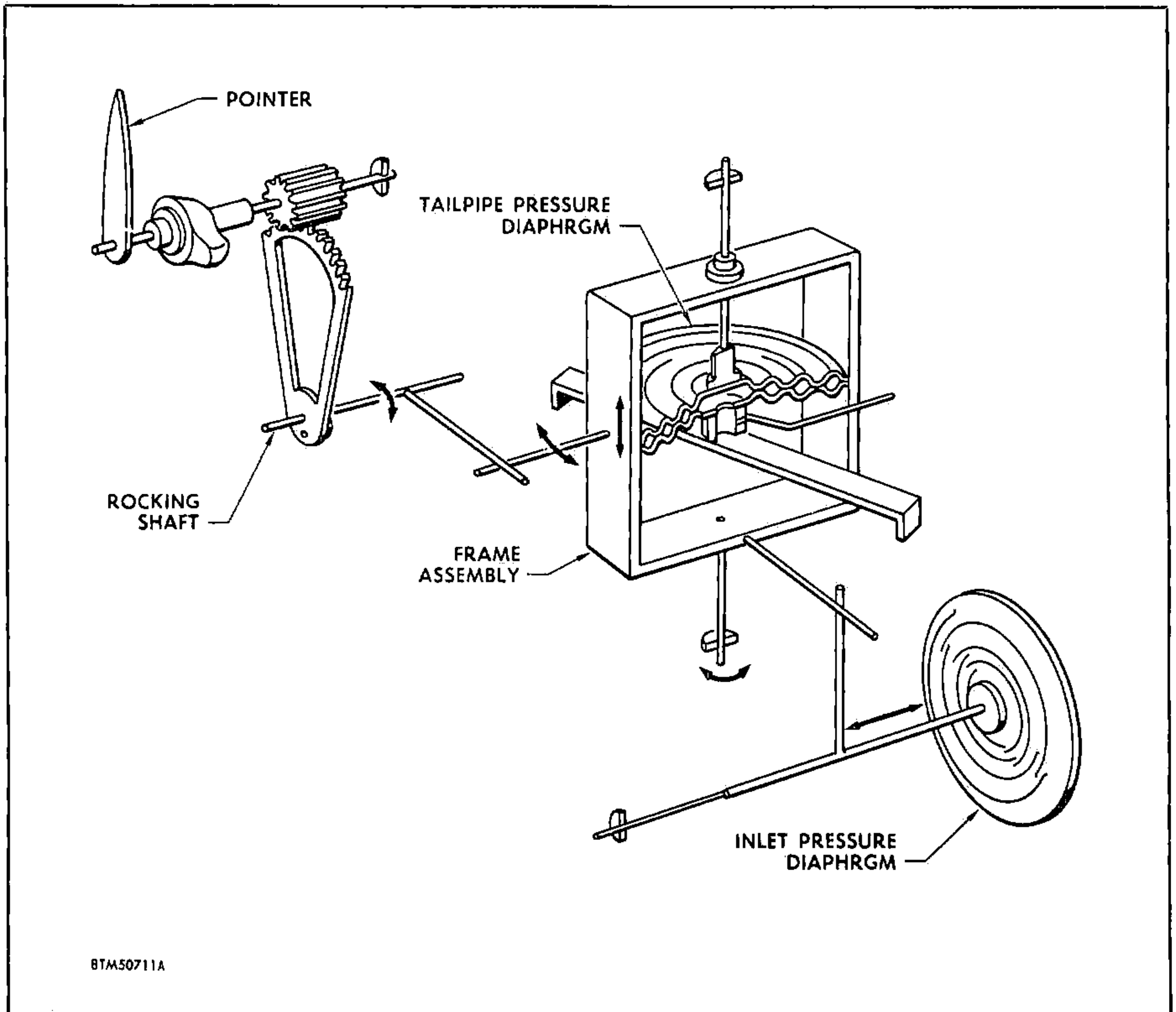


Figure 4-8. Direct Reading Pressure Ratio Indicator Operational Schematic

controlled by the inlet pressure diaphragm. This diaphragm is a sealed, evacuated unit (aneroid type). Since the internal pressure of this diaphragm is always constant, the diaphragm will expand and contract according to pressure changes on its outer surface. The pressure on the outer diaphragm surface is the engine inlet pressure. As the engine inlet pressure increases, the diaphragm contracts and allows the frame assembly to turn away from the rocking shaft. When the inlet pressure decreases, however, the diaphragm expands and pushes the frame assembly around towards the rocking shaft.

Note that the rocking shaft is geared to the indicator pointer. The rocking shaft is turned by the up-and-down movement of the frame assembly. As just noted, the up-and-down movement of the frame assembly is controlled by the tailpipe pressure diaphragm. However, the amount of pointer movement that can result from any specific tailpipe pressure depends upon how close the frame assembly is to the rocking shaft; and, as we have also noted, this depends upon the inlet pressure diaphragm. Note that the closer the frame assembly moves to the rocking shaft, the more the rocking shaft will be turned by a given amount of vertical displacement of the frame assembly.

From the operational schematic, (figure 4-8), you can see that the indicator pointer movement is controlled by the tailpipe pressure and biased by the engine inlet pressure. By using constant mechanical advantages at each of the mechanical links in the indicator, the direct reading indicator can show the ratio of the engine exhaust (tailpipe) pressure to the engine inlet pressure.

There is one disadvantage in using a direct reading type of pressure ratio indicator; exhaust gases are piped directly into the cockpit of the airplane. A dangerous condition could result if the exhaust gas line were to leak in the cockpit area. To eliminate this possibility, a remote reading type of indicating system is installed in the later models of the F-102A.

THE REMOTE READING PRESSURE RATIO INDICATOR SYSTEM.

If you understand the operation of the direct reading type of indicator, you should have no trouble understanding the remote reading type of indicator. It furnishes the same information to the pilot, and in a similar manner. The measuring mechanism, however, is mounted in the fuselage area between the two engine intake ducts. This measuring mechanism is called a transmitter, and it connects electrically to the remote reading indicator in the cockpit. As you will remember from figure 4-1 in this chapter, the remote reading pressure ratio indicator uses the same mounting hole on the instrument panel that the direct reading indicator does.

The remove transmitter has two diaphragms just as the direct reading indicator does. Any pressure differences in either the engine inlet pressure or the exhaust pressure are sensed by these two diaphragms. But, instead of transmitting the movement of the diaphragm to the rocking arm and then to a pointer, the remote transmitter has a small synchronous motor which the sensing diaphragms control.

According to the amount of diaphragm movement, the remote transmitter will send an electrical signal to the remote reading indicator in the cockpit. The indicator also has a synchro which is controlled by the signal from the transmitter, and it moves the indicator pointer an amount that is proportional to the amount of diaphragm movement in the transmitter.

The Remote Pressure Ratio Transmitter.

Figure 4-9 shows a side and end view of the remote pressure ratio transmitter assembly. Note on the end view that there are two tube fittings—one marked LOW and the other marked HIGH. The low-pressure connection receives the inlet air pressure while the high-pressure connection receives the exhaust pressure. In the center, you can see the electrical receptacle that provides power to the synchro transmitter inside the transmitter assembly, and carries the output signal from the transmitter to the synchro receiver in the indicator. The pressure and electrical connections are all on the base of the unit, rather than on the transmitter itself, because the base is attached firmly to aircraft structure and the transmitter is shock-mounted to the base. The curved tubes from the base to the transmitter carry the pressures to the transmitter case without interfering with the shock absorbing mountings.

HOW THE REMOTE TRANSMITTER PRODUCES A SIGNAL. As previously mentioned, the remote transmitter has two diaphragms inside an airtight case. One diaphragm is an evacuated bellows acted upon by inlet air pressure alone; the other diaphragm has exhaust pressure inside and inlet air pressure outside. An increase or decrease in either of these pressures will tend to move the diaphragms. The movement of the diaphragms is passed through mechanical linkage and gears and results in the rotation of a rocking arm. This rocking arm is connected to the rotor of a synchro-transmitter inside the case. As the rocking arm is rotated, the rotor of the synchro-transmitter is rotated proportionally.

The rotor is connected to an electrical source of power; and, as it is positioned within the stator windings, it will induce a specific current within each of the three windings. The rotor of the synchro-transmitter and the rotor of the synchro-receiver (in the indicator) are connected in series; and, as a result, both rotors will seek identical positions within their respective stators. This means that the same voltage is being induced into the stator of each of the transmitters. This, in effect, transmits an accurate indication of the ratio of pressures to the calibrated scale on the indicator dial.

The Remote Reading Indicator.

There are several differences between the remote indicator and the direct reading instrument. Note in figure 4-10 that the dial of the remote indicator is calibrated for the same limits as the direct reading indicator, but it is positioned differently. There are two index markers and two windows in the face of the dial. The readings in the small windows, and the position of the index markers, can be changed by the push-pull knob on the front of the indicator.

When the knob is pushed in, it will turn the numbers in the lower window and reposition the lower index marker. When the knob is pulled out, it will turn the numbers in the upper window and reposition the upper index marker. The upper window and the marker indicate the cruise setting desired by the pilot, while the lower window and marker indicate the takeoff reading which should show during takeoff. These dials and markers are mechanically connected to the knob and manually set by it. They are in no way affected by the indicating mechanism inside the indicator that turns the pointer.

Inside the indicator case is a synchro-receiver, a magnetic amplifier, and a servo motor. The position of the synchro in the transmitter unit provides the signal to the indicator through an electrical connection in the rear of the indicator case. The signal positions the synchro-receiver which in turn sends a signal via the amplifier to the servo motor. The servo motor then turns the point shaft to deflect the pointer and indicate the pressure ratio.

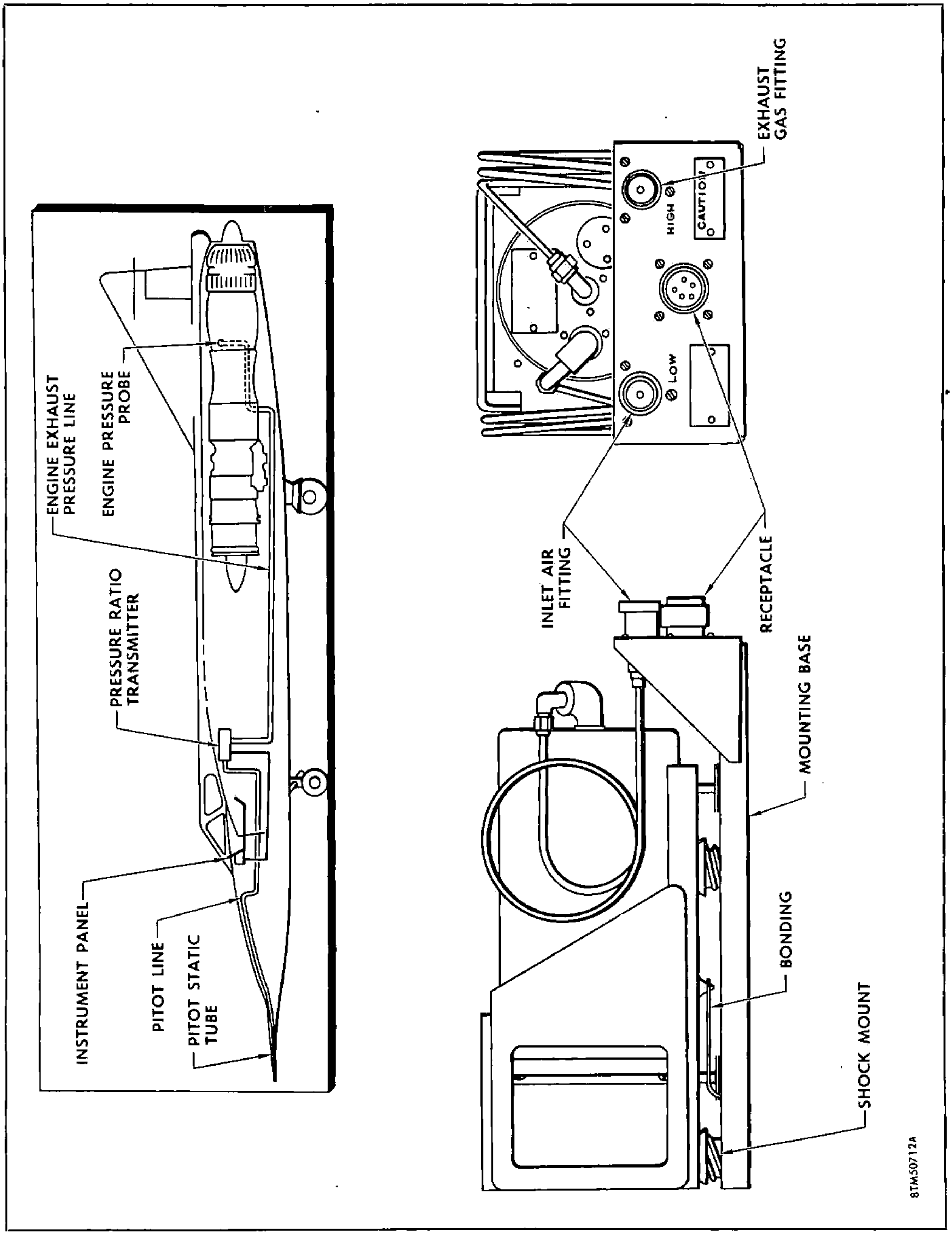


Figure 4-9. Remote Pressure Ratio Transmitter

To set the remote indicator properly, the pilot of the F-102A must do two things. First, he contacts the control tower to get the existing temperature at the field and the estimated temperature at the altitude at which he intends to cruise. He then consults his performance charts to determine what the settings on the indicators should be for the two temperatures. When he turns the push-pull knob, the index marker and the numbers in the windows will show the same reading.

You can see an example of these readings in figure 4-10. Note that the upper index marker is between 2.5 and 2.6 while the indication in its corresponding window is the same, or 2.55. Note also that the lower window shows a reading of 1.80 and that its index marker is at the corresponding position on the dial.

The windows serve only to give the pilot an accurate and simple method of telling what his index setting is. Note that the lower index (for takeoff) has a raised portion on one end. This is the most desirable position for the pointer during takeoff. However, it is safe to proceed with takeoff even if the needle goes beyond that point, or as long as it remains within the zone covered by the range marker. When the aircraft reaches the selected cruise altitude, the pointer should match the upper cruise index marker. The pilot then knows that his engine is performing efficiently.

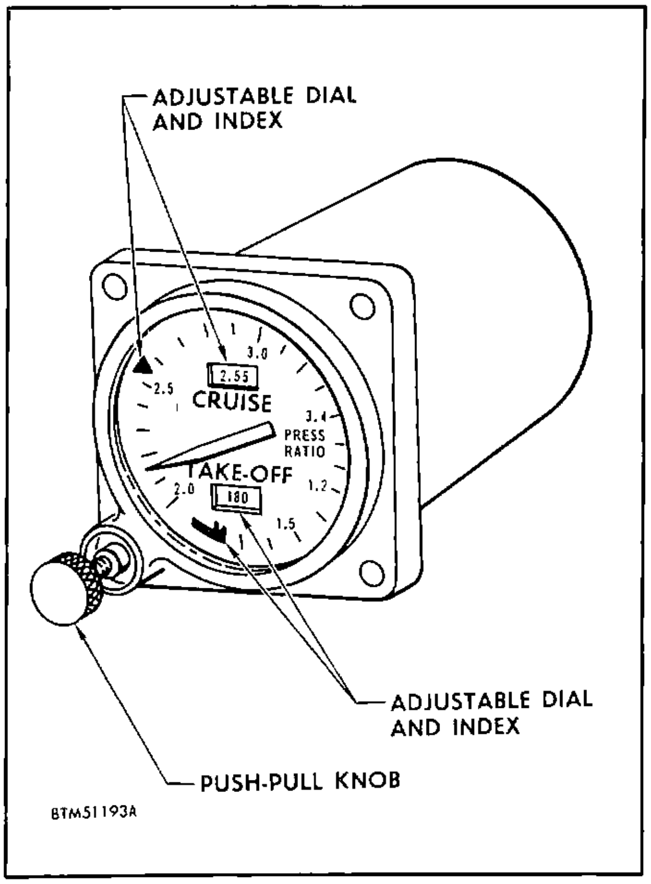


Figure 4-10. Remote Pressure Ratio Indicator

MAINTENANCE OF THE PRESSURE RATIO INDICATOR SYSTEM.

Any troubles in the pressure ratio indicating system can usually be traced to a faulty transmitter, faulty indicator, or clogged sensing lines. When trouble shooting a malfunctioning indicating system, it is best to first replace the indicator with a unit that is known to be serviceable. If the replaced indicator does not cure the trouble, the same thing should be done with the transmitter. When these two steps do not cure the trouble, you should then check the sensing lines to see whether they are clogged. This involves connecting a low-pressure air source to the sensing lines. It is possible to damage some of the other components in the pitot-static system if the exact pressures and procedures are not used during the line check. Always refer to the F-102A Instruments Maintenance Handbook (T.O. 1F-102A-2-9) whenever you troubleshoot the sensing lines. The last thing to check in the pressure ratio indicating system is the electrical wiring between the transmitter and the indicator. This can be done easiest by disconnecting the wiring from the transmitter and indicator, and then using a continuity light to check each wire for an open circuit or short to ground.

AFTERBURNER EXHAUST NOZZLE POSITION INDICATOR SYSTEM.

All jet engines that have afterburners must have some means of controlling the opening at the engine exhaust nozzle. This control of the afterburner nozzle is necessary, because pressures and temperatures produced by the engine vary between normal burning and afterburning. As previously mentioned, the nozzle is closed during normal burning—that is to say, the opening is restricted. In this position, the exhaust opening is the best size to provide maximum exhaust velocity for the existing temperature and pressure conditions at the exhaust nozzle. If this opening is too large, the maximum thrust cannot be developed by the engine.

When afterburning is initiated, the temperature at the exhaust nozzle increases greatly. This temperature increase causes additional expansion of the exhaust gases, and if the nozzle opening is not enlarged the pressure and temperature in the afterburner will become excessive. The J57 engine has an adjustable exhaust nozzle with an indicator to tell the pilot whether the nozzle is in the open or closed position. This indicator is located on the right side of the instrument panel next to the pressure ratio indicator.

THE EXHAUST NOZZLE POSITION INDICATOR.

The exhaust nozzle position indicator is a simple instrument. It consists of two solenoids which rotate a spring-centered, three-position card within a small

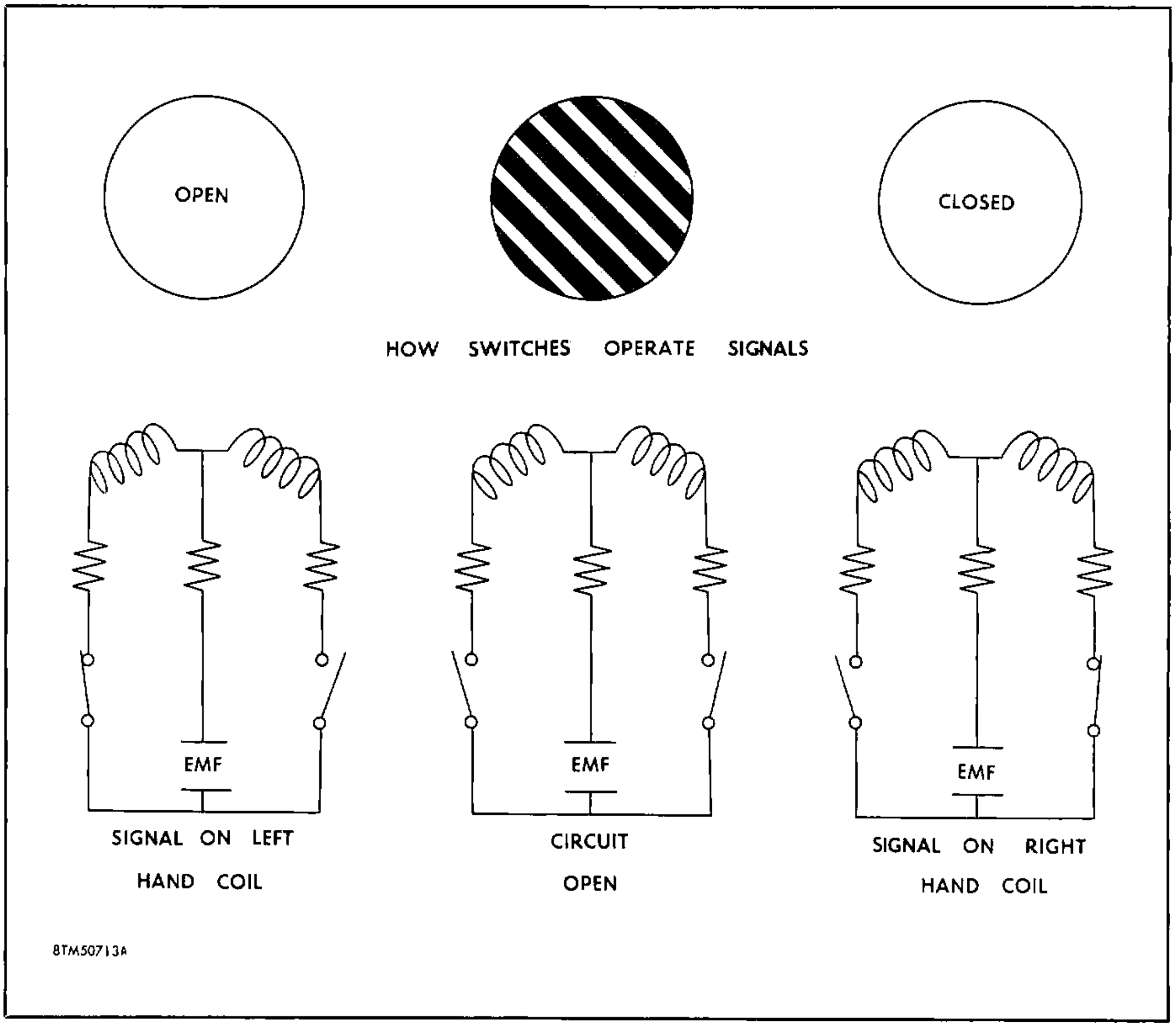


Figure 4-11. Afterburner Exhaust Nozzle Position Indication Circuit

sealed case. A window in the front of the case shows one of the three positions of the card.

There are three indications which the exhaust nozzle position indicator can give; the one which appears in the window at any particular time depends upon whether one of the two solenoids is energized or neither solenoid is energized. In figure 4-11 you can see the three types of indications and the corresponding electrical schematic that causes each of them to appear. Note that there are two switches in the system, only one of which can be actuated at any particular time. In the left diagram, you can see how the *open* solenoid is energized when the left switch is closed. The circuit is completed to ground from the airplane's 28-volt, d-c system, represented by emf (electromotive force). The middle diagram shows both switches open, so neither solenoid is energized. The diagonal lines (barber-pole) appear on the dial because a spring holds the rotating card in the center position. This indication appears when both switches are open or when power failure occurs. In the third diagram, the circuit is completed through the other switch energizing the *closed* solenoid. The CLOSED indication will appear as long as that switch is closed and the power supply is not interrupted.

HOW THE SYSTEM OPERATES.

The two switches are attached to the right side of the engine just below the compressor bleed valve. As you can see in figure 4-12, two cables connect the switch to the nozzle positioning cylinders. The upper cable is the nozzle position cable; the lower cable is a temperature compensating cable. Both are attached to spring plates in the switch assembly and serve to

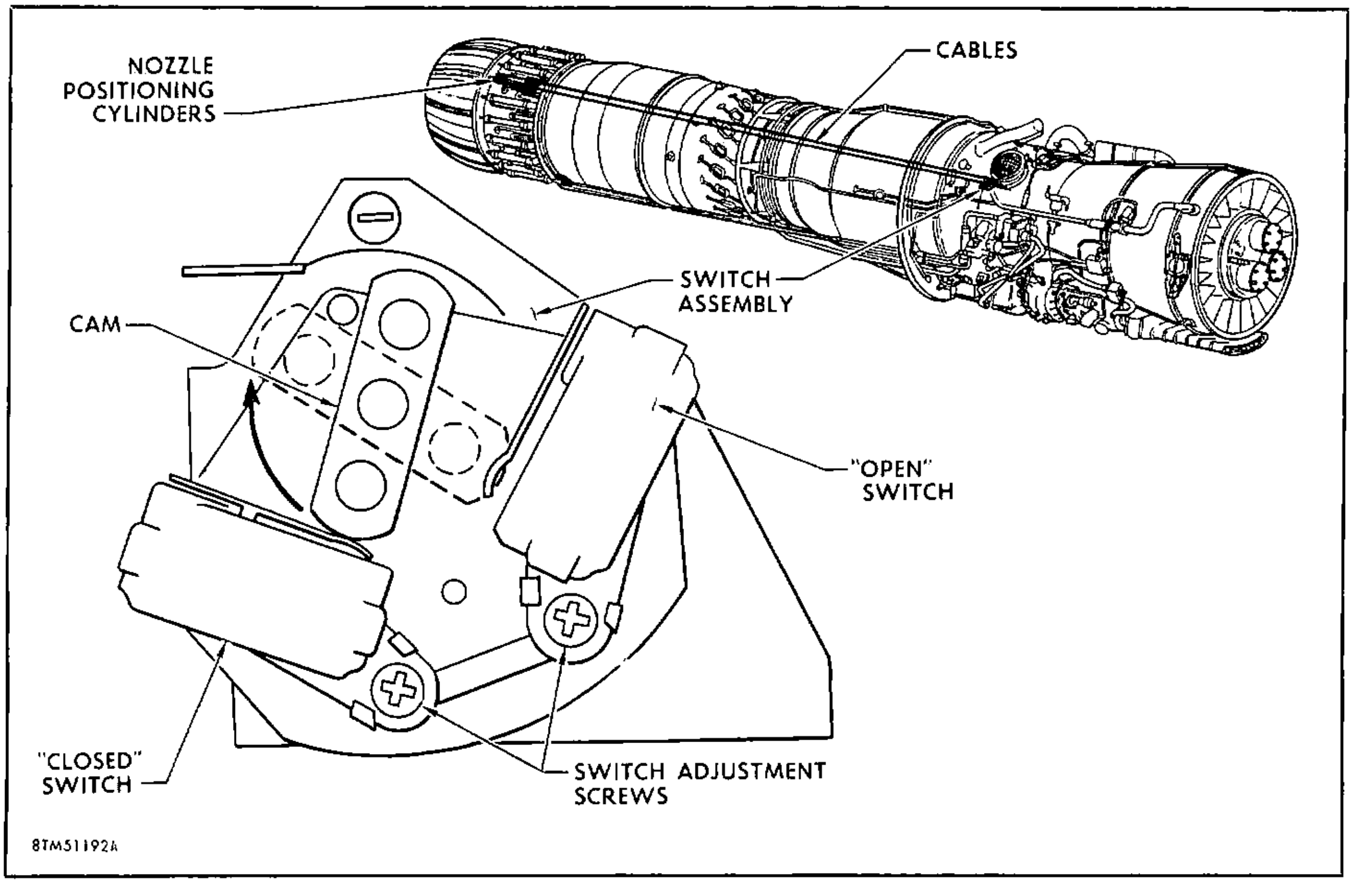


Figure 4-12. Afterburner Exhaust Nozzle Position Switches

rotate a cam. The enlarged view of the switch assembly shows how the cam actuates the switches.

Note that figure 4-12 shows the actuating arm of the *closed* switch depressed by the cam. With the switch in this position, the circuit of the indicator is energized as shown on the right view of the electrical schematic which was just discussed and the CLOSED reading shows in the window. The dotted line showing the outline of the cam shows where the cam stops when the afterburner nozzle is open. This position results in an OPEN indication because the *open* solenoid is energized. It should be obvious from the illustration that both switches are open momentarily as the cam rotates from one switch to the other. Thus, the "barber-pole" shows on the indicator very briefly each time the afterburner nozzle is opened or closed.

You should keep in mind that the switching mechanism does not open or close the afterburner; that operation is accomplished automatically when fuel pressure in the afterburner fuel control reaches certain limits. The sole purpose of the switch is to energize the correct indicator solenoids at the correct time.

HOW THE SYSTEM IS RIGGED.

From time to time, it becomes necessary to rig a cable-operated system. The afterburner position indicator system is no exception. Changing of a component in the system, a faulty indication in the cockpit, or a broken cable is sufficient reason to re-rig the system. First of all—what is rigging? Rigging is the act of making a series of adjustments to a mechanical or electro-mechanical system so that it will perform its function properly. In the afterburner position indicator system we adjust two turnbuckles (and, if necessary, the two switches) to achieve this goal.

To rig the system properly, we must have an SE-0947 rigging gage. Two models of the rigging gage are used: the J57-P-41 engine uses an SE-0947 gage, while the J57-P-23 engine uses an SE-0947-801 gage. They are quite similar and perform exactly the same function. The only difference between the two tools is that their methods of attachment to the engine differs. The gage should be attached to the position indicator switch mounting bracket as shown in view B of figure 4-13. As you will note, the gage has a movable pointer, a scale, and two rig pin holes. The rig pin holes (with rig pins installed) are used to keep the correct preload on both the cam and switch plate return springs. The dial indicator tells you whether you have changed your rigging while tightening the cables. If the rigging has been changed, the pointer will be pulled away from its index (zero) point. The switches are adjusted

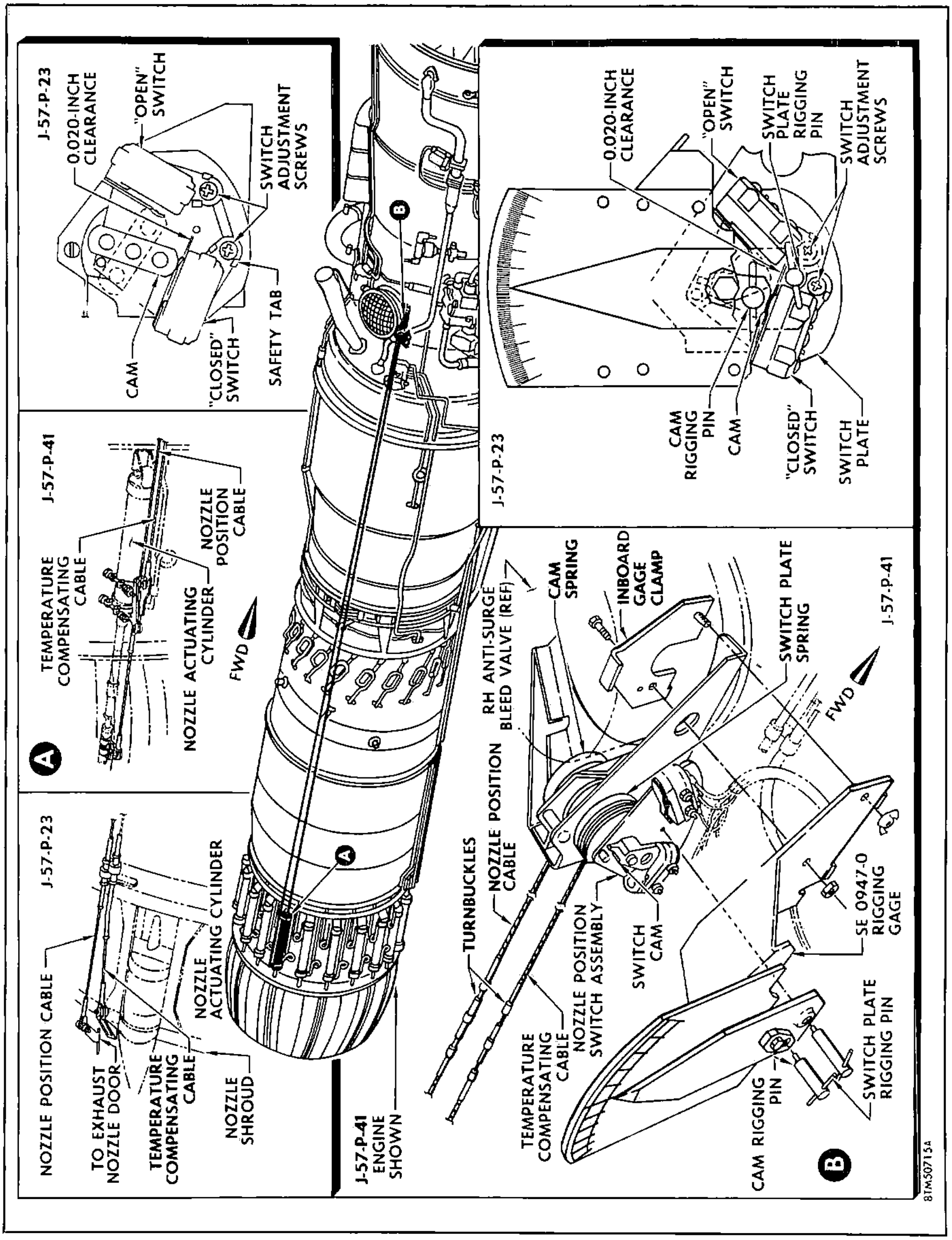


Figure 4-13. Rigging the Afterburner Exhaust Position Indicator System

by moving them physically—that is, the entire switch is moved either closer to or farther from the operating cam.

These instructions are for familiarization purposes only and are not intended as instructions for rigging the system. For specific rigging instruction, refer to your Power Plant Maintenance Handbook, T.O. 1F-102A-2-4.

MAINTENANCE OF THE AFTERBURNER POSITION INDICATOR SYSTEM.

Usually, any trouble encountered in the afterburner position indicator system can be immediately corrected by the rigging process just mentioned. However, if the rigging of the system is satisfactory, the indicator itself should be checked against a known satisfactory unit. In case the trouble still exists, a continuity check of the wiring should isolate the trouble.

EXHAUST TEMPERATURE INDICATOR.

The exhaust temperature of a jet engine, like the cylinder head temperature of a reciprocating engine, is a good indication of the over-all operating temperature of the engine. By way of further comparison, the jet engine exhaust temperature indicator is very similar to the instrument used to measure the cylinder head temperatures of piston engines. Both are thermocouple thermometers which measure the difference between electrical potentials of two metals in contact with each other. Figure 4-14 shows the exhaust temperature indicator and one of the four thermocouples used in the F-102A exhaust temperature indication system.

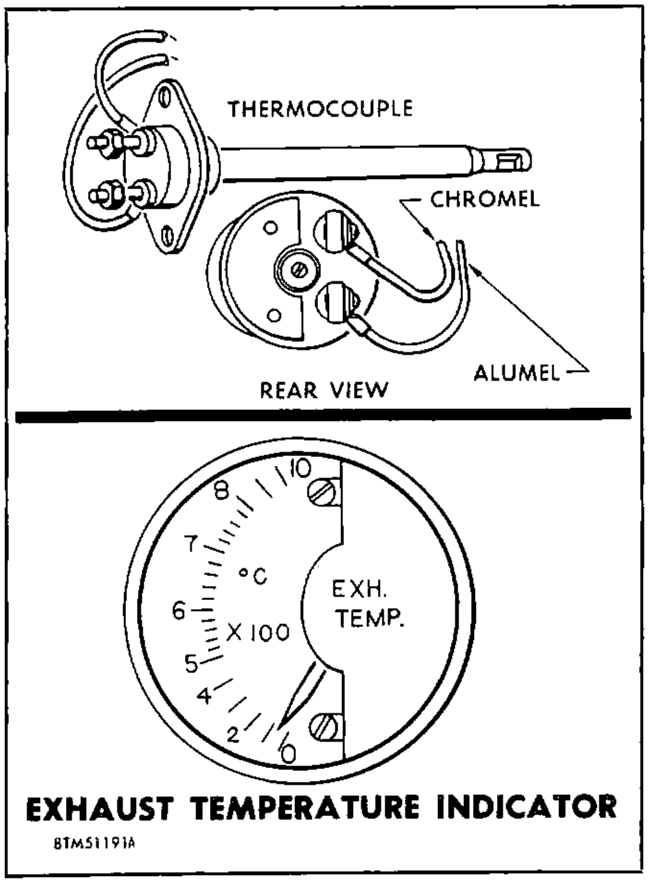


Figure 4-14. Exhaust Temperature Indicator and Thermocouple

THERMOCOUPLES.

A thermocouple is a combination of wires, each wire made of a different metal with each having a different electrical potential. If two such wires are connected together at one end and that junction point is heated above the ambient temperature of the opposite end of each wire, the joined wires become a source of electricity, the potential of which varies with the temperature. This physical phenomenon, known as thermo-electric effect, is the principle behind every thermocouple.

The thermocouples used in the exhaust temperature indicating system are made of chromel-alumel material. They are mounted in probes which project into the tailpipe section of the engine just aft of the turbines, and are approximately 90° apart. The thermocouples are connected in parallel with each other by leads made of chromel and alumel material.

THE EXHAUST TEMPERATURE INDICATOR.

The temperature of the thermocouples is shown on the dial of the indicator in degrees centigrade—the scale is calibrated from 0° to 1000°. As you can see in figure 4-14, there are no external controls or adjustments on the indicator. Internally, the indicator consists primarily of a moving coil mounted on pivots within a curved permanent magnet. Rotation of the coil is limited by two springs, one on each end of the coil. A pointer moves with the coil to show the temperature indication. The entire indicator is sealed and filled with helium.

Now take a look at the schematic illustration (figure 4-15). Note that the leads from the thermocouples connect to the coil through the springs, making a complete circuit. As you know, when current flows through a conductor, such as this coil, a magnetic field is set up. This magnetic field around the coil has both strength and direction, just as the field around a permanent magnet.

Note also that the coil is situated directly between the ends of the curved permanent magnet. The magnetic flux around such a magnet is concentrated between the two ends or poles. Thus, the coil tends to take a definite position between the ends of the permanent magnet. As you already know, like poles repel and unlike poles attract, so the coil tends to turn until each of its poles is close to the opposite pole of the permanent magnet.

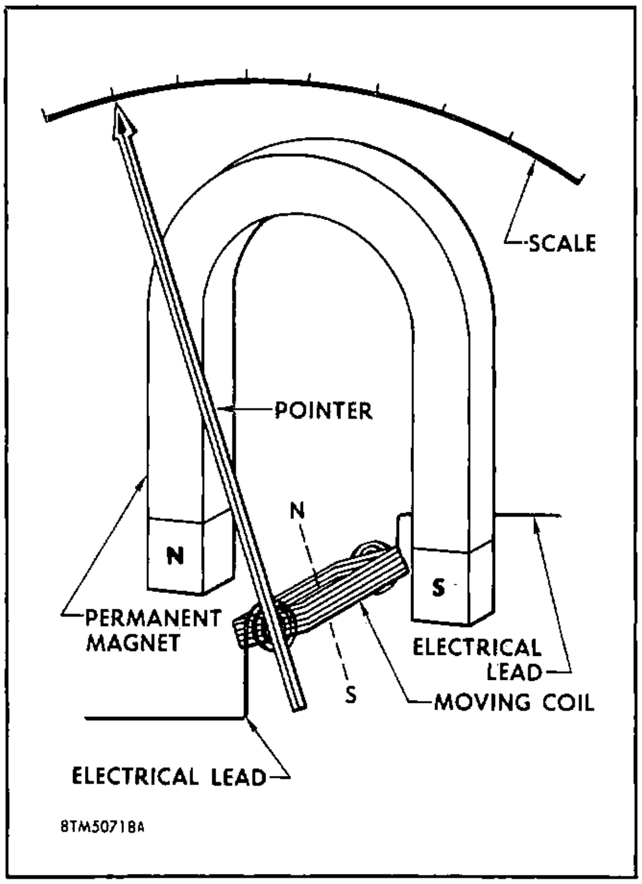


Figure 4-15. Exhaust Temperature Indicator Operational Schematic

If there were no restraining springs, the coil would always line up perfectly (with the poles in the normal relative positions) whenever there was a temperature difference between the "hot" end (thermocouple) and the "cold" end (indicator) of the system. Obviously, such an indicator would be of no value since it would always read the same. By the use of springs of the correct tension, the coil is permitted to move only an amount proportional to the strength of the magnetic field around it, which, as mentioned earlier, varies with the temperature differential.

Temperature Compensation.

The coil springs just discussed also serve another purpose; they are temperature compensators. The amount of rotation of the coil depends on the strength of its magnetic flux, which in turn is proportional to the difference in temperature between the thermocouple and the indicator. But that isn't exactly what we want to know. If the exhaust temperature is 600°C, we want the indicator to say 600°C, regardless of the temperature in the cockpit.

You can see then that the indicator must be set to read the cockpit temperature first so that the additional rotation of the coil will cause it to read actual exhaust temperature. The springs accomplish this for us because they are made from laminations of different metals which react differently to temperature changes.

Figure 4-16 shows how these bimetallic springs work. Note that the strip of brass in the laminated metal expands more than the strip of iron when heat is applied, causing the laminated strips to bend. In the same manner, the springs in the exhaust temperature indicator tighten up or straighten out with changes in cockpit temperature. In this way the coil, and therefore the pointer, are rotated to indicate the cockpit temperature.

The additional rotation, caused by the difference in temperature between the indicator and thermocouple, brings the pointer to a position proportional to the total exhaust temperature. Thus, the indicator pointer reflects the total of the temperature at the indicator, plus the difference between the temperatures at the indicator and the thermocouple. If you disconnect the thermocouple leads the indicator will show the approximate cockpit temperature.

Another temperature compensation problem results from the variations in electrical resistance of most metals with temperature changes. At a given temperature difference, the voltage generated in the circuit will produce a current inversely proportional to the resistance (Ohm's law). For any particular temperature difference between the "hot" and "cold" ends of the system, the current must always be the same. Therefore, a "neutralizer" is included in the indicator. This neutralizer is a resistor in which the resistance becomes less as the temperature increases. In that way

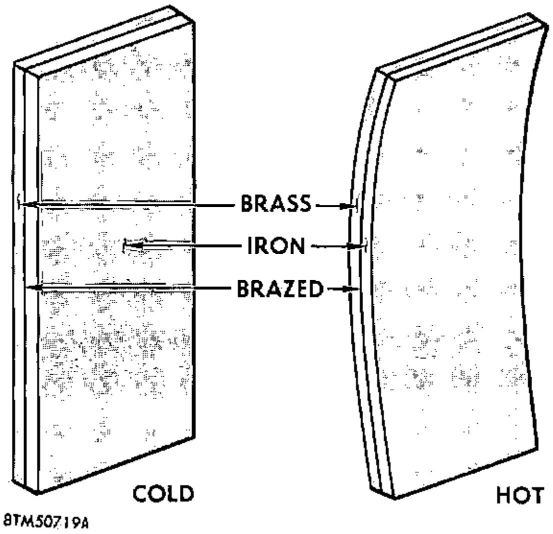


Figure 4-16. Effect of Temperature Changes on Bimetallic Strip

it keeps the total resistance of the system constant for any temperature difference.

SYSTEM CALIBRATION PROVISIONS.

We have discussed the importance of keeping the resistance of the system completely constant so the indicator will receive the right amount of current and give a correct indication. There are several things that might alter this resistance and consequently affect the accuracy of the indication. For example, if you replace any of the components of the system, as would happen when you change engines, there could be a slight change in the resistance. Even changing a terminal on the leads could alter the resistance. Therefore, a calibrating resistor is included in the system and is located on the right side of the cockpit above the master warning control box. You can use an ordinary Wheatstone bridge-type of tester to determine the correct adjustment of the resistor.

A special testing unit (SE-0783) is required to test the accuracy of the complete system. This test unit operates on a 115-volt, 60-cycle power source. You will note in figure 4-17, that the test unit has a lead incorporating a probe heater with an extension handle. The calibration and functional check of the system is made by heating each probe individually and checking the temperature indication in the cockpit against the reading of the test unit. When each probe has been checked individually, probes should all have heaters installed and the complete system checked against the test unit. If this indication is not within $\pm 10°C$ of the indication on the test unit, the system must be calibrated. This is accomplished by adjustment of the calibrating resistor in the cockpit.

FUEL FLOW INDICATING SYSTEM.

Any large jet engine uses tremendous quantities of fuel. The rate at which the engine uses fuel varies greatly, depending upon the power lever setting. Since airplane space and weight restrictions limit the amount of fuel that can be carried, the jet pilot is always vitally concerned with how fast the fuel supply is being consumed. Without this information, he cannot estimate accurately how long he can stay away from his base.

The rate of fuel consumption is also an indication of the efficiency of the engine. For these reasons, the F-102A uses a fuel flow indicating system. The two main components in the indicating system are shown in figure 4-18. The transmitter is mounted on the engine, while the indicator is situated on the engine instrument portion of the instrument panel.

THE FUEL FLOW TRANSMITTER.

The transmitter for the fuel flow indicating system is attached to the right side of the engine, adjacent to the oil pump and accessory drive housing. It is located in the main fuel line between the fuel control unit and the fuel oil cooler. An a-c synchro-transmitter is contained within the transmitter unit to send signals to the fuel flow indicator in the cockpit.

From the discussion of the pressure ratio indicator, you should be familiar with how the synchro-transmitter position of this component operates, so just the mechanical portion will be discussed here. The operational schematic in figure 4-19 will help you to see how the flow transmitter operates. Fuel flowing through the transmitter enters the port on the right and leaves through the port on the left, as shown by the arrows. Notice that a hub, with a vane projecting above it, is mounted in the flow area. A spring within the hub resists the force of the fuel flow, and tends to keep the vane in the upstream position at all times. The actual position taken by the hub and vane assembly depends on the rate of flow of the fuel which surrounds it. Therefore, we can call the vane and hub assembly the gage unit of the fuel flow transmitter. Now, let's follow the train of action to see how the position of the vane and hub is carried to the synchro.

How the Transmitter Produces a Signal.

Notice in figure 4-19 that there is no direct mechanical connection between the gage unit, which does the measuring, and the transmitting unit, which signals the indicator in the cockpit. The shaft of the hub is not geared to the sector shaft. Instead, a permanent magnet on the hub shaft surrounds a permanent bar magnet which drives a pinion gear. The relative positions of these magnets will tend to stay the same; when the gage unit rotates the ring magnet, the bar magnet turns with it so that the opposite poles of the two magnets are always lined up. You can see then how the reaction of the bar magnet and pinion to the movement of the ring magnet will move the sector shaft and drive the synchro. This method of driving the synchro permits the electrical part of the fuel flow transmitter to be isolated from the fuel-carrying part.

THE FUEL FLOW INDICATOR.

The fuel flow indicator is calibrated to show the rate of flow from 0 to 12,000 pounds per hour. As you have seen in the illustration of the flowmeter (figure 4-18), the dial is graduated every 100 pounds up to 3000 pounds, and in 1000 pound increments from 3000 to 12,000 pounds.

Since the fuel flow transmitter positions a transmitting synchro, it should be obvious that the indicator is a synchro instrument containing a synchro-receiver. Power to operate this synchro system comes from the airplane's 26-volt, 400-cycle, a-c electrical source. You have already learned the fundamentals of the synchro-receiver, so further details on this receiver will not be discussed.

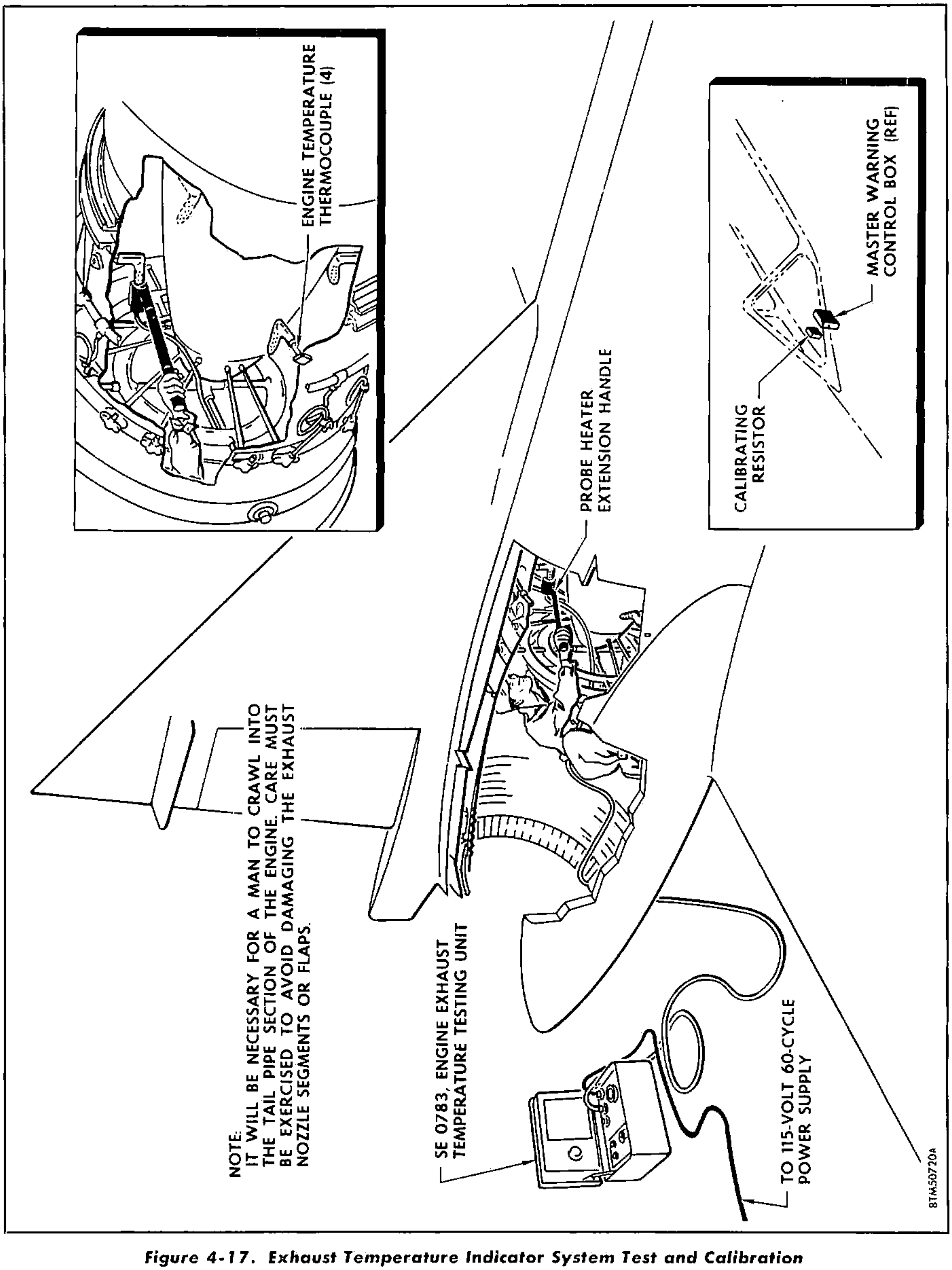
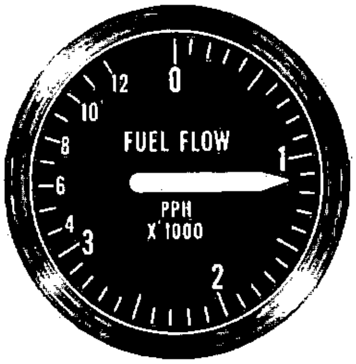
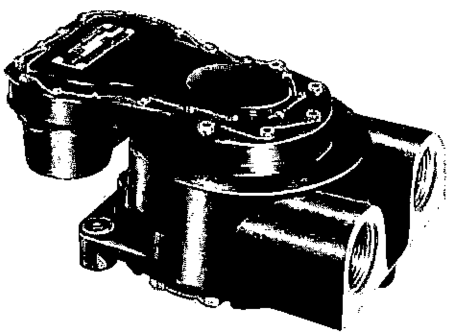


Figure 4-17. Exhaust Temperature Indicator System Test and Calibration



FUEL FLOW INDICATOR



FUEL FLOW TRANSMITTER

8TM50721A

Figure 4-18. Fuel Flow Indicator and Transmitter

Maintenance of the System.

If you should have any reason to doubt the accuracy of the indicator, disconnect the transmitter and plug the leads into a master synchro-transmitter. If the indicator pointer smoothly follows the movement of the test transmitter, the trouble lies in the airplane's fuel flow transmitter. There are no external provisions for adjusting either the indicator or the transmitter, so you must replace the faulty unit.

THE POWER PLANT WARNING SYSTEMS.

There are five individual warning systems for the J57 jet engine. These include the structural overheat, fire detector and overheat, fuel pump, oil pressure, and engine anti-ice warning systems. If you consider the meaning of the name "warning system," you will realize that these systems merely notify the pilot of the dangerous conditions in the power plant. The pilot himself must take the corrective action to eliminate the condition or to protect himself and the airplane from any possible danger.

The warning lights for all of the warning systems in the F-102A are centrally located in one master warning light panel. The panel, consisting of 16 different lights, is located on the right side of the pilot's instrument panel. The warning lights are numbered from 1 through 16. In addition to its number, each light also has the system or function name stencilled on the lens. These names are not visible, however, unless the bulbs in back of the warning light lenses are illuminated.

In addition to the 16 individual warning lights, the F-102A also has an amber-colored master warning light. This master light is located on the upper right side of the main instrument panel. When any condition in the airplane causes one of the 16 warning lights to light up, the master warning light also comes on. Since the master light is located directly in front of the pilot, it attracts his attention more quickly than the individual lights. It notifies him to check the individual lights on the warning light panel.

After the pilot has taken the necessary corrective action, he can press the RESET switch on the right control console. This switch extinguishes the master warning light. If another of the 16 warning lights should come on, the master warning light will light up again. The pilot then determines which warning light is on, takes the necessary corrective measures, and then presses the RESET switch again.

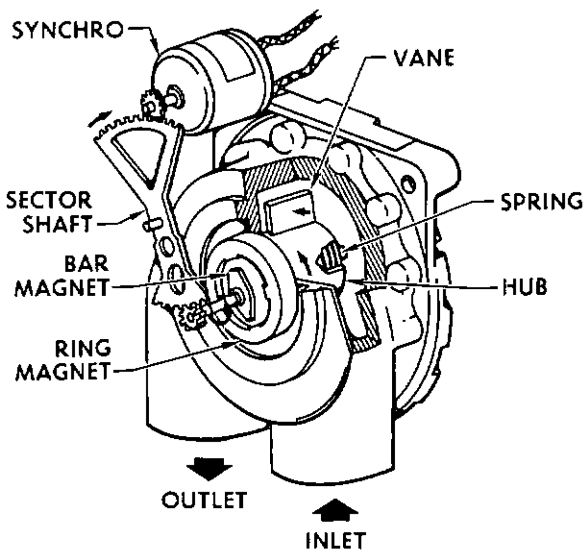


Figure 4-19. Fuel Flow Transmitter Operational Schematic

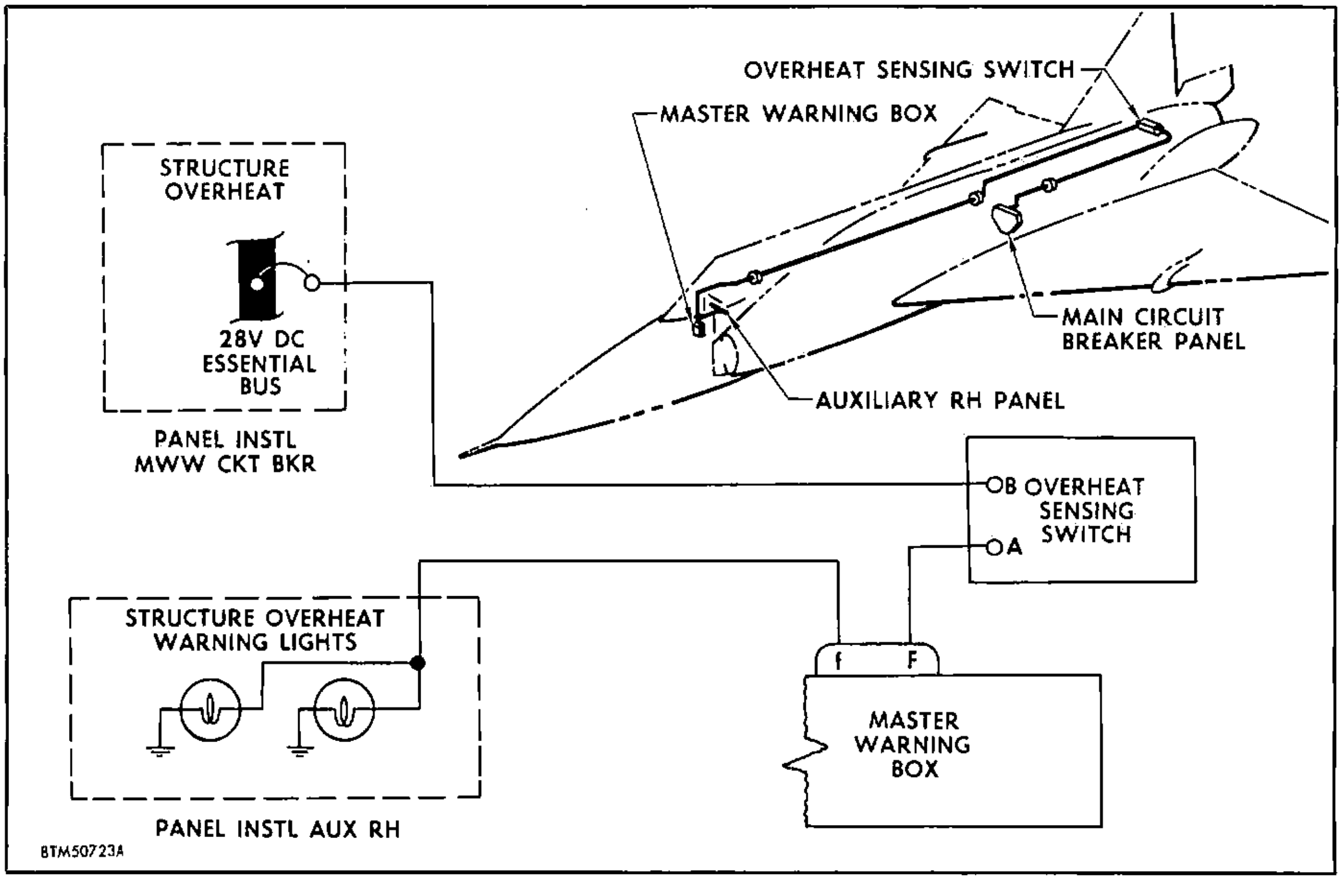


Figure 4-20. Structural Overheat Warning System Schematic

The power plant warning systems use the warning lights numbered 1, 4, 5, 7, and 8. Each of the power plant warning systems will be discussed individually in the following sections of this chapter.

THE STRUCTURAL OVERHEAT WARNING SYSTEM.

The structural overheat warning system uses the first (top) warning light on the warning light panel. This warning system notifies the pilot of excessive heat conditions in the fuselage structure at station 614.00 or the area just ahead of the tail cone fairings.

The detecting device for the structural overheat system is a mercury-operated temperature sensing switch which is attached directly to the fuselage structure. Only that portion of the switch which touches the fuselage structure is sensitive to temperature changes. This limits the switch to sensing structural temperatures only. The switch is preset to actuate whenever the structural temperatures rise above 245°F.

When the switch is actuated, both the master warning and structural overheat warning lights come on. These lights notify the pilot that a power reduction should be made and a reduced-speed flight condition established until the temperature decreases. If the system warning lights come on during ground run-up, you should shut the engine down immediately and investigate the condition.

Figure 4-20 shows the electrical schematic of the structural overheat warning system. Note that the current is taken from the 28-volt d-c essential bus. The five-amp, push-pull type circuit breaker, which protects the circuit from current overloads, is located on the main wheel well circuit breaker panel. Following the current from the essential bus and the circuit breaker, note that it travels to the switch. When this switch is actuated by an overheat condition, the current will flow to the master warning box where it causes the master warning light to illuminate, and then on the individual system warning light. The structural overheat warning system as described above is used on the very early models of the F-102A.

THE FIRE DETECTOR AND OVERHEAT WARNING SYSTEM.

The fire detector and overheat warning system, as its name implies, does two things—it notifies the pilot of an engine overheat condition and notifies him of a fire condition. Although each section of the warning system does just one job, both sections use the same warning light in the cockpit. By routing the electrical signal from the overheat circuit through a flasher, the warning light is made to flash on and off whenever an engine overheat condition exists. In the event of fire, the warning light will come on and burn steadily.

You should not confuse the engine overheat warning system with the structural overheat warning system. As just mentioned, the structural overheat warning system is used on just the very early models. The fire detector and overheat warning system, described in the following paragraphs *is* used on *all* models.

Both the overheat and the fire warning sections of the warning system use detector loops. These detector loops are the temperature sensing elements of the system. The loops are actually a type of coaxial cable with electrical connecting fittings on each end. The cables are constructed of an inner electrical conductor within an outer sheath of corrosion-resistant alloy.

The space in the cable between the inner conductor and the outer covering contains a thermistor-type heat-sensitive compound. The electrical resistance of this heat-sensitive compound varies inversely to the temperature. Under normal operating conditions the compound acts as a good insulator; but when a "hot" spot develops anywhere along the detector cable, the resistance of the compound drops and allows current to flow from the inner conductor to the outer portion of the coaxial cable. This completes the detector circuit to ground and the warning light comes on.

How the Fire Detector and Overheat Warning System Operates.

Figure 4-21 shows where the components of the warning system are located in the airplane and how the system operates. Note in the upper portion of the illustration that the overheat detector loop and the fire detector loop are situated in different sections of the engine compartment. The detector relays, the overheat flasher, and the detector control boxes are located in the upper electronics compartment.

To get a good idea of how the warning system operates, let us trace the current flow in the schematic in the lower portion of figure 4-21. The current for the warning system, like all of the other warning systems, is taken from the 28-volt d-c essential bus. After it passes through the push-pull 5-amp circuit breaker, the current flows to the two detector control boxes and to the junction point of the two relays. Note that the warning lights must receive their current from either one of the detector control boxes before the lights will illuminate. However, the control boxes cannot send current to the lights until the warning light circuit is grounded at some point.

In the preceding section, you learned that the detector loops ground out whenver they are subjected to a "hot" condition that lowers the internal resistance of the detector loops. Whenever one of the cables grounds out, its detector control box circuit will be completed, and current will flow to the lights. If the overheat loop is subjected to the "hot" spot, the current from the overheat detector control box has to flow to the overheat detector flasher before it passes on to the warning lights. This causes the warning light to flash on and off during the overheat condition. Since the fire detector circuit does not have a flasher unit, a fire condition causes the warning light to burn steadily.

The test switch, shown in the right hand portion of the electrical schematic, will allow you to ground check the warning system without heating any part of the coaxial cable detector loops. This switch is located to the left and above the tachometer. A small test light is mounted on the right side of the test switch. The test switch does the same job that the detector loops do during an overheat or fire condition —it grounds the circuit and allows the respective control box to send current to the warning lights. By positioning the switch to either the FIRE or OVHT position, the pilot can determine whether the warning systems are functioning properly.

Maintenance of the Fire Warning and Overheat Warning System.

Malfunctions developing in this warning system may stem from the control boxes or the detector loops. Both the boxes and the loops are delicate units and can be damaged during installation by rough handling or improper installation. Because of the nature of this warning system, the only indications of any malfunction will be the warning light coming on when neither an overheat or fire condition exists, or the lights not coming on when the undesirable conditions do exist.

From the discussion of how the warning system operates, you should know that a premature warning light indication is caused by some part of the detecting circuits shorting out to ground. The easiest way to determine whether the control boxes or the detector loops are defective is to replace the control box with a unit that is known to be good. If the substitute control box does not cure the difficulty, you will have to check all of the detector loops for possible breaks or loose electrical connectors. No rework is allowed on any of the loop segments, so you will have to replace any defective detector loop with a serviceable item.

THE ENGINE FUEL LOW-PRESSURE WARNING SYSTEM.

The engine fuel low-pressure warning system notifies the pilot when sufficient fuel pressure is not being supplied by the main engine stage of the fuel pump.

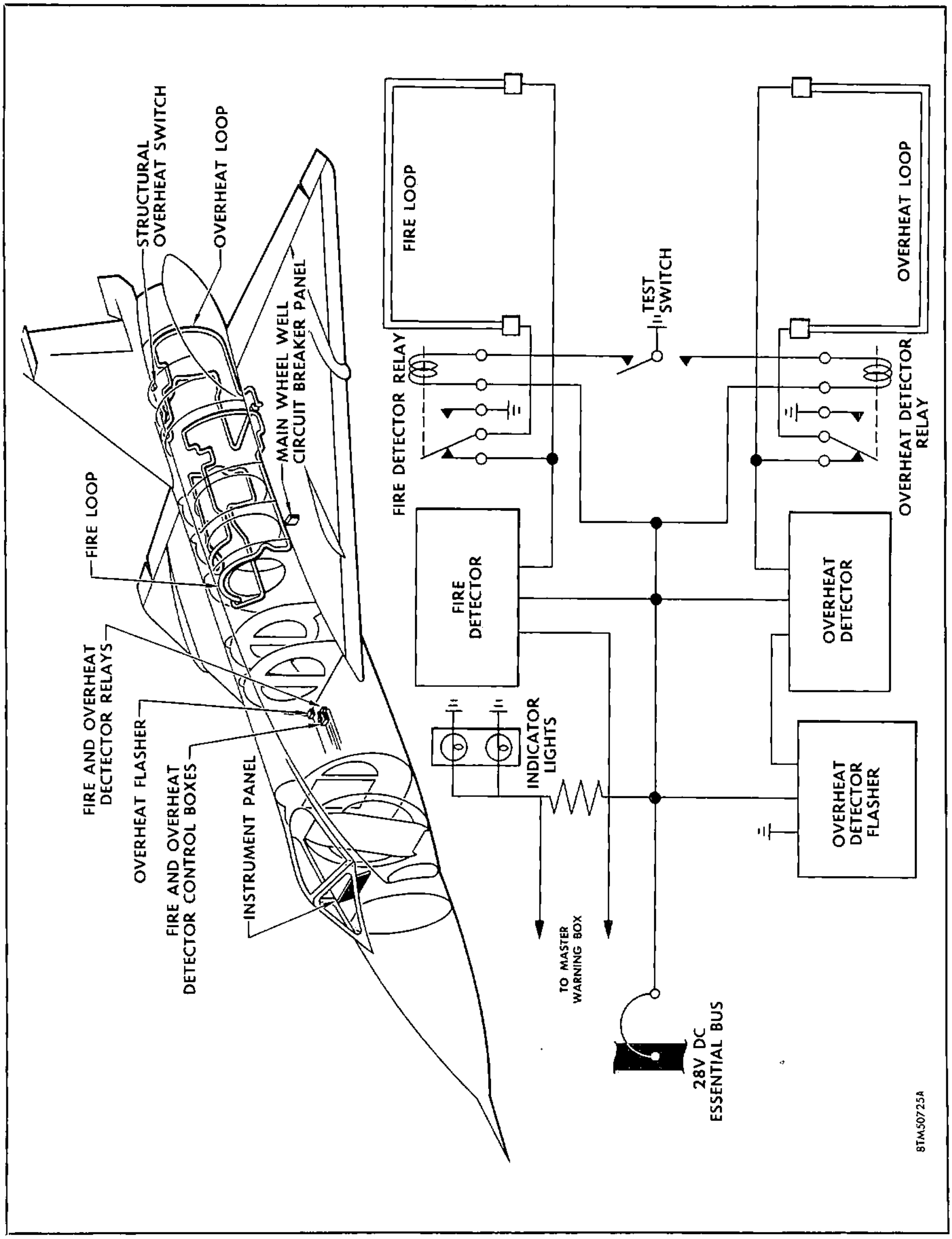


Figure 4-21. Fire Detector and Overheat Warning System Schematic

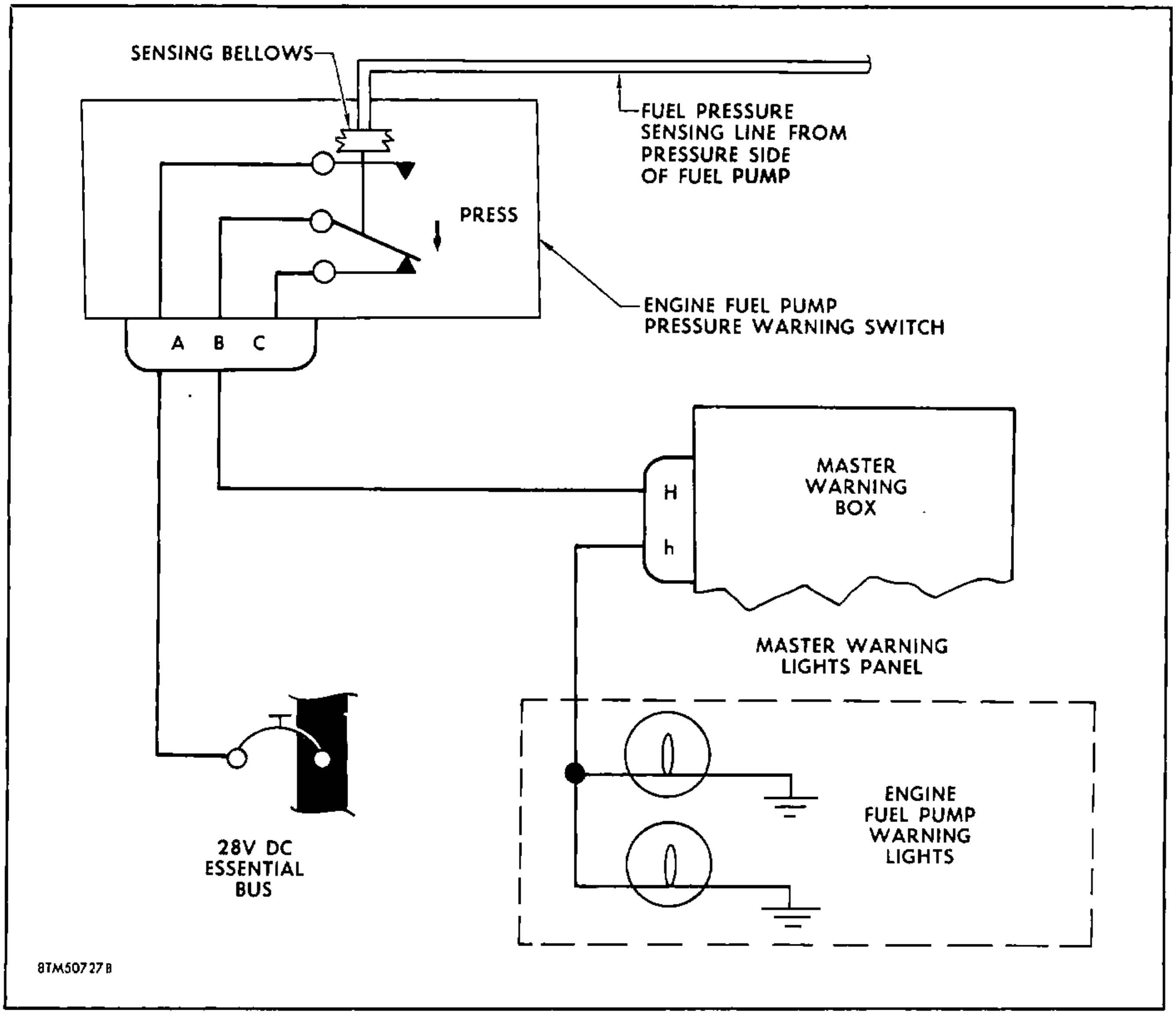


Figure 4-22. Engine Fuel Pump Low-Pressure Warning System

The system consists of a warning light on the warning indication panel, and a pressure switch on the lower left side of the oil pump and accessory drive housing of the engine. A pressure sensing line is installed between the switch and the pressure side of the fuel pump engine stage. Note in figure 4-22, that when fuel pressure drops in the fuel pump outlet line, the pressure warning switch closes and allows current from the 28-volt d-c essential bus to illuminate the warning light in the cockpit. As you can see in the illustration, the pressure warning switch is operated by the sensing bellows. Fuel pressure is routed to the outside of the bellows and forces the switch to open. The switch is shown in the open position in the schematic (figure 4-22). When the fuel pressure is low or when the engine is shut down, the sensing bellows expand and close the switch. This allows current to flow from terminal B to the master warning box and then to the engine fuel pump warning lights.

The warning lights will always come on when the engine is first being started. After the engine has attained starting speed, however, the fuel pump will put out sufficient pressure to force the bellows to open the switch and the lights will go off. The sensing bellows of the pressure switch are set to open the switch before the fuel pressure reaches 125 psi and to close the switch when the pressure drops below 105 psi.

Maintenance of the Fuel Low-Pressure Warning System.

You will find that the maintenance requirements on the fuel low-pressure warning system will be about the same as the requirements on the other warning systems. You should keep in mind, however, that the warning lights will light temporarily when the engine is being started. After the engine has attained idling speed, the lights should go off if the warning system is functioning properly.

The malfunctions of the warning system will include premature lighting of the warning lights or the lights will not illuminate at all. As you have noted in figure 4-22, the warning lights are supplied current whenever the pressure switch closes. When you trouble shoot a malfunctioning fuel low-pressure warning system, always check the light bulbs first. If they are not burned out, you will probably find it best to replace the pressure switch with a serviceable unit. Although this is not actually trouble shooting the warning system, the ease with which the switch can be replaced makes this a simpler remedy than checking the entire circuit with a continuity light or voltmeter. If replacing the switch does not correct the trouble, however, you will have to check out the entire circuit for an open or broken lead, or a short to ground. If the warning system still malfunctions after completing the circuit trouble shooting, the trouble lies in the fuel pump, and this requires replacing the pump.

THE ENGINE OIL LOW-PRESSURE WARNING SYSTEM.

The engine oil low-pressure warning system notifies the pilot whenever the engine oil pressure drops below 36 psi. The warning lights will go off when the oil pressure raises back to 40 psi. The pressure switch, which detects the oil pressure, is mounted on the left side of the engine just aft of the oil tank.

You will note in figure 4-23 that the oil low-pressure warning system is almost identical to the fuel pump low-pressure warning system. As you might well imagine, this warning system operates in the same manner.

Whenever the oil pressure drops below the 36 psi level, current from the 28-volt d-c essential bus will flow through the 5-amp push-pull type circuit breaker, pass through the pressure switch, and then illuminate the warning lights in the cockpit. When the oil pressure increases to 40 psi, the reverse situation takes place. The pressure switch is forced open by the oil pressure, and this breaks the warning light circuit. When this occurs, the warning lights go out.

If the oil low-pressure warning system should malfunction, you will find it best to follow the same procedure given for trouble shooting the fuel pump low-pressure circuit. After checking the warning light bulbs and the circuit breaker, replace the pressure switch. The last thing to do is trouble shoot the entire circuit with a continuity light or voltmeter. If the malfunction still persists, however, you will have to remove and replace the engine oil pump with a serviceable unit.

THE ENGINE INTAKE ANTI-ICE WARNING SYSTEM.

The engine intake anti-ice warning system notifies the pilot whenever the engine anti-ice system malfunctions. Icing is a very serious matter on any part of the airplane, especially in the engine air intake area. Therefore, it is imperative that the engine intake anti-ice warning system operates properly.

The F-102A uses hot air taken from the engine compressor section to heat the inlet duct guide vanes and the accessory fairing area. The anti-icing air flow is controlled by electrically-actuated valves and regulators installed in each of the lines. The valves operate in conjunction with the anti-icing control system. When the valves are open, heated air flows into the engine inlet guide vane manifold. The heated air then passes inward through the double-walled accessory fairing and vents out the cap of the fairing.

How The Anti-Ice Control System Operates.

Before you can understand the operation of the engine intake anti-ice warning system, you must know how the anti-ice system operates when it is functioning normally. Therefore, we will analyze the circuits shown in figure 4-24 and follow the operation of the control system. This schematic shows the system as it would be during flight under "no-ice" conditions.

First, locate the ice detector assembly and the interpreter assembly. These two assemblies are the heart of the engine anti-ice control system. The detector assembly detects the presence of ice in the engine intake duct and then signals the interpreter assembly. The Interpreter assembly energizes the anti-ice control relay which in turn actuates the anti-ice system.

The anti-ice system is controlled electrically by a three-position switch in the cockpit. This is shown in the lower part of figure 4-24. Note the three positions —AUTO, MAN, and OFF. First, let's see how the system works when the anti-ice switch is in AUTO. Note the 10-amp anti-ice power circuit breaker. All power for the detector and interpreter comes from the 28-volt d-c essential bus through the 10-amp anti-ice power circuit breaker. The current flows from the circuit breaker, through the ignition power relay, and then to the connection A1 on the interpreter assembly. The ignition power relay is energized only when the engine is being started. Consequently, the connection at A1 is always connected directly to power (across 2 and 3 as shown in the schematic) except during engine starting.

Following current flow A1, note that it travels through the switch in the ice detector assembly. Since a "no-ice" condition exists, the current flows back to A6 in the interpreter assembly. Note that relay R1 is connected to A6 and also to ground. Therefore, relay R1 is energized when the detector switch is in the "no-ice"

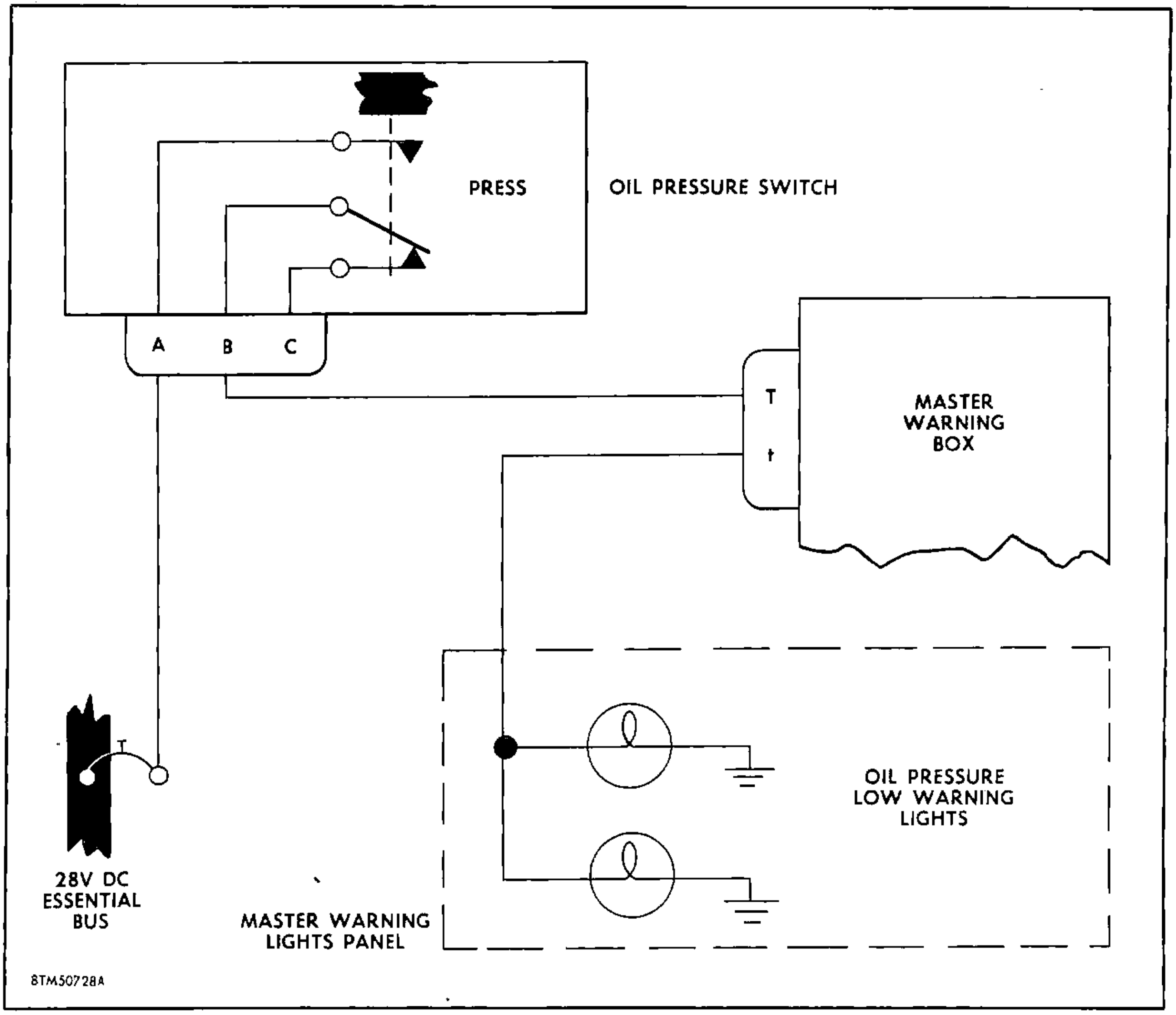


Figure 4-23. Engine Oil Low-Pressure Warning System Schematic.

position. Relays R3 and R4 are connected to B3 and contact C on the detector switch. These relays are deenergized in the "no-ice" condition as shown in the schematic. Since R2 is energized only when R4 is energized, R2 is also deenergized in the "no-ice" condition.

From the preceding discussion, you should have a general idea how the circuit functions when there is no ice in the detector tube.

Now, let's see what happens when the detector probe ices up. When the ice condition occurs, the detector switch closes. The operation of this switch is described later. Remember that A6 is connected to A1 through the switch in the R1 relay; therefore, B3 now has current since it is connected to A6 through C, B, and B2. Relays R3 and R4 now energize. This sequence results in the probe heater being connected to power through A1, the R3 relay switch and B4. The R2 relay and the heater at relay R1 are energized through R4. The R2 relay closes its switch to furnish a ground to the anti-ice control relay which then receives current from the 28-volt d-c bus through the 5-amp anti-ice control circuit breaker. The anti-ice control relay then closes to energize the anti-ice system.

The anti-ice system receives its power through the 5-amp anti-ice control circuit breaker. As the probe heater heats up, it melts the ice in the probe and the detector switch returns to the "no-ice" position. Relays R2, R3, and R4 deenergize, and the probe heater and the R1 relay heater are disconnected from power. Although the mechanical time delay at relay R2 is deenergized, it does not open at this time. This relay contains a clock mechanism that keeps the switch closed until 60 seconds after the relay has deenergized.

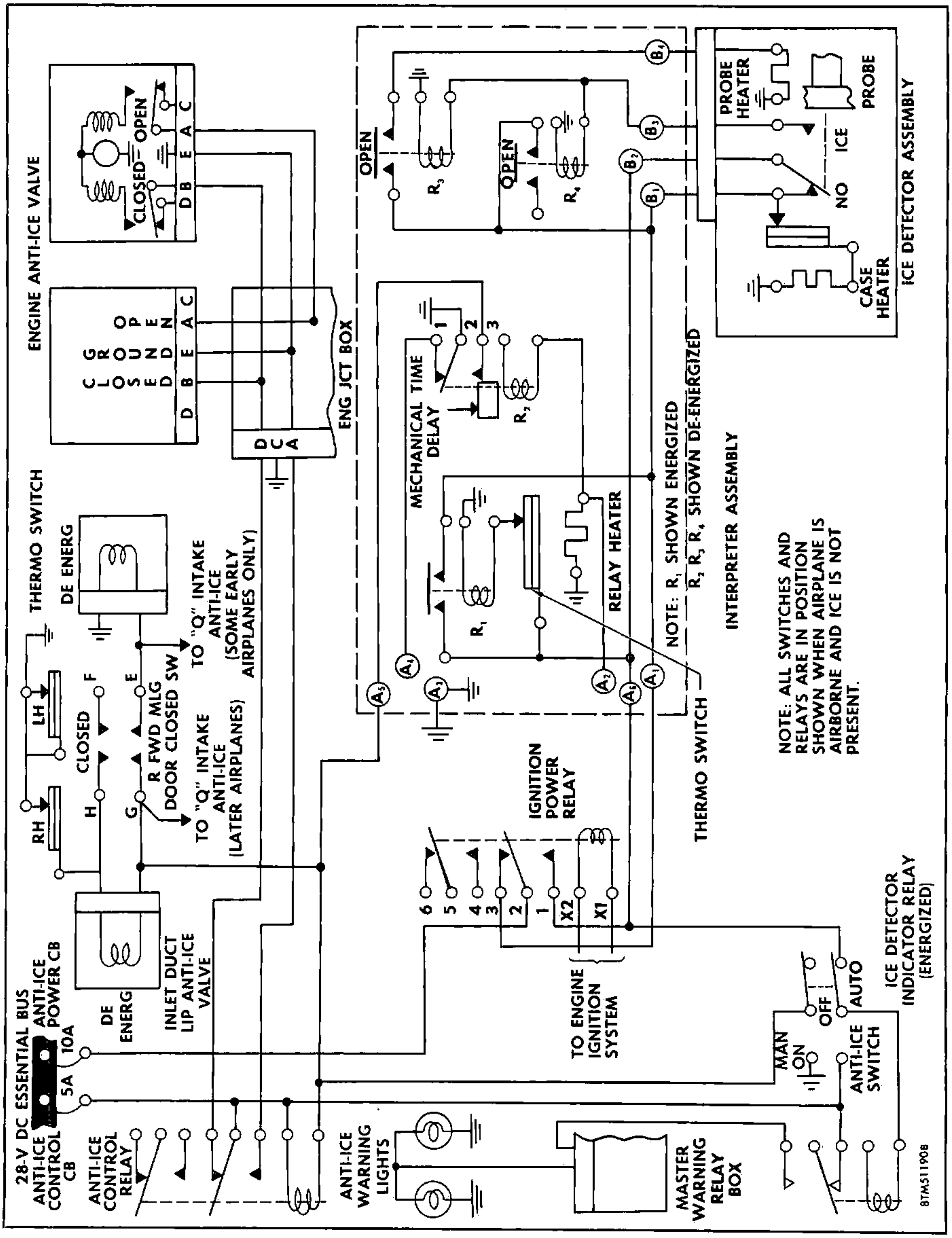


Figure 4-24. Engine Intake Anti-Icing Warning System Schematic

This delay feature insures that the anti-ice system will stay on long enough to do some good even though the detector switch returns to the "no-ice" position after just a few seconds.

How The Anti-Ice Warning System Operates.

Normally, the probe heater will melt the ice at the probe and return the probe switch to the "no-ice" position within 17 to 20 seconds. If it does not, it indicates that the detector is malfunctioning. To deenergize the relays during a malfunction and to keep the probe heater from overheating the probe, are the functions of the thermoswitch at relay R1. About 17 to 20 seconds after the detector switch moves to the ice position (across B and C) the R1 relay heater will cause the thermoswitch to heat up enough to break the circuit to R1 relay. The R1 relay switch will then open and A6 (also B) will no longer be connected to power. Since B3 is connected to B2 across the detector switch (at B and C) the R2, R3, and R4 relays will deenergize and the probe heater and the R1 relay heater will be disconnected. Sixty seconds later, the switch at the R2 relay will open the circuit to ground across 2 and 3; the anti-ice control relay will deenergize; and the anti-ice system will shut down.

Malfunction of the detector destroys the effectiveness of the automatic control system and the ani-ice system will not operate. If the pilot is informed of the malfunction, he can move the ANTI-ICE switch to MAN, thus grounding the anti-ice control relay. The anti-ice system will then operate until he moves the switch to OFF. Informing the pilot is the job of the anti-ice warning system. The warning lights on the right-hand auxiliary instrument panel illuminate and warn the pilot of the malfunction. These warning lights come on when the master warning box is energized. You can see in the schematic (figure 4-24) that the warning box is connected to contact 3 on the ice detector indicator relay. When the relay is energized, contact 3 will not be connected to power. Notice that the relay is connected to point A6 on the interpreter assembly through the anti-ice switch when it is at AUTO.

Now as we noted earlier, A1 is normally powered and A6 is normally connected to A1 (power); therefore, the detector indicator relay is normally energized except when the engine is being started and the ignition power relay is energized. The warning light will be on during a malfunction of the detector or interpreter assembly or when the anti-ice switch is at OFF. As you will learn below, the warning light will also be *on* when the switch is at AUTO *if* the air stream velocity in the engine intake duct is less than about 40 knots.

Ice Detector Assembly.

The ice detector assembly is mounted in the upper part of the engine air intake duct. The assembly consists of a probe and a main housing or case, with a pressure diaphragm inside the case. The chamber on one side of the diaphragm is connected to openings on the upstream side of the probe, while the chamber on the other side senses pressure from openings on the downstream side of the probe. When the airplane is moving, the upstream pressure will exceed downstream pressure. The diaphragm controls the ice detector switch. When the airstream is moving at less than 40 knots, the pressure differential across the diaphragm is not enough to move the switch to the "no-ice" position. If the ANTI-ICE switch is in AUTO at this time, the ANTI-ICE warning lights will illuminate. This does not necessarily indicate a malfunction. The warning lights should go out as soon as the airstream exceeds 40 knots.

When the airstream exceeds 40 knots, the pressure will force the switch to the "no-ice" position. However, if ice blocks the probe intakes, the pressure difference across the diaphragm will drop and the switch should move to the "no-ice" position. If the probe intakes should become clogged and will not open after 17 to 20 seconds, the interpreter will turn off the probe heater and the detector will be inoperative until the probe openings are no longer blocked. The heater in the detector case keeps the case warm to prevent condensation of moisture. The thermoswitch in the heater line interrupts the current to the heater when the temperature reaches a certain point. This heater will obviously cycle off and on continuously.

The detector assembly is a rather sensitive item, and it is better not to attempt maintenance on it without complete information. If the probe becomes clogged or otherwise becomes defective, replace the assembly with a new unit. If you must repair it, be sure to consult the technical order covering this component.

Maintenance Of The Engine Intake Anti-Ice Warning System.

From reading the preceding description of the engine anti-ice and anti-ice warning system, you should understand that a thorough comprehension of how the warning system operates cannot be had without first understanding the electrical wiring diagram of the anti-ice system itself. The maintenance requirements for the warning system will be somewhat more difficult than the requirements for the other engine warning systems. When the pilot reports that the anti-ice warning system is malfunctioning, however, troubleshoot the circuit with the same technique that you use on other circuits. Always start with the most logical trouble sources and end with the least likely. After assuring that the warning light bulbs, circuit breakers, and switch are functioning satisfactorily, check the electrical circuit with a voltmeter. Keep the electrical power ON. This type of trouble shooting will isolate the component that is malfunctioning. Any inoperative component must be replaced with a serviceable item.

SUMMARY.

This Training Supplement has presented you with a general review of jet engine principles and described the J57 turbojet engine as it is installed in the F-102A. After reviewing the development of jet engines in Chapter I, you acquired a general overall knowledge of the J57 engine and its associate systems. In Chapter II you learned how the fuel is metered to the engine and controlled in accordance with atmospheric and altitude conditions. Detailed descriptions and maintenance suggestions were also given for the oil, ignition and starting, in-flight and ground cooling, and the engine anti-ice systems. Chapter III described engine preservation and depreservation techniques and also covered the engine installation and removal operations. The last chapter explained how the engine instruments and warning systems function and outlined some of the common maintenance requirements.

Two points have been mentioned several times in the four chapters in this supplement, but they are important enough to be mentioned once more. These points concern the relation of F-102A Maintenance Technical Orders to this Training Supplement. In some cases, the description of the engine and its associate systems in this supplement contain specific values and pressures. These values and pressures are used for explanatory purposes only, and they should not be used on the flight line. The latest information of this nature can be found in the *Maintenance Technical Orders.*

In other instances, you have learned that all of the engines do not have the same type of components. Although specific engine model numbers have been given in some examples, this information has been omitted in others. This information, too, should always be obtained from the Technical Orders.

Keep in mind that this Power Plant Installation Training Supplement has not replaced the Technical Orders. It has only acquainted you with the J57 engine and outlined some of the maintenance needs. This knowledge has prepared you to use the Maintenance Technical Orders more intelligently so that you will be a more valuable member of the F-102A ground support team.

ALPHABETICAL INDEX

*The Symbol * Indicates An Illustration*

ALPHABETICAL INDEX (CONT)

*The Symbol * Indicates An Illustration*

ALPHABETICAL INDEX (CONT)

*The Symbol * Indicates An Illustration*

F-102A

MAINTENANCE TRAINING SUPPLEMENT

(EXPERIMENTAL)

To Be Used in Conjunction With

T.O. 1F-102A-2-3

T.O. 1F-102A-2-7

T.O. 1F-102A-2-10

FLIGHT CONTROL SYSTEM

CONVAIR
F-102A

MAINTENANCE TRAINING
SUPPLEMENT

FLIGHT CONTROL
SYSTEM

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE

AIR FORCE—AFPS—SAN ANTONIO, TEXAS—5-57-2600

1 JULY 1956

Foreword *Jafftb.*

The F-102A Training Supplements have been prepared by Convair, A Division of General Dynamics Corporation, to provide you, the system mechanic or maintenance technician, with basic information concerning the F-102A airplane and its operating systems. There are ten of these supplements, each covering an airplane operating system or major component. The text is direct and informal and is supplemented with illustrations and diagrams to aid you in understanding how the systems operate. The following is a list of the Training Supplements on the F-102A airplane:

Title

Flight Control System
Hydraulic System
High-Pressure Pneumatic System
Low-Pressure Pneumatic System
Airplane General
Airframe Fuel System
Power Plant Installation
Airframe Armament System
Electrical System
Instruments

Each supplement describes an operating system or major component, how and why it operates, and maintenance problems that you may encounter. The entire major system is further simplified by describing each of its subsystems separately. Thus, when you understand how all of the subsystems operate, you will be familiar with the functions of the major system. This knowledge of the airplane systems will enable you to use other operational, service, and maintenance instructions.

Since the purpose of these publications is to familiarize and acquaint you with the airplane systems and components, specific values, measurements, and tolerances are not given. Any values that appear in these supplements are approximate only and are used to emphasize more clearly a certain operation or condition. You must refer to your 1F-102A-2-3, -2-7 and -2-10 Technical Orders and other pertinent handbooks for specific values when adjusting or checking a system and its components. These Technical Orders are revised periodically to include the latest maintenance procedures and data on the equipment installed in the F-102A.

The F-102A Maintenance Technical Orders consist of the following handbooks:

T.O. 1F-102A-2-1	**General Airplane**
T.O. 1F-102A-2-2	**Ground Handling, Servicing, and Airframe Group Maintenance**
T.O. 1F-102A-2-3	**Hydraulic and Pneumatic Power Systems**
T.O. 1F-102A-2-4	**Power Plant**
T.O. 1F-102A-2-5	**Fuel Supply System**
T.O. 1F-102A-2-6	**Air Conditioning, Pressurization, and Anti-Icing Systems**
T.O. 1F-102A-2-7	**Flight Control Systems**
T.O. 1F-102A-2-8	**Landing Gear**
T.O. 1F-102A-2-9	**Instruments**
T.O. 1F-102A-2-10	**Electrical Systems**
T.O. 1F-102A-2-11	**Radio-Communication and Navigation Systems**
T.O. 1F-102A-2-12	**Armament and Armament Electronics**
T.O. 1F-102A-2-13	**Wiring Data (F-102A)**
T.O. 1F-102A-2-13A	**Wiring Data (TF-102A)**

The Air Force is vitally interested in seeing that its equipment functions properly and is maintained as efficiently as possible. The Air Force, like a manufacturer of an automobile or commercial appliance, provides printed instructions for use with its equipment. The printed instructions you will use most frequently in maintaining the F-102A are the dash-2 Technical Orders listed above. They are available for your reference at the Maintenance Office on your base. You must refer to them frequently so that you will always have the latest maintenance information. The Training Supplements will help you to use these Technical Orders by answering many of the "hows" and "whys" that may puzzle you from time to time.

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Introduction

This supplement consists of five chapters that describe the F-102A Flight Controls. Chapter I acquaints you with the theory of flight and the development of the F-102A flight controls. Chapter II gives you a detailed description of the elevon control surfaces and their associated hydraulic and mechanical actuating devices. Chapter III provides similar information on the rudder. Chapter IV explains the operation of the pitch and yaw damper systems. Chapter V concludes this supplement with comments on the evolution of the automatic flight control system.

Chapter I

THEORY OF FLIGHT CONTROLS

Contents *Page*

Since the historic first powered flight by the Wright brothers in 1903, flight controls have become one of the most important fields in the science of aerodynamics. In designing larger and faster airplanes, the method for "steering" them has had to be revised extensively to meet the demands imposed by aerodynamic forces and to compensate for the limitations of human pilots.

The necessity of revising the flight controls for airplanes is similar to the necessity of revising the steering mechanisms in automobiles. Early automobiles, with their small engines, low speeds, and light weight, were easily steered by the driver who turned a steering wheel that was mechanically connected by direct linkage to the front wheels. However, with the advent of high-powered engines, heavier bodies, and greater speeds, the forces directed on the steer-inch mechanism were too great for a driver to control adequately by human strength alone. Compensation for the external forces on the steering mechanisms was achieved with the use of *power steering.* This method allowed the driver to easily turn the wheel while a hydraulic or electrical system performed the actual work involved in turning the front wheels against the road forces. Power steering in automobiles brought automobile control within the physical limits of all humans regardless of their physical stature or strength.

Power steering is also used in modern high-speed aircraft. In addition to the human characteristics which are present in automobile steering, such as muscular strength, reaction time, and proneness to fatigue, the characteristics affecting the control of airplanes, such as sensitivity to changes of position and changes in forces on control surfaces, must also be considered. Although power steering was first used in airplanes to eliminiate the fatigue factor, the development of transonic and supersonic airplanes presented control problems which could be solved only with a fully-powered steering (flight control) system.

In this training supplement, you will learn about the flight control system that is used on the F-102A airplane. As you will discover in the first chapter, the control surfaces on the F-102A are somewhat different from those used on conventional-winged aircraft and the flight control system has been designed to compensate for these differences.

To fully appreciate the need for a complex flight control system such as the one used on the F-102A, you should be fully aware of the principles of aerodynamics and the forces, or loads, that the system must control as it performs its job. The following discussion in this chapter will deal with a review of fundamental aerodynamics and some of the problems of flight controls that have developed in the design of supersonic airplanes. There is no doubt that this review will be "old stuff" to you if you are a veteran Air Force maintenance man. If you are new to the aircraft maintenance field, however, this review can be just the right kind of tool to help you understand the importance of your maintenance and inspection duties on the F-102A flight controls.

PRINCIPLES OF FLIGHT.

You may have seen early illustrations that pictured man's attempts to fly, or you may have read about his struggles to gain mastery of the air. No doubt, man got the idea of flying from watching birds as they gracefully soared overhead. His early attempts at flight were, for the most part, fantastic—if not fatal. There were man-made bird wings strapped to arms and

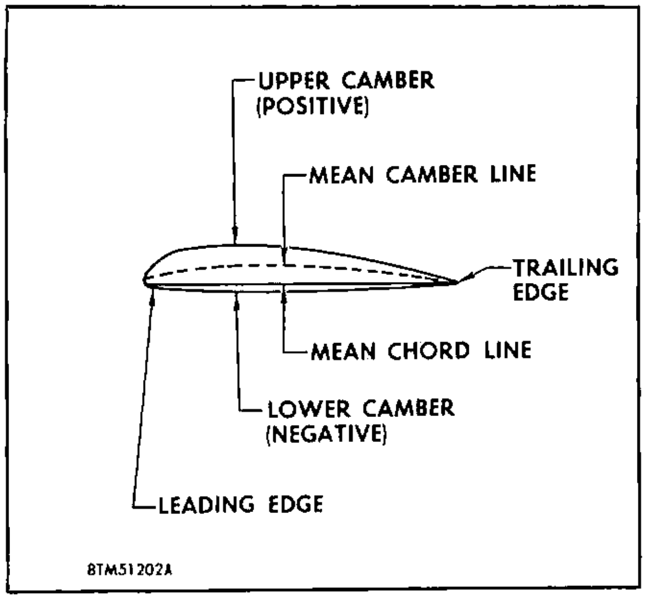

Figure 1-1. Typical Airfoil Section

shoulders, hot-air balloons, and gas balloons. Although man did achieve flight in the balloons, all of the flight attempts in heavier-than-air devices failed. Their failure lay in the fact that man did not fully comprehend the principles of aerodynamics.

AIRFOILS.

Let's take a look at a cross-sectional view of a typical airfoil and learn some of the airfoil characteristics and nomenclature. In figure 1-1, you will note that the two ends of the airfoil differ in appearance. The end that faces into the wind in flight is called the leading edge and is rounded; while the other end, the trailing edge, is narrow and pointed. A reference line which is often used in discussing an airfoil is the chord—a straight line drawn through the airfoil from the farthermost points of the leading and trailing edges. The distance from this chord line to the upper and lower surfaces denotes the amount of upper and lower camber (curvature). Another reference line, drawn from the leading edge to the trailing edge, is the mean camber line. This mean camber line is equidistant at all points from the upper and lower airfoil surfaces.

FORCES IN FLIGHT—LIFT, THRUST, WEIGHT, AND DRAG.

An airplane in straight and level flight has four forces acting on it. These forces are lift, drag, weight, and thrust. The *lift* of the airplane acts perpendicular to the direction of the relative wind. The *weight* (or gravity) acts vertically downward from the center of gravity of the airplane. *Thrust* is the force which moves the airplane forward during flight, and the *drag* is the resistance of the atmosphere against the airplane's forward motion. When the airplane is in straight and level unaccelerated flight, the lift equals the force of gravity and the thrust is equal to the force of drag.

Figure 1-2 shows you how the four different forces that determine flight actually cancel each other out. The upper view shows the two forces, lift and drag. The lift force pulls the airfoil up, but, at the same time, the resistance of the airfoil (drag) pulls the wing backward. The resultant action, consequently, is not just a straight up or straight back motion—it is a combination backward and upward motion.

In the second view, the other two forces, thrust and weight, cause the airfoil to have just the opposite movement. The thrust causes the wing to move forward, but the weight (or gravitational pull) causes the airfoil to fall toward the earth. The resultant airfoil motion is a combination of these two motions—it is forward and downward.

Putting these four forces together produces the motion which is shown in the third view. As you will note, the forces act in different directions and cancel each other. If the force of lift is as great as the weight, the airfoil neither rises or falls (climbs or dives); and, if the thrust is as great as the drag, the wing does not

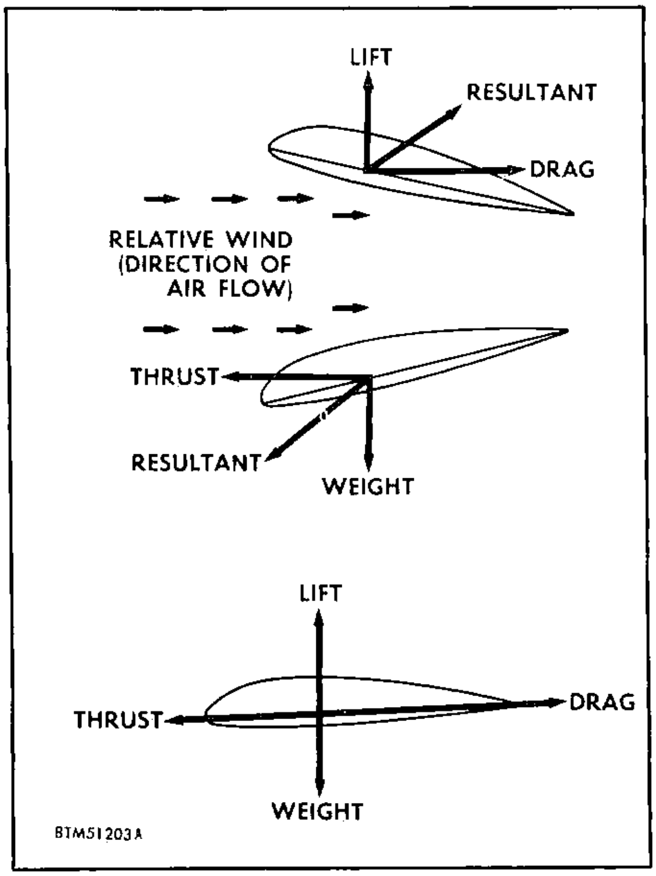

Figure 1-2. The Four Aerodynamic Forces On An Airfoil

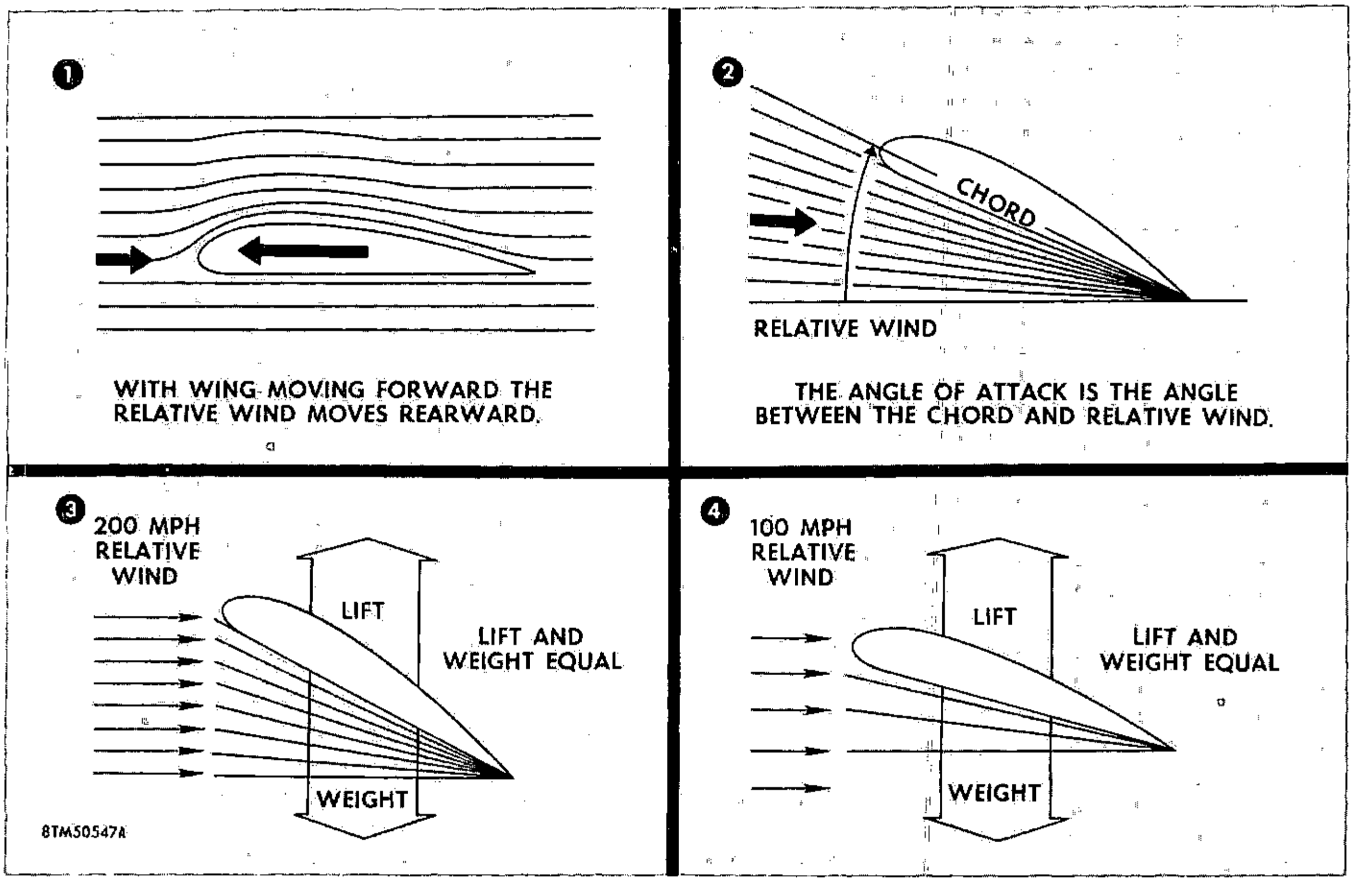

Figure 1-3. Relative Wind and Angle of Attack

move either faster or slower but moves at a constant speed. To go faster, we merely increase the thrust over the drag and the aircraft accelerates. Then when the thrust and drag again equalize, the aircraft no longer accelerates, but moves ahead at a faster constant speed than before.

While we are reviewing the principles of flight, we should consider the two neutral laws which combine to make it possible for an airfoil to support (lift) a heavy weight in air. The first law can be best illustrated by holding your hand out of the window of a moving automobile. As you incline your hand, the force of the air against your hand pushes it up. The airfoil, in this case, is your hand and it deflects the wind. This action creates a dynamic pressure effect on the lower surface of your hand, forcing it upward and backward. A similar effect on an airfoil produces about 25% of the total lifting force of the wing.

The second law, that of the Venturi tube principle, can best be demonstrated by grasping a piece of paper at each bottom corner with the thumbs and forefingers. Hold the paper a few inches from your lips, allowing the top edge of the paper to assume a gentle slope downward similar to the upper curvature of an airplane wing. Blow across the top of the paper and notice how the paper raises up at the far edge and assumes a horizontal position. The greater the velocity of air across the paper, the higher the paper will raise as the air passes across it.

Figure 1-3 shows how these two natural laws are applied to airfoils. When an airfoil moves through the air, both the airfoil shape and the airfoil angle of attack to the relative wind cause the air to be deflected. The deflection compresses the air below the airfoil and causes a high pressure on the under surface. The air traveling across the upper surface of the airfoil must travel a larger distance which consequently causes the speed of the air across the top portion to increase. This increase in the velocity of the air produces an area of low pressure next to the upper surface of the airfoil. In this manner, a pressure differential is created. The high pressure on the bottom surface of the airfoil causes the airfoil to be forced into the area of low pressure on the top surface. As the speed of the air flow increases, the pressure differential acting on the airfoil also increases. You can see from this how the necessary lift for flight is produced.

RELATIVE WIND AND ANGLE OF ATTACK.

As already mentioned, drag is the resistance of the atmosphere against the airfoil's forward motion and the

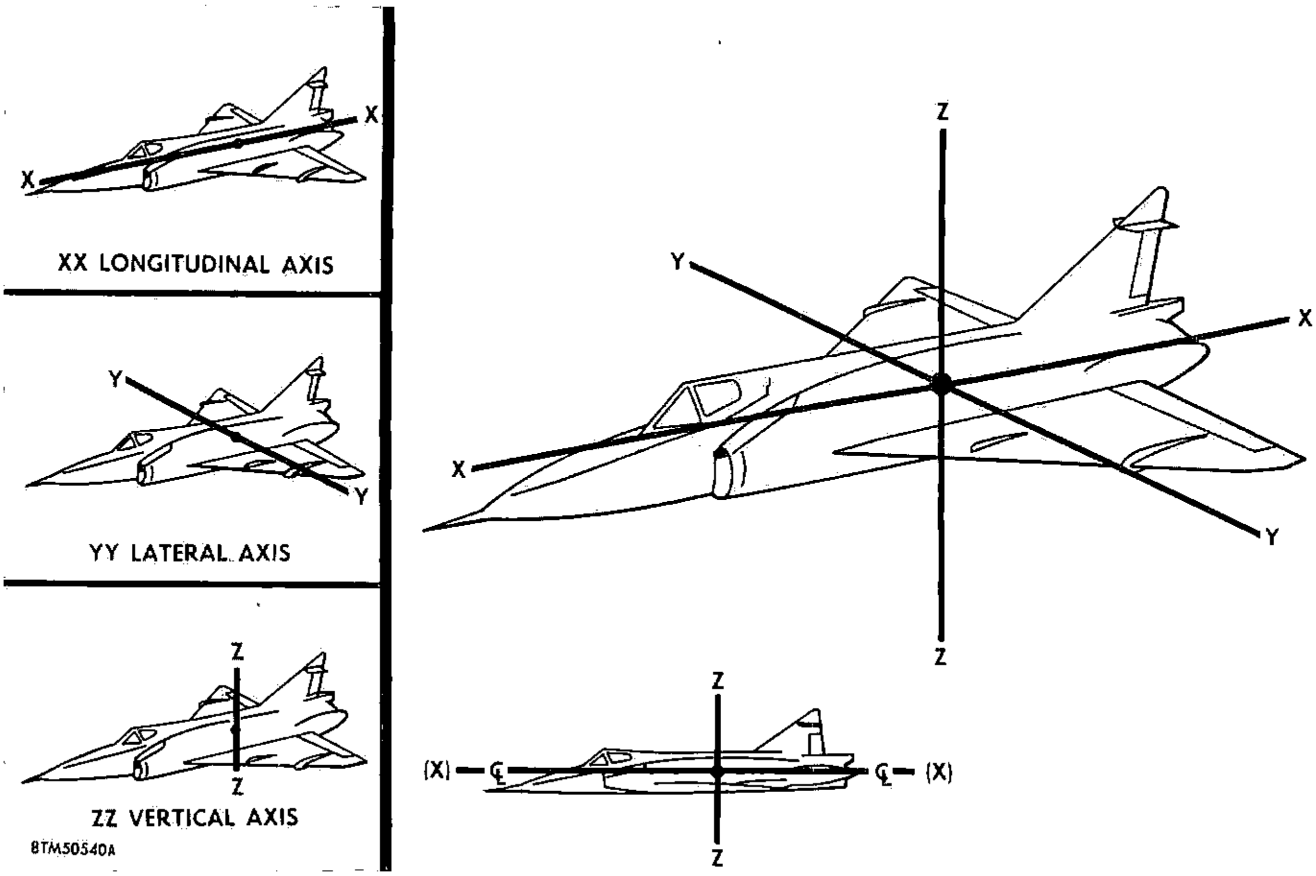

Figure 1-4. The Three Axes of an Airplane

drag will always act parallel to the relative wind. The relative wind is the direction of the air flow with respect to the airfoil. If an airfoil is moving forward horizontally, the relative wind moves backward horizontally. If the airfoil is moving forward and downward, the relative wind moves backward and upward. This is shown in the first view of figure 1-3. In flight, the actual path of the airplane determines the direction of the relative wind. It is the only motion of the aircraft in flight that produces a relative wind.

The angle of attack of an airfoil directly controls the distribution of pressure below and above it. The angle of attack can be defined as the angle between the chord of the airfoil and the direction of the relative wind measured in the manner shown in the second view of figure 1-3.

The shape of the airfoil cannot be effective unless it continually keeps attacking new air. If an airplane is to keep flying, it must keep moving. The speed with which the airfoil attacks the air, however, has a definite effect upon the lift generated by the airfoil. The airfoil lift is proportional to the square of the airfoil velocity in respect to the relative wind. An airplane traveling 200 mph has 4 times as much lift as one traveling 100 mph, if the angle of attack and the other factors are kept constant. Actually, you could not continue to travel in level flight and maintain the same angle of attack if you increased your speed; the airfoil lift would increase and the airplane would climb. For each angle of attack each airplane has a definite speed at which it will fly straight and level. To maintain the constant lift force which balances the weight in straight and level flight, the airfoil velocity must be decreased as the angle of attack is increased.

PLANES OF ROTATION.

There are three axes about which an airplane may turn. Whenever an airplane changes its attitude in flight in respect to the ground or any other fixed object, it must turn about one or more axes. Figure 1-4 shows you these axes which are imaginary lines passing through the airplane's center of gravity. You might think of these axes as imaginary axles around which the airplane turns like a wheel. At the center of gravity, where all three axes intersect, each of these axes is perpendicular to the other two. The axis which extends lengthwise through the fuselage from the nose to the tail is called the *longitudinal axis.* The axis which extends crosswise, from wing tip to wing tip, is the *lateral axis.* The axis which passes vertically through the center of gravity is called the *vertical axis.*

Motion about the longitudinal axis resembles the roll of a ship from side to side. In fact, the names used in describing the motion about an airplane's three axes were originally nautical terms, and have been adapted to aeronautical terminology because of the similarity of motion between an airplane and a ship. Consequently, the motion about the longitudinal axis is called *roll;* and, for the same reason, the motion about the lateral or crosswise axis is called *pitch*. You are probably familiar with the pitching motion of an ocean vessel as it plows into heavy seas. Finally, an airplane moves about its vertical axis in a motion which is called *yaw,* deviating from its course in an angular motion about the vertical axis, such as you would use in sculling a boat.

FLIGHT CONTROL SURFACES.

Roll, pitch, and yaw (the motions an airplane makes around its longitudinal, lateral, and vertical axes) are controlled by the three control surfaces. As shown in figure 1-5, roll is produced by the ailerons which are located at the trailing edges of the wing; pitch is affected by the elevators, the rear portion of the horizontal tail assembly; and yaw is controlled by the rudder, the rear portion of the vertical tail assembly.

You should remember that an airplane often rotates about all three axes at the same time. You can see a good example of this when an airplane begins a climbing turn. Coordinated movements of ailerons, elevators, and rudder cause the airplane to make, in one turning movement, rotations about the longitudinal, lateral, and vertical axes. Consequently, roll, pitch, and yaw can happen at the same time.

Flight control surfaces are hinged, or movable, airfoils. They are divided into two groups, the primary and secondary control surfaces. The primary group of control surfaces is made up of the ailerons, elevators, and rudder. You should also keep in mind that the elevators and the ailerons are combined on the F-102A and are called *elevons.*

Ailerons.

The ailerons control the movement about the longitudinal axis. There are two ailerons—one at the trailing edge of each wing. They are movable surfaces at the rear outer part of each wing. Moving the control stick or wheel to lower the aileron on one wing also raises the aileron on the other. The wing with the lowered aileron goes up because of its increased lift; the wing with the raised aileron goes down because of its decreased lift. The moving of either aileron, then, is aided by the simultaneous and opposite movement of the aileron on the other wing.

Rods or cables connect the ailerons to each other and to the control stick or wheel in the cockpit. When you apply pressure to the right on the control stick, the left aileron goes down and the right aileron goes up—rolling the airplane to the right. This happens because the down movement of the left aileron changes the wing camber and increases the angle of attack. The right aileron moves upward and changes the camber, resulting in a decreased angle of attack. Thus decreased lift on the right wing and increased lift on the left wing causes a roll and bank to the right.

Elevators.

The elevators control the movement of the airplane about its lateral axis and produce the motion known as pitching. They form the rear part of the horizontal tail assembly and are free to swing up and down. Like the ailerons, the elevators are attached to the control stick or wheel by means of cables or rods. Pushing the stick forward causes the elevators to move down, bringing the tail up and the nose down. Conversely, pulling the stick backward causes the elevators to move up, forcing the tail downward and the nose upward into a climb.

Like the ailerons, which are fastened to the trailing edge of the wing, the elevators are also hinged to a fixed surface—the horizontal stabilizer. Together, the stabilizer and the elevators form a single airfoil; and a change in the position of the elevators modifies the camber of the airfoil, increasing or decreasing the lift.

The elevators are the angle-of-attack control. When back pressure is applied on the control stick, the tail lowers and the nose raises, thus increasing the angle of attack. Under normal cruise conditions, this enables the pilot to control the air speed by regulating the elevators. If the airspeed is too low, forward pressure can be applied on the stick—lowering the elevators. This raises the tail and decreases the angle of attack. Of course this results in a loss of altitude, and the necessary correction is made by advancing the throttle.

Rudder.

The rudder controls the movement of the airplane around its vertical axis and causes the motion called yaw. Like the other primary control surfaces, the rudder is a movable surface hinged to a fixed surface —in this case the vertical stabilizer (fin). Rudder action is very much like that of the elevators, except that it swings in a different plane. The rudder swings from side to side instead of up and down. When the rudder is moved to one side, the shape of the airfoil is changed, producing a horizontal lift in the opposite direction.

The primary purpose of the rudder is to give vertical balance to the airplane and counter the effect of adverse yaw. As you know, the rudder does not turn the airplane; this is accomplished by banking the airplane through the use of the ailerons. However, in this maneuver, the lowered aileron on the outside of the turn

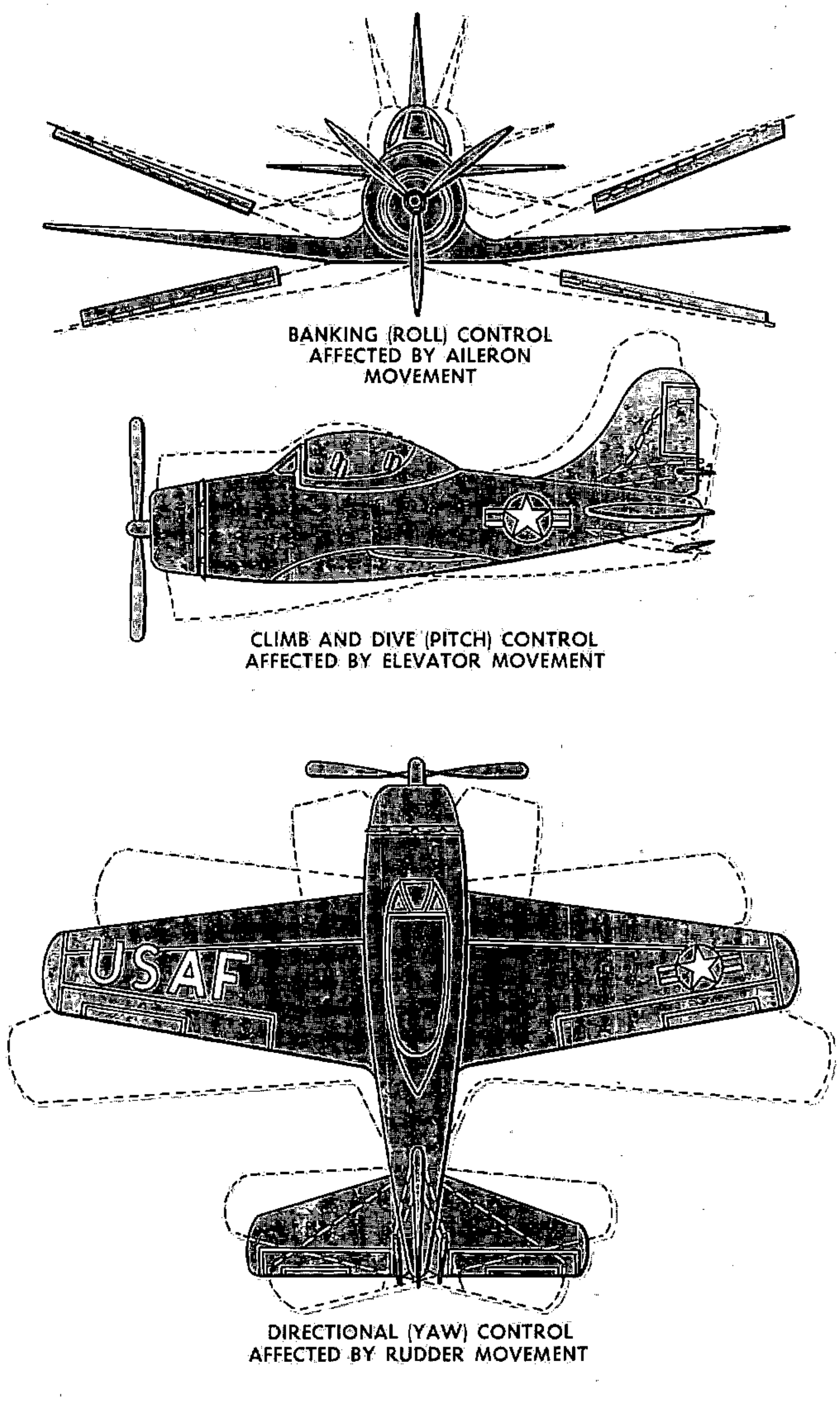

Figure 1-5. Pitch, Yaw, and Roll

acquires extra drag along with lift, and this force produces an opposite yawing which must be compensated for by adjusting the rudder. This is the only purpose of the rudder in normal flight—to offset the drag produced by the lowered aileron.

Secondary Control Surfaces.

The secondary control surfaces are the trim tabs, balance tabs, and servo tabs. They are used to reduce the force required to actuate the primary controls surfaces and for trimming and balancing the airplane in flight. These tabs are really small airfoils attached to, or recessed into, the trailing edge of the primary control surfaces.

TRIM TABS. Sometimes an airplane is loaded in such a way that it is slightly wing-heavy, tail-heavy, or nose-heavy. To offset such unbalancing forces, the pilot must exert a constant pressure on the stick or pedal in the opposite direction. To relieve this rather fatiguing effort, ailerons, elevators, and rudders are often provided with trim tabs.

A trim tab is a small, adjustable hinged surface, located on the trailing edge of the aileron, rudder, or elevator control surface. It is used to maintain balance in straight and level flight without pressure on the controls. This is accomplished by moving the tab in the opposite direction to that in which the primary control surface is to be moved. The airflow striking the tab causes the main control surface to move to a position that will correct the unbalanced conditions of the airplane. Trim tabs may be controlled from the cockpit or they may be the ground-adjustable type.

BALANCE TABS. Balancing tabs look like trim tabs and are hinged in approximately the same places as trim tabs. The essential difference between the two is that the balancing tab is connected to the wing structure by a rod, so that when the main control surface is moved in any direction the tab is moved in the opposite direction. The airflow striking the tab counterbalances some of the pressure against the primary control surface and enables the pilot to move and hold it in position.

SERVO TABS. Servo tabs are used primarily on large airfoils to help the pilot move the heavy primary control surfaces. The tab control wires or rods are linked in a manner that allows the tab movement to precede the movement of the main control surface. The airflow striking the tab moves the primary surface in the opposite direction. If this first pressure on the tab from the pedal is not sufficient to move the control surface, increased pressure releases a spring which brings the primary control wires into action. This type of tab enables the pilot to move the controls with considerably less control pressure than is required in airplanes of comparable size and speed without servo tabs.

Later in this chapter, you will find that the F-102A does not require any of the types of trim tabs discussed here. Since the flight controls are actuated by hydraulic power, the F-102A pilot does not need any trimming devices to assist him in controlling the airplane. And, as you will also learn, all trimming action is accomplished by an electrically actuated trimming system.

STABILITY IN AIRCRAFT.

All airplanes must possess stability to varying degrees for both safety and ease of operation. An unstable airplane requires constant adjustment of the controls, and consequently is difficult to operate. An unstable airplane is dangerous, for a sudden stall can send it into a fatal dive or spin. On the other hand, an excessively stable airplane can be equally dangerous, for it will tend to remain in its original path in spite of the pilot's efforts to control it.

There are three general types of stability—positive, neutral, and negative. Positive stability is usually referred to simply as stability, and negative stability is referred to as instability. An airplane with positive stability possesses the characteristics previously mentioned. If outside forces should disturb it from its normal flight, it tends to return eventually to its original position. This is shown in the upper view of figure 1-6.

A neutrally stable airplane does not tend to change its attitude in flight. However, if this attitude is changed by outside force or an adjustment in controls, the airplane does not tend to return to the original position. It remains in the new position until other forces influence it. Such an airplane is likely to tire the pilot, since he must control it constantly.

A negative stable, or unstable, airplane changes constantly from abnormal flight, and must be held straight and level by continuous use of the controls. If an unstable plane is put into a climb, it tends to climb more and more steeply until it finally stalls. In a dive, it tends to dive more and more steeply. There are no airplanes of this type in the Air Force.

Positive Stability.

There are two types of positive stability, static and dynamic. Static stability merely means that, if the airplane's course is disturbed, the airplane will tend to return to its original position. Dynamic stability is concerned with the oscillations an airplane goes through in attempting to return to its normal position of flight after that position has been disturbed. The swinging of the tail from one side to the other or up and down is called oscillation. The magnitude of these oscillations is measured by the extreme distance of the tail at each swing from an imaginary mean line. In a pendulum on a clock, the mean line is vertical, and

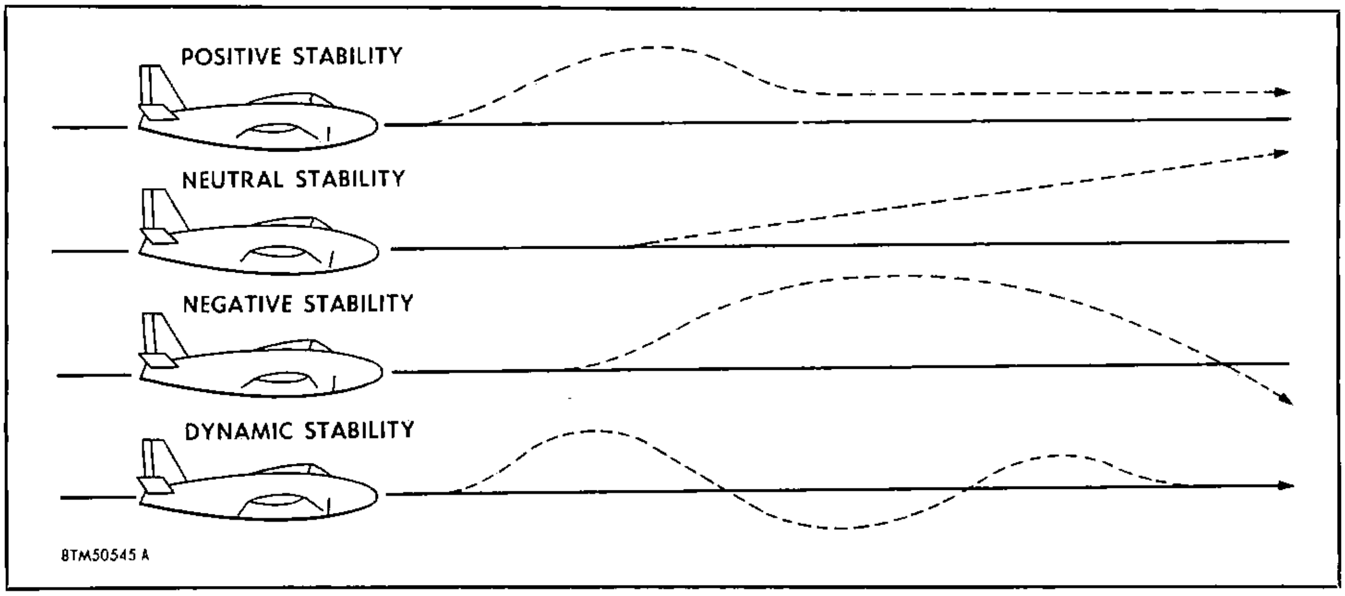

Figure 1-6. Aircraft Stability

the distance of each swing to the right and to the left indicates the size, or *amplitude,* of the oscillation. If, in the case of an airplane, these oscillations steadily decrease in amplitude, the airplane is stable. It will eventually return to the original flight attitude as shown in the bottom view of figure 1-6. However, if the oscillations increase in amplitude, the airplane is dynamically unstable.

Stability About the Axes.

Air Force planes are designed to have stability about all three axes. If the nose drops, if one wing lowers, or if the tail swings to one side, the airplane should quickly return to the normal position. There are certain features incorporated in the design of an airplane which give it stability about the longitudinal, lateral, and vertical axes.

LONGITUDINAL STABILITY. Longitudinal stability is the quality which makes an airplane stable about its lateral axis—the axis which runs through the wings from tip to tip. It involves the pitching motion of the airplane as its nose goes up and down in flight. A longitudinally unstable airplane tends to climb or dive until it goes into a stall or steep dive from which it cannot pull out. Thus, the airplane with longitudinal instability becomes dangerous to fly.

LATERAL AND DIRECTIONAL STABILITY. Stability about the *longitudinal axis,* which runs from nose to tail, is called lateral stability. This helps to stabilize the rolling effect when one wing lowers and the other raises. Of course, you must remember in considering stability, that no airplane has complete stability. If a plane were perfectly stable, the controls would be slow in responding, and maneuvers would be difficult, if not impossible. If an airplane is longitudinally stable, it does not necessarily remain fixed in a level, forward path, but does tend to maintain its attitude. Likewise, an airplane that is laterally stable does not always keep its wings level, but it does tend to return to its original attitude.

Stability about the *vertical axis,* which concerns the yawing moment, is known as directional stability. This does not mean that the airplane keeps a compass direction, but that it keeps a constant direction with regard to the relative wind.

DEVELOPMENT OF FLIGHT CONTROLS.

The early pioneers of flight contributed much to the science of aeronautics. Although all of the experimenters prior to the Wright brothers never achieved flight in a heavier-than-air machine, they formulated some of the fundamental concepts. Their early attempts were concerned only with getting off the ground. Control or maneuverability was not a prime consideration.

In December 1903, the Wright brothers proved to the world that man could fly, although their flying machine is almost comical when compared to our modern supersonic airplanes. Figure 1-7 shows one of the earlier models of the Wright brothers. Note that the concept of flight control surfaces was somewhat different from that which we know today. One of the distinguishing characteristics of this model was the forward vertical surface which the pilot used to offset the yawing tendencies. The wings have no small control surfaces to control airplane roll movement; they were merely "warped" to induce the rolling motion. The elevators at the aft end of the machine were built as one integral unit instead of two (stabilizer and elevator). When the pilot wished his machine to go up or down, he moved the whole elevator.

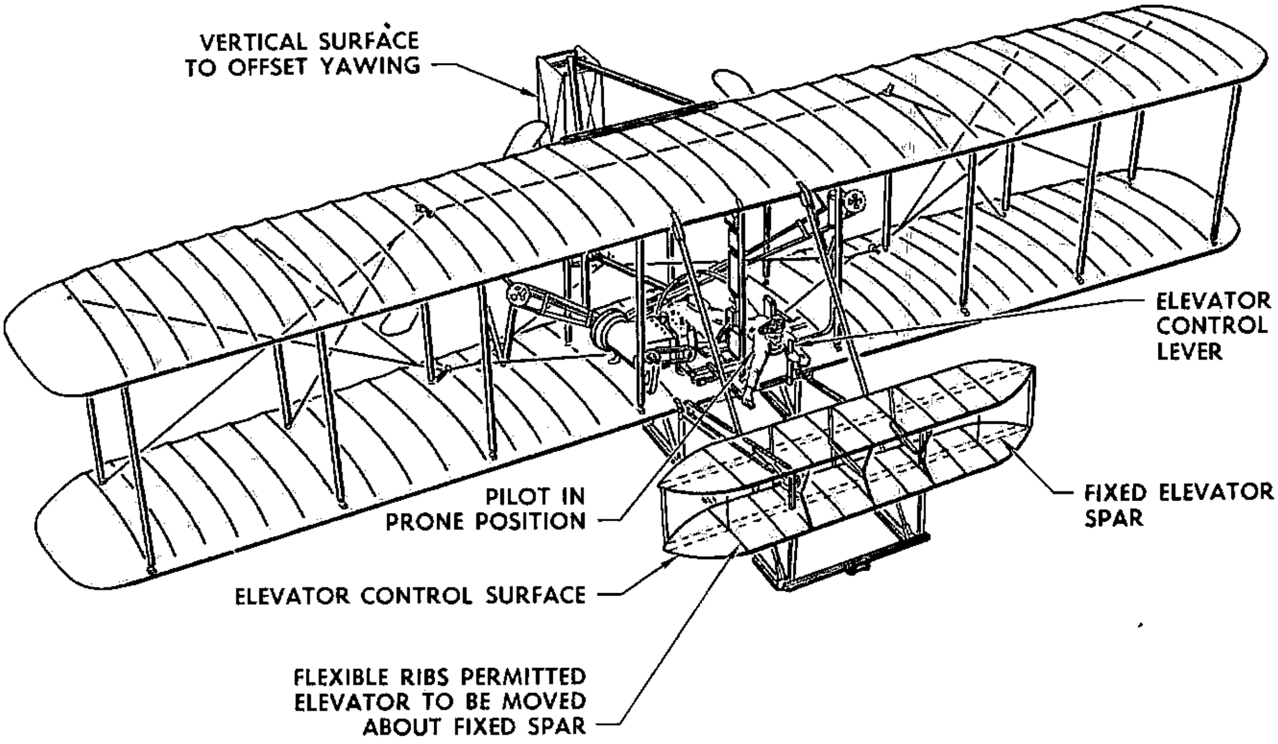

Figure 1-7. Flight Control Surfaces on Early Airplanes

By the beginning of the first World War, the airplane had been developed to a point where airplanes could be successfully controlled in almost all attitudes of flight. The airplane of this age had practically all of the basic control surfaces of today's conventional airplanes. As shown in figure 1-8, the rudder had been moved to the rear of the airplane, the elevators were being designed as a movable surface attached to the horizontal stabilizer, and the wings were incorporating ailerons, which were movable surfaces attached to the aft edge of the wings.

Shortly after the end of World War I, progress made in the design of airplanes necessitated the development of better flight control surfaces. It is quite interesting to note, however, that the general location of the flight control surfaces has not changed extensively from those of the early World War I airplanes. Figure 1-9 shows the location of flight control surfaces on conventional-winged and delta-winged aircraft. In the left view, note that tabs have been added to the control surface group. These tabs are used in the manner described earlier in the review of flight principles. The airplane shown in the right view has only three control surfaces. The elevons have replaced the elevators and the ailerons, but the rudder has remained the same in respect to function and location.

OPERATING THE FLIGHT CONTROL SURFACES.

The operation of flight control surfaces has also passed through several development stages. In a general sense, the three most prominent stages are the *direct manual* control, the *fully-powered* control, and finally, the *autopilot* control. Although these types of flight control operation have spanned almost half a century, all of them are in use today.

Direct Manual Control.

The direct manual control of the control surfaces was used on early types of airplanes. In the case of the Wright brothers' airplane, the pilot lay prone in the fuselage and directed the airplane by pulling on cables which were attached to the control surfaces. As flight controls were improved, the linkage which attached the control surfaces to the pilot controls in the cockpit became more complicated, but the basic principle was the same. As shown in figure 1-10, the pilot moved the control stick and the mechanical linkage then operated the control surface. With this type of system, the pilot had a definite "feel" of the amount of air pressure that he was forced to overcome with control stick operation. This, however, turned out to be one of the biggest disadvantages of the non-power-operated systems. The air load forces on the control surfaces vary with the speed of the airplane. Although

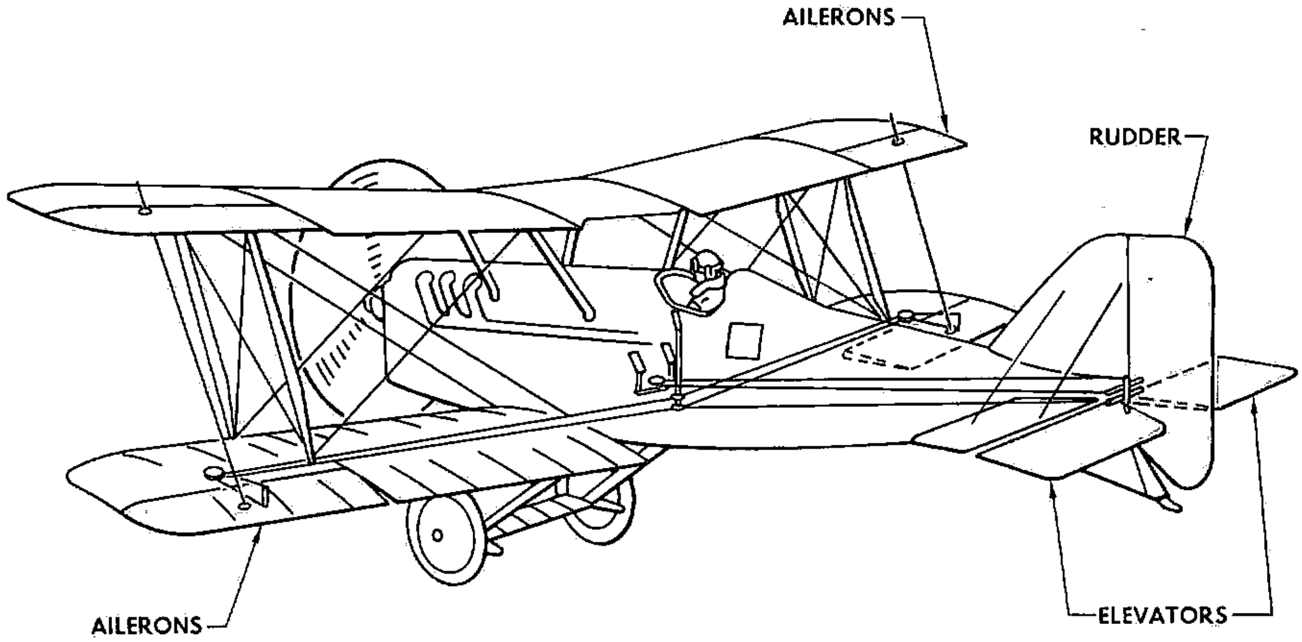

Figure 1-8. Flight Control Surfaces on World War I Airplanes

little effort was needed to control the very early airplanes, the faster models encountered such great air loads on the control surfaces that the task of flying the airplane began to exceed the physical limitations of the pilot.

Fully Powered Control.

The first attempts to compensate for the increased air loads dealt with the moving of the control surface hinges to counterbalance the loads, but this did not prove satisfactory in that a perfect balance would not be obtained that would suit all air load applications. The problem was finally solved by the development of the powered flight control system. Two typical systems of this type are shown in figure 1-11. In the upper view, note that the control stick does not physically connect to the control surface—it connects to and operates only the control valve shown in the center of the schematic. You will note that the control valve can direct hydraulic fluid to either side of the actuating cylinder which moves the control surface.

There are several distinct disadvantages of this type of control system. The pilot has no method for controlling the amount of control surface travel other than moving the stick just a small amount at a time. He also has no method for obtaining any "feel" of the control surface pressures.

The system shown in the lower view shows how one of the disadvanages has been removed in the modern flight control system. Note that the control stick linkage still connects to the control valve but that the additional feedback linkage also connects to the control valve. When the pilot positions the control stick for control surface displacement, the control valve opens and directs fluid to the actuating cylinder. This actuating cylinder in turn causes movement of the control surface and the feedback mechanism. When the control surface reaches a position that corresponds with the position dictated by the control stick, the feedback mechanism will have actuated the control valve so that the flow of hydraulic fluid to the actuating cylinder is shut off. This effectively locks the control surfaces in the position desired by the pilot. When the pilot moves the stick back to neutral, the reverse action takes place and the control surface stops in its neutral position.

The Autopilot.

The type of control system described above is quite satisfactory in most respects. One drawback of the system, however, was that it required the pilot to operate

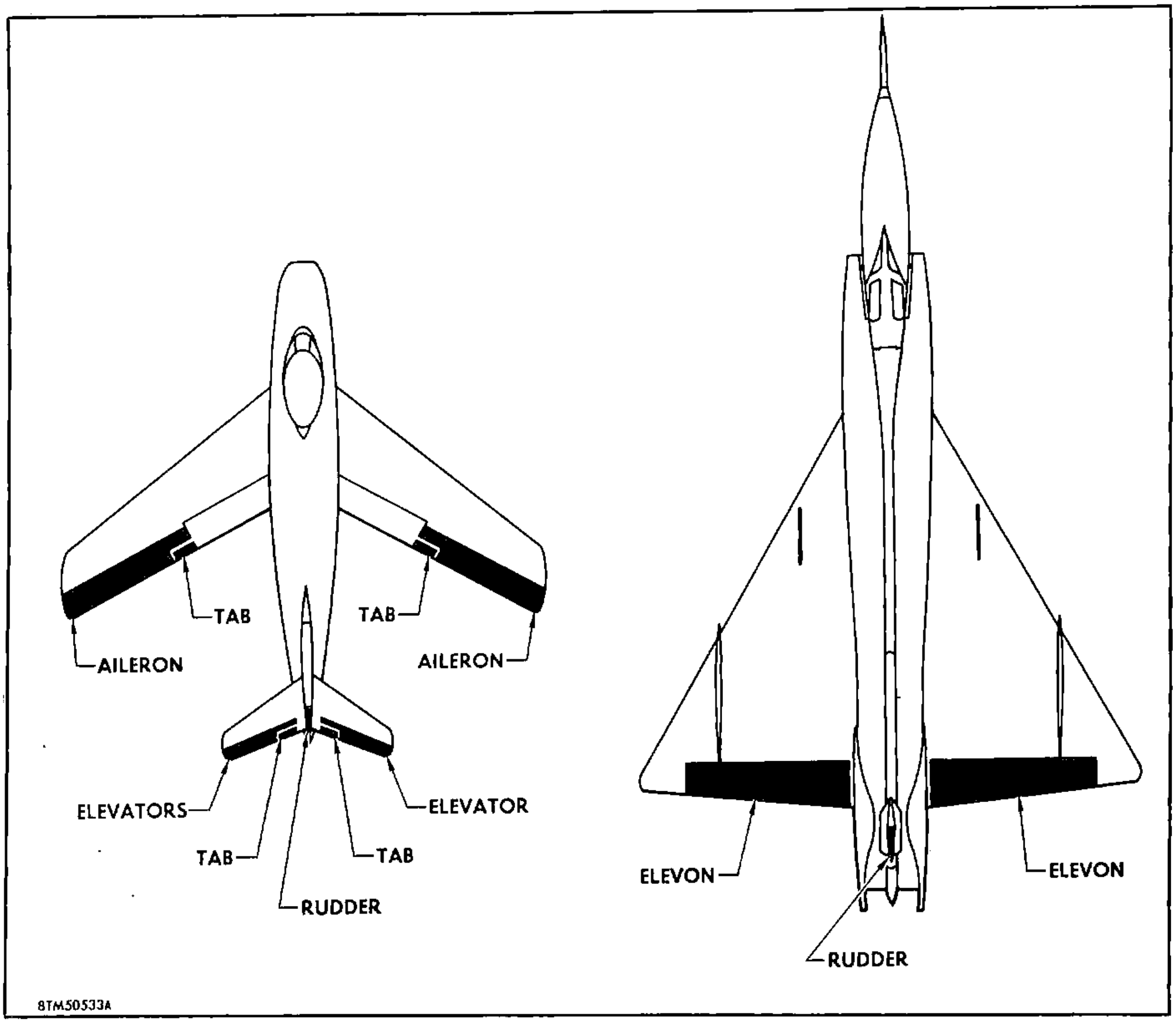

Figure 1-9. Flight Control Surfaces on Modern Airplanes

the controls all of the time. Because of the human fatigue factor, the next logical step in the development of flight control systems was to develop a flight control system that would relieve the pilot of all flight control duties during straight and level flight. This type of system is called the autopilot control system.

Figure 1-12 shows an autopilot control system in simplified schematic form. By comparing this schematic with the previous one, you will note the addition of the gyro. This is the heart of the automatic pilot control system. Once the pilot has selected a heading and established the flight attitude, he can leave the control problems to the autopilot. As the airplane strays from the present course and attitude, the gyro detects the changes and makes the necessary changes in the position of the hydraulic control valve. The control valve directs fluid to the actuating cylinder which then moves the control surface. When the control surface reaches the corrected position as indicated by the gyro, the feedback system repositions the control valve and stops the supply of hydraulic fluid to the actuating cylinder. When the course has been reestablished, the reverse operation takes place and the control surface returns to its neutral position.

PROBLEMS ENCOUNTERED ON SUPERSONIC AIRCRAFT.

With the advent of supersonic aircraft, many new problems in control and stability presented themselves, and new flight control systems had to be designed to solve them. Even the language is constantly changing. It is no longer sufficient to give airspeed in miles per hours. The speed of aircraft is now spoken of in terms of speed of sound. The term "Mach

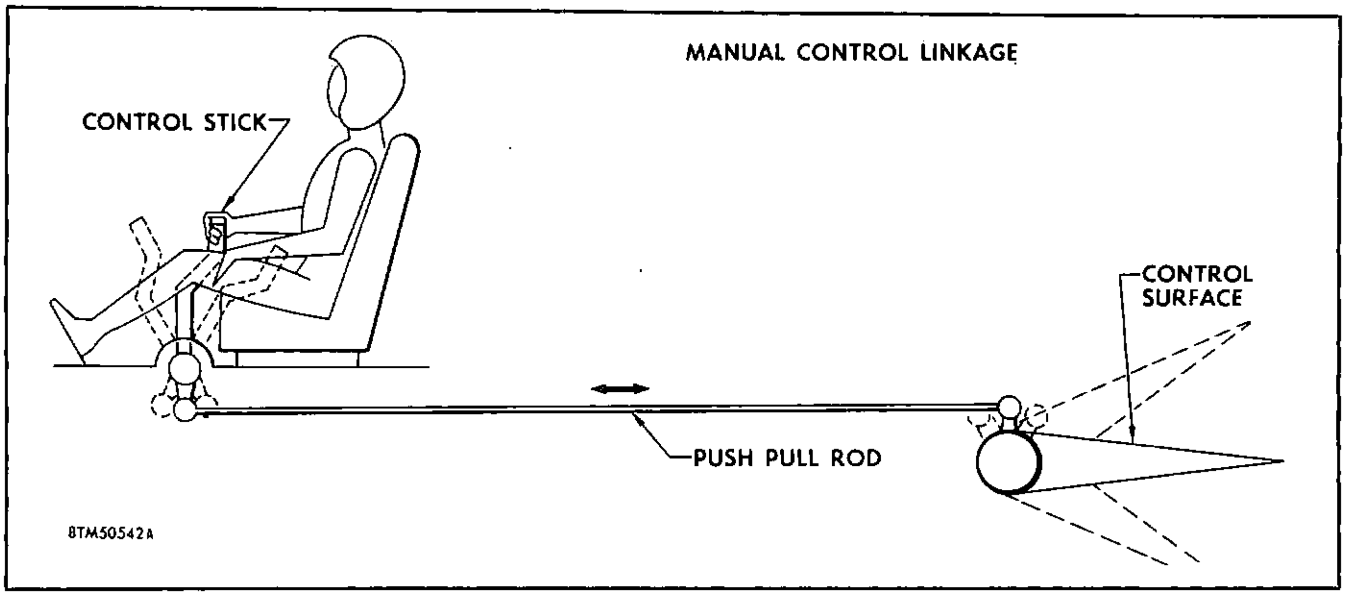

Figure 1-10. Direct Manual Operation of Control Surfaces

number" has come into accepted use. This is the ratio of the aircraft's speed to the speed of sound at the flying altitude.

Another term which has come into general use is "compressibility," or "compressibility limit." As a wing moves through the air, the air passing over the top of the wing flows at higher speeds than the air flowing under the wing. This difference in speeds causes a large pressure differential between the two masses of air, and shock waves result. Although the wing and control surfaces do not actually enter into the area of compressibility above the wing, they are still subject to the resulting shock waves.

The problem of controlling an airplane becomes very difficult as airplane speeds approach the sonic area. As shock waves are formed on both the upper and lower surfaces of the airfoil, the lift factor is greatly reduced and the aircraft begins to dive. As a safety measure, early high-speed aircraft had dive recovery flaps under the wings or fuselage. When the dive flaps were extended, it increased the drag and forced airflow downward so that the nose would come up.

Ailerons and rudders on conventional-winged airplanes also came in for their share of trouble on high-speed aircraft. The unstable transonic flow over the wings caused the ailerons to oscillate rapidly, creating a condition known as "buzz." Another oscillation called "snaking" also occurred about the vertical axis around the rudder area.

One of the worst problems of control and stability on airplanes was encountered just at transonic speeds. The occurrence of shock waves over the rear part of the surface generally created a wide wake so that the control surfaces at the rear of the wing lay completely in a "dead water" area. Then, no matter how much the controls were moved, no effect was obtained, from them. In other words, the control surfaces became useless. As the airplane speed increased, this "dead water" region gradually moved farther back; and at supersonic speeds, the control surfaces would be out of the wake of the shock. In these conditions, the control surfaces again became effective.

Another problem which has presented itself in the development of supersonic airplanes is the difference of critical Mach numbers for different types of wings. As you can readily understand, the airfoil section must span the transonic region without encountering strong normal or oblique shock waves. The lift of the wing must not deteriorate and any increase in drag must be due to the increase in velocity and not energy losses. The center of pressure, or point through which the lift is acting, should remain fixed near the center of the wing so that changes in direction will not develop. The wing must also give high lift at low subsonic speeds so that the aircraft may be landed at reasonable speeds.

Two methods for increasing the airplane's velocity without exceeding the critical Mach number are the use of swept-back wings with very low aspect ratios. The pressure fields of any wing always act perpendicular to the wing leading edge. Air flowing from any other direction has no effect on the wing's characteristics. Because of this, airflow across the wing on a swept-back design will always be lower than the actual free stream velocity. Sweepback also helps to raise the critical Mach number and reduce the drag by keeping the wing as far as possible inside the angular shock wave (Mach cone) of the fuselage. A Mach cone will form on the nose of the fuselage

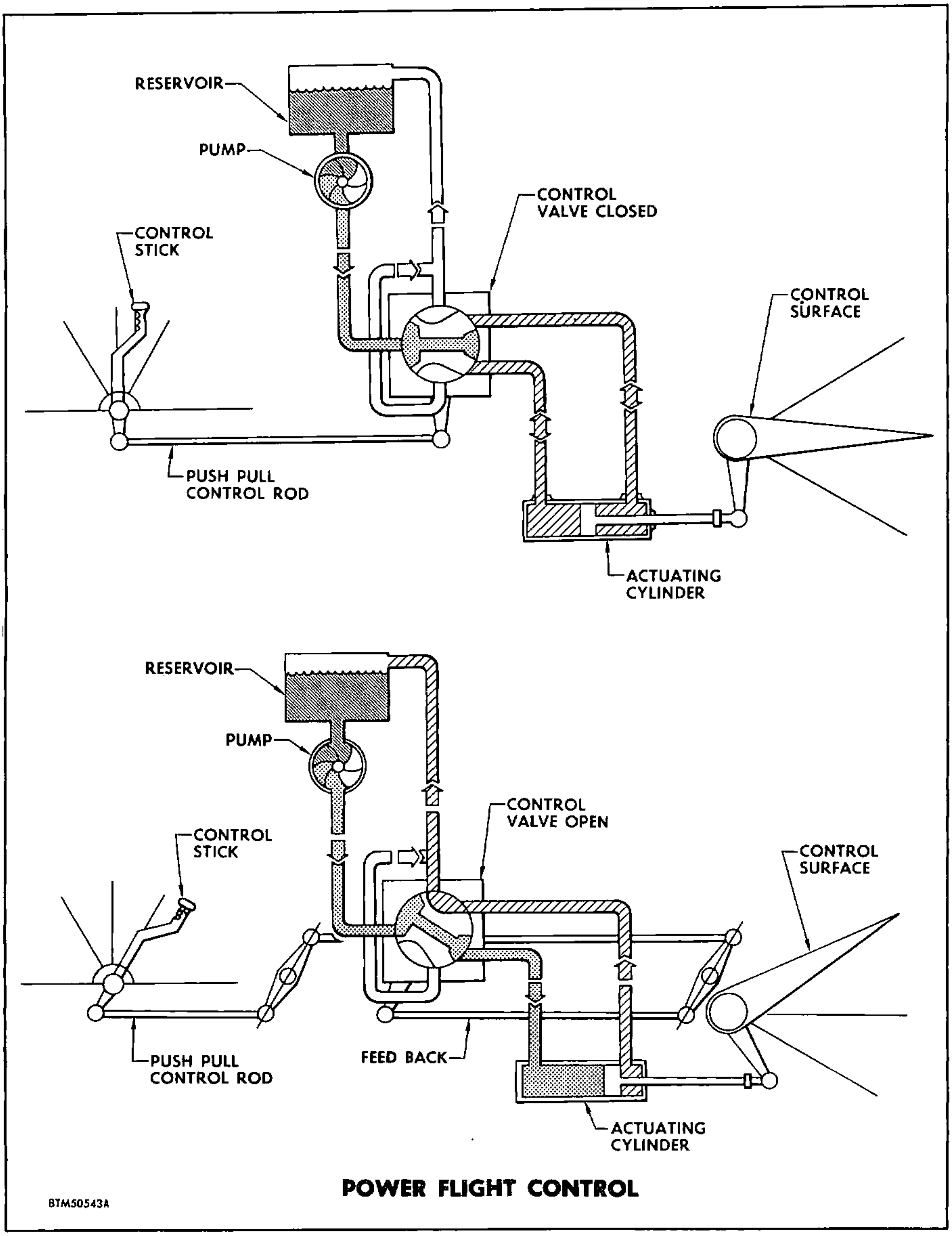

Figure 1-11. Fully Powered Operation of Control Surfaces

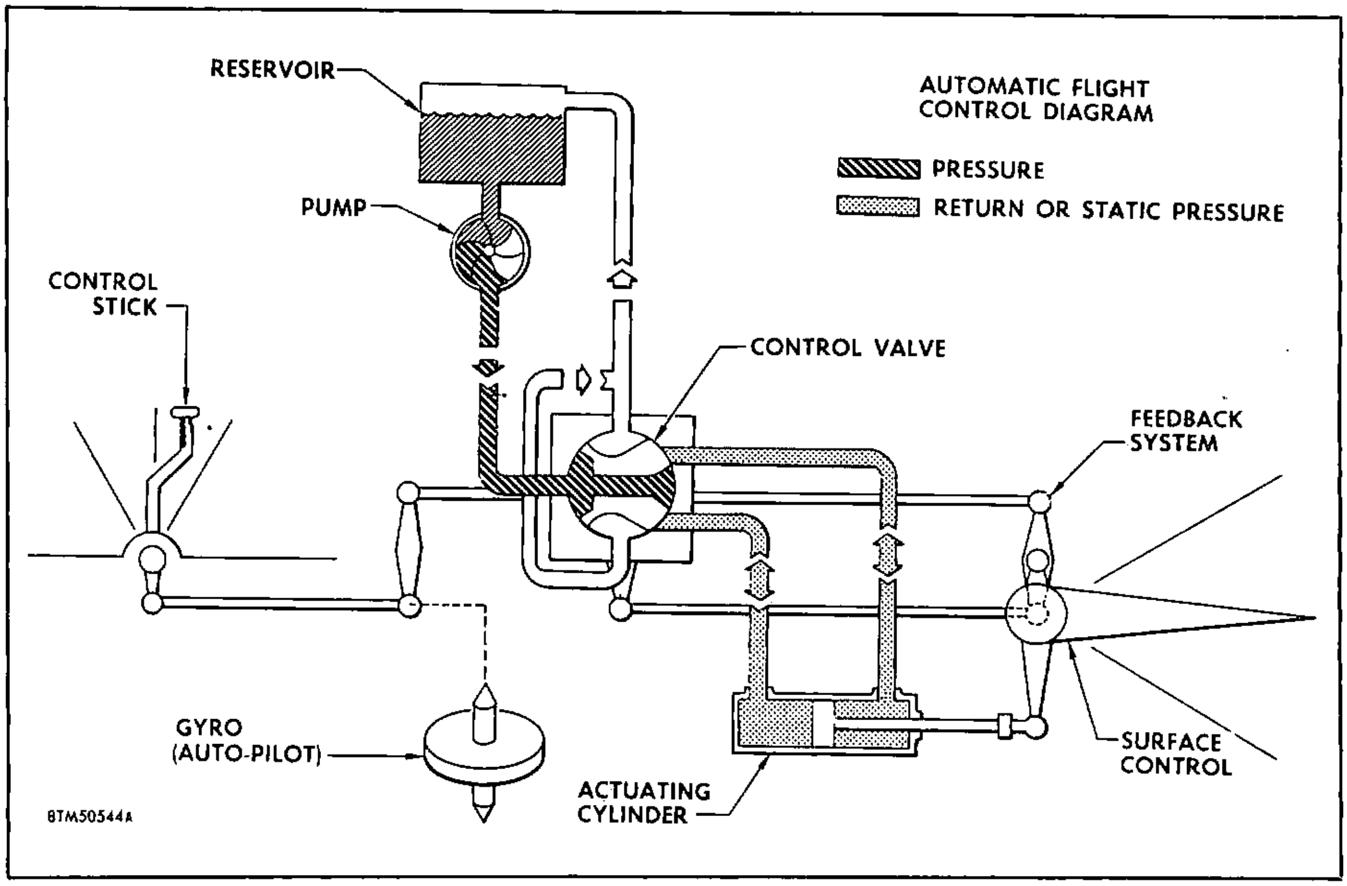

Figure 1-12. Automatic Operation of Control Surfaces

and the velocity inside the cone will be lower. It is usually desirable to keep the wing inside the area of lower velocity. If the wing is moved as far back on the fuselage as possible, most of it will fall inside the Mach cone.

It is very difficult to build a strong swept-back wing, so another idea was to build a delta-shaped wing to give the wing the needed structural strength and also provide the necessary airfoil characteristics for a stable flying armament platform. With the added structural strength, the wing can be made exceptionally thin with a low aspect ratio. This, of course, makes the airflow over the top and bottom of the wing travel at about the same velocity, and the transition from subsonic, through transonic, and into supersonic speeds can be made with a minimum amount of shock waves induced by compressibility. The delta wing also eliminates the use of elevators and horizontal stabilizers. This in turn, eliminates the need for controlling the air flow for both the wing control surfaces and the empennage control surfaces.

THE F-102A FLIGHT CONTROL SYSTEM.

The flight control surfaces on the F-102A, as in most supersonic airplanes, are moved by hydraulic power. Hydraulic power is used because of the high air loads imposed on the control surfaces during high-speed flight operation. Both the primary and secondary hydraulic systems (or one of them in case of malfunctioning of the other) supply the hydraulic pressure that moves the control surfaces.

The F-102A flight control surfaces shown in figure 1-13 consist of only elevons and rudders. Although the control surface arrangement is somewhat unique, the pilot controls the airplane in the same manner that he would control a conventional-winged airplane. The pilot operates the elevons by means of control stick and moves the rudder by means of rudder pedals. In figure 1-14, note that the forward and aft motion of the control stick gives elevator action while the left and right stick motion produces aileron action.

Since the hydraulic portion of the flight control system is irreversible, airloads on the flight control surfaces are not transmitted back to the pilot. Consequently, an artificial "feel" system is required to simulate an airload condition. This system of simulating airloads on the control surfaces is known as the artificial "feel" system. The function of the feel system is to provide the pilot with feel forces which vary the resistance of movement of the control stick and rudder pedals according to the speed of the airplane.

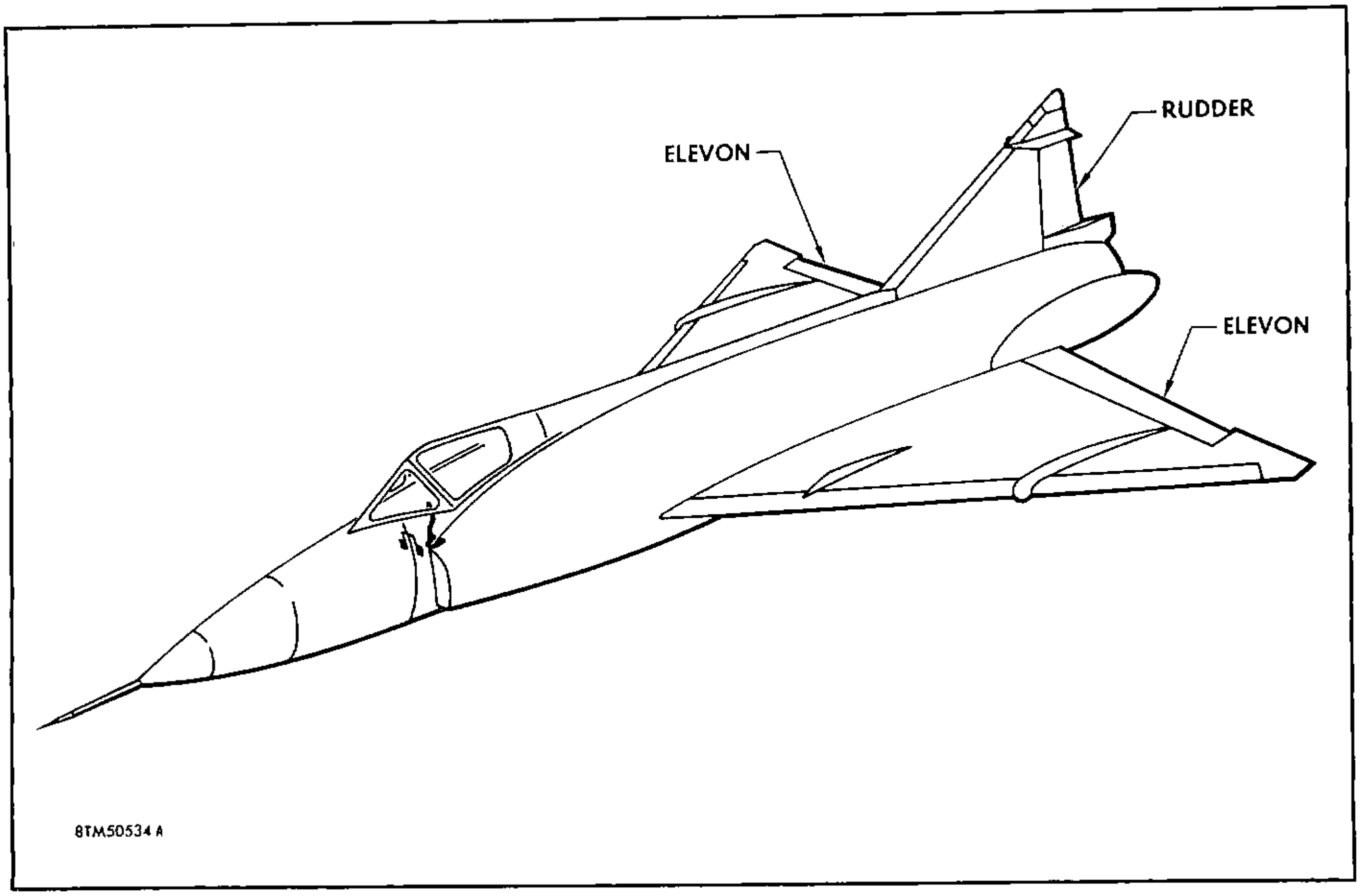

Figure 1-13. F-102A Control Surfaces

THE ELEVON CONTROL SYSTEM.

The elevon control system combines elevator and aileron motion in a single pair of elevons. By observing the diagram shown in figure 1-15, you can see how this control system operates. Movement of the control stick fore and aft produces elevator motion; movement of the stick to either side produces aileron motion. A series of mechanical linkages and cables transmits control motion from the stick to the "T" shaped bar at the elevon trim actuator. At the actuator, aileron and elevator motion are "mixed" and a proportionate amount of linkage movement is transmitted to the walking beam at each elevon.

When the walking beam rotates in either direction, it displaces the elevon control valve in a corresponding direction. This valve displacement is through a direct mechanical linkage between the walking beam and the control valve. The displacement of the elevon control valve moves two internal spool valves. These spool valves direct primary and secondary hydraulic system pressure to the inboard and outboard actuating cylinders which in turn move the elevons in a corresponding up or down direction.

When the elevons reach the position called for by the stick movement, the walking beam pivot point will also have moved. Walking beam movement causes the mechanical linkage between the beam and control valve to move in an opposite direction from the initial movement set up by the control stick and thus return the control valve to neutral. With the control valve in neutral, hydraulic pressure is cut off to the actuating cylinders and the elevons stop moving.

Simulated "feel" of the elevons is put into the system by the elevator "Q" feel cylinder shown in the diagram. This cylinder uses ram air that enters through the "Q" intake and air from the low pressure pneumatic system. The variable air pressure regulator controls the air to the cylinder, and the faster the airplane flies the more tension the cylinder puts into the controls.

THE RUDDER CONTROL SYSTEM.

The rudder control system actuates the conventional rudder surface on the vertical stabilizer. By observing the illustration in figure 1-16, you will note that pilot control is effected by conventional rudder pedals. When the rudder pedals are moved, the force is transmitted through cables and mechanical linkage to the rudder control torque tube. Rotation of this tube displaces the hydraulic control valve which then meters

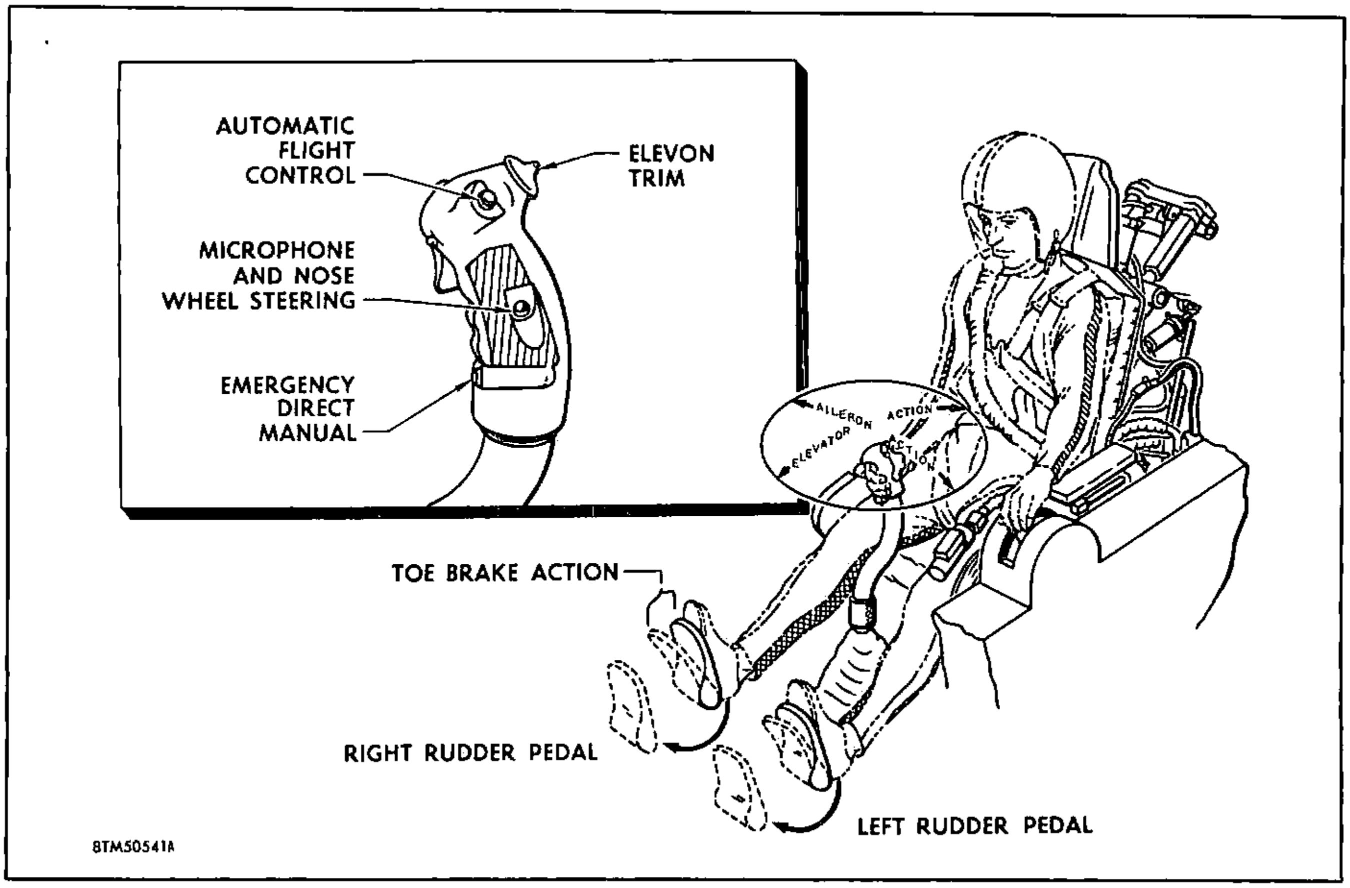

Figure 1-14. Control Stick and Rudder Pedal Action

hydraulic pressure to the rudder actuator cylinder. The rudder actuator piston rod is attached to the airplane structure. Hydraulic pressure moves the cylinder and its attached control valve and servo actuator which in turn moves the rudder surface. The rudder moves only as long as the pilot moves his pedals. When rudder pedal motion stops, the mechanical follow-up mechanism between the torque tube and the control valve returns the control valve to neutral and stops rudder surface movement.

The rudder control has a "feel" system for a simulated air load condition that operates much the same as the elevon feel system, except for one source of air pressure. This system uses "Q" (ram air) pressure and air from the high pressure pneumatic system. This resulting air pressure moves the cylinder shown just below the centering cylinder in the illustration. This "feel" cylinder puts tension into the rudder control just as the "feel" cylinder did in the elevon control. The faster the airplane flies the more tension the "feel" cylinder puts into the rudder control system.

MODES OF FLIGHT.

There are three modes or means available to the pilot for controlling flight attitude of the airplane. These are the Direct Manual, Manual, and the Automatic Flight Control System (AFCS) modes which are discussed separately in the following paragraphs. The Manual and AFCS modes serve as an automatic pilot on the F-102A.

Direct Manual.

When the two-position flight mode selector switch, located on the flight mode panel, is placed in the Direct Manual mode of flight, the pilot has direct control of the elevons and rudder. Direct Manual is normally used for takeoffs and landings, and any time the pilot desires to have only control stick, rudder pedal, and trim switch movements determine actuation of the control surfaces.

Manual.

Placing the flight mode switch in the MANUAL position completes a circuit which brings the flight control system's pitch and yaw damper systems into operation. In the Manual mode the pilot still flies the airplane much the same as he does in the Direct Manual mode, except that he now has the assistance of the two automatic stabilization systems—the pitch damper and yaw damper systems. If the pilot were required to make control corrections in high speed

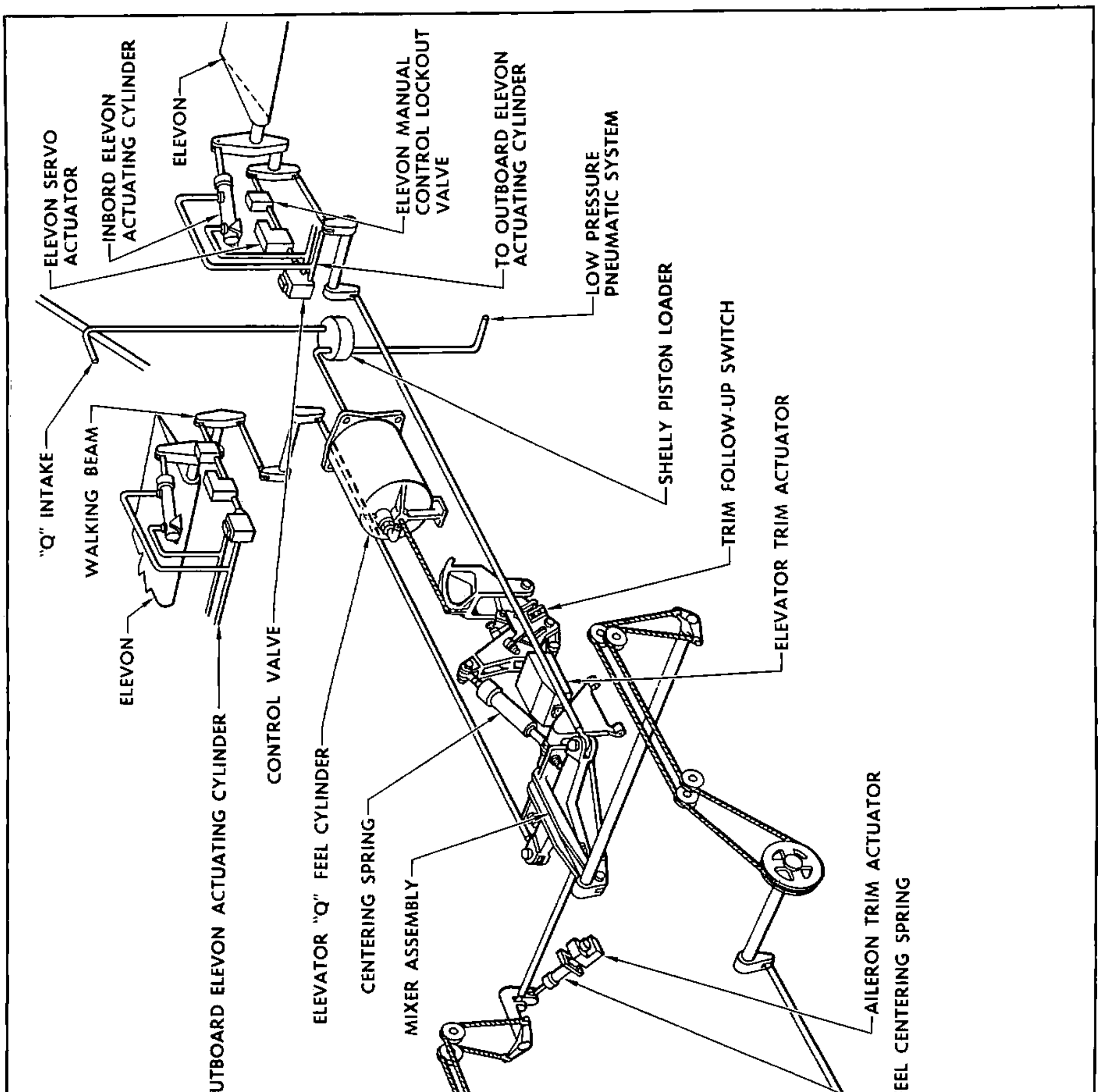

Figure 1-15. Elevon Control System Diagram

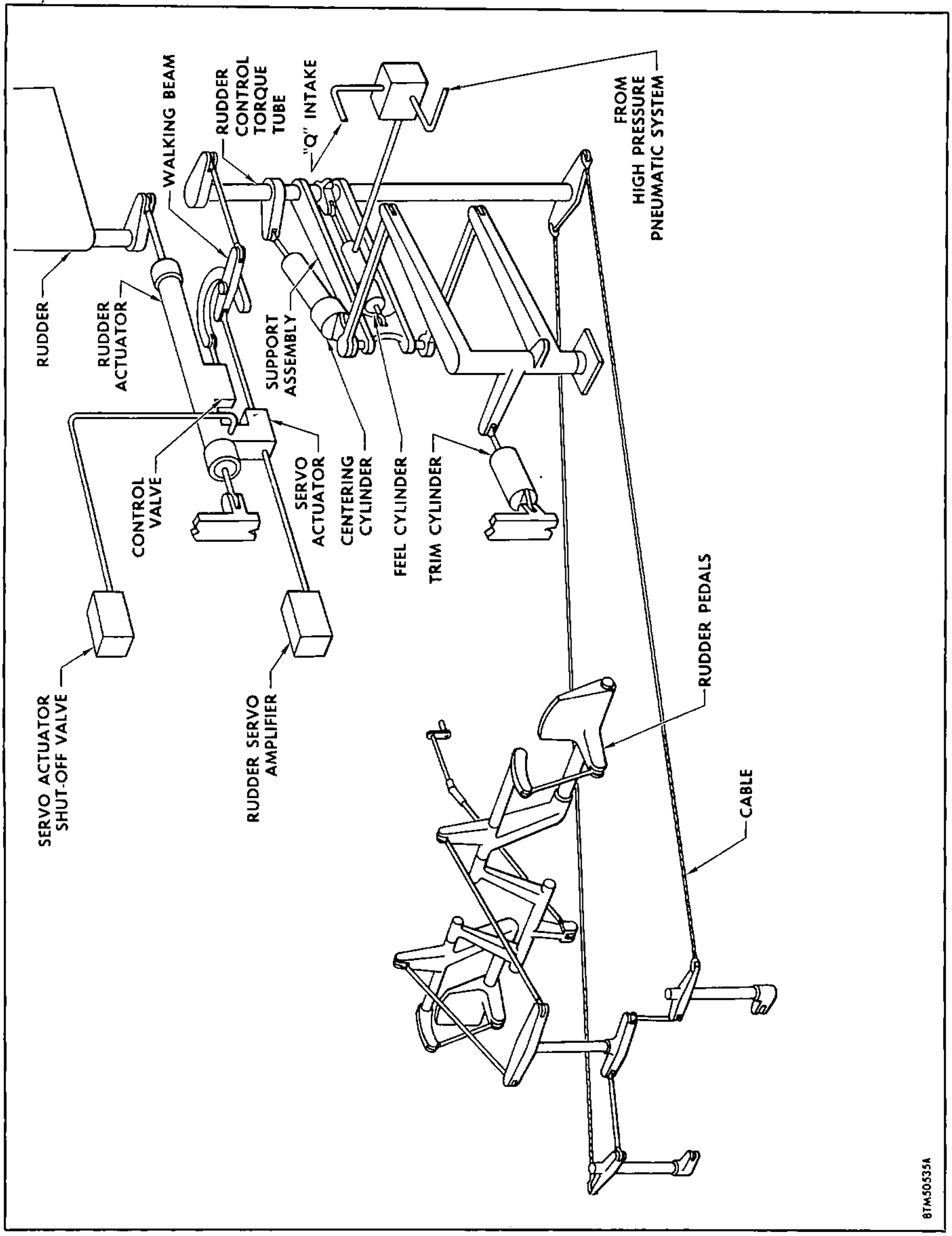

Figure 1-16. Rudder Control System Diagram

flight, he would probably over-control in the opposite direction and thereby cause an unstabilized condition. However, in the Manual mode of operation, the pitch and yaw damper systems sense the necessary flight control corrections and send these corrective signals to the servo actuators (see figures 1-15 and 1-16) which in turn actuate the control surfaces through the control valves as in the Direct Manual mode. These flight control corrections are sensed and made by the pitch and yaw damper systems before the pilot is even aware that flight corrections are necessary. Thus, in the Manual mode the pilot still has direct control plus the assistance of the damper systems to keep the airplane stable and provide turn coordination.

Automatic Flight Control System.

When the pilot places the flight controls in the Automatic Flight Control System, all mechanical pilot input and feedback motion is "locked-out" of the elevon control system. The Automatic Flight Control System (AFCS), which is a part of the MG-10 Weapon Control System, takes over and provides elevon control and turn (rudder-aileron) coordination.

When the aircraft is controlled by AFCS, the control stick is "locked-out" and the pilot is relieved of direct flight control. The pilot, however, can override the AFCS, if necessary, by exerting sufficient force on the control stick. Operation under AFCS allows the pilot to devote his attention to the operation of the fire control system. In the AFCS mode, the MG-10 system sends signals to the pitch and yaw damper systems to control flight. Therefore, these AFCS signals must control the aircraft as does the pilot when he has stick control. Thus, in AFCS the control signals provide "full authority"; that is, the AFCS has complete control of the elevons and turn coordination and can move the elevons to their full limits of travel (25° up and 8° down). In the Manual mode, the automatic type of stabilizing control provided by the damper systems is limited to a very small amount of elevon movement. This limited damper control of the elevons is due to the damper system arrangement of mechanical and electrical feedback. When the mechanical feedback arrangement is "locked-out" in the AFCS mode, as is the control stick, feedback is governed by a control surface potentiometer. Since the AFCS is thus freed from the mechanical feedback restriction, it has full control of the elevons.

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Chapter II

THE ELEVON CONTROL SYSTEM

Contents *Page*

In the preceding chapter you learned why flight controls are necessary. You also learned about the different types of controls that are used on airplanes and how they control the direction and attitude of the airplane about the longitudinal (roll), lateral (pitch), and vertical (yaw), axes. In Chapter I, you also received an overall picture of the flight control systems on the F-102A interceptor.

This chapter will be devoted to a detailed discussion of the F-102A elevon control system and how it operates, the mechanical and hydraulic components and their functions within the system, the artificial feel system, and the trim systems. The last part of the chapter will be devoted to a discussion of operational checkout, maintenance, and rigging of the elevon control system.

The pitch and yaw damper system and the automatic flight control system, as applied to elevon control, are discussed only briefly in this chapter. The systems are discussed fully in later chapters.

DESCRIPTION OF THE ELEVON CONTROL SYSTEM.

In figure 2-1 which shows the elevon control system, only the right elevon control is shown since the right and left elevons move in an identical manner. In later models of the F-102A, the servo actuator shutoff valve, the servo actuator, the lockout valve (and its solenoid), and the control valve are incorporated in a single hydraulic package (HEP) valve for each elevon. The hydraulic elevon package is described in subsequent paragraphs of this chapter.

The design of the F-102A is not conventional in that there are no horizontal tail surfaces. Elevators are normally attached to a horizontal stabilizer to provide pitch control on the conventional airplane. However, on the F-102A, control surfaces called "elevons" (one on the left and one on the right side) are attached to the rear structure of the delta wings to provide elevator control. The elevons combine elevator-aileron functions in the one set of control surfaces. That is, fore and aft movement of the control stick moves the elevons up or down together to provide elevator action

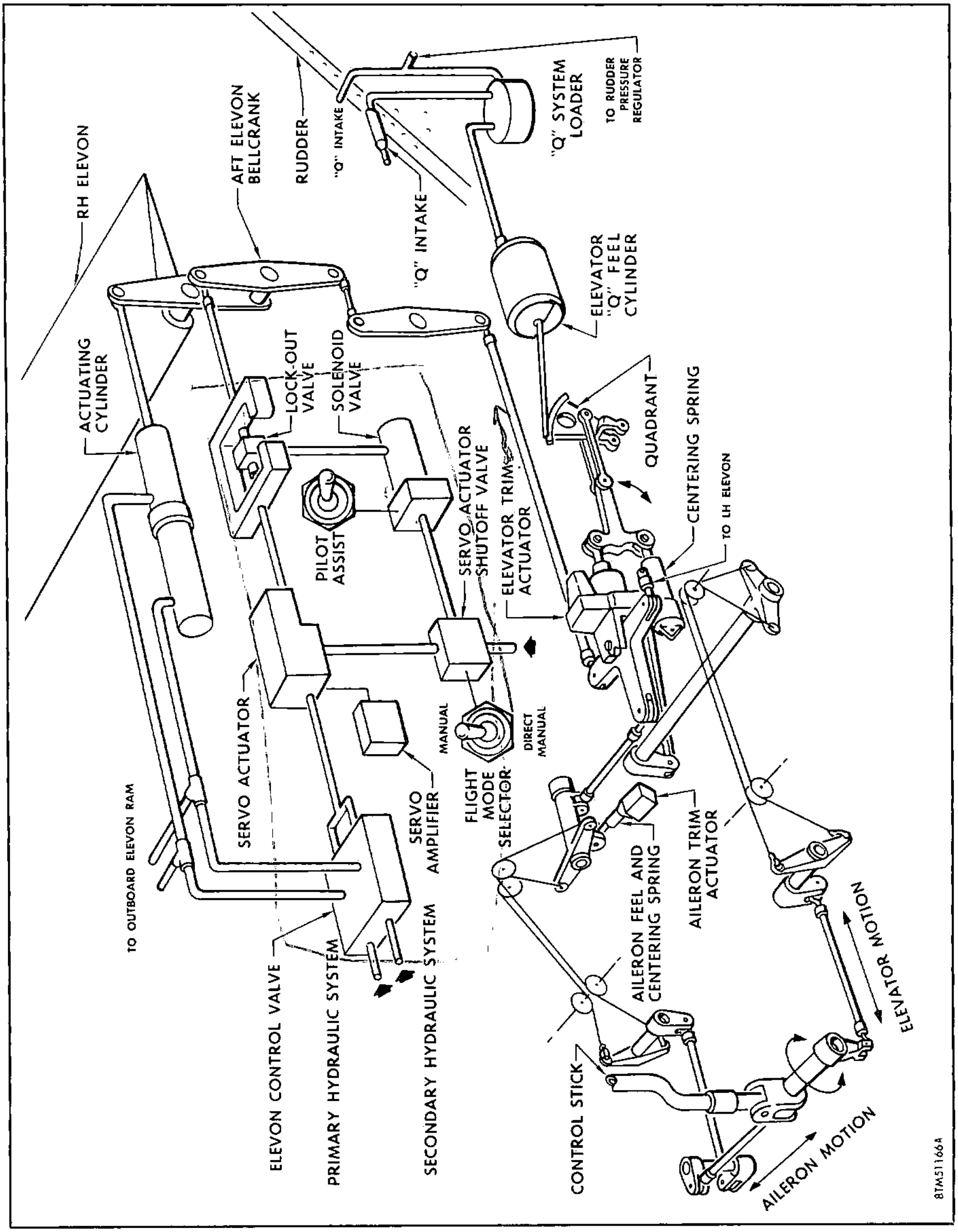

Figure 2-1. Elevon Control System Schematic

while movement of the stick from side to side moves one elevon up and the other down to provide aileron action for roll or bank control. A combination of the two movements, along with coordinated rudder movement, provides for normal climbing or descending turns. The word "elevon," obviously, is a combination of the words elevator and aileron broken down to form the one word to describe this control surface.

The mixer assembly, shown on figure 2-1, is a mechanical device which functions to supply elevator or aileron action to the elevons as determined by the pilot's control stick movements. As you can see, the control stick is connected to the mixer assembly through mechanical linkage. Movement of the mixer assembly is then transmitted to the left and right elevon hydraulic components through the left and right elevon control linkage.

The flight control system on the F-102A is a full hydraulic power system in which all control surface motion—movement of the elevons and rudder—is accomplished by hydraulic actuating cylinders. Full hydraulic power is used because of the high forces imposed on the elevons and rudder at high speeds. It would be extremely difficult for the pilot to manually move the control surfaces due to these high forces. Hydraulic power to move the control surfaces is supplied by the primary and secondary hydraulic systems. Each actuating cylinder has two hydraulic chambers with two pistons operating on one ram shaft so that either one or both of the two separate hydraulic systems can operate the flight control surfaces. In normal operation both hydraulic systems are used at the same time, the two systems provide a safety measure in the event one of the hydraulic systems should fail.

Two hydraulic actuating cylinders are provided for each elevon; one is connected to the upper inboard elevon horn on each elevon and the other is connected to the lower outboard elevon horn on each elevon. The cylinders on each elevon are interconnected so that when positioning the elevon one cylinder is etxending while the other is retracting. Mechanical linkage set in motion by the pilot's movement of the control stick, or through action of the serve actuator on signals from the damper system, opens hydraulic control valves. The control valves then meter hydraulic fluid to the actuating cylinders to move the elevons. A mechanical followup or feedback linkage then reverses the initial motion and returns the control valves to neutral. The elevons remain in the position established by the pilot's control motion until the pilot again moves the control stick.

Since the hydraulic action of the elevon control system is not reflected back to the mechancal linkage, airloads on the elevon control surfaces are not transmitted back to the pilot. To overcome this condition, an artificial feel system is incorporated in the mechanical system linkage. The feel system furnishes the pilot with feel forces at the control stick, and prevents him from overloading the elevon surfaces. This system will be discussed in more detail later in the chapter.

The control surfaces on the F-102A airplane do not have conventional trim tabs. Trimming of the airplane is accomplished by electrical trim actuators that move the entire elevon and rudder control surfaces for trim action. A five-position toggle switch on the control stick routes 28-volt, d-c power to the elevon trim actuators to move the elevons. Aileron trim is applied to move the elevons differentially for trim about the roll axis. Normal elevator trim is provided by the trim servo system; in addition, this system automatically changes elevator trim to compensate for speed and altitude changes and also for speed brake extension. The engagement of the elevator trim servo system is controlled by a solenoid-held switch on the utility switch panel in the cockpit. The elevon trim system will also be discussed in more detail later in the chapter.

MODES OF FLIGHT.

There are three modes or ways available to the pilot for controlling the flight attitude of the airplane. These are the Direct Manual, Manual, and Automatic Flight Control modes which are discussed separately below.

Direct Manual.

When the flight mode switch is placed in the DIRECT MANUAL position, the pilot has direct control of the airplane and only his control movements result in a change of course or attitude. The Direct Manual mode of control is normally used for takoffs and landings, and any time the pilot desires to have only control stick, rudder pedal, and trim switch movements determine displacement of the control surfaces.

Manual.

The Manual mode is that method of flight control wherein the pilot's control movements are dampened to some degree by counteracting signals from the damping system through the servo actuators. The servo actuators receive signals from the damper system and convert them to control motion. Placing the flight mode switch in the MANUAL position completes a circuit that sends 28-volt, d-c current to energize the elevon and rudder actuator shutoff valves. When these valves are energized, they route hydraulic pressure to the elevon and rudder servo actuators and also to each elevon lockout valve. The lockout valve is not actuated in the Manual mode. The elevon and rudder servo actuators are connected by linkage to their respective hydraulic control valves to superimpose damping impulses on the control surfaces. The manual mode of control is selected by the pilot to provide a stable armament platform, or when the airspeed is above Mach 0.75.

Automatic Flight Control.

In the Automatic Flight Control mode, the automatic flight control system of the MG-10 Aircraft and Weapon Control System governs the flight control system. Initially, the pitch and yaw damper systems must be engaged. Closing the AFCS-PILOT ASSIST circuit breaker and the AFCS ENGAGE switch on control box "105" of the MG-10 Automatic Flight Control System initiates automatic flight control. When the AFCS-PILOT ASSIST circuit breaker and the AFCS ENGAGE switch are closed, an electrical signal is sent to the lockout valve solenoid, which allows hydraulic fluid to extend two plungers that lock the control shaft from the aft elevon bell crank to the airframe. This action locks out pilot input control so that motion is not transmitted from the control stick to the elevon control valve. In lockng out pilot stick control, the mechanical followup or feedback loop is locked out. The mechanical feedback is replaced by electrical feedback.

AFCS signals are sent to the elevon servo actuators by means of the damper systems. The servo actuators move the valve selector spools in the elevon control valves, thereby causing hydraulic fluid to be ported to the elevon actuating cylinders. The elevons are thus displaced to the extent called for by the AFCS signals. As either elevon actuating cylinder is displaced, it also moves the wiper arm of a control surface feedback potentiometer. The feedback potentiometer sends a feedback signal to the damper systems where it counteracts the control signal. As a result, the damper system signal is decreased. When it has decayed, it will have stopped elevon movement at the desired elevon displacement. In later models of the F-102A, the servo actuator, the elevon control valve, the lockout valve, and the servo actuator shutoff valve are incorporated in a hydraulic elevon package (HEP) valve. The HEP valve is described in subsequent paragraphs of this chapter.

When flight control is switched to the Automatic Flight Control mode, the airplane will remain in the attitude established by the pilot at the time of engagement, except when a bank angle of less than 5 degrees exists. In this case, heading control will engage and the plane will roll out as required to maintain heading. When a bank angle greater than 5 degrees exists at time of engagement, heading control will not engage, and roll control will function to maintain the bank angle.

The rudder hydraulic system does not incorporate a lockout valve as does the elevon hydraulic system. Therefore, the rudder is always at the command of the pilot regardless of the mode of flight. The rudder servo actuator receives signals from the damping system and initiates appropriate rudder movement to coordinate a turn. The essential components for turn coordination are the roll-rate gyro, a high-pass network and filter, the aileron position potentiometer, and the airspeed compensator. These components originate the signals to the damper system for turn coordination.

Operation of the elevons in the three modes of flight will be discussed in detail later in the chapter.

ELEVON MECHANICAL SYSTEM.

The illustrations accompanying this discussion show the arrangement and identify the mechanical system components. For ease of discussion and understanding, the system is divided into four groups. The groups are composed of the forward linkage, which includes the control stick and the forward aileron and elevator control linkage; the mixer assembly and linkage; the aft linkage, including the linkage from the aft elevon bell crank to the hydraulic control valve as well as the actuating cylinder and its attachment to the elevon; and the control cables which transmit stick movement to the elevon control linkage and the mixer assembly.

FORWARD LINKAGE.

A conventional control stick, torque tubes, control rods, bell cranks, and support assemblies comprise the forward linkage, as shown on figure 2-2. The grip on the control stick incorporates a five-position, spring-loaded toggle switch for aileron and elevator trim and also a button-type switch for disengaging the Automatic Flight Control System (AFCS). The linkage on the right side of the stick transmits aileron movement as the pilot moves the stick from side to side. The linkage on the left side of the stick transmits elevator movement as the pilot moves the stick fore and aft.

The control stick assembly is attached to and supported by the elevator torque tube. Four roll pins attach the base of the stick assembly to the torque tube. The stick base in turn is supported by a single bolt through a support fitting attached to the airplane structure at its inboard end and upon which the stick base is free to rotate. The outboard end of the torque tube is attached to and supported by a bell crank and box assembly with a single bolt upon which the bell crank is also free to rotate. Both the inboard and outboard support bolts are screwed into nut-plates installed in the end of the bell crank and stick base. Fore and aft movement of the control stick rotates the torque tube which in turn transmits this movement to the elevator control linkage by means of the bell crank attached to the torque tube at its outboard end.

Note how the control stick is attached to the stick assembly base by one bolt and upon which it is free to pivot for side-to-side motion to provide aileron action. A push-pull control rod is attached to the lower end of the stick at the rod inboard end and to a bell crank and torque tube assembly at the rod outboard end.

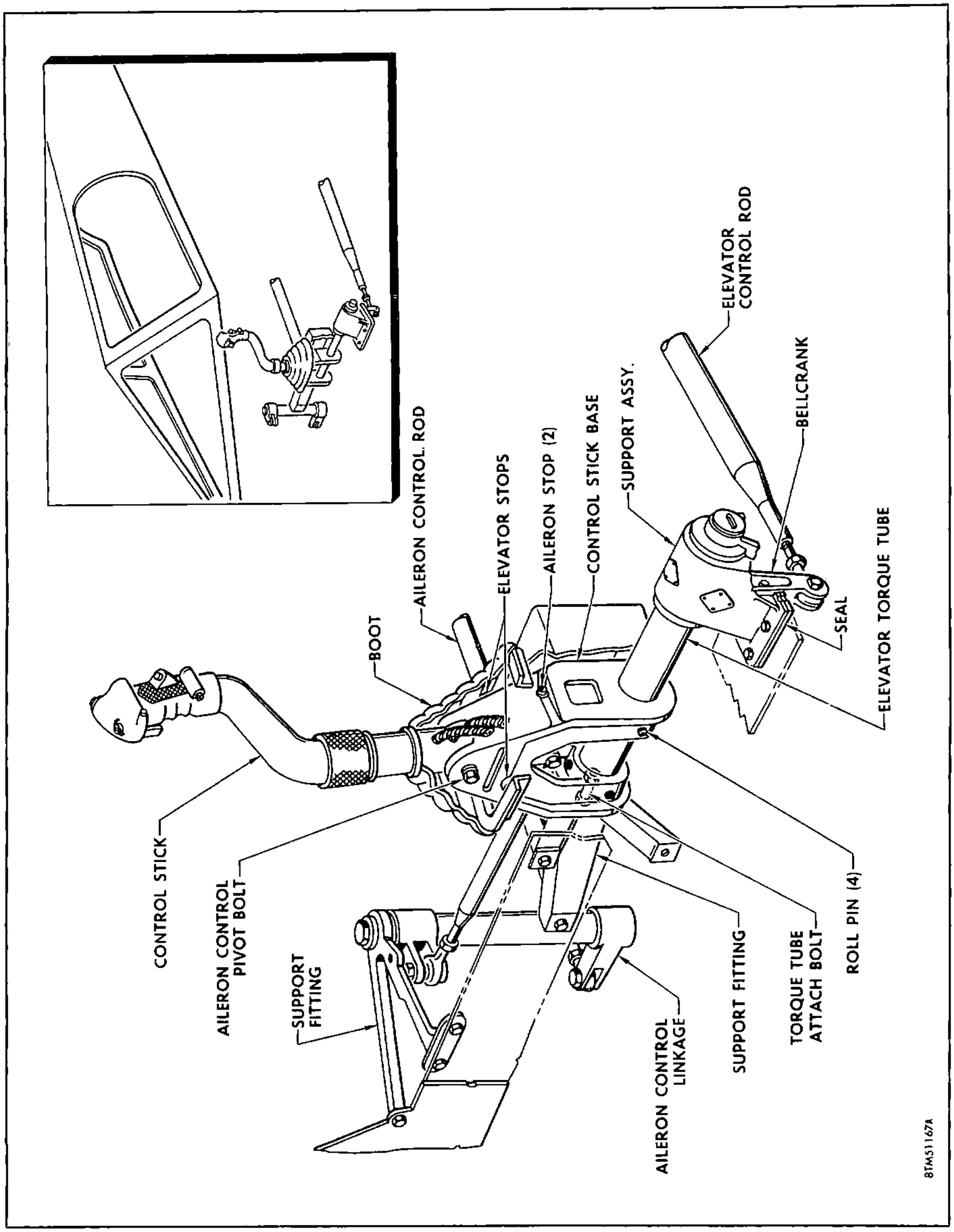

Figure 2-2. Elevon Control Stick and Forward Linkage

Side-to-side movement of the stick is transmitted through the push-pull tube to the upper bell crank to rotate the torque tube and the lower bell crank to provide aileron control movement to the mixer assembly.

Adjustable stop bolts, which limit side-to-side and fore and aft travel of the stick assembly, are provided as shown on figure 2-2. Refer to your F-102A handbook, T.O. 1F-102A-2-7, for the proper setting of these stops. Remember that all control stick and linkage attach bolts must be tightened to specified torque values when these components are being installed. Refer to the above mentioned handbook for these torque values.

MIXER ASSEMBLY.

Figure 2-3 shows the mixer assembly and the connecting linkage. The assembly consists of a cradle that moves fore and aft for elevator action and an elevon mixer bell crank that pivots from side to side for aileron action. The mixer bell crank is attached to the cradle by a single bolt around which it is free to pivot. Note that the forward end of the cradle assembly is attached to and supported by the elevator bell crank, while the aft end is supported by the elevator trim actuator shaft which is attached to the upper arm of the feel system bell crank. The feel system bell crank pivots on a single bolt through a support assembly attached to the airplane structure. Note that the centering spring is also attached to the lower arm of the feel system bell crank. The elevator trim actuator body is attached to the aft end of the mixer assembly cradle with four bolts; therefore, the shaft end of the trim actuator provides the aft support for the mixer assembly. This arrangement allows elevator trim and feel forces to be introduced into the mechanical system. Pilot's stick motion is transmitted then through the forward linkage to the mixer assembly which moves the mixer cradle in a fore or aft direction for elevator action, or pivots the mixer bell crank for aileron action. Either of these movements or combination of movements results in moving the elevon control rods in a fore or aft direction in proportionate amounts to actuate the aft control linkage. The aft control linkage then opens the hydraulic control valves, which meter hydraulic fluid to the actuators to move the elevons.

Adjustable stop bolts, which limit cradle assembly travel, are provided as shown in figure 2-3. Refer to your F-102A Maintenance Handbook for the proper setting of these stops.

ELEVON AFT LINKAGE.

Figure 2-4 shows the arrangement and identifies the components in the aft elevon control linkage located on the aft left and right sides of the fuselage. Only the installation for the right elevon is shown, since the installation on the left side is identical. Note that the linkage from the aft elevon bell crank to the hydraulic control valve includes the servo actuator and a yoke as part of the linkage. In the Direct Manual mode of control, the servo actuator piston shaft is locked in a rigid position by its centering springs and serves as only a part of the rigid link to the control valve. In the Manual and Automatic Flight Control modes of control, however, the shaft extends or retracts in response to electrical signals from the damper system to open the hydraulic control valve.

The yoke in the linkage functions to lock out pilot-initiated control movements when the pilot selects the Automatic Flight Control mode of control; the lockout valve, which is attached to the airplane structure, is energized and extends two plungers against the inside ends of the yoke, thereby locking the mechanical linkage to the control valve. The servo actuator then responds to signals from the AFCS components through the damper system to control elevon movement.

The control valve is opened either as a result of pilot-initiated control stick movements through the mechanical linkage or through the action of the elevon servo actuator which responds to signals from the damper system. When the control valve is opened, hydraulic fluid is metered to the inboard and outboard elevon actuating cylinders to move the elevons. The follow-up mechanism accordingly closes the control valve when the elevons reach a position corresponding to the pilot's stick position when in the Direct Manual and Manual mode of control. An electrical feedback system closes the valve during the AFCS mode of control. The operation of the elevons in the three modes of control are discussed in detail on the following pages.

In figure 2-4 note how the elevon actuating cylinder is attached to the inboard elevon horn; the horn pivots on the bolt shown just above the aft elevon bell crank. The potentiometer attached to the actuator is used only in the Automatic Flight Control mode and its use described later in this chapter. The hydraulic components of the aft control linkage, including the servo actuator, the elevon actuator, the control valve, the lockout valve, and the HEP valve (which combines these components), as well as maintenance problems concerned with them, are also discussed later in the chapter.

CONTROL CABLES.

Figure 2-5 shows how the elevator and aileron control cables are routed through pulleys to connect the forward linkage from the control stick to the linkage at the mixer assembly. The elevator linkage and control cables are on the left side (looking forward) and the aileron linkage and control cables are on the right side (looking forward). Note that each of the four cable assemblies consist of two cables joined together

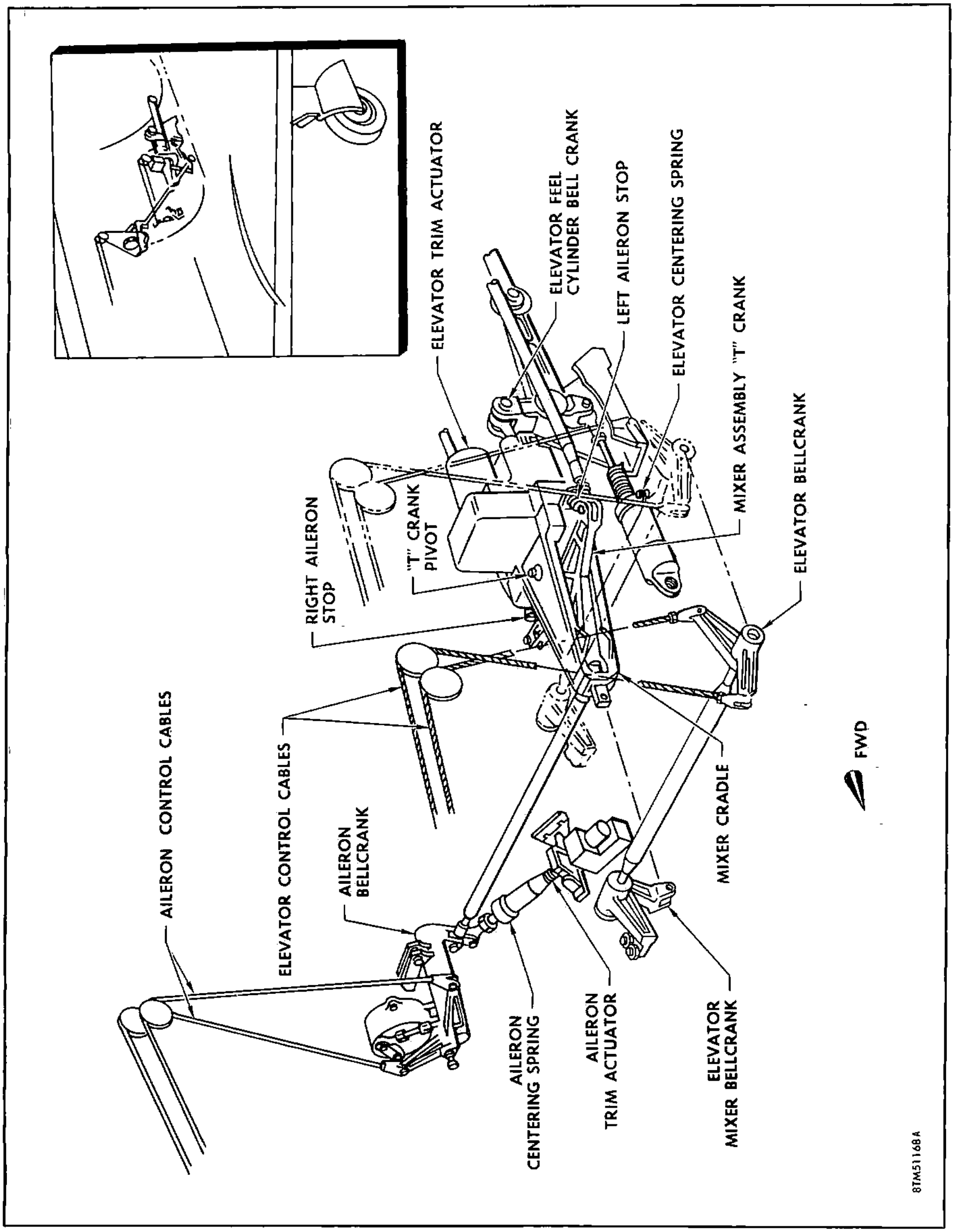

Figure 2-3. Mixer Assembly and Control Linkage

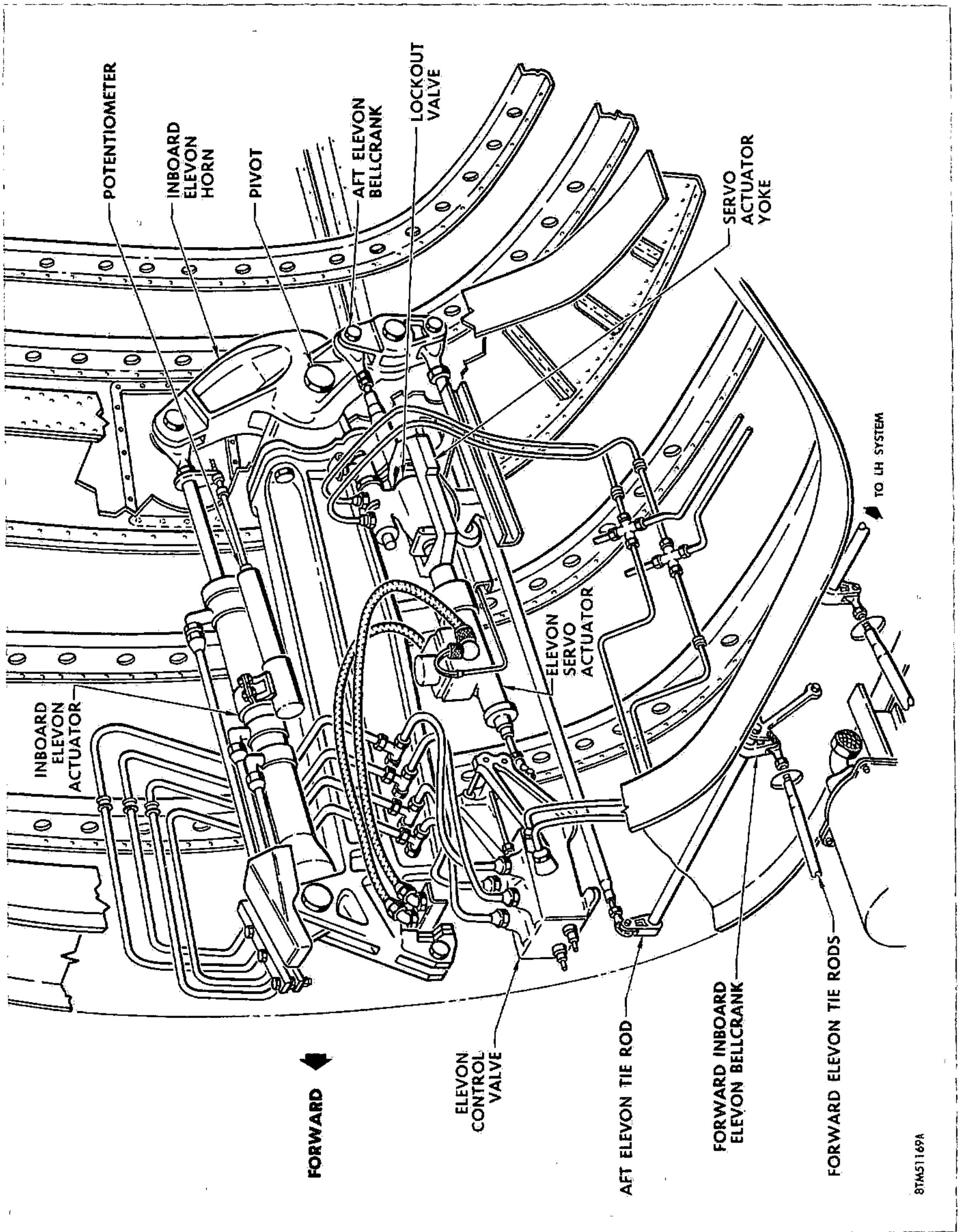

Figure 2-4. Elevon Aft Mechanical Linkage

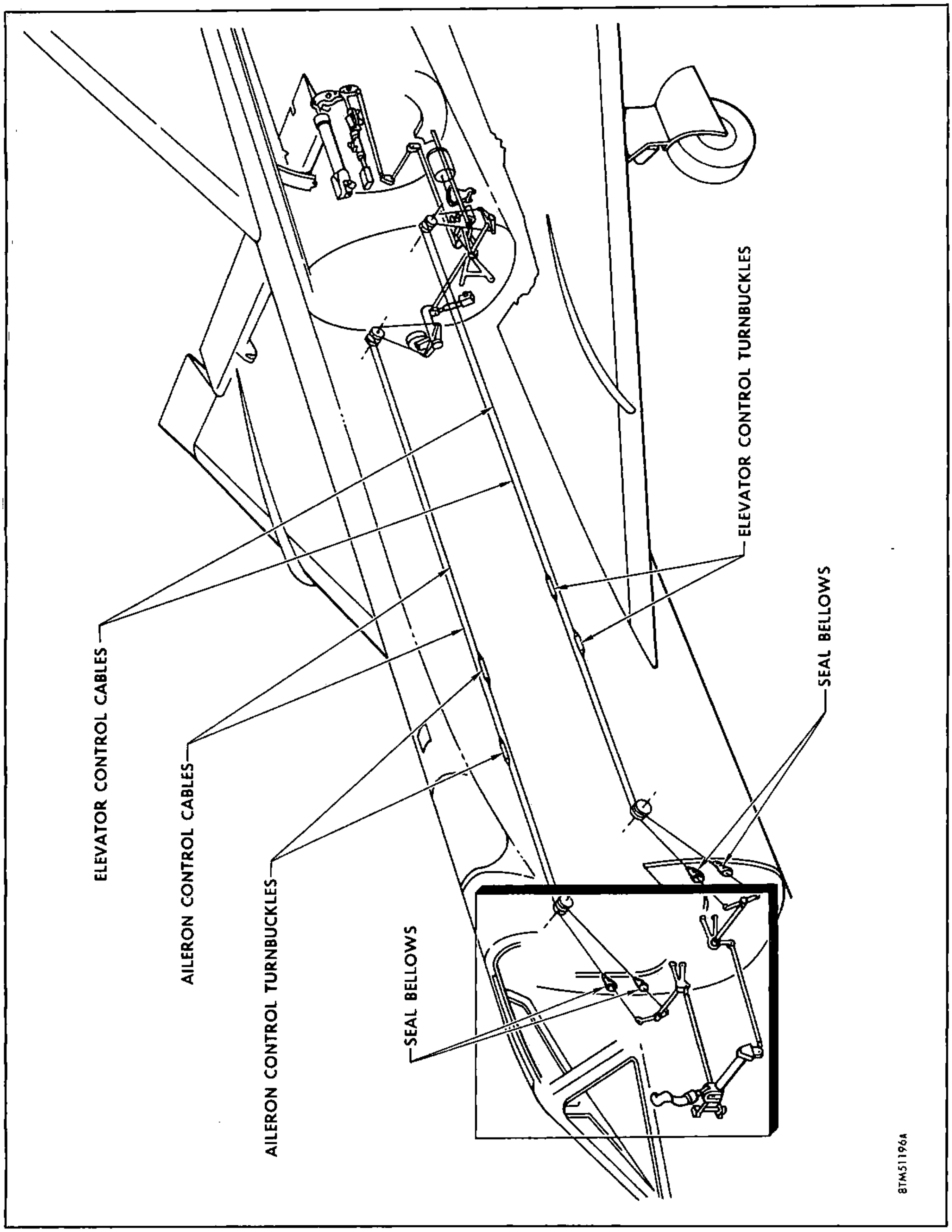

Figure 2-5. Control Cables and Linkage

with turnbuckles. The terminal ends of the cables are attached to the forward and aft bell cranks with bolts and self-locking nuts.

Cable Tension.

As you know, metals contract and expand when exposed to changes in temperature. For this reason control cable tension must be set at a value (in pounds) established for the degree of temperature existing at the time the cables are being installed or reinstalled in the airplane. The diameter of the cable must also be considered in establishing this value. These values have been predetermined and can be found by referring to the cable tension chart in the applicable F-102A maintenance handbook, T.O. 1F-102A-2-7. The above mentioned handbook also outlines in a step-by-step fashion a procedure for rigging the control cables.

Lubrication Of Control Cables.

A protective coating of corrosion-resistant compound for the full length of the cables and also lubrication of cables in areas of pulleys is required at periodic intervals. Again refer to the above mentioned handbook for the time intervals and the type of compound and lubricant used on the cables.

ELEVONS.

Each elevon is attached to the trailing edge structure of the delta wings by seven hinge bolts. Each elevon is actuated by two hydraulic actuators, one on the inboard end (inside the fuselage), and the other near the outboard end on the wing lower surface. The inboard elevon horn assemblies are attached to the inboard ends of the elevons with eight special bolts; the horns extend into the fuselage where they are connected to the inboard actuating cylinder rod ends. The horn cranks are also attached to the tubing and actuator support fittings by a single bolt around which the crank is free to pivot.

The elevons are stressed-skin, aluminum structures with a front spar and full and half ribs, and are designed to flex at approximately midspan to conform to the wing contour as it flexes under airloads.

Access doors are provided at the elevon attach fittings and on the left and right sides of the fuselage at the inboard elevon actuators. The outboard elevon actuators are enclosed with a removable fairing. Removal and installation procedures of the elevons as well as maintenance and lubrication requirements are discussed at the end of this chapter.

ELEVON CONTROL SYSTEM OPERATION.

As you recall, there are three modes or ways of controlling the F-102A elevons in flight. These are the Direct Manual, Manual, and Automatic Flight Control modes. As stated before, the elevon control system receives hydraulic power from the primary and secondary hydraulic systems to supply power to the hydraulic control valves. In the Direct Manual mode of control, the pilot's control stick motion is transferred through mechanical linkage to open the elevon control valves. The control valves then port hydraulic fluid to the elevon actuating cylinders to move the elevons in the amount and direction as indicated by the pilot's stick movement. The mechanical followup linkage then reverses the initial motion of the control valve linkage and returns the control valve spool to neutral. This stops the elevons in a position corresponding to the pilot's stick position. The elevons will remain in this position until the pilot again moves the control stick. The elevons will move only when the stick moves and in the direction determined by the stick position. If the pilot should release the stick, the centering springs will return the system to trim position.

In the Manual and Automatic Flight Control modes of operation, the control linkage moves the elevons in response to signals from the damper system. We will now discuss in detail the elevon control system operation in each of the three modes of control.

OPERATION IN THE DIRECT MANUAL MODE.

As you learned earlier, the Direct Manual mode is the method of flight control wherein the pilot has direct control of the airplane and only his control motions result in a change of course or attitude. When the control stick is moved fore and aft, it produces *elevator motion.* When the stick is moved to either side, it produces *aileron motion.* Both elevator and aileron motion may also be effected simultaneously for climbing or descending turns. Movement of the control stick is transmitted through mechanical linkage to the mixer assembly shown on figure 2-6. The function of this mixer assembly is to transmit linkage movement in proportionate amounts to the bell crank at each elevon. When the elevon bell crank rotates in either direction, it displaces the elevon control valve in a corresponding direction. The mixer assembly crank moves the elevon control rods in fore and aft directions only, in response to either aileron or elevator stick controlled movements.

Fore and aft movement of the stick results in a fore and aft movement of the mixer assembly and further results in equal movements being transmitted through the linkages to the elevon control valves. The elevons then move equally in the same direction to produce elevator action.

Differential motion of the elevons, that is elevons moving in opposite directions for aileron control, is initiated by moving the control stick to either side. Side motion of the stick is then transmitted by control

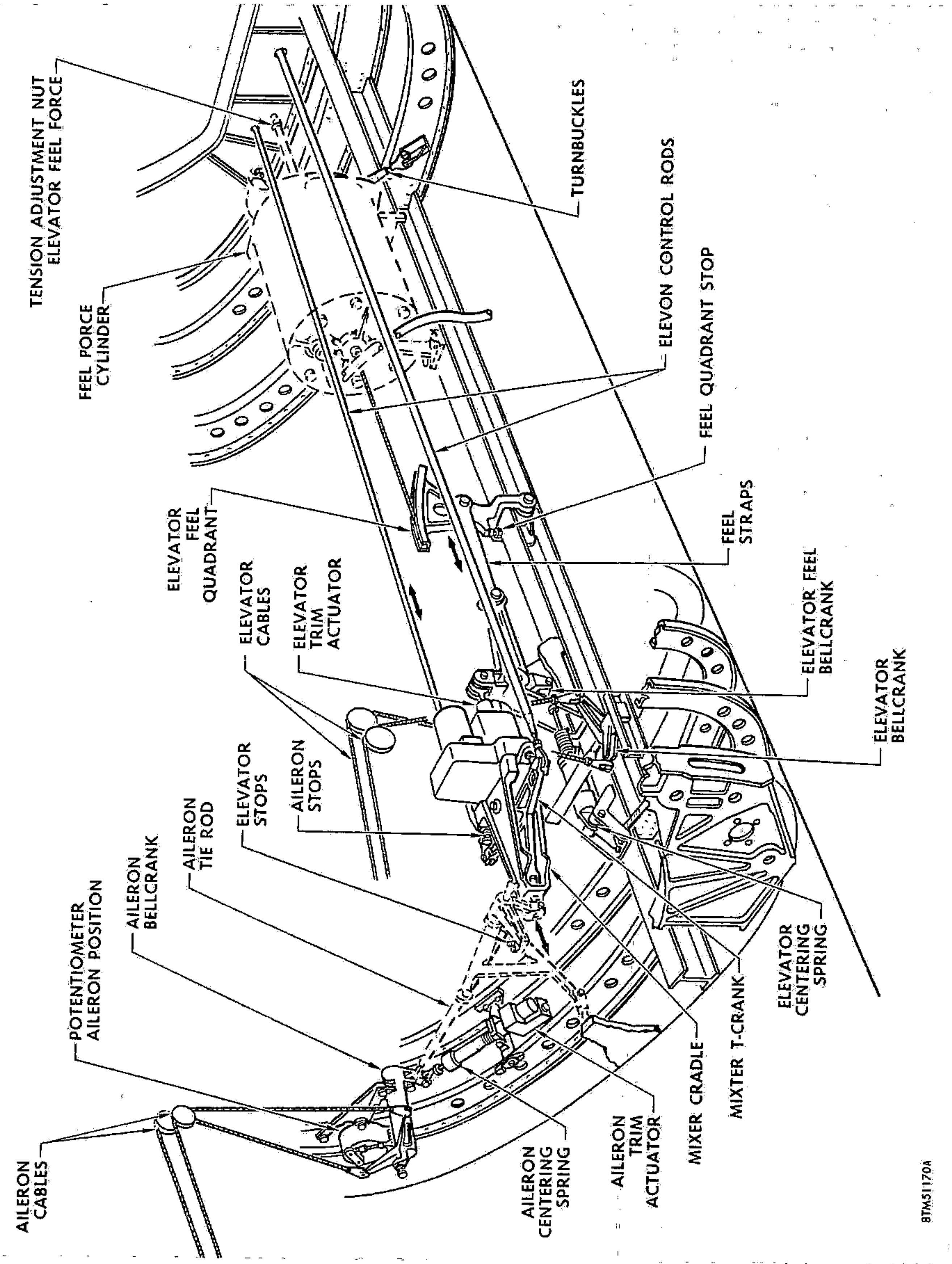

Figure 2-6. Elevator-Aileron Mixer Assembly Action

rods and bell cranks to the right hand control cable assembly. The cable assembly, in turn, actuates the bell crank opposite the mixer assembly. Now note that the bell crank is connected to the forward leg of the mixer assembly by a control rod and transmit the pilot's stick motion to rotate the mixer bell crank assembly for aileron action. The mixer bell crank is attached to the cradle assembly with a single bolt around which it is free to rotate. Side motion of the forward end of the leg results in opposite forward and aft movement of the cross member ends of the mixer bell crank. Since the ends of the cross members are attached to the elevon control rods, one rod will therefore move forward while the other rod will move aft. This will result in the mechanical linkage moving one control valve spool forward and the other will, at the same time, be moved aft a like amount. The elevons will then move in opposite directions for aileron action. The design of the mixer assembly is such that it can provide elevator and aileron movements in appropriate amounts for climbing or descending turns.

In figure 2-7 you can see the elevon control assembly on the right side of the fuselage, inboard of the elevon. (The control assembly on the left side is identical.) This view shows how the mechanical linkage actuates the hydraulic components to move the elevon in the Direct Manual mode of operation. Note that the elevon control rod from the mixer assembly is connected to the lower end of the aft elevon bell crank. The upper shaft on the elevon bell crank is connected to the control valve; also, the shaft to the control valve forms a yoke around the lockout valve. The lockout valve is attached to the airplane structure and the yoke permits the shaft to move forward and aft without coming in contact with the valve. The lockout valve is a component of the Automatic Flight Control mode of control and functions to lock the shaft and "lock out" the pilot's control movements to the elevon during automatic control. This mode of control will be discussed later.

The servo actuator is also a component connected to the shaft from the elevon bell crank to the control valve. The servo actuator, in the Manual and Automatic Flight Control modes of control, moves, the elevon control valves for elevon damping and automatic control. The function of the servo actuator will also be discussed later. In the Direct Manual mode of control, the servo actuator merely acts as a rigid link between the elevon bell crank and the control valve. Since the servo actuator and the lockout valve do not function during the Direct Manual mode we can consider the rod between the bell crank and the control valve as being a solid shaft, directly linking the bell crank and the control valve.

As we discussed previously, pilot movement of the control stick is transferred through linkage to the mixer assembly for elevator, aileron, or coordinated elevator and aileron movement for descending or climbing turns. The mixer assembly actuates the elevon control rods to transmit the appropriate movement to the elevon bell crank. The bell crank in turn actuates the shaft to the control valve which opens and ports hydraulic fluid to the elevon actuating cylinders. The actuating cylinders then move the elevons to a position corresponding to the pilot's stick position.

Movement of the shaft toward or away from the control valve to which the shaft is attached moves the spools of the control valves. As the spools move, they port the pressurized fluid to the actuating cylinders of the elevon and also open ports to the return side of the power supply systems. The direction away from neutral that the spools move determines the direction in which the elevon actuating cylinders move the elevon.

The two illustrations, figures 2-8 and 2-9, show the action of the mechanical linkage in directing the control valve to move from neutral and then return to neutral. Figure 2-8 shows the direction the bell crank moves when responding to a push from the mixer assembly. The arrows indicate the direction in which the crank moves and how it rotates around the pivot point. (Note that the pivot point of the crank is attached to the elevon horn.) The rotation of the shaft around its pivot point moves the shaft to displace the control valve spools. The spools then admit hydraulic fluid to the elevon actuating cylinders. The inboard actuating cylinder (attached to the upper elevon horn) retracts its cylinder to move the elevon UP. (The outboard actuating cylinder, attached to the lower elevon horn, extends at the same time for elevon UP movement.)

Figure 2-9 shows how the control spool is returned to its neutral position. The bottom of the bell crank is now being held in a rigid position by the pilot-held control stick and serves as the pivot point. As the top of the elevon horn moves forward, the lower part of the horn to which the bell crank is attached, moves aft. Since the lower end of the bell crank is being held in a rigid position, the upper portion is moved aft with the horn to pull the control valve spool back to its neutral position. As a result of this mechanical followup linkage action, the control valve spools automatically arrive at their neutral position at the same time the elevons arrive at the position determined by the pilot's stick movement.

Action of the elevon cylinders reverses the initial ment of the control valve by the pilot through this followup action. The elevons remain in the position determined by the pilot-held control stick until the pilot again moves the stick, at which time the cycle of motion and stabilization at another position is again accomplished. When the control valve spools return to neutral, they port an equal, though restricted, flow

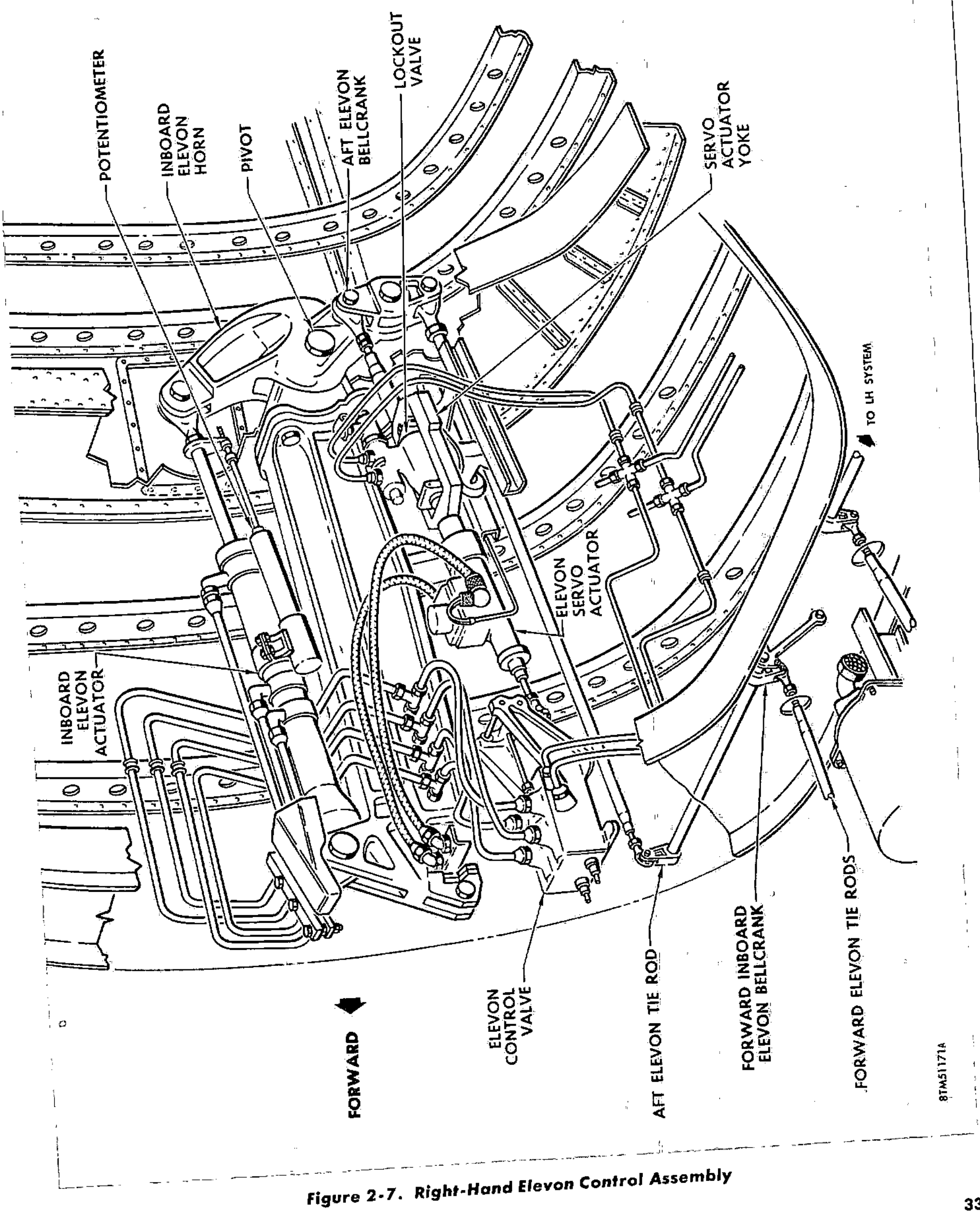

Figure 2-7. Right-Hand Elevon Control Assembly

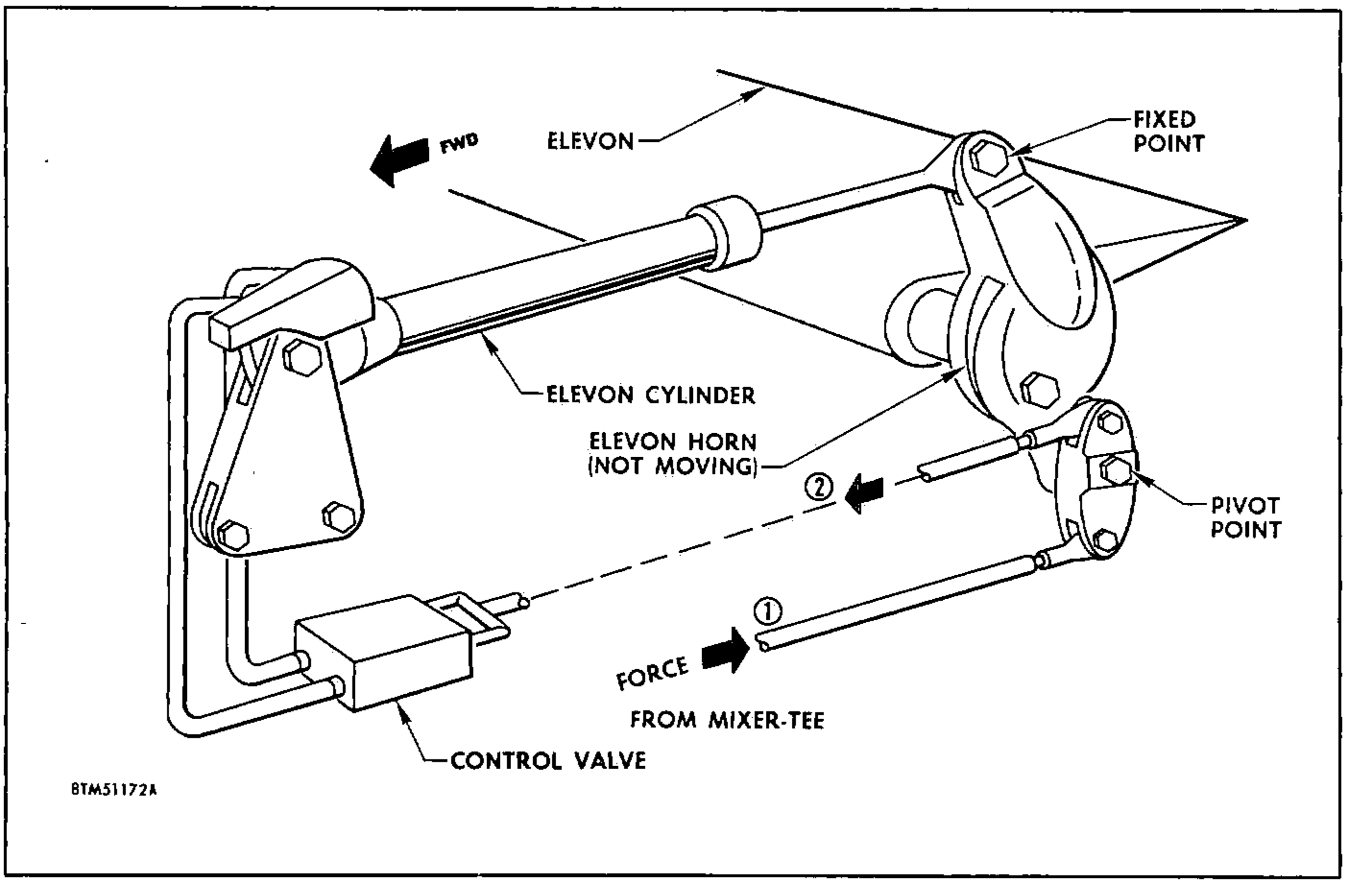

Figure 2-8. Elevon Mechanical Action, Initial Motion

to both sides of the cylinder pistons, thus locking both pistons in the actuating cylinders so that they are immovable and hold the control surface rigidly in position. Trim and artificial feel forces are introduced into the control system through the mechanical linkage. These systems are discussed later in this chapter.

OPERATION IN THE MANUAL MODE.

As you will recall, the Manual mode of control damps pilot-initiated movement of the elevons to some degree. The servo actuator responds to signals from the damping system to correct for slight pitch variations to maintain stability. In Chapter IV, you will learn that placing the flight mode selector switch in the MANUAL position completes a circuit which sends 28-volt, d-c power to energize the unlock solenoid in the elevon servo actuator shutoff valve; the shutoff valve then opens and supplies hydraulic pressure to the servo actuator. The servo actuator is now hydraulically energized and is free to move in response to signals from the damping system. Earlier we discussed how, in the Direct Manual mode of control, the servo actuator merely acts as a rigid link in the control shaft from the elevon bell crank to the control valve (shown on figure 2-7). Now the servo actuator partially displaces the control valve and thus moves the elevons for any stabilization corrections that may be required.

When hydraulic power is supplied to the servo actuator, the actuator begins to respond to the servo amplifier signals from the damping system. The actuator piston then extends or retracts, which ever is appropriate. Since the actuator piston shaft is a part of the link between the elevon bell crank and the control valve, this movement shortens or lengthens the link to position the spool in the control valve. This action then results in appropriate elevon movement to damp pilot-initiated movement of the elevons.

In other words, the pilot flies the airplane in the Manual mode as he does in the Direct Manual mode; the only difference is that in the Manual mode minor stabilization corrections are made automatically through the servo actuator on signals from the damping system. If the pilot were required to make manual control corrections at high speed, he would be apt to overcontrol too much in the opposite direction and therefore increase an unstabilized condition. The damping system anticipates needed corrections before the pilot is aware corrections are necessary.

The damper system servo amplifiers continually supply signals to the torque motors in the servo actuators, but until hydraulic power is supplied from the shutoff valve, the servo actuator does not respond. The

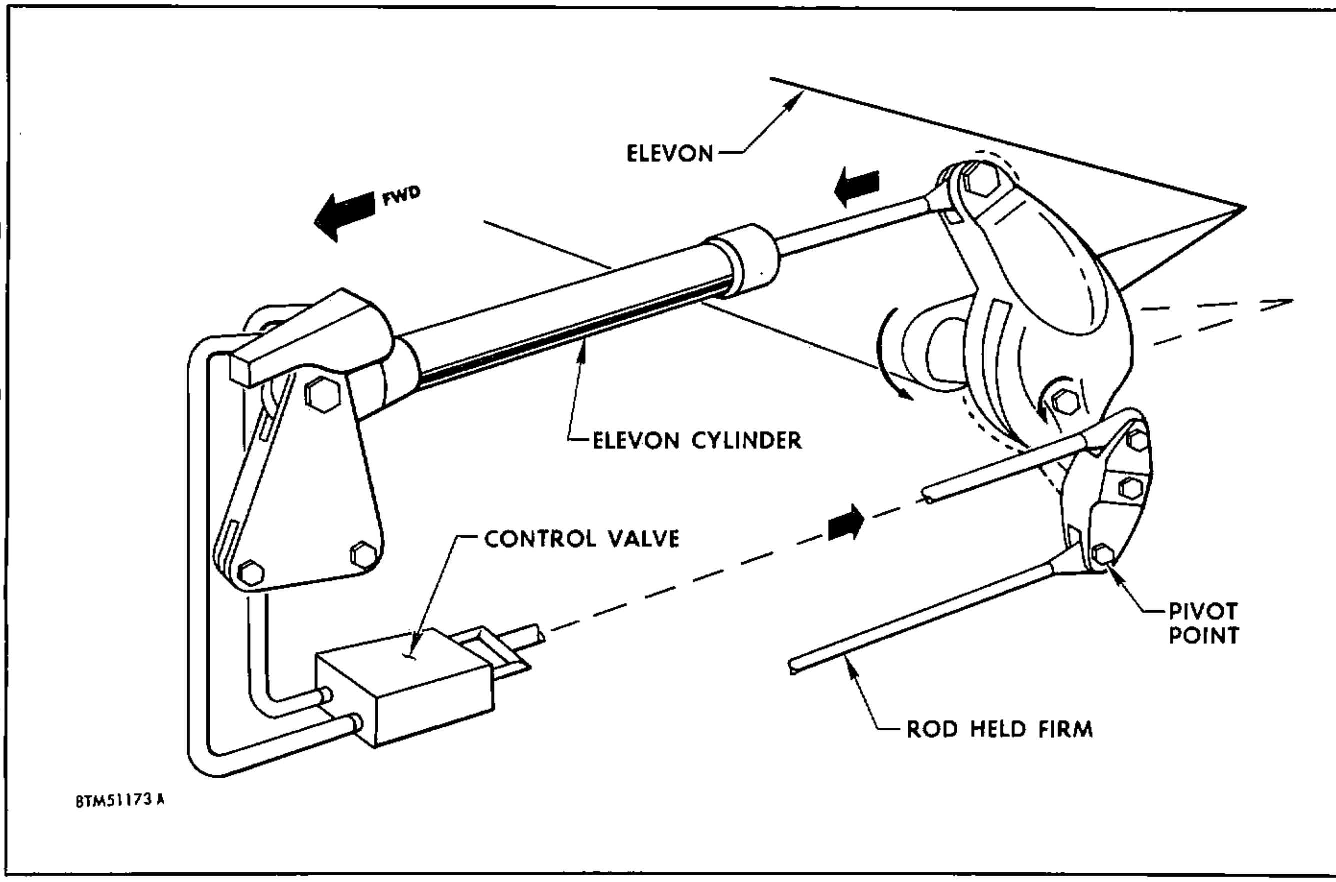

Figure 2-9. Elevon Mechanical Action, Follow-Up Motion

amplifier is ON at all times on a standby basis when electrical power is available and the circuit breakers are engaged.

The servo actuator receives power from the secondary hydraulic system. Should the secondary hydraulic system fail, operation in the Manual and Automatic Flight Control modes is not possible. The servo actuator piston is centered by its centering springs in event of secondary hydraulic system failure and the actuator again becomes a rigid link in the system. The airplane may still be controlled by the primary hydraulic system through pilot control in the DIRECT MANUAL mode of operation.

OPERATION IN THE AUTOMATIC FLIGHT CONTROL MODE.

When operating in the Automatic Flight Control mode, the pilot can devote his attention to the operation of the fire control system. We will not attempt to explain the Automatic Flight Control System in detail, because an entire chapter is devoted to it later in this supplement. However, we will concern ourselves with the mechanical and hydraulic functions of the components as they respond to electrical signals from the AFCS.

Like the Manual mode of operation, the servo actuator is the means by which the elevons are moved in the Automatic Flight Control mode of operation. Unlike the Manual mode, however, the pilot is "locked out" from direct control of the elevons in this mode.

As you have learned, a lockout valve is attached to the airplane structure in line with the linkage to the hydraulic control valve. The lockout valve is surrounded by a yoke which is a part of the control linkage between the elevon bell crank and the hydraulic control valve as shown on figure 2-7. In the Direct Manual and Manual modes of operation, the yoke moves with the control linkage since it is a part of the linkage. In the Automatic Flight Control mode however, the yoke is the means by which direct pilot control of the elevons is locked out. When the pilot selects the Automatic Flight Control mode of control, he engages the AFCS ENGAGE switch to route 28-volt, d-c power to energize a solenoid on the lockout valve. Opening of the solenoid valve permits hydraulic pressure to actuate the lockout valve and extend two plungers against the inside ends of the yoke. Since the body of the valve is attached to the airplane structure, the control shaft can no longer move and the pilot is locked out from direct stick control. The pilot can,

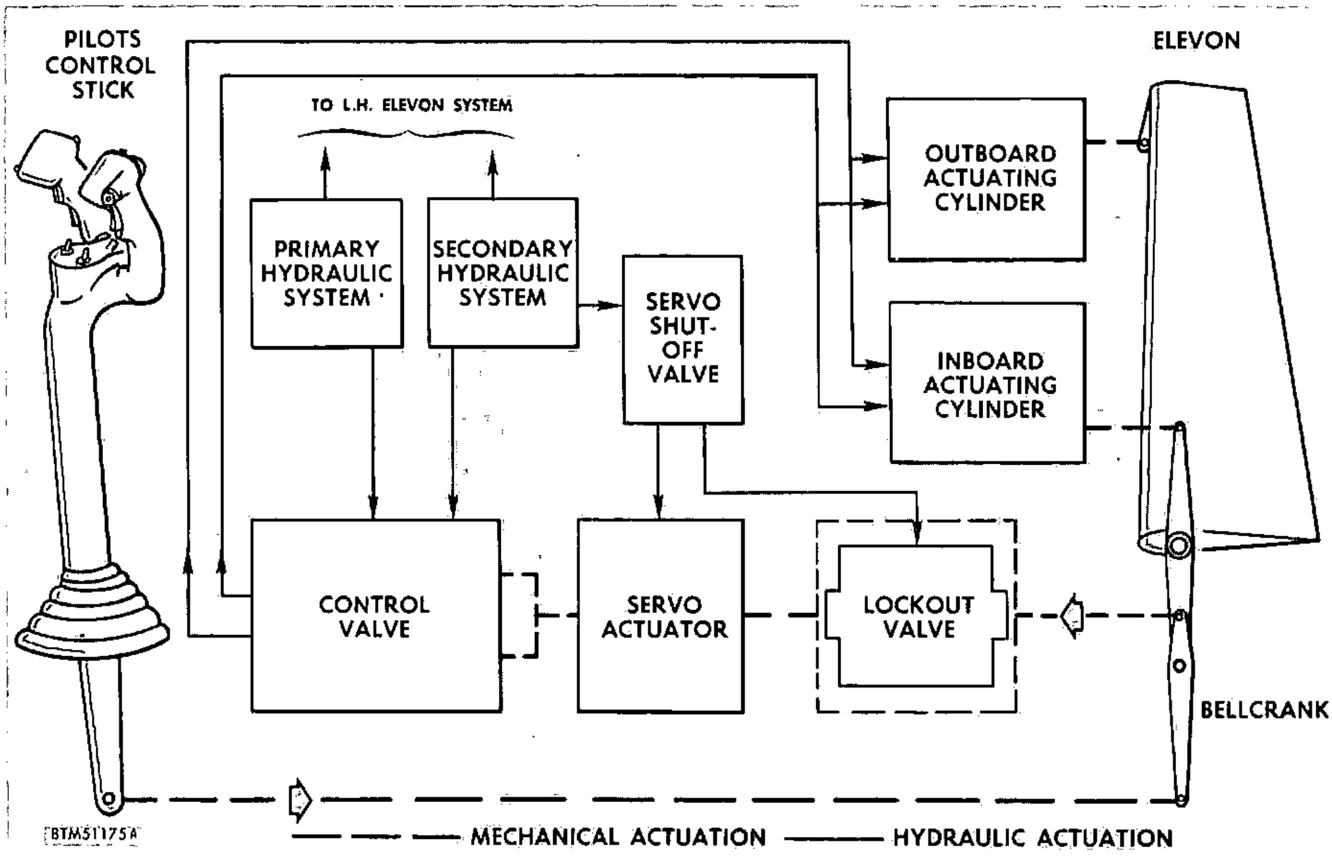

Figure 2-10. Elevon System Diagram

however, overpower the lockout valve when necessary by exerting a force of approximately 20 pounds on the control stick.

The AFCS components then act automatically to control the movement of the elevons. The servo actuators are energized hydraulically and are free to answer electrical signals from the damping system to control elevon movement. When a signal is received from the damping system for a course or attitude correction, the servo actuator extends or retracts its shaft to move the hydraulic control valve spools and port hydraulic fluid to the elevon actuators.

Although the pilot is locked out from direct control of the elevons, means of varying the stabilized attitude of the airplane are provided by a trim switch, located on the pilot's control stick. The trim switch operates through the AFCS components and not through the electrical trim actuators to effect trim changes; this is called the "Beep" trim.

The mechanical followup linkage we discussed earlier is now replaced by an electrical feedback system. This feedback system moves the control valve to stop elevon movement when the desired displacement has been reached.

Damping signals from the AFCS are superimposed on the elevons through the damping system to insure stability. The airplane remains in the attitude established by the pilot at the time of AFCS engagement, except when a bank angle of less than 5 degrees exists; in that case, heading control engages and the plane rolls out as required to maintain heading. When an angle of bank greater than 5 degrees exists at time of engagement, heading control does not engage and roll control functions to maintain the bank angle.

A detailed discussion of the lockout valve and its maintenance is given later in this chapter; Automatic Flight Control mode discussed in Chapter V. We should also mention here that the rudder control system does not include a lockout valve as does the elevon system, therefore the rudder is always at the command of the pilot regardless of the mode of operation. AFCS components not only control movement of the elevons automatically, but also provide turn coordination movements for the rudder.

ELEVON HYDRAULIC SYSTEM.

Full hydraulic power is supplied to the right and left elevon flight control systems from the primary and secondary hydraulic power supply systems. The right elevon hydraulic system is shown in the block diagram on figure 2-10; the left elevon hydraulic system

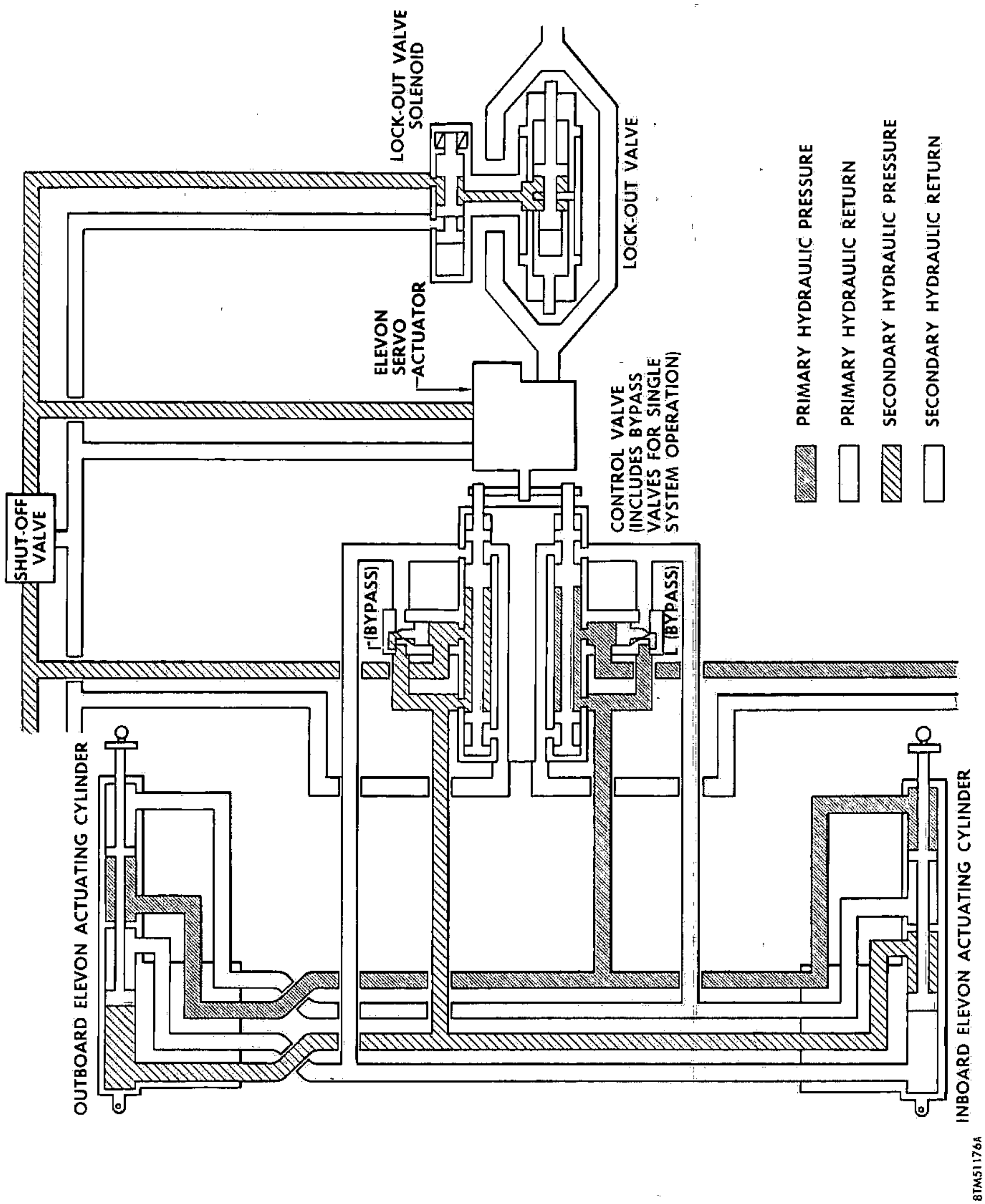

Figure 2-11. Elevon Hydraulic System Schematic

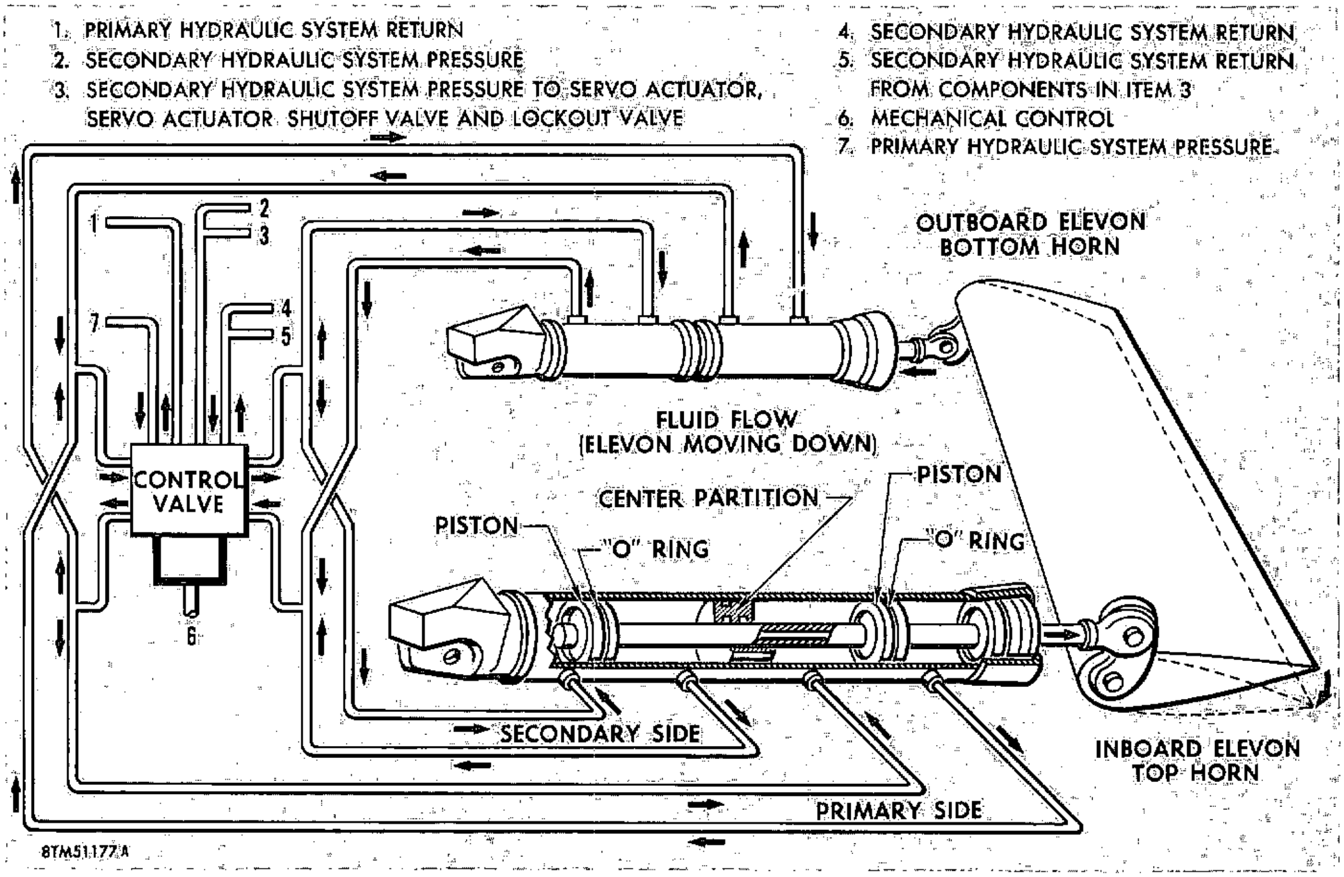

Figure 2-12. Elevon Hydraulic Flow Diagram

is identical. The mechanical action from the control stick and the actuating cylinders is represented by dotted lines; hydraulic actuation from the primary and secondary hydraulic systems is represented by solid lines. Both the primary and the secondary hydraulic systems supply power by way of the control valve to the inboard and outboard actuating cylinders at the elevons. Note however, that the secondary system also supplies power to other hydraulic components in each elevon control system. These components are the servo shutoff valve, the servo actuator, and the lockout valve. These components operate only when the Manual and Automatic Flight Control systems are engaged.

Pilot control of the elevons is by means of a conventional control stick and the mechanical positioning of the hydraulic control valves. The control valves in turn meter primary and secondary hydraulic fluid to the control surface actuating cylinders (rams) to move the elevons. The elevons move only as long as the pilot moves the control stick. When pilot movement of the control stick stops, a mechanical followup mechanism neutralizes control forces in the hydraulic control valve. The valve then equalizes fluid pressure in the actuating cylinders, causing the control surface movements to stop. The position in which the elevons stop is determined by the position at which the pilot stops the control stick. Operation of the pilot's control stick is the same as any conventional airplane.

SYSTEM OPERATION.

Since the elevon systems on both the left and right sides of the airplane are identical, only the hydraulic operation of one system is described here. Note in figure 2-11 that pressurized fluid is routed from the primary system (lower center of schematic) to one section of the control valve. The fluid passes through the primary section of the control valve when the valve is displaced, and then to the primary section of each actuating cylinder. Now, note how fluid from the secondary system (shown at top center of schematic) passes through its side of the control valve and on to its section of the actuating cylinders. Note also that hydraulic pressure from both systems does not enter the corresponding cylinders in the two actuating cylinders, since hydraulic pressure in the outboard cylinder extends the piston rod while pressure in the inboard cylinder retracts the piston rod. The outboard cylinder piston rod connects to a horn below the centerline of the elevon while the inboard cylinder piston rod connects to a horn above the elevon centerline (see figure 2-12). Therefore, retraction of one cylinder rod and extension of the other rod moves the elevon in the same direction. The details of how these cylinders function and why this method of operation is used will be explained when we discuss the actuating cylinders.

In addition to supplying power to the actuating cylinders, the secondary hydraulic system also supplies power to other components in the control system. Note how another line branches off the secondary system pressure line and is routed through the shutoff valve (shown at top of schematic). This valve controls fluid to the elevon servo actuator and the lockout valve. Note that because of the location of the shutoff valve in the line, fluid cannot be admitted to the lockout valve unless it is also admitted to the servo actuator. With this in mind, note also that the servo actuator may be energized hydraulically, but that the lockout valve solenoid can at the same time prevent fluid from energizing the lockout valve hydraulically. These combinations of control, as we learned earlier, are used in varying ways during different phases of operation.

ELEVON HYDRAULIC ACTUATING CYLINDERS.

Dual hydraulic actuating cylinders and full hydraulic power are used to move the control surfaces. The hydraulic power is supplied simultaneously from both the primary and the secondary hydraulic systems. This is accomplished by the internal arrangement of the actuating cylinders.

Inside the cylinders two power pistons are mounted on a single piston rod and act within two separate sections of the cylinder bore. The piston rod is sealed with two O-rings at the point where it passes through the center partition of the cylinder bore. This prevents interflow of fluid from one hydraulic system to the other when both sides are pressurized. Also, a small drain hole drilled in the housing between the two O-rings permits any leakage due to a defective O-ring to drain off, thus preventing build-up of back pressure on the other O-ring. As you know, an inboard and an outboard cylinder are provided for each elevon, and the rod ends of these cylinders are attached to the inboard top and outboard bottom elevon horns. The inboard and outboard cylinders are hydraulically cross-connected as shown in figure 2-12. So, when positioning the elevon, one cylinder rod extends while the other retracts.

Unbalanced Design of the Elevon Cylinders.

In figure 2-12 note that in the secondary (forward) side of the cylinder shown in the cutaway view, the piston rod attaches only to one side of the piston. Now note that the piston in the primary (aft) side of the cylinder has the rod passing completely through it and into the secondary side of the cylinder. The piston rod attaches to both sides of the primary piston and to one side of the secondary piston. It occupies a percentage of the total piston area that would otherwise be presented to fluid pressure. The forward side of the secondary system piston has no rod attached to it so its whole area on that side is presented to fluid forces. Therefore, there is an unequal area presented to the fluid forces. This unequal area presented to the fluid forces causes a greater force to be exerted in the piston rod extend direction than in the rod retract direction by an amount equal to the operating pressure times the piston rod area of the retracting piston.

In other words, the fluid has more area to work against if the rod is not there, and therefore can push harder—do more work. Both the outboard and inboard cylinders operate on the same principle. Therefore, since the piston rods of the inboard and outboard cylinders move in opposite directions to produce the same elevon movement, one cylinder is always providing more work thrust than the other. In one direction of elevon movement the outboard cylinder does more work; in the other direction, it is the inboard cylinder. As you can see in figure 2-12, when the elevon is moving *down*, the inboard cylinder is doing more work than the outboard cylinder; when the elevon is moving *up*, the outboard cylinder will do more work.

Since one actuating cylinder is attached to the outboard bottom horn of the elevon and the other to the inboard top horn of the elevon, the elevon is forced slightly out of alignment at its trailing edge by the unequal thrust. The thrust is just enough to present a slightly twisted control surface to the airstream. One purpose of this relatively small twisting movement, or windup force, is to assist in overcoming any tendencies toward flutter or other unstable characteristics. The main purpose of this inequality of forces, however, is to take up the accumulation of slack in the various mechanical linkages to each actuating cylinder. This helps to prevent excessive wear of the linkage and prevents lag and jerkiness in the system.

FLUID FLOW TO THE CYLINDERS.

Referring again to figure 2-12, follow the directional arrows indicating fluid flow when the elevon is moving down. Also follow the arrows on the cylinder piston rods near the elevon horns denoting mechanical movement. Note also that there are four hydraulic lines connected to each cylinder. These lines connect to both cylinders and to the control valve. At the top center of the control valve, note the cluster of lines numbered 1, 2, 3, 4, 5, and 7. These lines are the pressure and return lines from and to the main primary and secondary systems, which are the source of fluid pressure for the elevon system.

Note also the mechanical actuator of the control valve, numbered 6. When the valve is operated mechanically, two spools inside move a small amount and allow the pressure from the main hydraulic system to actuate the cylinders. From the crossed-over lines between the cylinders and from the flow direction arrows, note that when fluid pressure is ported to the extend side of one cylinder, the return lines are also crossed over so that the return fluid flows in a similar way. As an example, on figure 2-12 note the line coming out of the control valve at the upper right side. This line extends to a tee which branches in two directions—to the right side of the secondary piston of the upper cylinder, and to the left side of the secondary piston of the bottom cylinder. Tracing the flow arrows in this part of the system, you can see that when pressure is admitted into these lines by the control valve, they will route pressurized fluid to the sides of the pistons described above. The primary side of the cylinders is pressurized in the same manner, and at the same time.

Since the inboard cylinder is attached to the top horn and the outboard cylinder to the bottom horn, and since the cylinders move in opposite directions from each other because of the crossed lines, they both push the elevon in the same direction. Remembering the unbalanced feature of the cylinders though, we know now that the inboard end of the elevon will be pushed down harder than the outboard end. The elevon retains the established twist in its trailing edge when it is returned to neutral or moved up, because the outboard cylinder will then be pushing harder than the inboard cylinder.

You can trace the arrows in figure 2-12 and see that when the inboard cylinder is pushing its piston and rod toward the elevon and the outboard cylinder is pulling its piston and rod away from the elevon, the return fluid from the return side of the pistons will be pushed through the control valve and to the main hydraulic systems again. When the cylinders reverse the direction of the elevon, the return sides of the pistons become the pressure sides, and the sides that are now pressurized become the return sides.

ELEVON HYDRAULIC CONTROL VALVE.

The elevon hydraulic control valve is actually two valves in one—it controls both primary and secondary hydraulic system fluid to its respective actuating cylinder. This complete valve assembly consists mainly of two selector spools, two bypass valves, and four fluid ports for each hydraulic system. Figure 2-13 shows a cutaway view of one side of a control valve —the other side is identical. The valve spools are linked together mechanically at one end so that both valves stroke uniformly and meter an equal amount of primary and secondary system fluid to each side of the actuating cylinders.

Note that the valve selector spool shown incorporates a bypass valve which is built into the control valve body. The other spool (not shown) also incorporates a bypass valve and is identical to the one shown. (Also see figure 2-11.) These bypass valves are incorporated to provide a low-resistance flow path through the valve if one side becomes inoperative. For example, when one hydraulic system—say the primary—becomes inoperative, the other system can still move the actuating cylinder pistons, thus causing the piston in the inoperative side of each cylinder to move its existing static fluid through the inoperative side of the control valve. If the bypass valve were not incorporated, the static fluid on the inoperative side would be forced across the lands of the spool. This is a relatively high-resistance flow path, and would cause damage to the system.

Control Valve Operation.

When both systems are in their normal operating condition (and they would seldom, if ever, be otherwise), both hydraulic systems are supplying equal pressure to their respective sides of the actuating cylinders. Mechanical actuation of the valve is instigated either by direct pilot action of his controls, or by the servo actuator when it receives an electrical impulse from the damper system. The normal flow and action of one of the valve spools is shown in the upper schematic of figure 2-13. You can see that normal pressure is passing through the spool area in an equal amount to both sides of the cylinder piston. In the upper view of the operating positions, the spool is shown in position No. 1. This would move the spool to your left and allow full pressure to leave port No. 1. In this position return fluid from the actuating cylinder can flow through the control valve to return No. 1.

If the spool is moved in the opposite direction to position No. 2, the pressure and return flows would be reversed. Note that when the spool is in the neutral position, the spool lands allow equal pressure past their inside edges to both No. 1 and No. 2 ports in equal amounts. This locks the cylinder pistons in the position they have been placed in at the moment. The only thing that will move them from this position is movement of the spools as described above.

In the upper schematic, note that system pressure entering the valve is routed to the back of the bypass valve and holds the bypass valve to the right. In this position the valve closes the bypass passage between port No. 1 and port No. 2. Now in the lower schematic a condition is set up which shows how the bypass valve functions. Here system pressure is inoperative and the control valve spool is displaced for elevon movement. Note that pressure No. 1 is now static (no pressure). Movement of the actuating cylinder by the remaining operating system causes fluid in the inoperative side of the cylinder to be displaced from one side of the piston to the other. This displaced fluid entering the control valve moves the bypass valve to the left, thus allowing pressure to bypass through the

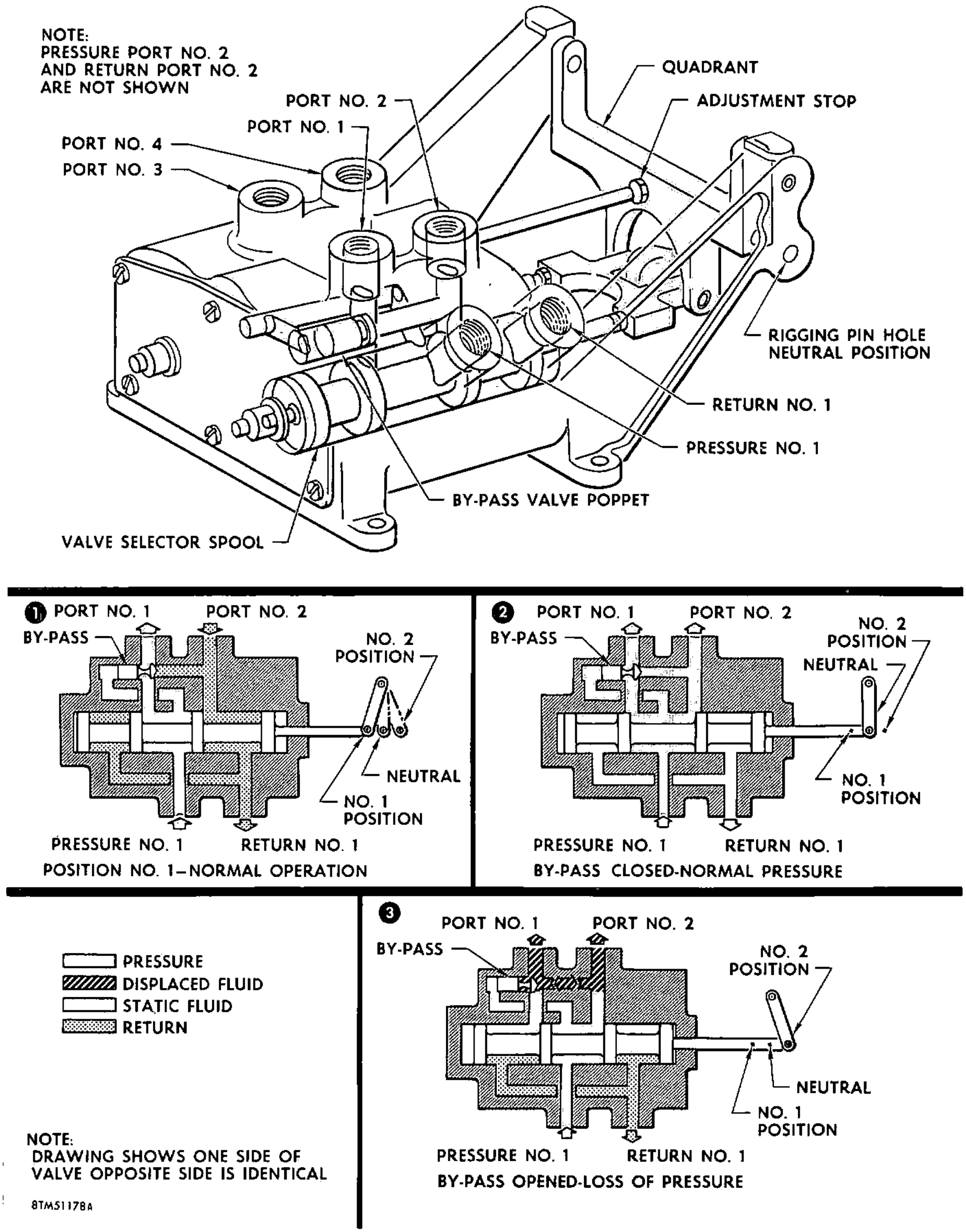

Figure 2-13. Elevon Control Valve

valve between port No. 1 and port No. 2. The bypass valve will remain in this position until system pressure again enters pressure No. 1.

Maintenance.

The valves are machined to very close tolerances, and any clogging in the valve body or particles of foreign matter around the spools affects the performance of the actuating cylinders. During maintenance, these conditions may be detected by sluggish response to motion of the control stick. Also, defective O-rings may cause leaking, which of course affects control performance too. If control is sluggish and hydraulic pressure is up to standard in the system, the control valve is the most likely unit to be suspected. Replacement of the unit is the remedy.

To remove a faulty valve, you should first make sure that there is no pressure in the two hydraulic systems, then you can disconnect the eight hydraulic lines from the valve. The four center lines which connect to the top of the valve must be removed completely to allow the valve to be lifted out of position. Therefore, mark or tag the lines in some manner so that you will know where they go when you reinstall them. Also cap or plug all openings in disconnected tubing and components when working on any part of a hydraulic system. This is to insure that no foreign matter or moisture enters the system.

After removing or disconnecting the tubing, disconnect the push-pull shaft of the servo actuator from the control valve by removing its connecting bolt and nut. *Do not* disturb the shaft adjustment on the servo actuator. All that remains to be done then is to remove the mounting bolts from the valve and lift the valve out of its position.

Installation of a new valve is the reverse of the removal procedure; that is, the new valve is bolted in position, the push-pull shaft is connected to the servo actuator shaft, and the tubing is reinstalled. The new valve comes pre-adjusted for neutral position and throw of the spools and will be identical to the old valve in adjustments. Therefore, it should not be necessary for you to disturb the adjustment stop on the quadrant. However, always install a rigging pin in the rigging hole, shown on figure 2-13, so that the valve mechanism is held rigid when connecting the push-pull shaft to the servo actuator. Always be sure that you remove this pin when you complete the installation.

This description has pointed out the highlights of the valve replacement. Refer to your F-102A Maintenance Manual for detailed step-by-step instructions and checkout procedure for the system after valve replacement.

Servo Actuator.

Figure 2-14 shows the servo actuator as it appears in the control shaft. This actuator is comprised of a spring-loaded pilot spool, a spring-loaded slave piston, an electrical torque motor, and flow passage restrictors. In the illustration note how the shaft connects to the elevon control valve (also see figure 2-13). This shaft is part of the slave piston shown in the lower schematic view. When signals come from the damping system, this shaft can move either in or out about 0.045-inch maximum depending on the corrective signal received. The 0.045-inch movement can displace the control valve spools about ⅛-inch, or 1/16-inch on either side of neutral. Thus you can see that this valve affects elevon movement to a very small degree (one degree maximum on each side of neutral during damper system operation).

The other (aft) end of the servo actuator is rigidly connected to the control shaft so that pilot-initiated motion of the elevons still displaces the control valve as in the Direct Manual mode. On top of the actuator is the torque motor that receives the corrective signals from the damper system.

Servo Actuator Operation.

Earlier in this chapter you learned that the servo actuator cannot be energized hydraulically unless the shutoff valve is energized. As you will recall, in the Manual and Automatic Flight Control modes the shutoff valve is open, and the servo actuator is energized hydraulically and is free to respond to electrical signals from the damping system to control flight.

In the schematic diagram on figure 2-14, the servo is shown in the neutral position. Note that hydraulic pressure entering the actuator bleeds in around the center portion of the pilot spool and goes to the two nozzles at the flapper. When the torque motor is de-energized (no signal from the damping system), the flapper is an equal distance from the two nozzles. Pressure bleeding through the two nozzles then passes on out the return outlet. However, when the damper system sends an electrical signal to the torque motor, the torque motor moves the flapper toward one of the two nozzles in an amount equal to the intensity (voltage) of the signal. As the flapper moves toward a nozzle, a fluid pressure unbalance is set up at the pilot spool, thus causing the pilot spool to move to one side. The pressure unbalance is set up because a back pressure is created at the restricted nozzle.

As an example of this pressure unbalance at the pilot spool, let's suppose that the torque motor moved the flapper against the right nozzle. This flapper then restricts flow through the right nozzle, and, at the same time, in moving away from the left nozzle allows more flow through it. Pressure in the passageway between the restricted right nozzle and the right end of

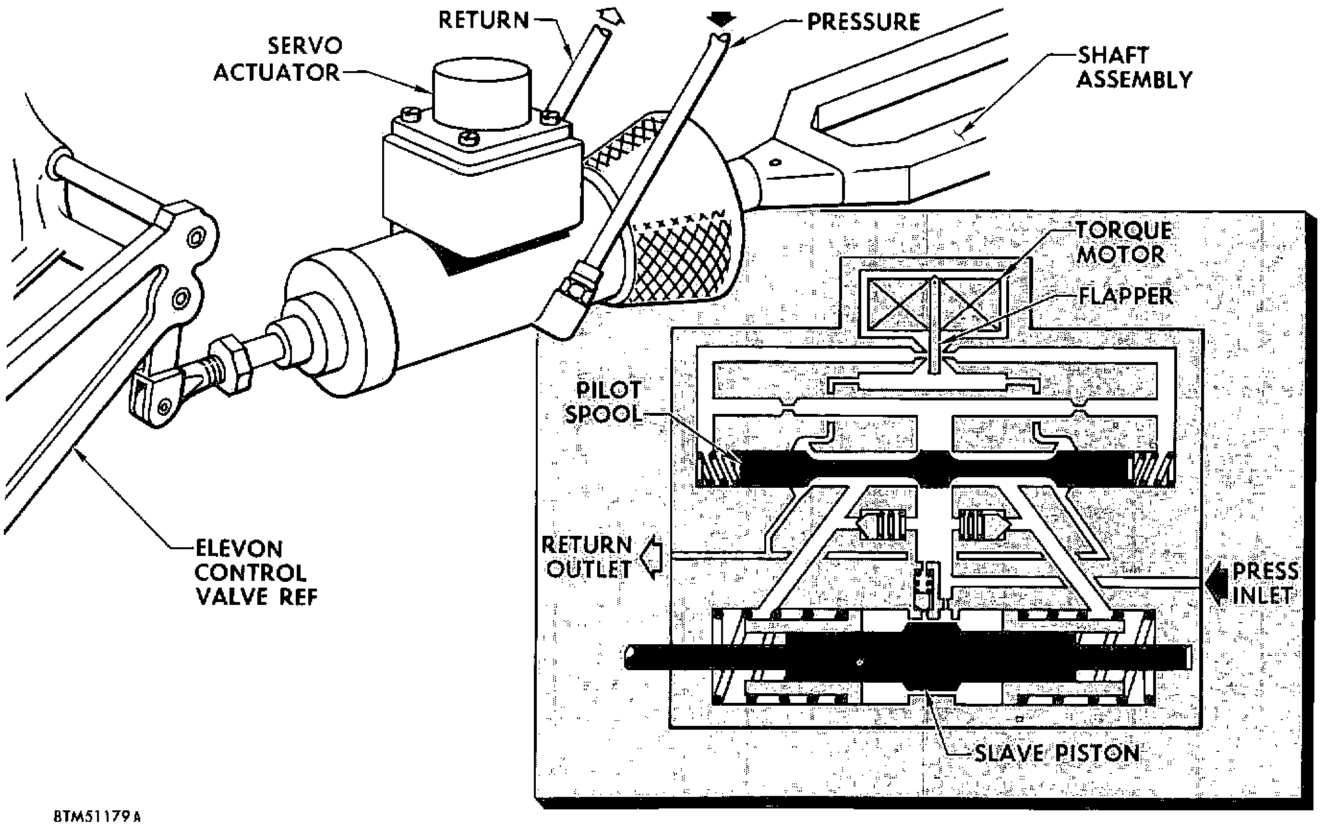

Figure 2-14. Elevon Servo Actuator

the pilot spool is increased because of this restriction. This pressure increase causes the pilot spool to move to the left against the spring. Pressure, which in neutral bled past both sides of the pilot spool, now is diverted exclusively to the right end of the slave piston (which is part of the servo actuator shaft) and moves the control valve spools away from neutral. This control valve repositioning admits fluid pressure to, and ports return fluid from, the appropriate sides of the elevon actuating cylinders. The elevon is thus moved to its new position to correct flight attitude.

As the elevon actuating cylinders move, the feedback potentiometer connected to the cylinder (shown on figure 2-4) sends another electrical signal to the damping system. The signal intensity depends upon the distance the cylinder moves. This second signal is of opposite electrical sign to the first signal and tends to null it out. The second signal thereby rapidly reduces the differential current (opposite current) to the torque motor to zero. As the two signals are balancing, they are moving the servo actuator flapper towards neutral. When the two signals are completely balanced, the flapper has returned to neutral, again equalizing the fluid pressure on both sides of the pilot spool, thus causing the servo actuator slave piston to return to neutral. Consequently, the control valve spools have also been returned to neutral, but the elevon actuators have been locked in the new position called for by the original input signal to the torque motor. The elevon actuators stay in the new position because the return side of the control valve spools are at neutral, thus locking the fluid on each side of the actuating cylinder piston. The only way the actuating cylinders can move further out, or back to neutral, is for the control valve to again move to direct fluid pressure to the appropriate sides of their pistons. This control valve action, as you know, can either be originated by damping signals through the servo actuator or by pilot initiated motion.

Maintenance.

If the servo actuator does not function properly, or not at all, you should first check to see if signals are being received from the damping system. To do this, you should check with the maintenance personnel in charge of the automatic flight control equipment. They can tell you if signals are arriving at the servo actuator. If signals are being received, then the trouble is in the actuator. In this case, you would replace the entire servo actuator assembly.

Troubles in the actuator would either be electrical or hydraulic. If the electrical circuitry and torque motor

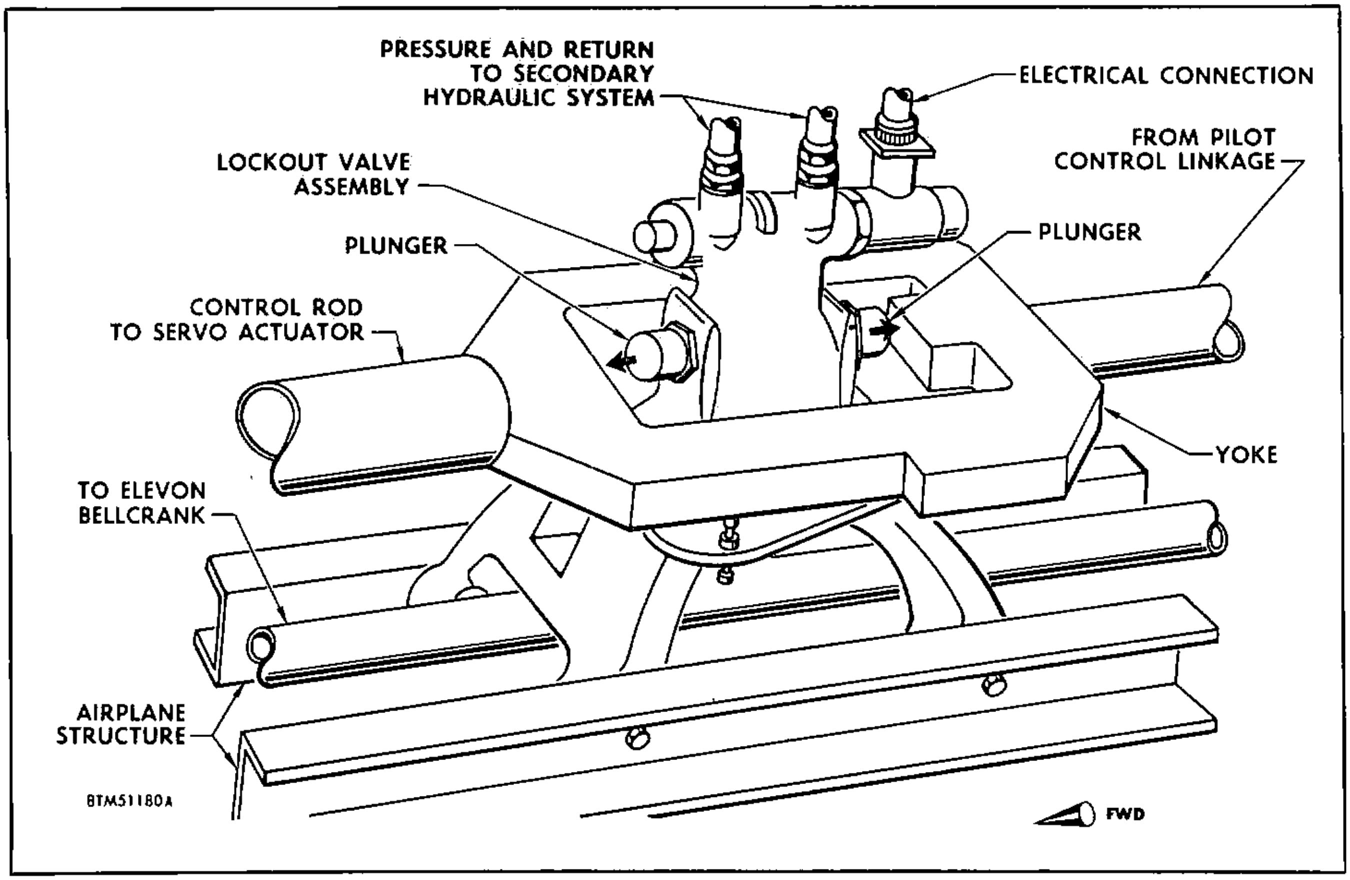

Figure 2-15. Elevon Lockout Valve and Yoke

is satisfactory, and the secondary hydraulic system is delivering proper fluid pressure to the servo, then leaks or restrictions within the servo would cause the trouble. If exterior leaks are causing actuator malfunction, you may be able to correct the difficulty without replacing the actuator. If a leak is discovered in the fittings, remember to return the hydraulic pressure to zero before attempting to tighten or replace the fittings. A fitting could crack while being tightened and cause personal injury if the line were pressurized.

To remove a faulty servo actuator, you must first return the hydraulic system pressure to zero. You will then disconnect the two flex hoses from the servo, disconnect the electrical fitting, and remove the bolt connecting the push-pull shaft to the control valve mechanism. After everything has been disconnected, the servo is removed by unscrewing the knurled collar nut from the back end of the assembly.

Before installing the new unit, you must first install two rigging pins in the system linkage. The forward pin inserts in the rigging pin holes in the control valve quadrant. These pin holes are located in the forks immediately above the point where the servo push-pull shaft connects to the elevon control valve, as shown in the illustration, figure 2-14. To install the aft rigging pin, you must first remove the upper push-pull rod connecting bolt from the aft elevon bell crank assembly attached to the elevon horn. With the bolt removed and the elevons in the neutral position, you can push the rigging pin through the bell crank and rod until it enters the hole in the end of the elevon horn pivot bolt outboard of the bell crank.

Now you can install the servo assembly on the end of the push-pull control shaft and safety wire the knurled collar. With the rigging pins still in place, you then connect the actuator shaft to the control valve quadrant. If the actuator shaft connecting point is too far forward or too far aft, making it difficult to install the bolt, loosen the lock nut on the shaft and adjust its length to suit so that the connecting bolt can be installed. Don't forget to tighten and re-safety the lock nut. The flex hoses and the electrical connector can then be attached and the rigging pins removed. Again refer to your F-102A maintenance T.O. for the step-by-step instructions on actuator replacement and checkout procedures for the system.

Elevon Lockout Valve.

In figure 2-15 you can see how the lockout valve is installed. Note that it extends up from the airplane structure and through the yoke of the control shaft.

This valve cannot be energized hydraulically unless the automatic flight control system is engaged. The shutoff valve which hydraulically energized the servo actuator also routes hydraulic pressure to the lockout valve. The lockout valve consists of a lockout valve solenoid and two plungers as shown in the hydraulic schematic on figure 2-11. When the automatic flight control system is engaged, the solenoid is energized and allows hydraulic fluid to enter the lower part of the lockout valve. This hydraulic pressure extends the two plungers that lock the control shaft so that motion is not transmitted from the control stick to the elevon control valve. The pilot can, however, overpower this valve when necessary by exerting approximately 20-pounds overcontrol to the stick.

The maintenance of this valve is comparatively simple. If the plungers do not extend when the automatic flight control system is engaged, or do not retract when the system is disengaged, you should first determine that power is reaching the solenoid for its operation. If power is at the solenoid and secondary hydraulic system pressure is up to standard, then the lockout valve is malfunctioning and will require replacement. Do not try to disassemble this valve to correct for malfunctioning; instead, replace it.

Hydraulic Elevon Package (HEP) Valve.

The HEP valve has been mentioned in preceding sections of this chapter. You learned that the HEP valve incorporates the servo shutoff valve, the lockout valve, the elevon servo actuator, and the elevon control valve in one package. Figure 2-16 shows the HEP valve and its external connection. If you will compare figure 2-16 with figure 2-7, you will note the absence of the above mentioned components in the HEP valve arrangement. The four hydraulic lines connected to the top ports of the valve are the input and return lines from the primary and secondary hydraulic system; there is an input and a return line for each system. The four hydraulic lines connected to the bottom ports of the valve are routed to the inboard and outboard actuators for the elevon. Two of these lines carry the primary hydraulic system input and return fluid to extend or retract the elevon actuators. Each of these primary lines is tapped so that there is a primary system input and return to both the inboard and outboard elevon actuators. The other two lines connected to the bottom ports of the HEP valve are the secondary system lines to the elevon actuators. They, too, are tapped to provide a secondary system input and return line to each of the inboard and outboard elevon actuators.

In figure 2-16, the torque motor is shown on the side of the HEP valve facing the viewer. Note the rods from the mixer tee to the aft elevon bell crank (aft elevon tie-rod) and from the elevon bell crank to the HEP valve (the pilot input rod). The pilot input rod is connected to a pivoted side arm. This side arm is connected by a rod to another pivoted side arm on the opposite side of the valve. The movement of both side arms is coordinated by the rod.

HEP Valve Operation.

Figure 2-17 is a schematic of the HEP valve and the right-hand inboard and outboard elevon actuators. The left-hand system functions in the same manner. In the schematic we see the torque motor valve, the shutoff valve and its unlock solenoid, the control valve, and the lockout valve and its associated pistons. The function of the HEP valve, to control the application of hydraulic pressure to the elevon actuators, is centered in the control valve. The control valve consists of a single spool and two independent end sections. The spool is centered between two high-rate, preloaded, compression springs and is displaced mechanically by pressure from the side arms against the end sections. Notice the parallelogram arrangement by which pilot input is coordinated in both side arms by a connecting rod. The fourth side of the parallelogram is the control valve. Note that each side arm of the parallelogram is displaced about a pivot.

The control valve spool is moved hydraulically by pressure controlled by the torque motor valve. This pressure is applied to the valve spool ends and is independent of the mechanical input. In the Manual mode of control, this pressure is determined by pitch damper system signals and is added to the mechanical pilot input to move the valve spool. In the Automatic Flight Control mode, the pressure controlled by the torque motor valve is determined by AFCS signals which are fed to the pitch damper system to control its output to the torque motor. In the Automatic Flight Control mode, the hydraulic pressure controlled by the torque motor valve is the only force acting on the control valve spool; the pilot input or mechanical control is locked out.

At this point, let us consider what happens when the control valve spool is displaced. We are not concerned with how it is displaced; we will determine that when we consider HEP valve action in the Manual, Direct Manual, and Automatic Flight Control modes. Figure 2-18 shows an internal view of the HEP valve. The valve spool is in its neutral position, shutting off the input and return ports to the primary and secondary systems. Suppose that the spool is moved to the *right* sufficiently to open both the primary and secondary input ports. Primary and secondary hydraulic system input pressure is then applied to the primary *down* elevon line and secondary *down* elevon line, respectively. The down elevon pressures cause the elevon actuators to move the elevons downward. At the same time that the two input ports are opened, two of the four return ports are opened. The primary and secondary system *up* elevon lines are now hydraulically

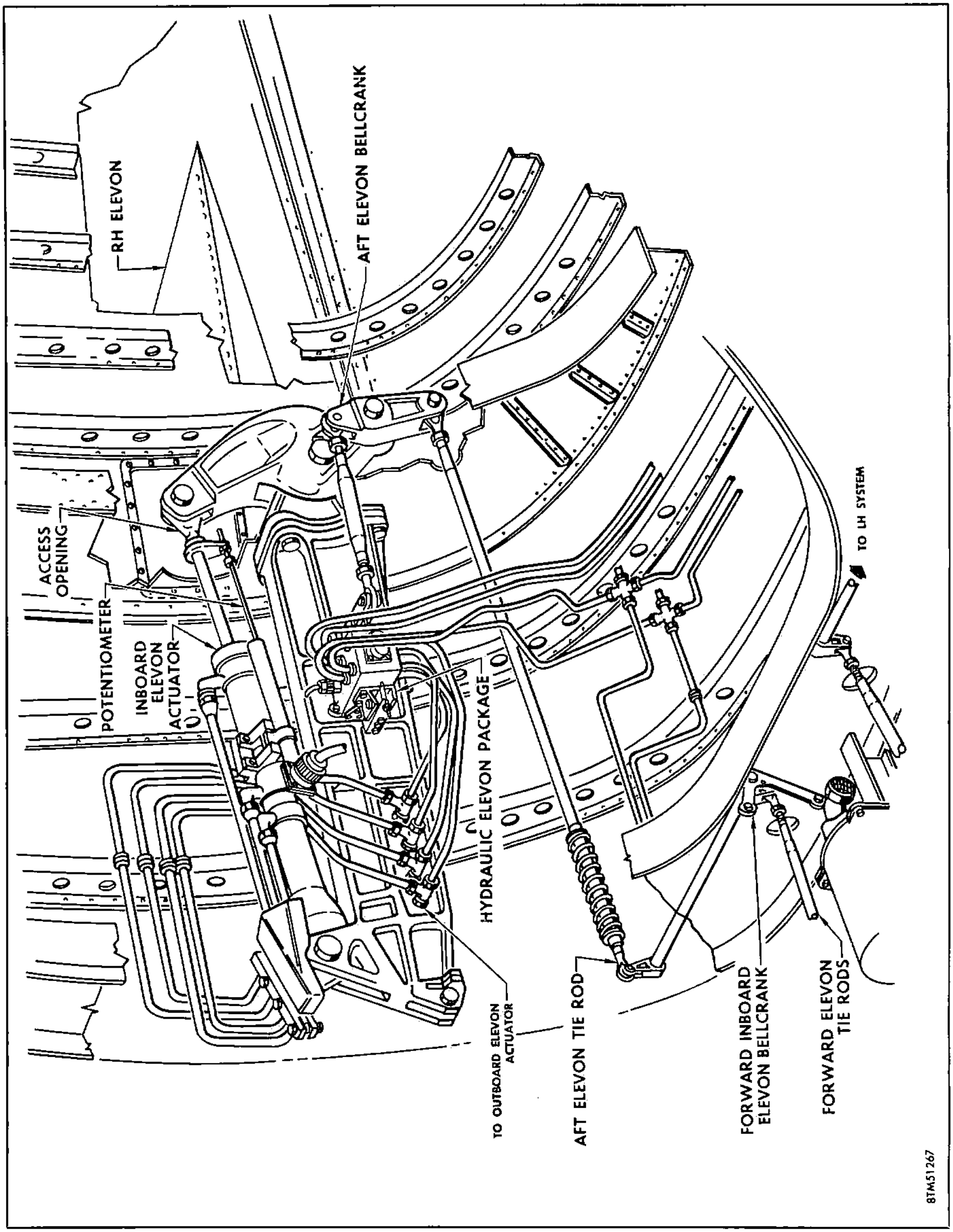

Figure 2-16. HEP Valve Installation

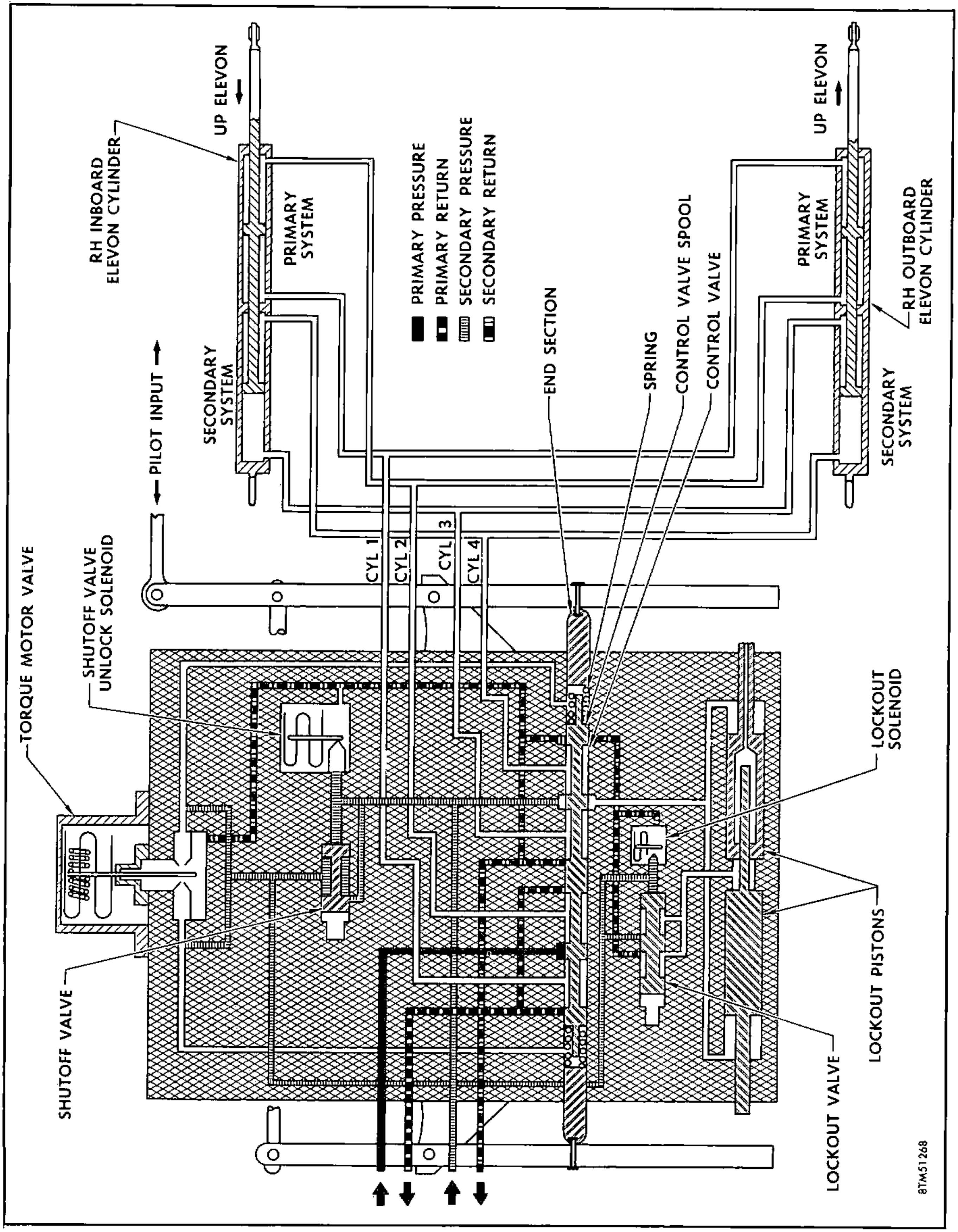

Figure 2-17. HEP Valve Elevon Control Schematic

connected to the return ports and serve to return the fluid displaced by *down* elevon-actuator action to the respective systems. You can see the displacing and the return flow to and from the elevon actuators by again referring to figure 2-17.

If the control valve spool is moved to the *left* sufficiently from its neutral position, it will again open the primary and secondary input ports and the two other return ports. But this time input pressure is channeled to the *up* elevon lines, and the *down* elevon lines serve as return lines.

In the Direct Manual mode, displacement of the control valve spool is effected only by pilot input and the parallelogram linkage which moves the valve end sections and springs in a push-pull action. The valve spool then controls the application of primary and secondary system pressures to the elevon actuators.

In the Manual mode, damper system control of the HEP valve is superimposed on pilot control. Damper system control is applied through the torque motor valve. To understand the function of the torque motor valve, we refer again to figure 2-17. Tracing the secondary hydraulic system input to the HEP valve, observe that this input is applied to the shutoff valve (in addition to the control valve). In the Direct Manual mode, the shutoff valve blocks secondary system flow to the torque motor valve and bypasses this flow through the shutoff valve unlock solenoid to the secondary system return line. In the Manual (and in Automatic Flight Control) mode of operation, the unlock solenoid is energized by the engagement of the pitch and yaw damper systems. The shutoff valve is then opened by its unlock solenoid and passes the secondary system input to the torque motor valve. With the torque motor flapper in its centered postion, equal secondary pressure is present at both ends of the control valve spool and at the nozzles of the torque motor valve. When a pitch damper system signal is applied to the torque motor, the flapper closes one of the two nozzles. The hydraulic balance at the torque motor valve is upset and an increased pressure is applied to one end of the control valve spool. The spool then opens its ports to direct hydraulic pressures to the elevon actuators.

As the signal applied to the torque motor from the pitch damper system diminishes, the torque motor flapper is returned to its neutral position by springs (shown in figure 2-18). In the Manual mode of operation, it should be understood that hydraulic pressure controlled by the torque motor and mechanical pressure from the pilot input linkage can both displace the control valve spool simultaneously. The hydraulic and mechanical pressures are simply added together.

In the Automatic Flight Control mode, AFCS signals control the output of the pitch damper system that is fed to the torque motor. In this mode the torque motor valve operates in the same manner that it did in the Manual mode. To make the torque motor valve the only source governing control valve action, pilot input is locked out in the following manner. When AFCS is engaged, the pitch and yaw damper systems are already engaged (or AFCS would not engage). Therefore, the shutoff valve will be open and secondary hydraulic system pressure will be present at the torque motor valve and also at the lockout valve. At AFCS engagement the lockout valve solenoid is energized and the lockout valve is thus opened. The secondary hydraulic system pressure present at the lockout valve is then applied to the lockout pistons. As a result, the lockout pistons are extended against the side arms of the parallelogram linkage, holding them rigid and forcing the parallelogram linkage and, therefore, the control valve spool, to a neutral position. No mechanical pilot input displacement of the control valve spool can then normally be made. However, a mechanical override provision is incorporated in the lockout components to permit the pilot to regain manual control of the aircraft should any failure of the shutoff or lockout valves occur. Mechanical input of a force slightly greater than the lockout load displaces the lockout piston.

There is one other function of the HEP valve which has an important bearing on complete engagement of the AFCS. That function is provided by the lockout switch and its cam on one of the lockout piston rods. This arrangement is provided on each of the two HEP valves (one for the right elevon and the other for the left elevon). The function of the lockout switches on each HEP valve is to make certain that both elevons are under AFCS full-authority control when in the Automatic Flight Control mode; that is, to prevent the condition wherein one HEP valve locks out mechanical feedback (correct AFCS operation), and the other HEP valve does not (permitting mechanical feedback to counteract AFCS signals). When the lockout piston rods are fully extended, the arm of the lockout switch falls into a detent in the piston rod and the switch is closed. These lockout switches for both HEP valves are in series, and, when both switches are closed, the AFCS engage circuit in the damper systems will be completed. If the lockout switch at either one of the HEP valves is not closed by the extended lockout piston rod, the AFCS engage circuit will not close and the AFCS will not engage.

Maintenance.

Maintenance of the HEP valve consists primarily in checking the hydraulic connections to the unit for leaks and the electrical input for faulty connections. Inspection of mechanical parts such as mounting bolts, pivot pins, and access plate bolts and lock wiring

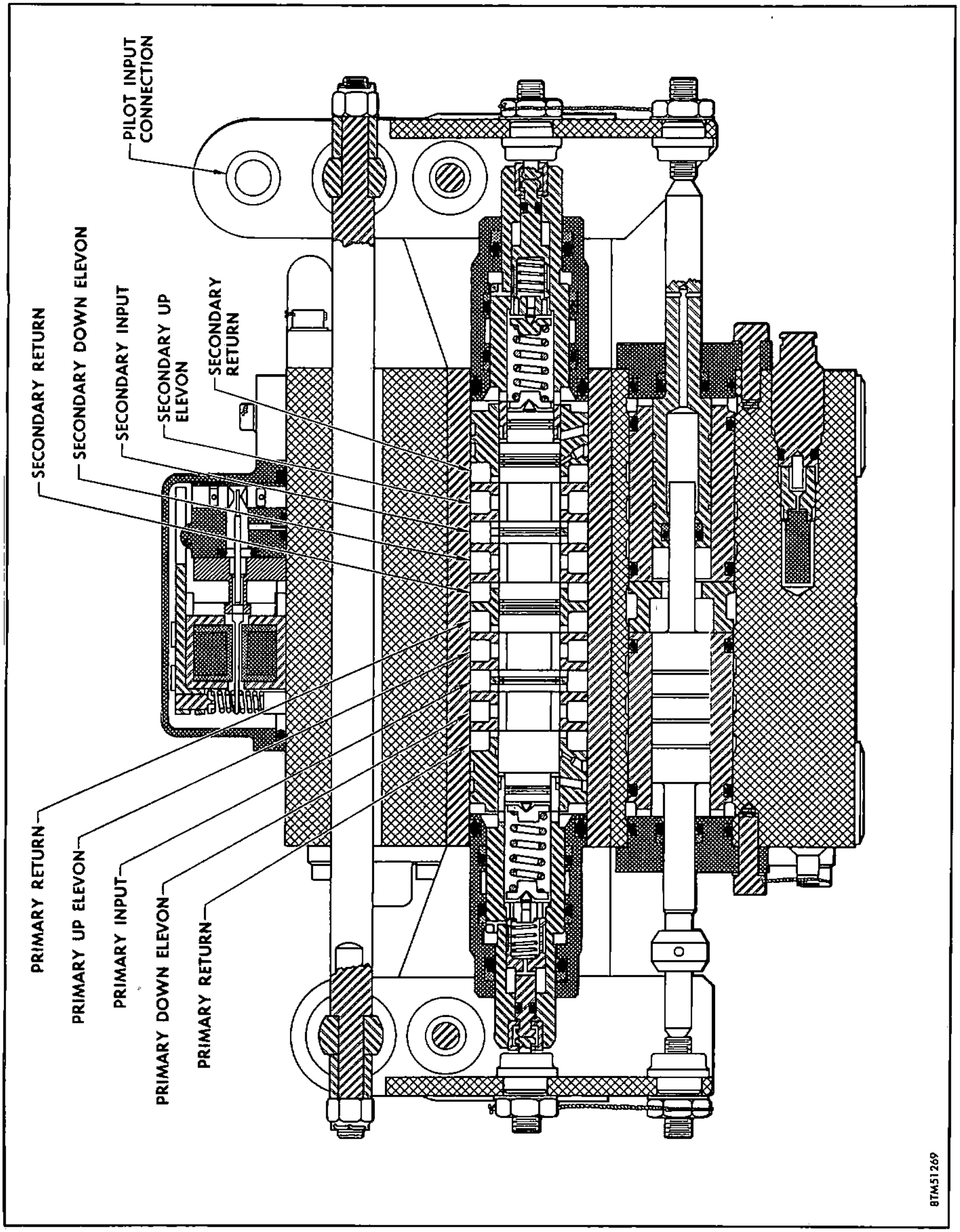

Figure 2-18. Control Valve Operation

should be made periodically. No internal trouble shooting should be performed. If the HEP valve is not functioning or is functioning improperly, check that electrical signals are being received from the damper system. Qualified personnel only should make this electrical check. If electrical input to the actuator is correct, hydraulic input to the unit should be checked, again by qualified personnel. If the electrical and hydraulic inputs to the HEP valve are correct, then the trouble is in the unit itself. The unit should be replaced. If external leaks are causing the valve to malfunction, you may be able to repair the leak. Return hydraulic pressure to zero before starting any repair work on hydraulic lines.

To remove a faulty HEP valve, you should make sure that there is no pressure in the two hydraulic systems and that there is no electrical power applied to the valve. Detach the pilot input rod and disconnect all electrical input. Remove the hydraulic lines from the top and bottom ports of the valve. Label the lines as you disconnect them to make sure that you reconnect them to the proper ports. Cap or plug all openings in the tubing after you have disconnected them to prevent any foreign matter from entering. After you have removed the hydraulic lines, you can then unscrew the mounting bolts and remove the valve.

To install a HEP valve, you simply reverse the removal procedure. The preceding maintenance information is only intended to acquaint you with the maintenance approach. For specific maintenance instructions, refer to your F-102A maintenance handbook, T.O. 1F-102A-2-7.

ARTIFICIAL FEEL SYSTEM.

As you recall, we learned earlier in the chapter that it would be extremely difficult for a pilot to manually operate the control surfaces of a high speed, supersonic airplane without the aid of some power generating device. High forces imposed on the control surfaces as a result of this high speed, make it necessary that the control surfaces be moved by a powered force. This force is provided by hydraulic actuators to move the rudder and the elevons on the F-102A airplane.

The pilot's control stick and rudder pedals, therefore, are not connected directly to the control surfaces, but act through linkage and control cables to position hydraulic control valves; the control valves then meter pressurized hydraulic fluid to the actuating cylinders to position the control surfaces in proportion to stick and pedal movement. This arrangement isolates the pilot from the normal feel of the airloads on the control surfaces, which he would feel were he directly connected to the control surfaces. Consequently, an artificial feel system is required to simulate an airload condition.

This system of simulating airloads on the control surfaces is known as the *Artificial Feel System.* The function of the feel system is to provide the pilot with feel forces which vary resistance to movement of the control stick and rudder pedals. The feel system, through a combination of "Q" (ram air from the intake tubes on the vertical fin) pressure and pneumatic pressure acting on feel cylinders, determines what control stick and pedal forces shall exist for varying stick and pedal displacements, altitudes and airspeeds. The feel system is added to the mechanical linkage of the flight control system and uses both springs and programmed air to provide these feel forces.

We will discuss the elevon feel system and the components that make up the system in detail in the following paragraphs. The rudder feel system will be discussed in the next chapter.

ELEVON ARTIFICIAL FEEL SYSTEM.

The elevon feel system is composed of separate systems for elevator and aileron control. The aileron feel system consists simply of a combined feel and centering spring cylinder as a part of the mechanical linkage of the aileron control system. This provides the pilot with aileron feel forces at the stick. The cylinder can also be pressurized by the high-pressure pneumatic system through the action of the roll rate limiter system to give additional forces to return the control stick to a neutral position.

The elevator feel system is more complex and consists of a feel force cylinder, the "Q" system loader, "Q" (ram air) pressure system, an articulated link and quadrant, and a "T" shaped bell crank. Low-pressure pneumatic system air is also used to provide motive power for one of the bellows in the "Q" system loader.

How the Elevator Feel System Operates.

The elevator feel system is shown on the schematic diagram in figure 2-19. Note that the "Q" system loader receives "Q" pressure air from both the rudder and elevator intake tubes on the vertical fin, static air pressure, and low-pressure pneumatic air. These air pressures are regulated in the "Q" system loader and supplied to the feel cylinder in amounts proportional to the speed of the airplane to provide the pilot with normal feel forces at the stick. Both "Q" intake pressures are equal and will vary with the speed of the airplane. The faster the speed, the higher the "Q" pressure, which results in more resistance put into the controls.

You will remember that forward and aft movement of the control stick results in forward and aft movement of the mixer assembly to produce elevator action. Referring to the schematic, it can be seen that a "T" shaped bell crank is connected to the feel cylinder quadrant at its aft end, and to the mixer assembly

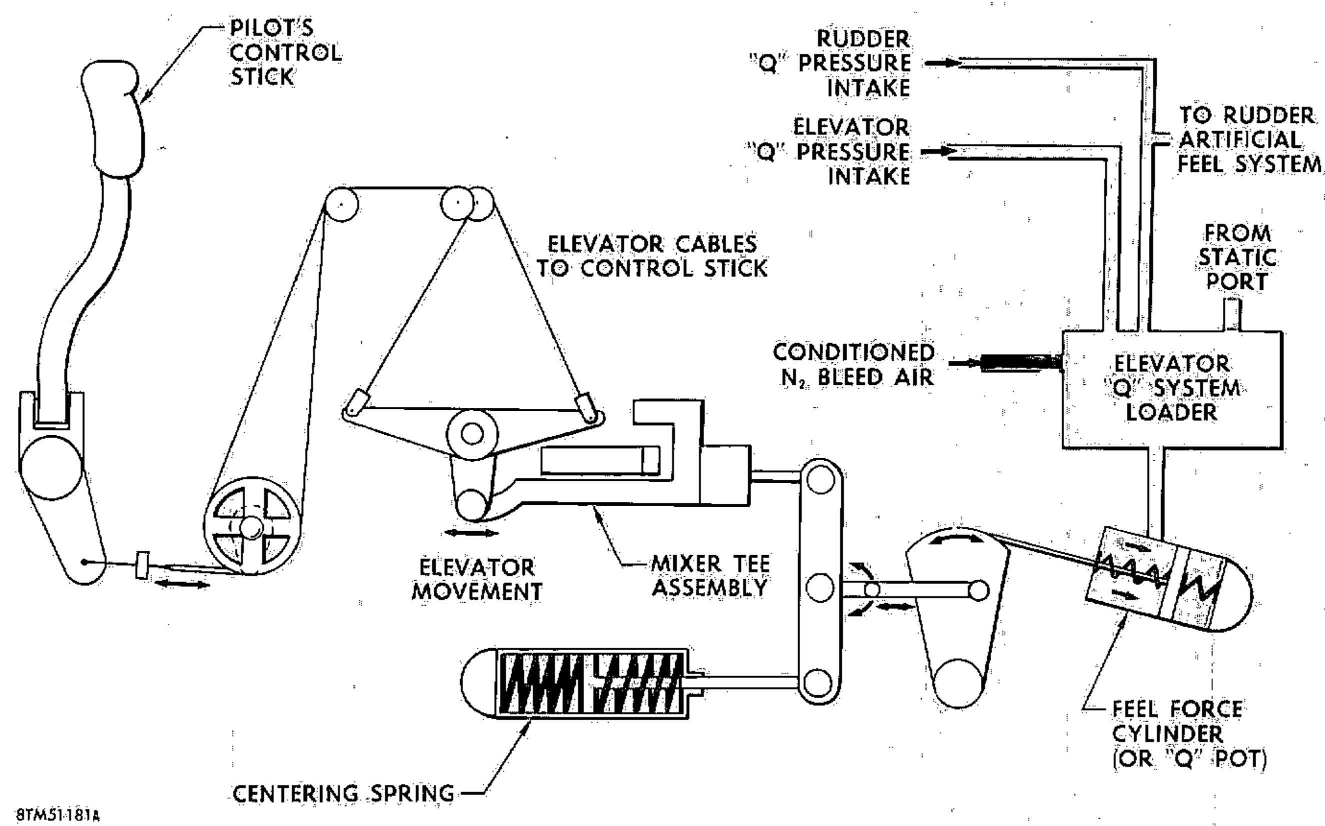

Figure 2-19. Elevator Artificial Feel System Schematic

and the centering spring at its forward arms. Also, it can be seen that as the mixer assembly is moved forward and aft, the bell crank will pivot about its attach point at the cross of the "T"; this will cause the pin-connected joint of the aft leg of the "T" crank to rotate in an arc as the mixer moves forward and aft. As the pin-connected joint rotates, the rotary motions are transmitted to the quadrant to which the articulated link is attached. This motion is always in the same direction regardless of the direction of motion of the "T" shaped bell crank. The motion will always result in a pull force in the cable from the quadrant to the feel force cylinder. Since the "Q" pressures are operating against the "low side" (small area side) of the piston in the feel cylinder, as indicated by the arrows, the motion of the quadrant is resisted with a force that is proportional to the pressures in the feel cylinder. This force is felt by the pilot at the control stick, since it is his movement of the stick that moves the mixer assembly and the "T" shaped bell crank.

Since the function and purpose of the feel system is to provide the pilot with feel forces at the control stick, the feel forces must generally increase with airspeed. On the F-102A, howver, elevator effectivity is not directly proportional to airspeed. At high subsonic speeds the elevator effectivity decreases; that is, more elevator deflection is required for a given amount of airplane response. As airspeed increases from this point, elevator effectivity again increases and the amount of elevator necessary for a given response decreases. Consequently, the pressures scheduled into the feel cylinder must decrease for a period to allow the pilot to obtain more elevator through the region where elevator effectivity decreases, and then the scheduled pressures must increase again as the elevator effectivity increases. The problem, then, is to schedule pressures into the feel cylinder that will simulate the forces that the pilot would normally feel (increase in feel forces with an increase in airspeed) were he directly connected to the control surface. This is the function of the "Q" system loader which will be discussed in the following paragraphs.

Components of the System.

As you recall, the components that make up the elevator feel system include the "Q" system loader, the feel force cylinder, and the "Q" pressure system. We will discuss these three items in more detail in the following paragraphs. The mechanical linkage of the feel system and how the system is introduced into the elevator control system was discussed in preceding paragraphs.

"Q" SYSTEM LOADER. This unit, as we learned, is a mechanism which, by means of a series of bellows and a cam, regulates and programs actual ram "Q" pressure air into the feel force cylinder. The loader is located in the engine accessories compartment and consists of a shock mounted computer assembly and a valve assembly. Ports are provided in the loader for static air pressure, ram air pressure, low-pressure pneumatic air, and regulated air pressure. No tubing is connected to the static port and the port should not be capped when the loader is installed in the airplane. The elevator "Q" system loader receives "Q" system pressure and "Q" pressure sensing from the pitot tubes on the leading edge of the vertical fin. "Q" pressure from the elevator "Q" pitot (lower tube) is metered or regulated by the loader's computer section and applied to the feel force cylinder. The computer section uses low-pressure pneumatic system air (engine bleed air) as a power source for operating the regulator valve. The applied "Q" pressure air in the feel cylinder is felt by the pilot as resistance to control stick movement.

When the "Q" system loader is suspected of malfunctioning, as may be indicated by improper elevator system feel forces, the unit should be replaced with a unit known to be serviceable. If such a replacement is not available, a test may be performed to determine the serviceability of the unit installed. Your F-102A maintenance handbook, T.O. 1-F102A-2-7, outlines the procedure for conducting this test; the handbook also outlines in a step-by-step fashion the procedure for removing and installing the "Q" system loader.

FEEL FORCE CYLINDER. The elevator feel force cylinder is also installed in the engine accessories compartment. The unit consists of a spring-loaded piston and cylinder assembly. Figure 2-20 shows the feel cylinder and linkage installation. Note how the cable passes through the hollow piston rod and is attached at the aft end of the rod. The forward end of the cable is attached to the quadrant. The cylinder is pressurized by variable air pressure from the "Q" system loader, as we just learned. This varying air pressure in the cylinder causes a varying force on the piston and is felt by the pilot as resistance to movement of the control stick in elevator motion.

When the feel cylinder is being installed, the cable should be aligned to center in the hollow tube. This can be accomplished by installing washers under the heads of attaching bolts at the forward end of the cylinder and by adjusting the turnbuckles at the aft end. The nuts on the aft end of the cable assembly are the means of adjusting the cable to impose the proper preload on the feel cylinder. The nuts should be tightened until the tube has moved ¼-inch into the cylinder. Complete installation and removal procedures of the feel cylinder are outlined in a step-by-step fashion in T.O. 1F-102A-2-7.

"Q" SYSTEM INTAKE TUBES. You will note by referring to the schematic diagram on figure 2-21 that two intake tubes are installed on the leading edge of the vertical stabilizer. The tubes provide the source of "Q" (ram air) pressure and route the air to the elevator "Q" system loader and rudder variable air pressure regulator in the elevator and rudder feel systems. The upper inlet tube supplies "Q" pressure air to the rudder pressure regulator, the airspeed compensator, and to the ¼-inch ram air inlet on the elevator "Q" system loader. The lower inlet tube supplies "Q" pressure air to the ¾-inch ram air inlet on the elevator "Q" system loader. The intake tubes are electrically heated through the airplane anti-icing system. Care must be exercised when installing the intake tubes to insure that the tube couplings and electrical connectors are properly engaged. Refer to the F-102A maintenance handbook, T.O. 1F-102A-2-7, for removal and installation procedures.

System Maintenance.

If low control stick forces are experienced in the flight control system, the most likely cause would be a leak in the "Q" pressure system. A test procedure for checking the "Q" pressure system for leaks is outlined in the applicable F-102A maintenance handbook. Leak tests must also be conducted whenever new or reinstalled equipment or lines are installed in the system. Another cause of low control stick forces could be malfuntioning of the elevator "Q" system loader. If after conducting the leak test, the low stick force condition is not corrected, the "Q" system loader is probably malfunctioning and must be replaced with a unit known to be serviceable. If such a unit is not available, a test may be performed to determine the serviceability of the unit installed.

TRIM SYSTEMS.

The elevons and the rudder on the F-102A airplane do not have conventional trim tabs. Trimming of the airplane is accomplished by electrical actuators that move the entire elevon and rudder control surfaces. The elevon trimming is controlled by a five-position toggle switch located on the pilot's control stick for elevator and aileron trimming. An elevator trim servo system is also provided to automatically change elevator trim to compensate for speed and altitude changes and speed brake extension in addition to the normal elevator trim function.

The elevator trim servo switch is located on the utility switch panel; when the system is engaged, the switch will remain in the ON position. The trim systems are interconnected through the takeoff trim circuit to provide a means of automatically trimming the airplane for takeoff. The takeoff trim switch is also located on the utility switch panel. Electrical power for takeoff

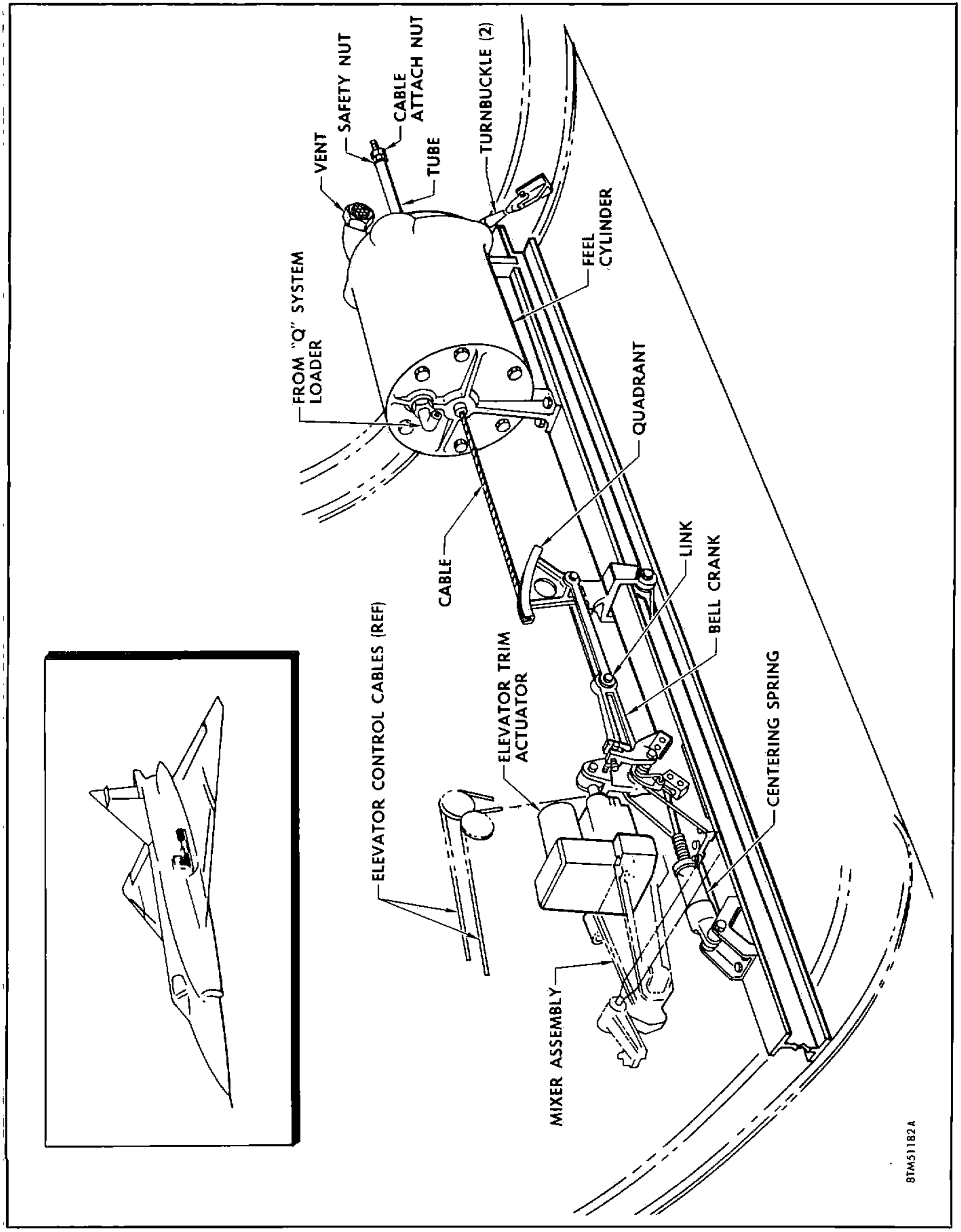

Figure 2-20. Artificial Feel System Perspective

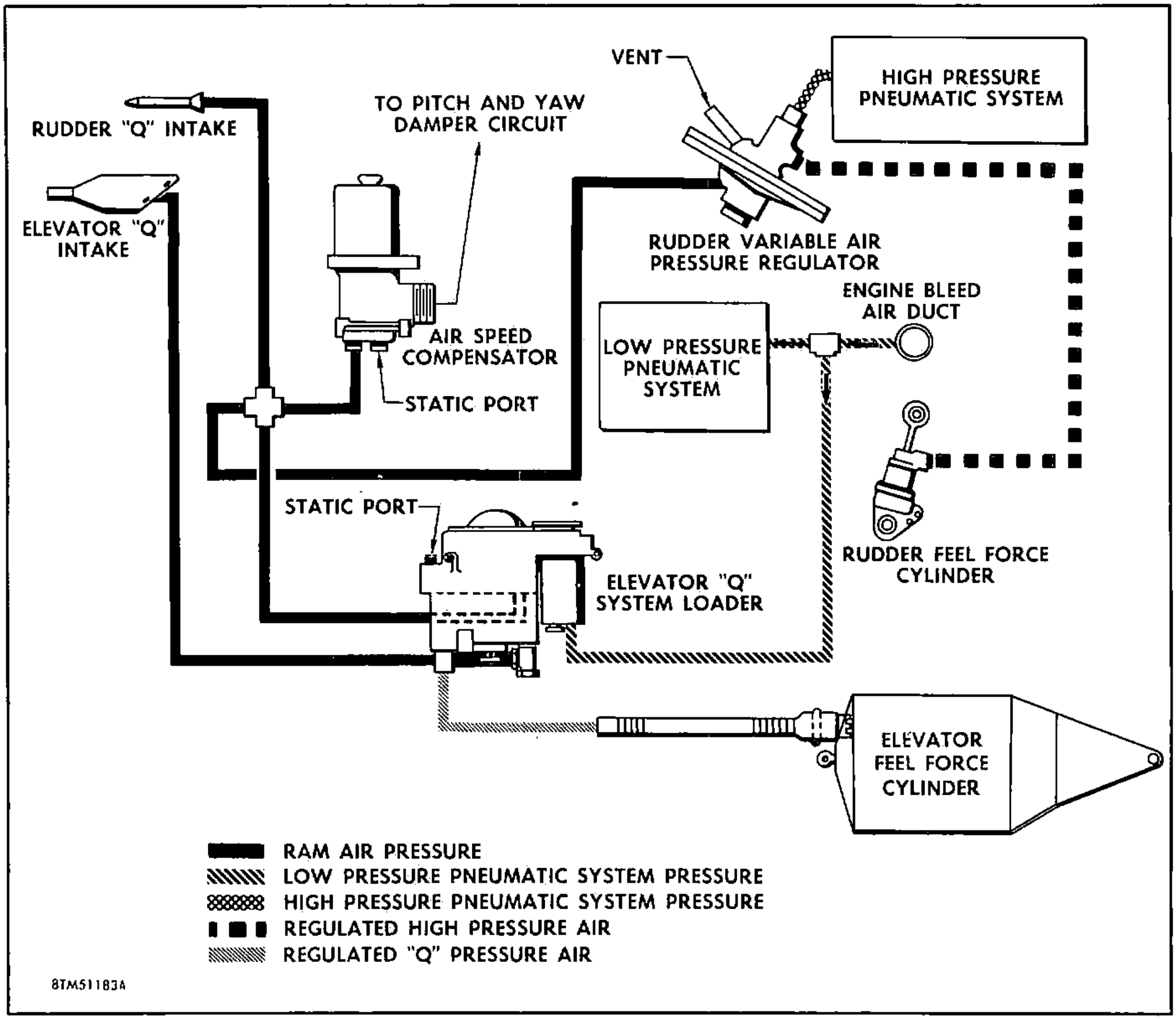

Figure 2-21. Elevator and Rudder "Q" Pressure System Schematic

trim is routed through the nose landing gear UP position switch so that the system is inoperative when the airplane is airborne.

WHY CONTROL SURFACES ARE TRIMMED.

When an airplane is loaded in such a way that it is slightly nose-heavy, wing-heavy, or tail-heavy, the pilot must exert a constant pressure on the control stick or rudder pedals, in the opposite direction to offset these unbalancing forces. To relieve the pilot of this tiring effort, ailerons, elevators, and rudder are often provided with trim tabs. The trim tabs maintain the control surfaces in exact positions away from neutral to maintain balance in straight and level flight. This is accomplished by moving the trim tab in the opposite direction to that in which the primary control surface is to be moved. The airflow striking the trim tab causes the main control surface to move to a position that will correct the unbalanced condition of the airplane. As we have learned, the control surfaces on the F-102A airplane do not have trim tabs; instead, the entire control surface is trimmed to provide the necessary trim action for maintaining balance of the airplane in straight and level flight. The following paragraphs will be devoted to a detailed discussion of the elevon trim system. We will discuss the system components and also learn how the system operates.

ELEVON TRIM SYSTEM.

The elevon trim system consists of separate trim systems for the aileron and elevator trim, as well as the trim servo system. The takeoff trim system electrical circuit is also routed through the elevon and rudder trim systems to trim the control surfaces for takeoff. We will discuss each of these systems individually for ease of understanding.

AILERON TRIM SYSTEM.

The aileron trim system consists simply of the electrical trim actuator connected in series with the feel and centering spring. The actuator shaft is attached to the aft aileron bell crank at the shaft rod end, while the actuator body is attached to the airplane structure, as shown on figure 2-22. When the five-position toggle switch, located on the pilot's control stick, is positioned for aileron trim, 28-volt, d-c power is routed to the aileron trim actuator. Aileron trim motion, is then introduced into the control system through the aileron feel and centering spring to the aileron bell crank, then through the push-pull control rod to the mixer assembly. The mixer assembly and control linkage transfer the trim motion to the hydraulic control valve to meter hydraulic fluid to the elevon actuator. The elevons then move to the trim position. Since trim is imposed against the feel and centering spring, this causes the system to center at the trimmed position. In aileron trim, the entire elevon surface is moved differentially to trim the airplane about its roll axis; that is, one elevon moves up while the other is moving down. The schematic on figure 2-23 illustrates schematically the aileron electrical trim circuit.

Aileron Trim Actuator.

The aileron trim actuator is located inboard of the mixer assembly at the aft aileron bell crank. The actuator consists of four internal switches (two limit switches and two position switches) and a reversible d-c motor to drive the actuator shaft. The two limit switches protect the motor by interrupting the circuit when the actuator reaches the fully extended and the fully retracted positions. The two position switches determine the neutral position of the actuator and provide the connections to drive the actuator to its neutral position for takeoff trim.

As we have learned, the aileron trim actuator is energized when the switch on the control stick is positioned for aileron trim. The trim actuator is also energized when the takeoff trim switch is actuated and the trim actuator is not in its neutral position. Power to the takeoff trim circuit is routed through the nose landing gear UP switch so that the system is inoperative when the airplane is airborne. A procedure for adjustment of the internal limit switches is outlined in T.O. 1F-102A-2-7.

ELEVATOR TRIM SYSTEM.

Normal elevator trim action is controlled by the five-postion toggle switch located on the pilot's control stick. The switch is spring-loaded to the center or *off* position. When the pilot positions the switch for elevator trim, 28-volt, d-c power is routed to the elevator

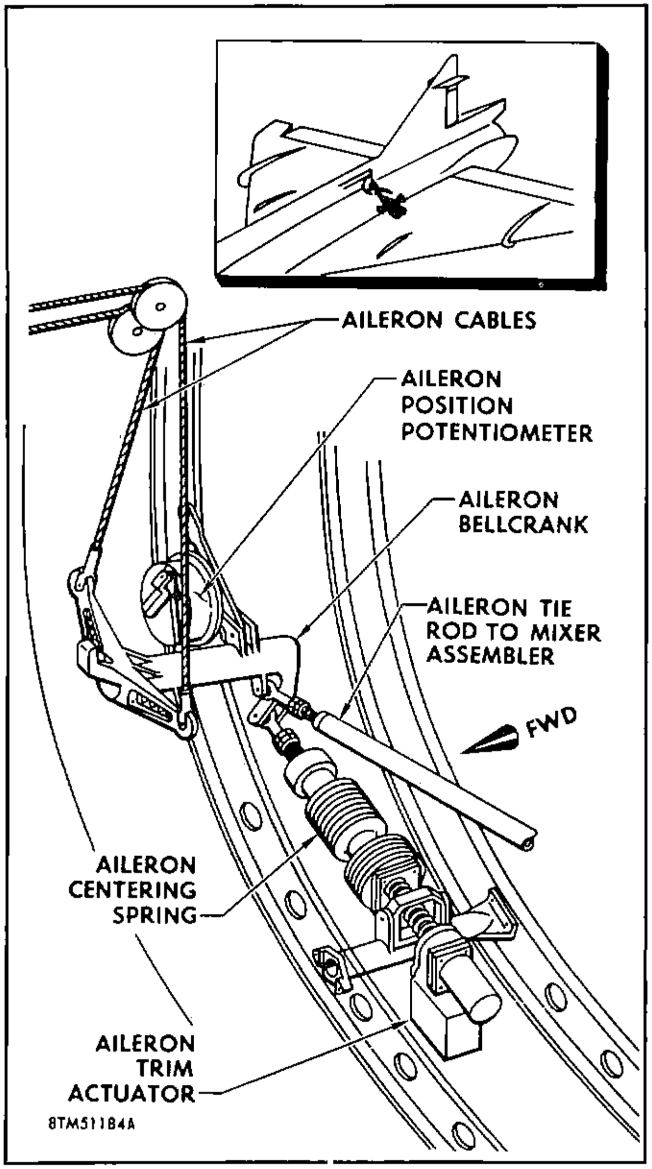

Figure 2-22. Aileron Trim System Perspective

trim actuator; the trim actuator then repositions the elevons through the mechanical linkage to provide the desired trim. When the trim switch is returned to the neutral position, elevator movement stops.

The elevator trim actuator is installed between the mixer assembly and the top leg of the elevator feel system bell crank, as shown on figure 2-24. The elevator trim system also consists of additional components to provide longitudinal stability functions, and is referred to as the *Trim Servo System*. This system provides the means of automatically changing the elevator trim to compensate for speed and altitude changes and speed brake extension, in addition to normal elevator trim functions. The elevator trim electrical circuit diagram is shown schematically on figure 2-25. Details on the elevator trim system and components

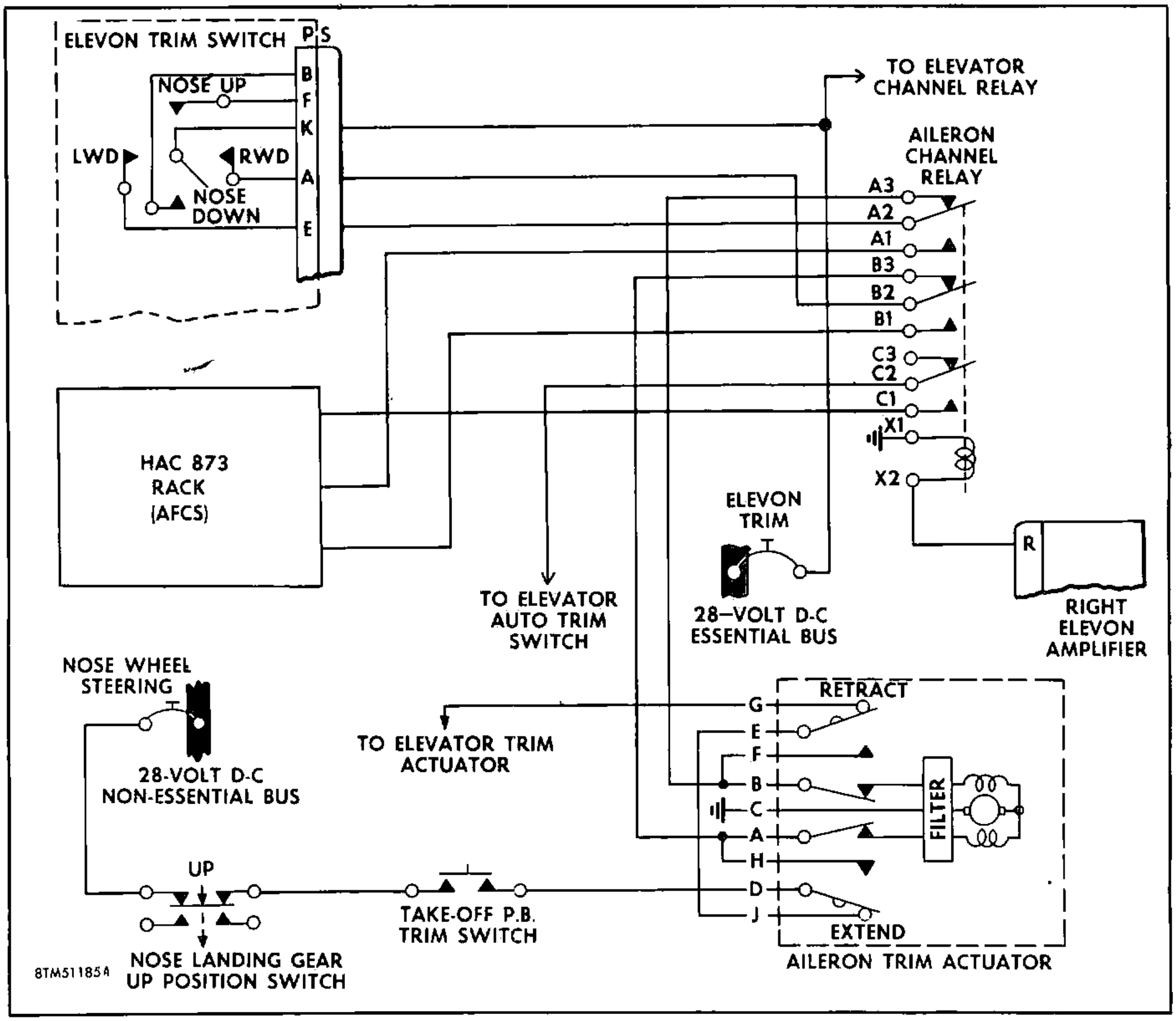

Figure 2-23. Aileron Trim Circuit Schematic

will be discussed in the following paragraphs as we learn how the elevator trim system operates.

Elevator Trim System Operation.

When the pilot places the trim switch in the elevator *up* or elevator *down* position, the elevons are repositioned to provide the desired elevator trim for the airplane. The trim switch routes 28-volt, d-c power to the elevator trim actuator which then *extends* or *retracts* the electrical trim actuator shaft. As shown in the elevator trim system illustration (figure 2-24), the trim actuator is installed between the mixer assembly cradle and the top leg of the feel system "T" bell crank; the lower leg of the "T" bell crank is attached to the centering spring. Since the centering spring is preloaded, and the force required to move the cradle assembly and the linkages to the elevon control valves is less than that required to move the centering spring, the mixer assembly cradle will move when the trim actuator shaft extends or retracts. The cradle assembly motion is then transmitted through the control linkage to open the hydraulic control valves. The control valves then meter hydraulic fluid to the elevon actuators to move the elevons to the desired trim position, as determined by the forward or aft movement of the mixer assembly cradle. The followup mechanism closes the control valves when the elevons are positioned to the elevator trim position selected by the pilot. When the pilot selects the Automatic Flight Control mode of control, the trim switch operates through the AFCS components instead of through the electrical trim actuators to effect the trim changes; this is called the "Beep" control.

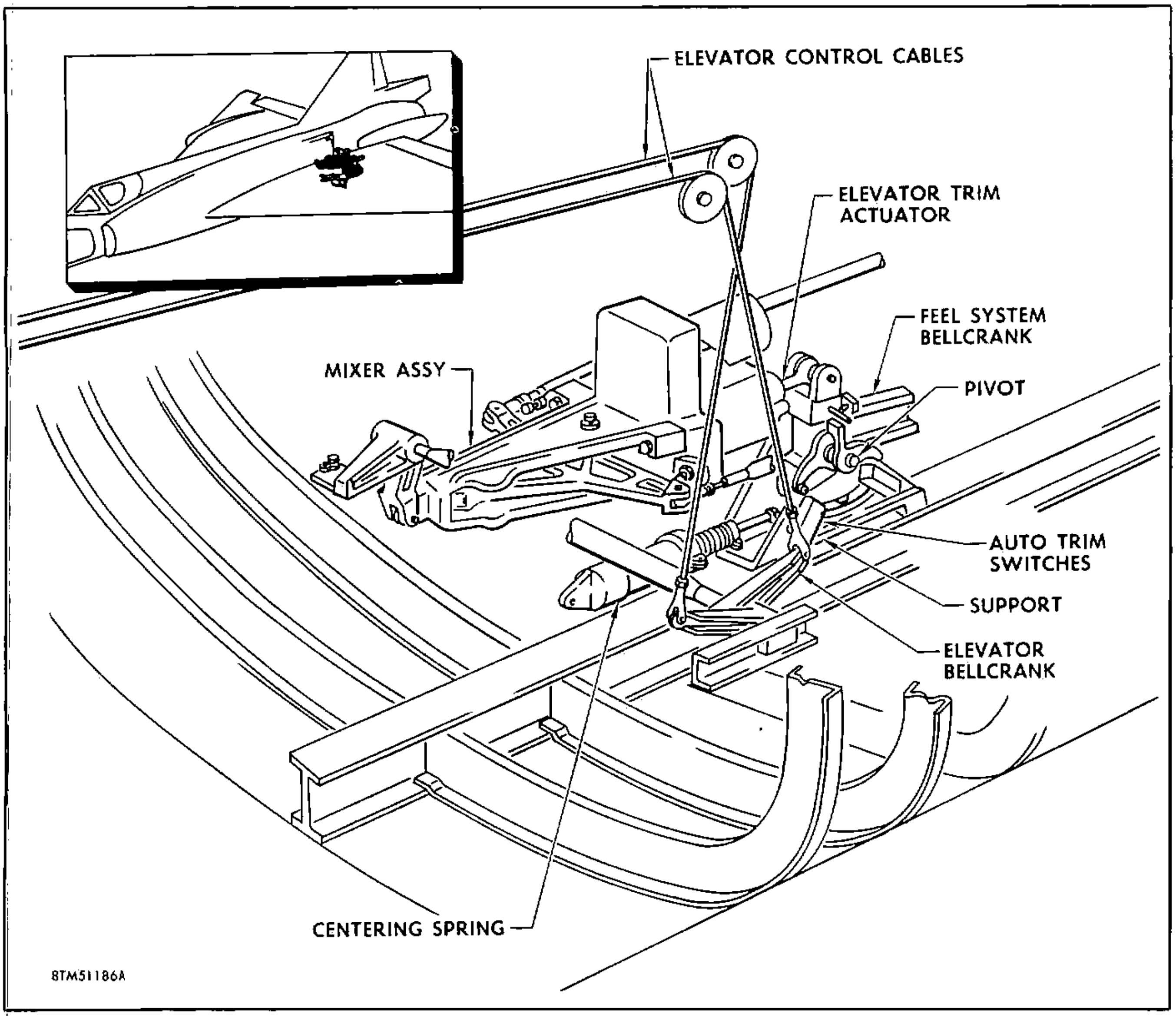

Figure 2-24. Elevator Trim System Perspective

Note on figure 2-24 the two elevator auto trim switches mounted on the left side of the elevator feel bell crank. The switches provide automatic trim followup when the airplane is operating in the Automatic Flight Control mode of flight. When the AFCS is engaged, d-c power is routed to the switches by the aileron channel relay located in the aft electronic compartment. Elevator motion of the elevons causes a displacement of the elevator feel bell crank. This bell crank movement actuates the switches which then route power to the elevator trim actuator. The trim actuator then follows elevator movements to extend or retract the actuator shaft. This action of the shaft rotates the feel bell crank arm of the articulated link toward the neutral feel point. This prevents violent airplane response if the AFCS is disengaged at a time when a large displacement of the mechanical control from the neutral feel point exists.

ELEVATOR TRIM ACTUATOR. As you have learned, the elevator trim actuator is installed between the aft end of the mixer assembly and the feel system bell crank. The actuator consists of a reversible d-c motor and four internal switches. The motor, of course, drives the actuator shaft; the four internal switches consist of two limit switches which interrupt the circuit when the actuator reaches its travel limits, and two position switches which determine the takeoff position of the actuator. A position potentiometer is an integral part of the actuator and provides feedback signals to the elevator trim servo system. The elevator trim actuator is energized by 28-volt, d-c power controlled by either the elevator trim switch on the control stick, the elevator trim servo box, or the elevator auto-trim switches. The actuator is also energized when the takeoff trim switch is actuated and the trim actuator is not positioned to the correct elevator trim for

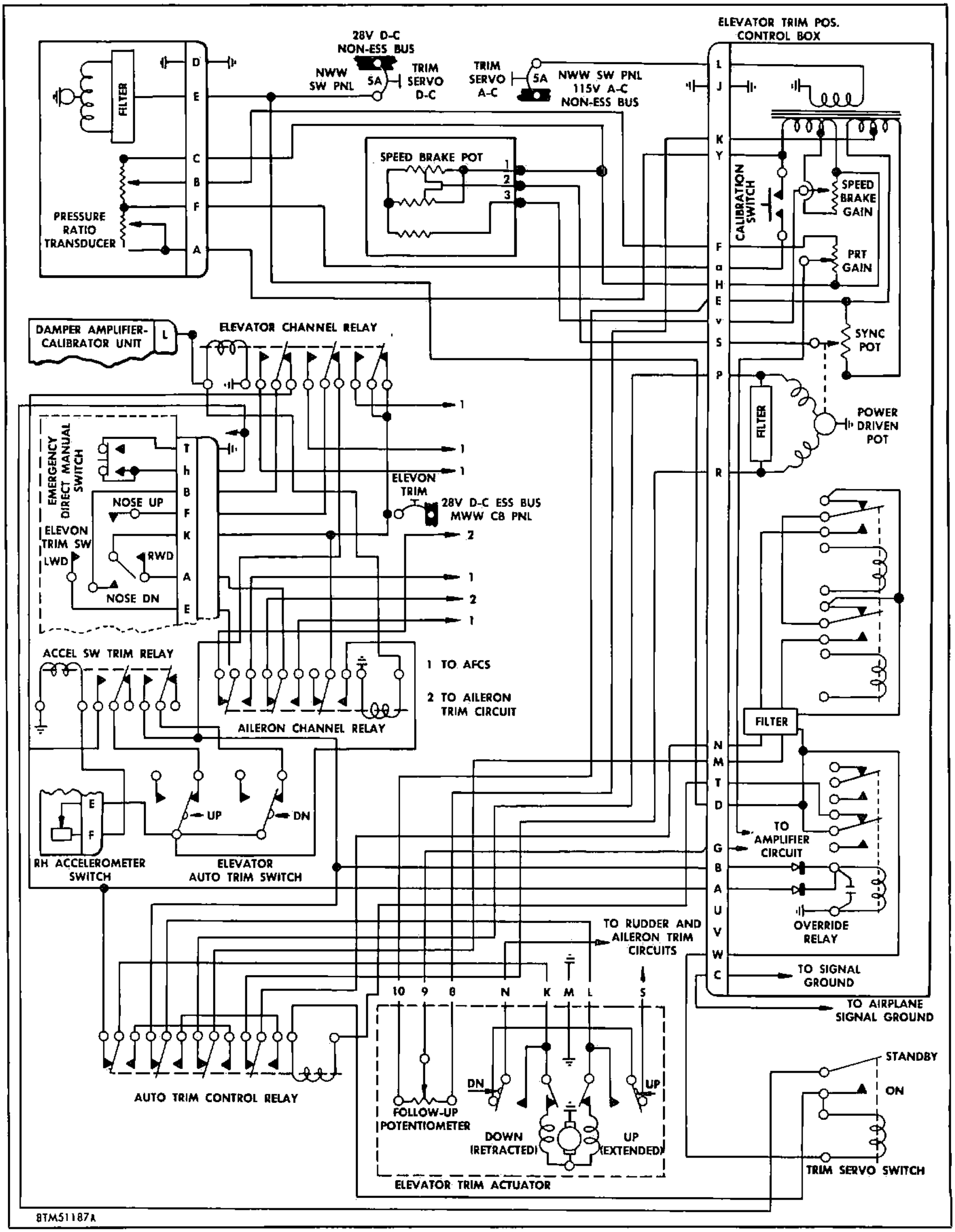

Figure 2-25. Elevator Trim Servo Circuit Schematic

takeoff. A procedure for adjustment of the internal limit switches is outlined in your F-102A maintenance handbook, T.O. 1F-102A-2-7.

ELEVATOR TRIM SERVO SYSTEM.

Before we discuss the trim servo system it may be well to have an understanding of the purpose of the system. A peculiarity of the F-102A airplane is that, as the airplane accelerates or decelerates through the high subsonic speed range, a condition occurs called "stick lightening." In most airplanes, in order to maintain level flight with an increase in airspeed, the pilot must hold a forward force on the stick or increase the trim of the airplane in the nose-down direction by use of the trim controls. With a decrease in airspeed, he must normally hold a back-stick pressure or trim nose up to maintain level flight. The F-102A, however, does not respond in this manner to an increase or decrease in airspeed in the high subsonic or supersonic speed range. Through a portion of the speed range, an increase in airspeed results in an increased need for up elevator trim. As a result of this peculiarity, a device called the *Trim Servo System* is installed to compensate for this tendency.

The elevator trim servo system, therefore, provides the means of automatically changing elevator trim to compensate for speed and altitude changes and speed brake extension, in addition to a normal elevator trim function. The trim servo system consists of a pressure ratio transducer, an altitude compensating unit, a servo amplifier, a circuit to the elevator trim actuator which incorporates a feedback potentiometer, and a synchronization circuit.

Operation Of The Trim Servo System.

The engagement of the elevator trim servo system is controlled by a solenoid held switch on the utility panel. The switch is placarded STANDBY and ON. When the servo system is not engaged, 28-volt elevator trim signals are routed to the elevator trim actuator when the pilot places the trim switch on the control stick in the NOSE UP or NOSE DOWN positions. When the trim servo system is engaged, electrical signals proportional to airspeed and altitude are routed from the pressure ratio transducer to a servo amplifier in the elevator trim position control box. Another signal, proportional to speed brake extension, is routed from the speed brake potentiometer to the servo amplifier in the control box. These error signals are summed and amplified, and then routed to the elevator trim actuator, causing the actuator to move. A feedback signal from the trim actuator potentiometer cancels out the error signal when the actuator reaches the correct position and the actuator movement stops.

As you remember, the actuator moves the mixer assembly in a forward or aft direction; this movement in turn is transmitted through the linkage to the hydraulic control valve, which then meters hydraulic fluid to the elevon actuators to position the elevons for the desired elevator trim. If the pilot desires additional trim, he may again actuate the trim switch on the control stick. Actuation of the trim switch by the pilot energizes an override relay in the control box to disconnect power from the auto-trim control relay.

The auto-trim relay re-routes power directly to the trim actuator to provide elevator trim. The auto-trim relay also connects the error signals coming from the control box to a power-driven potentiometer in the control box. Motion of the trim actuator changes the feedback signal from the actuator potentiometer and thereby upsets the balance of the error signal. This unbalance of the error signal drives the power driven potentiometer to again cancel out the error signal, thereby zeroing the trim servo system at the trim set by the pilot. When the trim switch is released, the servo system resumes command about this new point of reference.

Takeoff Trim Circuit.

The aileron, elevator, and rudder trim systems are interconnected through the takeoff trim circuit to provide a means of automatically trimming the airplane for takeoff. The takeoff trim circuit is shown schematically on figure 2-26. The TAKEOFF TRIM switch is located on the utility switch panel. When the switch is actuated, 28-volt, d-c essential power drives the aileron and rudder trim actuators to a neutral position (if the actuators are not in a neutral position) and the elevator trim actuator to a five-degree *up* elevator position. The takeoff trim circuit is routed through the nose-wheel steering circuit breaker panel, through the nose landing gear *up* position switch, to the aileron trim actuator.

When the aileron trim actuator reaches the neutral position, the neutral position switches route power to the elevator trim actuator. When the elevator actuator reaches the five-degree *up* elevator position, the position switches route power to the rudder trim actuator. Then, when the rudder trim actuator reaches the neutral position, the internal switches route power to the takeoff trim indicator light to indicate that the control surfaces are trimmed at the takoff position. When the takeoff switch is released, the light will go out. Since the power is routed through the nose landing gear *up* position switch, the takeoff trim system is inoperative when the airplane is airborne.

MAINTENANCE OF THE ELEVON CONTROL SYSTEM.

Maintenance problems of various parts of the elevon control system were discussed throughout the chapter. You learned that the best way to handle a maintenance

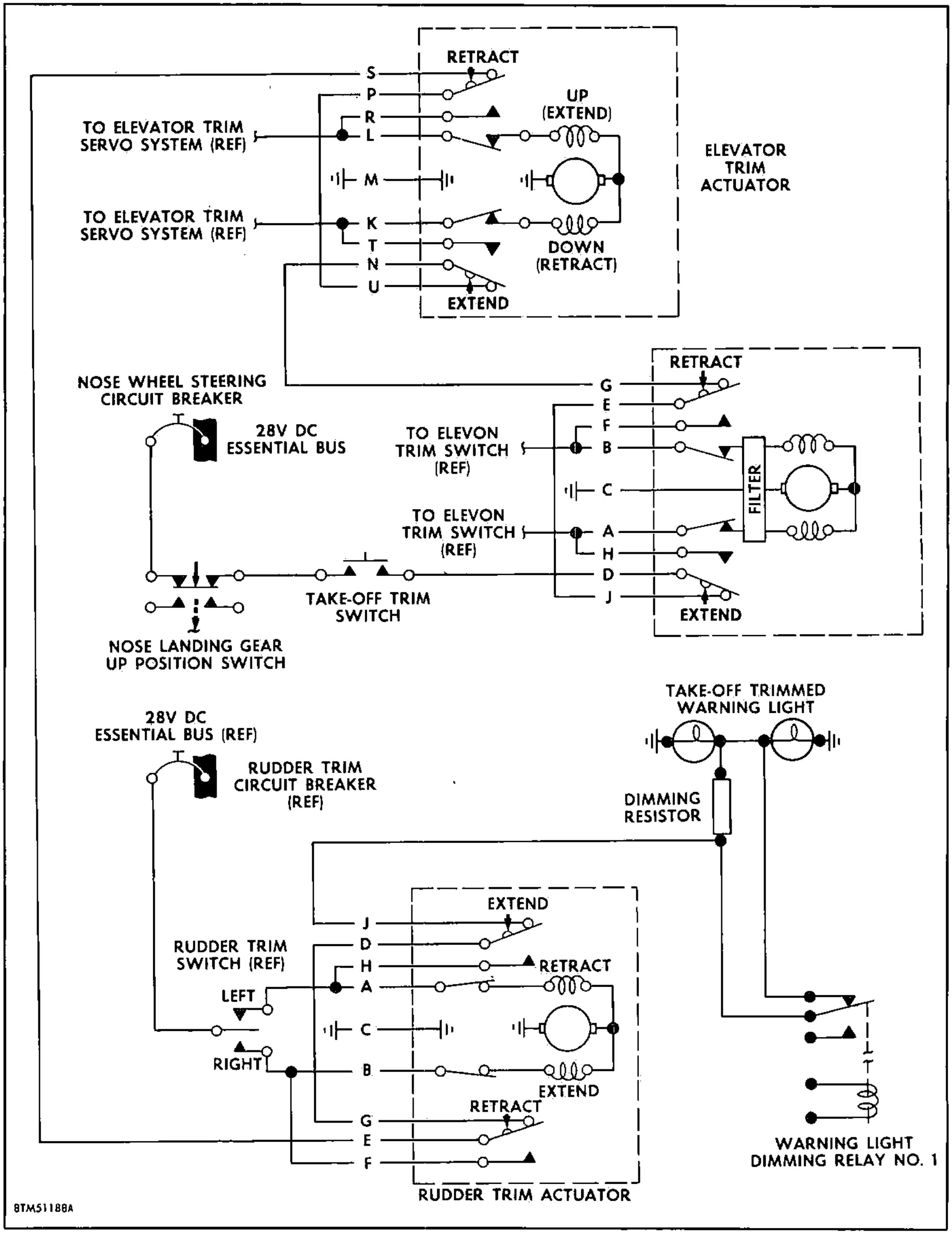

Figure 2-26. Takeoff Trim Circuit Schematic

problem is to refer to the trouble shooting tabular lists in your F-102A maintenance handbook, T.O. 1F-102A-2-7. These lists outline the probable cause of a malfunction and corrective measures to be taken. The above mentioned handbook also establishes procedures for removal, installation, and adjustment of the control system components, in addition to information concerning ground servicing and lubrication of the control system.

Trouble shooting should begin at the power systems. Make sure that the control systems are receiving power from the separate sources involved before trouble shooting the control systems or their components. Also, before replacing a component that is suspected of malfunctioning, make sure that the component is receiving power.

OPERATIONAL CHECKOUT AND TESTING.

To maintain the operating efficiency of the F-102A flight control system and to insure that all parts of the system are functioning within specified limits, you will be required to perform operational checkout and testing of the system and its components. Operational checkouts are performed after reinstallation or replacement of any component, after a malfunction of the system has been corrected, and at established periodic intervals. If any malfunctions are encountered during the operational check of the system refer to the Trouble Shooting tabular list as explained above. Your F-102A maintenance handbook also outlines in a step-by-step fashion, a procedure established for performing the operational check and testing.

Before performing the operational check on the flight control system, certain preparations must be made, safety precautions must be taken, and certain conditions must be observed. These steps are discussed in the following paragraphs.

Pre-Check Preparations.

Preparations that you will be required to perform before an operational check of the control system include a visual inspection of the system for obvious errors in installation, clearances, cracks, distortion, and security; providing a portable hydraulic test stand and connecting the stand to the primary and secondary hydraulic systems of the airplane; connecting external electrical power to the airplane electrical system; and providing a controllable source of dry, filtered air or nitrogen.

Safety Precautions.

It will be your responsibilty to see that all areas adjacent to movable control surfaces and components are cleared of all objects and personnel before performing the operational check. Also, station personnel at switches and controls that will be powered when you connect external electrical power or hydraulic power to the airplane; this will prevent inadvertent actuation of the system and its components that would endanger personnel or damage equipment.

Conditions To Be Observed.

As you will recall, a portable hydraulic test stand is required to perform an operational check of the control system. When this test stand is used, it must be connected to both the primary and secondary hydraulic systems. If this is not done, pressure in the secondary system alone will cause the pressurized hydraulic fluid to bleed through interconnecting components to the primary system.

Before performing an operational check of the control system, be sure that the trim systems are properly rigged. An improperly rigged trim system will adversely affect the proper functioning of the control system. In other words, an operational check of the trim systems must be performed before an operational check of the control system.

REPLACING THE ELEVONS.

Each elevon, as you have learned, is attached to the left and right delta wing by seven hinge bolts. Access doors are provided at the hinge points on the lower surface of each elevon, above the hinge centerline on the fuselage to provide access to the elevon actuator and to the control valve actuating rods, and on the inboard upper and lower elevon surface to provide access to the elevon horn attach bolts. The outboard elevon actuators are enclosed with a removable fairing to provide access to the actuators. A bonding jumper is installed at each elevon hinge assembly and is detached from the elevon hinge when removing the elevon. Elevon removal and installation procedures are outlined in a step-by-step fashion in T.O. 1F-102A-2-7, your maintenance handbook.

During installation of the elevon and at established time intervals, lubrication of elevon components is necessary; also, elevon attach and linkage bolts must be torqued to specified values. Refer to the above mentioned handbook for components that require lubrication, the type of lubricant specified, the time interval when lubrication is required, and for specified bolt torquing values. Figure 2-27 shows the left elevon and its attachment points. The right elevon is similar.

RIGGING THE ELEVON CONTROL SYSTEM.

The basic requirement of the rigging procedure is to maintain the mechanical and hydraulic components in a neutral position while rigging the system. The neutral position of the system and its components is maintained by inserting rigging pins in the rigging pin holes provided in the system components and the airplane structure. The rigging pin holes in the system components and the airplane structure, when properly matched and locked in this position with the

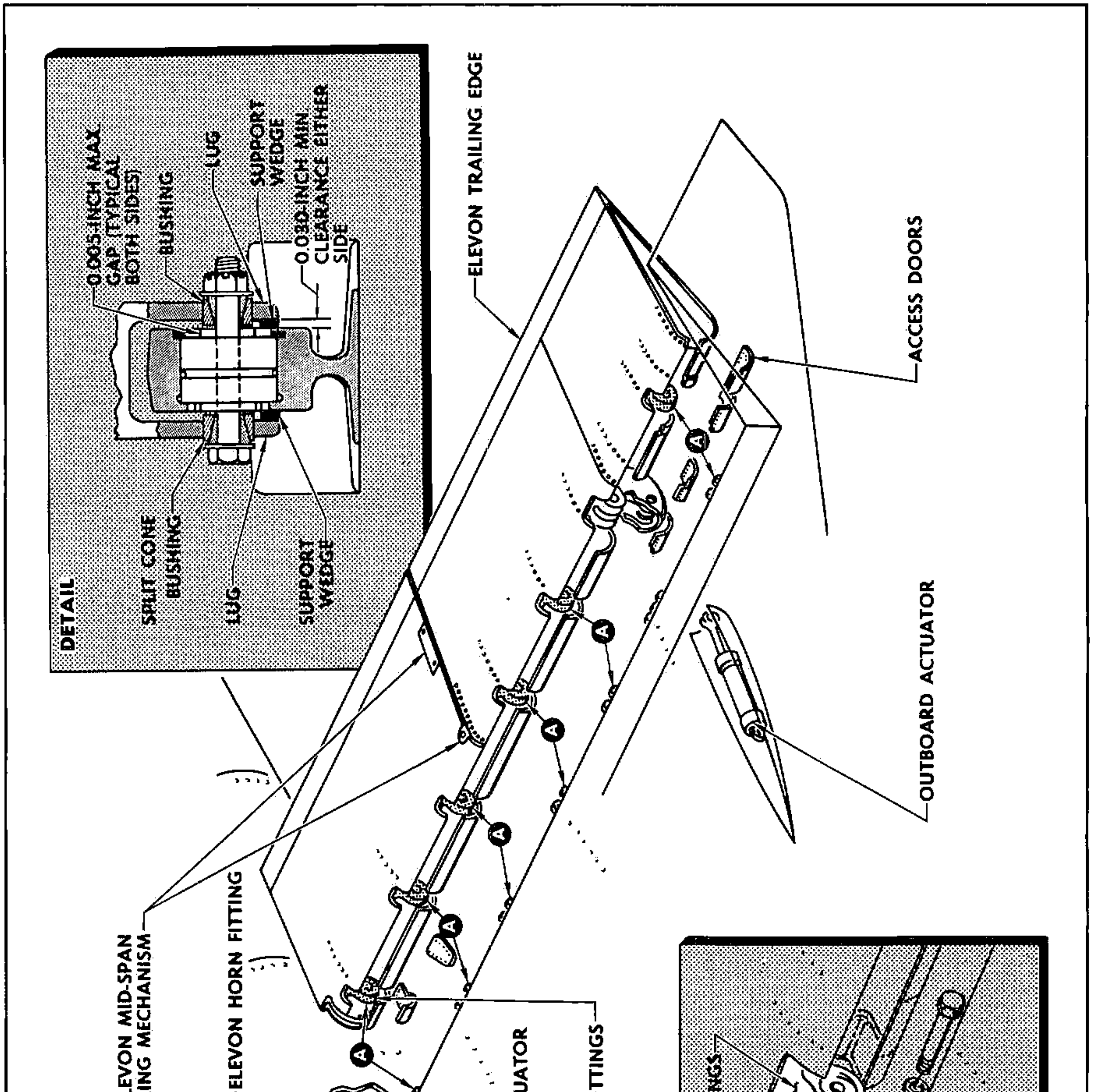

Figure 2-27. Elevon Installation

rigging pins, maintain the system in neutral. For convenience, the elevon control system rigging is broken down into three parts: First, elevator system rigging which includes the forward portion of the control system concerned with elevator action, the elevator portion of the mixer assembly, and the elevator trim and feel system. Second, aileron system rigging which includes the forward portion of the controls system concerned with aileron action, the aileron portion of the mixer assembly, and the aileron feel and trim system. And third, the aft portion of the system in which aileron and elevator action are combined. Portions of the system between points fixed by rigging pins or rigging fixtures may be rigged independently of the rest of the system. However, care must be taken to insure that elevator and aileron neutrals are established where this is applicable.

Rigging of the elevon control system should be performed whenever a component has been removed from the system and replaced or reinstalled. After rigging of the system is complete an operational check must be performed to insure that all parts of the system are functioning within specified limits. Before performing an operational check of the elevon control system, be sure that the trim systems are properly rigged. An improperly rigged trim system will adversely affect the proper functioning of the elevons.

Before rigging the control system, make sure that hydraulic pressure is relieved from the hydraulic system and the accumulators. No specific rigging procedures will be given in this training supplement. Refer to your F-102A maintenance handbook, where you will find the rigging procedure outlined in a step-by-step fashion.

SPECIAL TORQUE VALUES.

All bolts that connect the various linkages and secure the components in the control system must be torqued to specified values. Torque values, as you probably know, are the measurement (in inch-pounds or foot-pounds) of how much torque can be applied to a wrench to tighten a bolt or other type of fastener. The force applied is dependent on the diameter of the bolt and the material of which the bolt is made. Special-type wrenches called "torque wrenches" are used to apply this measured force. One type of torque wrench has a dial indicator to measure the amount of torque applied; there are other types also. Torque values are established to prevent undertorquing or overtorquing a fastener. Too tight is as bad as not tight enough. If you overtorque a bolt for example, you stretch the bolt beyond its elastic limit. The bolt either breaks or weakens to the point where it is not doing its job. Overtorquing can also distort the material under the bolt head or nut. No torque values will be given in this training supplement. Refer to your F-102A maintenance handbook for the specified torque values.

LUBRICATION OF THE SYSTEM.

Various components of the control system require lubrication during installation and at other specified intervals. Refer to your applicable F-102A maintenance handbook for these lubrication requirements. Lubrication charts in the handbook specify the type of lubricant to be used and the time intervals at which the various components of the control system require lubrication. The charts also indicate, by visual reference, the points to be lubricated and any special type equipment that will be needed.

EMERGENCY OPERATION OF FLIGHT CONTROL SYSTEM.

There are no emergency provisions directly incorporated within the flight control system. However, there are indirect emergency systems governing the power sources. As you remember, hydraulic power is supplied to the control system through two completely independent systems, the primary and secondary hydraulic systems. Maneuverability, however, is reduced when operating on only one system; the Automatic Flight Control System and the pitch and yaw damper systems will not function if the secondary hydraulic system becomes inoperative. An emergency hydraulic system is provided to supply power to the primary system in the event of an engine pump failure.

The electrical power systems provide two sources of emergency electrical power. An emergency a-c generator supplies power in the event the normal a-c generator fails, while the battery supplies d-c power in the event of a d-c generator failure. Since the AFCS and the pitch and yaw damper systems receive power from nonessential buses, which are disconnected in emergency operation, the systems are automatically disengaged during an electrical power failure.

SUMMARY.

The first chapter of this supplement was concerned with the basic theory of flight controls and a general description of the F-102A flight control system. You were made aware of the differences that exist between a more conventional-type of airplane and the F-102A airplane.

In this chapter you learned about the elevon control system and how it operates, and also how the elevons function as both ailerons and elevators. We also discussed the mechanical and hydraulic components and their function within the control system, the artificial feel system, and the trim system. The last part of the chapter was devoted to a discussion of maintenance procedures, including operational checkout and testing, and rigging of the elevon control system. In the following chapter, you will learn about the rudder control system on the F-102A airplane.

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Chapter III

THE RUDDER CONTROL SYSTEM

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In the preceding chapter you learned about the elevon control system and how the elevons combine elevator and aileron motion in a single pair of elevons to control the airplane about its longitudinal (roll) and lateral (pitch) axes. You also learned that it would be extremely difficult for a pilot to manually operate the control surfaces of a high-speed, supersonic airplane without the aid of some power generating device. High forces imposed on the control surfaces, generated by high speed, make it necessary that the control surfaces be moved by a powered force. This force is provided by hydraulic actuators to move the elevons and the rudder on the F-102A airplane.

In this chapter you will learn about the rudder control system and how it operates, the mechanical and hydraulic components and their functions within the system, the artificial feel system, and the trim system. The last part of the chapter will be devoted to a discussion of operational checkout, maintenance, and rigging of the rudder control system. The pitch and yaw damper systems and the automatic flight control system, as applied to rudder control, are discussed only briefly in this chapter; however, the systems are discussed fully in following chapters.

REVIEW OF FLIGHT CONTROL SYSTEM.

Before getting under way with a discussion of the rudder control system it may be well to have a brief review of the flight control system on the F-102A airplane. This can serve two purposes; first, refresh your memory about what has been said before, and second, bring to your attention again the various components that make up the flight control system.

The flight control system on the F-102A airplane is a full hydraulic power system in which all control surface motion, movement of elevons and rudder, is accomplished by hydraulic actuating cylinders. Hydraulic power to operate the actuating cylinders is provided by the primary and secondary hydraulic systems. Each actuating cylinder, two on each elevon and one on the rudder, has two hydraulic chambers and two pistons operating one ram shaft so that either one or both of the two separate hydraulic systems can operate the flight control surfaces. In normal operation both hydraulic systems are used at the same time. The two systems provide a safety factor in the event that one of the hydraulic systems should fail.

There are three modes of operation that control the movement of the rudder and elevons—*Direct Manual,* which permits pilot movement of the conventional control stick and rudder pedals; *Manual,* which includes automatic damper signals to the hydraulic servo actuators in addition to pilot control; and *Automatic Flight Control,* which utilizes a fully automatic flight system that also operates through the servo actuators.

The elevons and rudder on the F-102A airplane do not have conventional trim tabs. Normal trimming is obtained by movement of the whole control surface. An electrically actuated trim system is incorporated in the mechanical linkage so that airplane trim is superimposed on the pilot's control—in other words, the trim set into the control system is maintained regardless of pilot control movement. The elevon trimming action is controlled by a five-position toggle switch on the control stick. The rudder trimming action is controlled by a three-position toggle switch on the utility switch panel at the pilot's pedestal.

An artificial feel system is also incorporated in the mechanical linkage of the control system. This artificial feel system is necessary since the airloads imposed on the control surfaces cannot be felt through the hydraulic actuating cylinders that move the control surfaces. The artificial feel is put into the flight control system by the feel cylinders in the mechanical linkage. Air that enters through the "Q" (ram air) intakes on the vertical stabilizer is amplified by pneumatic system air pressure through the variable air pressure regulators to control air pressure to the feel cylinders. The system resists movement of the controls with a force that is relative to airplane speed and altitude to provide a feel in the control system. The faster the airplane flies the more tension the feel cylinder puts into the control system.

DESCRIPTION OF RUDDER CONTROL SYSTEM.

The components and linkage of the rudder control system are shown in the schematic diagram on figure 3-1. The rudder control system, like the elevon control system is hydraulically powered. Note that the conventional rudder pedal mechanism incorporates a pilot-to-pedal length adjustment, brake control cylinder linkage, and nose wheel steering linkage. A system of bell cranks and cables transmit pilot pedal motion aft to the vertical torque tube in the fin island below the vertical fin. Rotation of the vertical torque tube actuates linkage to open the hydraulic control valve; the control valve then ports hydraulic fluid through drilled passages to the actuating cylinder to move the rudder.

In the Manual and Automatic Flight Control modes of control, the servo actuator responds to signals from the yaw damper system to dampen the pilot's control movements and also to initiate appropriate rudder movement to coordinate turns. The servo actuator, control valve, actuating cylinder, and the followup mechanism are mounted as one assembly in the fin island. The piston end of the actuating cylinder is attached to the airplane structure and the cylinder fixed end is attached to the rudder horn. The followup loop, which establishes a correlation between pedal position and rudder position, makes this arrangement necessary. Therefore, the cylinder piston is immovable and the cylinder body moves to displace the rudder.

As you have already learned, an artificial feel system is incorporated in the mechanical linkage of the rudder control system. The function of the feel system is to provide the pilot wth feel forces at the rudder pedals. Since the hydraulic action of the flight control system is not reflected to the mechanical control linkage, airloads on the control surfaces are not transmitted back to the pilot. Consequently, an artificial feel system is required to simulate an airload condition. The components that make up the feel system include a feel cylinder, a variable air pressure regulator, and the "Q" (ram air) pressure system. The feel system is discussed in detail later in the chapter.

The control surfaces on the F-102A airplane do not have conventional trim tabs. Trimming of the airplane is accomplished by electrical actuators that move the entire elevon and rudder surface for trim action. A three-position toggle switch on the utility switch panel routes 28-volt, d-c power to the rudder trim actuator to move the rudder to the left or right. When the trim switch returns to the neutral position actuator movement stops. Trim motion is introduced into the rudder control system through the mechanical linkage. The trim system is also discussed in detail later in the chapter.

MODES OF FLIGHT.

There are three modes, or ways, available to the pilot for controlling the flight attitude of the airplane. These are the Direct Manual, Manual, and Automatic Flight Control modes which are discussed separately below.

Direct Manual.

A two-position flight mode switch is located on the flight mode panel. When the switch is placed in the DIRECT MANUAL position, the pilot has direct control of the airplane and only his control motions result in a change of course or attitude. The Direct Manual mode of control is normally used for takeoffs and landings, and any time the pilot desires to have only control stick, rudder pedal, and trim switch movements determine displacement of the control surfaces.

Manual.

The Manual mode is the method of flight control in which the pilot's control movements are dampened to some degree by counteracting signals from the damping system through the servo actuators. The servo actuators receive signals from the damping system

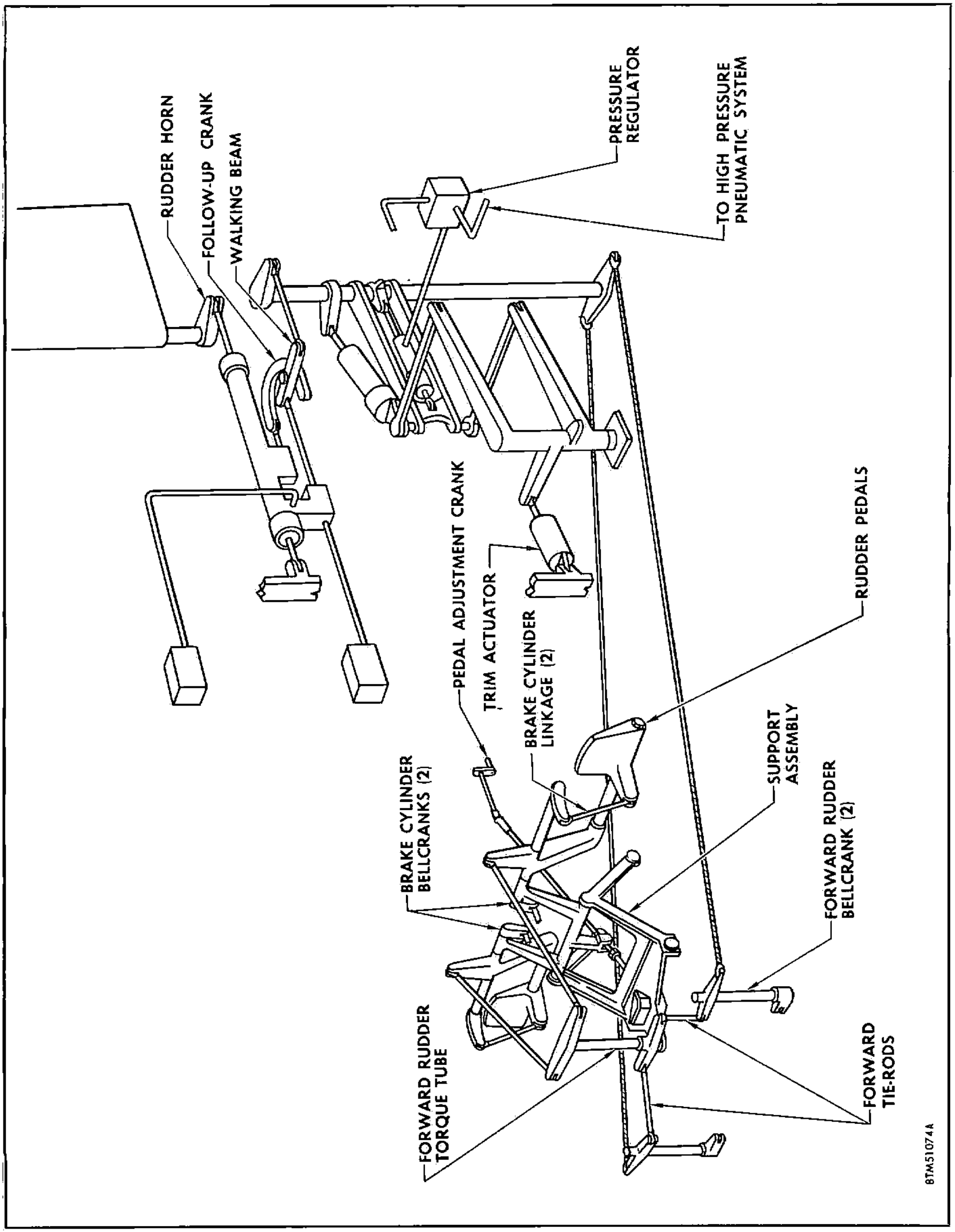

Figure 3-1. Rudder Control System Schematic

and convert them to control motion. Placing the flight mode switch in the MANUAL position completes a circuit that sends 28-volt, d-c current to energize the elevon and rudder actuator shutoff valves. When these valves are energized, they route hydraulic pressure to the elevon and rudder servo actuators and also to a solenoid valve in the elevon manual control lockout system. The elevon and rudder servo actuators are connected by linkage to their respective hydraulic control valves to superimpose damping impulses on the control surfaces. The Manual mode of control is selected by the pilot to provide a stable armament platform or when the airspeed is above Mach 0.75.

Automatic Flight Controls.

As you will recall, the Automatic Flight Control Subsystem (AFCS) is a part of the MG-10 Aircraft and Weapon Control System and governs the airplane flight control system when it is in the Automatic Flight Control mode. In this mode, the AFCS has "full authority" or complete control of the elevons. This complete control is effected by a lockout valve which prevents pilot control of the elevons.

The rudder hydraulic system does not incorporate a lockout valve as does the elevon hydraulic system. Therefore, the rudder is always at the command of the pilot regardless of the mode of flight. The rudder servo actuator receives signals from the damping system and initiates appropriate rudder movement to coordinate a turn. The essential components for turn coordination are the roll rate gyro, a high-pass network and filter, the aileron position potentiometer, and the airspeed compensator. These components originate the signals to the damper system for turn coordination.

MECHANICAL LINKAGE.

The following illustrations (figure 3-2 through 3-6) show the arrangement and identify the mechanical system components. For ease of discussion we will divide the system into four groups. These groups comprise the forward linkage (including the rudder pedals), the aft linkage housed in the fin island below the vertical fin, the control cables which connect the forward and aft linkage, and the rudder actuating cylinder assembly which includes the yaw damper servo actuator, the hydraulic control valve, and the follow-up mechanism. We will also discuss at this time maintenance problems or maintenance procedures concerned with each group or individual components of the mechanical system.

Forward Linkage.

A conventional rudder pedal assembly, fork assembly, forward rudder torque tube, tie rods, and bell cranks, comprise the forward rudder control linkage. The rudder pedals also incorporate a pedal length adjustment mechanism. As you can see in figure 3-2, the nose wheel steering and brake control linkage is also actuated by the rudder pedals. These two systems are discussed in another supplement of this training series.

RUDDER PEDAL ASSEMBLY. The main support for the rudder pedal assembly and the forward torque tube is a one-piece magnesium alloy casting. The casting is attached to floor rails by four bolts, two on the left and two on the right side. Note that the fork assembly is attached to this main support casting with one long bolt on which it is free to rotate for adjustment of the pedal length.

The inboard and outboard brake bell crank and the rudder pedal hangar bell crank, on the left and right sides, are attached to the fork assembly upper arms. A shaft supports bearings on which the bell cranks rotate. Each outboard brake bell crank is linked to a crank on the rudder pedal by a control rod and each inboard crank is linked to its brake cylinder by linkage. The inboard brake bell crank is attached to the outboard brake bell crank axle with two pins installed at 90 degrees to each other. This arrangement also secures the pedal hangar bell cranks to the fork assembly.

The rudder pedals are attached to the lower end of the hangar arm bell crank by the rudder pedal shaft extending through the hangar bell crank and locked to the bell crank with two pins installed at 90 degrees to each other. The pedals rotate on the shaft for brake action on the inboard and outboard bearing. The outboard bearing is locked to the shaft with a retaining ring.

Periodic lubrication of the rudder pedal assembly is required for proper functioning of the assembly. Periodic lubrication is required at the bearing in the pedal adjust linkage at the fork assembly, the inboard and outboard bearings in each pedal, and the inboard and outboard bearings on each brake bell crank. Check your F-102A maintenance handbook for the time interval and the type of lubricant to use.

FORWARD TORQUE TUBE. The forward torque tube assembly includes the torque tube and the upper and lower bell cranks to which the tube is attached. Pins installed at 90 degrees to each other secure the torque tube to the bell cranks. The lower bell crank is supported by the one-piece magnesium alloy support casting and the upper bell crank by a brace attached to the airplane structure. A single bolt, screwed into a single nutplate at the center of each bell crank, secures each bell crank to its upper and lower support. Thus the bell crank is free to rotate. Adjustable stops are installed on the lower bell crank to limit rudder pedal travel. After the stops on the aft rudder torque tube at the feel cylinder crank have been set to allow for specified rudder travel, the forward torque tube bell crank stops are set. The forward stops are set to

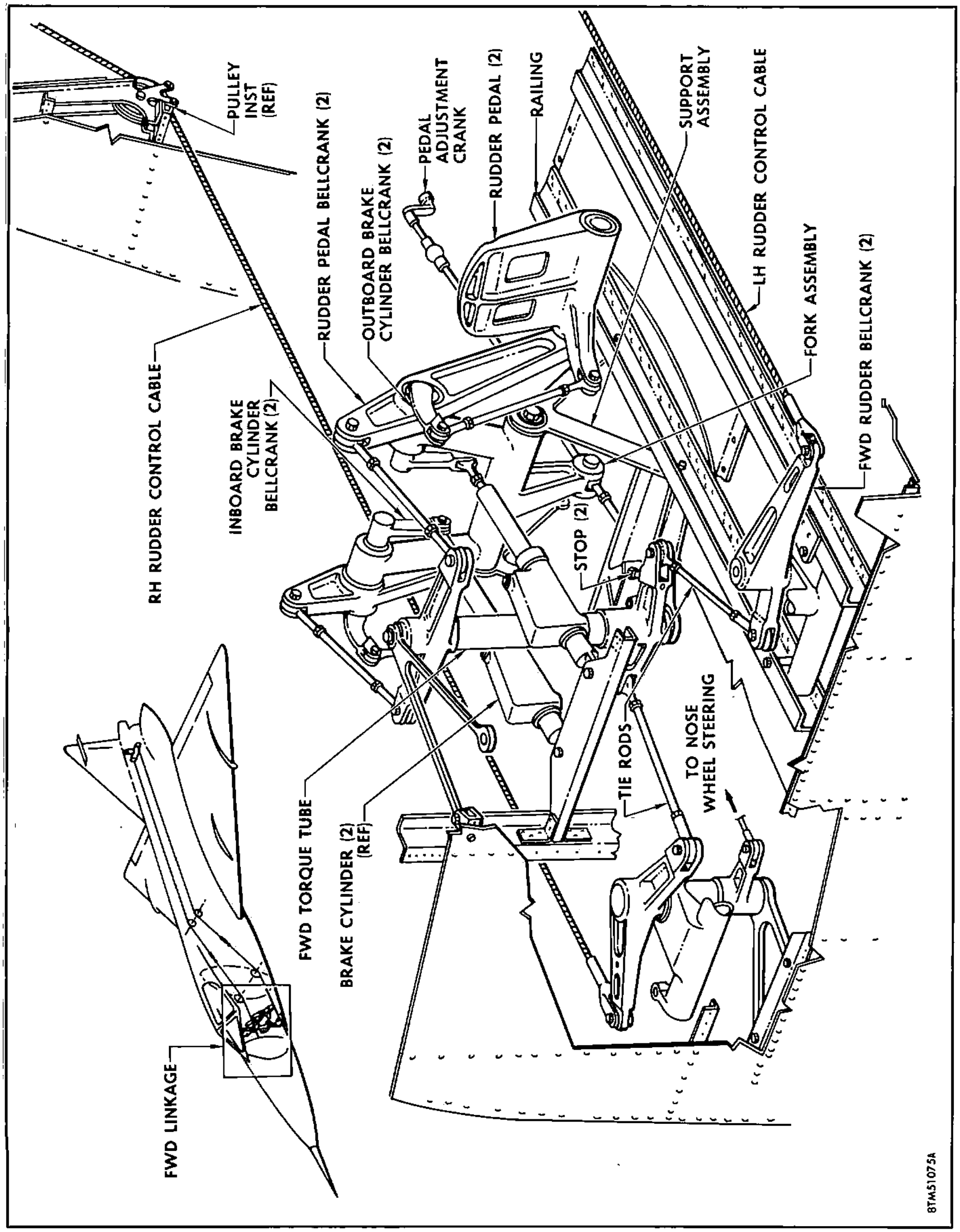

Figure 3-2. Rudder Forward Control Linkage

allow a specified gap between the stops and the bearing surface. Check your applicable F-102A maintenance handbook, T.O. 1F-102A-2-7, for these settings.

The upper bell crank on the forward torque tube assembly is connected to the rudder pedal bell cranks with control rods and the lower bell crank is connected to the left and right control cable bell cranks with control rods. Displacement of the rudder pedals by the pilot actuates the pedal bell crank, which transfers the movement to the forward torque tube, which in turn transfers the movement to the control cable bell cranks to move the control cables. Movement of the control cables rotates the vertical torque tube in the tail section, which actuates linkage to open the hydraulic control valve. The valve then ports hydraulic pressure to the actuating cylinder to displace the rudder.

If you are called upon to install the forward torque tube, always check for freedom of movement in the torque tube assembly mounts before attaching the rudder pedal and push-pull linkage. Also, the roll pins that attach various components of the forward linkage must extend through the outer attach holes a specified minimum distance. Check your F-102A maintenance handbook for this specified minimum distance.

Aft Linkage.

Figure 3-3 shows the arrangement and identifies the components in the aft rudder control linkage. The aft linkage and components are housed in the fin island below the vertical fin and includes a vertical torque tube assembly to which is attached a rudder control crank, rudder centering crank, and the feel cylinder crank. Other components are the walking beam, linked to the rudder control crank by a control rod at its outboard end and to the shaft of the damper servo actuator at the other end; the followup crank, attached to the center of the walking beam at one end and to the hydraulic control valve at the other end; and the rudder feel and centering support bracket which is attached to, but rotates freely on, the rudder torque tube. The rudder trim actuator crank is attached to the free end of the centering and feel support bracket through linkage.

The yaw damper servo actuator, rudder hydraulic control valve, rudder actuating cylinder, and the followup mechanism are mounted as one assembly. Note that the piston end of the rudder actuator cylinder is attached to the airplane structure and the cylinder stationary end to the rudder control horn. In this case the cylinder end and not the piston moves the rudder. The reason for this arrangement is because of the design of the followup loop mechanism, which establishes a correlation between rudder pedal and rudder position; or in other words, the pilot's pedal motion opens the hydraulic control valve to move the rudder to a position determined by the displacement of the pedals, while the followup mechanism closes the control valve when the rudder reaches this position. The feel cylinder, the centering spring, and the trim actuator are also incorporated in the mechanical linkage. These components will be discussed individually later in the chapter.

SERVO ACTUATOR LINKAGE. In the Direct Manual mode of control, the servo actuator piston shaft is locked in a rigid position by its centering springs and serves only as a pivot point in the mechanical linkage. In the Manual and Automatic Flight Control modes of control, the shaft extends or retracts in response to electrical signals from the yaw damper system to superimpose the desired damping motions on pilot mechanical control movements to the rudder. The end of the servo actuator piston shaft is attached to and acts as a pivot point for the inboard end of the walking beam. Therefore, movement of the damper servo actuator accomplishes the addition to, or subtraction from, pilot input.

Also, it can be seen that, if the servo actuator shaft extends at the same time and equally with a left rudder pilot input (link between torque tube bell crank and outboard end of walking beam moving forward), the walking beam merely rotates about the center attachment to the followup crank and the damper servo actuator has subtracted total pilot input. In this manner, the servo actuator adds to and subtracts from pilot input to correct overcontrol movements of the pilot and also to initiate appropriate rudder movement for turn coordination.

The servo actuator shutoff valve is a solenoid-actuated hydraulic valve, which is energized by the damper engage circuit. When the valve is energized, it opens and admits secondary hydraulic system pressure to the servo actuator, which then responds to signals from the damper servo amplifier. The amplifier is ON at all times when electrical power is available and the circuit breakers are engaged.

The servo actuator shaft motion will rotate the walking beam about the outboard end of the beam when the pilot input is zero. This action in turn will result in rotation of the attach point of the followup bell crank to open the hydraulic control valve. As you remember from the preceding discussion, when the rudder moves in response to the servo input, the mechanical followup loop closes the control valve. This rudder motion is not transmitted back to the rudder pedals since the torque tube does not move. In fact, positive back-up from the centering spring is necessary to insure faithful rudder response to damper commands. Since the damper servo inputs result in pivoting the walking beam about its outboard end, any play or slop in the linkages from the walking beam to and including the trim actuator results in the

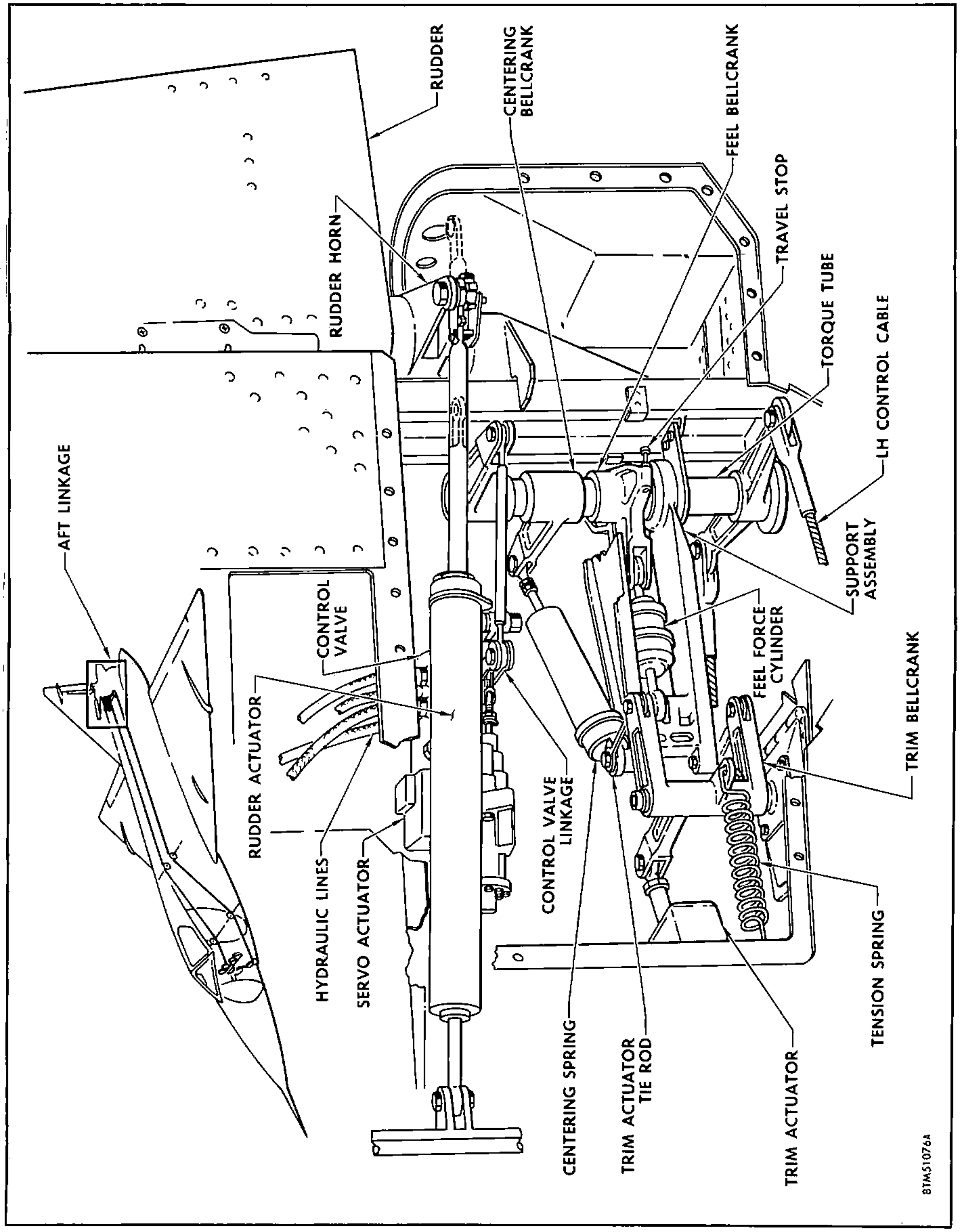

Figure 3-3. Rudder Control Aft Linkage

servo shaft motion being partially lost (in the slop) rather than being transmitted to the hydraulic control valve. This results in a phase lag and under-response of the rudder to damper commands. Therefore, as you can see, it is very important in the maintenance of the system to insure minimum or complete lack of play in these linkages.

Control Cables.

The control cables, which connect the forward and aft rudder control linkage, are routed from the forward left and right rudder cable bell cranks aft, through pulleys along the upper fuselage to the vertical torque tube bell crank in the tail section. Note on figure 3-4 that each left and right cable assembly consists of two cables joined together with turnbuckles. The terminal ends of the cables are attached to the forward cable bell cranks and the aft torque tube bell cranks with bolts and self-locking nuts.

CABLE TENSION. As you know, metals contract and expand when exposed to changes in temperature. For this reason control cable tension must be set at a value (in pounds) established for the degree of temperature existing at the time the cables are being installed or reinstalled in the airplane. The diameter of the control cable must also be considered in establishing this value. These values have been predetermined and can be found by referring to the cable tension chart in the Flight Controls maintenance handbook, T.O. 1F-102A-2-7. The T.O. also outlines, in a step-by-step fashion, a procedure for rigging the control cables.

LUBRICATION OF CONTROL CABLES. A protective coating of corrosion-resistant compound for the full length of the cables and also lubrication of cables in the area of pulleys is required at periodic intervals. Again, refer to the above mentioned handbook for the time intervals and the type of compound and lubricant used on the cables.

Rudder.

The rudder is attached to the rear spar of the vertical stabilizer with two hinge bolts and to the airplane structure at the lower support fitting with 10 bolts. The support fitting acts as a third hinge point when attached to the airplane structure. A bearing, held in the rudder support fitting with lock rings, allows the rudder to rotate within the fitting. Three bonding jumpers join the rudder to the rear spar of the vertical stabilizer. The rudder horn is a part of the lower support fitting and connects the rudder to the hydraulic actuator end fitting to move the rudder when the cylinder is actuated. The rudder is of aluminum honeycomb construction. The procedure for replacing the rudder is discussed at the end of this chapter and is illustrated on figure 3-15.

RUDDER CONTROL SYSTEM OPERATION.

As you remember, the rudder control system receives hydraulic power from the primary and secondary hydraulic systems to supply power to the hydraulic control valve. Displacement of the rudder pedals by the pilot transfers the pedal motion through control linkage to the aft vertical torque tube. Rotation of the torque tube then actuates linkage to open the hydraulic control valve, which then ports hydraulic fluid to the rudder actuating cylinder to move the rudder. The followup mechanism returns the control valve to neutral at which point movement of the rudder stops. When the pilot releases his applied force from the rudder pedals the centering spring returns the rudder to its original trim-neutral position.

To give you a complete understanding of what actually takes place, we will go through a complete cycle of operation in the Direct Manual mode of operation. Direct Manual is normally used for takeoff and landings, and anytime the pilot desires to have only control stick, rudder pedal, and trim switch movements determine displacement of the control surface. The Manual and Automatic Flight Control modes of control will then be discussed as they apply to rudder control operation.

OPERATION IN THE DIRECT MANUAL MODE.

As you recall, the Direct Manual mode is that method of flight control whereby the pilot has direct control of the airplane and only his control motions result in a change of course or attitude. We will assume that the pilot applies a force to the left rudder pedal, moving it forward. This force initiates a forward movement of the left rudder control cable and a clockwise rotation (viewed looking down) of the rudder torque tube in the tail section, to which the cable is attached. This rotation of the torque tube results in a forward movement of the link from the crank at the top of the torque tube to the outboard end of the walking beam. The other end of the walking beam is attached to the shaft of the damper servo actuator. Since we are operating in the Direct Manual mode of control, the shaft of the servo actuator is held at its center position by its centering springs. This results in the walking beam pivoting about the servo actuator attachment point.

In the Direct Manual mode of operation the servo actuator is rigid because the servo actuator shutoff valve is closed. Since it is not functioning hydraulically, the servo actuator acts merely as part of the straight shaft, or rigid link, between the mechanical bell crank linkage and the control valve. You will note by referring to figure 3-5 that the followup bell crank is attached to the center of the walking beam and pivots about an attachment to the rudder actuator cylinder body. The remaining end, which is an offset

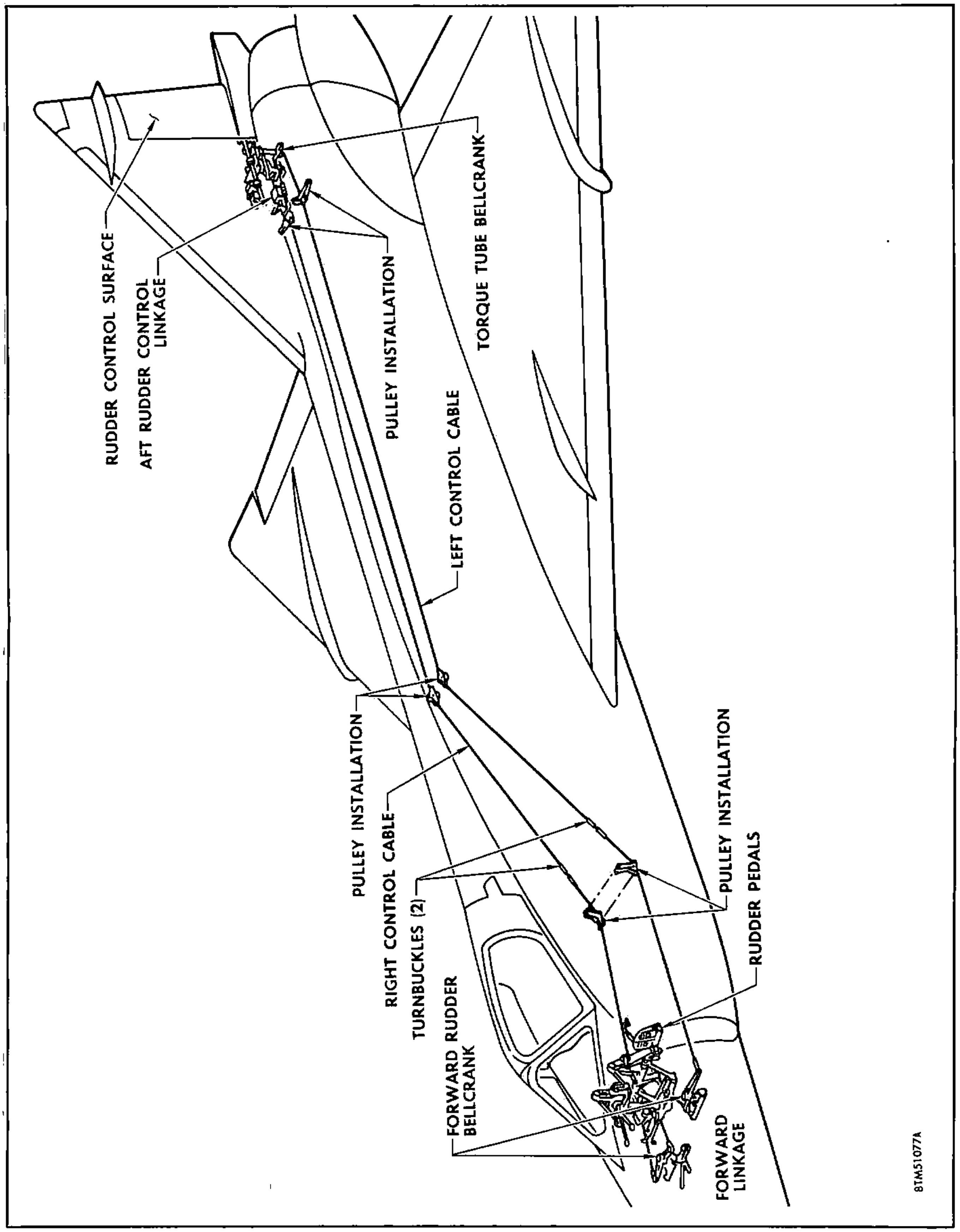

Figure 3-4. Rudder Control Cables Installation

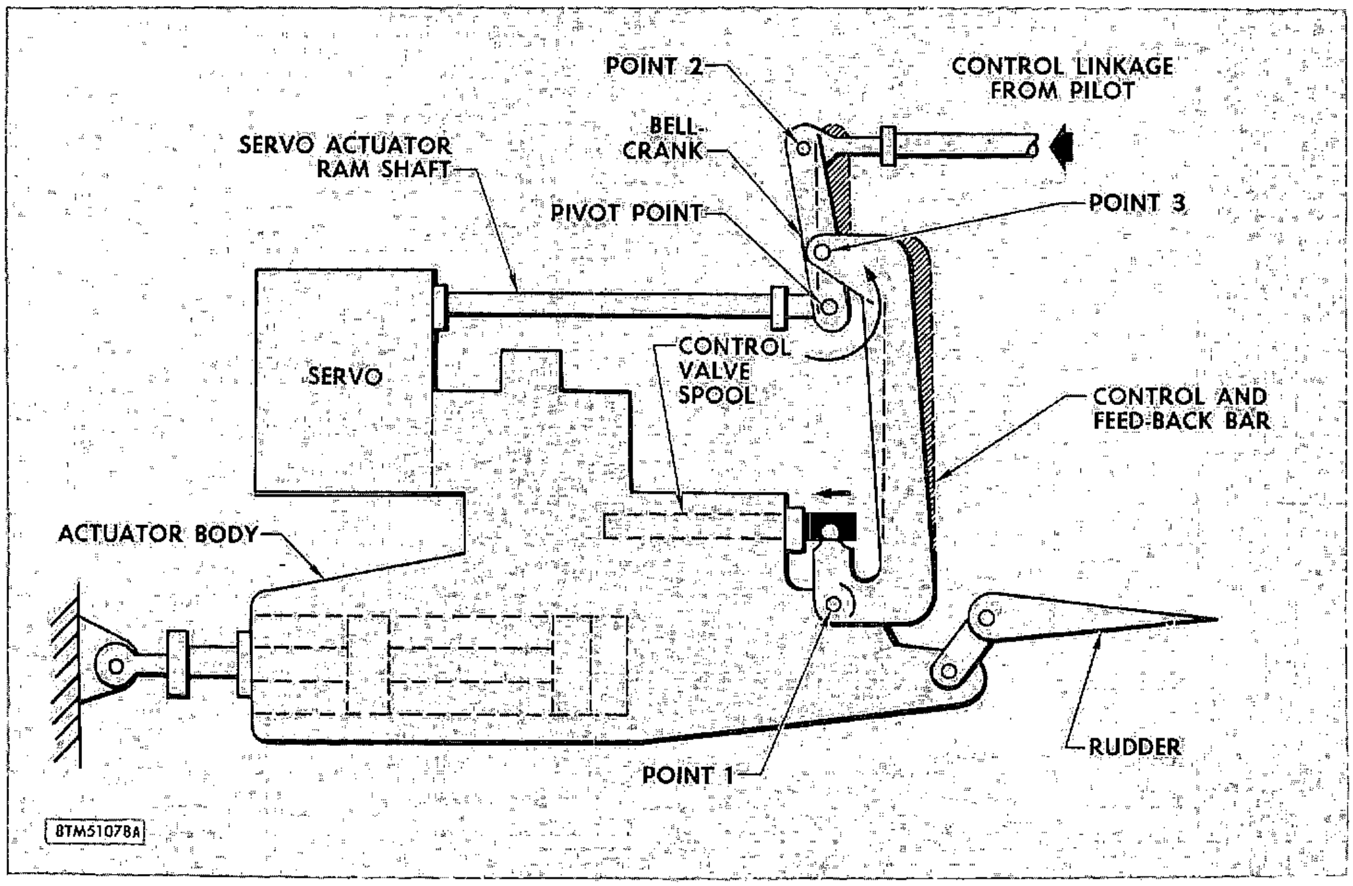

Figure 3-5. Rudder Mechanical Linkage Action, Initial Motion

of the pivot, is attached by linkage to the control valve spool. Now as the walking beam is pivoting about the servo actuator attachment, motion of the walking beam is transmitted to the followup bell crank. As a result, the linkage from the followup bell crank to the control valve moves aft, opening the control valve. The control valve then ports pressurized hydraulic fluid to the actuating cylinder in such a manner that the cylinder retracts and the rudder moves left.

As the actuating cylinder moves forward, the motion causes movement of the followup crank which results in returning the control valve to neutral; at this point the movement of the rudder is stopped. Refer to figure 3-6 to follow this action. During this forward movement of the actuating cylinder, the outboard end of the walking beam is held as a fixed point since the pilot is holding a definite rudder pedal position. Since the damper servo actuator is attached to the rudder actuator cylinder and moves with it, the end of the walking beam attached to the servo actuator shaft also moves forward and the walking beam pivots about its outboard attachment. The followup bell crank is then rotated by the walking beam about its pivot point on the actuating cylinder assembly. As you can see in figure 3-5, this pivot point is attached to the cylinder assembly and moves with it. The rotation of the followup bell crank with respect to the attach point is counterclockwise as viewed in the illustration. This rotation, which is in the opposite direction to the original rotation of the followup crank, moves the control valve toward neutral.

The control valve is again at neutral when the actuating cylinder has moved through the distance as determined by the dimensions of the followup mechanism. As a result, the rudder moves a distance which is proportional to the movement of the rudder pedals. When the pilot releases his applied force from the rudder pedal, the feel cylinder and centering spring return the rudder torque tube to its original position.

The reverse sequence of events in the followup mechanism returns the rudder to its original position. The net result is that rudder position is determined by the angular position of the rudder torque tube. The one exception is when rudder motion results from damper servo action inputs. In this case the rudder torque tube and the rudder pedals do not move.

You will note by referring to figure 3-3 that the rudder feel cylinder crank and the rudder centering crank are attached to the rudder control torque tube. Also attached to the tube, but rotating freely, is the rudder

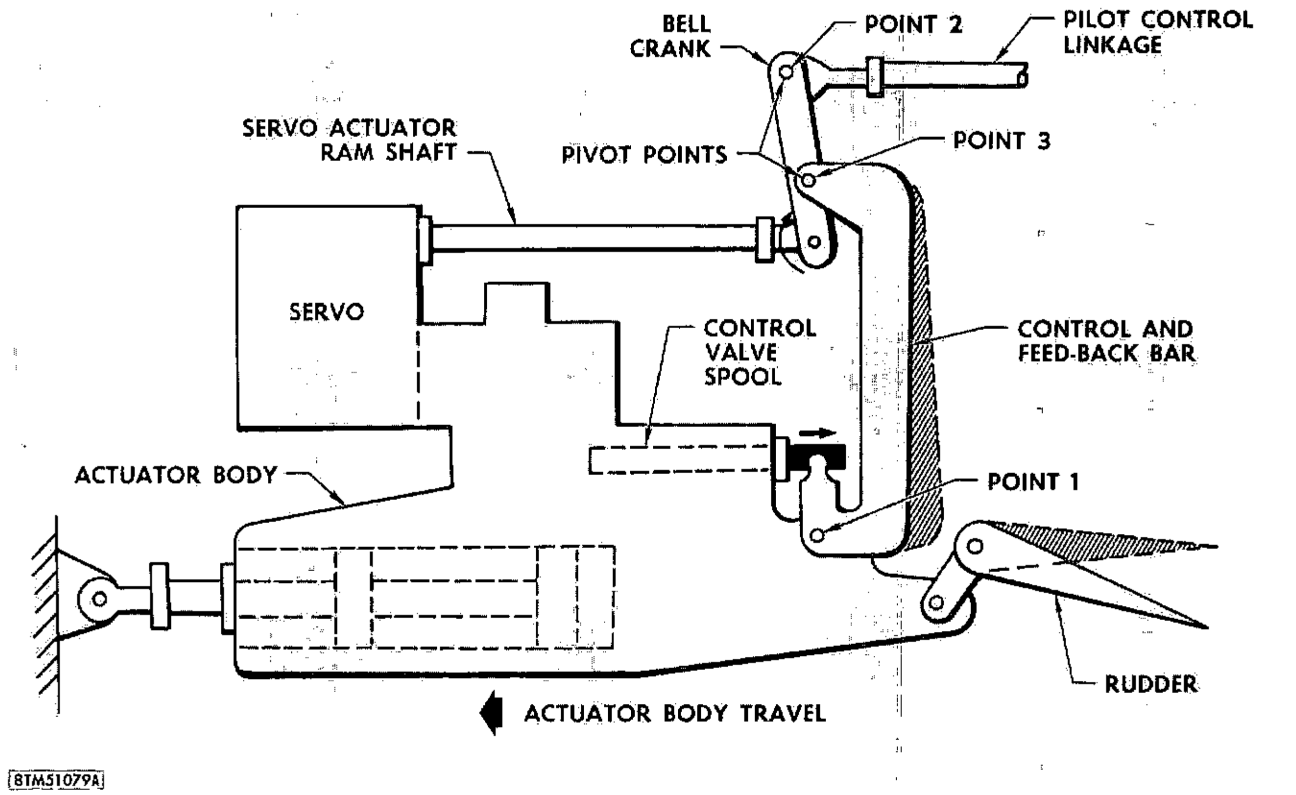

Figure 3-6. Rudder Mechanical Linkage Action, Followup Motion

feel and centering support. The trim actuator is connected to the free end of the rudder feel and centering support through linkage. Trim motion is introduced in the system through the rudder trim bell crank to the free end of the rudder feel and centering support when the trim switch is engaged. This causes the rudder feel and centering support to rotate in the direction of trim. The trim force operates against the centering spring to move the control surface. The motion that results from the combination of movement of the rudder pedals and the trim introduced in the system is transmitted to the rudder actuating cylinder control valve by the rudder torque tube bell crank.

OPERATION IN THE MANUAL MODE.

As you recall, the Manual mode is that method of flight control whereby the pilot's control movements are dampened to some degree by counteracting signals from the damping system through the servo actuators. The yaw rate gyro originates the signal to the yaw damper system to provide a directionally stable airplane.

The rudder servo actuator is hydraulically energized through the servo actuator shutoff valve when the pilot selects the Manual mode of operation. When energized, it opens and admits secondary hydraulic system pressure to the servo actuator, which then responds to signals from the yaw damping system amplifier. However, the pilot may override or boost rudder motion while the servo actuator is in service.

As far as the mechanical control system is concerned, it is necessary only to know that the servo actuator piston extends or retracts in response to electrical signals from the yaw damper servo amplifier. The end of the servo actuator piston ram is attached to and acts as a pivot point for the inboard end of the walking beam. As the servo actuator piston extends, the piston ram moves the walking beam to pivot it off the pilot-held control rod. The walking beam then moves the attached followup crank, which action in turn opens the control valve. The valve then ports hydraulic fluid to the rudder actuating cylinder to move the rudder. A feedback potentiometer is connected to the servo actuator ram. As the servo actuator ram moves in response to servo action, the potentiometer feeds back a signal of opposite polarity to cancel the original signal and stop the action.

When the two signals are completely balanced, the servo actuator piston is returned to neutral by the mechanical followup linkage. Consequently, the control valve also has returned to neutral by the reverse action of the linkage. However, the rudder actuator has been locked in the new position called for by the original input signal to the servo actuator. This rudder movement as a result of servo actuator action is not transmitted back to the rudder pedals, since the torque tube does not move. The servo actuator internal mechanism and how it works will be discussed under the rudder hydraulic system, farther along in this chapter.

OPERATION IN THE AUTOMATIC FLIGHT CONTROL MODE.

In the Automatic Flight Control mode, the AFCS will maintain the attitude of the airplane at the instant of engagement if engaged within attitude limits of the system. If engaged in a climb, the AFCS will maintain the climb attitude of the airplane. If engaged in a turn, that attitude is maintained. Means to vary the stabilized attitude of the airplane after the engagement of the Automatic Flight Control mode are provided by a trim switch. The trim switch in the Automatic Flight Control mode now operates through the AFCS components instead of the electrical trim actuators to effect trim changes, and is called the "Beep" control.

The rudder control system does not incorporate a lockout valve as does the elevon control system. Therefore, the rudder is always at the command of the pilot. As you recall, in the Automatic Flight Control mode, when the AFCS is engaged, it completes a circuit which supplies 28-volt, d-c power to open the solenoid valve at the elevon lockout valve. Opening the solenoid valve permits hydraulic pressure to actuate the lockout valve, thereby "locking out" the mechanical followup loop and pilot input control movements to the elevons. The AFCS signals are sent to the elevon servo actuators through the damper system for control of elevon movement. The mechanical followup loop in the elevon system is replaced by an electrical feedback system. This feedback system moves the control valve, thereby stopping elevon movement when the desired displacement has been reached.

In the Automatic Flight Control mode, as in the Manual mode of control, the rudder servo actuator receives signals from the yaw damping system to dampen pilot's control movements and to initiate appropriate rudder movement to coordinate a turn. The essential components for turn coordination are the roll rate gyro, a high-pass network and filter, the aileron position potentiometer, and the airspeed compensator. These components originate the signals to the yaw damper system for turn coordination.

A rolling acceleration, as you know, is produced by movement of the ailerons. Therefore, the aileron displacement and roll rate act to move the rudder to minimize the yawing produced. The aileron position potentiometer and the roll rate gyro originates the signals to the yaw damper system which initiates appropriate rudder movement to coordinate the turn. When the aileron action returns to neutral, as the desired roll attitude is approached, the aileron movement reduces to zero. However, rolling velocity due to momentum, must then be coordinated. This is accomplished by the lagging roll rate network which produces a delayed signal to coordinate the roll.

Since the amount of rudder required to coordinate a turn is less at high speeds than at low speeds, the airspeed compensator reduces the strength of the signal from the roll rate gyro and the aileron potentiometer as the airspeed of the airplane increases.

In order to coordinate turns in the speed range where opposite rudder is required, a signal termed "minus aileron signal" is taken from the aileron position potentiometer and scheduled by the airspeed compensator in such a way that the strength of the signal increases with increasing airspeed. The minus aileron signal becomes larger than the combined aileron and roll rate signals. This is done so that an opposite rudder signal is available for turn coordination, as is the case when the yaw is in the direction of the turn.

The rudder servo actuator piston extends or retracts in response to the turn coordinator signals through the yaw damper system to coordinate a turn with appropriate rudder movement. The rudder mechanical linkage functions in the same way as in the Manual mode of rudder control. A more thorough discussion of the damper systems and the Automatic Flight Control mode will be found in Chapters IV and V of this supplement.

F-102A FLIGHT CONTROL HYDRAULIC SYSTEM.

Full hydraulic power is supplied to the right and left flight control systems from the primary and the secondary hydraulic power supply systems. The primary system supplies hydraulic power by way of a control valve, to dual chamber actuating cylinders in each flight control system. The secondary hydraulic system not only supplies power to its half of the same dual chamber cylinders, but also supplies hydraulic power to other hydraulic components of each control system. If one of the two hydraulic power supply systems becomes inoperative, the remaining system will supply the required hydraulic power for continued operation of the flight controls. Control is slower, however, if only one system is operating.

RUDDER CONTROL HYDRAULIC SYSTEM.

The components of the hydraulic portion of the rudder flight control system are shown in the schematic diagram on figure 3-7. The components include the rudder actuating cylinder and control valve, the rudder servo actuator, and the servo actuator shutoff valve.

The rudder actuating cylinder and control valve are incorporated within the same casting with interconnecting drilled passages between the two. Note also that the drilled passages route the flow of pressurized hydraulic fluid to the appropriate sides of the actuating cylinder pistons. Following the flow of fluid in the schematic diagram, we see that the primary hydraulic system fluid supply serves only the rudder actuating cylinder; the secondary hydraulic system fluid supply serves not only its chamber of the actuating cylinder but also the servo actuator. The rudder servo actuator is attached to the rudder actuating cylinder with bolts. The rudder control hydraulic system does not incorporate a lockout valve as does the elevon control system; therefore, the rudder is always at the command of the pilot regardless of the mode of flight.

HYDRAULIC SYSTEM OPERATION.

Pilot displacement of the rudder pedals is transmitted by mechanical linkage to open the hydraulic control valve. The valve then meters pressurized hydraulic fluid through drilled passages to the rudder actuating cylinder to move the surface in the direction of the pilot's control movement. A mechanical feedback linkage between the rudder control surface and the hydraluic control valve shuts off the valve when the control surface reaches a position corresponding to the rudder pedal position.

The servo actuator shutoff valve supplies hydraulic pressure to the servo actuator when the damper system is engaged. The servo actuator shaft extends or retracts in response to electrical signals from the yaw damper servo amplifier to superimpose the desired damping motions over pilot control motion through the hydraulic control valve. (The yaw damper servo amplifier is discussed in detail in the next chapter.)

The servo actuator shutoff valve is a solenoid-actuated hydraulic valve which is energized by the damper engage circuit. When energized, it opens and admits secondary hydraulic system pressure to the servo actuator, which then responds to signals from the damper servo amplifier. Mechanical features of the damper servo actuator and shutoff valve were discussed earlier in this chapter.

How Hydraulic Power Moves the Rudder.

The direction in which the hydraulic control valve spool is moved determines which side of the rudder actuator pistons receives hydraulic pressure. Unlike the elevon actuators, the rudder actuator is of balanced design, so that there is equal piston area throughout. Therefore, equal reaction to control forces is obtained in either direction. The piston rod of the rudder actuator is attached to the airplane structure and the cylinder body is attached to, and moves with, the rudder horn (arm). When the hydraulic force is applied to the actuator, the actuator cylinder extends or retracts along its own ram shaft causing the actuator cylinder fixed end to move the rudder.

Rudder Actuator and Control Valve.

The control valve cylinder bore is separated into two sections. One section controls flow of primary hydraulic system fluid and the other section controls flow of secondary hydraulic system fluid to the corresponding tandem pistons of the rudder actuator. Note on the rudder hydraulic schematic diagram (figure 3-7) that the control valve is also separated into two sections in tandem; this arrangement allows extension of one double spool through both sections of the valve cylinder. To move the rudder requires fore and aft movement of the spool in the cylinders. This routes hydraulic pressure to the rudder actuating cylinder by way of the drilled passages mentioned above. The mechanical features and operation of the mechanical linkage of the rudder control system were discussed earlier in this chapter.

Servo Actuator and Shutoff Valve.

As you remember, we discussed the mechanical control of the servo actuator earlier in the chapter and learned that it is used in the Manual mode of rudder control wherein the pilot's control movements are dampened to some degree by counteracting signals from the damping system through the servo actuator. Let us now discuss the servo actuator in more detail to learn how and what it does.

The servo actuator is bolted to the rudder actuating cylinder casting and is controlled by a solenoid-operated, three-way, shutoff valve. The valve is located between the servo actuator and the secondary hydraulic system pressure line to the control valve. The servo actuator is comprised of spring-loaded pistons, fluid flow passages, flow passage restrictors, and an electrical torque motor. When the pilot selects the Manual mode of operation, the servo actuator is hydraulically energized through the shutoff valve and can then respond to electrical signals from the damper system. The signals position the servo actuator piston to pivot the rudder control bell crank (walking beam) around the pilot-held control rod as shown on figure 3-8. This action in turn moves the attached followup bell crank to operate the control valve. The control valve then meters pressurized hydraulic fluid to the rudder actuator in amounts indicated by the signal received to dampen the pilot's rudder control movements.

To see how the servo actuator accomplishes this, assume that the damper system sends an electrical signal to the servo actuator torque motor. The torque motor then moves the flapper, shown on figure 3-9, toward one nozzle or the other in an amount depending on

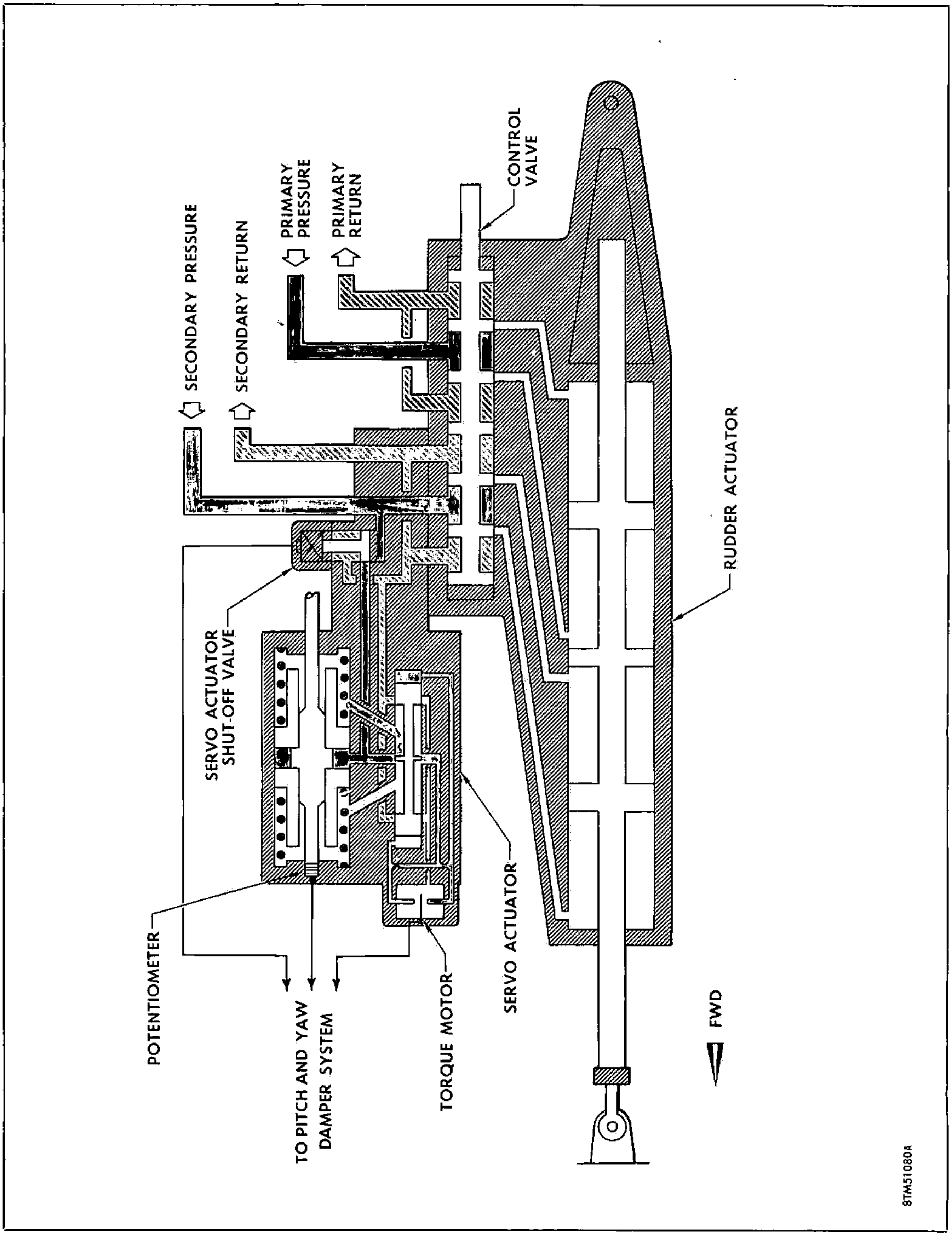

Figure 3-7. Rudder Hydraulic System Schematic

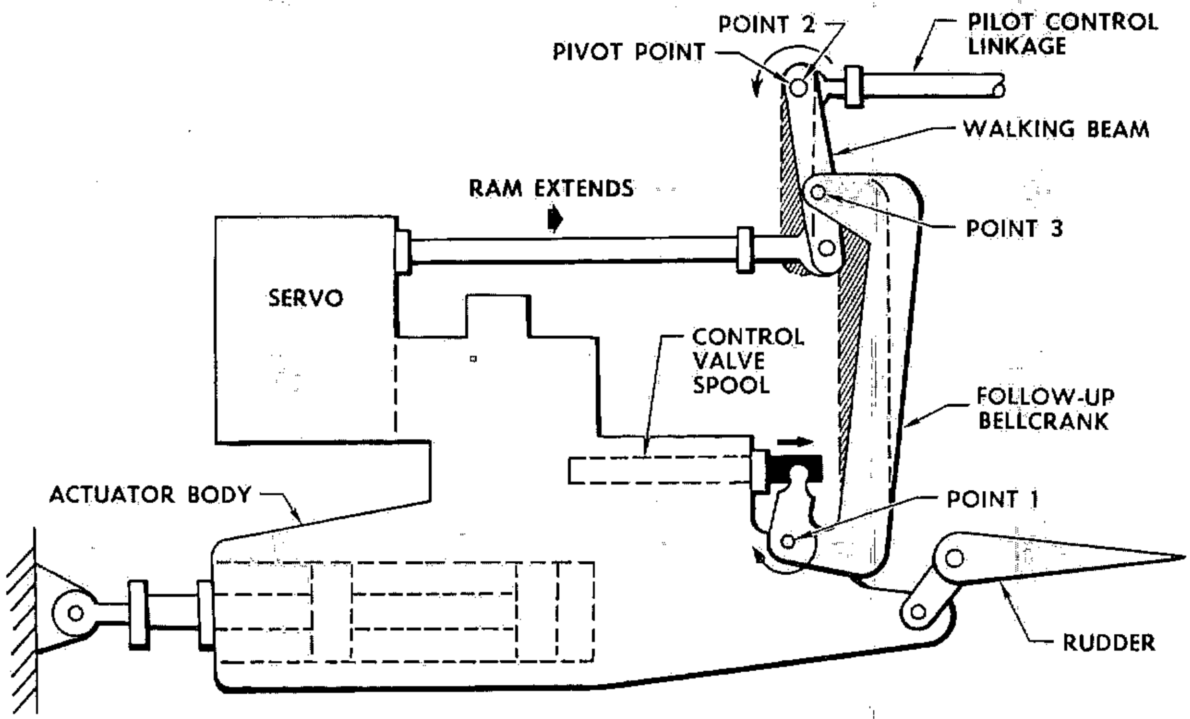

Figure 3-8. Rudder Mechanical Linkage Action, Initial Motion in Manual Mode

the intensity of the signal. As the flapper moves towards a nozzle, a fluid pressure unbalance is set up in the servo actuator. This is a result of the movement of the flapper toward one nozzle which causes restricted flow through that nozzle, and at the same time, in moving away from the other nozzle, allows more fluid to flow through it. Pressure in the passage is then increased between the restricted nozzle and one end of a control spool in the servo because of the restriction. This pressure unbalance causes the servo actuator control spool to move in the desired direction to port fluid pressure to the correct side of the servo actuator ram shaft piston, and port fluid from the other side of the piston to return. Consequent motion of the servo actuator ram pivots the walking beam and the followup bell crank to position the control valve spool away from neutral.

The control valve repositioning admits fluid pressure to, and ports return fluid from, the appropriate sides of the rudder actuating cylinder piston. The piston ram of the rudder actuator then moves the rudder to the new position. As the rudder actuator piston and ram move, a feedback potentiometer connected to the ram sends another electrical signal, the voltage of which is determined by the amount of ram displacement, to the torque motor. The second signal is of opposite electrical sign to the first signal, and tends to null it out. This second signal thereby rapidly reduces the current of the first signal to the torque motor to zero.

As the two signals are balancing, they are moving the flapper towards neutral. When the two signals are completely balanced, the flapper has returned to neutral, again equalizing the fluid pressure. The equalized pressure thus causes the servo actuator pistons to also return to neutral. Consequently, the control valve has been returned to neutral by reverse action of the linkage, but the rudder actuator has been locked in the new position called for by the original input signal to the torque motor.

The effect on the mechanical linkage of the above action is indicated by arrows on figure 3-10. As you can see, the push rod from the pilot's control linkage does not move as the walking beam rotates around the pivot point. The servo ram retracts to draw the other end of the walking beam back to its original position. This causes the followup bell crank also to return to its original position. Of course, as the actuating cylinder assembly moves in response to the hydraulic actuating force the cylinder will also contribute to the return action through the movement of the followup bell crank. The rudder stays in the new position that

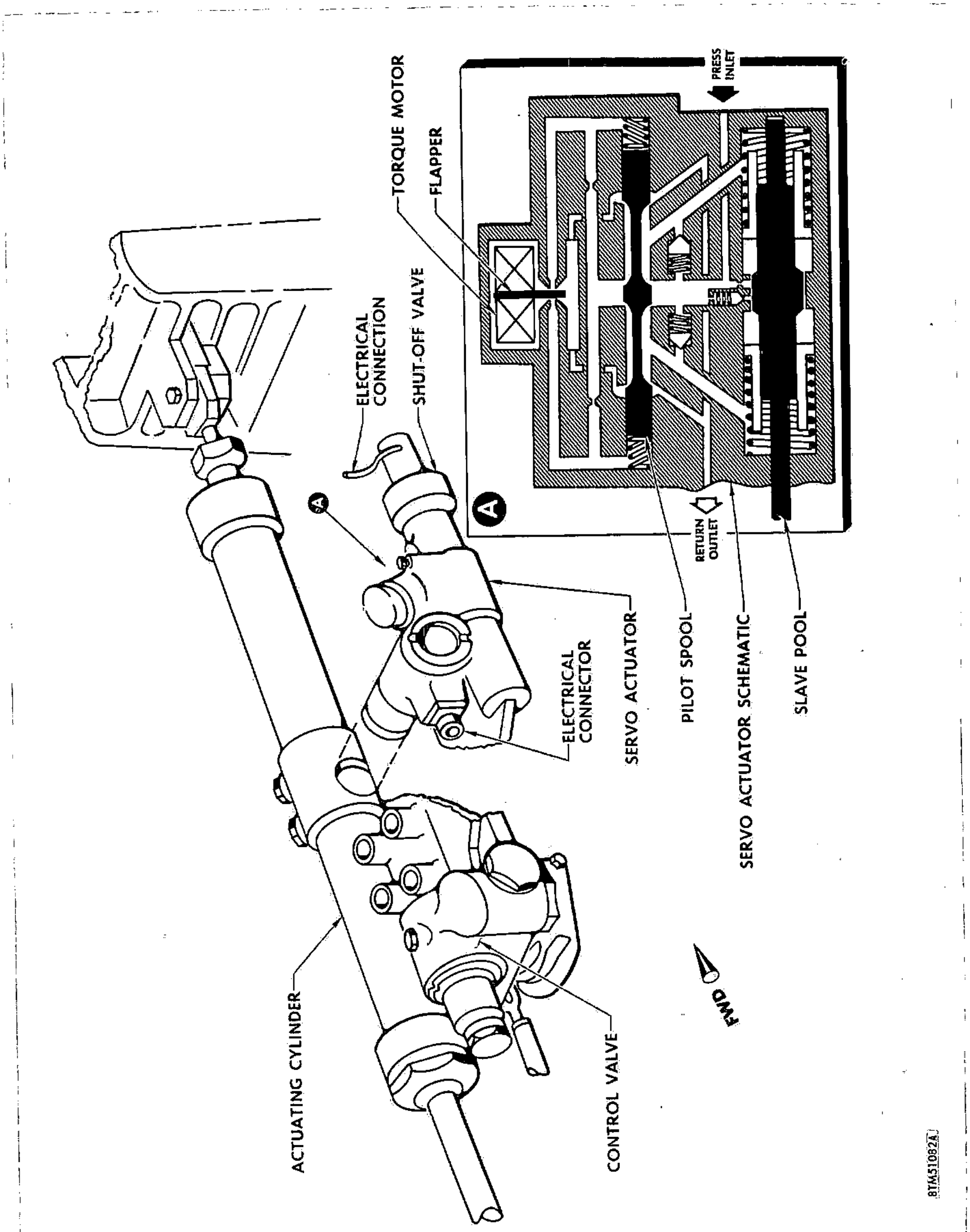

Figure 3-9. Rudder Servo Actuator

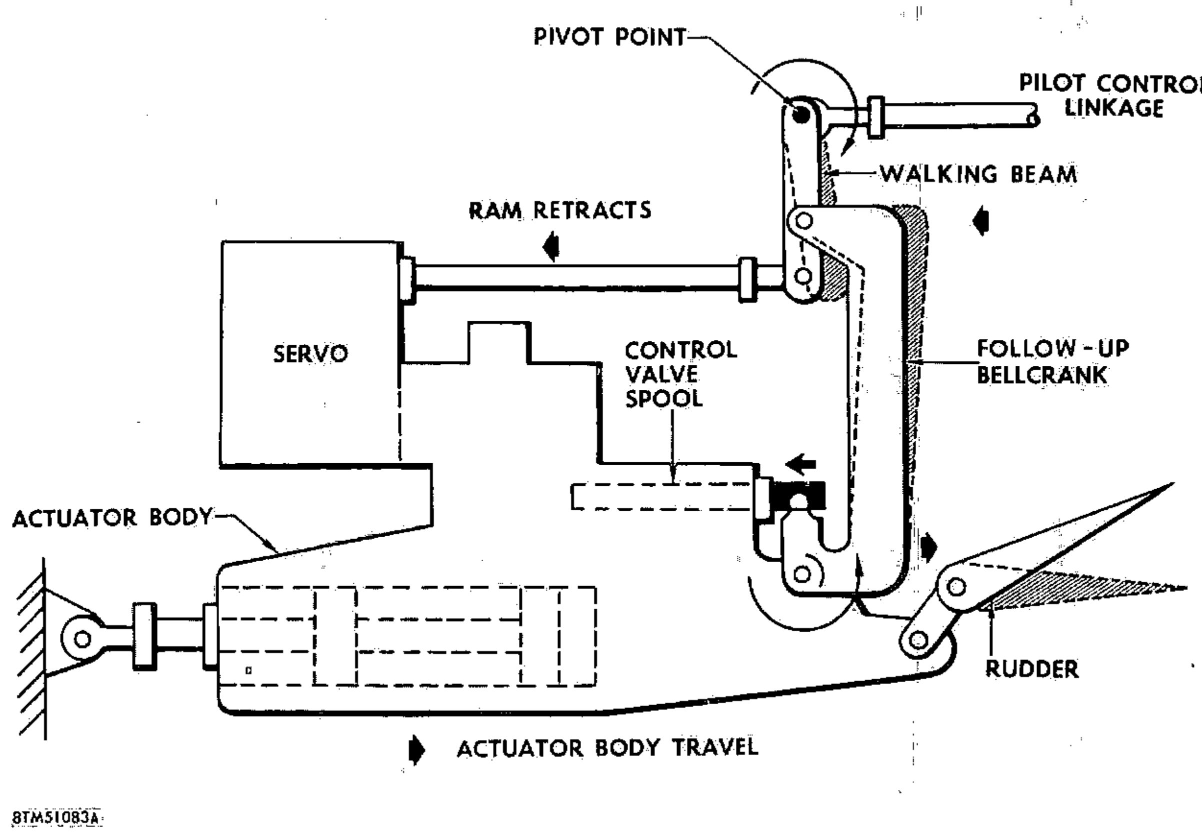

Figure 3-10. Rudder Mechanical Linkage Action, Followup Motion in Manual Mode

has been established since the return of the control valve to neutral locks the fluid in on each side of its piston. The only way it can move farther out, or back to neutral, is for the control valve to direct fluid pressure to the correct side of its piston again.

HYDRAULIC SYSTEM MAINTENANCE.

Since the control valve is incorporated within the casting of the actuating cylinder, malfunctioning of either section will necessitate replacement of the entire unit. If control surface motion becomes sluggish, and hydraulic test stand pressure is correct, the control valve is the most likely unit to be at fault. An inspection of the valve for evidence of clogged element screens, or chips of foreign matter, is indicated if the above occurs. A faulty O-ring is also a possibility. If leakage is noted around the actuator cylinder, this also would indicate a faulty O-ring. Air in the hydraulic system components will cause malfunctioning. If air is present in the system, it must be bled off. This can be accomplished by repeated cycling of the system.

During maintenance of the servo actuator system, here are some points to bear in mind. Erratic operation of the system can result from improper rigging, mechanical interference, air in the servo actuator, or a faulty servo actuator. After checking for improper rigging and mechanical interference, you should next check for air in the servo actuating cylinder at the action limits (to bleed off any air that may be in the system). If the condition is not corrected, the servo actuator is probably faulty and must be replaced. You must remember that when hydraulic lines are disconnected to replace a component, or for any other reason, the system will have to be bled of air.

A faulty electrical circuit to the servo actuator shutoff valve, or a faulty shutoff valve, may prevent release of hydraulic power to the servo actuator. A continuity check of the electrical circuit will help to determine if this is the trouble. An indication of this type of trouble is when the system engages and then disengages too soon.

ARTIFICIAL FEEL SYSTEM.

Since the hydraulic action of the flight control system is not reflected to the mechanical control linkage, airloads on the flight control surfaces are not transmitted back to the pilot. Consequently, an artificial feel system is required to simulate an airload condition. This system of simulating airloads on the control surfaces is known as the *artificial feel system*. The function of

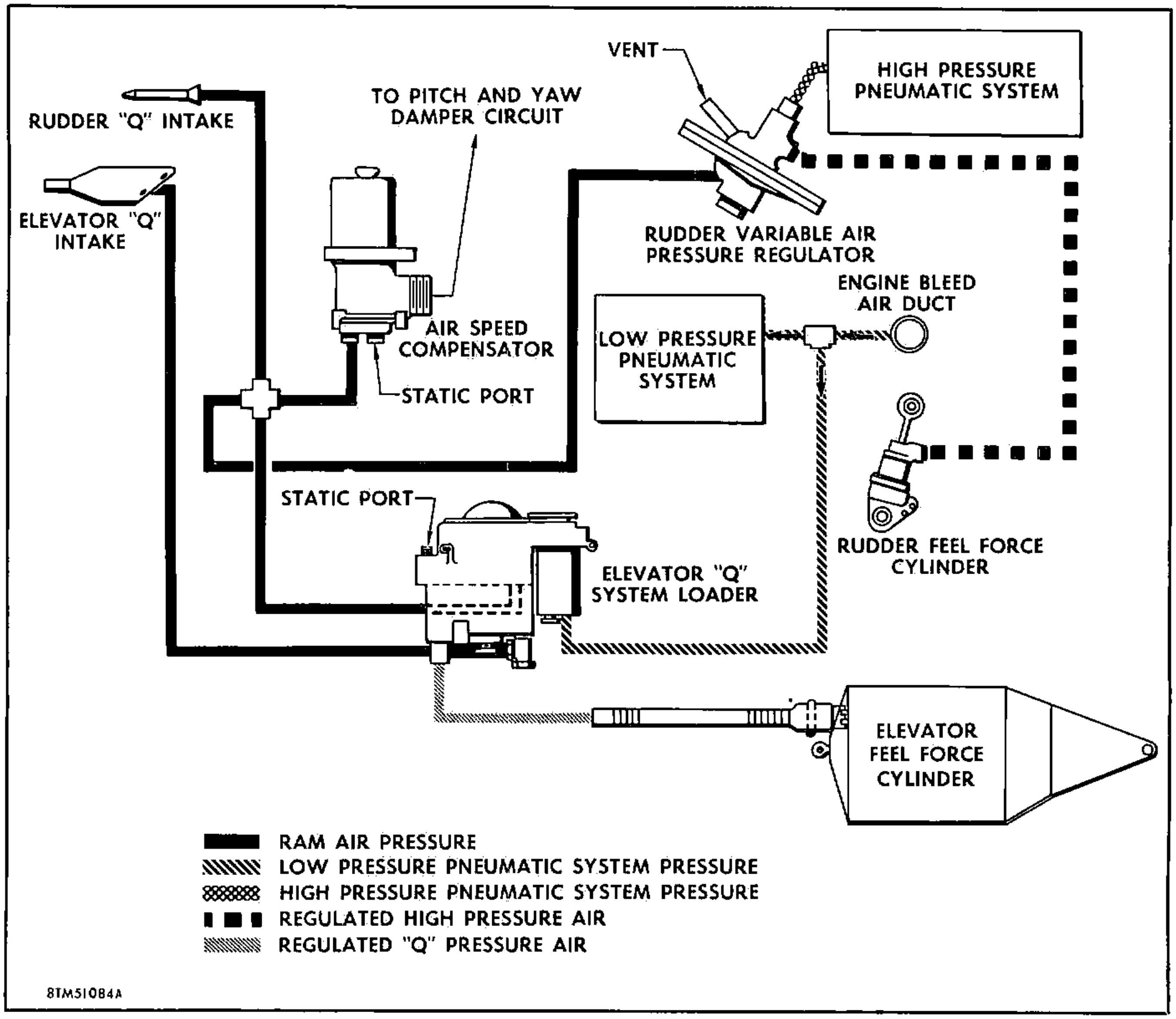

Figure 3-11. Rudder and Elevator Artificial Feel System Schematic

the feel system is to provide the pilot with feel forces which vary resistance to movement of the control stick and rudder pedals. The feel system is added to the mechanical linkage of the flight control system and uses both springs and programmed air to provide these feel forces.

Aileron "feel" is determined by springs only, while rudder and elevator "feel" is determined by a combination of springs and air cylinders. The feel system, through a combination of "Q" (ram air) pressure and pneumatic pressure acting on the feel cylinders, determines what control stick and pedal forces will exist for varying stick and pedal displacements, altitudes, and airspeeds. You learned about the elevon feel system in the preceding chapter. Let us now discuss the rudder feel system, how it operates, the components making up the system, and the maintenance problems you may encounter.

HOW THE RUDDER FEEL SYSTEM OPERATES.

The function of the rudder feel system (shown on figure 3-11) is to provide the pilot with feel forces at the rudder pedals. The feel force varies the resistance to pedal movement proportionately to the airspeed and altitude. As you recall, airloads on the control surfaces are not transmitted back to the pilot because the hydraulic action of the flight control system is irreversible (not reflected to the mechanical control linkage). Consequently, an artificial feel system is required to simulate airload conditions.

Referring to figure 3-12, note that the feel cylinder piston is attached to a bell crank on the vertical torque tube. The cylinder end fitting is attached to a bracket

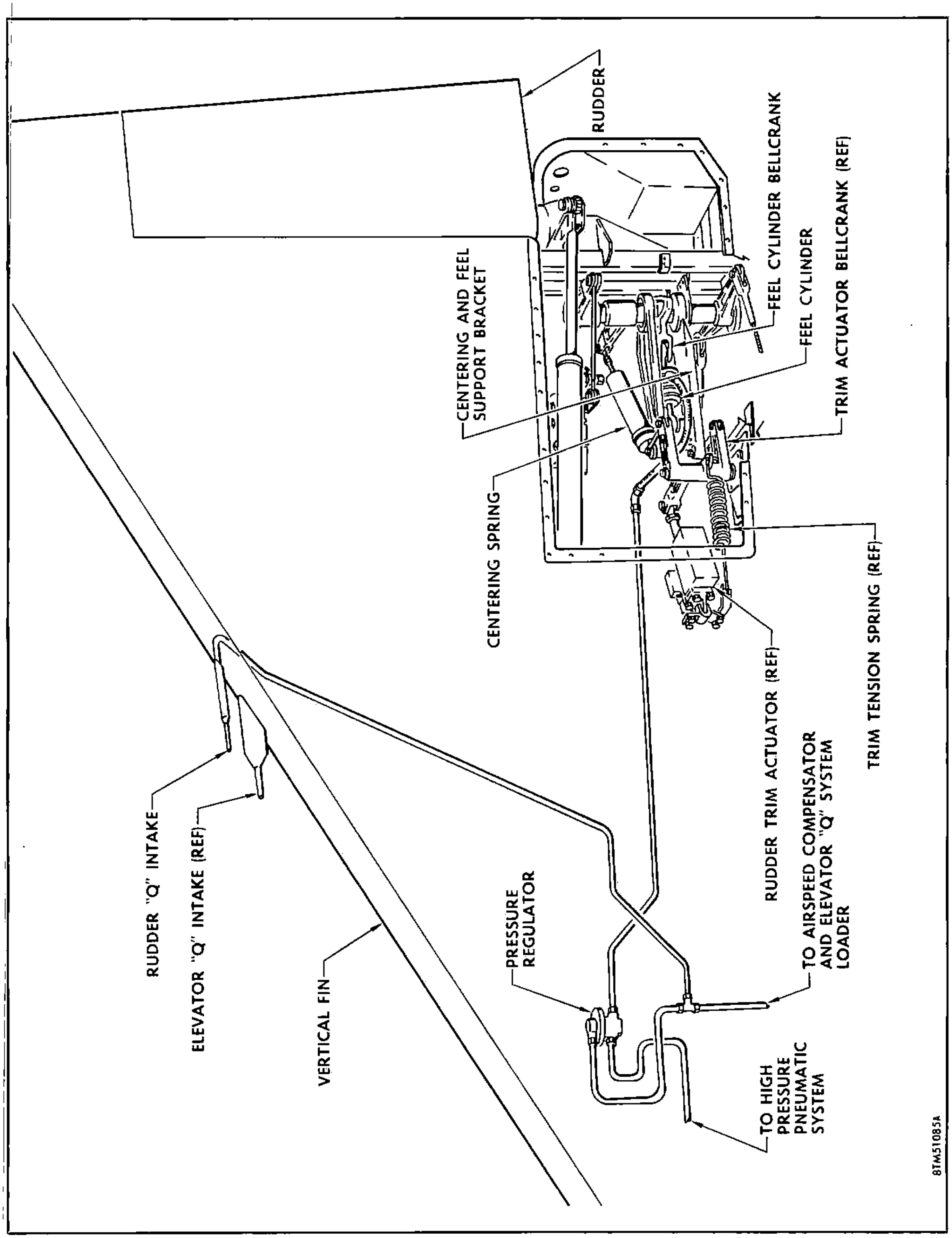

Figure 3-12. Rudder Artificial Feel System

on the free end of the feel and centering support assembly. The arms of the support assembly are attached to the torque tube through bearings which allow the feel and centering support to rotate freely on the torque tube; that is, the support assembly may rotate without rotating the feel cylinder holding bracket. The feel cylinder is mounted so that it is lined up with the rudder feel and centering support so long as the controls are centered with respect to trim.

Pilot displacement of the rudder pedals will rotate the vertical torque tube within the feel and centering support arms and the feel cylinder bell crank will rotate with the torque tube. Since the feel and centering support is held in a fixed position through the rudder trim actuator linkage, any rotation of the feel cylinder bell crank pulls the feel cylinder piston to increase its extension from the cylinder. This arrangement allows trim to be introduced into the system without affecting the feel system.

Air pressure proportional to airspeed is supplied to the rod end of the feel cylinder piston and the force produced by this pressure resists rotation of the torque tube. This force is felt by the pilot because he is supplying the initial force at the rudder pedals. Air pressure from the high pressure pneumatic system is regulated by a variable air pressure regulator to supply air to the feel cylinder. The air pressure regulator is supplied "Q" (ram air) pressure from the intake tube on the vertical stabilizer. The variable air pressure regulator then amplifies the ram air with the high-pressure air. This amplified ram air is then supplied to the rudder feel cylinder in amounts proportionate to the speed of the airplane.

When airplane speed is decreased, some of the high-pressure air is vented to the atmosphere through the regulator so that the pressure in the feel cylinder continues to be proportionate to the airplane speed. The faster the airplane flies, the more tension the feel cylinder puts into the flight control system. The "Q" pressure system also furnishes an airspeed reference to the pitch and yaw damper systems through the airspeed compensator for turn coordination.

COMPONENTS OF THE SYSTEM.

The components that make up the rudder artificial feel system include a feel cylinder, a variable air pressure regulator, and the "Q" (ram air) pressure system. The feel system is added into the mechanical linkage of the rudder control system.

"Q" System Intake Tubes.

You will note by referring to the schematic diagram on figure 3-12 that two intake tubes are installed on the leading edge of the vertical fin. The tubes provide the source of "Q" (ram air) pressure for the pressure regulators in the elevon and rudder control systems. The rudder pressure regulator amplifies the "Q" pressure air with high-pressure air and supplies this air to the feel cylinder in amounts proportionate to the speed of the airplane. The upper tube, shown on the diagram, supplies "Q" pressure air to the rudder variable air pressure regulator, the airspeed compensator, and to the 1/4-inch ram air inlet on the elevator "Q" system loader. The lower tube supplies "Q" pressure air to the 3/4-inch ram air inlet on the elevator "Q" system loader. The intake tubes are electrically heated through the anti-icing system which is discussed in another section of this training supplement.

Pressure Regulator.

The rudder variable air pressure regulator is located in the lower forward portion of the vertical stabilizer. It is mounted on a bracket assembly and attached to the bracket with three bolts through nutplates. Ascess to the regulator is through the dorsal duct access door on the stabilizer.

The regulator consists of a spring-loaded diaphragm and a pressure regulator. The function of the regulator is to amplify "Q" (ram air) pressure with high-pressure air (from the high-pressure pneumatic system). This amplified ram air pressure is then supplied to the feel cylinder in amounts that are proportionate to the speed of the airplane. When airplane speed is decreased, some of the high-pressure air is vented to the atmosphere through the regulator so the pressure in the feel cylinder is proportionate to airplane speed. A check valve in the pressure supply line to the regulator isolates the circuit from the rest of the pneumatic system to maintain a rudder feel force in the event of loss of pneumatic pressure.

Feel Cylinder.

The rudder feel cylinder, actuated by regulated high-pressure air from the high-pressure pneumatic system, is a simple cylinder and piston assembly. The air pressure in the cylinder varies in proportion to the "Q" (ram air) pressure, which is a function of airspeed and altitude. As the air pressure in the cylinder varies, the force on the piston varies. The force is felt by the pilot as resistance to movement of the rudder pedals. The piston is so mounted that movement of the rudder pedals in either direction from the neutral position works against the force in the feel cylinder. One end of the feel cylinder is attached to a bell crank on the vertical rudder torque tube; the other end is attached to a bracket on the free end of the rudder feel and centering support which is attached to, but rotates freely on, the torque tube.

SYSTEM MAINTENANCE.

If low rudder pedal forces are experienced in the flight control system, the most likely cause would be a leak

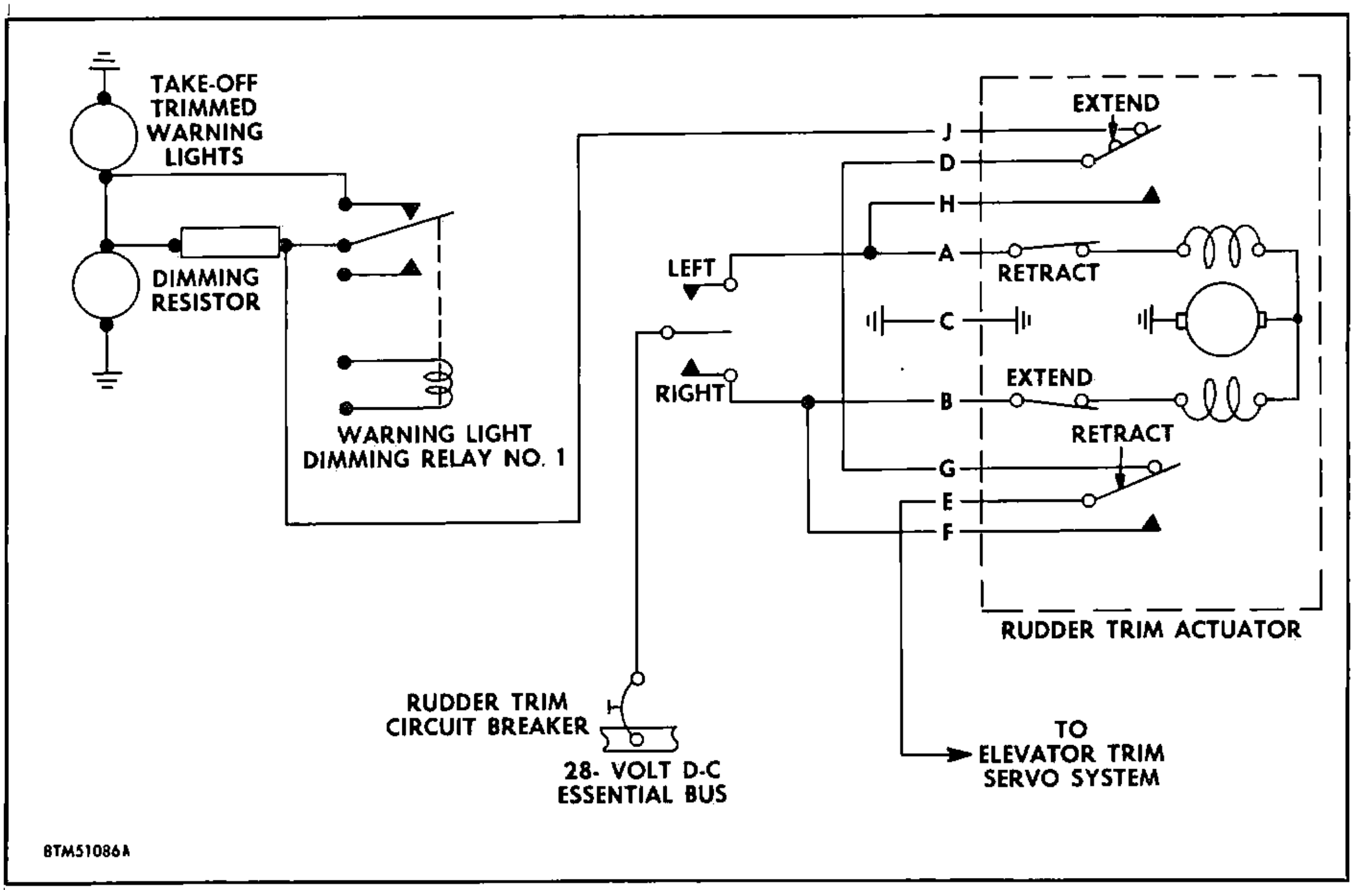

Figure 3-13. Rudder Trim Circuit Schematic

in the "Q" pressure system. A test procedure for checking the "Q" pressure system for leaks is outlined in the applicable F-102A maintenance handbook, T.O. 1F-102A-2-7. Leak tests must also be conducted whenever new or reinstalled equipment or lines are installed in the system. Another cause of low rudder pedal forces could be malfunctioning of the variable air pressure regulator. If, after conducting the leak test, the condition is not corrected, the regulator is probably malfunctioning and must be replaced.

TRIM SYSTEMS.

When an airplane is loaded in such a way that it is slightly wing-heavy, tail-heavy, or nose-heavy, the pilot must exert a constant pressure on the stick or rudder pedal in the opposite direction to the unbalanced axis to maintain straight and level flight. To relieve the pilot of this tiring effort, ailerons, elevators, and rudders are often provided with trim tabs. The trim tabs maintain the control surfaces in exact positions away from neutral to maintain balance in straight and level flight.

The elevons and the rudder on the F-102A airplane do not have conventional trim tabs. Trimming of the airplane is accomplished by electrical actuators that move the entire elevon and rudder surfaces. The elevon trimming is controlled by a five-position toggle switch located on the control stick. The rudder trimming action is controlled by a toggle switch located on the utility switch panel on the pilot's pedestal. The trim systems are interconnected through the takeoff trim circuit to provide a means of automatically trimming the airplane for takeoff. The takeoff trim switch is located on the utility switch panel also. Electrical power for takeoff trim is routed through the nose landing gear UP position switch so that the system is inoperative when the airplane is airborne.

RUDDER TRIM SYSTEM.

The rudder trim action is controlled by a three-position toggle switch located on the utility switch panel of the pilot's pedestal. The switch is spring-loaded to the center OFF position. Pushing the switch to the left side closes the circuit to supply 28-volt, d-c power to the retract side of the trim actuator for left rudder motion; pushing the switch to the right side supplies power to the extend side of the trim actuator for right rudder motion. When the switch is returned to neutral (OFF) position, the actuator movement stops. A schematic diagram of the rudder trim electrical circuit is shown on figure 3-13. The rudder trim actuator is located in the vertical fin island below the vertical stabilizer; it is attached to the airplane structure

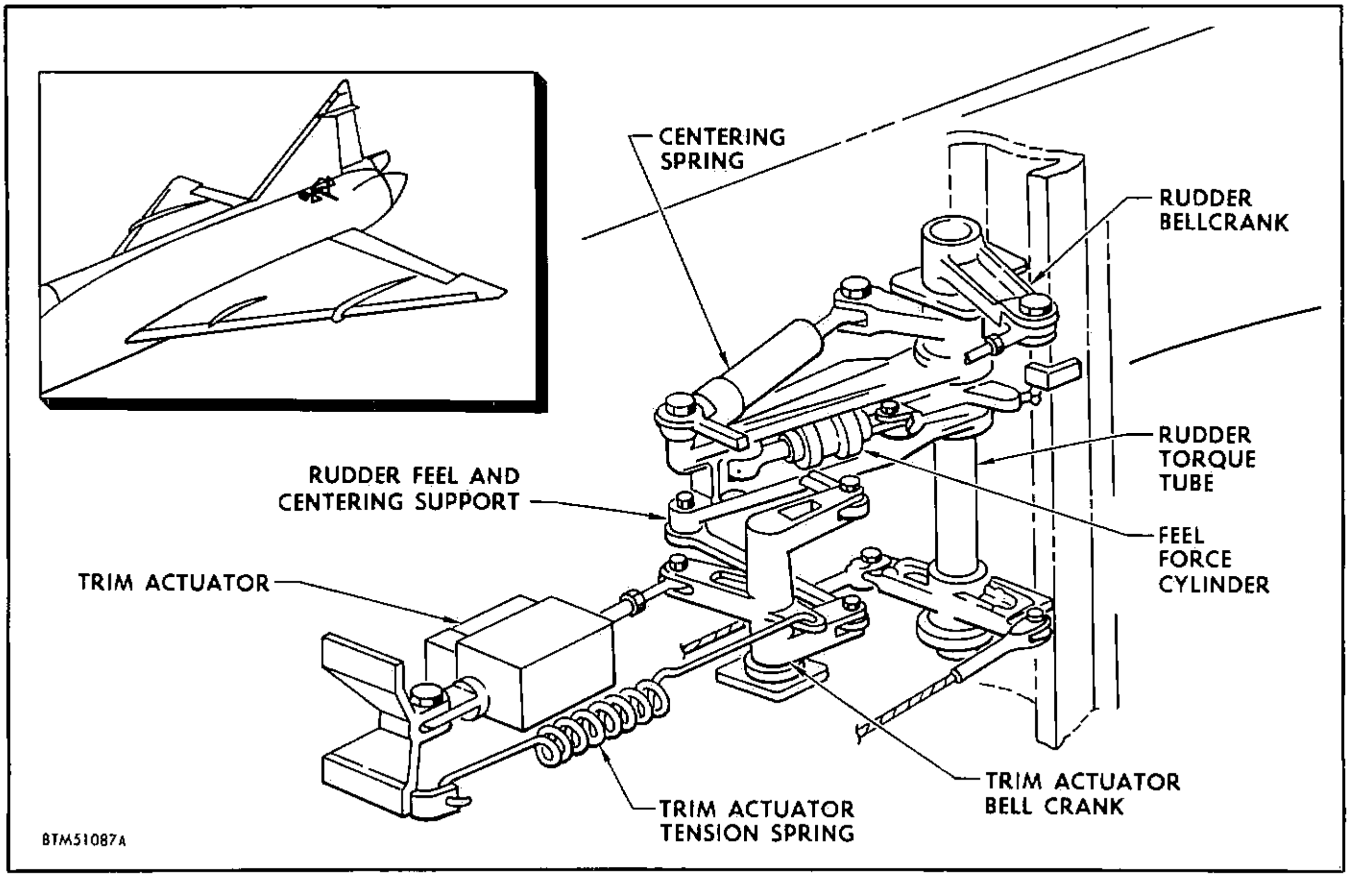

Figure 3-14. Rudder Trim Actuator and Linkage Perspective

at its forward (fixed) end and to the bell crank assembly at its aft (shaft) end. Integral limit switches in the actuator control the maximum travel limits.

Rudder Trim System Operation.

Figure 3-14 shows the rudder trim and also the rudder feel and centering components. Note that the trim actuator bell crank is connected through linkage to the free end of the rudder feel and centering support. The other end of the feel and centering support is attached to, but rotates freely on, the rudder torque tube. The feel cylinder is attached between the free end of the rudder feel and centering support and the feel cylinder bell crank. The bell crank is solidly attached to the rudder torque tube. Trim motion is introduced into the control system through the rudder trim bell crank to the free end of the rudder feel and centering support. This causes the rudder feel and centering support to rotate in the direction of trim. The trim force operates against the centering spring to move the control surface.

The feel cylinder is mounted so that it is lined up with the rudder and centering support so long as the controls are centered with respect to the trim. Any motion of the rudder feel cylinder bell crank offsets this arrangement so that the pilot is working against the feel cylinder when he moves the rudder pedals. This arrangement allows trim to be introduced into the system without affecting the feel-force system. The rudder centering spring operates in tension or compression to center the control system with respect to trim. The free end of the feel and centering support becomes a fixed point at whatever trim position has been selected. The centering spring then rotates the rudder torque tube through the centering spring bell crank to center the system. As the rudder torque tube is rotated, the torque tube bell crank actuates linkage to open the hydraulic control valve; pressurized hydraulic fluid is then metered to the rudder actuating cylinder to move the rudder. The rudder moves until the followup mechanism closes the control valve and the rudder has been displaced in an amount proportional to the trim actuator shaft movement.

Rudder Trim Actuator.

The rudder trim actuator, located in the lower aft section of the vertical stabilizer, consists of four internal switches (two limit switches and two position switches) and a d-c motor that drives the actuator rod. The two limit switches protect the motor by interrupting the circuit when the actuator reaches the fully extended and the fully retracted positions; the two position switches determine the neutral position of the actuator, provide the electrical connections to drive the actuator to its neutral position for takeoff trim, and illuminate the takeoff trim warning light.

The rudder trim actuator is energized by 28-volt, d-c power when the trim switch on the utility switch panel is positioned. The trim actuator is also energized when the takeoff trim switch is actuated and the trim actuator is not in its neutral position. Power to the takeoff trim circuit is routed through the nose landing gear UP switch so that the system is inoperative when the airplane is airborne. A procedure for adjustment of the internal limit switches is outlined in your F-102A maintenance handbook, T.O. 1F-102A-2-7.

EMERGENCY OPERATION OF FLIGHT CONTROL SYSTEM.

There are no emergency provisions directly incorporated within the flight control system. However, there are indirect emergency systems governing the power sources. As you remember, hydraulic power is supplied to the control system through two completely independent systems. Either of these systems is capable of powering the flight control systems. Maneuverability, however, is reduced when operating on only one system; the automatic flight control and pitch and yaw damper systems will not operate if the secondary hydraulic system becomes inoperative. An emergency hydraulic system is provided to supply power to the primary system in the event of an engine failure.

The electrical power systems provide two sources of emergency electrical power. An emergency alternator supplies power in the event of an alternator failure while the battery supplies power in the event of a generator failure. Since the pitch and yaw damper systems and the automatic flight control system receive power from nonessential buses, which are disconnected in emergency operation, the systems are automatically disengaged during an electrical power failure.

MAINTENANCE OF THE RUDDER CONTROL SYSTEM.

Maintenance problems of various parts of the rudder control system were discussed throughout the chapter. You learned that the best way to handle a maintenance problem is to refer to the trouble shooting tabular lists in the F-102A maintenance handbook, T.O. 1F-102A-2-7. These lists outline the probable cause of a malfunction and corrective measures to be taken. The above mentioned handbook also establishes procedures for removal, installation, and adjustment of the control system components, in addition to information concerning ground servicing and lubrication of the control system. Trouble shooting should begin at the power systems. Make sure that the control systems are receiving power from the separate sources involved before trouble shooting the control systems or their components. Before replacing a component that is suspected of a malfunction, make sure that the component is receiving power.

OPERATIONAL CHECK AND TESTING.

To maintain the operating efficiency of the F-102A flight control system and to insure that all parts of the system are functioning within specified limits, you will be required to perform operational checkout and testing of the system and its components. Operational checkouts are performed after reinstallation or replacement of any component, after a malfunction of the system has been corrected, and at established periodic intervals. If any malfunctions are encountered during the operational check of the system, refer to the trouble shooting tabular lists as explained above. Your F-102A maintenance handbook also outlines in a step-by-step fashion a procedure established for performing the operational check and testing. Before performing the operational check on the flight control system, certain preparations must be made, safety precautions must be taken, and certain conditions must be observed. These steps are discussed in the following paragraphs.

Precheck Preparations.

Preparations that you will be required to perform before an operational check of the control system include a visual inspection of the system for obvious errors in installation, clearances, cracks, distortion, and security; providing a portable hydraulic test stand and connecting the stand to the primary and secondary hydraulic systems of the airplane; connecting external electrical power to the airplane electrical systems; and providing a controllable source of dry, filtered air or nitrogen.

Safety Precautions.

It will be your responsibility to see that all areas adjacent to movable control surfaces and components are cleared of all objects and personnel before performing the operational check. Also station personnel at switches and controls that will be powered before you connect external electrical or hydraulic power to the airplane. This will prevent inadvertent actuation of the system or components that would endanger personnel or damage equipment.

Conditions To Be Observed.

As you recall, a portable hydraulic test stand is required for performing an operational check of the control system. When this test stand is used, it must be connected to both the primary and secondary hydraulic systems. If this is not done, pressure in the secondary system alone will cause the pressurized hydraulic fluid to bleed through interconnecting components to the primary system.

Before performing an operational check of the rudder control system, be sure that the trim system is properly rigged first. An improperly rigged trim system will adversely affect the proper functioning of the rudder control system. In other words, an operational check of the trim system must be performed before an operational check of the rudder control system.

REPLACING THE RUDDER.

The rudder, shown on figure 3-15, is attached to the rear spar of the vertical stabilizer with two hinge bolts and to the airplane structure at the lower support fitting. Also, three bonding jumpers join the rudder to the rear spar of the vertical stabilizer. To remove the rudder, it is first necessary to disconnect the rudder actuator at the rudder horn. The bonding jumpers are then disconnected at the rudder. After removing the bolts that attach the lower support fitting to the airplane structure, support the rudder and remove the two hinge bolts. The rudder is moved aft to clear the hinge fittings and then tilted to clear the upper section of the fin.

During installation of the rudder and at established periodic inspections, lubrication of the rudder hinge points and the actuator cylinder bearing at the rudder horn is required. Also, when installing the rudder, there are bolt torquing requirements to be observed. Refer to your F-102A maintenance handbook, T.O. 1F-102A-2-7, for the type of lubricant specified, the inspection interval when lubrication is required, and for bolt torquing requirements. The above mentioned handbook also outlines in a step-by-step fashion the precedure for installing the rudder.

RIGGING THE RUDDER CONTROL SYSTEM.

The basic requirement of the rigging procedure is to maintain the mechanical and hydraulic components in a neutral position while rigging the system. The neutral position is maintained by inserting rigging pins in rigging pin holes in the system bell cranks and airplane structure. The rigging pin holes in the system components and the airplane structure, when properly matched and locked in this position with the rigging pins, maintain the system in neutral. It is possible to use portions of the rigging procedure rather than the entire procedure, provided that portion of the system lies between two rigging pins, or between a rigging pin and the neutral position of the control surface. However, neutral positions of other components must be checked. Rigging of the rudder control system should be performed whenever a component has been removed from the system and replaced or reinstalled.

After rigging of the system is complete, an operational check must be performed to insure that all parts of the system are functioning within specified limits. Before performing an operational check of the rudder control system, be sure that the trim systems are properly rigged first. An improperly rigged trim system will adversely affect the proper functioning of the rudder.

Before rigging the control system, make sure that hydraulic pressure is relieved from the hydraulic system and the accumulators. No specific rigging procedures will be given in this training supplement. Refer to your F-102A maintenance handbook where you will find the rigging procedure outlined in a step-by-step fashion.

SPECIAL TORQUE VALUES.

All bolts that connect the various linkages and secure the components in the control systems must be torqued to specified values. Torque values, as you probably know, are the measurement (in inch-pounds or foot-pounds) of how much force can be applied to a wrench to turn a bolt or other type fastener. The force applied is dependent on the diameter of the bolt and the material of which the bolt is made. Special type wrenches called "torque wrenches" are used to apply this measured force. One type of torque wrench has a dial indicator to measure the amount of torque applied; however, there are also other types.

Torque values are established to prevent undertorquing or overtorquing a fastener. Too tight is as bad as not tight enough. If you overtorque a bolt, for example, you stretch the bolt beyond its elastic limit. The bolt either breaks or weakens to the point where it is not doing its job. Overtorquing can also distort the material under the bolt head or nut. Refer to your applicable F-102A maintenance handbook for specified torque values.

LUBRICATION OF THE SYSTEM.

Various components of the control system require lubrication during installation and at other specified intervals. Refer to your applicable F-102A maintenance handbook for these lubrication requirements. Lubrication charts in the handbook specify the type of lubricant to be used and the time intervals at which the various components of the control system require lubrication. The charts also indicate, by visual reference, the points to be lubricated and any special type equipment that will be needed.

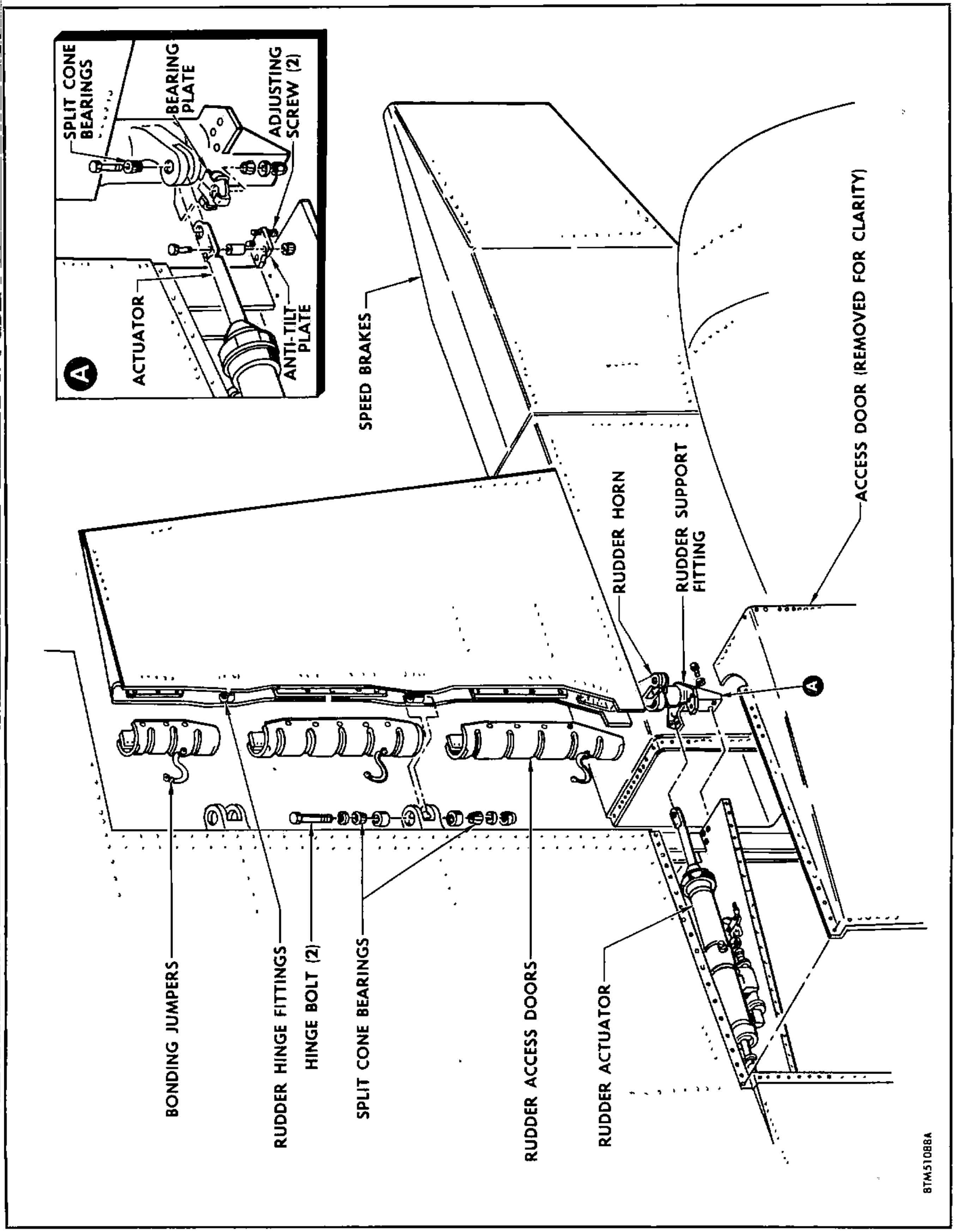

Figure 3-15. Rudder Installation

SUMMARY.

The first two chapters of this flight control system training supplement were concerned with the basic theory of flight controls, the F-102A flight control system, and the elevon control system. In this chapter, you learned about the rudder control system and how it operates, the mechanical and hydraulic components and their functions within the system, the artificial feel system, and the trim system. The last part of this chapter was devoted to a discussion of maintenance procedures, including operational checkout and testing and rigging of the rudder control system.

In the following chapter, you will learn about the pitch and yaw damper systems and their function in the F-102A flight control system.

Chapter IV

PITCH AND YAW DAMPER SYSTEM

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WHY A DAMPER SYSTEM IS NECESSARY.

A damper is a device that controls the vibration or oscillation of a moving body about its axes. One of the most common dampers we encounter today is the shock absorber on the automobile. You are probably aware of the function of the shock absorber—especially if you've driven a car a great number of miles without replacing these parts. You may have noticed that every small bump in the road caused the body to pitch and sway in an uncontrolled manner. This happens because the shock absorbers did not *dampen* the action of the springs.

You can easily observe this damping action while the car is at rest by raising the front bumper and quickly releasing it. If the front end of the car rebounds once and returns to a normal position, the shock absorbers are doing their job. On the other hand, if the front continues to rise and fall several times, the shock absorbers are worn out and the springs continue to toss the body up and down.

Airplanes flying through the air require shock absorbers the same as automobiles traveling down the highway. If some force—such as a gust of wind—causes the airplane to start oscillating, it is going to deviate from its course and will continue to oscillate until the natural stability of the airplane returns the craft to normal flight. Depending on the design of the aircraft, there are varying amounts of "built-in" stability. A general rule is that large passenger and cargo planes possess great natural stability due to their design, but consequently they suffer in maneuverability and performance. On the other hand, fighter-type aircraft—designed for maximum performance and maneuverability—are relatively unstable.

When planning a new airplane such as the F-102A, designers must follow certain specifications as to performance, range, maneuverability, and size. Since the requirements for the F-102A were so extreme, the compromises in design to attain the desired characteristics (of high speed, maneuverability, and relatively small size), resulted in some sacrifice of natural stability. However, the overall design of the F-102A makes it one of the most stable aircraft of its type in production.

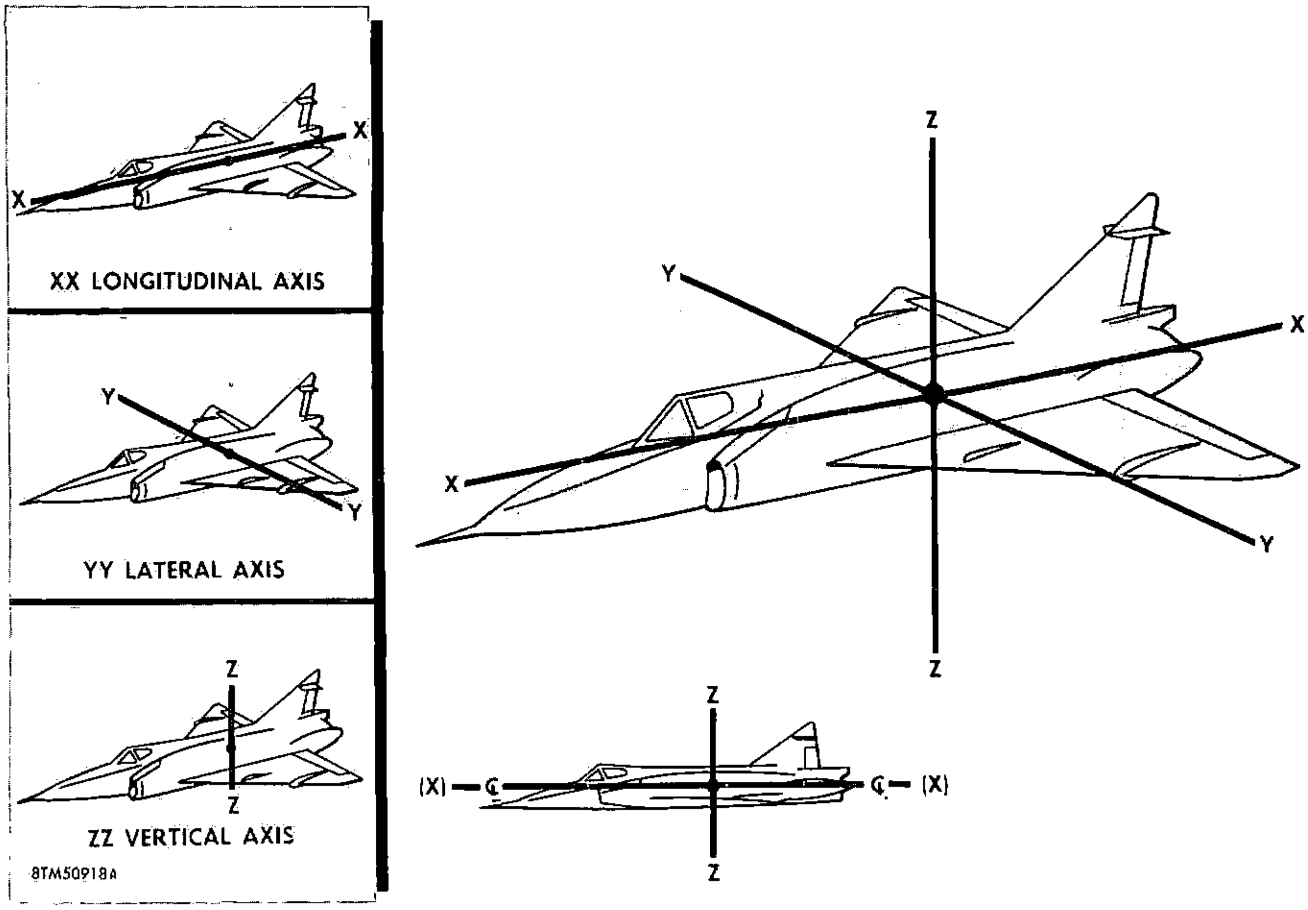

Figure 4-1. The Aircraft Axes

WHAT AN AIRCRAFT DAMPER SYSTEM IS.

To compensate for the lack of stability, designers have substituted an artificial means of providing stability in the F-102A. This artificial control is known as a damping system. Such systems not only superimpose damping control surface movements on the normal pilot control movements but also they automatically operate in special automatic flight modes of operation. This results in a stable aircraft at all times.

THE DAMPER SYSTEMS ON THE F-102A.

This chapter will acquaint you with the damper systems employed on the F-102A. You will learn of the components comprising these systems and their operation. With one major exception, these damper systems —the pitch and yaw—are quite similar in circuitry, components, and theory of operation. This one exception is the axes that the systems control. To acquaint you with the axes of the aircraft, let us consider the examples shown in figure 4-1.

Note in the illustration, figure 4-1, the three axes of the aircraft. The rotation of the aircraft around the longitudinal axis (x-x) is known as *roll.* The rotation around the vertical axis (z-z) is called *yaw.* The rotation around the lateral axis (y-y) is *pitch.* Picture in your mind the surfaces that control the airplane movement around these axes. The ailerons control the *roll,* the rudder controls the *yaw,* and the elevators control the *pitch.* Therefore, a damper system that would correct a deviation from any of these axes would naturally be connected to the control surface for this particular axis.

A TYPICAL DAMPING SYSTEM.

Figure 4-2 shows a block diagram of a damper system similar to that installed in early series F-102A aircraft. Later versions do not incorporate an electrical feedback, but utilize a mechanical centering device that accomplishes the same purpose as the servo actuator feedback. Notice that the main components of this damper system are a rate gyro, an amplifier, a servo motor, and a feedback potentiometer. The rate gyro and the control surface actuator each have wiper arms that pick off voltages from a potentiometer. In the case of the rate gyro, this voltage corresponds to the *rate of deviation.* In the control surface actuator, the voltage is proportional to control surface movement.

Assuming that the aircraft tries to move off course due to an external force, the rate gyro senses this force and sends a signal proportional to the rate of deviation to

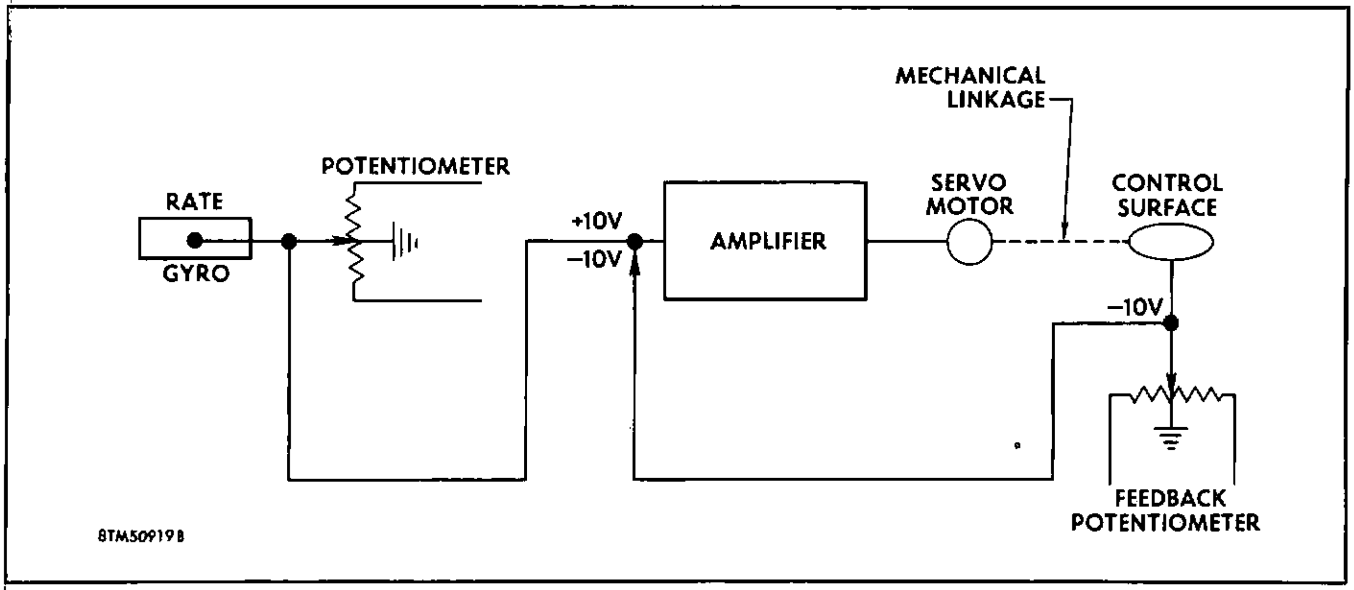

Figure 4-2. Block Diagram of a Typical Damper System

the amplifier. The purpose of the amplifier is to amplify the signal to a strength sufficient to drive the servo motor. The servo motor moves the control surfaces at a rate proportional to the amount of deviation sensed by the rate gyro. The movement of the control surfaces counteracts the deviation and quickly corrects the flight path.

However, if we leave the system in such a state, the control surface remains in the position necessary to counteract the original deviation. Thus, the airplane will attempt to go from the neutral position to the position directed by the new control surface position. The rate gyro senses the new deviation and again sends corrective signals to the amplifier. The servo motor receives the amplified signal and moves the control surface to a new position. Such a device would not be satisfactory since the system would keep making a continuous correction.

To overcome this undesirable characteristic of damper action, designers have added a feedback circuit from the control surface actuator back to the amplifier. The part of the amplifier that receives this feedback signal is called the *summing stage* because it actually adds the signals algebraically.

The summing stage of the amplifier operates in the manner of a balance. For example, if the rate gyro sends a +10 volt signal to the amplifier, the control surface moves a distance proportional to the rate gyro signal. When the control surface moves, the wiper arm attached to the actuator moves along the potentiometer until it picks off a —10 volts. Referring to the block diagram (figure 4-2), you can see that this feedback signal is sent to the summing stage of the amplifier. Since the rate gyro signal and the control surface feedback signal are in the summing stage of the amplifier at the same time, the two signals (of opposite polarity) balance each other. Consequently the amplifier no longer sends a signal to the servo motor and the control surfaces stop moving.

As the forces acting on the aircraft become less and less, the signal from the rate gyro diminishes. But the summing stage of the amplifier is still receiving the —10 volt signal from the wiper arm at the control surface actuator. The amplifier sends this signal to the servo motor and drives the control surface back toward the original or neutral position. When the control surface moves to the neutral position, the damper system has completed a cycle of operation. Let us briefly recount what has happened. The rate gyro sensed a deviation, the wiper arm mounted on the gyro case picked off a deviation signal proportional to the rate of deviation and sent the signal to the summing stage of the amplifier. From here the amplified signal traveled to a servo motor. The servo motor moved the control surface and counteracted the deviation of the aircraft from the desired course. During the movement of the control surface, a wiper arm picked off a voltage of opposite polarity from its potentiometer—the amount of signal corresponding to the position of the control surface. This feedback signal went into the summing stage of the amplifier and balanced or cancelled the deviation signal from the rate gyro. Thus the servo motor stops driving as the control surface has compensated for the deviation. This leaves the summing stage of the amplifier with only the negative signal from the control surface actuator. The rate gyro is then in a neutral position. The amplifier sends the negative signal to the servo motor causing it to drive the control surface back towards the neutral position.

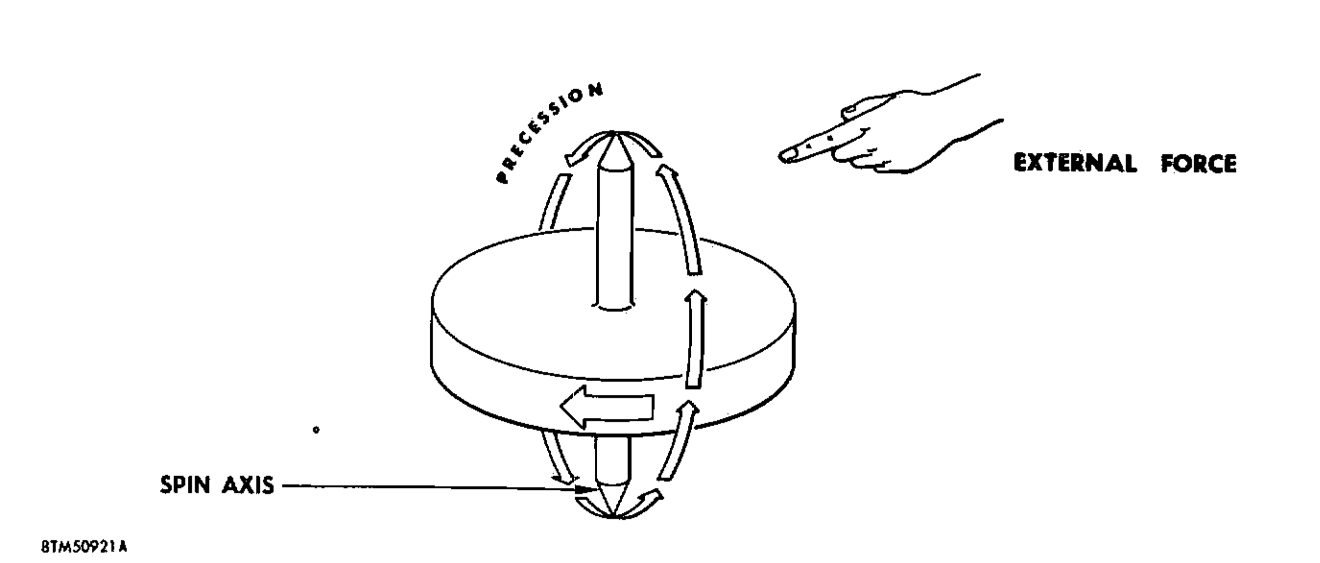

Figure 4-3. Gyro Precession

Thus, the damper system has sensed a deviation, counteracted the deviation (with a movement of the control surface), and moved the control surface back to neutral. Incidentally, this happens so quickly and smoothly, the pilot never knows that it has taken place.

COMPONENT DESCRIPTION.

To adequately understand the sequence of events described in the operation of a typical damper system, you must learn the theory of operation of the separate components. Since the rate gyro initiates the action of the damper system, let us examine the construction and operation of a typical rate gyro.

The Rate Gyro.

The rate gyro operates on the principle of the gyroscope. A gyroscope is nothing more than an accurately balanced flywheel spinning rapidly around a central axis. Two important characteristics of gyroscopes are shown in figures 4-3 and 4-4. In figure 4-3, note that the gyro moves at right angles to the applied force. This characteristic of a gyroscope is known as *precession.* The term precession means the act of preceding or leading. Actually, the movement of the gyro does precede the force 90° in the direction of rotation.

Another kind of precession, *apparent precession,* explains the other characteristic of gyroscopes; this is shown in figure 4-4. Gyroscopes resist any force which tries to change the direction of their axis. This rigidity, or gyroscopic inertia, can be illustrated in the following manner. If you mount a gyro in a universal mounting that is free to move in any direction, and maintain the rotation of the gyro for 24 hours, the gyro will remain "fixed" in space regardless of the earth's rotation. If the axis of the gyroscope is vertical at the equator at 6 a.m., it will appear horizontal at noon, upside down at 6 p.m., and return to the original position at the end of the 24 hour period.

These characteristics of the gyroscope make it an ideal device to serve as a reference plane for direction and attitude of the aircraft—it will tend to remain in its original position and its movement can be predicted from an applied force. Now let us examine how the gyroscope is able to tell the controls that the airplane is deviating from the desired course.

PRINCIPLES OF THE AUTOPILOT. Pitch and yaw dampers are only refinements of autopilots using principles which have long been known. In some applications, especially in early systems, the autopilots depended on *displacement* gyros for their deviation signals. A *displacement* gyro is one in which the attitude of the airplane is measured in reference to the fixed axis of rotation of the gyro. A potentiometer is attached to the gyro in such a manner that as the airplane deviates from a neutral attitude, a signal voltage is picked from the potentiometer that is proportional to the *amount of airplane movement* from the neutral attitude. Consequently, a corrective control surface displacement results which is proportional to the deviation of the airplane from neutral. This system of automatic flight control in which the control surface is moved a distance proportional to the airplane attitude deviation is a very common one.

The autopilot has used a satisfactory proportional system for many years, but it is not intended for damping functions. Variations in airplane attitude are too great *before* corrective control is applied, to be of

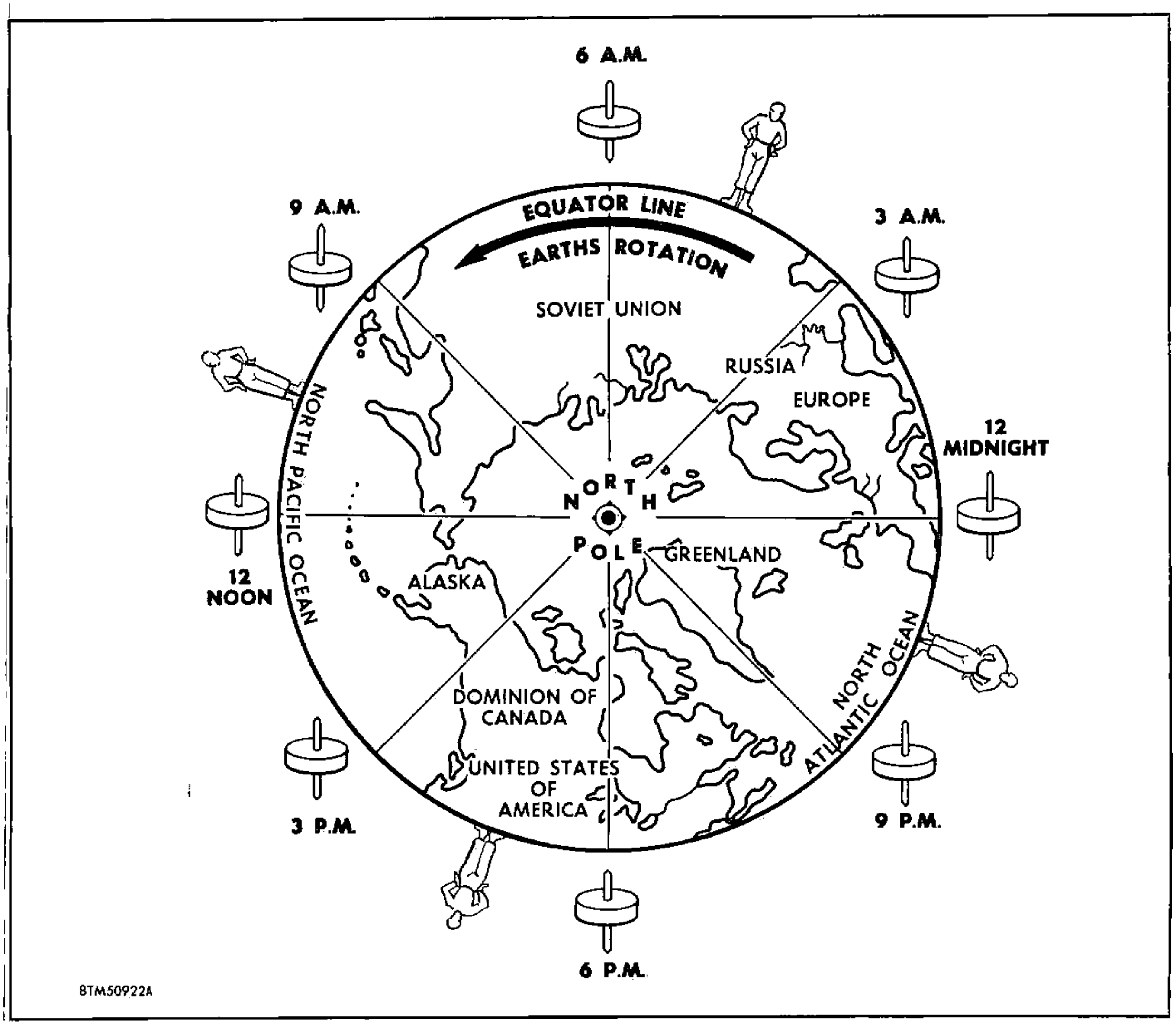

Figure 4-4. Gyro Rigidity

service for armament fire control purposes. Consequently, damping systems employ a system mechanization which applies a corrective surface movement that is proportional to the *rate of deviation* of the airplane.

To better understand what is meant by rate of deviation, consider the airplane as a pendulum. When the pendulum swings to one of its limits, as shown in figure 4-5, you can see that it has traveled its maximum displacement. When the pendulum reaches maximum displacement, it stops before swinging back in the other direction. From this we can say that maximum displacement results in *zero rate* of displacement. The opposite of this condition would be the dead center point in the return swing. In this condition, the *rate* of displacement is maximum and the displacement is zero.

To effectively understand how the damping system enters into this picture—let us see what would be the best way to stop the pendulum from swinging. Considering the airplane, the most logical place would be near the center of the swing. This is the point where rate of movement is maximum and displacement is zero. Thus, displacement could be kept to a minimum.

The rate gyro's job is to sense the *rate* of displacement of the aircraft from a particular axis, and through a center-tapped potentiometer and a wiper arm pick off a positive or negative voltage that is proportional to the rate of deviation. An important fact to remember is that the rate gyro is mounted in such a manner that it is free to move only around one axis. The axis in which the gyro is restrained (or fixed so that it cannot move) is referred to as the rate input axis. A deviation of the aircraft from this axis causes the gyro

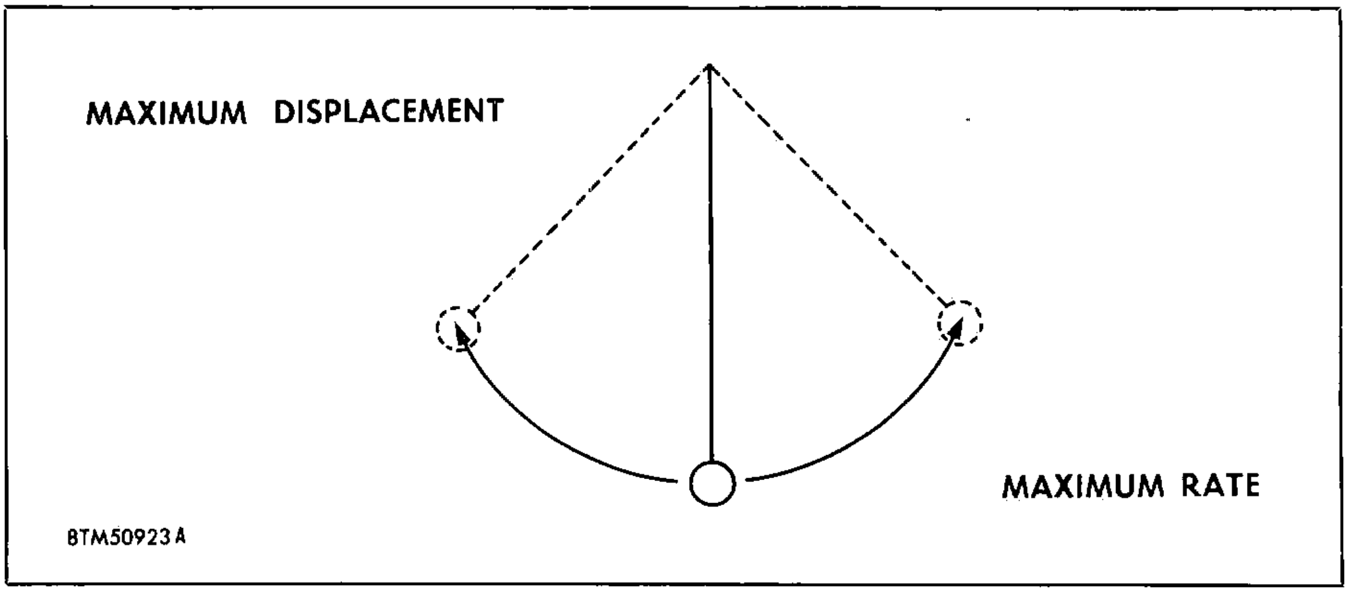

Figure 4-5. Theory of the Rate Gyro

to precess in its free axis. Another feature of rate gyros is the rate springs that limit or control the amount of precession. The reason for this is that the rate can be varied according to the sensitivity desired in the system. For example, the pitch and yaw rate gyros each have 20° rate springs. This means that a deviation along the pitch or yaw axis of 20° per second will give maximum potentiometer wiper arm displacement. On the other hand, the roll rate gyro has 90° rate springs. These springs are designed to give maximum wiper arm displacement when the aircraft roll rate reaches 90° per second.

Let us review rate gyro operation. Note in figure 4-6 that when the airplane changes attitude or deviates from its course, the rate gyro fixed along the axis of deviation will precess. The gyro will precess until it is stopped by the rate springs. Thus, the gyro will act against the rate springs in an amount that is proportional to the *deviation rate*. With the wiper arm attached to the gyro, the precessing movement will result in an appropriate pickoff voltage from the potentiometer fixed on the case of the gyro. Thus, the purpose of a rate gyro is to sense the *direction* and *rate* of displacement from a reference state of rest.

The gyro consists of a rapidly spinning rotor mounted in a housing which in turn is supported on ball bearings in a frame. The rotor housing is free to move or precess about *one* axis only—the axis through the ball bearing supports. The gyro is sensitive to and measures the rate of rotation of the airplane through the base of the gyro and perpendicular to the pivot points. Two coil springs, attached to the housing and to a bracket on the frame, return the gyro to neutral whenever it tilts in either direction. Rotation of the reference plane will cause the rotor to tilt (precess) about the mounting axis. One of the coil springs will then apply a force tending to return the rotor to the original position when the returning force of the spring equals the precessional force of the gyro, and the movement of the gyro will be proportional to the rate of turning of the reference plane.

If a potentiometer is attached to the housing, and the wiper arm of the potentiometer is attached to the frame, a signal voltage will be picked off which will be proportional to the *rate* of the turn. As you have already learned, the strength of the springs which restrain the tilting of the gyro determines the maximum turning rate at which the potentiometer will be fully displaced. The spring strength and the dynamic characteristics of the gyro together determine the response of the gyro. For example, a spring whose strength is such that a 10 degree-per-second turning rate produces a full potentiometer displacement is said to be a 10 degree spring.

FUNCTION OF THE RATE GYRO. The rate gyros develop electrical signals which are proportional to the rate of deviation of the airplane about its axes. These signals are such as to produce control surface displacement opposing the airplane's deviation.

Note in figure 4-7 that the gyroscope is mounted to a support ring in a manner so that it is free to precess only about an axis perpendicular to the line of flight of the airplane. The spin axis of the gyroscope is normally maintained parallel to the line of flight, and to the base of the gyro, by means of two coil springs whose ends are attached to the support ring and the base of the gyro. An adjusting screw on each of the spring supports, enables individual adjustments of the spring tensions for proper centering of the gyroscope support ring assembly.

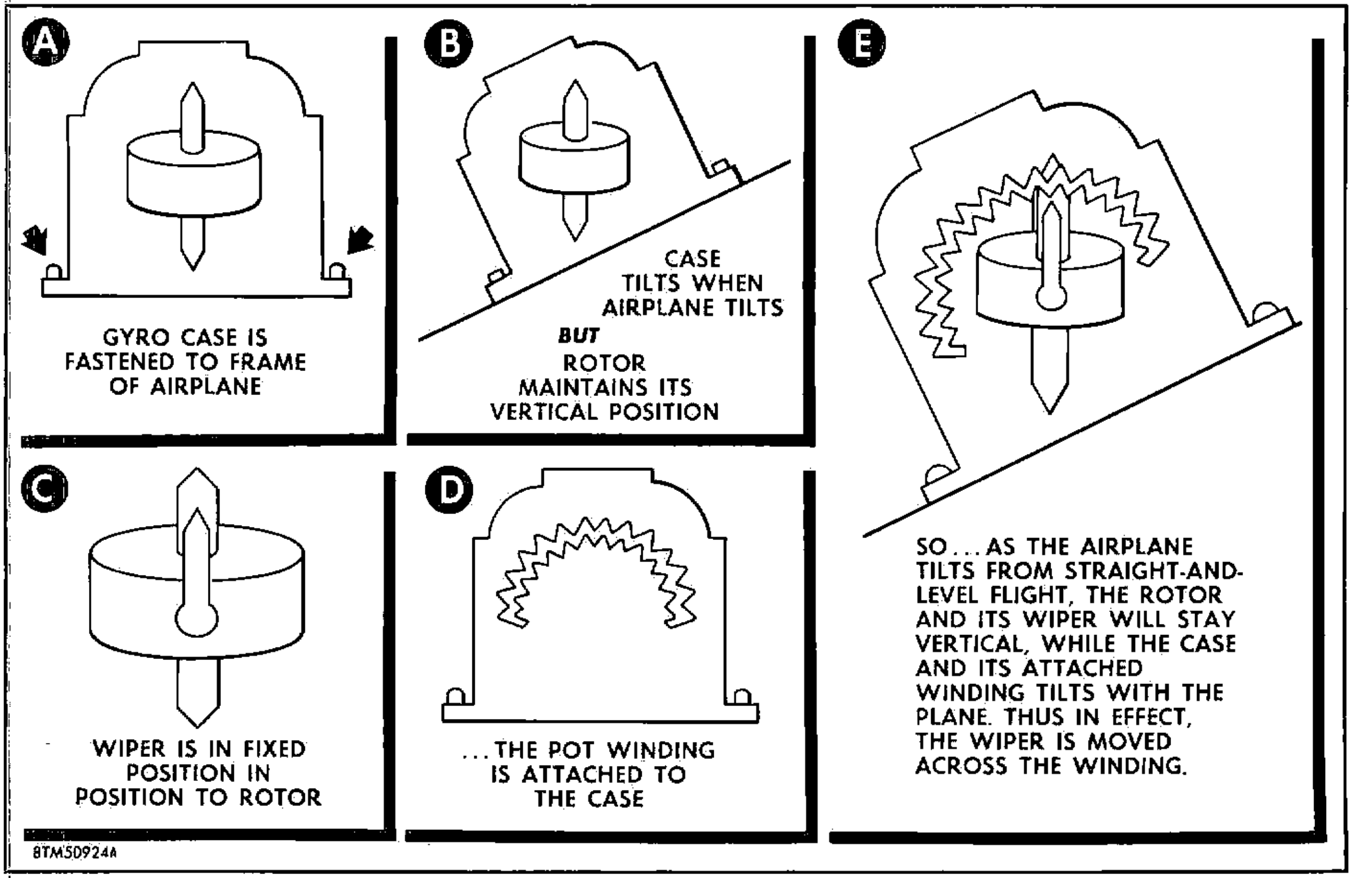

Figure 4-6. The Origination of the Rate Gyro Signals

The precessing force, applied to the rapidly spinning gyroscope rotor whenever the airplane rotates about a particular axis, causes the gyroscope support ring to tilt from its parallel position with respect to the gyro base. This stretches one of the coil springs and introduces an opposing force which tends to balance the precessing force. The gyro support ring thus reaches a position of rest where the spin axis of the gyroscope is tilted at an angle which is proportional to the airplane's rate of turn. To measure the gyroscope angle of tilt, note that the support ring is provided with a potentiometer wiper which contacts a cylindrical, wire-wound potentiometer attached to the gyro base assembly.

The gyroscope portion of the rate gyro consists of an electric motor which operates on 115-volt, 400 cycles-per-second, single-phase, alternating current. The assembly differs from a conventional motor in that the rotor consists of a cylindrical shell which completely encloses the stationary stator assembly. Spinning with a velocity of 18,000 rpm, and with the rotating mass outside the stator, maximum gyroscopic action is thus obtained per unit of overall weight.

The gyroscope proper is mounted in the support ring with rotor freedom being provided by ball bearings. The support ring is mounted on the gyro base by means of additional ball bearings which reduce the frictional resistance to gyro precession to a minimum. The gyroscope rotor is dynamically balanced, and the gyroscope support ring assembly statically balanced to eliminate the possibility of gyro precession due to unbalance within the gyro itself.

The entire rate gyro assembly is protected from dirt and accidental damage by a cover attached to the base assembly. A line-of-flight arrow on the gyro base indicates the proper mounting position of the unit and must be aligned with the desired airplane axis.

Due to the highspeed of the gyro and the critical tolerances of the bearings and shaft, there is a provision for keeping the gyro at a fairly even temperature for maintaining the tolerances. Later in the schematics of the damper systems you will find that there are thermostatically controlled heating devices within the cover of the gyro. These devices maintain the temperature at 75°(±10°)F.

The Amplifier in a Typical Damping System.

Direct current voltages are used almost exculsively in the damper system amplifiers of the F-102A. For this

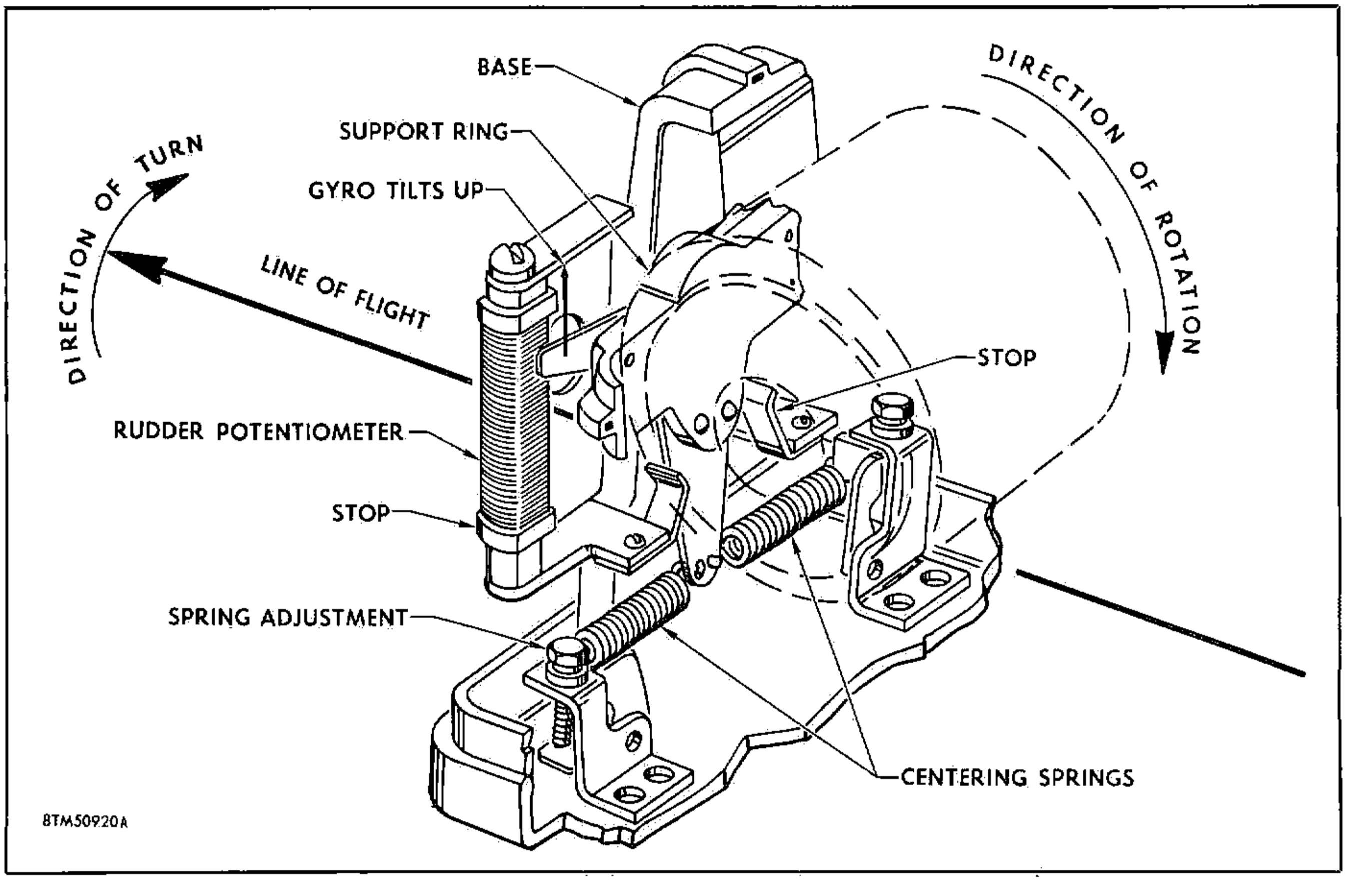

Figure 4-7. The Rate Gyro

reason, we will not discuss a-c amplifiers, but will limit our discussions to the specific application of the d-c amplifiers found in the F-102A.

In most cases, direct-current amplifiers are used to amplify d-c voltages. A simple d-c amplifier is shown in figure 4-8. As you can see, it consists of a single vacuum tube with a grid resistor across the input terminals and with the load in the plate circuits. The load may be some sort of mechanical device, such as a relay or a meter, or may be the input to another amplifier. The d-c voltage that is to be amplified must be *applied directly to the grid of the amplifier tube.* For this reason, only direct coupling can be used in the amplifier input circuit.

In the F-102A, d-c amplifiers are used as voltage regulators where small changes of voltages must be amplified for control purposes. Figure 4-9 shows the two stages of a direct-coupled amplifier. Note that a d-c voltage is applied directly to the grid of the first stage. The output of the plate of this first stage is coupled directly to the grid of the second stage. An interesting point that you should remember is that the d-c voltage on the first stage does not necessarily represent a signal from only one source. It can be a combination of signals from several sources, as you will learn later in the discussion of the yaw damper system. For this reason, the first stage (tube) of a d-c amplifier is called a summing stage. In the summing stage there is a small amplification as well as summing, because the input signals to the second stage grid must always be greater than the inputs to the summing grids.

THE TRIODE. From the previous discussion, you have learned the general principles of amplification. Now let us examine the theory of amplification at the source of the amplification—the triode. As you know, a triode is a tube that contains three active elements—a cathode, a control grid, and a plate. The cathode has a negative potential with respect to the plate, the grid has a varying potential, and the plate is always positive with respect to the other elements. Variations of the voltage on the grid permit regulation of the plate current. You can see how this is accomplished by studying the detail views in figure 4-10. Note that the electrostatic field arrows are drawn to indicate the direction in which the emitted electrons are attracted.

In detail A, the control grid (represented by the small circles) is made so negative with respect to the cathode that all lines of force terminate on the grid and no field exists between the cathode and the plate. This field is in such a direction that it causes the return of the electrons emitted from the cathode. Thus, there are no electrons that reach the plate from the cathode, and the plate current is therefore zero.

Detail B shows what happens when the grid is made less negative. There are two forces at work in this detail B. The first force is that which exists between the plate and the cathode. The plate potential is much more positive than the cathode. This positive potential is an attracting force to the electrons on the cathode. The second force is that of the grid. The grid is more negative than the plate. This force tends to repel the electrons traveling to the plate and sends them back to the cathode. But, if this grid potential is made less and less negative, the plate current flow will increase. The variation of the voltage on the grid has more effect on the electron flow than a similar amount of variation of the plate voltage. This can be explained by the fact that the velocity of the electrons at the time they reach the grid is not as great as the velocity of those that reach the plate. Because of this distance-velocity relationship, you can see that the potential on the grid will have much more effect on the electron flow than the plate potential.

In detail C, the negative voltage on the grid has been reduced to zero. This leaves only one force, plate voltage, to affect electron flow. Plate current is high under these conditions, since the emitted electrons no longer experience a repelling force from the grid. However, some electrons strike the control gird and cause a small grid current flow.

An interesting case is shown in detail D. Here the control grid is made more positive than the cathode, and the grid and plate potentials create electrostatic fields that attract the emitted electrons. Under these conditions, both plate current and grid current flow are quite high.

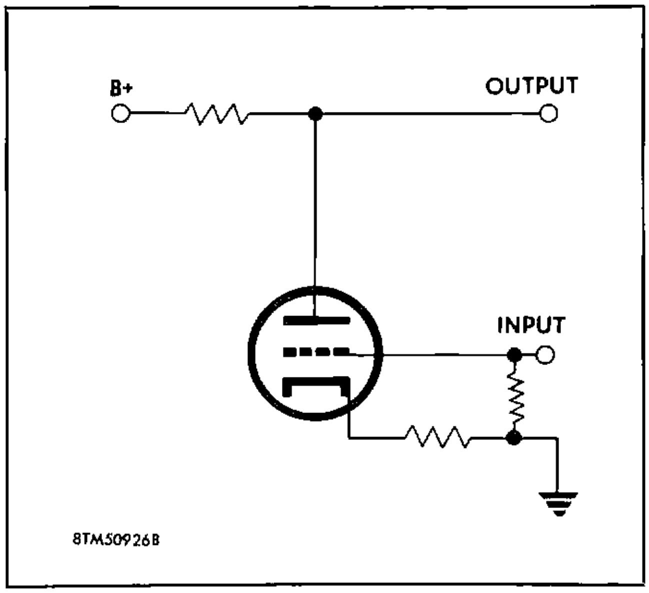

Figure 4-8. A Simple D-C Amplifier

You have seen that the grid voltage has a great effect on the flow of plate current in a triode. By varying the potential on the grid from a very negative to a slightly positive voltage, plate current flow varies from zero to a maximum. Thus, the grid may be considered to be a valve, and variations in its potential cause either an increase or decrease in the flow of electrons to the plate.

The Servo Motor of a Typical Damping System.

The term SERVO, as used in electronics, means a device that supplements a primary control operated by a comparatively feeble force. In the case of the damping

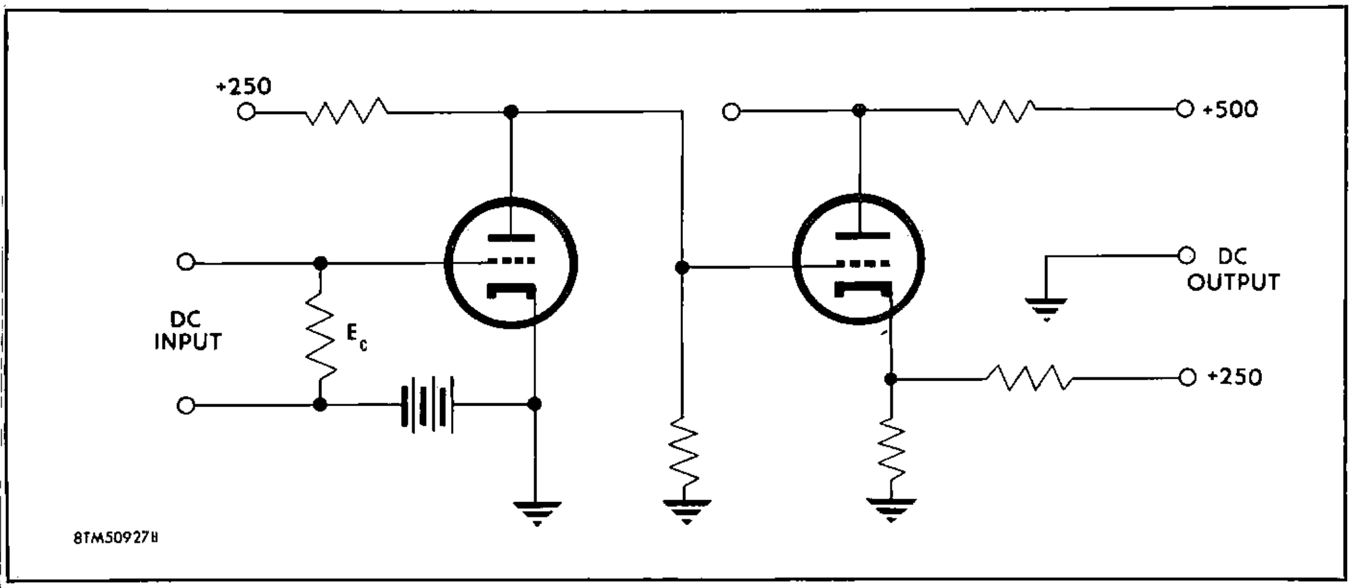

Figure 4-9. A Direct Coupled Amplifier

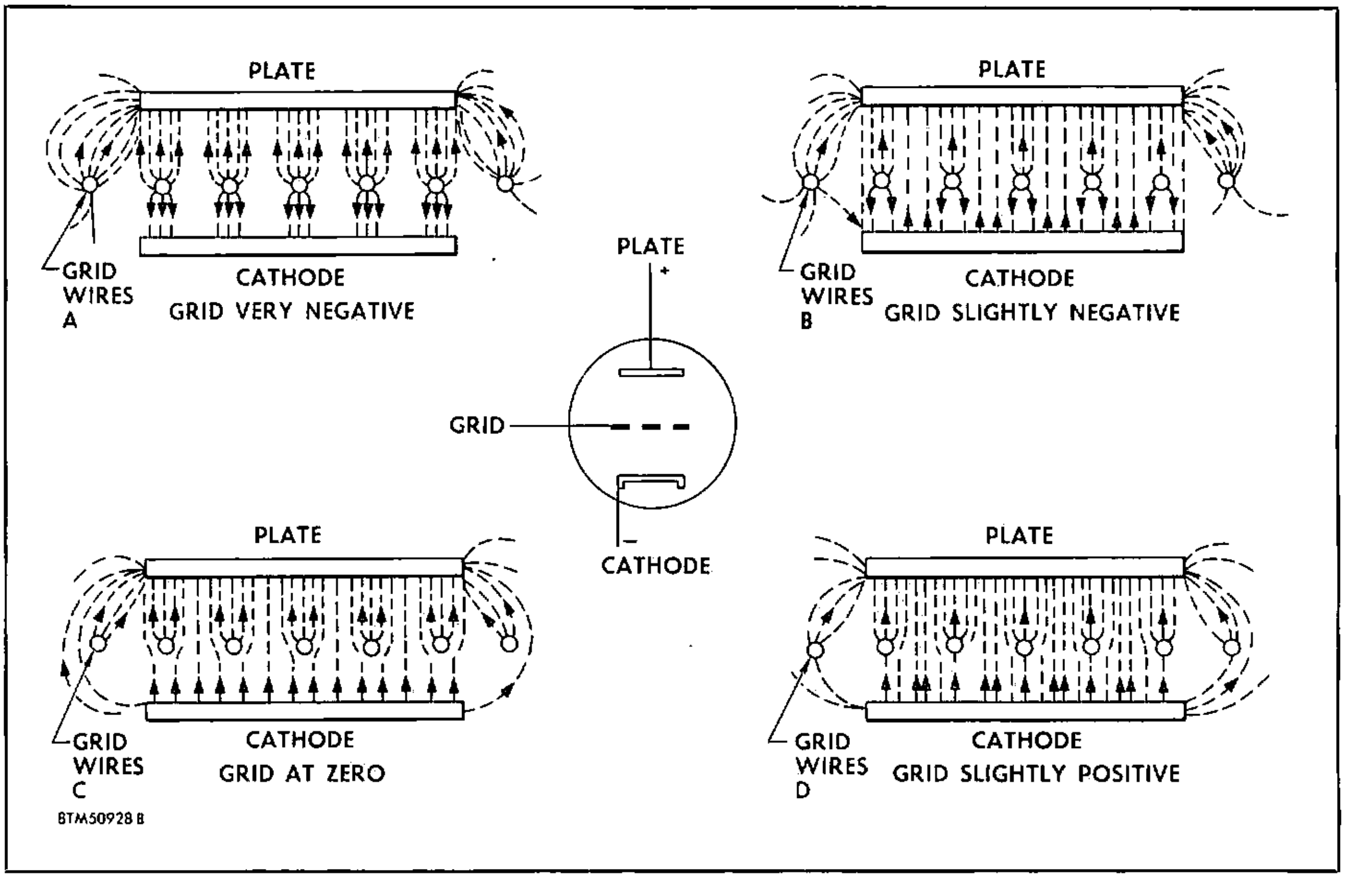

Figure 4-10. Operation of a Triode

system, the servo actuator torque motor causes the control surfaces to move as a result of the amplified error pick-off voltage from the rate gyro potentiometer. The primary control of servos is usually an automatic device for measuring position or voltage and the variations of this primary control cause the servo motor to respond. Thus, the primary control can be used as a correction or compensating device such as the rate gyro in a damping system.

The torque motor in the servo actuator (and the HEP valve in later flight control systems) corresponds to the servo motor discussed previously. Actually the torque motor is not a motor in the conventional sense; it is simply a flipper in the servo actuator that restricts the flow of hydraulic fluid through two ports within the value chamber. The flipper position is determined by differential current flow through coils in the torque motor.

You remember that a servo motor operates only in response to the error pick-off voltage from the primary control or the feedback signal from the potentiometer. Thus, it affects movement of the control surface only at a rate proportional to the deviation measured in the summing stage of the amplifier.

The Control Surface Feedback Circuit.

To make the conventional servo system complete, it is necessary to have a feedback from the control surface to the summing stage of the amplifier. Early versions of the flight control system incorporating the servo actuator have electrical feedback, as shown in figure 4-11. You can see that the feedback is located in the lower left portion of the servo actuator and moves according to the position of the actuator piston rod. Depending on the precession of the gyro, the output voltage from the rate gyro pick-off will fall within the maximum positive and negative voltage of the potentiometer. As you know, the voltage from the rate gyro is placed on the grid of the summing stage tube in the amplifier.

Later versions of the flight control systems which incorporate the HEP valve do not require an electrical feedback. A unique mechanical spring-centering device in the torque motor chamber causes the control surface to return to neutral as the signal from the rate gyro to the torque motor decreases. In the damper system, the electrical and mechanical components are designed in such a manner that the feedback signal is of the same quantity, but of opposite polarity (180° out of phase) to the rate gyro signal. Assuming that the servo motor is phase-sensitive, the introduction of

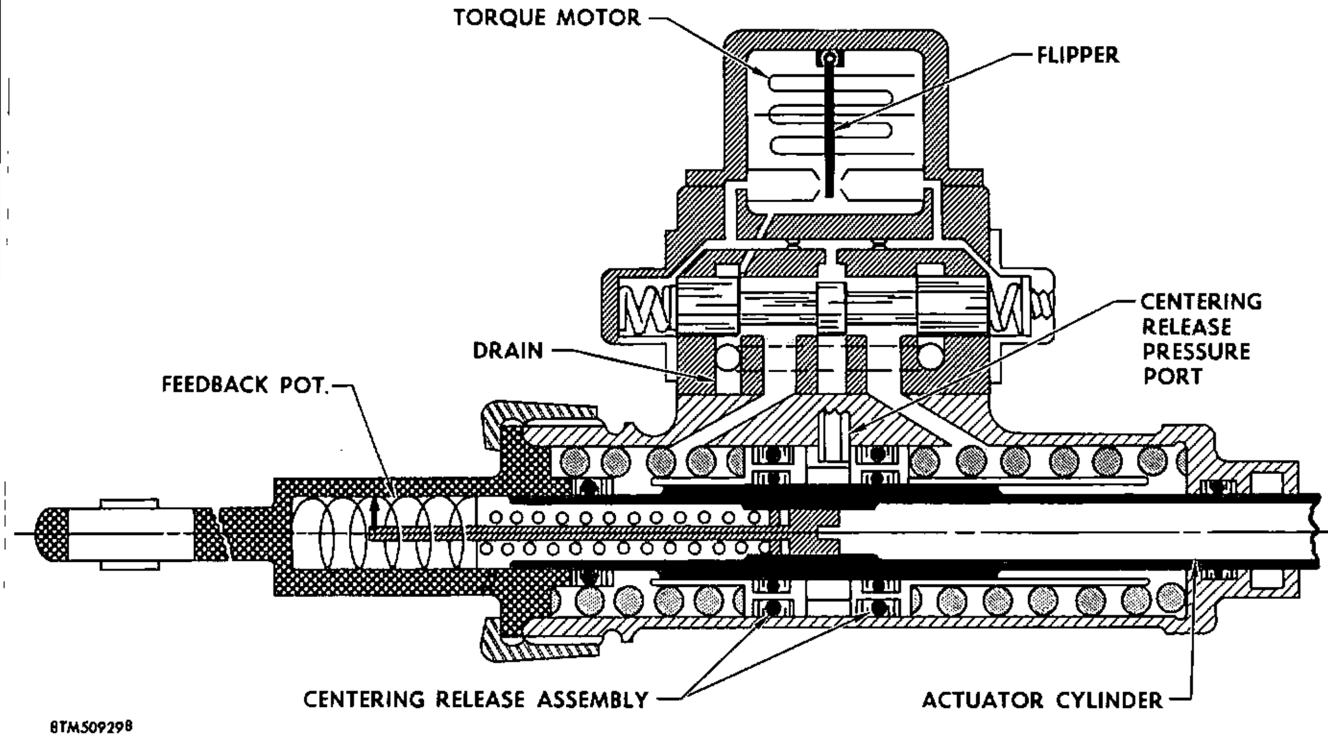

Figure 4-11. The Servo Motor of a Typical Damper System

this feedback signal causes the motor to drive in the opposite direction. This causes the control surface to move back to the original position. Referring back to the block diagram of a typical damper system (figure 4-11), you can see that there is a complete tie-in between the amplifier, the servo motor, the control surface, and the control surface feedback potentiometer. This tie-in forms a loop that is commonly referred to as a servo loop.

DAMPER SYSTEM POWER SOURCES AND DISTRIBUTION.

As you will learn later in this chapter, the damper systems require a variety of voltages to perform their function. These values are +200, —200, +105, —105 and +28-volts d-c and 200/115-volts, 400-cycle a-c with a 55-volt a-c pick-off to the monitor relays. In this section, you will learn of the power sources and the distribution of these voltages.

A-C POWER SOURCE FOR THE DAMPER SYSTEMS.

The 200/115-volt, 400-cps, phase B power for the damper system is taken from the FLIGHT CONTROL NON-ESSENTIAL BUS located in the main wheel well. What is meant by the 200/115-volt portion of the power description? As you know, most aircraft a-c power is obtained from 3-phase, 400-cycle alternators. In these alternators, there are three armature coils, each producing a separate phase of voltage. The alternator in the F-102A is rated at 208/120-volts output. If you examine the diagram in figure 4-12, representing the three armature coils, you can see how these various voltages are obtained.

The output voltages between T1 and T2, T2 and T3, or T3 and T1 will measure 208 volts. You will also find that the output between T1, T2, or T3 and ground is 120 volts. Note that there are three phases of voltage from the alternator—Phase 1 or (A), 2 (B), and 3 (C). Normally on schematics and bus identifications in the aircraft, you will see 200/115-volt with various phase designations. A wye-to-delta, 200/115-volt transformer is designed so that the 208/120-volt, 3-phase, 400-cycle a-c output can be tapped off as 115-volt, phase B and phase C. Thus, the voltage taken from the FLIGHT CONTROL NON-ESSENTIAL BUS (phase B), shown in figure 4-12, is a single phase voltage. For a more detailed account of the alternator, you should refer to the Electrical Maintenance Training Supplement in this series.

Why A-C Power Is Used in the Flight Control System.

The one direct use of the 115-volt, 400-cycle, a-c voltage in the damper systems is gyro power. Otherwise

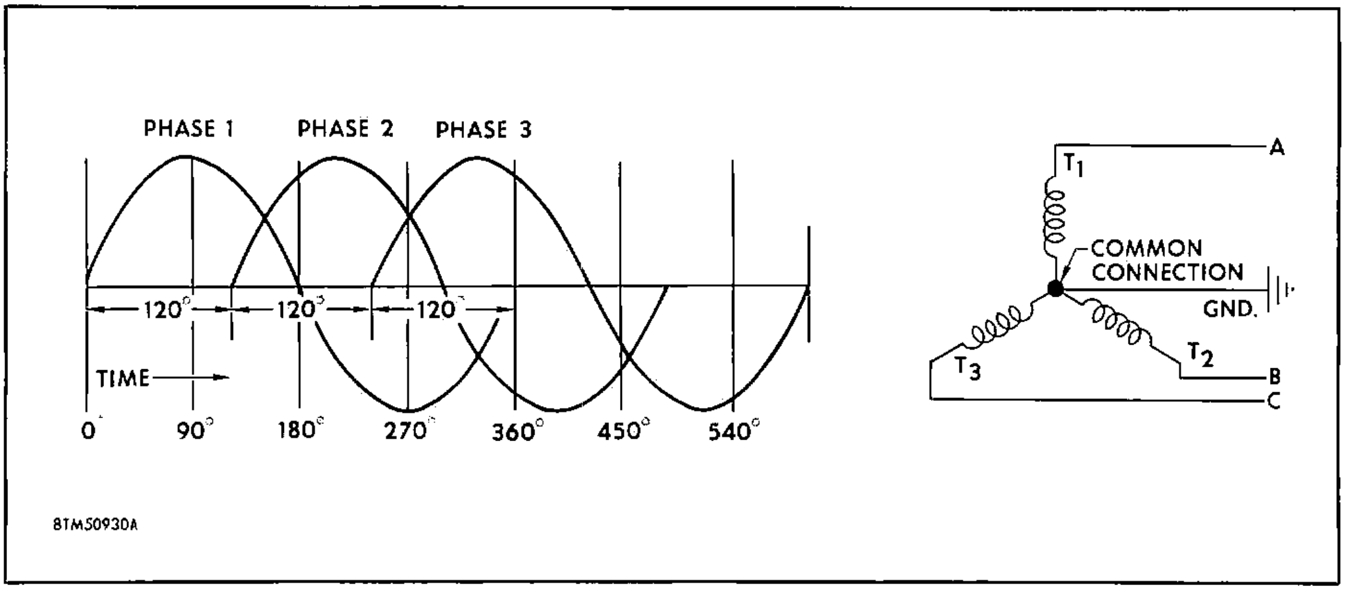

Figure 4-12. The Three Armature Coils of the Alternator

the a-c voltage is rectified to direct current. One of the outstanding features of alternating current is its versatility. It is far more practical to transmit high a-c voltages in aircraft than similar d-c voltages. There are several reasons for this—a-c line losses are not as high as d-c losses, and the size of components used in the system can be much smaller and therefore lighter in the a-c system. Another consideration is the fact that alternating current can readily be converted to direct current by means of rectifiers. Such a system permits many advantages—the most obvious being that relatively high a-c voltages can be generated at a remote source, transmitted efficiently to the using components, and converted to similar high d-c voltages in close proximity to their use. This is the most important application of a-c voltage you will find in the F-102A flight control system. There is a small heater voltage (6.3-volt a-c) taken from the rectifier transformer that is used for the tube filament circuits.

D-C REQUIREMENTS OF THE DAMPER SYSTEMS.

As you already know, the damper systems require five d-c voltages. Four of these voltages are high (±200 and ±105) while the remaining voltage is low (28). The four high d-c voltages are rectified a-c voltages while the 28-volt d-c is obtained from the aircraft's 28-volt, d-c non-essential bus. As you can see in figure 4-13, the output from the 28-volt bus—located in the main wheel well—is controlled from the FLIGHT CONTROL DAMPER circuit breaker.

Application of D-C Power in the Damper Systems.

Deviation signals from the rate gyros originate from a potentiometer with a center tap to ground. One end of the potentiometer connects to +105 and the other to —105-volts direct current. The signals from the rate gyro potentiometers are impressed on the grids of triode vacuum tubes. The plates of these tubes connect to a +200-volt d-c source (B+) while the cathodes connect to a —200-volt d-c line.

The +28-volt, d-c voltage is used to operate relays in the amplifiers. From this 28-volt d-c source there are various smaller d-c voltages derived through resistors. These are important only as they apply to individual circuits and are not used generally throughout the system.

The Source of the Four Large D-C Voltages.

Referring to the power distribution schematic, figure 4-13, you can see that the power amplifier is located in the same unit with the rudder amplifier. The power amplifier has the function of rectifying the 115-volt, a-c input into the four d-c voltages which you learned about earlier in the chapter. The 115-volt alternating current is changed into direct current through a twin diode, full-wave rectifier circuit.

Full-Wave Rectifiers.

Figure 4-14 shows a typical full-wave rectifier. It operates similar to the one used in the F-102A damper system. There are two important facts to remember about full-wave rectifiers:

1. The full-wave rectifier employs two diodes.
2. The high-voltage transformer secondary of the full-wave rectifier is center tapped (CT).

Note in figure 4-14 that the center tap is returned to ground and then through *RL* to the cathode of tubes *V1* and *V2*. Point 1 of the high-voltage winding is connected to the plate of *V1* and point 2 to the plate of *V2*. Thus the a-c voltage developed from 1 to *CT*

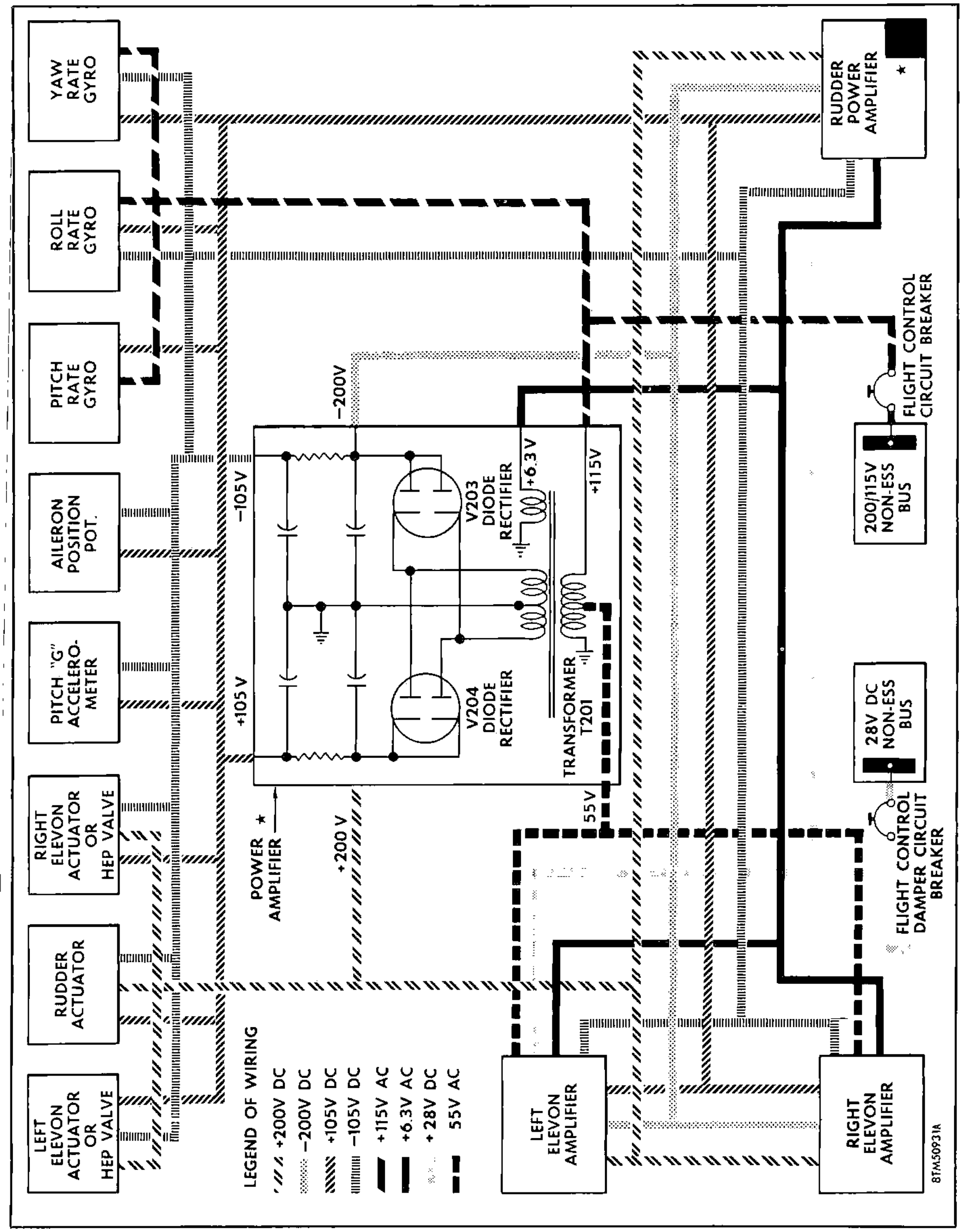

Figure 4-13. Power Distribution in the Damper System

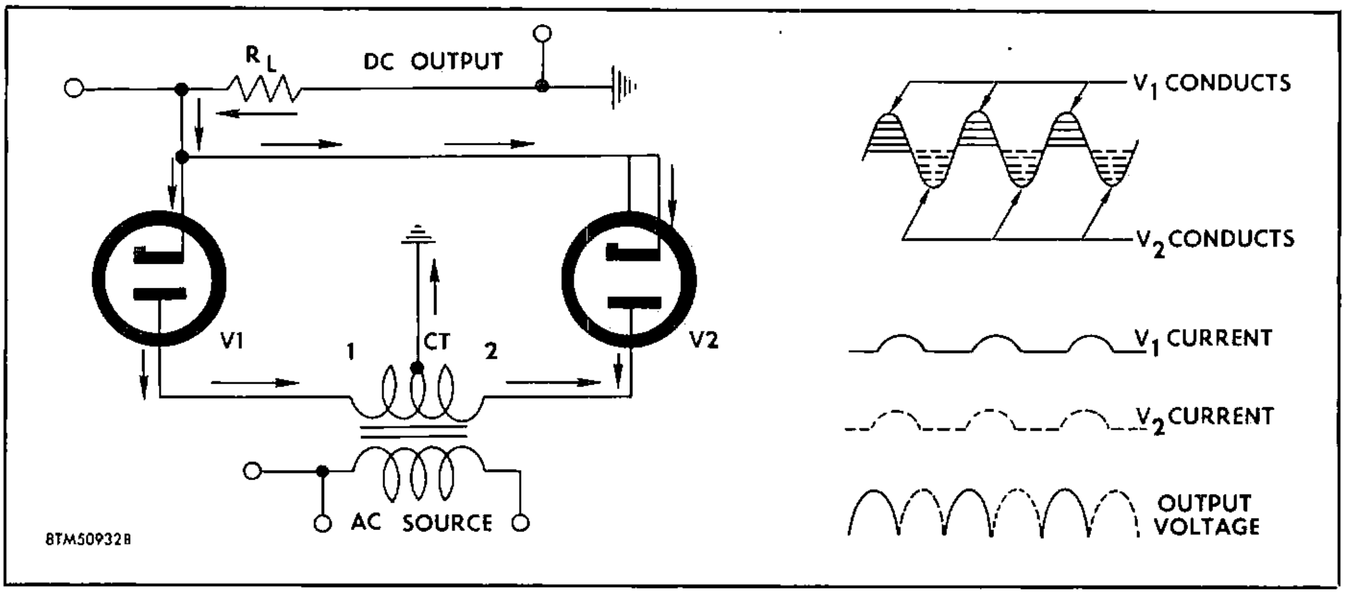

Figure 4-14. The Full Wave Rectifier

is applied across *V1* and the voltage from 2 to *CT* is applied across *V2*.

The two diodes conduct alternately, since at any given instant one plate is positive and the other negative. One-half cycle later the polarity of the voltages is reversed. The direction and path of current through *V1* is shown by the solid arrows and through *V2* by the dotted arrows. Both currents flow through the load resistor in the same direction. The output and input voltage waveshapes are shown in figure 4-14. Note that the frequency of the pulses is twice the frequency of the a-c source. If 60-cycle a-c is rectified by a full-wave rectifier, the ripple frequency is 120 cps.

In full-wave rectifiers, where one-half of the high-voltage secondary is connected across each diode, the d-c output voltage is proportional to one-half of the secondary's a-c voltage. On the other hand, in half-wave rectifiers, the entire voltage of the high-voltage secondary is applied across the diode. The high-voltage secondary winding of a full-wave rectifier must therefore provide approximately twice the a-c voltage to give the same d-c output. Applying the principles of the full-wave rectifier to the schematic on figure 4-13, you can see that the 200/115-volt, a-c voltage is first stepped up through transformer *T-201*. This transformer steps the voltage from its original 115-volt value to 185 volts. The secondary coil of the transformer is center tapped to ground, so that the phase B voltage (single phase) now splits into two components of an a-c voltage. The positive cycle goes to the cathode and plate half of each of the twin diodes, while the other one-half cycle goes to the cathode and plate of the remaining half of the twin diodes. This makes it possible to achieve full-wave rectification and develop two d-c voltages—one of —200 volts and the other of +200 volts. This end result of a ±200 volts is somewhat surprising considering the characteristics of the diode. You know that we placed 185 volts on the cathodes and plates of the diodes—and here we obtain a 200-volt output—so what is the explanation?

The Meaning of RMS.

In most instances, the value of alternating current or voltage is compared with direct current. This is done by finding the value of the sine curve of alternating current that will have the same power—or ability to produce heat—as a similar value of direct current. Thus you can see that the figure denoting quantity of voltage in an a-c circuit is neither the instantaneous nor the average value of the voltage.

From looking at the usual diagrammatic plotting (figure 4-15) of alternating current through a complete cycle (the sine wave) you might think it possible to find the average value of alternating current and use this figure for comparing it with direct current. It is not possible to do this because in the following power formulas — $P = I^2R$ and $P = \frac{E^2}{R}$ — you can see that power is proportional to the square of current (I^2), or the square of the voltage (E^2). From this relationship the term "Root Mean Square" (RMS) is derived to mean the same as effective value.

An example of the relation that exists between the maximum value of alternating current and its effective value can be found in the following example. Assuming the maximum value of an alternating current is 100 amperes, as measured with precision equipment, you would then find that it has the same heating effect as 70.7 amperes of direct current. We can thus make the statement that the effective value of a-c is 70.7% of the maximum value. This can also be expressed by saying the effective value equals the maximum value multiplied by 0.707. Thus, it can be seen that the maximum value is 1.41 times the effective value. Applying this principle to the voltages in the power amplifier, we find that 185 volts times 1.41 equal 260+ volts. Actually then, the alternating current reaching the cathodes and plates of the two twin diodes used in the rectifier has a maximum or peak voltage of 260+ volts.

If we put a capacitor in the output circuit that will charge on the upward swing of the cycle and discharge on the downward swing, you can see how the voltage is smoothed out and also the approximate relationship between maximum and effective voltage. Thus, the peaks of the direct current output of the rectifier is 260+ volts with an effective rating of 200. The resistors in the top of the schematic on figure 4-13 drop the ±200 volts to ±105 volts for use in the damper system circuits.

DAMPER SYSTEM ENGAGE CIRCUITRY FOR EARLY MODEL AIRPLANES.

To enable the damper system on early production models of the F-102A to operate, hydraulic fluid under pressure must act on the hydraulic servo actuators. Unlock solenoids control the flow of the hydraulic fluid to these servo actuators. In the schematic on figure 4-16, you can see that a 28-volt, d-c signal causes the solenoids to unlock. When the servo actuator does receive the hydraulic fluid, the damper signals can operate through their control of the flipper valve in the servo actuators. As you will soon learn, there are two unlock solenoids in the engage circuitry. One of these is called the rudder unlock solenoid and controls the rudder servo actuator. The other is the elevon unlock solenoid and controls both elevon servo actuators.

The circuit in figure 4-16 shows the complete damper engage circuitry. You can see how these circuits operate from the following description. You know the rudder and elevon unlock solenoids located in the upper right corner of the schematic are the critical units in the engagement of the damper systems. The first problem then is to find out how these particular units are energized.

In the upper left corner of figure 4-16, you can see that 28-volt, d-c power enters the damper engage circuitry through the FLIGHT CONTROL DAMPER circuit breaker. The problem is to route power from this circuit breaker to the unlock solenoids. In the aircraft this circuit breaker is located on the forward bulkhead of the main wheel well. When the power enters the engage circuitry through the FLIGHT CONTROL DAMPER circuit breaker, the next control point is a switch in the DAMPER ENGAGE relay, *K201*. This relay is located in the RUDDER-POWER amplifier unit on the aft electronics compartment door.

In order to energize the DAMPER ENGAGE relay, you can see that there must be a path to ground from relay K201 through contacts of relay K305 and K304. These relays, *K305* and *K304*, are the B+ monitor relays—they are located in the right elevon amplifier. These relays will automatically close, provided you have closed the FLIGHT CONTROL DAMPER 28-volt, d-c circuit breaker and the FLIGHT CONTROL 115-volt, 400 cps circuit breaker (and the power supplies are delivering the correct voltages to these circuit breakers). The energizing of these relays has provided a ground for the solenoid of relay *K201*.

You can see that the EMERGENCY DIRECT MANUAL relay must also be energized in order to deliver power to the solenoid. To supply power to the EMERGENCY DIRECT MANUAL relay, you must actuate the DAMPER ENGAGE switch in the cockpit. This switch is shown in figure 4-17. This is a manually operated device; however, when you actuate the DAMPER ENGAGE switch, you can see that it remains locked or held in position due to the action of a solenoid. You must also provide a bypass around the elevator limit switch through a set of contacts on the FULL AUTHORITY engage relay No. 2, *K202*. These contacts will normally be closed when engaging the damper system, because *K202* cannot be energized until after the damper system is engaged.

You must also allow the contact of the 60 second time delay relay to close. This relay is a temperature sensitive relay that takes one minute for the bimetallic contacts to heat and close the circuit from the 28-volt, d-c FLIGHT CONTROL DAMPER circuit breaker.

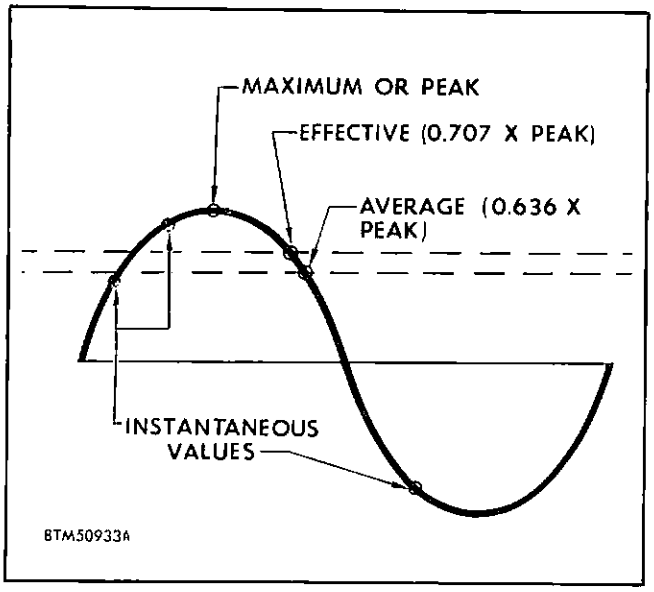

Figure 4-15. Values of A-C Voltage

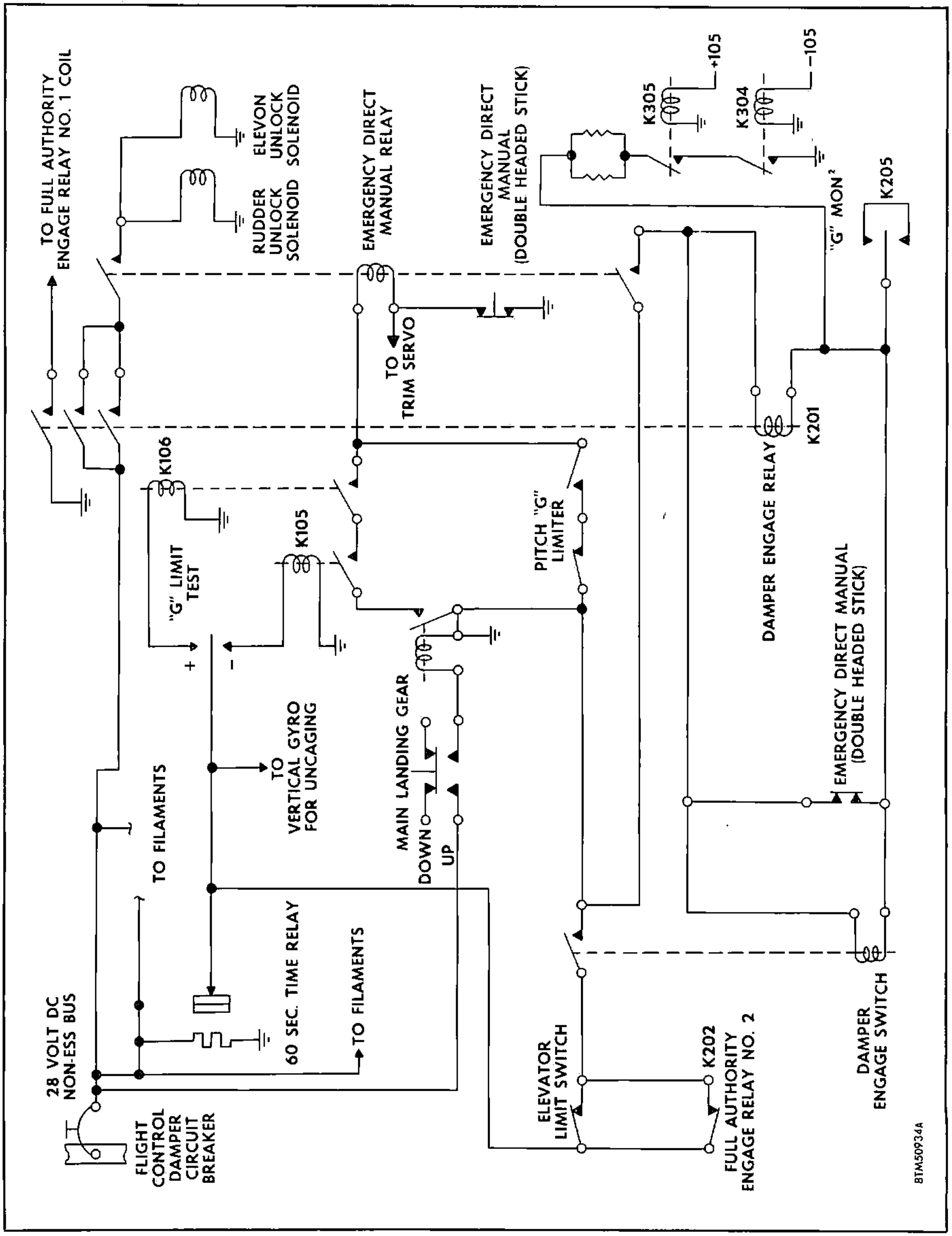

Figure 4-16. Damper Engage Circuitry for Early Model Flight Control System

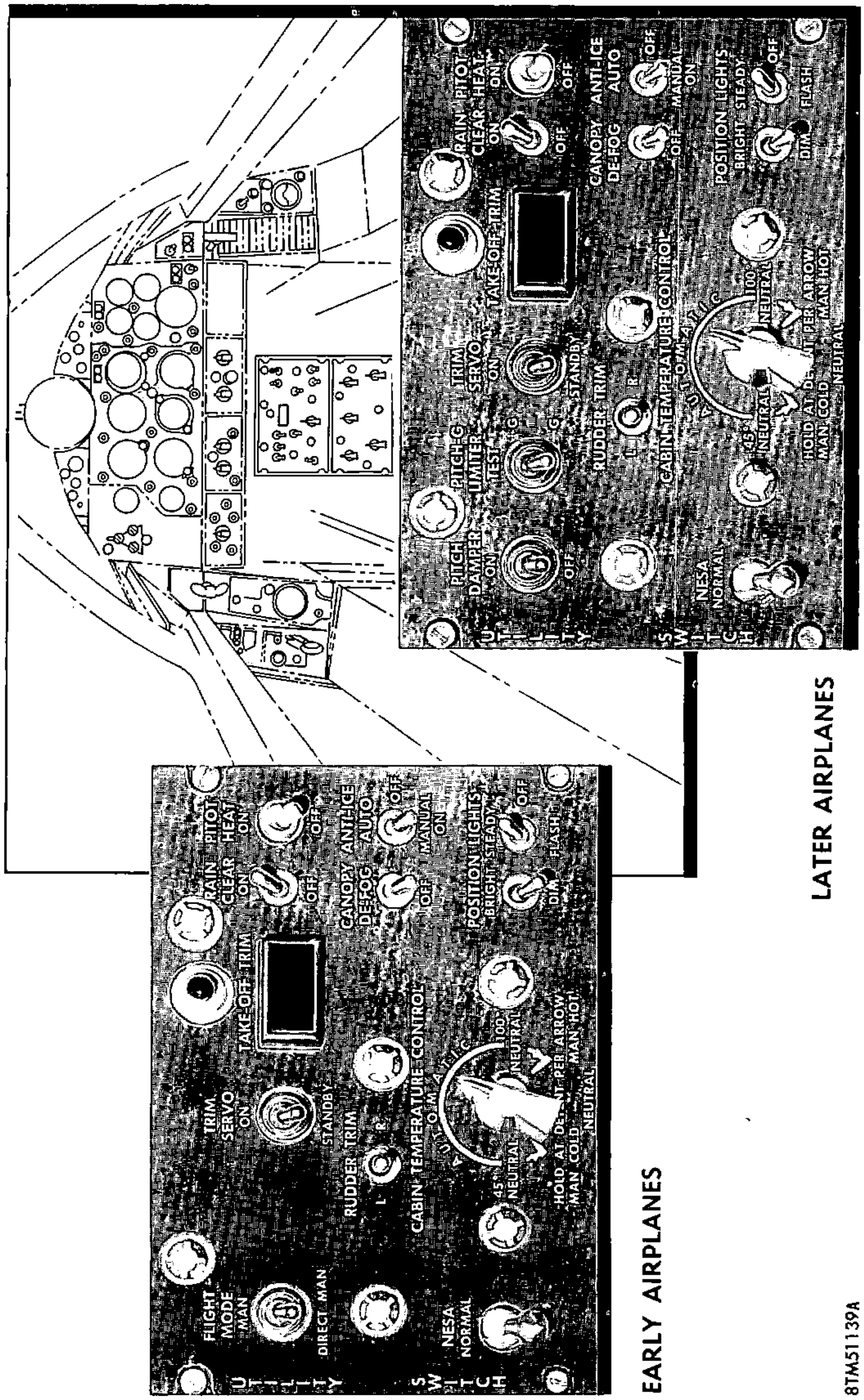

Figure 4-17. The Damper System Engage Switch

When these contacts close, you can see that a path has now been provided from the circuit breaker; through the time-delay relay; around *K202;* through the DAMPER ENGAGE SWITCH; through a switch in the EMERGENCY DIRECT MANUAL relay; through the DAMPER ENGAGE relay, *K201;* and completed to ground through the B+ monitor relays *K304* and *K305.* With the exception of maintaining the contact of the "G" monitor relay *K205* in the open position, this circuit results in the energizing of the DAMPER ENGAGE relay. The only problem left is to energize the EMERGENCY DIRECT MANUAL relay.

Looking now at the circuit from the FLIGHT CONTROL DAMPER circuit breaker to the solenoid of the EMERGENCY DIRECT MANUAL relay, let us see how the EMERGENCY DIRECT MANUAL relay is energized. Notice in the schematic that the ground for the EMERGENCY DIRECT MANUAL relay is through the EMERGENCY DIRECT MANUAL switch located on the control stick. When the switch is placed in the energized position, the circuit is completed to ground. You can see from the schematic that the contacts of the pitch "G" limiter or their bypass network must remain in the closed position. This is no problem as they will remain closed if the main landing gear is extended. You can also see that the DAMPER ENGAGE switch must be actuated. This connects the circuit between the ELEVATOR LIMIT switch and the PITCH "G" LIMITER network. Now follow the circuit from the bypass around the elevator limit switch via a set of contacts on the FULL AUTHORITY relay No. 2, *K202.* This bypass permits the damper system to engage without having the PILOT ASSIST system engaged at the same time. When the 60 second time delay relay has closed its contacts, power is furnished to the EMERGENCY DIRECT MANUAL relay, and the damper system is operational.

DAMPER ENGAGE CIRCUITRY FOR LATER MODEL AIRPLANES.

In the damper engage circuitry of later model F-102A airplanes, the pitch and yaw damper systems are engaged separately. This was not the case with the early damper systems that were engaged simultaneously from the same damper engage switch. When engaging the systems separately, there is a definite sequence of engagement between the Yaw, Pitch, and AFCS systems. We will begin the study of the damper systems with the engagement of the yaw damper system.

YAW DAMPER ENGAGE CIRCUITRY.

The YAW DAMPER ENGAGE switch is a manually actuated switch on the "105" box located on the lower mid-section of the instrument panel, figure 4-18. The switch is spring-loaded to the normally open position. However, once it is actuated to the ENGAGE position, it will remain engaged until the holding relay is de-energized. Note in figure 4-19, that 28-volt, d-c power from the damper circuit breaker is supplied to the switch. You can see that the yaw damper engage switch must be closed before power can continue through the circuit to the EMERGENCY DIRECT MANUAL (EDM) relay. The ground for the EDM relay is through the EDM switch. This switch is in the normally closed position except when the pilot disengages the system by depressing the EMERGENCY DIRECT MANUAL switch on the control stick, thus opening the yaw damper engage circuit.

Note also, that the yaw damper system must be engaged before the TRIM SERVO ENGAGE switch connects power to the trim servo circuits from the EDM relay. When the EDM relay energizes, power is supplied to the holding solenoid of the YAW DAMPER ENGAGE switch and the solenoid of the damper engage relay. A ground for the solenoid of the damper engage relay and the holding solenoid of the yaw damper engage switch is provided through the +105 and —105-volt, d-c monitor relays located in the elevon amplifiers. These two relays operate only if +105 and —105-volts d-c are available to the damper system from the rudder-power amplifier. When all of the previous items are operating properly, the yaw damper system will engage.

PITCH DAMPER ENGAGE CIRCUITRY.

The yaw damper system must be engaged before the pitch damper system will engage. Following the circuit from the DAMPER circuit breaker, note that the damper engage relay and the EDM relay must be energized before power can reach the PITCH DAMPER ENGAGE switch. Also note that the rudder unlock solenoid is energized when these two relays are energized.

The PITCH DAMPER ENGAGE switch is a manually actuated switch located below the YAW DAMPER ENGAGE switch. It occupies the same position as the DAMPER ENGAGE (Flight Mode) switch of early model F-102A aircraft (see figure 4-17). When the switch is actuated, note that power is connected to the right and left HEP valve unlock solenoids. These solenoids must be energized to permit hydraulic fluid to enter the chamber around the torque motor flipper. Unless pressurized hydraulic fluid surrounds the flipper, signals from the pitch damper system will have no effect on the control surfaces.

The PITCH DAMPER ENGAGE switch is also spring-loaded to the normally open postion. When you manually depress the switch, power connects to a holding solenoid that retains the switch in the engaged position. The switch will remain in the engaged position until the yaw damper system disengages or the "G" limit switch in relay *K105* fails to energize when the full authority relay No. 2 energizes.

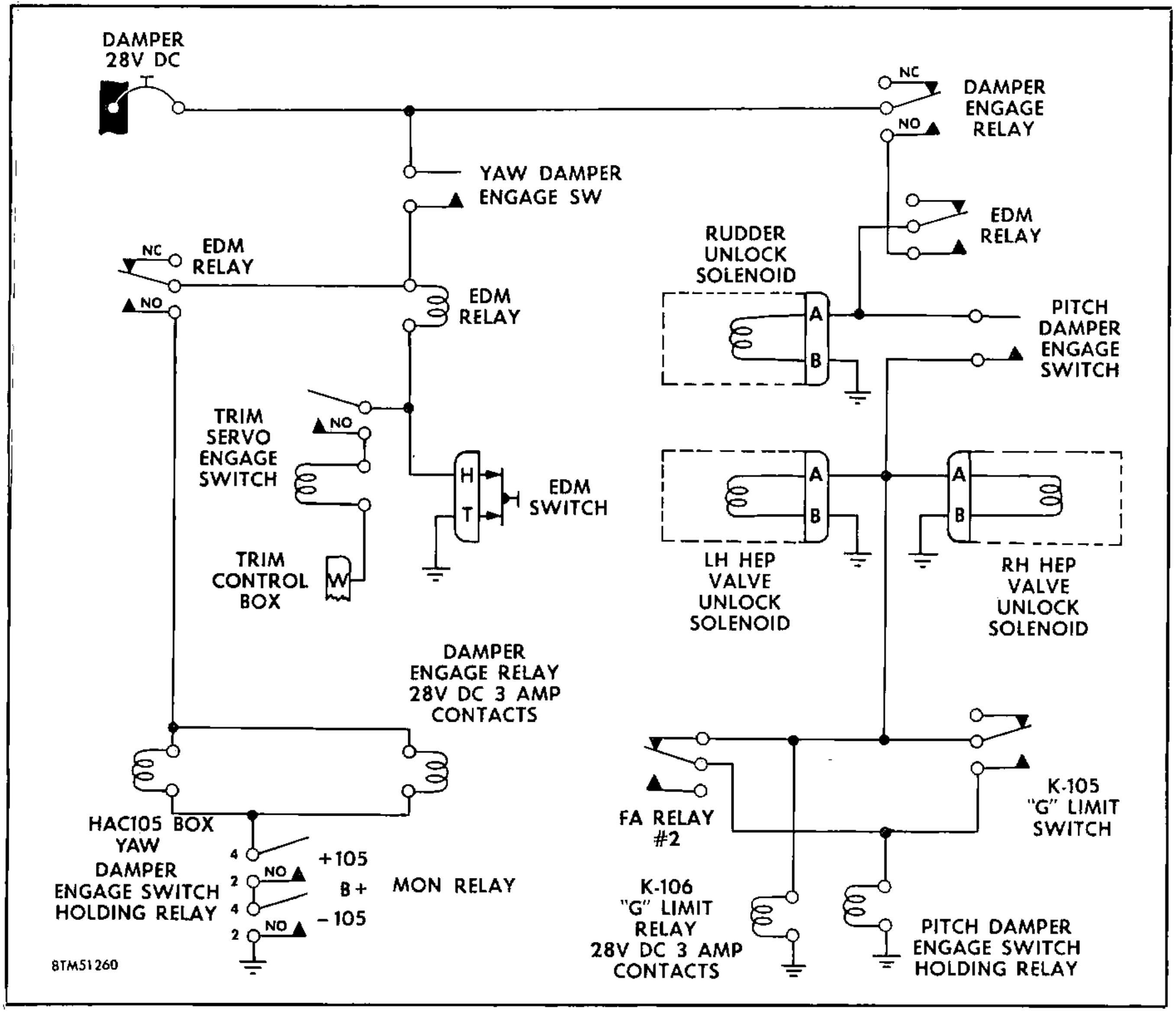

Figure 4-18. Damper System Controls

The Full Authority relay is energized only when the AFCS switch is actuated. The "G" limit relay *K105* is also energized only when the AFCS is engaged. Thus, you can see that the energizing of the full authority No. 2 relay in AFCS mode has no effect on the holding relay of the PITCH DAMPER ENGAGE switch because the circuit is completed through the "G" limit switch. Note also, that relay *K106* is energized when the pitch damper system engages. When you study the AFCS engage circuitry in Chapter V you will find that this relay is critical to AFCS engagement.

THE YAW DAMPER SYSTEM.

In the previous portions of this chapter you learned the theory and operation of a typical damper system. You also became acquainted with the power distribution and means of engaging the damper systems. Having become familiar with the background of damper system operation, you are now ready to learn specifically how the damper systems perform their function. Let us first examine the yaw damper system.

The purpose of the yaw damper system is to control the directional stability of the aircraft. Since the rudder controls the direction of the aircraft, the end result of the yaw rate gyro signals is the positioning of the rudder control surface. Auxiliary components, known as the variable parameter yaw damper, perform the additional function of automatic turn coordination with the ailerons and rudder acting together. To distinguish between these two systems, the primary system is known as the basic yaw damper system and the auxilary system as the turn coordinator.

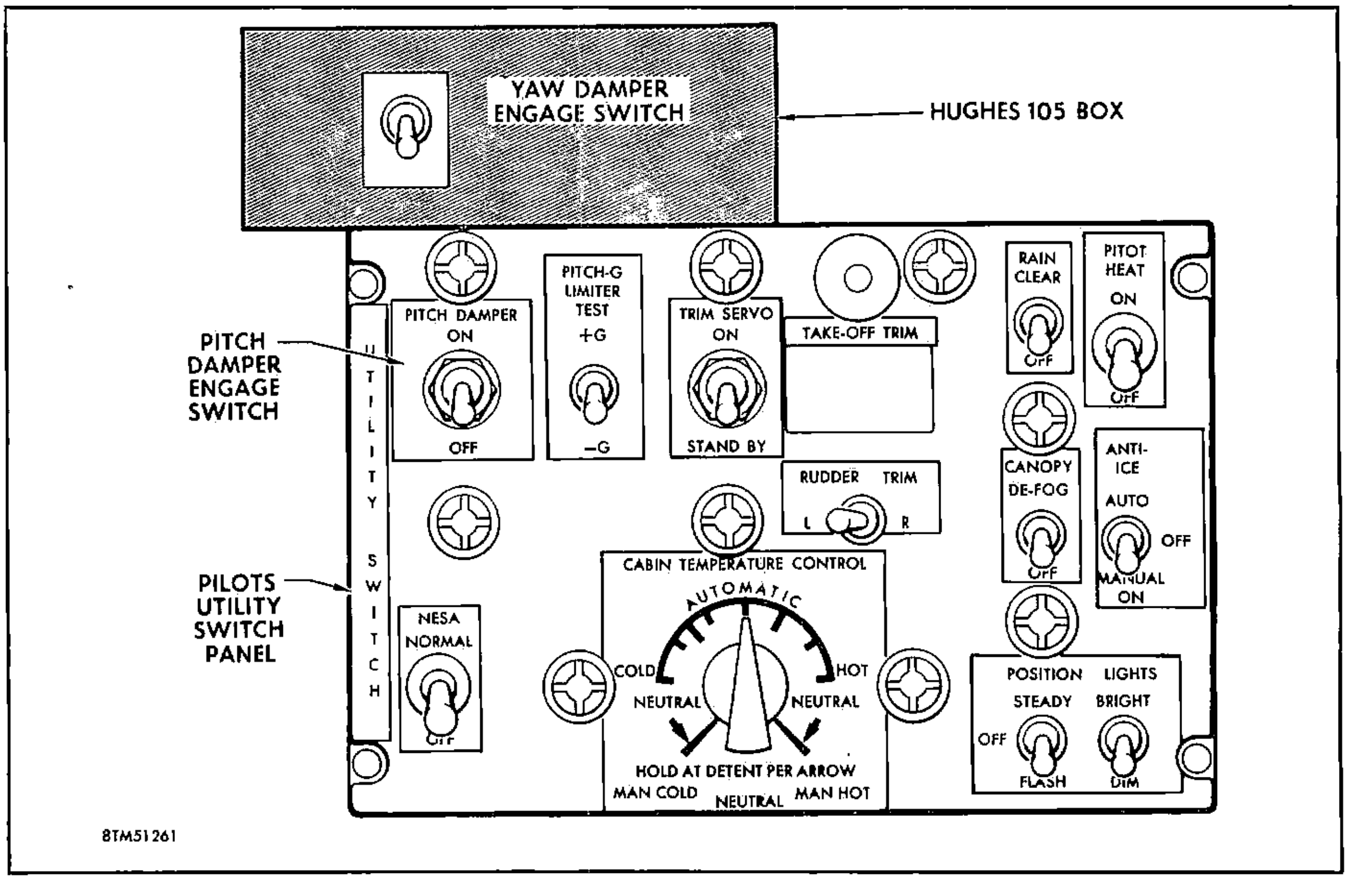

Figure 4-19. Damper Engage Circuitry for Later Model Flight Control Systems

COMPONENTS OF THE BASIC YAW DAMPER SYSTEM.

The block diagram in figure 4-20 shows the components of the basic yaw damper system. As you can see, the most important components are the yaw rate gyro, the high-pass network and filter, the amplifier, and the servo actuator with its torque motor and feedback potentiometer. Note the similarity between the basic yaw damper system and the typical damper system you studied earlier in this chapter. The yaw rate gyro originates the signals for this system, so let us begin our study with this unit.

The Yaw Rate Gyro.

The yaw rate gyro is located in the main wheel well. It receives its power directly from the 115-volt, 400-cycle, a-c power line and the potentiometer receives +105 and —105 volts direct current. The gyro location and external features are shown in figure 4-21.

As you can see in figure 4-22, the gyro is mounted on the top of the main wheel well. It is a small sealed unit held to the airframe by four self-locking nuts—there is only one electrical connection to make to this unit. You can also see the roll rate gyro which is mounted on the forward bulkhead on the main wheel well. Both of the gyros are quite similar in external appearance.

Internally the yaw rate gyro consists of an umbrella-type rotor mounted in a spring-retained gimbal. As the gyro precesses during aircraft deviations, a wiper arm attached to the gimbal moves either up or down on a center tapped potentiometer. Since the center tap is at ground potential, the gyro pickoff signal to the amplifier may be either a positive or negative voltage. Thus, you can now understand what is meant by varying voltages on the grids of the summing tubes. As an item of interest, the gyro rotor operates at a speed of 18,800 rpm and requires only 70 seconds to reach this speed.

Note that the yaw rate gyro initiates a signal that goes to the high pass network and filter. Let us examine the circuits of this network before continuing with the other sections of the yaw damper system.

The High Pass Network and Filter.

The schematic, figure 4-22, shows the high pass network and filter that is part of the damper circuit located in the amplifier. As you have already learned, the purpose of this network is to filter out "noise" (interferences) and signals of undersirable frequency as well as reduce a steady yaw rate signal in a steady turn to zero. As a result of the network, the damper does not "fight" a turn produced by the pilot's movement of the controls.

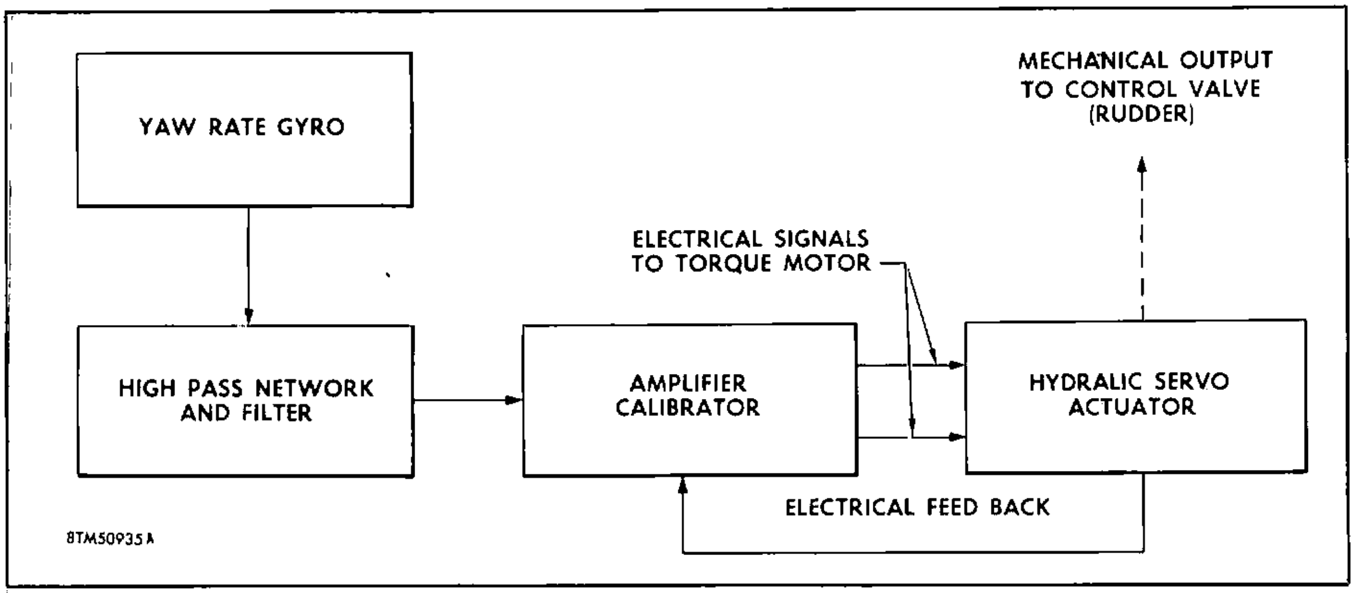

Figure 4-20. The Basic Yaw Damper System

As a deviation signal from the gyro builds up on capacitor *C1*, an apparent current flows through the capacitor. When the signal reaches maximum strength, the apparent current flow stops and the voltage on the capacitor (*C1*) decays as current bleeds through resistor *R1* to the ground. The values of the resistance and capacitance determine the time required for the decay of the signal. Thus, a steady yaw rate signal in a steady turn causes a constant signal to be sent from the yaw rate gyro. A steady signal cannot be felt through the high pass network, so the damper does not fight a pilot controlled turn.

The filtering portion of the network consists of resistor *R2* and capacitor *C2*. The value of *R2* and *C2* are such that a signal frequency above 15 cycles per second is greatly attenuated or filtered. This action eliminates signals from the rate gyro due to vibration of its mounting structure. Consequently, the flight controls do not jitter as a result of the high frequency signals.

Referring to the yaw damper system schematic in figure 4-23, locate the potentiometer mounted on the gyro case. Note that +105 volts is applied to one end of the potentiometer while —105 volts is applied to the other end. When the gyro is not precessing, the spin axis is perfectly level with the case, and the voltage from the potentiometer wiper is zero because there is a center tap ground on the potentiometer. Therefore, the potential on the grid of summing tube *V201* is zero. As you know, an oscillation of the aircraft about its vertical axis will cause the gyro to precess either up or down. You can see this causes a varying voltage from the potentiometer to be sent to the amplifier.

Note also, the gyro motor that receives power from the 115-volt, a-c power line. The heater shown in the schematic, figure 4-23, maintains the entire operating components at 75°(±10°)F for proper operation.

The Servo Actuator Torque Motor and Feedback Potentiometer.

It is not necessary to picture again the servo actuator torque motor and feedback pot as they were shown earlier in the chapter in the discussion of a typical servo motor and feedback. Referring to the yaw damper schematic on figure 4-23, note that the torque motor schematically appears as two coils coming together in a vee. Effectively these coils act much the same as the solenoid in a relay, because a differential current flow between the coils will position the flipper in the servo actuator. The bottom of the vee is connected to the +200-volt, d-c line—the other two ends of the coils connect to the plates of the amplifying stage of the damper amplifier.

On the inside of the vee, the flipper valve is in a neutral position when the current flow is equal in both of the coils. Thus, the flow to both sides of the control valve in the servo actuator is equal. However, when the signal on either of the grids of the summing amplifier causes a differential current flow in the plates of the amplifier, the current on one coil of the torque motor will increase and the other will decrease. The differential between these currents in the two coils permits the actuator torque motor to position the flipper, thus regulating the pressures to either side of the control valve.

The feedback pot is located in the case of the servo actuator, and the movement of the servo actuator

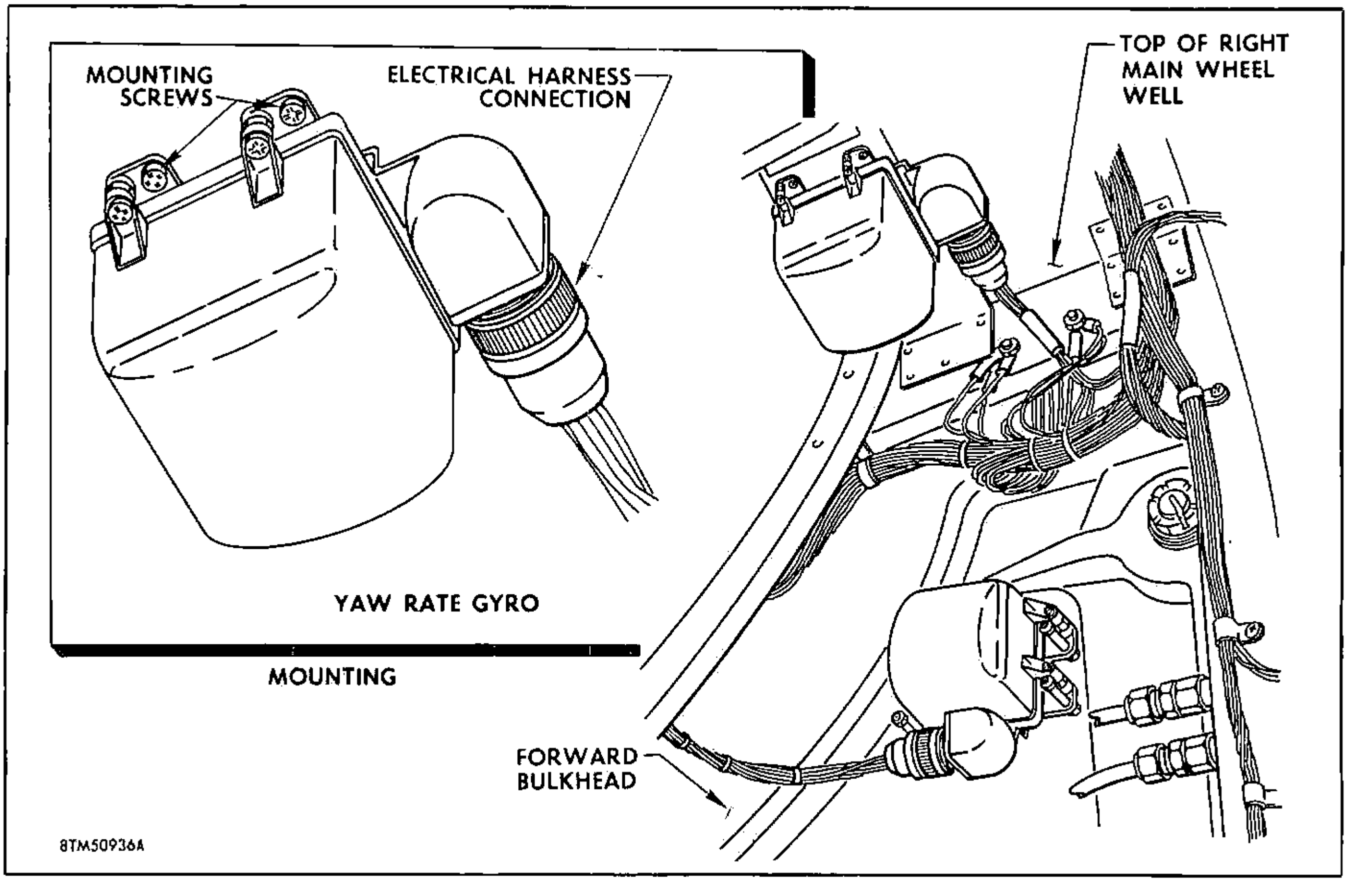

Figure 4-21. The Yaw Rate Gyro

positions the wiper proportionally to the servo actuator movement. This feedback pot is center-tapped to ground, permitting the wiper to pick off either a negative or positive voltage, depending on the position of the control surfaces.

THE CIRCUITS OF THE YAW DAMPER SYSTEM.

The schematic on figure 4-23 shows the basic yaw damper system. Starting at the top right corner of the schematic, note the yaw rate gyro potentiometer. This potentiometer connects to the plus and minus 105-volt, d-c power circuits you studied earlier in this chapter. When the gyro precesses, you can see that it is possible for the wiper to pick off either a plus or minus voltage. The direction of precession depends of course, on the deviation of the aircraft. Let us say, for example, that when the aircraft yaws to the left, the gyro precesses in such a manner that the wiper picks off a negative voltage. A yaw to the right would thus result in the wiper arm picking off a positive voltage.

Now let us picture what happens when the aircraft yaws to the left. The wiper picks off a negative voltage and the first thing in the circuit you encounter is the high pass network and filter. The capacitor and gain adjustment potentiometer perform the high pass functions of the circuit. As the signal from the rate gyro builds up on the gyro side of the capacitor, an apparent current flow occurs through the capacitor, then stops, and the voltage decays as current bleeds through the resistance to ground. The effect of these characteristics on the system is to govern the rate and time of the control surface response.

The next resistor and capacitor in this circuit compose the filtering portion of the yaw rate circuit. The values of these components are such as to attenuate (diminish) and pass to ground signals above 15 cycles per second. The controls could not respond to signals above this frequency with effective results; thus, the reason for incorporating this network into the system.

Before applying the positive signal to the left grid of tube *V201*, let us examine the condition of the circuits in the summing and amplifying stages of the schematic. You know that in a normal condition, the coils of the torque motor should have the same voltage on each of the coils. Under this condition, the flipper valve in the servo actuator will remain centered. Tracing the circuit backward from the coils of the torque motor, you can see that the potential of both plate circuits from tube *V202* would have to be equal for the torque motor to maintain equilibrium. The potentials of each of the plate circuits would in turn be governed by the inputs to the grids of *V202*. Following the grid circuits back to their origin, you find that they connect to the plates of tube *V201*. It should not be necessary to explain further how the equal plate potentials of this tube affect the torque motor coils.

When there are no inputs to the grids of tube *V201*, the outputs from the tube must be equal; otherwise the grids of *V202* will receive unequal signals and differential plate currents cause the torque motor coils to position the flipper to one side or the other. From practical experience, you know that tube manufacturers can not assemble components so that both sides of the tube will conduct equally. Therefore, you will find a zeroing switch (at the lower right of tube *V201)* that grounds all inputs to the grids of tube *V201*. Now locate the microammeter in the cathode circuits of tube *V202*. This meter is designed to detect the balance of current flow in the cathode circuits of *V202*. If there is a differential flow through *V202* as a result of *V201* not conducting equally, the zeroing pot in the left grid circuit of *V202* permits you to compensate for the difference in tube and circuit characteristics.

You can operate the zeroing switch from a push button on the front of the amplifier. Located adjacent to the push button is the microammeter. If the meter indicates an unbalanced condition in the tubes while you are depressing the zero switch, you can balance the circuit by adjusting the zeroing knob on the front of the amplifier.

Picking up the negative yaw signal from the rate gyro circuit and following it through the high pass network and filter, let us see what happens when the signal reaches the right grid of the *balanced* amplifier network. A negative voltage on the right grid acts as a repelling force to the electron flow between the right cathode and plate. A reduction of electron flow naturally results in less current flow, but it causes a greater voltage drop between the plate and cathode than previously existed. Therefore, the plate circuit is more positive than it originally was.

The decrease of current flow in the right plate circuit should cause a decrease in the cathode potential. However, the value of the resistor between the two cathodes of tube *V201* is relatively small. Therefore, the potentials on the two cathodes are nearly equal at all times.

The tendency for the cathode potential to change results in an increased current in the right half of the tube that is approximately equal to the decrease in the left half. Thus, the cathode potentials remain relatively constant regardless of changes in the grid potentials. The increased current flow in the right half of the tube results in a lower potential in the right plate circuit. This decreased potential means that the plate circuit is now more negative than before.

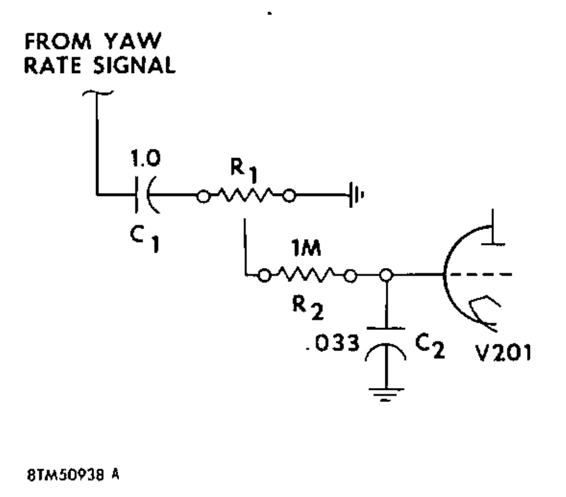

Figure 4-22. The High Pass Network and Filter

With the negative signal input, you can see that the left plate potential increases and the right plate potential decreases. These changes are reflected on the grids of *V202*. Since the left plate circuit of *V201* that connects to the left grid of *V202* became more positive due to the increased potential, the left plate of tube *V202* has a higher current flow. Consequently, the left coil of the torque motor receives more current than formerly. The opposite condition exists for the right plate circuit of *V202* because of the factors already considered. The differential currents in the torque motor coils results in the positioning of the flipper valve in the servo actuator.

When the differential current in the torque motor causes the flipper in the servo actuator to move, the servo actuator moves the control valve which then ports hydraulic fluid to the control surface actuator for positioning of the rudder. As the rod in the servo actuator moves, a wiper arm attached to the rod picks off a voltage that is proportional to the control surface movement. Notice in the schematic, figure 4-23, that the feedback goes to the left grid of the summing tube *V201*. You have already learned that the feedback potentiometer and wiper arm always send a signal opposite in sign to the original input signal.

You will recall the original signal from the rate gyro was a negative signal; so the feedback sigal is a positive signal. Let us see what effect this signal will have on the summing tube *V201*, and the associated circuits.

The servo actuator feedback pot will gradually send a signal to the grid as the servo acutator moves. When the control surface reaches a point that satisfies the rate gyro signal, the positive servo actuator feedback will have reached an equal but opposite sign signal

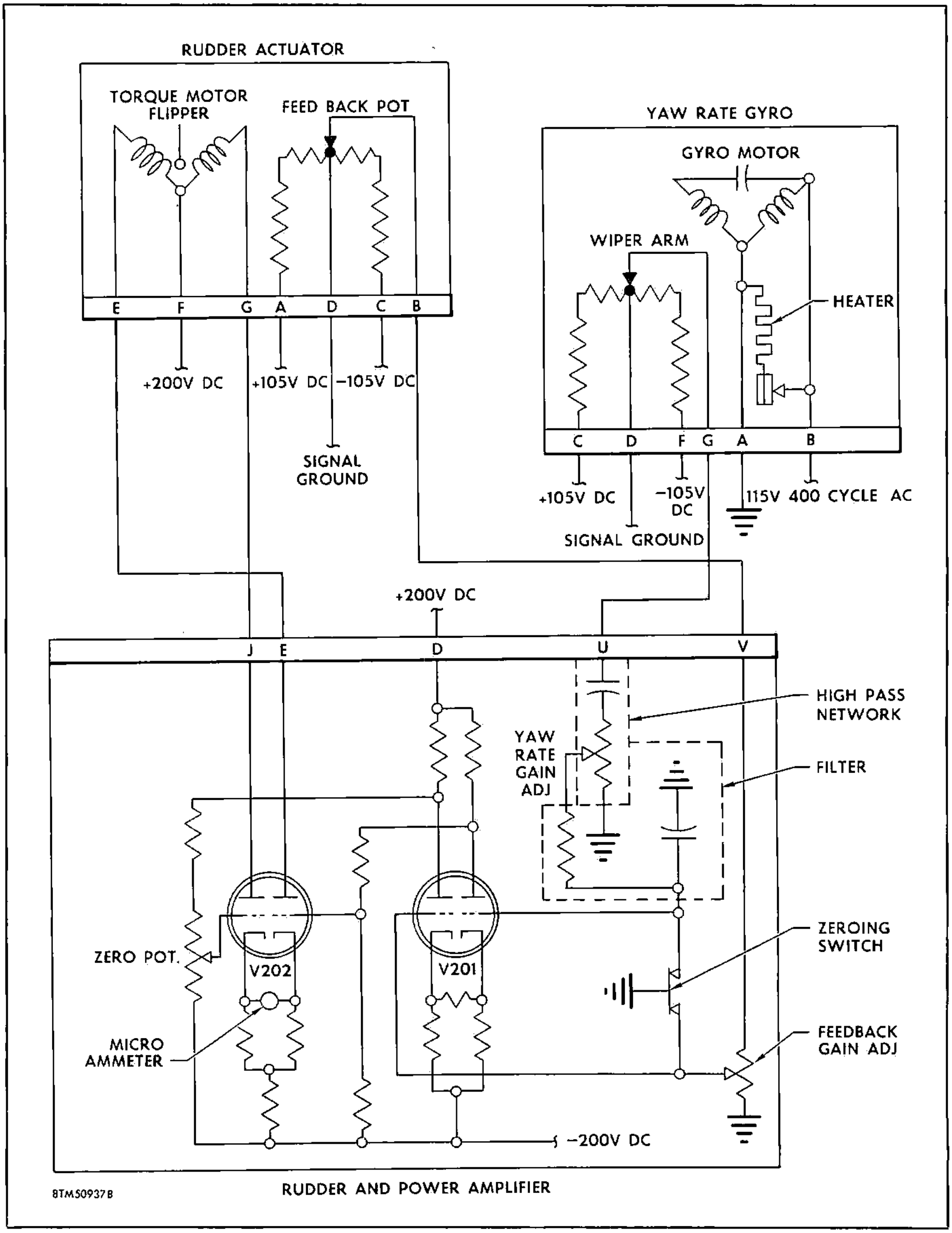

Figure 4-23. The Yaw Damper System Schematic

on the right grid. In this condition, what has happened to the electrical balance of the circuits? The right plate circuit of *V201*, which was formerly more negative that the left plate circuit, becomes even more negative due to the added positive potential from the feedback circuit. However, by this time the yaw rate gyro senses the yaw in the opposite direction brought about by the new position of the rudder. Therefore, it is sending a positive signal to the amplifier. The feedback signal is much more positive than the rate gyro signal at this time; so the differential between the two positive signals causes the torque motor coils to position the flipper valve in the other direction and thus, move the control surface back to neutral. As the surface moves back to neutral, a less positive signal is picked off the servo actuator feedback pot.

Also, because of the control surface movement, the rate gyro precesses back to the center position and the two positive signals now on the grids decrease until the surface is again at neutral. Thus, the original balanced condition exists in the amplifier, and the control surface remains neutral until the next deviation occurs.

THE RUDDER-POWER AMPLIFIER.

Figure 4-24 shows the location of the amplifier for the yaw damper system. As you can see, this unit is located on the aft electronics compartment door. Note the method of attaching the unit to the rack on the door. The pins at the bottom of the amplifier plug into a receptacle on the terminal strip at the base of the rack. At the top of the rack you can see a screw-type clamp that fits over a lip on the amplifier case. This device holds the top of the amplifier firmly to the rack, while the pins at the bottom retain the amplifier in the terminal strip receptacle.

The amplifier rack is shock mounted to a frame attached to the door structure. The terminal strip at the bottom of the rack has four plugs containing the input and output circuits to the other damper components. The tubes for the yaw and power amplifiers, as well as the relays for the damper engage and monitor circuits, are easily accessible for removal and replacement.

Note at the top of the amplifier, the small covered compartment containing the adjustments for the circuits of the yaw damper system. As you can see in the detail, there are eight knobs provided for the various adjustments. All of these controls are simple variable potentiometers in the circuits (labeled on the face of the control panel) and permit you to increase or decrease the strength of the signals. Also located at the top of the amplifier is the microammeter and the zeroing switch. This meter and switch are used together to check the electrical balance of the amplifier tubes; their operation will be explained later when you study the yaw damper circuits.

THE TURN COORDINATOR.

The turn coordinator consists of four major units shown in the block diagram on figure 4-23. Note that these units are not in the familiar servo loop pattern of the former systems. However, these units do connect to the basic yaw damper system and its servo loop functions the same as described in the basic yaw damper operation.

Note in the block diagram that the signal from the roll rate gyro and the aileron position potentiometer connect to the airspeed compensator. In the following discussion you will learn the purpose of these various components and their operation.

PURPOSE OF THE TURN COORDINATOR.

The components of the turn coordinator work together to provide automatic turn coordination of the rudder with respect to aileron position. To better understand what control surface movements are necessary for a coordinated turn, let us follow the pilot's action in a normal turn.

When the pilot desires to turn the aircraft to the right, he moves the stick to the right. The stick movement causes the right aileron to move up and the left aileron to move down; therefore, the right wing drops and the left wing rises. At the same time that he moves the stick, he also depresses the right rudder pedal, moving the rudder to the right. Thus, the airplane goes into a turn in a bank or tilted attitude. The banked attitude prevents the aircraft from skidding around the turn in much the same manner that a banked curve on a highway prevents a speeding automobile from skidding off the road.

Looking at the turn problem from another aspect, let us assume there is not a turn coordination feature when the aircraft rolls. You know the ailerons alone produce roll, and in most aircraft, a roll results in a yaw. There are two types of yaw—proverse and adverse. In the F-102A, both types of yaw are encountered; each type being peculiar to the speed range in which the aircraft is flying.

Below the speed of sound, *adverse yaw* is present in a roll attitude. For example, when the aircraft rolls to the right without the necessary rudder correction, the plane yaws to the left and skids through the air. The skidding as a result of the yaw is referred to as side slip.

It is beyond the scope of this supplement to explain all the conditions causing this yaw condition, but designers attribute much of it to the inertia coupling existing between the roll and yaw axes. The coupling is dependent to such a degree on each axis that a rolling acceleration causes a yawing motion that is similar to the precession of a gyro. Other factors are the aerodynamic coupling between the axes of the aircraft.

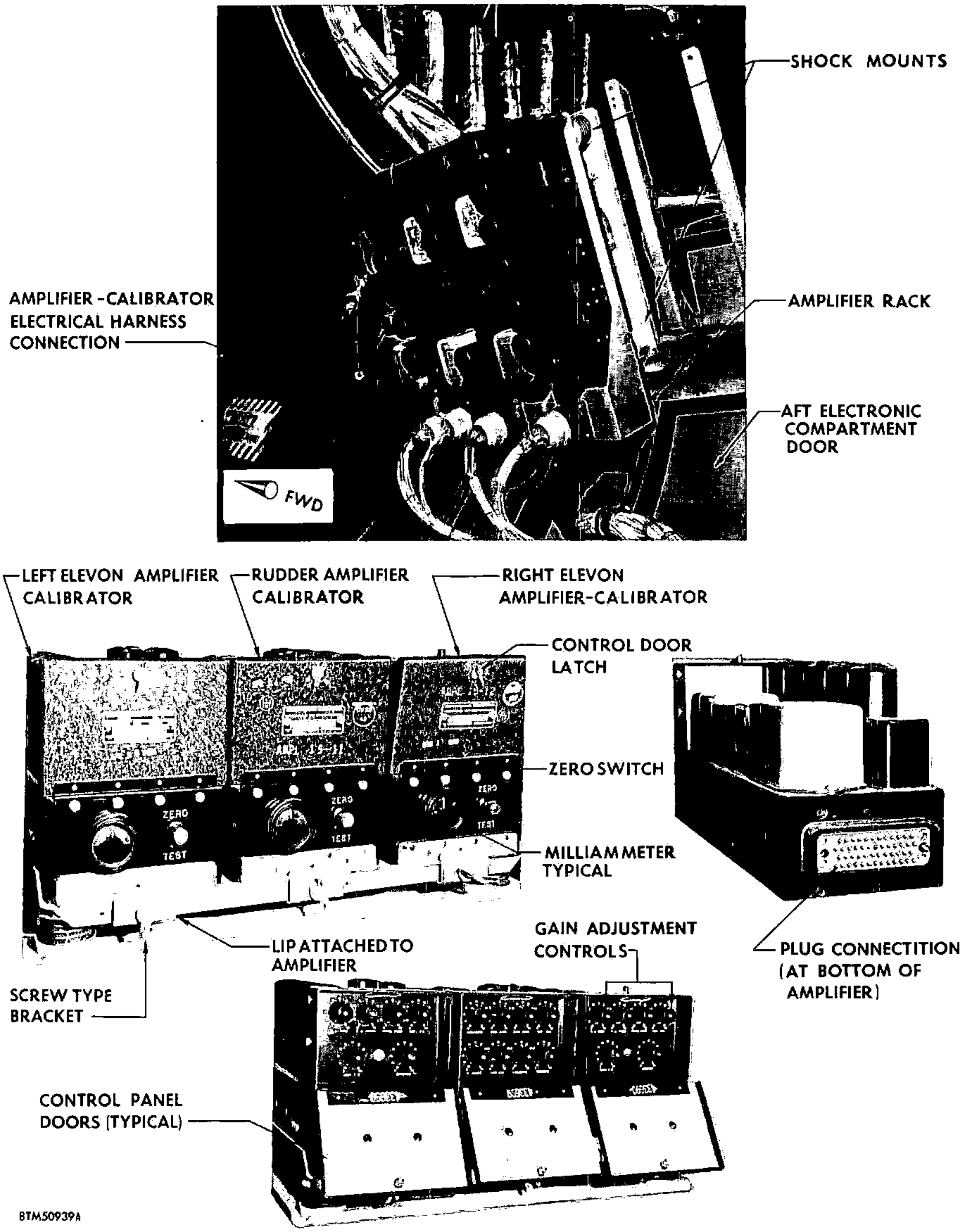

Figure 4-24. The Rudder-Power Amplifier-Calibrator

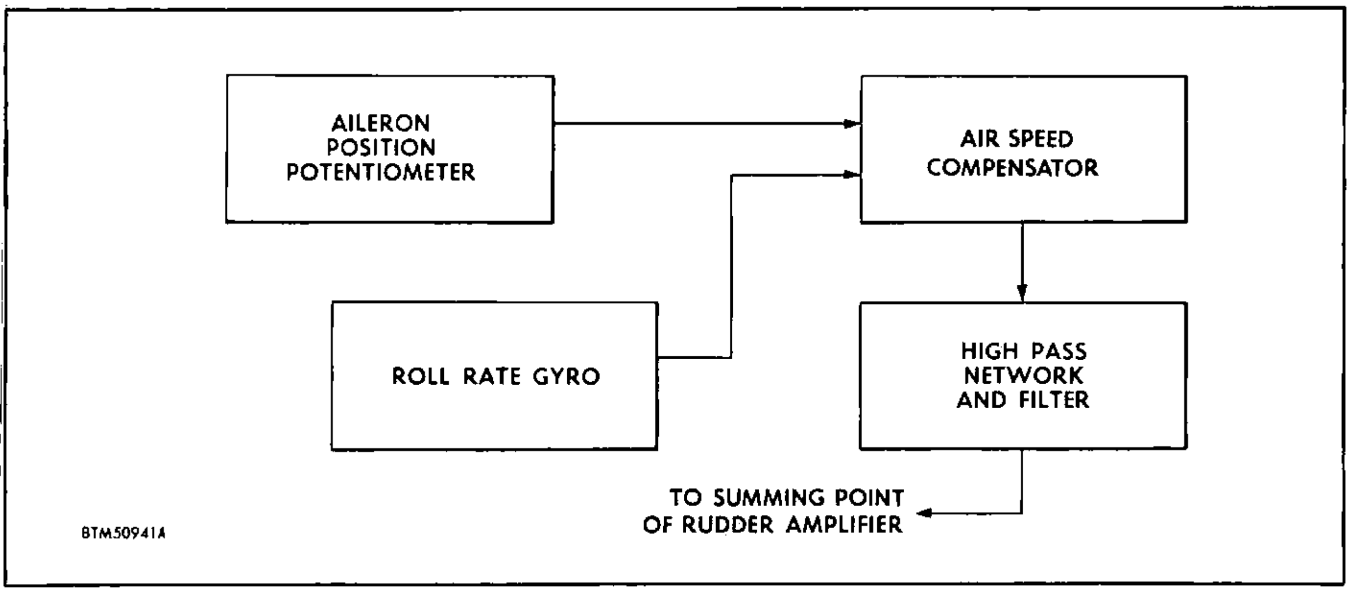

Figure 4-25. Turn Coordinator Block Diagram

Generally, this yawing, or side slip condition, can be explained by saying that a rolling velocity causes airflow conditions that in turn induce a yawing motion. You know that aileron movement causes this rolling motion, and you know that the rudder controls the yaw. Therefore, you can assume that signals from the aileron position potentiometer and roll rate signals from the roll rate gyro would be used to overcome this problem and produce coordinated turning.

It is known that less rudder correction is necessary for yaw correction as the speed of the aircraft increases —up to a certain point. A study of the flight characteristics of the F-102A reveals that there is a speed range where no rudder correction is required to make a coordinated turn. When the aircraft passes through this critical speed range, the condition known as *proverse* yaw is encountered.

COMPONENTS OF THE TURN COORDINATOR.

In the following section, you will learn of the turn coordinator components and their operation. Keep in

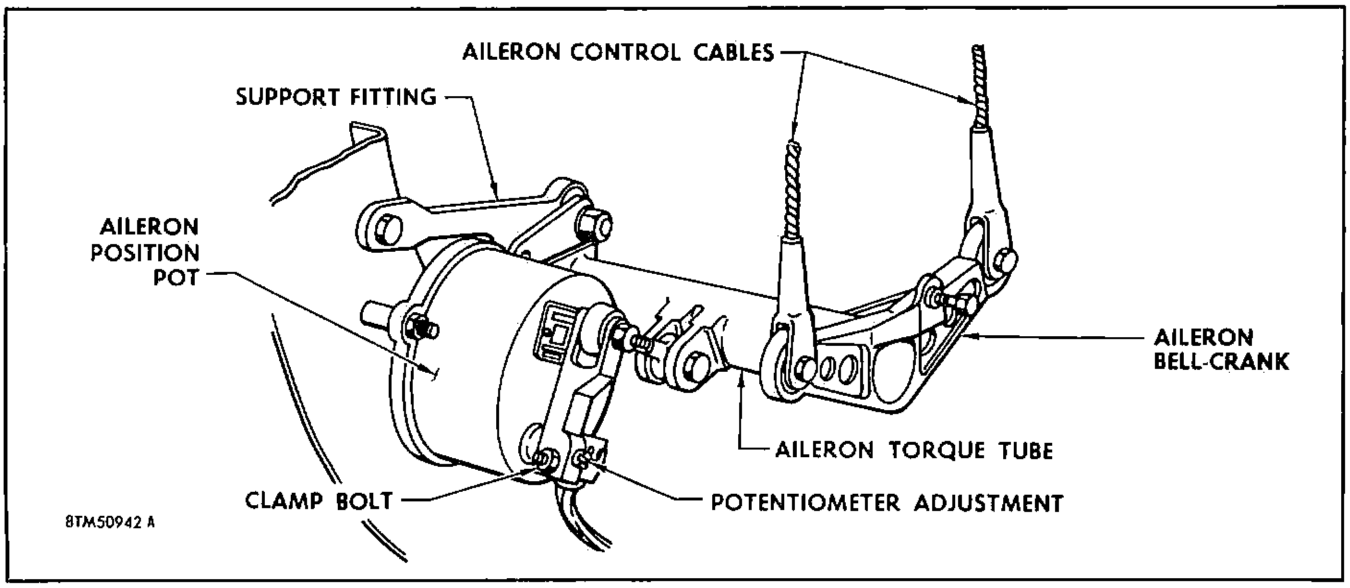

Figure 4-26. The Aileron Position Potentiometer

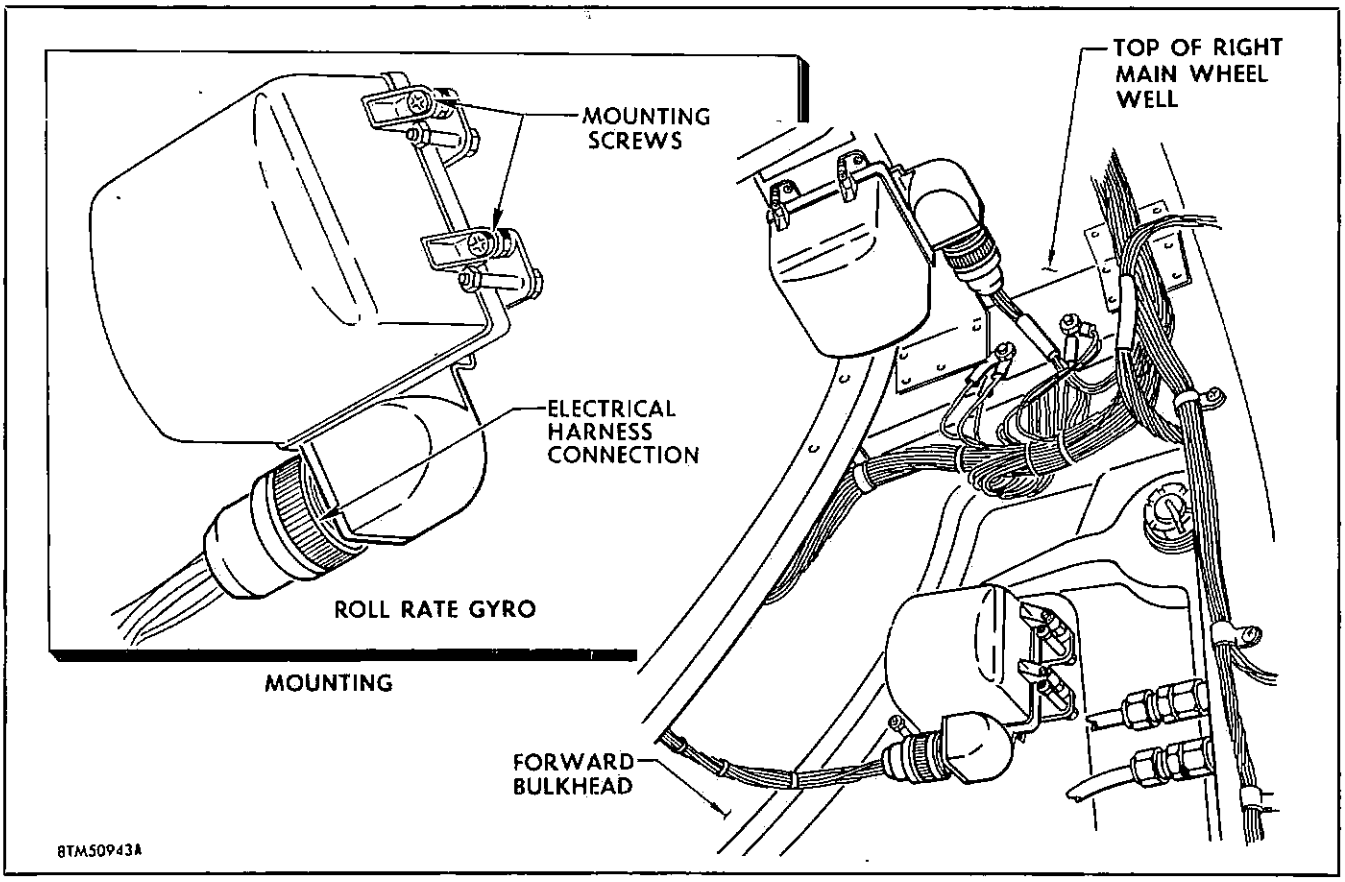

Figure 4-27. The Roll Rate Gyro

mind as you study these components, how they perform a coordinated turn by coordinating the rudder and aileron movements.

The Aileron Position Potentiometer.

The aileron position potentiometer is located in the aileron bell crank at the mixer (see figure 4-26). As you can see, the movement of the aileron cables positions the wiper arm of the potentiometer. The resulting signal is proportional to the amount of rudder correction necessary for a coordinated turn. Coupled with this signal from the aileron position potentiometer is the roll rate signal from the roll rate gyro.

The Roll Rate Gyro.

Note the roll rate gyro in figure 4-27; this unit, located in the main wheel well, supplies the yaw damper system with a signal that is proportional to the rate of roll of the aircraft.

When the aircraft first starts to turn, the yaw rate gyro signal tends to oppose the turn; but the combination of the roll rate gyro and the aileron position signals are much larger than the yaw rate gyro signal, and they compensate for the opposing signal. Therefore, you can see why it is necessary to introduce roll rate signals to the yaw damper system.

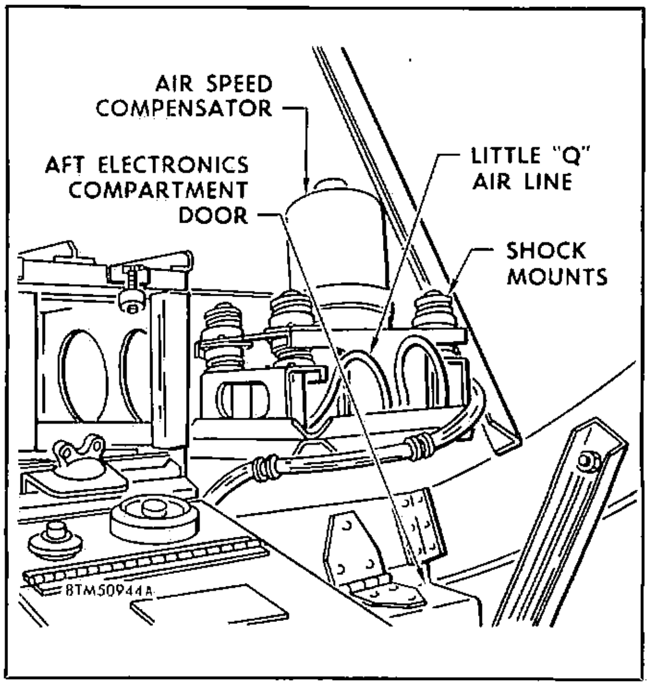

Figure 4-28. The Airspeed Compensator

The Airspeed Compensator.

Figure 4-28 shows the airspeed compensator; this unit is located in the top shelf of the electrical equipment rack. Note the hose connection to this unit. This is the ram air input hose from the little "Q" probe on the leading edge of the vertical stabilizer (fin). The air entering the tube is proportional to airspeed and operates a bellows device within the airspeed compensator. The bellows moves the wiper arms in the compensator to develop four signals: —aileron; +aileron; lagged roll rate; and roll rate. The direction of the arrows in figure 4-29 shows the movement of the wiper arms with increasing airspeed. Thus, you can see that the roll rate and aileron position signals are regulated in accordance with airspeed through the operation of the bellows device.

THE CIRCUITS OF THE TURN COORDINATOR.

Figure 4-29 shows the interconnection of the units in the turn coordinator. The first circuit you will study is the distribution of the roll rate signal from the rate gyro. Note that the roll rate signal enters the airspeed compensator and connects to two circuits. In the airspeed compensator, note the circuit from the potentiometer at the right. Following this circuit to the amplifier, you can see that it forms the lagged roll rate circuit that was mentioned in the introduction to the turn coordinator. As you will recall, the purpose of this circuit is to compensate for the residual roll caused by the rolling momentum of the aircraft in a turn.

The residual roll is, of course, dependent on the airspeed. Since the amount of rudder required to coordinate a turn is less at high speed than at low speed, the airspeed compensator reduces the strength of the signal from the roll rate gyro as the airspeed increases. Note the direction of the arrows in the schematic; they point in the direction the "Q" potentiometer wiper arms will move with increased airspeed. (The letter "Q" is used to denote the airspeed impact pressure from the "Q" probe on the rudder.) The inlet for the "Q" line is on the leading edge of the vertical stabilizer. The faster the aircraft flies, the greater the air pressure is on the bellows. This causes the bellows to expand and the wiper arm to move down the resistor. You can see that this movement adds more resistance to the circuit and attenuates the roll rate signal.

There is a lagged roll rate gain adjustment knob on the front of the calibrator. The knob permits you to adjust the gain of the lagged roll rate signal during calibration and alignment checks of the damper system.

Now let us see what happens in the other circuit of the airspeed compensator that receives the roll rate signal. Note that once again there is a "Q" potentiometer operated by the bellows. The same thing happens in this potentiometer as in the other circuit. When the airspeed increases, the roll rate signal diminishes and there is less roll rate signal sent to the left grid of tube *V201*. Naturally, less roll rate results in less signal to the rudder amplifier.

Directly below the roll rate gyro on the schematic is the aileron position potentiometer that sends a signal to two circuits in the airspeed compensator. Let us first discuss the minus (—) aileron signal.

The purpose of the minus aileron signal is to coordinate the turns in the speed range requiring an opposite rudder correction (proverse yaw speed range). As you can see, the bellows positioned wiper arm in this circuit picks off less resistance as airspeed increases. This permits a larger aileron position signal to enter the circuit and thus, cause a greater rudder correction. The most important thing to notice about this circuit is that it connects to the right grid of tube *V201*. As you will recall, the roll rate gyro signal that passed through the airspeed compensator circuits were connected to the left grid of this tube. Therefore, you can see that an opposite rudder correction results from the minus aileron signal.

Now look at the plus (+) aileron signal circuit that is directly below the minus aileron signal circuit on the schematic. The purpose of this circuit is to introduce a rudder correction in the speed range where adverse yaw exists. Note that the wiper arm in this circuit operates similar to those in the roll rate circuits. Up to a certain point there is less and less signal permitted to pass to the left grid of tube *V201*.

The net effect of these two circuits is that as airspeed increases, the signal strength of the plus aileron circuit decreases and the minus aileron signal increases. At some point in the speed range of the airplane, the minus aileron signal will equal the total of the plus aileron signal and roll rate signal. At this point, the summing stage tube *V201* is balanced and the rudder is zero. As airspeed increases above this point, the minus aileron still increases while the plus aileron decreases, and the opposite rudder to that of the original, is obtained. In this manner, proverse yaw is coordinated at higher speeds. The schematic on figure 4-30 shows how the turn coordinator on figure 4-29 attaches to the basic yaw damper system shown on figure 4-23.

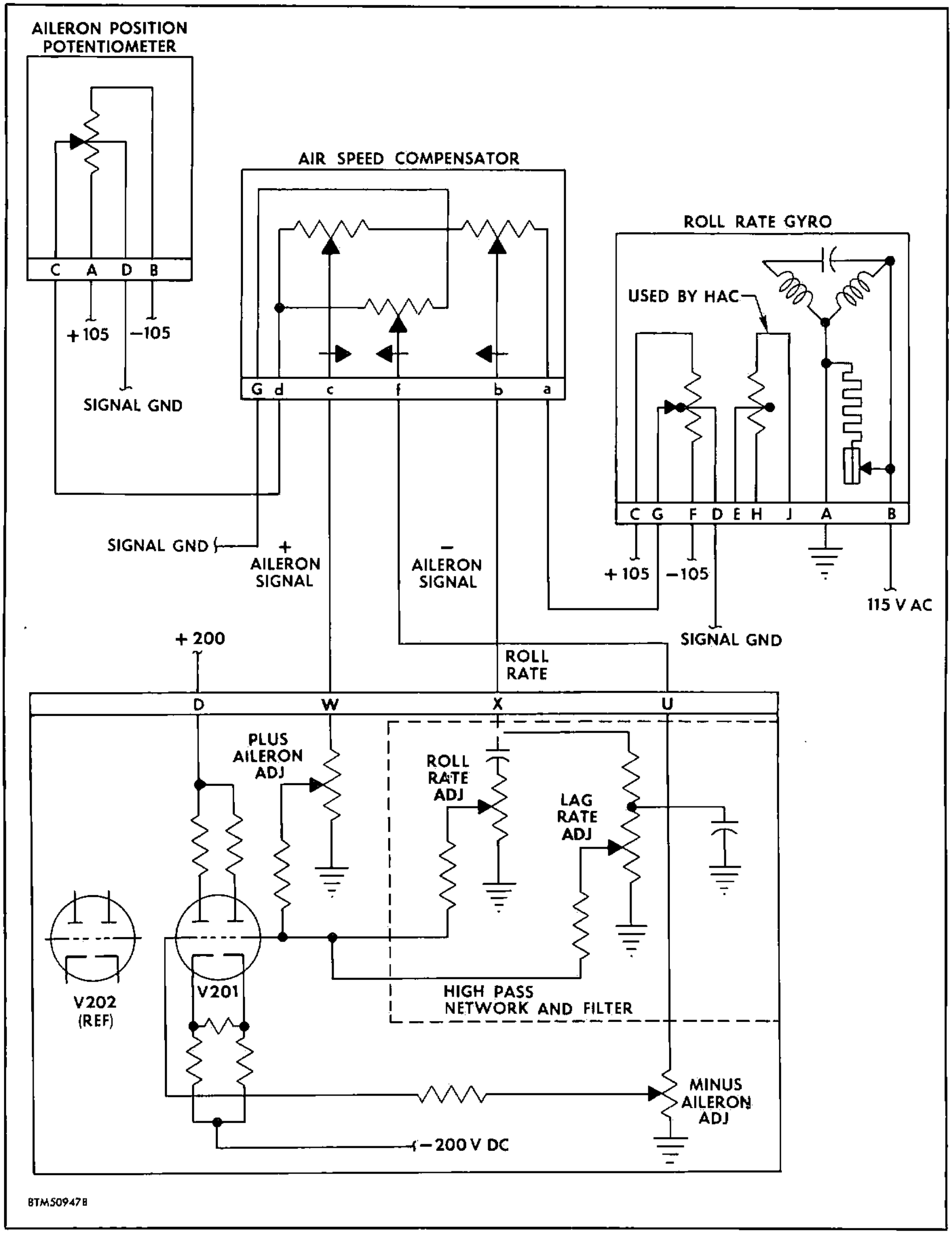

Figure 4-29. Turn Coordinator Schematic

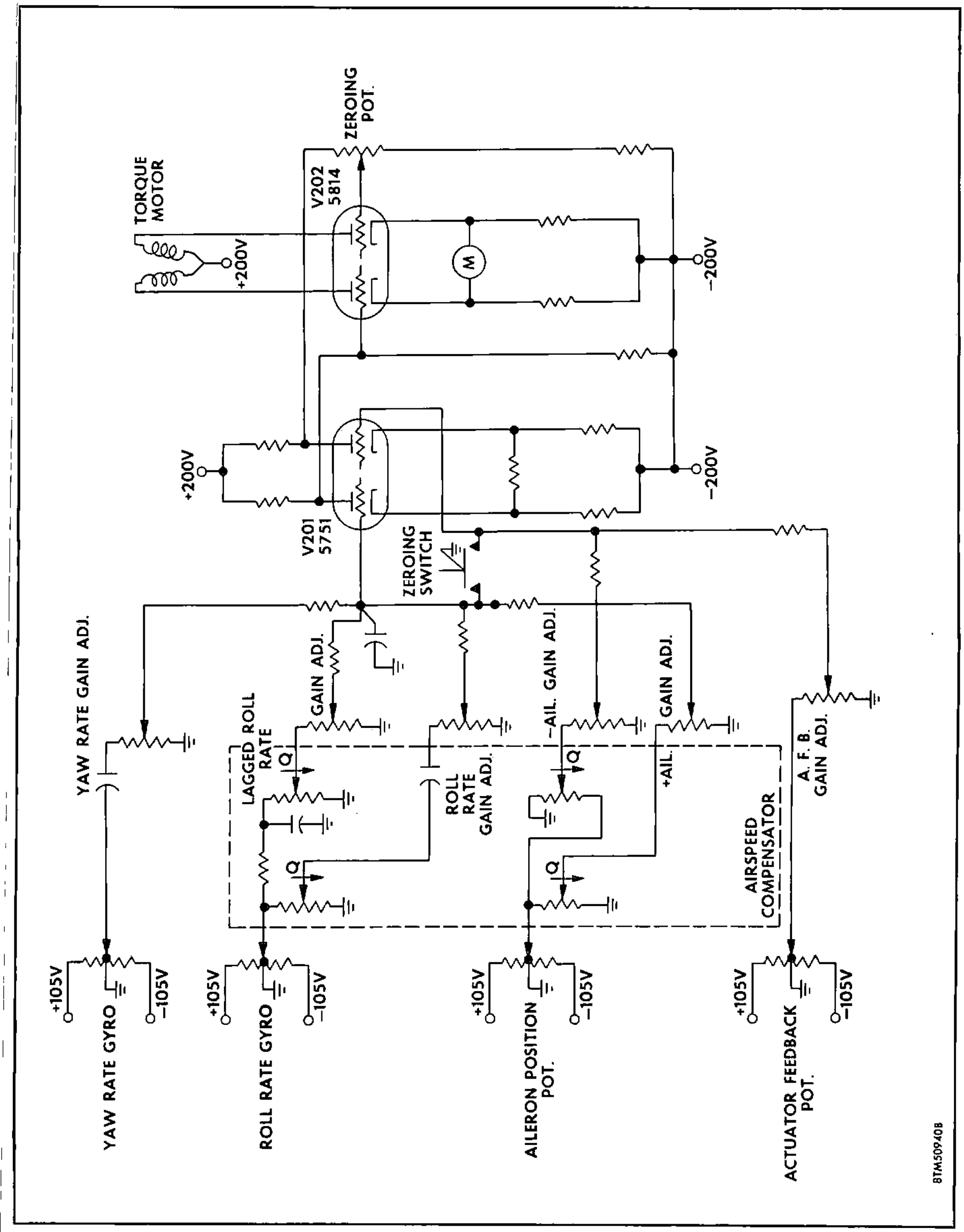

Figure 4-30. The Yaw Damper With Turn Coordinator

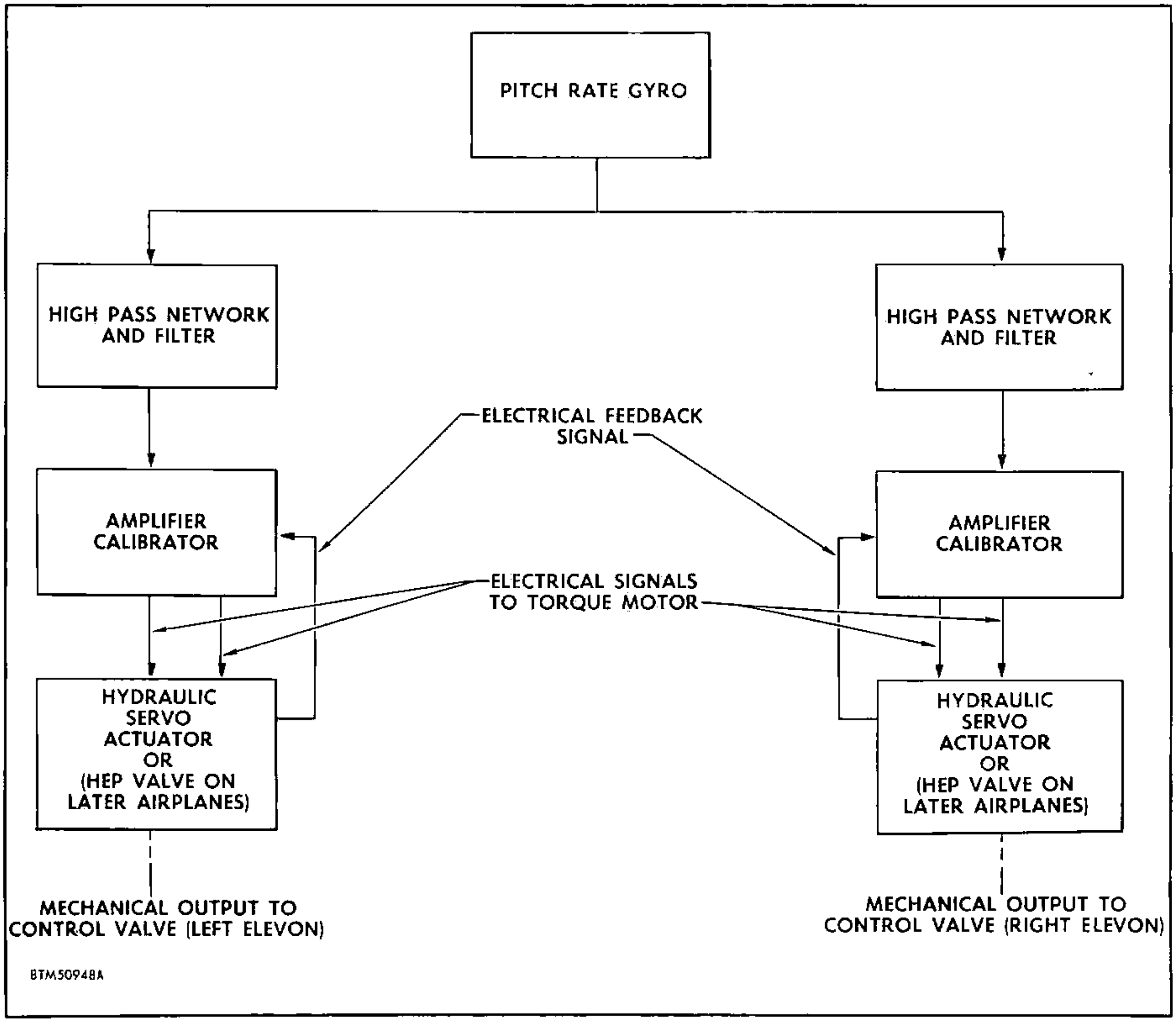

Figure 4-31. The Pitch Damper System

THE PITCH DAMPER SYSTEM.

The pitch damper system provides longitudinal stability through the elevons. Comparing the block diagram of the pitch damper system on figure 4-31 to the yaw damper system on figure 4-20, you can see that the systems are quite similar. Notice that the pitch system uses the same four types of components as the yaw system—a rate gyro, a high pass network and filter, a left and right elevon amplifier-calibrator, and a left and right elevon servo actuator. On later systems the HEP valve replaces the servo actuator. In either the HEP valve or the servo actuator, the signal from the amplifier is applied to the coils of a torque motor. The torque motor positions a flipper that controls the flow of hydraulic fluid to the elevon actuating cylinders.

As you can see, there is a difference in the quantity of some of the components—in the pitch damper system there are two amplifiers and two actuators (or two HEP valves). These additional components are necessary because there are two elevons while there is only one rudder. Since the elevons must work together, we can assume that the deviation signal from the pitch rate gyro must go to each elevon amplifier.

PRINCIPLE COMPONENTS OF THE PITCH DAMPER SYSTEM.

The four types of components in the pitch damper system—rate gyro, high pass network and filter, amplifiers, and actuators—need no extensive explanation as you are already familiar with them. Of course, there are certain minor differences that are required in adapting the basic damper system to pitch correction. In the following portion of this section, you will become acquainted with these differences as they apply to pitch damper functions.

The Pitch Rate Gyro.

From the discussion of gyroscopic principles, you know that the pitch rate gyro is mounted differently than the yaw rate gyro. This is necessary because it must sense deviations along a different axis of the aircraft. You can see the location of this unit in figure 4-32; it is located on the opposite side of the main wheel well from the roll and yaw rate gyros. Otherwise, in external appearance and internal circuitry, the pitch rate gyro is similar to the other rate gyros.

The High Pass Network and Filter.

The high pass network and filter of the pitch damper system performs the same function as in the yaw damper system. That is, it filters out "noise" and signals of undesirable frequency, and it reduces a steady pitch rate signal to zero so that the pitch system does not oppose desired elevon movement, as in a climb or dive.

The Elevon Amplifiers.

There is one particular feature of the elevon amplifiers, shown on figure 4-33, that is not shared by the yaw damper system. This is the coupling of signals from the Automatic Flight Control System (AFCS) to the amplifiers. This feature permits the automatic flight control portion of the fire control system to effect control surface movement through the damper mechanisms. This aspect of pitch damper operation compares with the turn coordination feature of the yaw damper system — in other words, it is an auxiliary function. This will be discussed fully in the operation of the AFCS in the next chapter.

The Torque Motor.

The torque motor in the servo actuator or in the HEP valve receives signals from the amplifier, and the movement of the torque motor flipper ports hydraulic fluid under pressure to a spool within the actuator. The spool receives the effect of this pressure change and moves in a corresponding direction. As the spool moves, it uncovers a port and permits hydraulic pressure to displace the actuating cylinders; this results in control surface movement. This operation is similar to that of the yaw damper system.

THE PITCH DAMPER CIRCUITS.

As you can see in figure 4-34, the signal from the pitch rate gyro connects to both the right and left elevon amplifiers. Note that the amplifiers shown in the schematic are similar. A signal from the rate gyro, acting on the grids of the summing stage tubes, will have the same effect on both control surfaces and the elevons will move together to provide appropriate pitch corrections. The operation of the amplifiers is similar to the rudder amplifier.

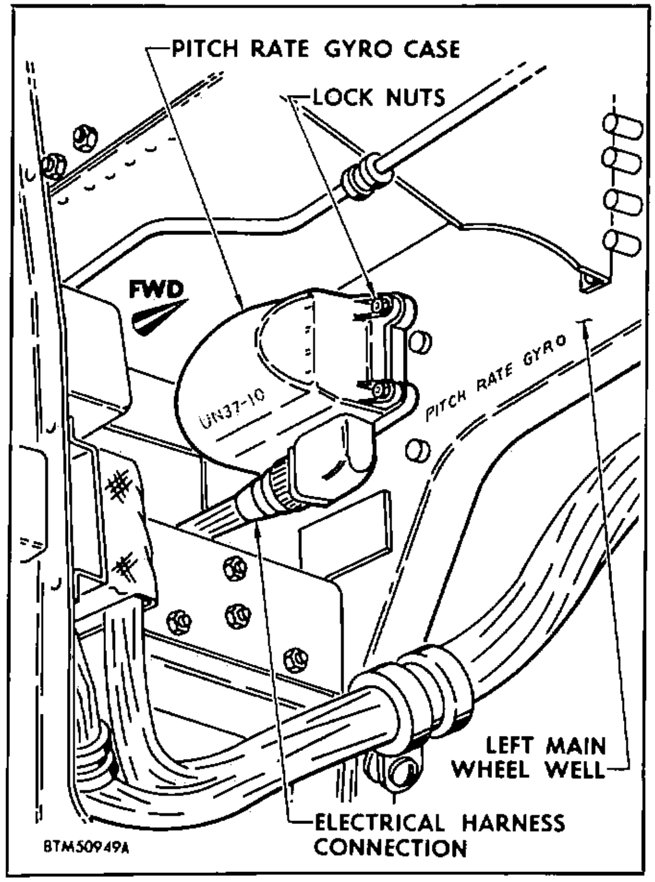

Figure 4-32. The Pitch Rate Gyro

On the control panel of the right and left elevon amplifiers, there is a pitch rate adjustment that compensates for differences in the circuits. The need for such an adjustment is quite obvious, because unequal movement of the right and left control surfaces results in an aileron action. The presence of aileron action in the pitch damper operation is not desirable, although in AFCS mode which operates through these same amplifiers, such an action is required.

As you will recall from the damper engage circuits, there is a provision for disengaging the damper system when a malfunction of the system causes a differential movement of the elevons in damper operation. You now have a basic understanding of the pitch and yaw damper systems and their effect on the flight controls. With this knowledge, you are better prepared to study the AFCS interconnecting circuits to the damper system. These interconnecting circuits, often referred to as the Full Authority (FA) subsystem, are dependent on the damper systems for accomplishing their purpose. In the next chapter, you will learn the function of the AFCS and Full Authority system.

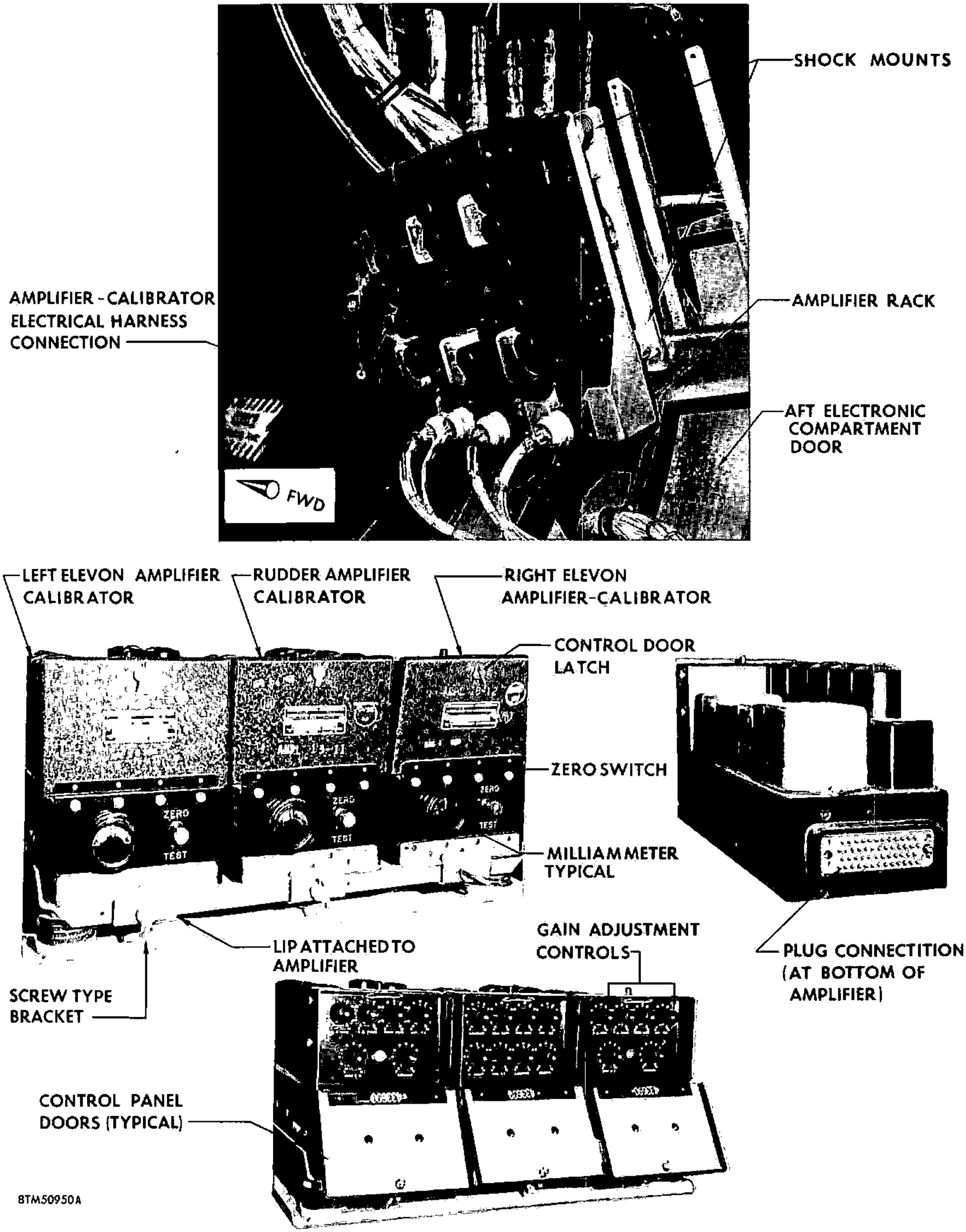

Figure 4-33. The Right and Left Elevon Amplifier-Calibrators

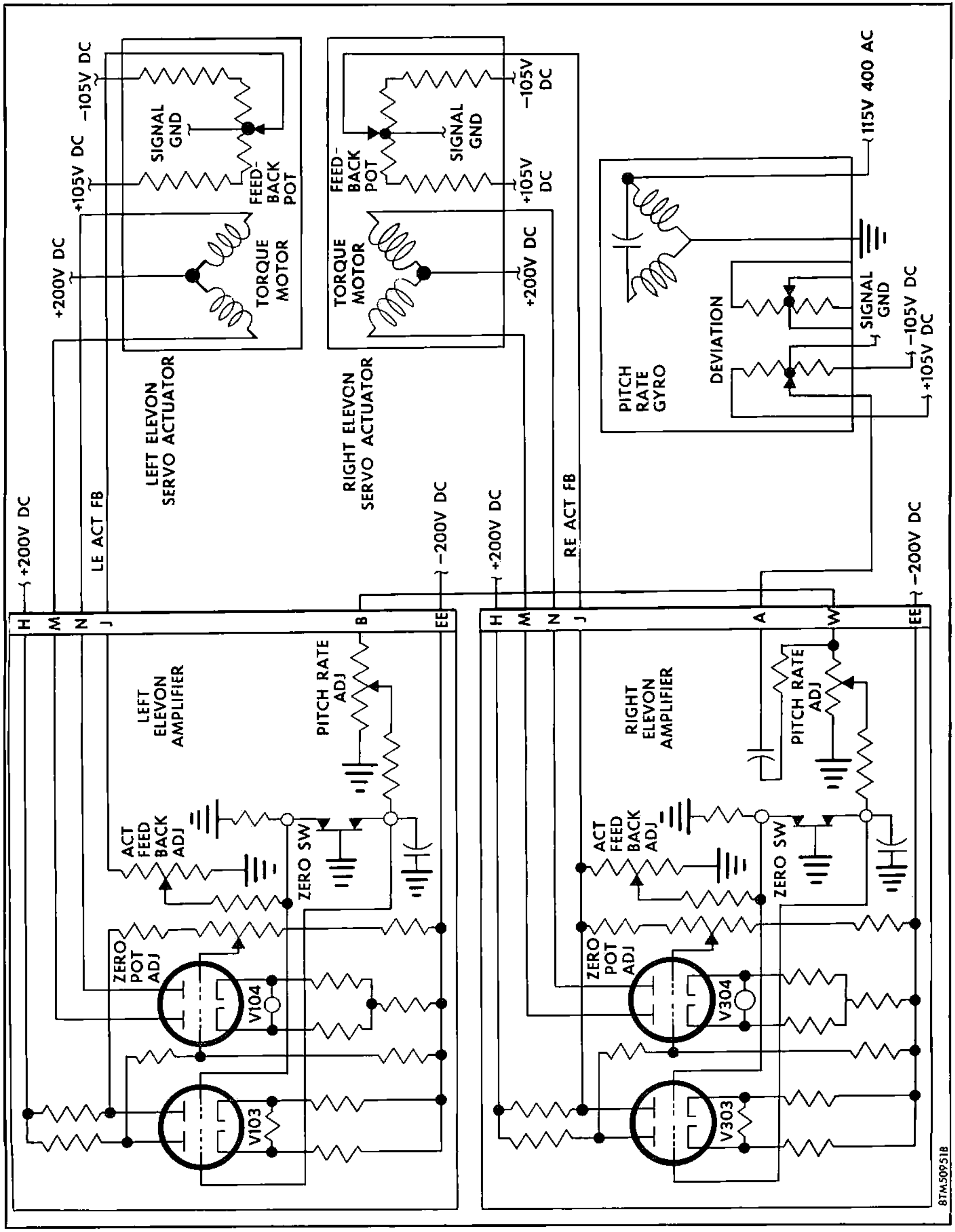

Figure 4-34. Pitch Damper System Schematic

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Chapter V

THE AUTOMATIC FLIGHT CONTROL AND FULL AUTHORITY SUBSYSTEMS.

Contents *Page*

The Automatic Flight Control Subsystem (AFCS) signals connect to the damper system through the Full Authority Subsystem. Since you are already familiar with the damper system operation (described in the previous chapter), there will only be a few new components or principles introduced in this chapter that you have not already studied about. First, let us see what the purpose of the Automatic Flight Control and Full Authority Subsystems is, and also learn a little about the evolution of the flight control system.

PURPOSE OF THE AUTOMATIC FLIGHT CONTROL AND FULL AUTHORITY SUBSYSTEMS.

The Automatic Flight Control Subsystem (AFCS) is part of the MG-10 Aircraft and Weapon Control System. In general, the purpose of the AFCS is to provide automatic direction and attitude signals to the flight controls. The signals originate in components similar to those of an autopilot. The AFCS sends the signals through the Full Authority Subsystem to the damper amplifiers. Thus, the Full Authority Subsystem acts as an interconnection network between the AFCS and the damper system. The AFCS full authority signals have a much wider range than the damper system signals. As a result of the larger signals, the control surfaces are permitted to move through their full range of travel while operating in the AFCS mode.

As you will recall, the turn coordinator dictates the position of the rudder from the position of the ailerons. Therefore, it is only necessary for the AFCS to supply signals to the elevon amplifiers in order to have complete control of the aircraft's flight controls. The only signals we are concerned with in this chapter, then, are the signals the AFCS sends to the damper system elevon amplifiers. These signals are the AFCS *full authority roll, full authority pitch,* and the *control surface feedback signals* originating at the right and left elevon control surface actuators.

EVOLUTION OF THE FLIGHT CONTROL SYSTEM.

To avoid confusion with the discussions in this and other supplements of this training series, you should

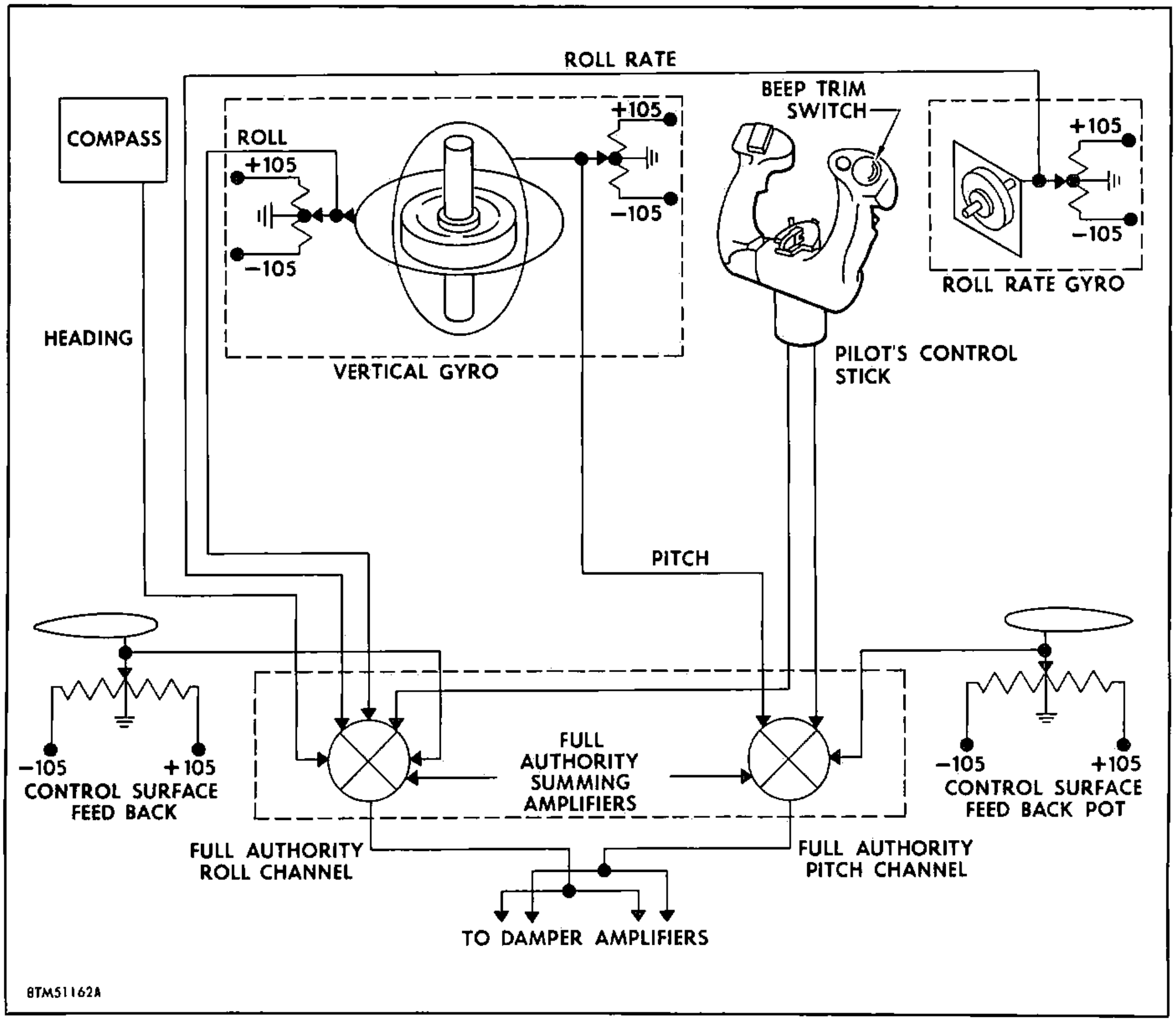


Figure 5-1. A Typical Automatic Flight Control System

be familiar with the evolution of the F-102A flight control system. In some of the supplements you will find reference to the Pilot Assist System. This system was an interim system installed on some of the first production series F-102A aircraft. It has since been deactivated and will be removed from these early aircraft. The Pilot Assist System was designed to operate only with the servo actuator type control valve.

Basically, the operation of the Pilot Assist System and the AFCS are similar. A brief description of the pilot assist components and their operation will acquaint you with the theory of AFCS operation and help you in understanding how the AFCS controls the flight attitude of the F-102A. Detailed description of the AFCS, its actual operation and maintenance procedures, are classified and can be found in the MG-10 System Maintenance Manual.

THE OPERATION OF A TYPICAL AUTOMATIC FLIGHT CONTROL SYSTEM.

Let us assume the Pilot Assist System is a typical Automatic Flight Control System. We can then use the block diagram in figure 5-1 to explain the basic principles of AFCS in the F-102A. Note that the main units of the system are the compass, the vertical gyro, the roll rate gyro, the control surface feedback potentiometers, the amplifiers, and a trim control switch. Let us see what each of these components accomplishes and how it affects the system.

THE PURPOSE OF THE COMPONENTS.

The compass (or heading unit) generates directional reference signals. The AFCS must have directional signals in order to correct for deviations of the aircraft from the desired heading and thus maintain a fixed heading. The vertical gyro is a displacement gyro (which you learned about in the discussion of gyros in the damper systems) and supplies the system with pitch and roll displacement signals. Note in figure 5-1 that the spin axis is perpendicular to the earth's surface and that the gyro is free to move in both the pitch and roll axis. The pitch signal is fed to the pitch channel of the full authority summing amplifier. The signal is applied to the damper amplifiers as the full authority pitch signal.

Note that the vertical gyro also originates a roll signal. This signal is combined with a signal from the roll rate gyro to form the full authority roll channel. The signal from the full authority roll channel is fed to the right and left elevon amplifiers as the full authority roll signal. The right and left elevon surface feedback potentiometers provide feedback which is summed with the inputs to the pitch and roll channel. The signal from the potentiometers and the manner in which it controls the operation of the AFCS will be discussed in a separate section of this chapter.

The beep trim switch on the pilot control stick connects to the full authority pitch and roll channels of the pilot assist summing amplifiers. When the pilot desires to make a change in the heading or attitude of the aircraft while the AFCS is engaged, he momentarily actuates the trim control on the stick until the desired change is made.

You can see in figure 5-1 that signals from the beep trim switch are sent to the full authority summing amplifiers as full authority pitch and roll signals. Thus, beep trim operates electronically through the damper amplifiers to trim the aircraft while the controls are in the AFCS mode.

With this brief introduction to the theory of AFCS operation and the development of signals to the damper system, let us now see what conditions are necessary for the engagement of the AFCS to the damper system.

THE FULL AUTHORITY SYNCHRONIZING ENGAGE AND MONITOR NETWORK.

The right and left elevon amplifiers each contain synchronizing motors that drive potentiometers. The potentiometers control the AFCS engage circuitry and the feedback to the full authority engage monitors. The synchronizing motors position the potentiometer wiper arms so that signals from the surface feedback potentiometers and the full authority pitch and roll channels do not cause violent engage transients when the flight control system is placed in AFCS mode of operation.

It is important to remember that certain functions in the pitch and yaw damper system as well as AFCS are dependent on the operation of the radar. Among these functions is the supplying of voltages to the synchronizing circuits of the damper system.

As you can see in figure 5-2, the 55-volt a-c power to the monitor tubes is dependent on the energizing of AFCS interlock relay No. 2. Note that this relay is energized only when the radar portion of the MG-10 system is operating. Also note that the MG-10 system furnishes the ±105-volts direct current to the motor-driven potentiometer as well as the aileron and elevon position potentiometers.

Thus, the synchronizing circuits can not function during Direct Manual mode because the damper amplifiers are not properly energized. The circuits also can not function during AFCS mode because relays *K101* and *K301*, the elevon amplifier engage relays, are energized. This fact is mentioned to remind you of the purpose of the synchronizing circuits — to null out surface feedback signals and full authority pitch and roll signals prior to AFCS engagement.

During damper operation and prior to engagement of the AFCS, inputs to the full authority summing amplifier must be nulled out. The reason for nulling the signals is to prevent sudden, hard-over signals reaching the amplifiers during AFCS engagement.

The surface feedback potentiometer will not necessarily be in the *no* signal position when the AFCS is engaged. If such is the case, a feedback signal will be applied to the left grid of tube *V102* and the aircraft will be thrown into a violent maneuver. To prevent this undesirable condition, note that the full authority summing signal is taken from the plate of tube *V102* and sent from the "873" box to the right grid of tube *V101*. As you can see, the plate circuits of tube *V101* are connected to the coil of the differential current relay. The left grid of tube *V101* is at ground potential when tube *K101* (or the corresponding tube and relay in the right elevon amplifier) is deenergized. Since the left grid has no bias voltage applied, there is a differential current flow in the coil of relay *K102*. This causes the arm of the differential current relay to be deflected to one of the contact points which connects 55 volts alternating current to one of the windings of the synchronizer motor. Remember that 55 volts alternating current is not available to the system unless the radar is ON.

The motor positions the potentiometer arms. As you can see in the schematic, when the wiper arm is in the center position, no voltage from the ±105-volt, d-c potentiometer is applied to the grid of tube *V102*. The ±105 volts direct current is also not available unless the radar is ON. However, the wiper will not be in the center of the potentiometer unless there are no signals from the full authority pitch, roll, or surface feedback circuits in the full authority summing

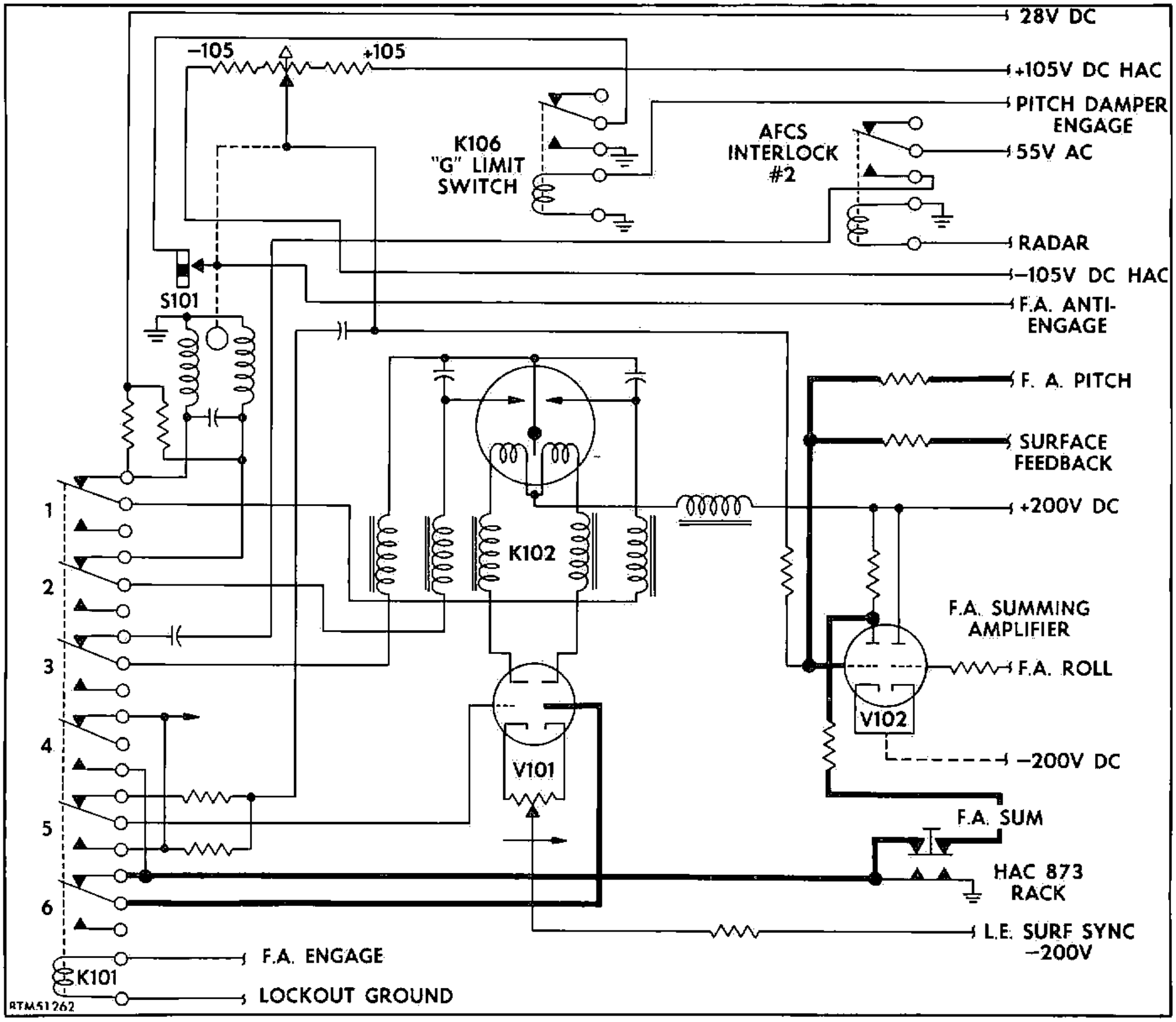


Figure 5-2. Full Authority Synchronizing Circuit

tube. If signals are present, the motor will drive the potentiometer arm in a magnitude and direction that will effectively cancel the full authority signal to the summing tube.

THE PITCH "G" LIMITER SYSTEM.

The pitch "G" limiter system is a monitor of the AFCS system. The purpose of the system is to disengage the AFCS when the F-102A reaches certain predetermined "G" limits. As you can see in figure 5-3, the limiter system consists of a lateral accelerometer, an amplifier, and a rate gyro. The limits of the system are +4.5 and —1.5 "G's" of normal acceleration or +45 and —15 degrees per second of angular acceleration or the equivalent combinations of both. Regardless of the amount or direction of the angular acceleration, the system will disengage when the normal acceleration reaches +5 or —2 "G's."

The linear accelerometer measures forces normal to the pitch axis. Note in figure 5-3 that outputs are taken from potentiometers and sector switches. The potentiometer is center-tapped at +4.9 to +5.0 "G's" and at —1.9 to —2.0 "G's."

The amplifier portion of the system contains dual-triode vacuum tubes and differential current relays. The tubes are biased so that the plate circuit relays are held closed during normal operation. When forces beyond the predetermined "G" limits are detected by the accelerometer or pitch gyro, signals pass through the pitch "G" networks causing the bias on the tubes to be overcome and the relays release. The AFCS engage circuitry is then broken and the flight controls revert to yaw damper control. Should the balanced condition of the limiter circuits be disturbed by the failure of an amplifier component, the "G" monitor

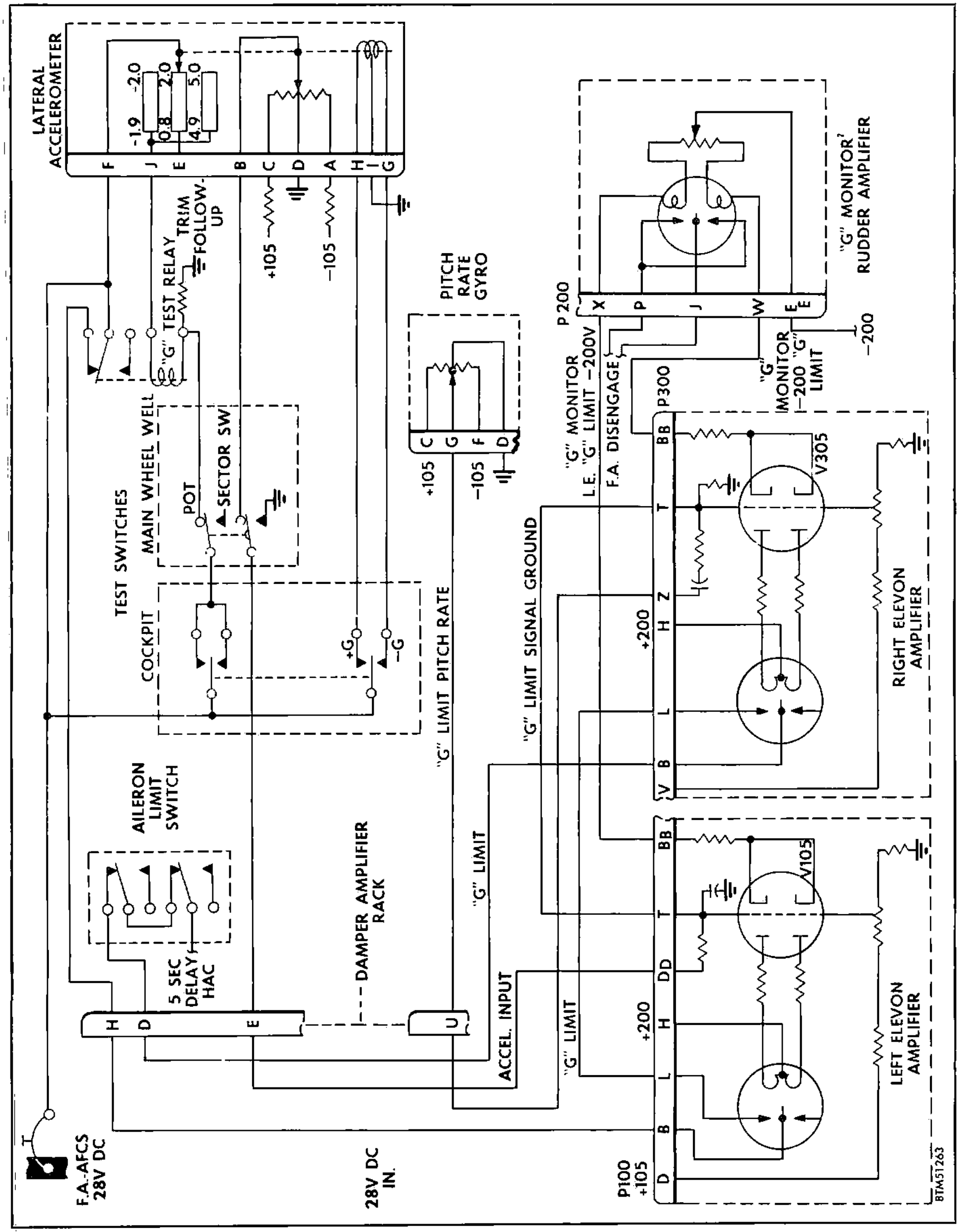


Figure 5-3. The Pitch "G" Limiter

squared relay, which compares the ±"G" circuits, will operate to disengage the pitch "G" limiter and consequently the AFCS and pitch damper system.

The components of this system are located in various places throughout the aircraft. The pitch rate gyro is the same gyro that furnishes pitch rate signals to the damper system; it is located in the left main wheel well and is shown on figure 4-32. The linear accelerometer is located in the aft section of the right main wheel well. The amplifier portion of the system is located in the right and left elevon amplifier-calibrators (the ±"G" monitors) and in the rudder amplifier ("G" monitor squared amplifier). The adjustments for these components consist of a "G" monitor adjusting knob on the rudder amplifier, a +"G" and a —"G" balance on the elevon amplifiers, and various control and test switches which are discussed in the following section.

Testing the Pitch "G" Limiter Circuits.

Two test switches are provided to test the functional operation of the pitch "G" limiter network. Note in figure 5-3 that one of these switches is located in the cockpit and the other in the forward section of the right main wheel well. The cockpit test switch, located on the UTILITY SWITCH PANEL, permits you to check either inflight or ground operation of the accelerometer's mechanical and electrical components from stop to stop, component interconnecting wiring, amplifier signal summing and switching circuitry, and most important, the limiter calibration.

Moving the cockpit test switch to either "+" or "—" energizes internal test solenoids which displace the accelerometer mass throughout its range of travel. Movement of this mass produces signals that exceed the desired "G" limits. Should the established tripping levels be greater than +5 "G's" or —2 "G's," the systems will remain engaged, indicating the miscalculation of the pitch limiter system.

The switch in the main wheel well permits you to check the circuitry that operates the +5 "G" and —2 "G" selector switches. The check consists of holding the main wheel well switch ON while actuating the cockpit switch. The system should disengage at each switch position.

There are no specific provisions for a test of the gyro circuits. You can detect malfunctions, however, by the oscillations in the full authority mode or disengagement of the systems (prior to aircraft response) due to hard-over gyro signals.

Calibration of the Pitch "G" Circuits.

Calibration of the pitch "G" limiter circuits consists of balancing the gain adjustments of the ±"G" monitor circuits so that engagement takes place within the specified limits. To calibrate the pitch "G" limiter circuits, apply 28 volts direct current between the amplifier and signal ground; measure the applied voltage in the circuit, and adjust the +G BAL knob so that the full authority and pitch damper systems barely disengage. You should observe the point of disengagement while you are adjusting the balancing control, then cycle the applied voltage about the calibration point and note that the systems always disengage at the desired level.

Next apply the calibration voltage as a step input between the amplifier pitch rate input and the signal ground, then note that the systems disengage. With the voltage maintained at the amplifier input, the "G" limiter should reset within 5 seconds to allow re-engagement. The above procedure is repeated to set the —G BAL knob but applying —19.7 volts instead of 28 volts.

THE AFCS ENGAGE CIRCUITRY.

The AFCS engage circuitry is the last in the sequence of flight control engage circuits. Before the AFCS will engage, the yaw and pitch damper systems must be engaged in the order mentioned. There are three main circuits in the AFCS engage circuitry. Note in the schematic on figure 5-4 that one circuit breaker goes to the "G" test relay. Normally, this test relay will be deenergized except when tests are being performed on the "G" circuits of the AFCS. As you already know, the +"G" and —"G" sector switches will be closed unless the pitch "G" limiter has sent signals that cause the ±"G" monitors to open the circuit.

The next portion of the circuit passes through the aileron limit switches. The switches will normally be closed. They open only when there is a 3° differential between the elevons. Passing through the aileron limit switches, you can see that the next break in the circuit is the 5-second time delay relay in the "239" box located in the "873" rack. This relay is energized independently of the damper and AFCS engage circuitry. Note that as soon as the circuit passes through this relay, two branches are formed. One branch goes to a switch in the EDM relay. The EDM relay, as you will recall from the yaw damper engage circuitry, is energized when the yaw damper is engaged. Thus, the circuit is completed to the solenoid of the AFCS ENGAGE switch located in the "105" rack. You must remember that we have not yet actuated the AFCS ENGAGE switch. The circuits discussed so far have merely been from the power source to the AFCS switch.

Following the circuit through the solenoid, you can see that the circuit passes through the surface feedback synchronizing potentiometers and the "G" limit switch to ground. In case the "G" monitor relay becomes

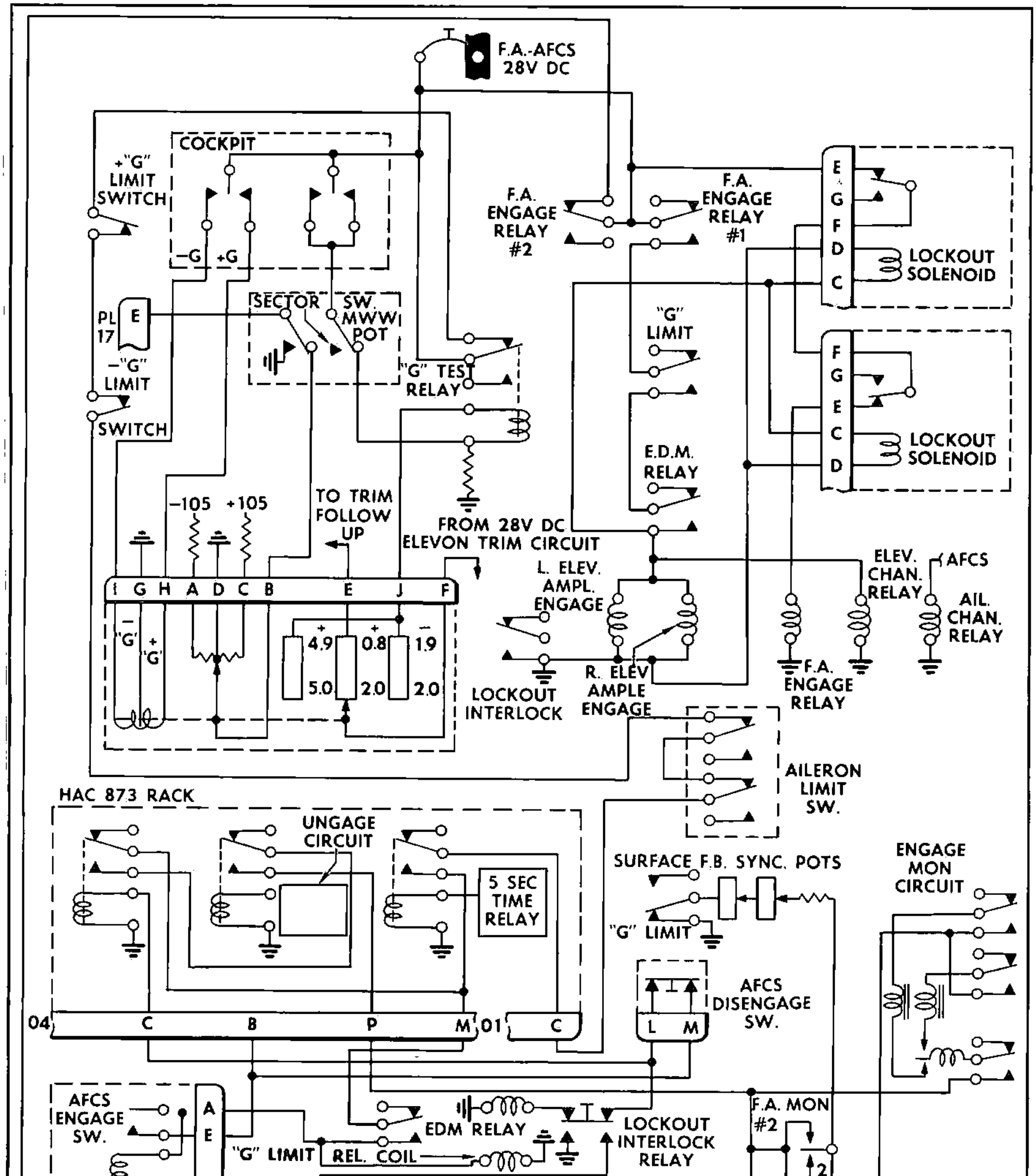


Figure 5-4. The AFCS Engage Circuitry

deenergized, the ground for the AFCS will be interrupted and the system will disengage. When the AFCS ENGAGE switch is closed, note that power is fed to the AFCS DISENGAGE switch. Unless the AFCS DISENGAGE switch is manually actuated (thus breaking the circuit), this 28-volt, d-c power is furnished to the solenoid of the AFCS ON relay. When the AFCS ON relay is energized, note that power from the 5-second time delay relay is routed through the relay to a contact of the 1-second uncage relay. When the uncage relay energizes, note that power goes to the solenoid of the full authority engage relay No. 1, the contacts of the partial engage monitor, and the MONITOR 2 circuits.

When you place the AFCS ENGAGE switch in the ON position, note that the interlock relay, *K107*, is energized if the AFCS DISENGAGE switch is not actuated. Later you will see that the lockout solenoids in the HEP valve cannot be energized until the lockout interlock becomes energized. Note also that the "G" limit relay solenoid, *K105*, is energized at the same time that the solenoid of the AFCS ENGAGE switch is energized.

The second 28-volt, d-c circuit from the circuit breaker is the one that energizes the HEP valve lockout solenoids, the left and right amplifier engage relays *(K101* and *K301)*, and the elevator channel relay. The 28-volt power from the circuit breaker is fed to these components through closed contacts of the full authority engage relay No. 1, "G" limit relay *K106*, and the EDM relay. The 28-volt circuit to the elevator channel relay is completed directly to ground; in the other components it is completed to ground through a closed contact of the lockout interlock relay *K107*. The contacts of the full authority engage relay No. 1 and relay *K107* were closed when the AFCS ENGAGE switch was closed; the "G" limit relay *K106* contacts were closed when the pitch damper system was engaged; and the EDM relay contacts were closed when the yaw damper engaged.

As you have learned, the HEP valve unlock solenoids are energized to allow full authority control of the elevons. When the lockout solenoids are energized, lockout piston rods on the HEP valve are extended and the arms of the lockout switches shown in the schematic fall into detents. Do not become confused by thinking that the lockout solenoids actuate the switches directly; they do not. The switches are in the closed position (shown) only when the lockout piston rods are in the fully locked-out position and the switch arms fall in the detents.

Following the switch circuits, you can see then when the switches are in the rod detents, the 28-volt, d-c circuit is completed from the circuit breaker to the full authority engage relay No. 2. The left and right amplifier engage relays, when energized, permit signals from the AFCS to enter the damper amplifiers. The elevator channel relay is energized so that trim signals originating at the ELEVON TRIM switch may be switched from the trim servo system to the AFCS. Thus, signals originating at the control stick trim control become "BEEP" trim signals.

THE PARTIAL ENGAGE MONITOR CIRCUITS.

The partial engage monitor circuits provide an anti-engage or disengage feature in the event either one or both lockout switches in the HEP valve fail to close when the AFCS is engaged. The monitor circuit will also disengage the AFCS after engagement if one of the lockout switches fail. You can see why such an arrangement is necessary, since full authority on one elevon and not on the other would cause loss of control.

Note in the schematic of the AFCS engage circuitry, figure 5-4, that when the AFCS is engaged, the full authority engage No. 1 relay *K203* and the right elevon amplifier engage relay *K301* are energized. The full authority engage No. 2 relay *K202* will not energize unless both lockout switches in the elevon HEP valves are closed. As you will recall from the description of the HEP valve, the lockout switches will close only when the lockout pins are fully extended and the switch actuating arms fall into the detents in the pins.

Let us assume that one of the lockout switches in the HEP valve fails to extend when AFCS is engaged. Power from the circuit breaker will then pass through deenergized full authority engage No. 2 relay to terminal 2 of the energized full authority engage No. 1 relay. Now consider the voltage divider network. As you can see, the network consists of a *2MEG* and a *56K* resistor connected in series from the —200-volt, d-c power source. The result of the network is a —5-volt signal impressed across the 0.33 MFD capacitor. When the 28-volt signal from the full authority engage No. 1 relay is applied to the junction of the *2MEG* and *56K* resistors, the capacitor will start charging to a positive voltage. The initial negative charge will be dissipated in the process of charging positively.

At the point where the charge on the condenser goes positive, the diode will conduct and introduce a different charging rate. The positive voltage on the capacitor is then impressed across the *300K* resistor, which is the grid resistor for the monitor tube. If the voltage becomes large enough, within a specified time limit, the differential current relay in the plate circuit of the tube will trip and disengage the AFCS mode. The reason for having the time delay in the circuit is that the lockout switches require a small amount of time to close after the lockout solenoids are energized. Otherwise, the monitor circuit would cause disengagement of the AFCS before the lockout switches extended, thus making it impossible to complete the circuit to the solenoid of full authority engage relay No. 2.

HOW THE FULL AUTHORITY SIGNALS AFFECT THE PITCH DAMPER AMPLIFIERS.

Figure 5-5 shows how the full authority pitch and roll signals affect the right and left elevon damper amplifiers. As you have already learned, the full authority signals connect to the summing stages of both elevon damper amplifiers. Note that the full authority pitch signal connects to corresponding grids of the full authority summing stage tubes, while the roll signal (for aileron movement) connects to the opposite grids. Since the yaw damper system depends on the position of the aileron position potentiometer for rudder movement, you can see that it is not necessary to have the full authority signals controlling the rudder. You will recall that the signals from the compass were connected to the full authority roll channel, thus the roll signal causes the aileron movement which in turn causes the rudder to move for a coordinated turn.

When the full authority roll signal enters the left elevon amplifier, note that it is applied to the right grid of the summing stage. Let us suppose the signal is positive. Since a positive signal on the grid accelerates the electron flow to the right plate, the output of the right plate circuit is higher than the left. As you will recall from damper system operation, the left plate circuit will now have even less output than before the signal was applied to the right grid. Note that the output for the summing tube is taken only from the left plate circuit. Under the circumstances, the output from the left plate circuit is more positive than the right plate circuit. Following the left plate circuit output, note that it is routed through the MG-10 system and then through the left elevon engage relay, *K101*. Relay *K101* is energized when the AFCS is engaged; thus, the signal connects to the damper summing stage. From this point, the operation of the circuits is the same as for damper operation.

In the right elevon amplifier, the roll signal is applied to the left grid of the summing stage. You can see that the opposite effect of that in the right damper amplifier will be present in the left stage. Consequently, a negative signal will be applied to the damper summing stage. You will recall that a positive signal was applied to the damper summing stage in the right elevon amplifier. In this manner, a differential movement of the elevons will occur and the airplane will assume a roll attitude.

The full authority pitch signals are applied to the same grids in the full authority summing stages. Thus each damper amplifier will receive the same signal and the aircraft will climb or dive since the control surfaces will move together.

"BEEP" TRIM CIRCUITS.

During AFCS mode, the pilot trims the control surfaces from the ELEVON TRIM switch on the control stick. The trim switch on the control stick controls trim in Direct Manual, Manual, and AFCS operation. The important difference is that AFCS trim is *electronic trim* operating through the elevon amplifiers, while Manual or Direct Manual trim is accomplished through the trim servo network.

As you can see in the schematic, figure 5-6, when the AFCS system is engaged, the elevon and aileron channel relays are energized. Since the radar must be turned on for the AFCS to engage, the aileron channel will be energized and the elevon channel relay is energized as part of the AFCS engage circuit. Therefore, both of these relays will normally be energized when the flight controls are in the AFCS mode.

Note that the TRIM switch has four positions: NOSE UP, NOSE DOWN, LEFT WING DOWN, and RIGHT WING DOWN. Thus, the signals originating from the elevon trim control correspond to pitch and roll signals. Note that the signals from the elevon trim switch are routed through either the elevon or aileron channel relays to the AFCS components. The signals in the AFCS components are controlled so that a fixed full authority pitch or roll signal is applied to the AFCS summing amplifier in the elevon amplifiers. Consequently, the control surfaces move at a constant rate while the pilot is holding the trim switch in the desired position. When the pilot releases the trim switch from a roll position, the AFCS will return the aircraft to a level attitude providing the roll angle is less than 5°. Above 5°, the AFCS will maintain the roll attitude.

THE DIFFERENCE BETWEEN AFCS AND DAMPER SIGNALS.

In the following section you will learn how the AFCS signals extend the movement of the control surfaces beyond the damper limits, and also the different type of feedback necessary as a result of this extended movement.

THE FULL AUTHORITY SUBSYSTEM.

In order for the AFCS to control the aircraft over the normal limits of pilot control, the system requires complete control of the elevons. As you will recall, the limits incorporated in the pitch damper for elevon movement was plus or minus (±) 1°. In the AFCS, the elevons must be permitted to move through their full limits (25° up and 8° down). With the damper system arrangement of mechanical and electrical feedback, the full limits of movement are impossible. Therefore, you will find that two components have been introduced to permit the expansion of the limits. These components consist of a potentiometer at the control surface that indicates the position of the elevons, and an electrical lockout device in the servo actuator linkage (or HEP valve on later aircraft) that prevents mechanical feedback during AFCS operation.

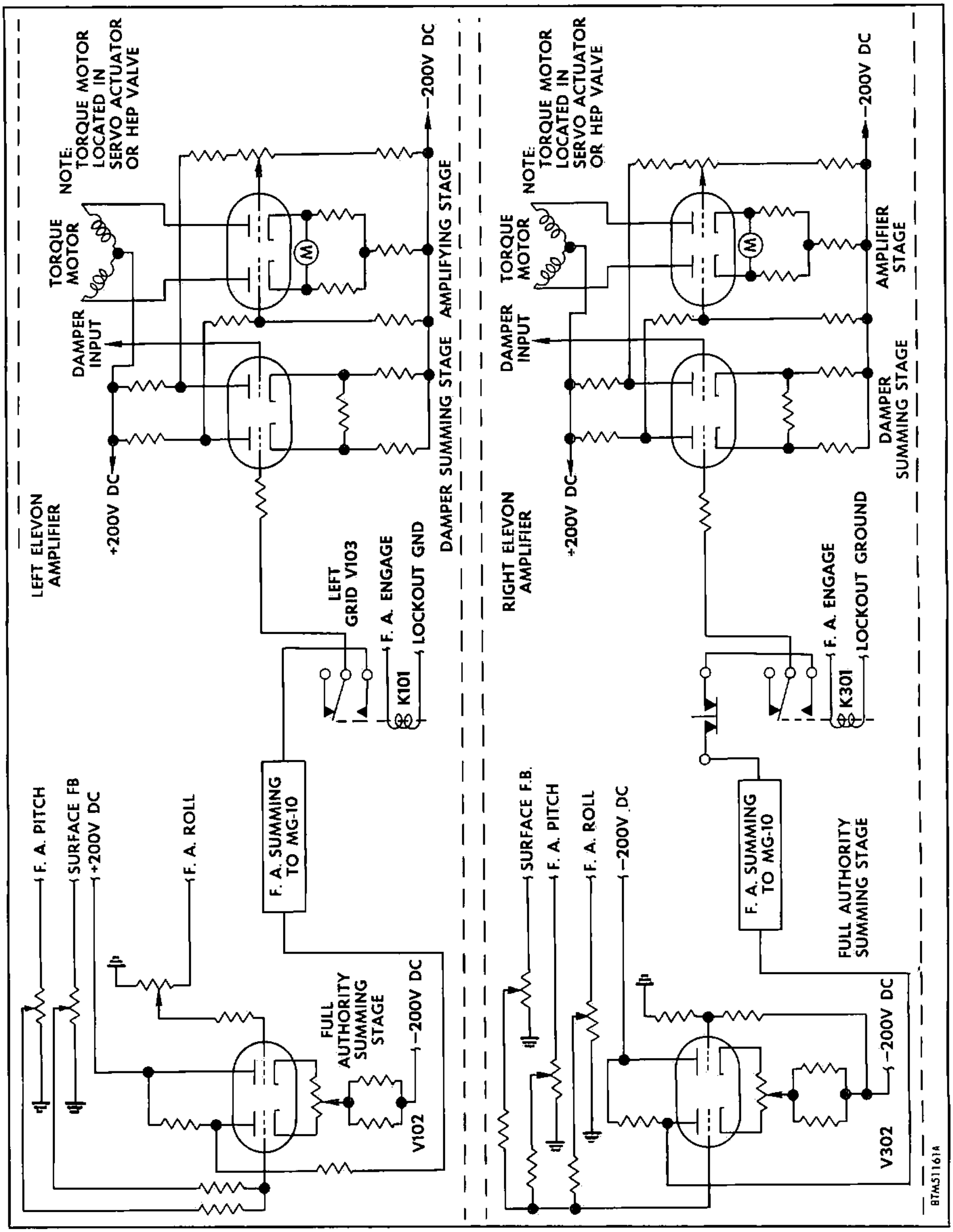


Figure 5-5. The Full Authority Connections to the Elevon Amplifiers

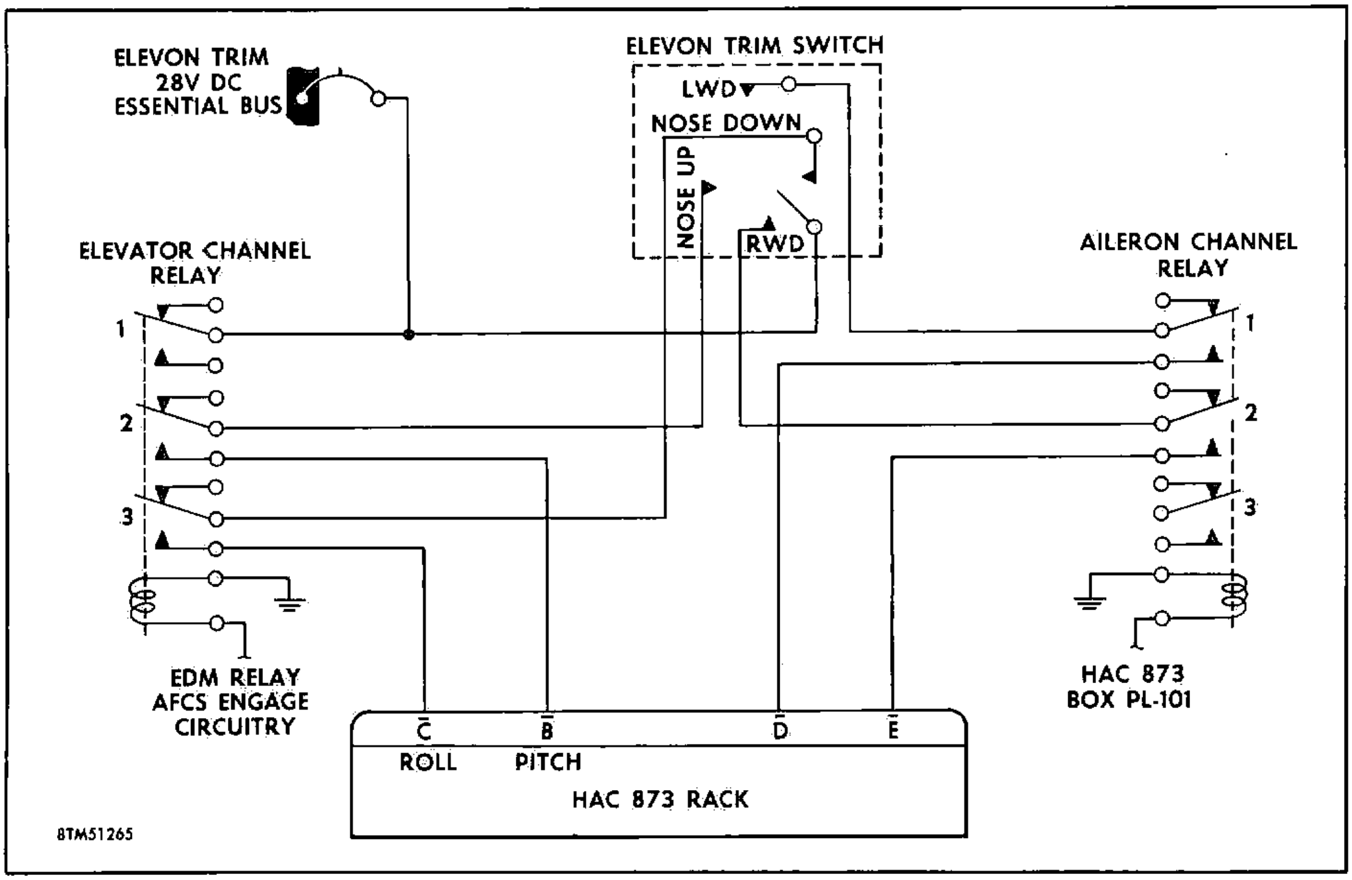


Figure 5-6. The "Beep" Trim Circuits

THE SERVO ACTUATOR AND LOCKOUT VALVE.

In figure 5-7 you can see the position of the lockout valve in relation to the other components of the servo actuator control linkage. When the AFCS is engaged, the lockout valves in both the right and left elevon control linkages are extended. Note that the motion of the elevon will move the top of the walking beam that normally furnishes the mechanical feedback. Thus when the control valve closes, the servo actuator cannot be centered because it must fill the distance between the control valve and the top of the walking beam.

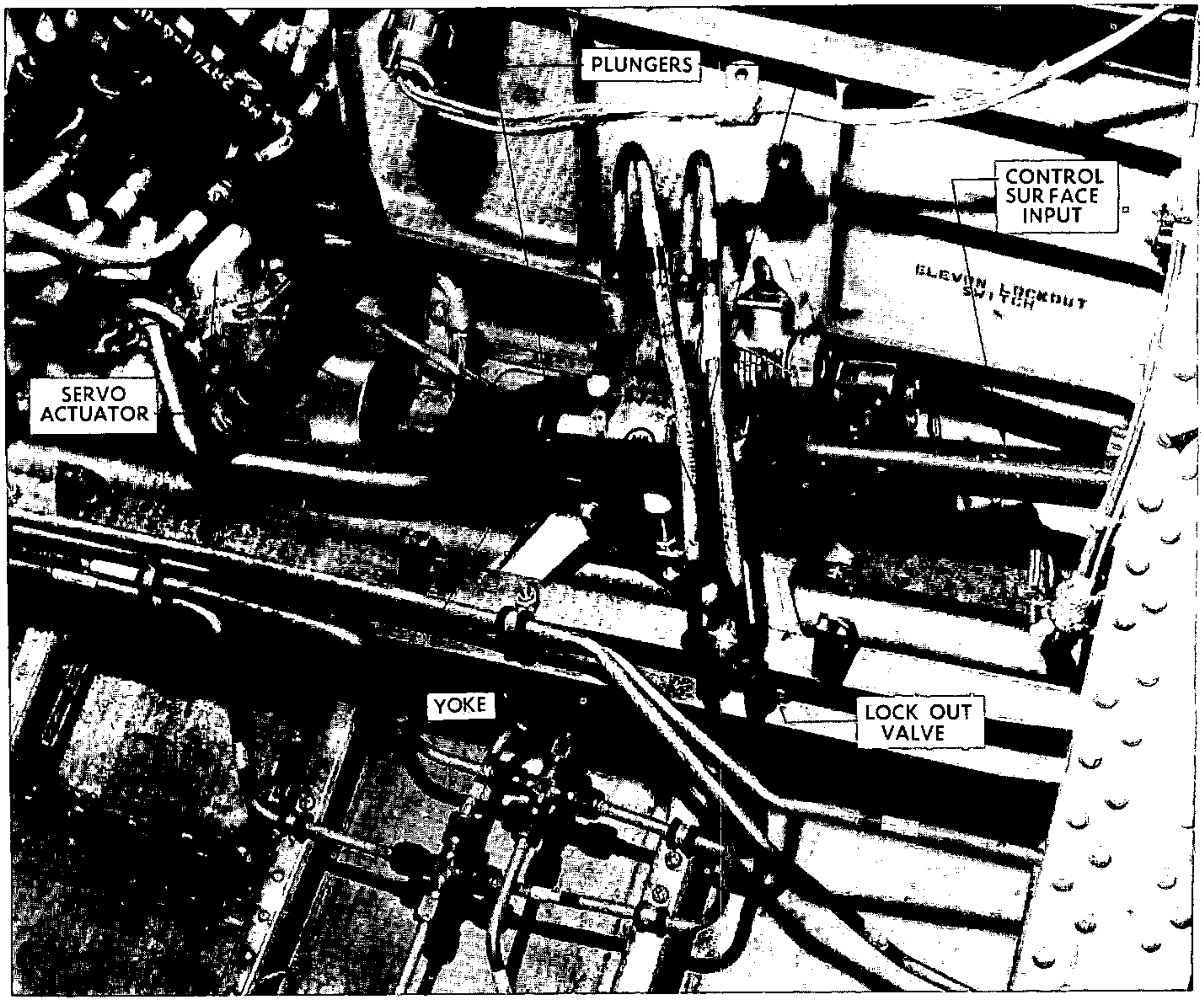
If the top of the walking beam were prevented from moving the servo actuator piston back and forth, the control valve would move as desired. The solution to this is to lock out the mechanical feedback. This is done by attaching the linkage between the servo actuator and the top of the walking beam to the structure of the airplane. (Remember that it is necessary to do this only in the AFCS mode—not in the damper operation.)

As you can see in figure 5-7, a yoke is formed in the linkage. Inside the yoke, note that there is a hydraulic valve attached to the aircraft structure. This valve is positioned in such a manner that the linkage is free to move a limited distance. However, in AFCS operation, full authority is required and plungers at each end of the hydraulic valve will extend to lock the linkage to the aircraft structure. This valve is known as the lockout valve because it *locks out* the mechanical feedback.

With the top of the walking beam locked so that it cannot move (due to the energizing of the lockout valve), the motion of the control surfaces will move the entire walking beam. This causes the stick to move according to the position of the elevons. The force exerted on the plungers of the lockout valve is not great enough to prevent the pilot from overriding the hydraulic valves and controlling the airplane in case of an emergency.

THE HEP VALVE.

In figure 5-8 you can see the HEP valve as it is installed in the aircraft. There are four electrical components associated with the HEP valve — the lockout solenoids, the unlock solenoids, the limit switch, and the torque motor. Only the torque motor can be observed from the exterior of the valve. The other components are located within the unit. The purpose of the limit switch is to complete the circuit for AFCS engagement. As long as the lockouts of the HEP valve



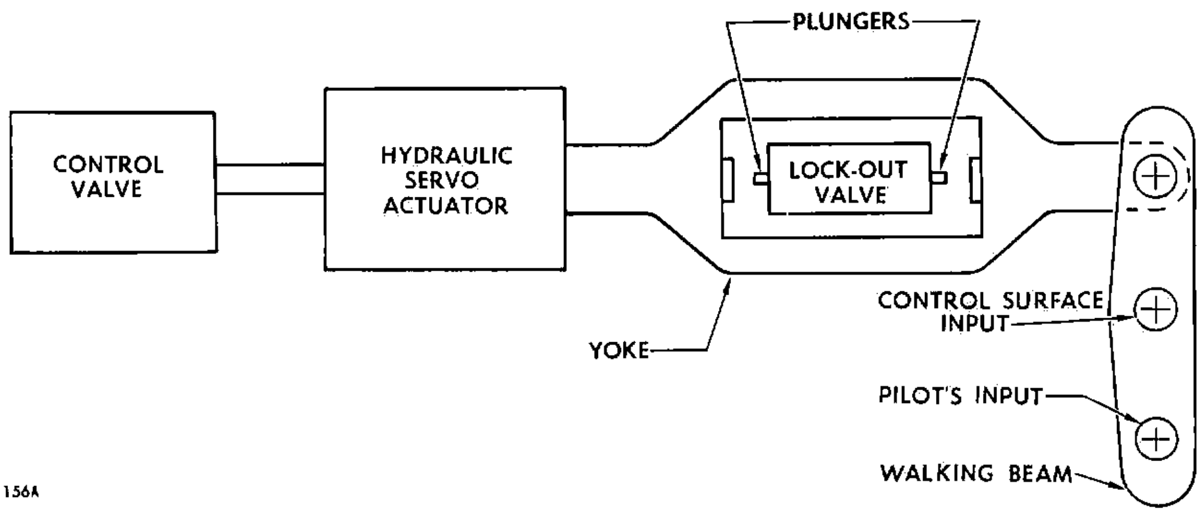


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Figure 5-7. The Lockout Valve and Servo Actuator

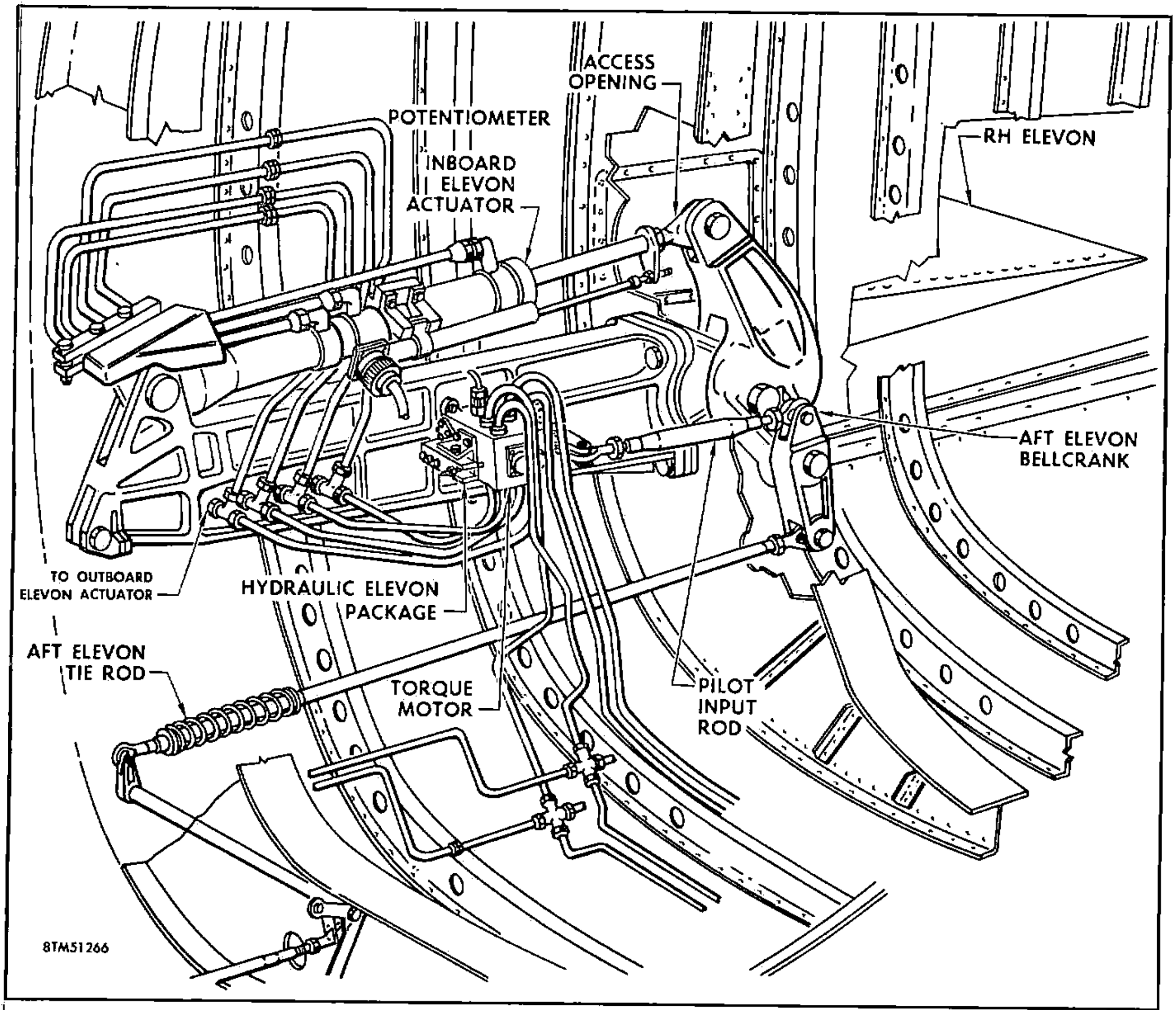

Figure 5-8. The HEP Valve Installation

are retracted, the switch is open; when the lockouts extend, the switch closes and the engage circuitry for the AFCS is completed.

Note the manner in which the lockouts prevent the pilot inputs and mechanical feedback from operating within the control valve. The mechanical feedback must be locked out during full authority operation because the AFCS could not hold the aircraft in a desired pitch or roll attitude. The reason for this is that as soon as a signal would be applied to the torque motor and the control surface started to move, the mechanical feedback would be shutting off the control valve and neutralizing the control surface movement. The AFCS operation requires full authority over the flight controls; therefore, mechanical feedback is undesirable.

You are already familiar with the purpose of the lockout solenoids. The purpose of the lockout valves is to control the porting of hydraulic fluid to the lockout pins and to lock the parallelogram linkage so that mechanical feedback is prevented from closing the flow of fluid to the control surface actuator.

The unlock solenoids are required for both damper and AFCS operation. These solenoids must be energized to permit fluid to enter the chamber around the torque motor and to permit damper and AFCS signals to operate the control surfaces.

THE CONTROL SURFACE POTENTIOMETER.

Now let us examine the potentiometer at the control surface and see how it affects the AFCS (figure 5-9). With the prevention of the mechanical followup by

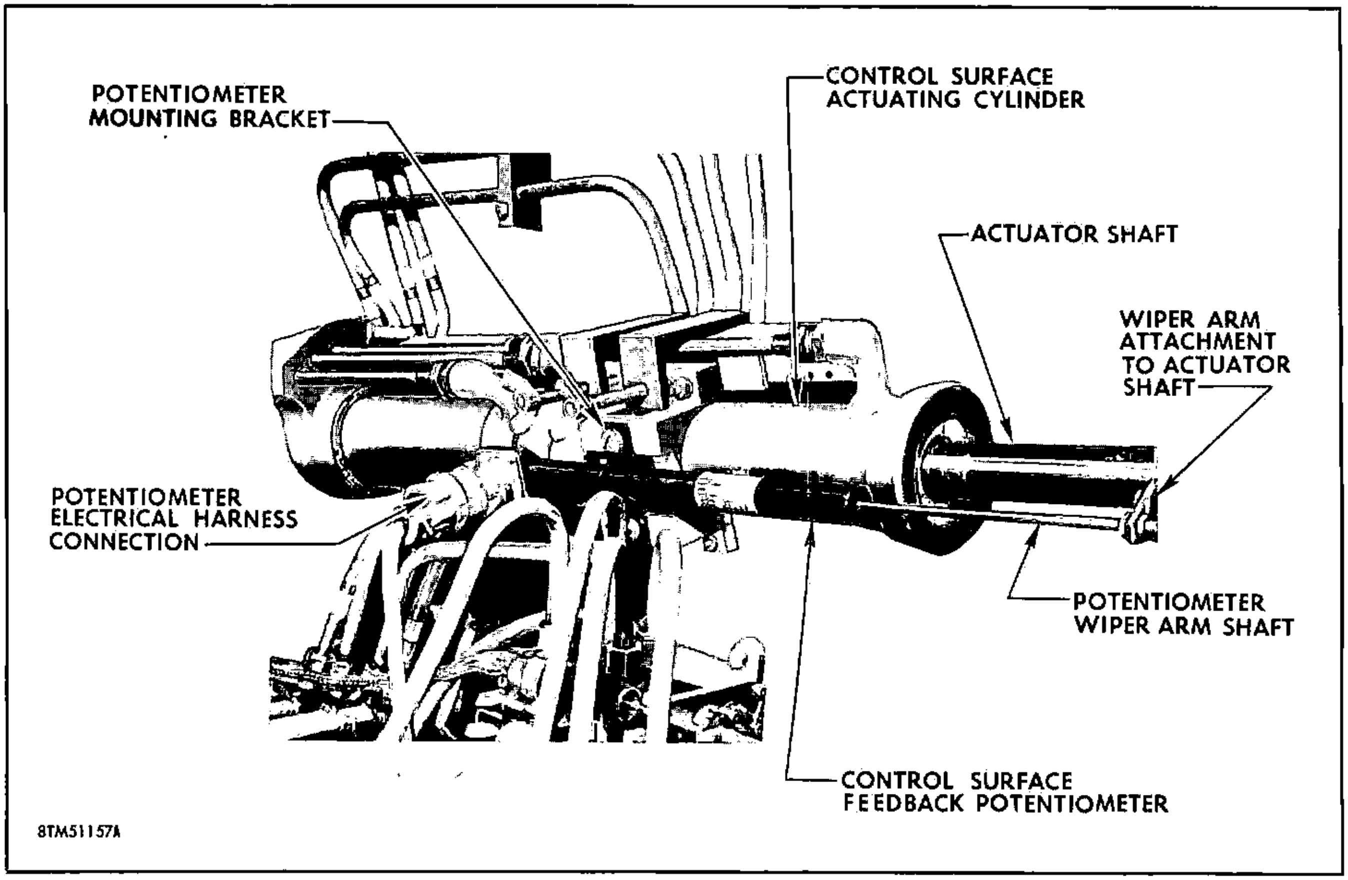

Figure 5-9. The Control Surface Potentiometer

the action of the lockout valve, an electrical surface feedback is necessary to establish closed loop control in AFCS operation. This results in two servo loops during AFCS operation with the servo actuator, but only one if the HEP valve is installed. You are already familiar with the servo actuator loop from your study of the damper system in Chapter IV. The other loop —the surface position loop—applies only to the AFCS.

THE TWO AFCS CLOSED LOOP SYSTEMS.

The block diagram in figure 5-10 shows you how the two closed loop systems operate in early model aircraft. As you can see, one loop is from the servo actuator to the amplifier and the other is from the control surface feedback potentiometer to the amplifier. For purposes of this discussion, let us assume the AFCS sends a 50-volt signal to the elevon damper amplifier.

The Servo Actuator Loop.

Follow the signal from the full authority summing amplifier and note that it goes to the elevon servo amplifier. From this unit, the operation is similar to the damper system operation. The signal goes through the elevon servo amplifier and to the torque motor of the servo actuator. The servo actuator extends its full throw (0.045 inch) and opens the elevon control valve. This results in the movement of the elevon surface.

Now let us examine what happens in the closed loops of this network. The rod of the servo actuator moves to its full throw position in this operation. You can see that it causes a wiper arm in the servo actuator to pick off a voltage from a potentiometer. (This operation is identical to damper system operation.) For example, let us say this voltage is 7.5-volts d-c. Following the flow in the block diagram, figure 5-10, you can see that this 7.5 volts is fed back to the full authority summing amplifier where it is summed with the original command signal of +50 volts. This gives an error voltage of +42.5 volts (+50 volts —7.5 volts) that goes to the elevon servo amplifier.

At this time the elevons begin to move at a maximum rate because the servo actuator is at full displacement, thus porting full hydraulic flow to the control surface actuators. As the control valve opens and permits the porting of hydraulic fluid to the control surface actuators, there is movement of the control surfaces. The control surfaces do not move at full rate until the servo actuator control valve is fully open and the error feedback is maximum. When the servo actuator loop error voltage reaches its maximum value, it immediately starts decreasing at a constant rate along with the surface position loop error signals.

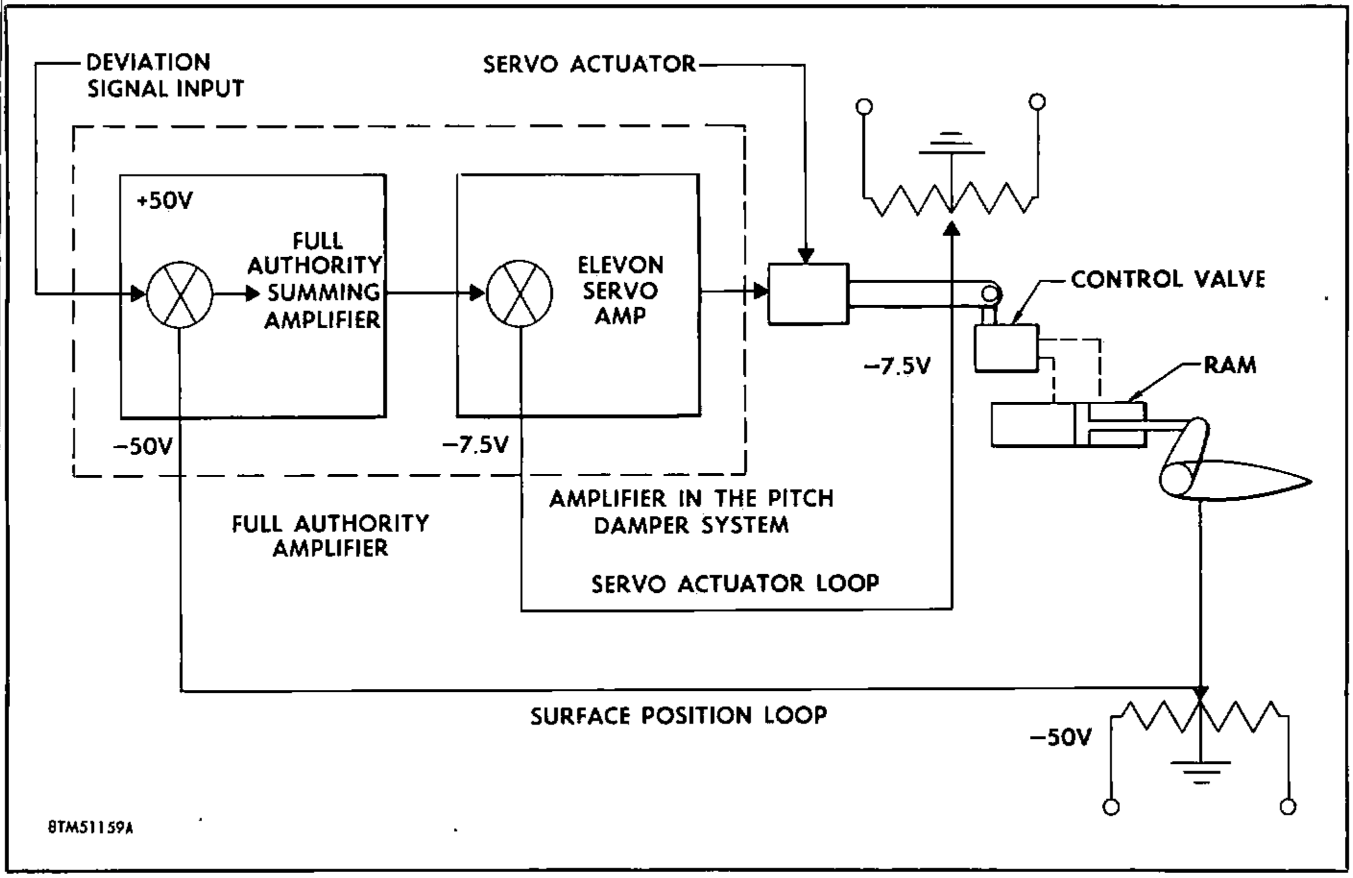

Figure 5-10. The Two Full Authority Closed Loop Systems

The servo actuator and surface position loop error signals are separated by a 7.5-volt feedback from the servo actuator feedback pot. You can see in figure 5-11 that the curves for error voltages in these loops drop at a constant rate. This happens because the control surfaces are moving at a maximum rate, and surface position feedback signals are therefore increasing in proportion to this rate.

The Control Surface Feedback Loop.

Now let us consider the control surface feedback loop. This is quite different from the loop you studied in the damper system operation, which is also the loop discussed above. The control surface feedback potentiometer has a much wider voltage range than the servo actuator feedback potentiometer. The reason for this is that the AFCS has full authority over the elevons, and can move the elevons through their maximum limits. The damper system, as you will recall, had elevon limits of $\pm 1°$. Thus, you can see why it is necessary to have a much larger potentiometer for the AFCS control surface feedback.

In figure 5-11 note that as the elevons move up, they reach a point that results in the control surface feedback pot sending a voltage (exceeding the 42.5-volts error signal in the servo actuator loop) to the amplifier. In the illustration, the dotted line represents the control surface loop error signal. When this occurs, the servo actuator loop error signal changes sign and the control valve starts to close. The elevons still continue to move up until the feedback voltage equals the decreasing input voltage from the full authority amplifier.

The reason the full authority amplifier sends a decreasing voltage is that the reference units sense the corrections being made by the control surfaces. Therefore, as the control surfaces move the airplane back on the desired course, the natural operation of the reference units is to send less deviation signal to the full authority amplifier.

When the control surface feedback voltage equals the input voltage, note that there is a short period of time before the negative servo actuator loop error voltage closes the control valve. You can see then that the servo actuator loop error voltage changed its sign as the control surface feedback voltage exceeded the control surface position loop error voltage and that the signal caused the control valve to close.

The elevons stop moving at the point where the control surface feedback equals the input from the full authority amplifier. They will remain in position as long as the +50-volt command signal continues from

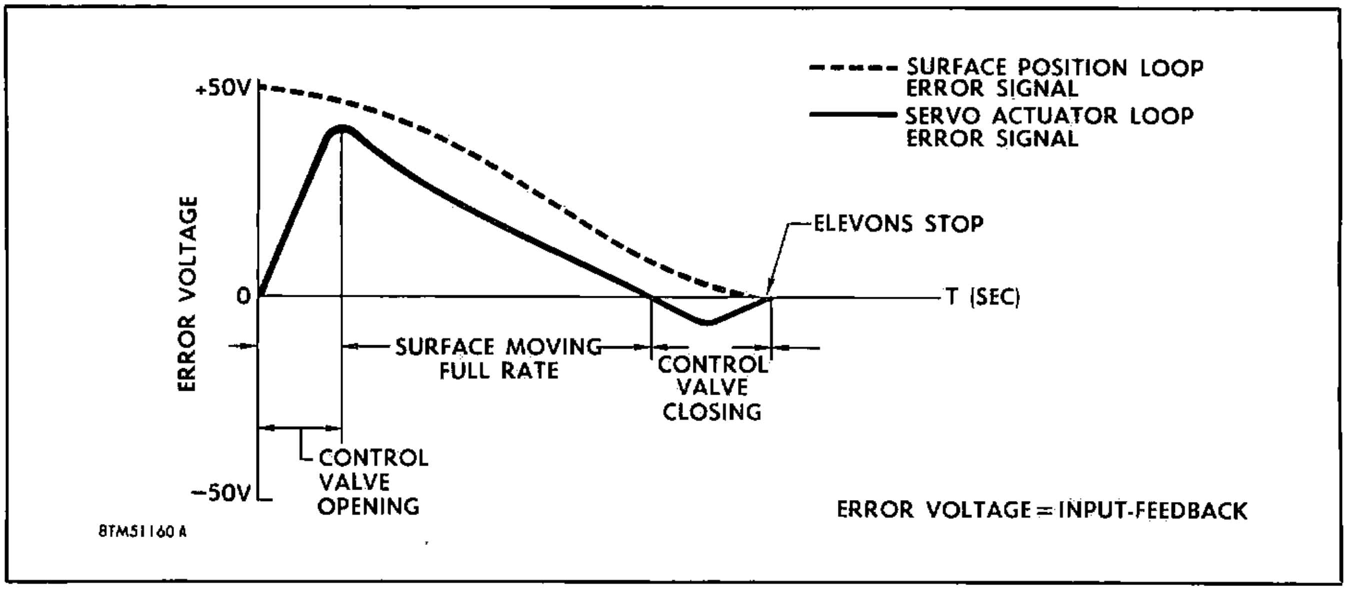

Figure 5-11. Error Signal Time

the full authority amplifier. When the command signal is withdrawn, the control surface position loop error signal is equal to the control surface feedback voltage, and the reverse condition exists in the system. This causes the elevons to move back to neutral.

Summary of Loop Operation.

Reviewing the operation of these two loops, you can determine the behavior of the system at any particular instant simply by studying the error signals. You must remember that the surface position error voltage is derived from the difference between the full authority amplifier input and the control surface feedback voltage.

When the input signals equal the feedback signals, the error voltages in the loops are zero and there is no control surface movement. The same condition holds true when the input signals and the feedbacks are zero. In addition, you can determine the direction the control surfaces will move by examining the size of the signals. When the inputs are greater than the outputs, the surfaces move away from neutral; and when the feedback signals are greater than the inputs, the surfaces move toward neutral.

The error signal time is shown in figure 5-11. You can see that the +50-volt input from the full authority amplifier is maintained until the elevons reach a position proportional to the +50-volt signal. Considering the dotted line—which is the surface position loop error signal—the value of the error signal is equal to the +50-volt input signal (at time zero) since the feedback is zero. The input signal—amplified in the elevon amplifier—is fed to the servo actuator. This causes the control valve to open and the elevons to start moving. Immediately, feedback from the servo actuator and the surface potentiometers begins to subtract from the input signal and the error curve starts to drop and reaches zero when the feedback equals the input signal. At this point, since the error signal is zero, the elevons stop moving and will remain in this position until the error signal changes from zero.

Examination of the servo actuator loop error signal curve illustrates the manner by which the servo actuator receives signals and opens and closes the control valve to obtain surface movement. You can see in the schematic on figure 5-1 that the surface position loop error signal summing is accomplished at the full authority summing amplifier. Therefore, the input to the elevon pitch damper amplifier is the surface position loop error signal.

When the large error signal is fed to the servo actuator loop, the servo actuator receives its full throw and the elevons start to move up. The servo actuator loop error voltage rises very quickly to a value almost equal to the surface position loop error signal. The servo actuator, in moving to its full throw position, feeds back the 7.5 volts that subtracts from the surface position loop error signal as full throw is reached. The servo actuator loop error curve then follows and parallels the surface position loop curve at a value of 7.5 volts below the surface position loop curve. During this time the elevons are moving up at a maximum rate since the control valve is fully open.

At the point where the surface position loop equals 7.5 volts, the servo actuator loop error voltage is zero. The elevons are still moving up, however, and the surface position error loop is still decreasing toward zero. The result is that the servo actuator loop error signal changes sign and the servo actuator receives a signal which has decreased beyond zero and changed sign. The actuator then starts to extend and close the control valve.

In figure 5-11, this point is where the servo actuator loop error crosses the line representing time. The surface position loop error signal is still decreasing because the elevons are still going up. This in turn causes the control surface feedback signal to increase. Soon, however, the elevons will reach a point where the feedback equals the input and the error is zero. Now the servo actuator loop error signal is provided by the servo actuator feedback. The negative servo actuator loop error is calling for a return of the servo actuator to neutral. Consequently the control valve is closed and the elevons stop moving. This elevon position is then held as long as the +50 volts is maintained from the full authority amplifier.

When the input from the full authority amplifier is decreasing, the elevons are returning to neutral because of the large negative (with respect to the input) signal from the control surface feedback pot. The opening of the control valves is provided in the same manner as before, only this time causing the elevons to return to neutral. A similar change in the sign of the servo actuator loop error signal closes the valve as the elevons reach neutral.

CALIBRATION OF THE FLIGHT CONTROLS.

It is not the purpose of this section to give you a detailed account of the flight controls calibration procedure. You will find these details in the F-102A Flight Controls Maintenance Manual, T.O. 1F-102A-2-7. As you examine the maintenance manual, you will find that there are separate procedures for calibration of the various types of equipment installed in the F-102A damper systems. Even though the components vary—HEP valve or servo actuator; —2, —4, or —6 amplifiers; and pitch "G" limiter—a general description of the purpose of calibration will help you to understand what the calibration procedure is designed to do.

The test equipment consists of a unit designed to simulate or control the signals from the sensing unit of the damper and from the AFCS system. As you can see in figure 5-12, calibration requires ground power units for both the electrical and hydraulic systems as well as a manometer regulator unit for simulating inputs to the "Q" probe on the vertical stabilizer. The maintenance manual gives detailed directions for the proper attachment and operation of the ground power units and directions for the proper attachment of the manometer regulator.

THE FIELD TEST UNIT.

The field test unit shown in figure 5-12 is typical of the units used to calibrate the damper systems (Stability Augmentation Subsystem) and the AFCS interconnection system (Full Authority Subsystem). Not only is the unit used for calibration but also as a troubleshooting aid in locating malfunctions in the subsystems.

The test unit introduces either d-c or one-half cps signals to the input channels of the damper or AFCS interconnection systems. The calibration signals are substituted for the normal aircraft signals, although normal aircraft circuitry can be routed through the test unit for calibration and checking purposes.

The cable connection to the amplifier rack will vary depending on the series of aircraft and the type of damper system components installed. On early model aircraft, you will find that engagement of the damper system, as well as the AFCS, must be accomplished from the cockpit. On later model aircraft, you will find that these systems may be engaged from controls on the test unit. There are other details peculiar to the various systems listed in the Flight Controls Maintenance Manual, T.O. 1F-102A-2-7. Be sure that you read carefully the directions for cable attachment before performing a calibration.

Note the main controls and areas of the test unit panel shown in figure 5-12. Let us locate the controls and briefly describe each of them. The POWER SWITCH is a push-on, push-off type switch located at the bottom left of the panel and directly below the power indicator light. The SIGNAL SOURCE SWITCH has two positions—AIRCRAFT and CALIBRATOR. In the AIRCRAFT position, the signal source switch maintains normal circuit conditions in the aircraft system; in the CALIBRATOR position, calibration signals are substituted for the normal aircraft signals.

In the first (AIRCRAFT) position, normal aircraft circuitry is restored. The CALIBRATOR SIGNAL SWITCH selects a signal of preset level for calibration (an adjustable one-half cps signal from the oscillator or an adjustable d-c signal). The lights above the POLARITY SELECTOR SWITCH indicate the type (a-c or d-c) of signal and the polarity if it is d-c. The D-C SIGNAL ADJUST CONTROL adjusts the level of the d-c signal when the CALIBRATOR SIGNAL switch is in the D-C SIGNAL ADJUST position. The adjustment does not permit you to apply a signal larger than the preset signal, but merely allows a variation from the maximum preset signal down to or approaching zero volts.

The OSCILLATOR ADJUST CONTROL adjusts the level of the one-half cps signal when the calibrator signal switch is in the OSCILLATOR ADJUST position. The POLARITY SELECTOR switch selects positive or negative signals, as desired, in the D-C SIGNAL ADJUST position of the CALIBRATOR SIGNAL switch, and in those channel selector positions which

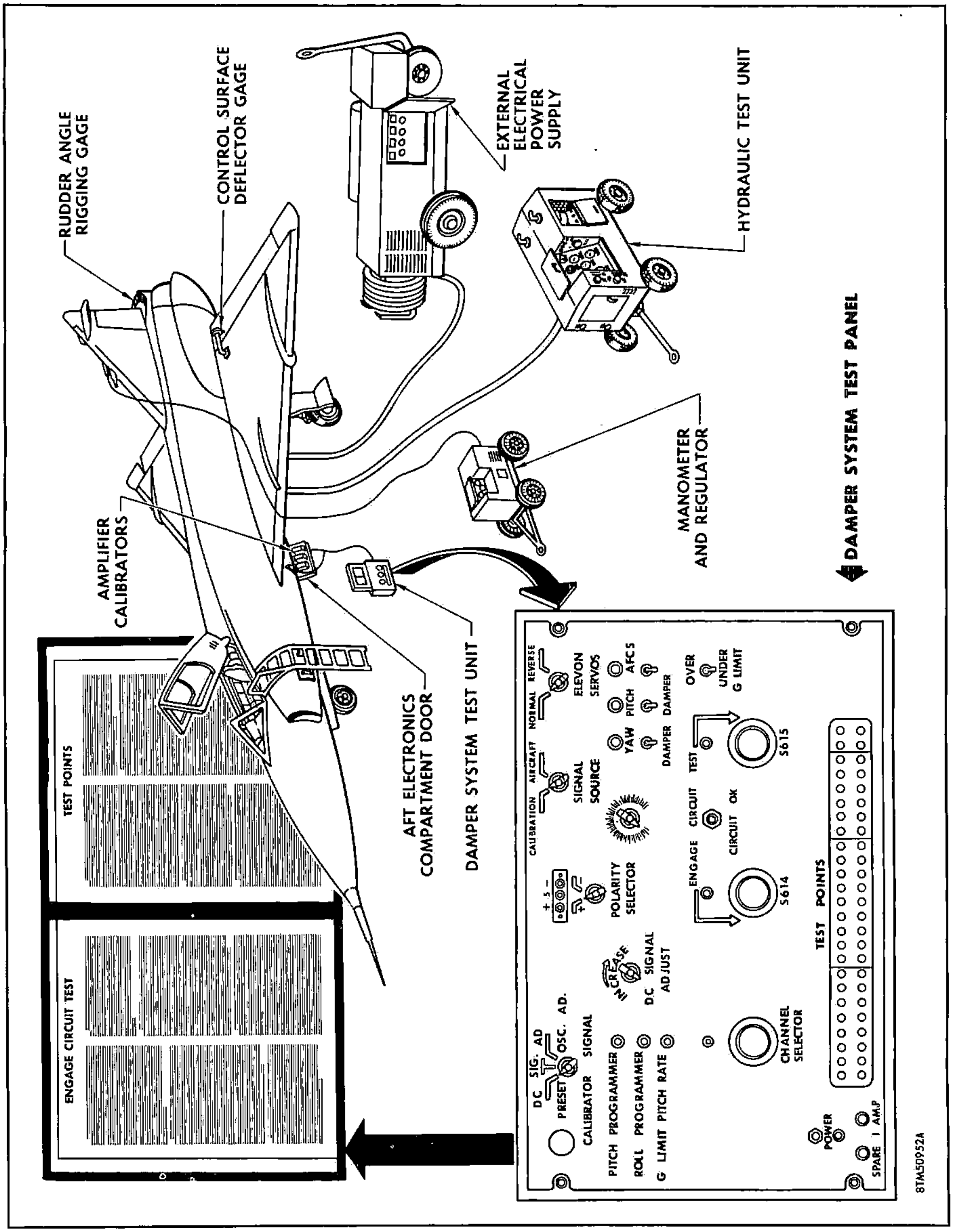

Figure 5-12. A Typical Calibration Scene

supply d-c signals in the preset position of the CALIBRATOR SIGNAL switch. (The POLARITY SELECTOR has no effect on a-c signals.)

The CHANNEL SELECTOR switch applies the calibrator signal to the desired channel for calibration and grounds those amplifier inputs not being calibrated. The switch is the rotary type that must be pulled out before switching from one position to the other. A lighted number above the switch indicates the switch position. The "G" LIMIT "UNDER AND OVER" switch selects signals preset at the upper and lower tolerance limits of the "G" limiter operation to provide a check of the correct settings of these limiters. The PITCH PROGRAMMER and ROLL PROGRAMMER switches are momentary contact switches that supply selected calibrator signals to the corresponding pitch and roll input circuits.

The "G" LIMIT PITCH RATE switch is the momentary type that supplies the selected calibrator signal to the "G" limit pitch rate inputs. The YAW DAMPER switch on the test unit corresponds to the MANUAL-DIRECT MANUAL switch in the aircraft. The PITCH DAMPER switch in the test unit corresponds to the PITCH DAMPER switch in the aircraft, and the AFCS switch parallels the AFCS/OFF switch. The ELEVON SERVOS switch in the REVERSE position connects left amplifier signals to the right actuator and the right amplifier signals to the left actuator. Under these circumstances, it is possible to isolate malfunctions in the elevon actuators or amplifiers.

The "S-614" and "S-165" ENGAGE CIRCUIT TEST switches connect an indicator lamp labeled "CIRCUIT OK" to various test points in the engage circuitry to indicate the proper circuit condition at that point. The test points consist of pin jacks at the bottom of the front panel. The pin jacks provide access to signal sources and system voltages for test purposes. In the lid of the calibrator case is a complete list of the test points, their functions, and their relation to the damper and AFCS interconnection system. By following the directions and tables given in the list you can check all applicable voltages in the system.

PURPOSE OF CALIBRATION.

The main reason for calibrating the pitch and yaw damper systems is to insure that signal outputs from the amplifiers provide the correct control surface movement. As you will recall, there are several points in the system where it is possible for unbalanced conditions to exist—the vacuum tubes in the differential amplifiers, the position of the wiper arms on the potentiometers of the feedback circuits, and variations in electronic components such as resistors and capacitors.

It is important to remember that calibration does not involve adjusting the control surfaces to these unbalanced conditions; but instead involves compensating in the circuits for the outputs to the control surfaces. The calibration of the system was planned to make it as convenient for you as possible. Therefore, you will find that the adjustments are located where they are the easiest to reach. There are no adjustments to perform on the rate gyros; neither are there any on the feedback potentiometers on the servo actuators. The controls are all located on the front of the amplifier-calibrator units so that you can work at ground level. You will remember that the amplifier-calibrators are located on the aft electronics compartment door.

CONTROL SURFACE RESPONSE.

One of the most important phases of calibration is the measurement of control surface response to damper system signals. To aid you in measuring the control surface response, there are two special measuring tools —one for the rudder, and the other for the elevons. Of course, prior to calibration you will have checked the rigging and determined that the mechanical movement of the surfaces is satisfactory.

THE RUDDER ANGLE RIGGING GAGE.

The measuring device for rudder movement is known as the rudder angle rigging gage. As you can see from the illustration in figure 5-13, this device is similar to a protractor. It attaches to the stub island at the bottom of the rudder. It is interesting to note that the rudder acts as its own indicator and that you read the rudder movement from neutral directly from the rudder position in relation to the protractor. If the initial position of the rudder does not correspond to the neutral position on the protractor, you can adjust it through actuation of the trim control system.

THE CONTROL SURFACE DEFLECTOR GAGE.

A somewhat different device measures elevon movement. Notice the control surface deflector gage in figure 5-14; this gage attaches to either wing at the position shown. The base of the gage is fixed to the wing by two installation pins. First, however, you must remove the studs in the wing hoist fittings to permit the pins to be attached to the fittings.

It is extremely important to align the elevons to the neutral position. There is a reference rivet on the fuselage (where the elevons meet the fuselage) that you must use in determining the neutral position. As with the rudder, you may use the trim system to assist in obtaining elevon neutral positioning.

After you have adjusted the elevons to the neutral position and attached the base of the gage to the position shown in the illustration, there are minor adjustments you must perform to assure accurate readings. As you can see in the illustration, there is an adjusting nut where the lever attaches to the base. You can

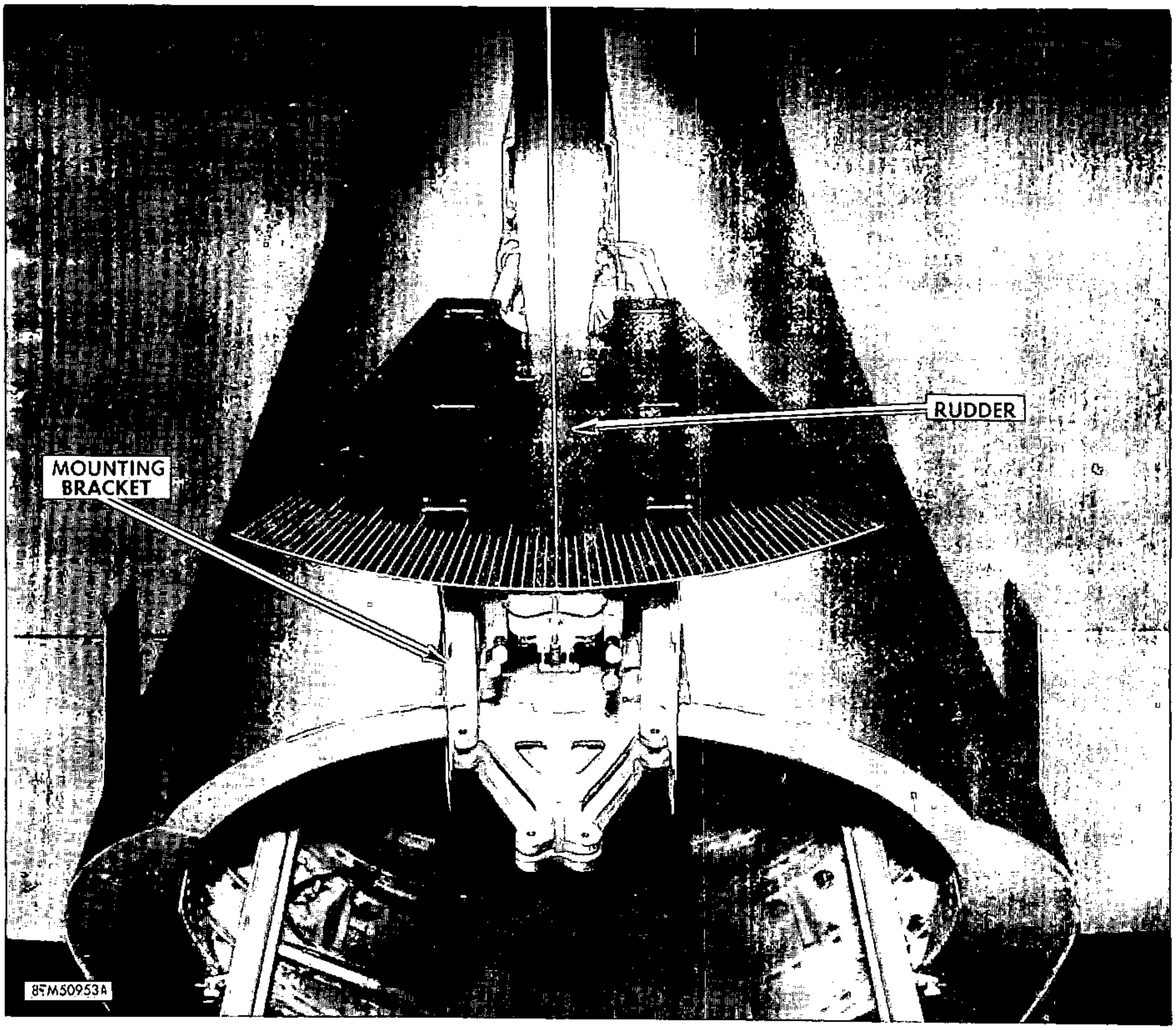

Figure 5-13. Rudder Angle Rigging Gage

adjust the lever to the desired position for neutral readings. The neutral position readings are 5 for the small dial and 0 for the large dial.

The dials operate in the following manner: the probe that touches the elevon surface is movable and follows elevon movement up and down, and as it follows the movement, a ratchet within the gage causes the indicator dials to move. The large dial moves one complete revolution for each degree of elevon travel; the small dial reads total degree travel and has a maximum reading of 10°.

You must adjust the lever arm so that the small dial reads 5°. This permits 5° measurement of elevon travel in either direction from the neutral point. Since the large dial is calibrated in thousandths of a degree, it would be impractical to adjust the lever arm until the large dial read 0. Therefore, there is a provision for adjusting the face of the dial so you can place the 0 under the pointer.

TROUBLESHOOTING AND MAINTENANCE PROBLEMS OF THE DAMPER SYSTEMS.

In troubleshooting the damper systems, there are several types of troubles you may encounter. These troubles may be grouped under two main headings: those problems dealing with power distribution within the damper system proper and those involving output signals of the damper system components.

Under this first grouping, power distribution, let us consider one of the most important areas of the damper systems — the circuit breakers connecting power from the buses to the damper system circuits.

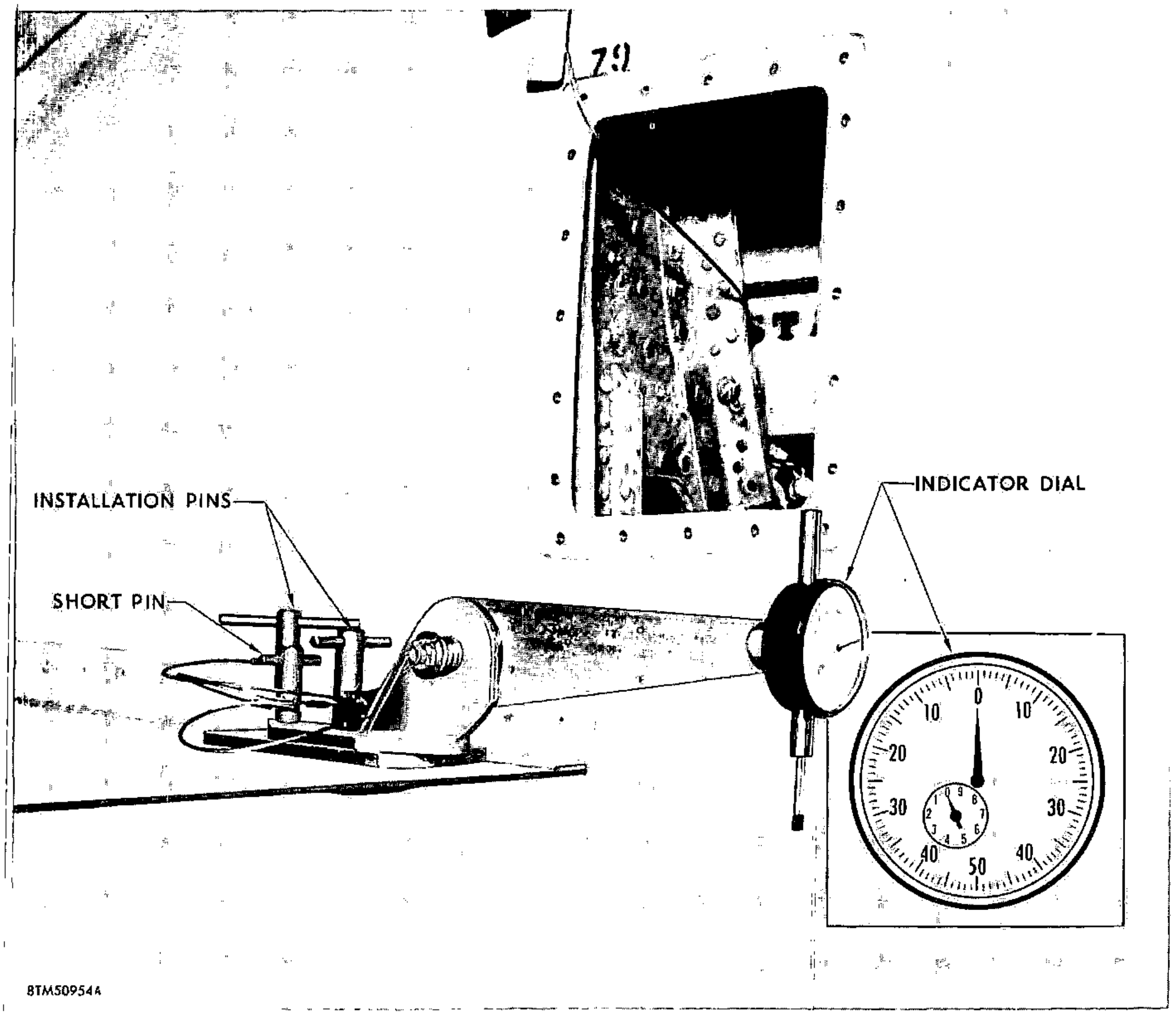

Figure 5-14. Control Surface Deflector Gage

A common malfunction is caused by the failure of a circuit breaker to stay in the closed position. This indicates that excessive loads are in the circuit, such as a direct short, or that the mechanism of the circuit breaker is defective.

To check for this condition, you must open the circuit breaker and place an ohmmeter between the deenergized portion of the circuit breaker and ground. This will indicate either that the circuit is satisfactory or there is a direct short. If the circuit does not reflect a direct short, then it is safe to assume the trouble is in the circuit breaker mechanism. You should replace it.

On the other hand, if the circuit reflects a direct short, you must disconnect all the components that connect to the circuit from the circuit breaker. With all the components disconnected, you must check the aircraft wiring from the circuit breaker to the using components. If this does not reveal the short, re-engage the circuit breaker and reconnect the components one at a time until the circuit breaker "pops" to the open position. The trouble lies in the component that you reconnected just prior to the "popping" of the circuit breaker.

Another problem is the high readings of the +105 and —105 d-c voltages. The initial check for this condition is to open the circuit breaker and look for disconnect components. If this condition does not exist, then you must check for an *open* circuit in the aircraft wiring that supplies these voltages to the various components. After determining that the trouble does not exist in the wiring, the next step is to check for an *open* potentiometer in the components. As a last resort, you should replace the rudder-power amplifier-calibrator.

The reverse of this situation is the reading of low +105 and —105 d-c voltages. In this case, you should first disconnect the circuit breaker and check the resistance between the 105-volt circuit and the signal ground. The resistance should be within 2700 to 3000 ohms. If the resistance is low, the next step is to disconnect all of the components of the system that use these voltages and measure the resistance between the 105-volt circuits and ground for an open circuit.

Another check is the reading of the potentiometers in each of the system components for their proper value as indicated from the schematic diagrams. A final check is the replacement of the rudder-power amplifier-calibrator.

When the DAMPER ENGAGE switch will not "hold" in the ON position once you have actuated it, you must check the FLIGHT CONTROL DAMPER circuit breakers (both 28-volts d-c and 115-volts a-c) to make sure they are closed. If they are closed, the next item is the damper engage circuitry. You have already learned the operation of this circuit; so you are familiar with the conditions for energizing the holding relay at the DAMPER ENGAGE switch.

Another problem is the placing of the damper switch to the MANUAL position, and the unlock solenoids not energizing. With the engage circuitry schematic, check the conditions for energizing the unlock solenoids.

Now we come to those problems concerning the output signals from the damper system components. Suppose when you engage the damper system, the rudder moves violently in either direction. This condition means the engage transient at the rudder is too great. The first item to check is the output from the aileron position potentiometer with the ailerons at zero; this output should read 0 $\pm$.2-volt.

If the trouble still exists following this check, you must now determine if the actuator feedback gain control on the front of the rudder-power amplifier-calibrator is not at zero. You check this by observing the position of the control knob with reference to the calibration marks on the front of the panel.

The next item to check is the zeroing control on the amplifier-calibrator. As you move the zeroing control slowly and evenly, observe the rudder movement. If it moves abruptly from one damper limit to the other (6° on either side of neutral) you must check for the presence of an actuator feedback signal. This is done by adjusting the zeroing control while observing the actuator feedback signal on the voltmeter. The next check to make is to push the zero button on the front of the amplifier and observe which way the needle moves as you adjust the zeroing control. As you move the zeroing control, the needle should move smoothly. However, if the needle on the meter remains in the center of the dial or at one extreme, this indicates the resistance in the torque motor coils might be out of tolerance. When such is the case, you will have to replace the servo actuator or HEP valve. If the torque motor coils are satisfactory, the next item to check would be the wiring from the actuator to the amplifier for continuity.

Another maintenance problem you may encounter is that the engage transient at the elevons is too large. This means they snap into position when you engage the damper system. In general, the check for this condition is quite similar to that for the rudder. Adjustment of the appropriate elevon amplifier zeroing controls, gain controls, and observation of the meter response to zero control adjustment applies here the same as in rudder troubleshooting.

A chattering of the control surfaces, when you engage the damper system, indicates "noise" is present in the electronic components. This check involves the isolation of the unit delivering these erratic output signals to the servo actuator. It is anticipated there will be a provision on future test units to ground all the circuits to the torque motor. This will permit you to cut off all signals to the control surfaces. When the controls are quiet, you can simply introduce the input signals one at a time until the chattering starts once again. The component sending the signal last introduced must then be changed.

When the control surfaces do not move smoothly in response to a smoothly applied signal, you should replace the servo actuator since there is probably a malfunction in either the torque motor or the ports in the servo actuator.

These maintenance problems we have discussed give you some idea of the types of problems you may encounter and the method for correcting them. With the use of the test equipment, the isolation checks will become easier; but you still must have a thorough knowledge of the data flow in order to understand the nature of the malfunction.

SUMMARY.

The basic principles of flight presented in the first chapter apply to all aircraft, but special emphasis was placed on those principles peculiar to the F-102A. The second and third chapters dealt specifically with the F-102A control surfaces—the elevons and rudder—and their associated hydraulic and mechanical controlling components. The fourth chapter gave you the pitch and yaw damper systems and the way these systems affected the control surfaces. The fifth and last chapter presented you with the general principles of AFCS. Throughout this supplement, you learned the many maintenance and operating practices to follow for proper system operation.

As you know, the F-102A interceptor is a result of the combined efforts of many people. In one lifetime, it is doubtful whether any *one* of them could hope to understand completely all the detailed aspects of F-102A operation. But you (and they) can become familiar with a particular portion of the F-102A.

The material in this Flight Control System Supplement, as well as the information in the F-102A Flight Controls Maintenance Manual, T.O. 1F-102A-2-7, will help you to become thoroughly familiar with the operation and maintenance of the F-102A flight controls.

Always keep in mind that you can never obtain all the information you need or desire in one particular book concerning a system on this airplane. You must constantly review and study all available sources of system information. This reminder applies to all aircraft systems, but is particularly emphasized for the flight controls. The reason for this is the growth and development of the fire control systems that will eventually dominate the flight control and armament systems of all military aircraft.

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ALPHABETICAL INDEX

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F-102A

MAINTENANCE TRAINING SUPPLEMENT

(EXPERIMENTAL)

To Be Used in Conjunction With

T.O. 1F-102A-2-3

T.O. 1F-102A-2-6

LOW-PRESSURE PNEUMATIC SYSTEM

CONVAIR
F-102A

MAINTENANCE TRAINING SUPPLEMENT

LOW-PRESSURE PNEUMATIC SYSTEM

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE

AIR FORCE—AFPS—SAN ANTONIO, TEXAS—5-57-2600

1 JULY 1956

Foreword

J O Fletcher

The F-102A Training Supplements have been prepared by Convair, A Division of General Dynamics Corporation, to provide you, the system mechanic or maintenance technician, with basic information concerning the F-102A airplane and its operating systems. There are ten of these supplements, each covering an airplane operating system or major component. The text is direct and informal and is supplemented with illustrations and diagrams to aid you in understanding how the systems operate. The following is a list of the Training Supplements on the F-102A airplane:

Title

Flight Control System
Hydraulic System
High-Pressure Pneumatic System
Low-Pressure Pneumatic System
Airplane General
Airframe Fuel System
Power Plant Installation
Airframe Armament System
Electrical System
Instruments

Each supplement describes an operating system or major component, how and why it operates, and maintenance problems that you may encounter. The entire major system is further simplified by describing each of its subsystems separately. Thus, when you understand how all of the subsystems operate, you will be familiar with the functions of the major system. This knowledge of the airplane systems will enable you to use other operational, service, and maintenance instructions.

Since the purpose of these publications is to familiarize and acquaint you with the airplane systems and components, specific values, measurements, and tolerances are not given. Any values that appear in these supplements are approximate only and are used to emphasize more clearly a certain operation or condition. You must refer to your 1F-102A-2-3 and -2-6 Technical Orders and other pertinent handbooks for specific values when adjusting or checking a system and its components. These Technical Orders are revised periodically to include the latest maintenance procedures and data on the equipment installed in the F-102A.

The F-102A Maintenance Technical Orders consist of the following handbooks:

T.O. 1F-102A-2-1	**General Airplane**
T.O. 1F-102A-2-2	**Ground Handling, Servicing, and Airframe Group Maintenance**
T.O. 1F-102A-2-3	**Hydraulic and Pneumatic Power Systems**
T.O. 1F-102A-2-4	**Power Plant**
T.O. 1F-102A-2-5	**Fuel Supply System**
T.O. 1F-102A-2-6	**Air Conditioning, Pressurization, and Anti-Icing Systems**
T.O. 1F-102A-2-7	**Flight Control Systems**
T.O. 1F-102A-2-8	**Landing Gear**
T.O. 1F-102A-2-9	**Instruments**
T.O. 1F-102A-2-10	**Electrical Systems**
T.O. 1F-102A-2-11	**Radio-Communication and Navigation Systems**
T.O. 1F-102A-2-12	**Armament and Armament Electronics**
T.O. 1F-102A-2-13	**Wiring Data (F-102A)**
T.O. 1F-102A-2-13A	**Wiring Data (TF-102A)**

The Air Force is vitally interested in seeing that its equipment functions properly and is maintained as efficiently as possible. The Air Force, like a manufacturer of an automobile or commercial appliance, provides printed instructions for use with its equipment. The printed instructions you will use most frequently in maintaining the F-102A are the dash-2 Technical Orders listed above. They are available for your reference at the Maintenance Office on your base. You must refer to them frequently so that you will always have the latest maintenance infromation. The Training Supplements will help you to use these Technical Orders by answering many of the "hows" and "whys" that may puzzle you from time to time.

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Introduction

This supplement is arranged in six chapters and covers the F-102A Low-Pressure Pneumatic System. Chapter I contains a general introduction to pneumatic principles and their application in the F-102A airplane. The Low-Pressure Pneumatic System is briefly described to give you a general understanding of the system. Chapter II describes the sources and distribution of all low-pressure air used in the F-102A. In chapter III you learn about the cockpit air conditioning and pressurization system and how it is controlled. Chapter IV describes how low-pressure air cools and ventilates the various electronic compartments in the airplane. Chapter V contains a description of the subsystems that are pressurized with air from the low-pressure pneumatic system. In chapter VI you learn about the airplane anti-ice and defog systems. The automatic anti-ice control system and the various anti-icing systems—both hot air and electrical—are completely described. A summary concludes this supplement.

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Chapter I

PNEUMATIC PRINCIPLES AND THEIR APPLICATION TO THE F-102A

Contents *Page*

Modern high-speed airplanes, like the F-102A interceptor, use compressed air to perform many functions. You probably know that a functional compressed air system is called a *pneumatic system.* A high-pressure pneumatic system uses highly compressed air to perform work functions. A low-pressure pneumatic system uses lower pressure air to operate many utility-type subsystems. You will find that the F-102A airplane uses both of these systems—but keep in mind that they are distinct and separate systems.

The high-pressure pneumatic system uses high-pressure air that is stored in flasks—at about 3,000 psi maximum pressure—for the operation of the armament bay doors and armament displacement assemblies, the emergency extension of the landing gear, the deployment of the drag chute, and the braking action of the main wheels. This independent system is completely discussed in another Training Supplement of this series.

The low-pressure pneumatic system uses a combination of outside *ram air* and engine *bleed air* to air-condition, pressurize, ventilate, de-ice, and de-fog the airplane.

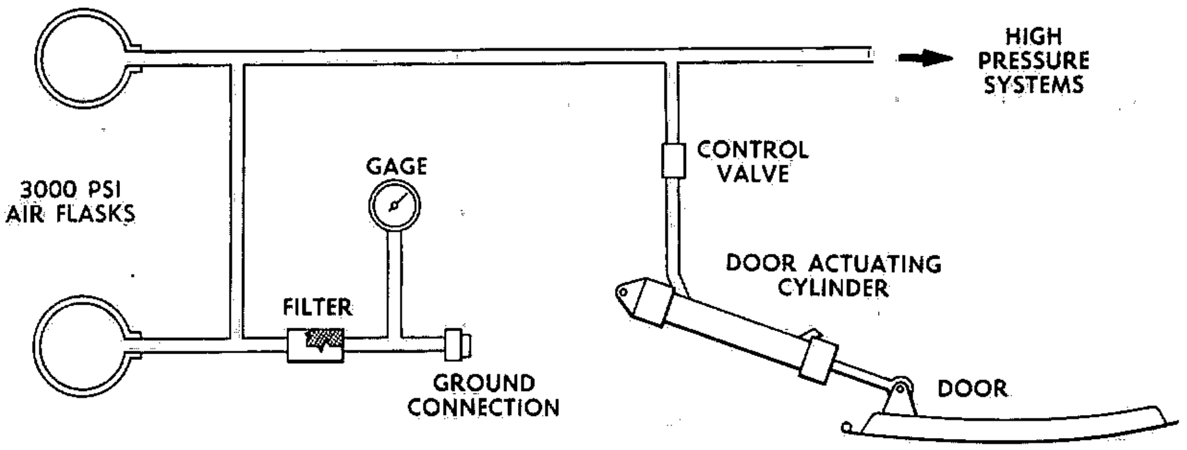
Figure 1-1 shows a simple high-pressure and a simple low-pressure pneumatic system. Note that the source of power for the high-pressure system is the air pressure stored in the air flasks. These flasks must be charged (filled with air) by a ground compressor while the aircraft is on the ground. The amount of work which the high-pressure system can accomplish is limited by the amount of air that the flasks can store.

As shown in the lower view of the illustration, the air for the low-pressure pneumatic system is taken from the atmosphere surrounding the airplane during flight. One source of air is the bleed point on the engine N_2 compressor section, while the other source is the boundary ram air intake. By using atmospheric air, the air supply for the low-pressure system is practically unlimited.

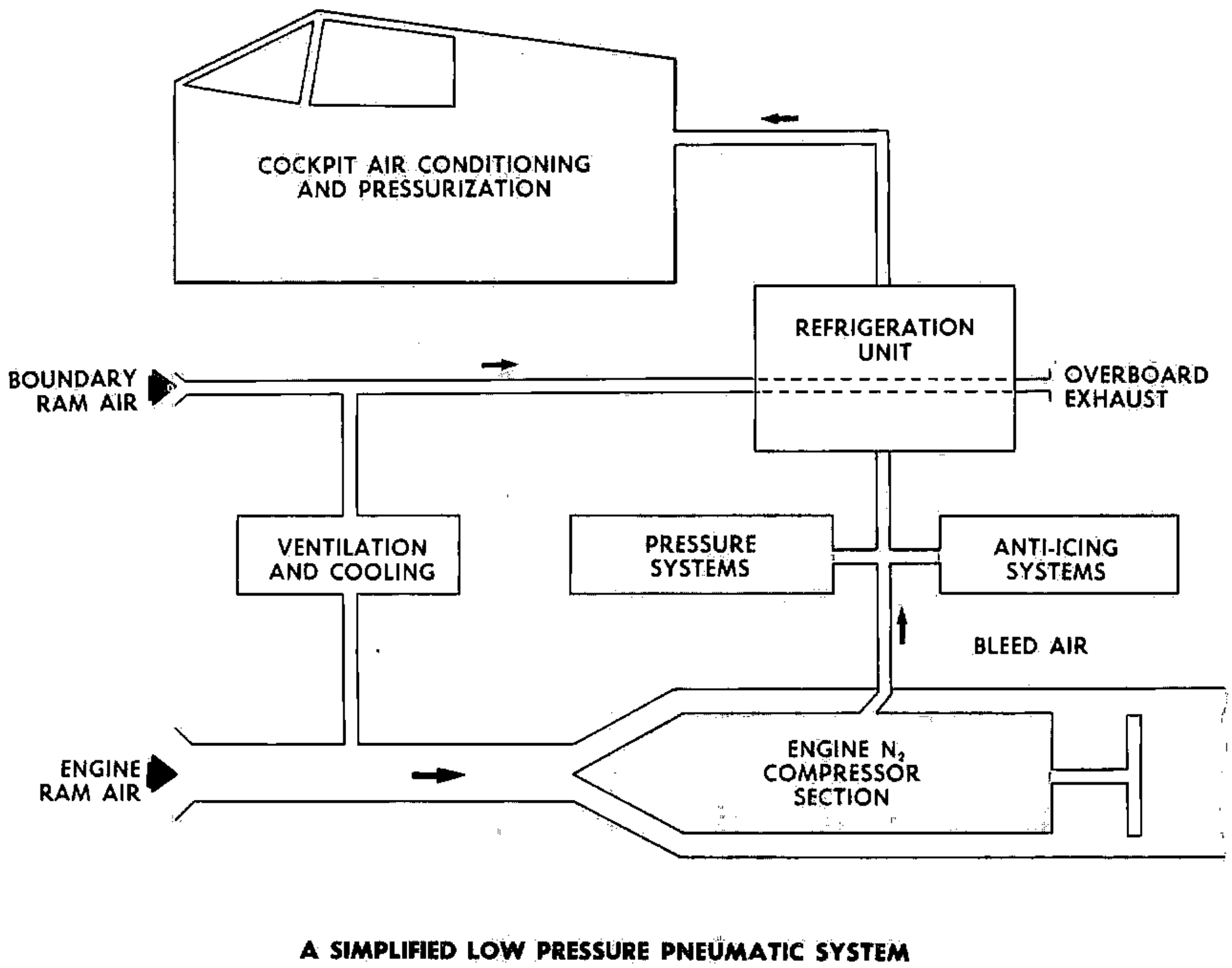
The success or failure of an F-102A intercept mission may often depend on the proper operation of the low-pressure pneumatic system. The icing of air intakes, the fogging of windshields, or the sudden loss of cockpit pressure could cause the failure of a mission and/or the loss of both the pilot and the airplane. The electronic equipment in the fire control and communication systems will function properly only when correctly ventilated. Because of the large electrical power losses involved in operating the electronic equipment, excessive amounts of heat are generated. If this heat were not dissipated, the electronic equipment would become overheated and would malfunction.

THE F-102A LOW-PRESSURE PNEUMATIC SYSTEM.

As you will note in figure 1-2, the F-102A low-pressure pneumatic system air-conditions and pressurizes the cockpit and ventilates and cools the electronic compartments, the IFF unit, and the aircraft generators. This system also furnishes compressed air to pressurize the fuel tanks, hydraulic reservoirs, elevon artificial-feel system, canopy seal, pilot's G-suit, and the glycol



A SIMPLIFIED HIGH PRESSURE PNEUMATIC SYSTEM



A SIMPLIFIED LOW PRESSURE PNEUMATIC SYSTEM

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Figure 1-1. High-Pressure Versus Low-Pressure Pneumatic Systems

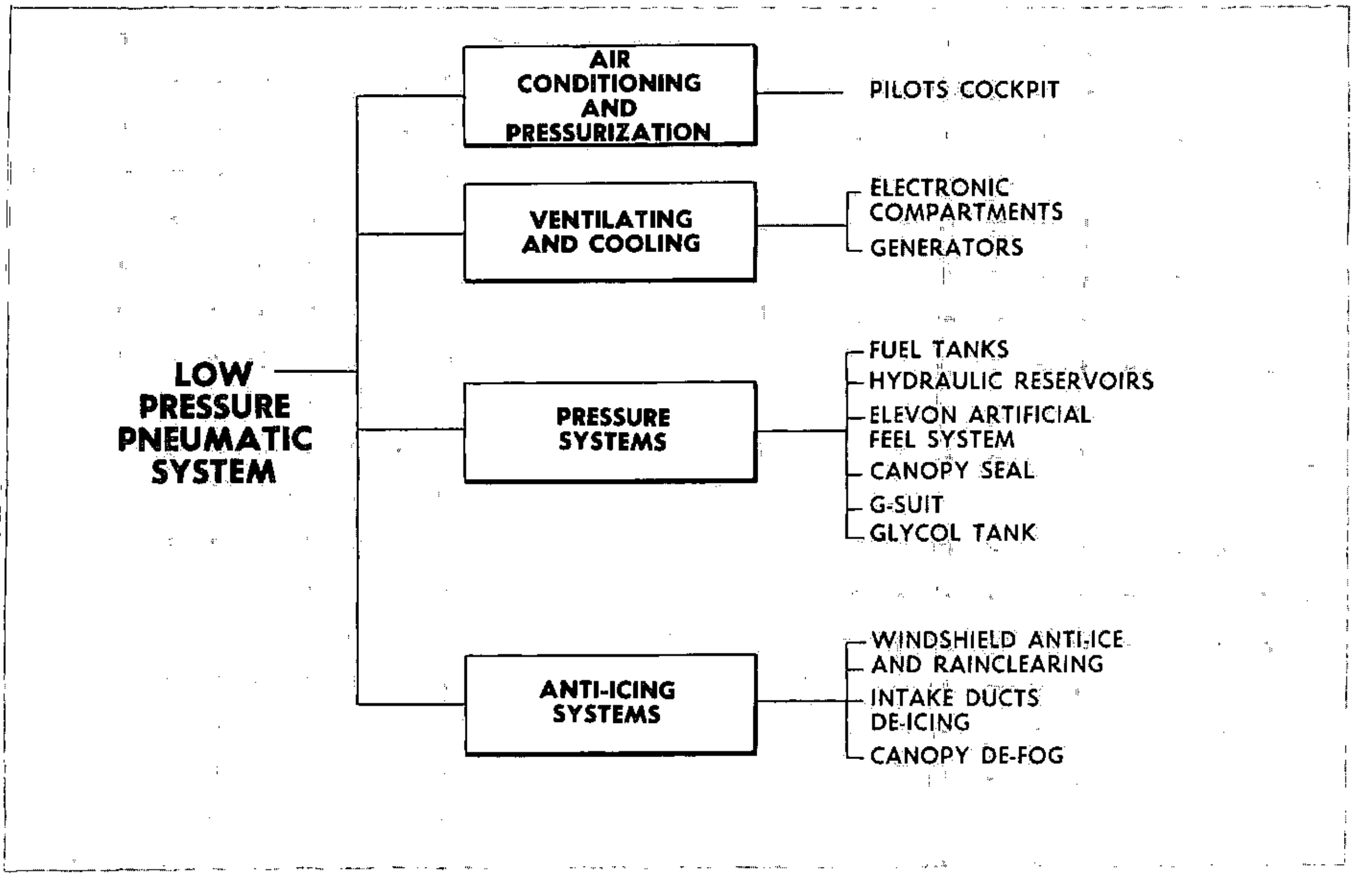

Figure 1-2. Functions of the Low-Pressure Pneumatic Systems

tank of the anti-icing system. The system also provides hot air to anti-ice the engine ram air intake ducts, to rainclear and anti-ice the pilot's left windshield, and to de-fog the canopy. Notice the many functions this system performs—it could easily be called a "jack of all trades."

AIR SOURCES FOR THE LOW-PRESSURE PNEUMATIC SYSTEM.

Figure 1-3 shows that the air used in the low-pressure system enters the boundary-layer air intake ducts or the engine air intake ducts as *ram air*. Part of this ram air is used in its untreated state for cooling and ventilating purposes. Most of the ram air, however, is drawn into the engine compressor where it is compressed and diffused before it enters the engine combustion chamber. A small part of the engine air is *bled* from the second, or N_2, stage of the engine compressor. Some of this hot, compressed *bleed air* is used in the anti-ice system and some is used to pressurize the hydraulic reservoirs and the elevon artificial-feel system. Most of this bleed air passes through a refrigeration unit where it is *conditioned*. Conditioned air from the refrigeration unit is used to air-condition and pressurize the cockpit, while partially conditioned air is used to pressurize the canopy seal and other airplane components. As shown in figure 1-3, the low-pressure pneumatic system consists of a combination of untreated ram air, unconditioned bleed air, partially-conditioned bleed air, and conditioned bleed air.

A thumbnail description of each use of the low-pressure pneumatic system is given in the following text. These short descriptions will give you a general concept of the overall low-pressure pneumatic system. In the following chapters of this Training Supplement, each application of the low-pressure system will be described in detail.

COCKPIT AIR CONDITIONING AND PRESSURIZATION.

As shown in figure 1-4, the cockpit of the F-102A airplane is both air-conditioned and pressurized by air bled at a high temperature and pressure from the engine compressor. Note that the bleed air is *conditioned* before it enters the cockpit by being routed through a refrigeration unit. The refrigeration unit consists of an air-to-air heat exchanger and an expansion turbine.

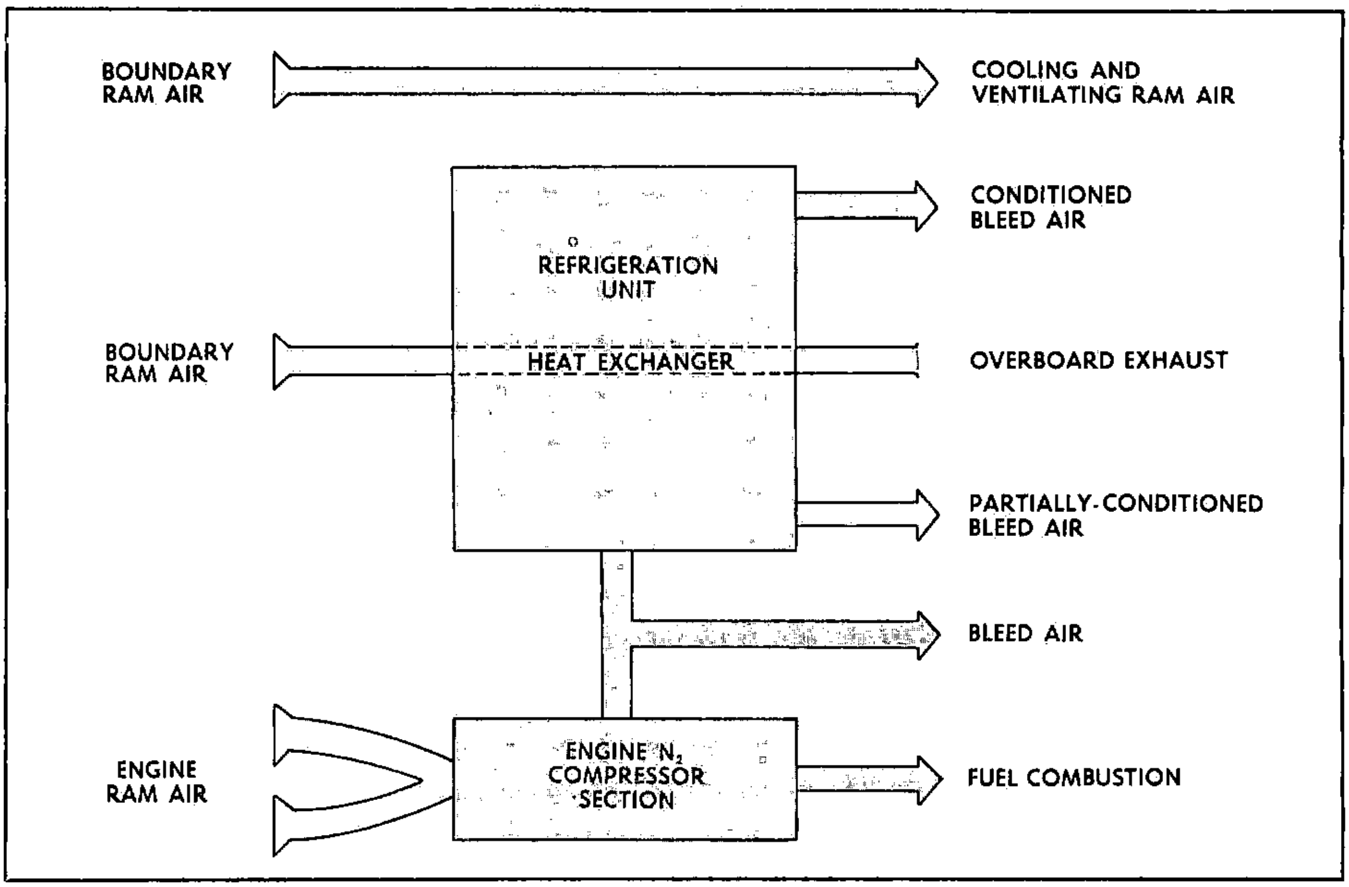

Figure 1-3. Air Sources for the Low-Pressure Pneumatic Systems

Note that ram air from the right boundary-layer air intake duct passes through the refrigeration unit heat exchanger. This air acts as a cooling agent for the bleed air. After passing through the refrigeration unit, the conditioned air mixes with unconditioned bleed air which has bypassed the refrigeration unit. It then enters the cockpit through *piccolo* tubes. The proportion of unconditioned, or *by-pass,* bleed air to conditioned bleed air determines the temperature of the air entering the cockpit. Note that this conditioned air flows from the cockpit, through a pressure regulator, to the forward electronic compartment.

An automatic temperature control system regulates the cockpit temperature by controlling the opening of the bypass valve. The amount of unconditioned air (hot air) is controlled by varying the bypass valve position. A temperature control switch permits the pilot to select the desired operating temperature.

The air-conditioning system will operate on the ground with the engine running; however, the conditioned air from the refrigeration unit will have a higher temperature than during flight. This is because little or no bleed air is mixed with the conditioned air and the total flow of air to the cockpit is therefore less than it would be during in-flight operations. Air pressure in the cockpit is kept tolerant during flight by means of a pressure regulator that controls the flow of air from the cockpit. When flying below 10,000 feet, the flow of air from the cockpit is unrestricted; therefore, the cockpit is unpressurized, and the air pressure in the cockpit is the same as that of the altitude at which the airplane is flying. (This is often referred to as "cockpit altitude and airplane altitude being the same.")

When flying between 10,000 and 26,000 feet, the pressure in the cockpit is maintained at the 10,000-foot level. Thus, the cockpit altitude is 10,000 feet and the airplane altitude may be 25,000 feet. Above the 26,500-foot altitude level, the cockpit pressure (or altitude) is regulated to about 5 psi above atmospheric pressure.

The chart (figure 1-5) shows the level of pressurization maintained at any altitude. For instance, at the 40,000-foot altitude level (step 1 on the chart), the cockpit pressure maintained equals the atmospheric pressure at an altitude of 17,000 feet (step 3 on the chart). The 5 psi differential is represented by step 2. When the difference between atmospheric and cockpit pressure exceeds 5 psi, an air safety valve relieves the cockpit pressure by venting air overboard.

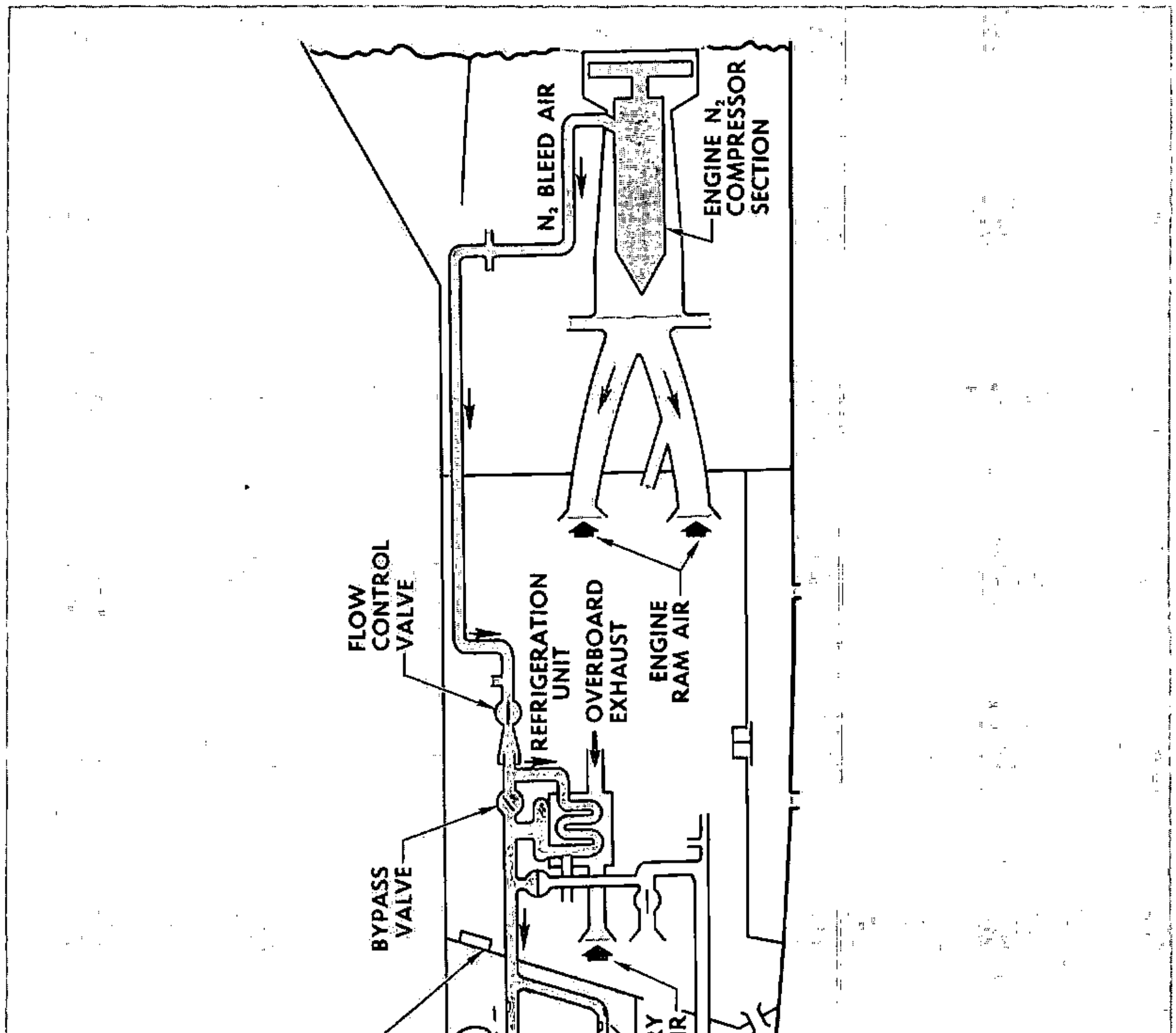

Figure 1-4. Cockpit Air Conditioning, In-Flight

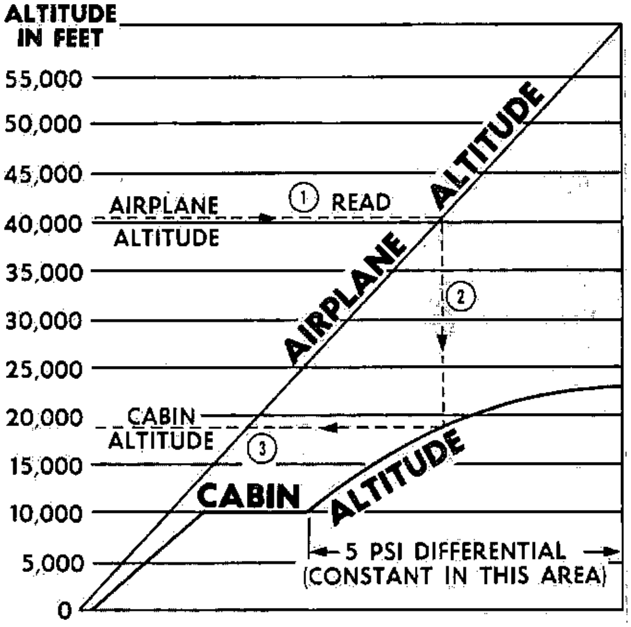

Figure 1-5. Cockpit Pressurization Schedule

In the event the air-conditioning and pressurization system becomes inoperative, the pilot can shut off the bleed air and turn on the ram air to ventilate the cockpit. The flow control valve (see figure 1-6) is closed to shut off the bleed air, the ram air valve is open to permit air from the left boundary-layer air intake duct to enter the cockpit, and the air safety valve is open to depressurize the cockpit. When the control switch is OFF, all air to the cockpit is cut off and the safety valve remains closed.

Whenever the bleed air is shut off from the cockpit, the cockpit pressure slowly decreases because of airframe leakage. A *rate sensor* unit controls the rate of bleed air flow to prevent pressure surges in the cockpit when the bleed air shutoff valve is opened. This control unit is described in detail in Chapter III of this Training Supplement.

VENTILATION SYSTEM, IN-FLIGHT OPERATION.

The various electronic compartments in the forward part of the airplane are ventilated mainly by ram air from the left boundary-layer air intake. In the figure 1-7, note that the ram air enters the intake and passes through a pressure regulator and temperature controlled shutoff valve before it is distributed through the ducts to the various parts of the electronic compartments. The pressure regulator component of the shutoff valve prevents the air pressure in the ducts from exceeding a set pressure. When the intake air becomes too hot, the temperature controlled shutoff valve stops the flow of ram air. The conditioned air discharged from the cockpit provides additional ventilation for the forward electronics compartment. The ventilating air from the forward electronic compartment and the intermediate electronic compartment is exhausted into the armament bay through a pressure relief valve and overboard through controlled leakage.

When the shutoff valve closes and stops the flow of ram air to the ventilating system, the discharge air from the cockpit provides the necessary emergency cooling for the forward electronic compartment and the intermediate and upper electronic compartments.

The aft electronic compartment, IFF compartment, and the generators are ventilated by ram air drawn from the engine air inlet ducts. See figure 1-8. Those compartments are not pressurized, and the air has an unrestricted exhaust from each compartment.

To prevent highly corrosive rocket exhaust fumes from damaging the electronic equipment, the flow of the ventilating system ram air is automatically cut off just before the rockets or missiles are fired. Air flow is automatically restored several seconds after the firing cycle is completed.

VENTILATION SYSTEM, GROUND OPERATION.

When the engine is *not* running, there are no provisions for ventilating the aft electronic and IFF compartments on the ground. However, the forward, intermediate, and upper electronic compartments can be ventilated by a ground air-conditioning unit that is connected to the forward electronic compartment through the nose wheel well. As you can see in figure 1-9, air from the ground air-conditioning unit flows into the forward electronic compartment. From there, note that the air flows to the intermediate and upper electronic compartments through the ram air distribution ducts. Note also that three exhaust outlets prevent the buildup of pressure in the cockpit and compartments. The conditioned air also flows from the forward electronic compartment to the cockpit. The pressure opens the cockpit pressure regulator. Thus, cockpit ventilation is provided on the ground.

When the airplane is on the ground and the engine is idling, the air pressure in the engine air intake ducts creates a negative pressure, or partial vacuum, in the aft electronic compartment, the IFF unit, and the generators. Following figure 1-10, you can see that this continuously draws outside air into the

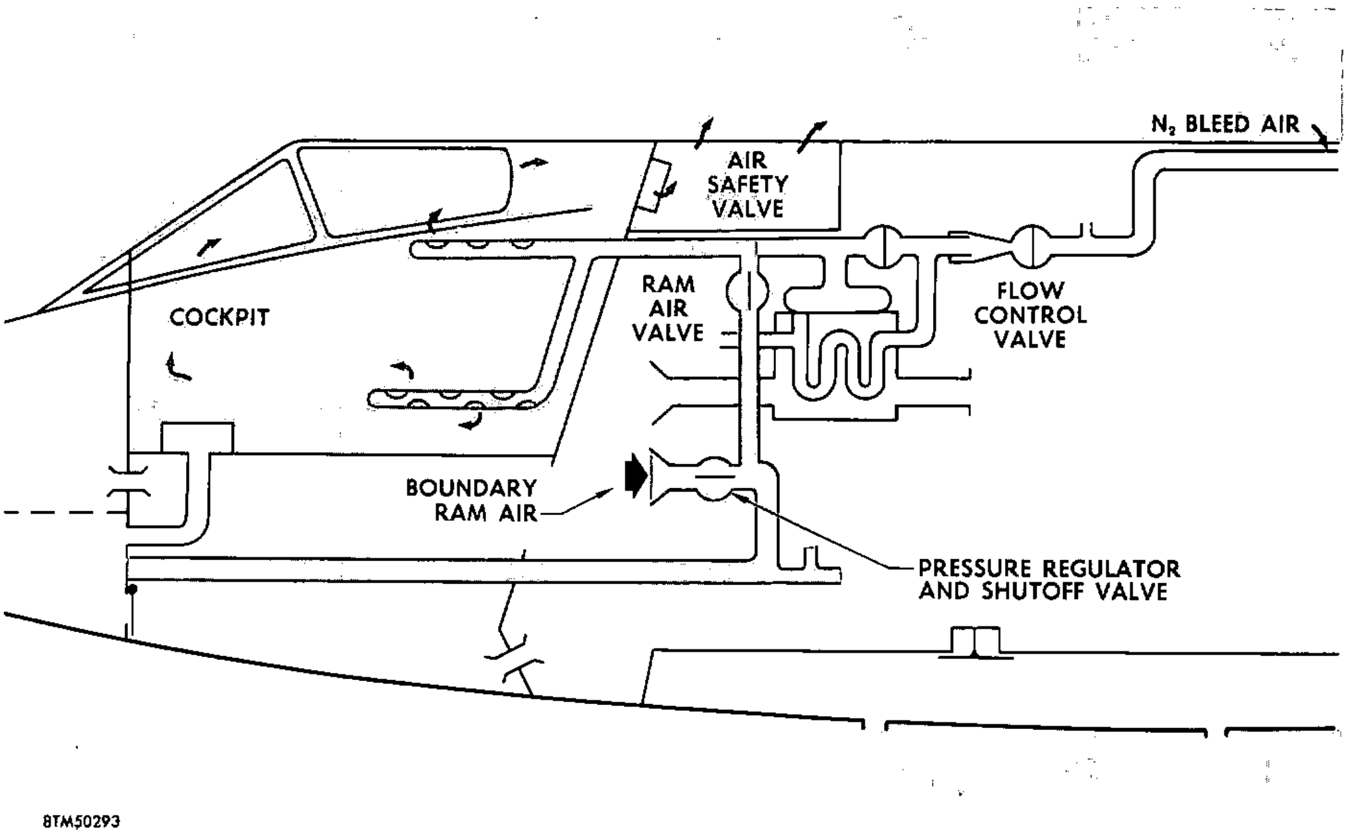

Figure 1-6. Cockpit Ventilation by Ram Air, In-Flight

generators through a flapper door and into the aft electronic compartment and IFF unit through vents into the fuselage. Notice an additional continuous flow of cooling air created by the engine intake. The engine draws air through the intake and check valve from the intermediate and upper electronic compartments, and the air drawn from the compartments is replaced by air entering through the nose wheel well.

On the ground, with the engine idling, the forward, intermediate, and upper electronic compartments are cooled by means of a *jet pump*. Figure 1-11 illustrates the general layout of the ventilating system. When you operate the system under these conditions, you can see that the jet pump valve opens and engine bleed air flows forward and out of the airplane through the left boundary-layer intake, thus reversing the normal flow of air in the duct.

This reverse air flow causes a partial vacuum which draws air from the forward, intermediate, and upper electronic compartments. The compartment air is replaced by outside air entering through the nose wheel well. If the airplane skin near the jet pump nozzle becomes too hot, the jet pump valve will automatically close completely.

When the control switch is in PRESSURE, the cockpit air-conditioning system operates on the ground. This permits conditioned air from the cockpit to enter the forward electronic compartment. This air, combined with the air from the jet pump, greatly improves the effectiveness of the ventilation system. When the cockpit canopy is open, the cockpit pressure regulator will not open and the air cannot enter the compartment.

PRESSURIZATION OF OPERATING SUBSYSTEMS.

Certain airplane components or systems are pressurized with bleed air from the low-pressure pneumatic system. The items concerned are: the canopy seal, the glycol tank used in the radome anti-ice system, the fuel tanks, the pilot's G-suit, the hydraulic fluid reservoirs, and the elevon artificial-feel system. As shown in figure 1-12, the hydraulic reservoirs and the artificial-feel system are pressurized with unconditioned bleed air whenever the engine is running.

The other four subsystems use partially conditioned bleed air from the heat exchanger. This partially conditioned air is cut off whenever the flow of bleed air to

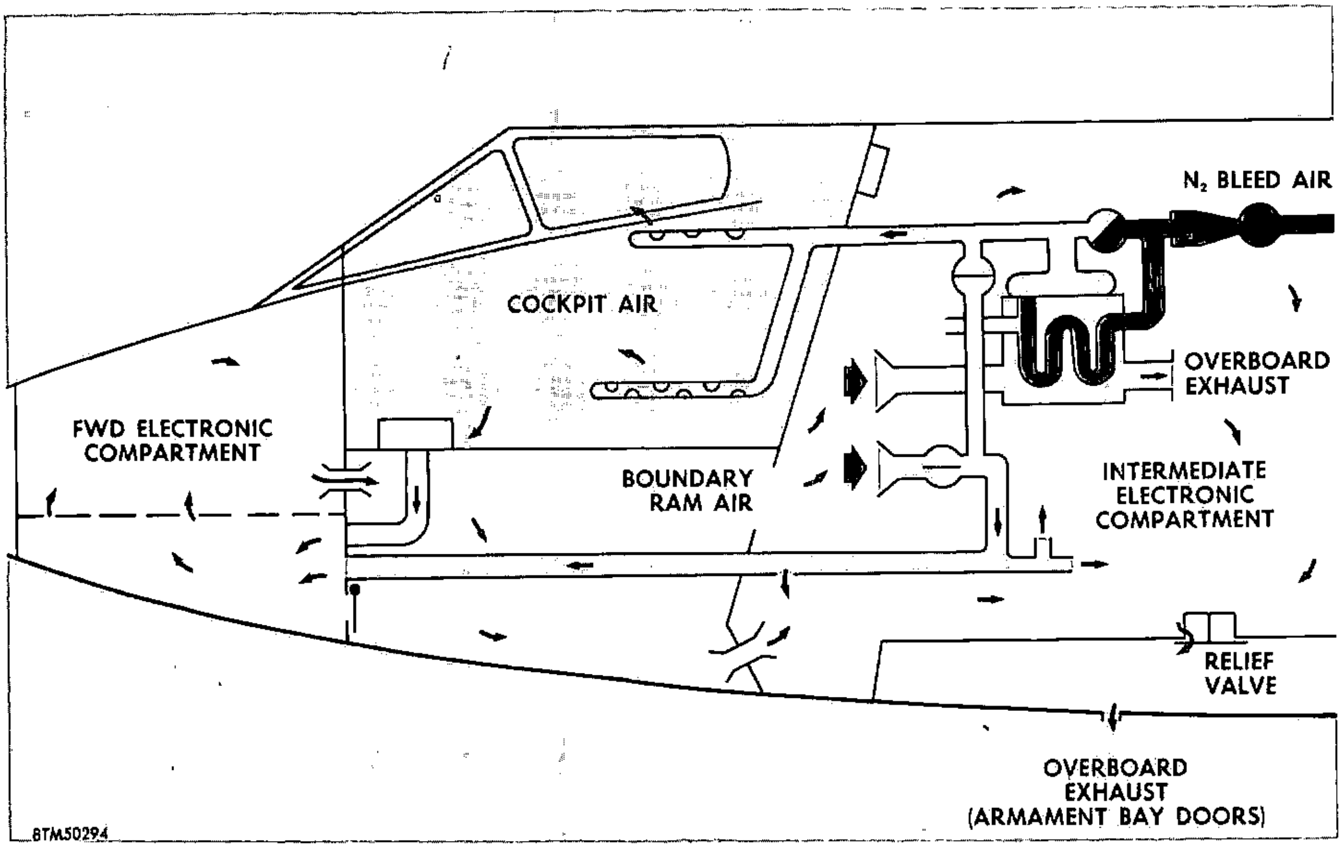

Figure 1-7. Ventilation of Electronic Compartments, In-Flight, Forward Portion.

the heat exchanger is halted—when armament is being fired or when ram air is being used for cockpit ventilation. All of these subsystems or components are briefly discussed below. You will find a more complete discussion in Chapter V of this Training Supplement.

The canopy is sealed to prevent the loss of cockpit pressurization and to prevent the entry of rain. Part of the seal system consists of a hollow rubber tube installed in a channel around the lower part of the canopy. Partially conditioned bleed air pressurizes this canopy seal tube whenever the canopy is closed and latched.

The fuel tanks of the F-102A airplane are pressurized to insure a continuous flow of fuel to the engine. Two separate sources of compressed air are used. In figure 1-12, you can see that one source is partially conditioned bleed air from the refrigeration unit heat exchanger. The second source is unconditioned air bled from the N_1 stage of the engine compressor. This second source is necessary because the flow of air from the heat exchanger is occasionally shut off. Fuel tank pressurization is completely discussed in another Training Supplement of this series which covers the F-102A Airframe Fuel System.

The pilot of the F-102A airplane always wears a G-suit to prevent his blacking out during sudden accelerations or changes in direction. Partially conditioned bleed air, from the refrigeration unit heat exchanger, automatically pressurizes the suit during these maneuvers.

Each of the two reservoirs in the hydraulic system is pressurized with unconditioned bleed air. This pressure is controlled and regulated by regulators in the hydraulic system. Pressurization of the reservoirs helps maintain a positive fluid flow to the hydraulic pumps. The hydraulic system is completely discussed in another Training Supplement of this series.

The elevon artificial-feel system resists movement of the elevon controls with a force that varies with airplane speed and altitude. Unconditioned bleed air from the N_2 section of the engine is fed into an air regulator that controls this system. This provides the pilot with the "feel" of the elevon movement that is lost through the use of a hydraulic power system.

ANTI-ICING SYSTEMS.

The F-102A airplane contains a complete anti-ice and de-fogging system that uses engine bleed air, glycol fluid, and electrical current. An automatic ice detec-

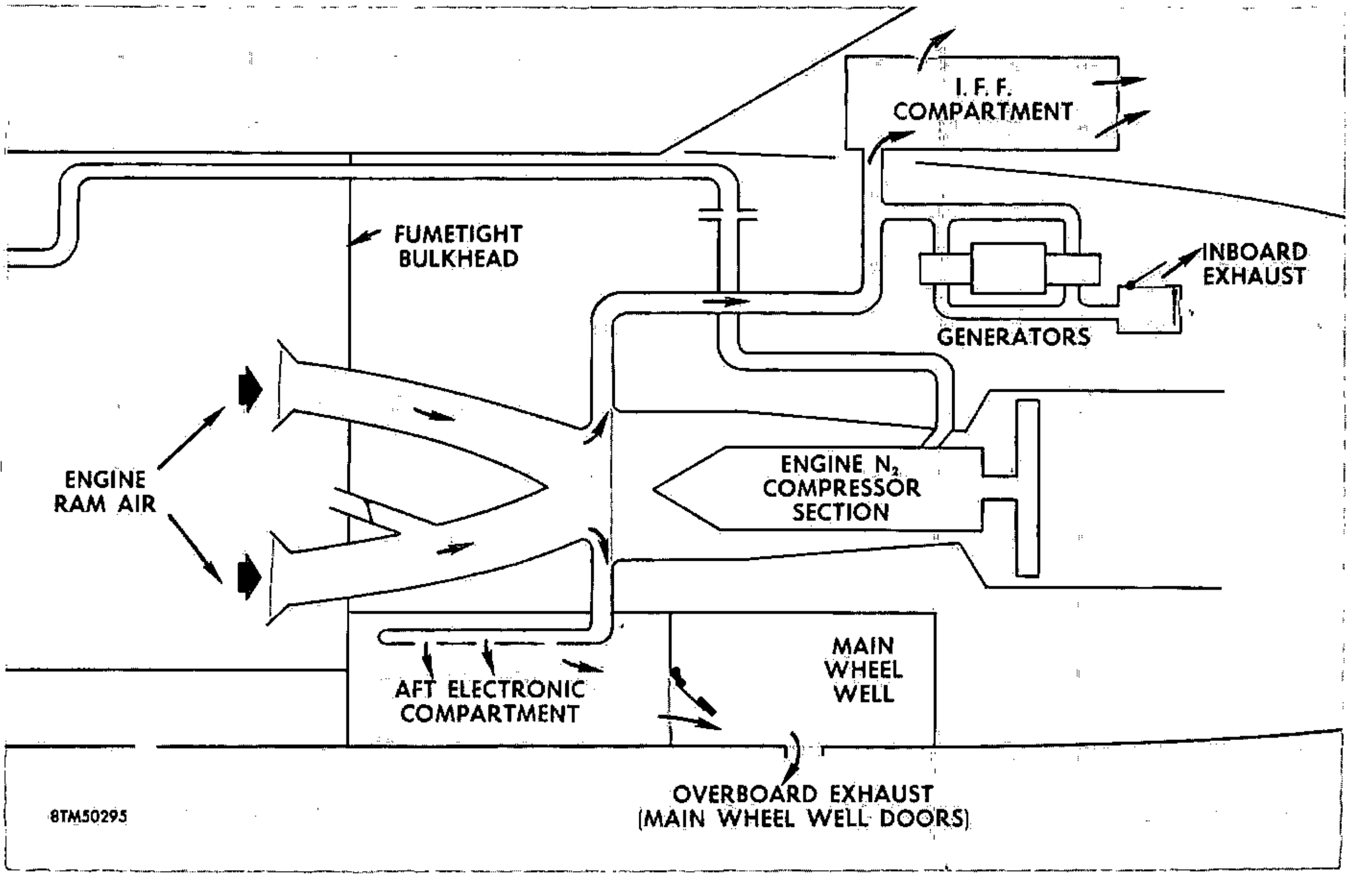

Figure 1-8. Ventilation of Electronic Compartments, In-Flight, Aft Portion

tion system controls parts of the system while other parts are controlled manually by the pilot. When the engine is not running, the entire anti-ice system is shut off on the ground with the exception of certain pilot controlled systems. On the ground, the radome anti-ice system is always shut off.

The ice detection system—consisting of an ice detector probe and a system of electrical switches and relays—controls the flow of bleed air that anti-ices the engine air intake and the forward edges of the ram air intake ducts. Figure 1-13 shows the parts of the anti-icing system that are supplied with hot air from the low-pressure pneumatic system. The ice detection system also controls the flow of glycol fluid that anti-ices the radome and controls the electrical current that de-ices the "Q" pressure intake assemblies (artificial feel system pressure intake) on the rudder. The windshield rainclearing and anti-icing, pitot-static tube anti-icing, windshield defogging, canopy defogging, and pilot's oxygen mask defogging systems all have individual controls operated at the pilot's discretion.

Hot Air Anti-Icing Systems.

The forward edges of the engine ram air intake ducts are of double-skin construction and have small slots along the inboard edges of the ducts. When the ice detector probe detects ice formation in the engine intake, the automatic control system operates and routes hot bleed air to the hollow forward edges of the intake ducts. The hot air melts the detected ice, and this air is then exhausted overboard through the slots. If the airplane skin becomes too hot, themostatic switches in the control system stop the flow of bleed air through the forward edge of the intake ducts.

The windshield rainclearing and anti-icing system is controlled by a switch in the cockpit. When the switch is ON, a flow of bleed air spreads out over the external surface of the left windshield, forming a screen of warm air—to prevent rain from reaching the windshield. This, combined with the defogging system that electrically heats the windshield, prevents the formation of ice on the windshield.

The canopy panels of the F-102A airplane are defogged by a hot-air system manually controlled by the pilot. When the control switch is turned ON, partially conditioned air from the refrigeration unit heat exchanger enters the defog ducting and creates a negative pressure, or partial vacuum, behind it. You can follow this action in figure 1-14. Warm air from the cockpit enters the end of the duct to fill the

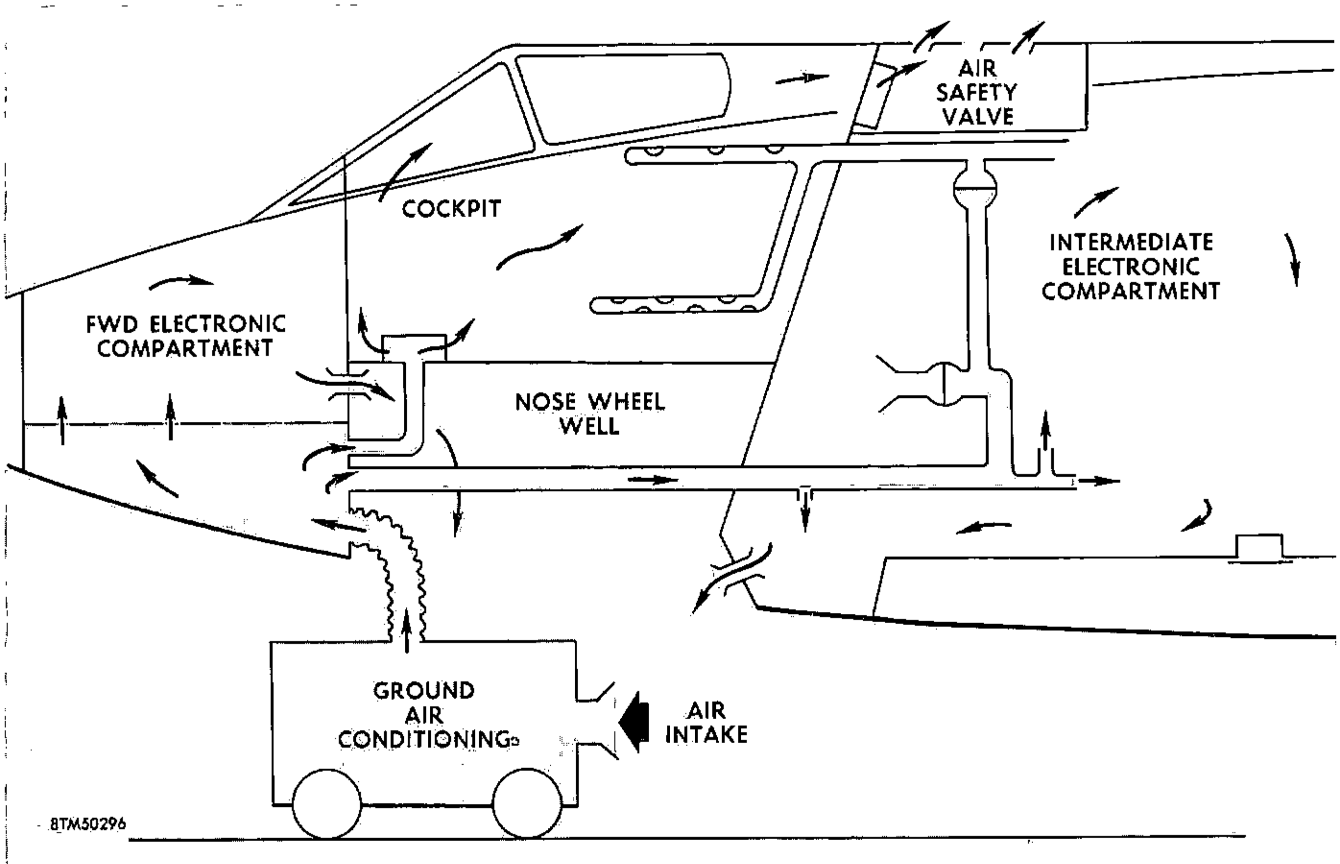

Figure 1-9. Ground Ventilation, Engine Not Operating

vacuum, and is sucked through the ducting and out through nozzles at the forward end of the canopy. This defogs the inner side of the canopy glass.

The engine anti-icing system, also called a de-icing system, is a component part of the engine and is completely discussed in another Training Supplement in this series which covers the Power Plant Installation.

Electrical Anti-Icing Systems.

The two windshield "Nesa" glass panels are defogged by electrical heating elements imbedded in the glass. See figure 1-15. The system is controlled by a toggle switch in the cockpit. Preset temperature sensing elements, in the glass, maintan the temperature of the glass by varying the flow of electric current in the heating elements.

The "Q" pressure intake assembly consists of two pitot-type tubes that pick up ram air pressure and feed it to the flight control feel systems. The feel systems supply the pilot with an *artificial feel,* or resistance to control movement, that varies with the "Q" pressure. Electrical heating elements prevent the formation of ice which would block the air intake. In the illustration, note that the instrument pitot-static boom and tube project forward from the nose of the airplane. Electrical heating elements prevent the formation of ice that might block the pitot and static air intakes. The flow of current to the heating elements is controlled by a switch in the cockpit.

The pilot's oxygen mask is defogged electrically. Note that the system consists of a control switch, a power source, a variable resistor to control the heat, and a quick-disconnect lead to the mask harness.

Fluid Anti-Icing System

The radome is anti-iced by glycol contained in a pressurized tank. The radome anti-icing shutoff valve is controlled automatically. When it is open, glycol is forced overboard, through a porous ring, to form an anti-ice film on the exterior of the radome.

PNEUMATIC PRINCIPLES.

There are certain fundamental principles that you should keep in mind when working with the low-pressure pneumatic system. These principles deal with the behavior of air under different conditions, and are discussed in the following paragraphs.

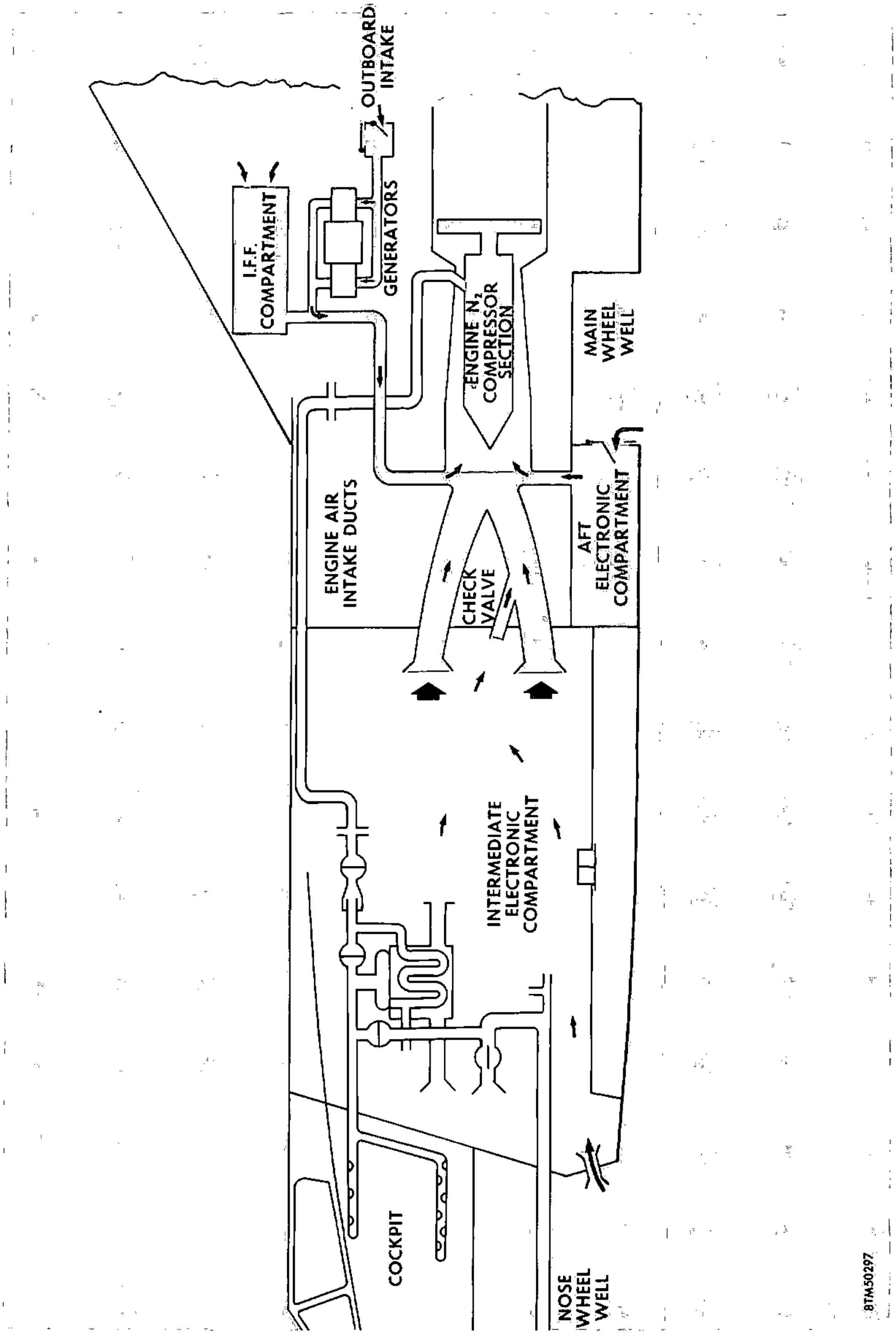

Figure 1-10. Ground Ventilation Aft Compartments, Engine Idling

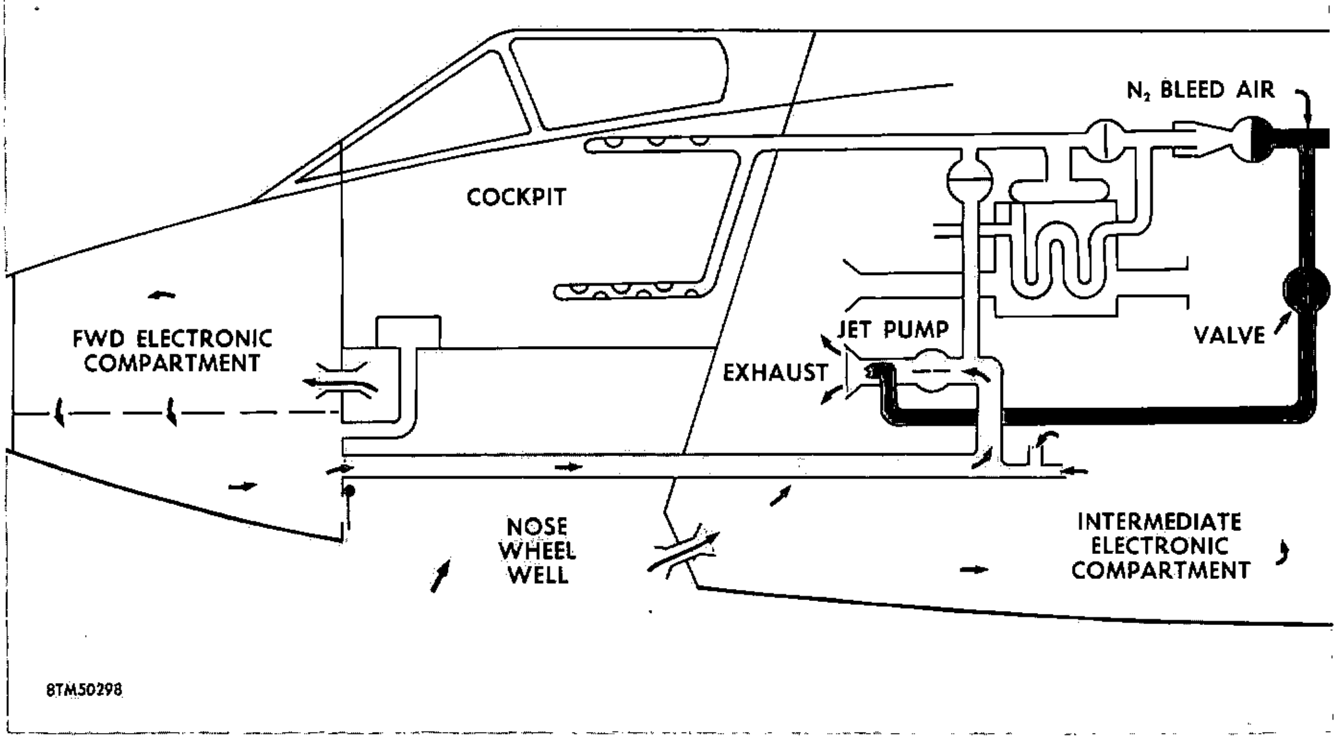

Figure 1-11. Ground Ventilation, Forward Compartments, Using Jet Pump, Engine Idling

ATMOSPHERE AND ATMOSPHERIC PRESSURE.

The atmosphere consists of a thick blanket of air surrounding the earth. Air is composed of tiny molecules of different gases, mostly nitrogen and oxygen. These molecules are very small, so small that you cannot see one even with the aid of the most powerful optical microscope. Actually, since the distance between molecules is very great in comparison to their size, air is mostly empty space. These molecules are continually in motion, colliding with each other and with surrounding objects. The combined force of all these collisions gives the effect of atmospheric pressure. Obviously, the fewer molecules of air in a given space, the fewer collisions there will be. The fewer collisions there are, the less the pressure. Study the illustration (figure 1-16) and you will understand this statement. Air pressure at sea level is relatively great. The weight of the air above presses on the air below, forcing more molecules into a given space. This increase in number of molecules causes an increase in the number of collisions and therefore an increase in pressure. At higher altitudes the molecules are farther apart, therefore the pressure is lower.

You have probably seen a mercury barometer. It is a device that measures atmospheric pressure. When a unit of air is raised higher above the earth, the column of air above becomes shorter. Study the example in figure 1-17. The cubic foot of air, shown at sea level, is compressed by the weight of the column of air extending above it to the upper limit of the atmosphere. This atmospheric pressure supports the column of mercury, shown in the tube, at a level where the weight of the column is equal to the weight of the column of air. This is 29.92 inches mercury (hg) at sea level. A change in atmospheric pressure causes a corresponding change in the height of the column of mercury.

WHAT HAPPENS WHEN AIR IS COMPRESSED.

We have said that air consists of tiny molecules of various gases, mostly nitrogen and oxygen. Actually, under normal atmospheric pressures and temperatures, air is primarily empty space with molecules speeding around in random directions. The distance between the molecules is great, in comparison to their size. When pressure is increased, the molecules are pressed closer together but are still widely separated.

Since these molecules are in continuous motion, they are constantly colliding. The collision of one molecule with another, or with the sides of a container, causes friction or heat. When air is compressed, the

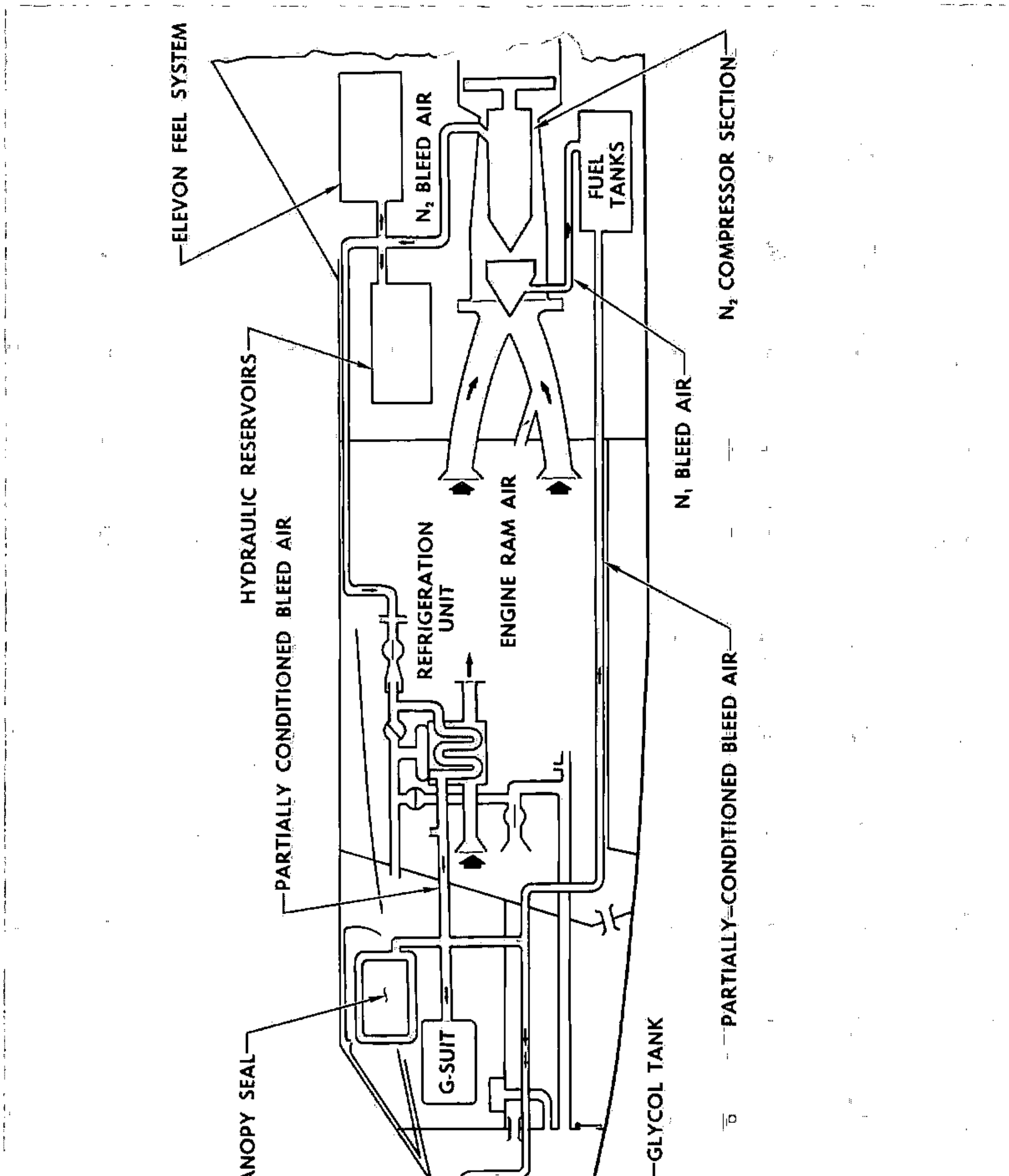

Figure 1-12. Pressurization of the Subsystems

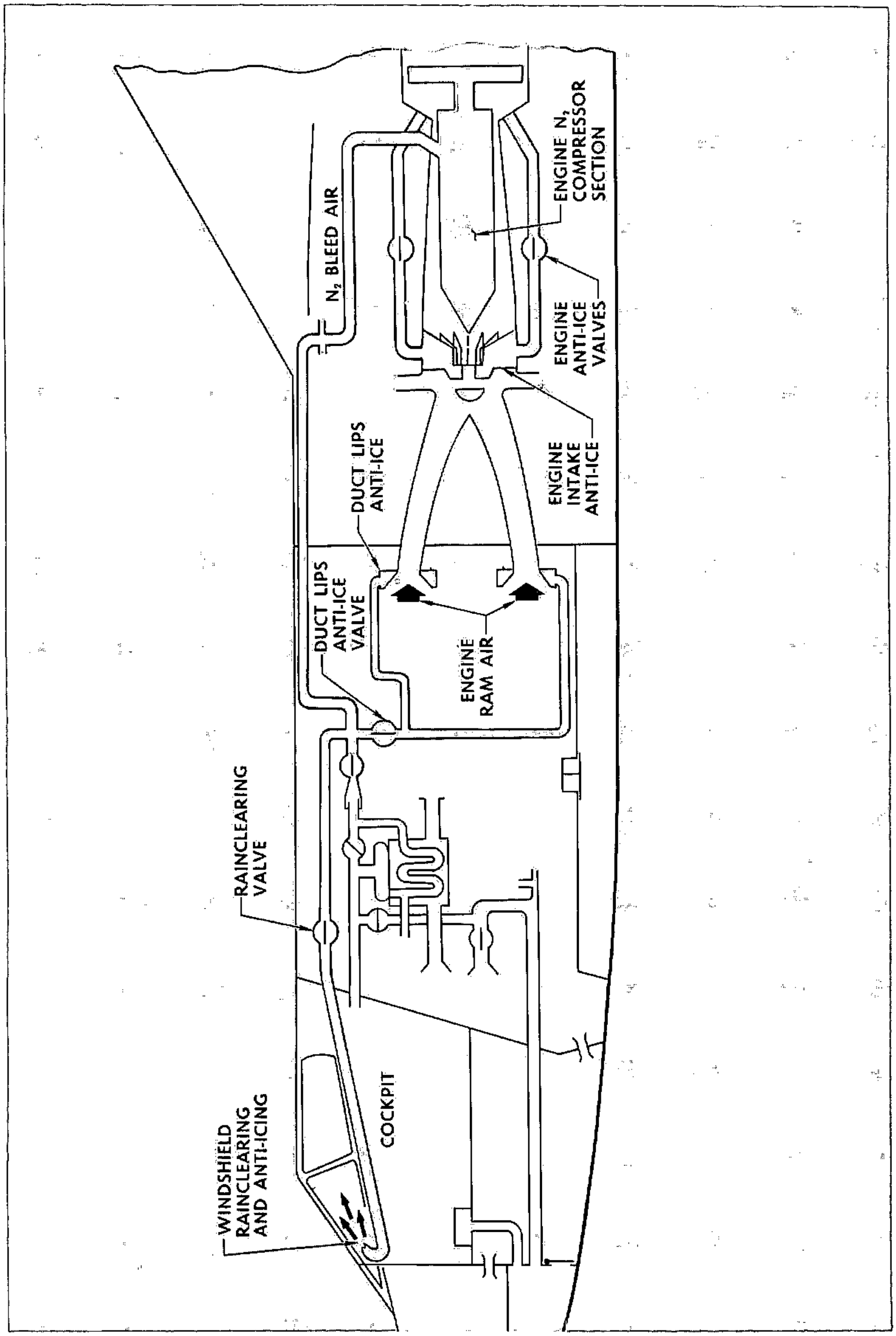

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Figure 1-13. Anti-Icing with Bleed Air

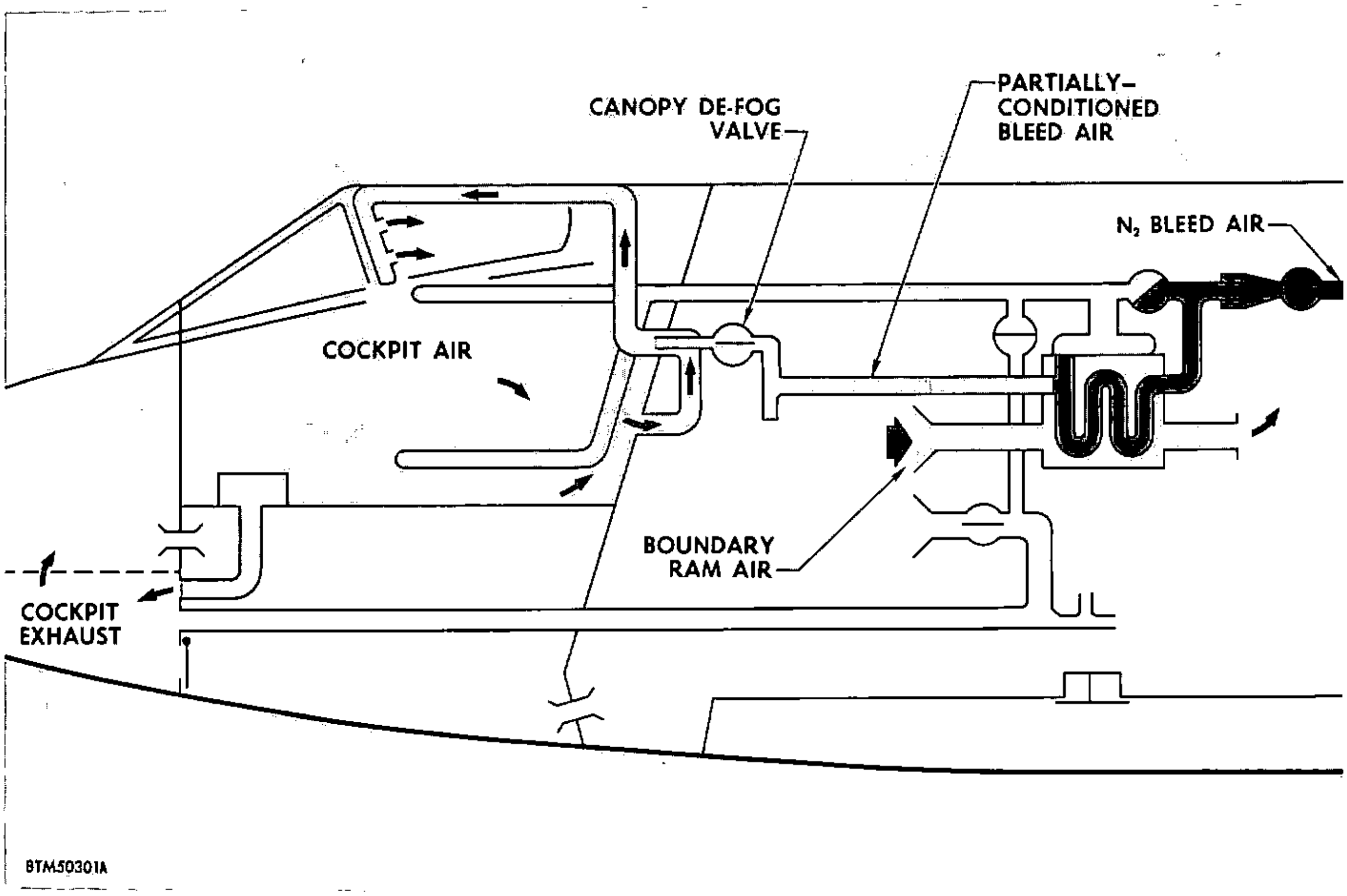


Figure 1-14. Canopy De-Fogging System

collision rate increases; therefore the temperature rises. The greater the compression, the higher the temperature. You can see this relationship in figure 1-18. Notice the piston before it is compressed. The molecules are far apart and the temperature bulb and the pressure gage have low readings. When the piston compresses the air, the molecules move together and the temperature and the pressure readings increase. This relationship works in the reverse direction, too. If the pressure drops, the temperature will drop.

This *heat of compression* may result in a very great temperature rise, the exact temperature being determined by many factors. For instance, if air is compressed to one-tenth of its original volume, the temperature may increase from an original 15°C (59°F) to 343°C (650°F). The heat of compression in the engine of the F-102A causes the temperature of this compressed air to rise as high as 427°C (800°F).

WHAT HAPPENS WHEN AIR IS HEATED.

We have seen that the air temperature will rise when air is compressed. Now we will consider what happens when air is heated. When free air is heated, the molecules travel faster. Because of the increased speed, the space between the molecules will increase. In other words, the air will expand. This is shown in figure 1-19. Notice that there are fewer molecules within a given space and thus there will be fewer collisions, but because of their increased speed they will have a greater force. This greater force will balance the reduction in the number of collisions. Therefore, the pressure will not change.

When the air in a closed container is heated, the air cannot expand and the greater force of each collision results in an increase in pressure. The reverse of the relationship holds true; that is, if the temperature of air in a closed container drops, the pressure also drops.

WHY AIRPLANES ARE PRESSURIZED.

Remembering that the molecules in air are evenly distributed, you can understand that atmospheric pressure (under the same conditions and altitude) will be equal at any point of contact. Thus, the air pressure on the human body is equal at all points and we are not aware of this pressure at any one point. We are, however, aware of changes in pressure—especially sudden changes like those felt when flying in a rapidly climbing airplane. This sudden drop in pressure wouldn't give the body time to adjust and the gases trapped in the body would therefore expand and cause much discomfort. It could even cause death. This is even more noticeable when the airplane is

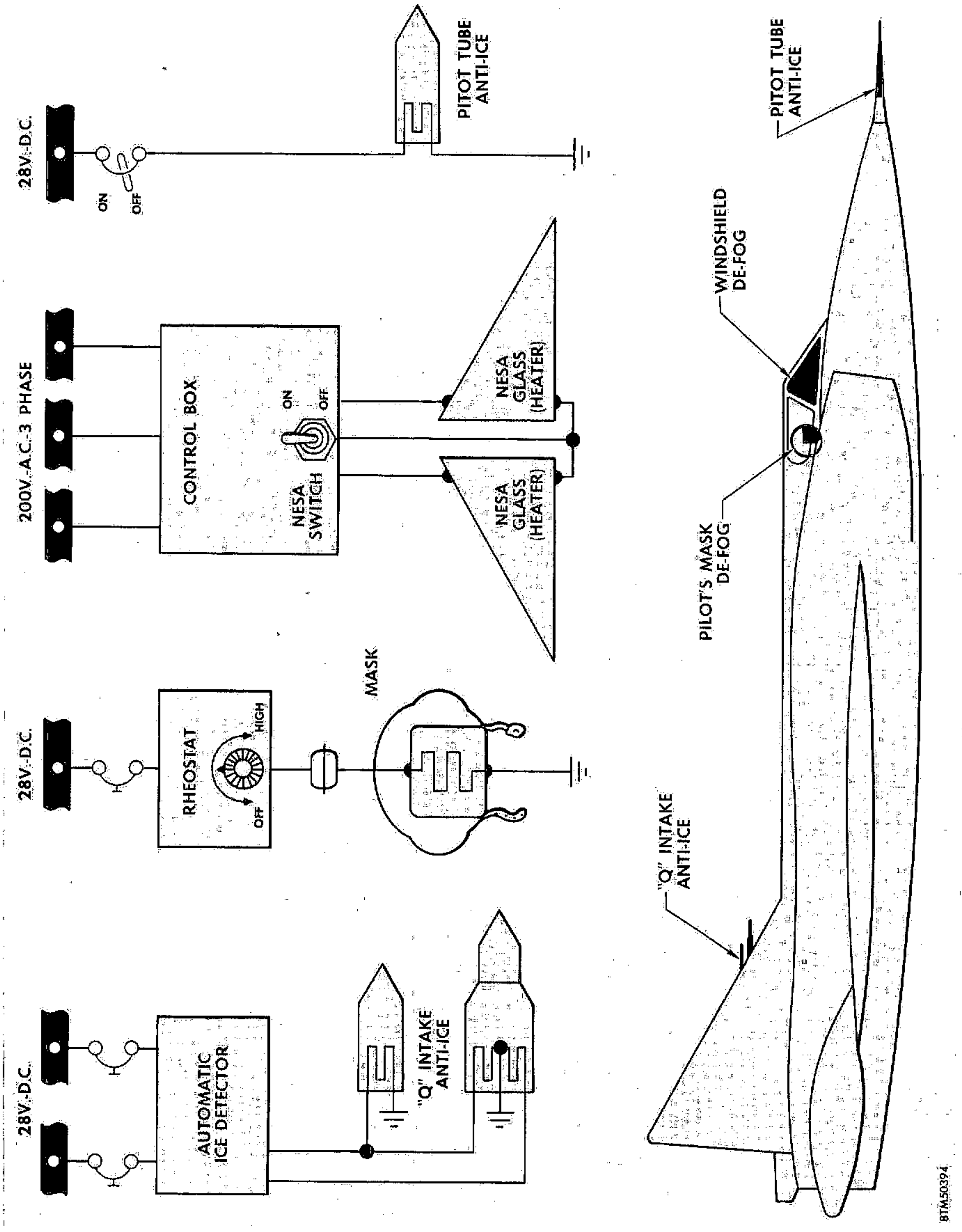


Figure 1-15. Electrical Anti-Icing System

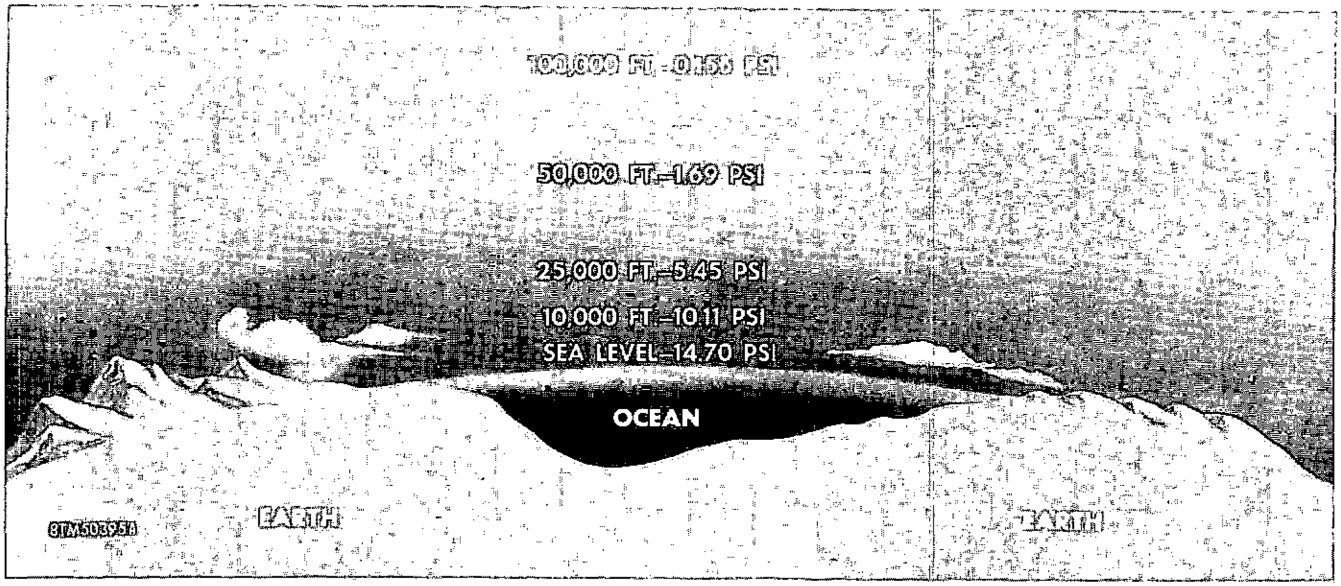


Figure 1-16. The Atmosphere

rapidly descending. Airplane cabins or cockpits are pressurized to prevent this rapid pressure change from occurring.

WHY A LOW-PRESSURE AIRCRAFT PNEUMATIC SYSTEM IS NEEDED.

In the early days of aviation, an airplane was relatively simple. It consisted of a wood-and-fabric frame, a small power plant, and a simple control system. The pilot sat in an open cockpit, protected from the weather by heavy clothing. Utility-type systems were not used; these came later as aviation science advanced and as improved power and higher altitude operation showed a need for them.

Perhaps the first use of air in a utility-type system was a crude heating arrangement in which ram air was heated by conducting it across a hot engine manifold and then directed at the pilot's feet. As time passed, airplanes became more powerful and began operating at higher altitudes. Cockpits were enclosed and more efficient heating and ventilating equipment was developed. Large, high-speed, long-range airplanes were too difficult and too tiring to control manually. Hydraulic, electric, and pneumatic power control systems were developed to eliminate this trouble.

As airplanes began flying at higher and higher altitudes, pressurization systems became necessary, not only for cockpits but also for fuel tanks and hydraulic reservoirs.

In piston-engine powered airplanes, pressurized air is supplied by one or more auxiliary air compressors driven by the main power plant. These compressors add weight to the airplane and supply only a limited amount of air. As you learned previously, air becomes hot when compressed. This *heat of compression* must be removed before the air can be used in the cockpit. To accomplish this, a *heat exchanger,* or similar device, was added to cool the pressurized air.

The development of the turbojet engine greatly simplified the problem of supplying compressed air. In this type of engine, air enters the engine in huge quantities and is highly compressed by multistage compressors before it enters the combustion chambers. A portion of this air is used for fuel combustion, but most of it is used for engine cooling. In most engines, from one to eight percent of this compressed air can be *bled* to supply different systems without lowering engine efficiency. Some sort of cooling device or heat exchanger is still necessary before using bleed air for cockpit pressurization or air conditioning.

Another problem that has kept pace with airplane development is that of *electronic cooling.* Modern aircraft carry an enormous amount of electronic equipment that must be kept cool to operate properly. Ram air, normally used for this purpose, is easily available, but the complicated distribution ducts necessary to route the air present many problems. Since bleed air is very hot, it provides a convenient method of de-icing or defogging various parts of the airplane. From the above discussion, you can see that each part of a low-pressure pneumatic system was developed to meet a particular need. Air-conditioning is necessary to provide a stable cockpit temperature despite extreme external temperature changes during flight. Pressurization is necessary to maintain body functions at extreme altitudes. Anti-icing systems insure the proper operation of instruments and controls and prevent ice from adding weight to the airplane.

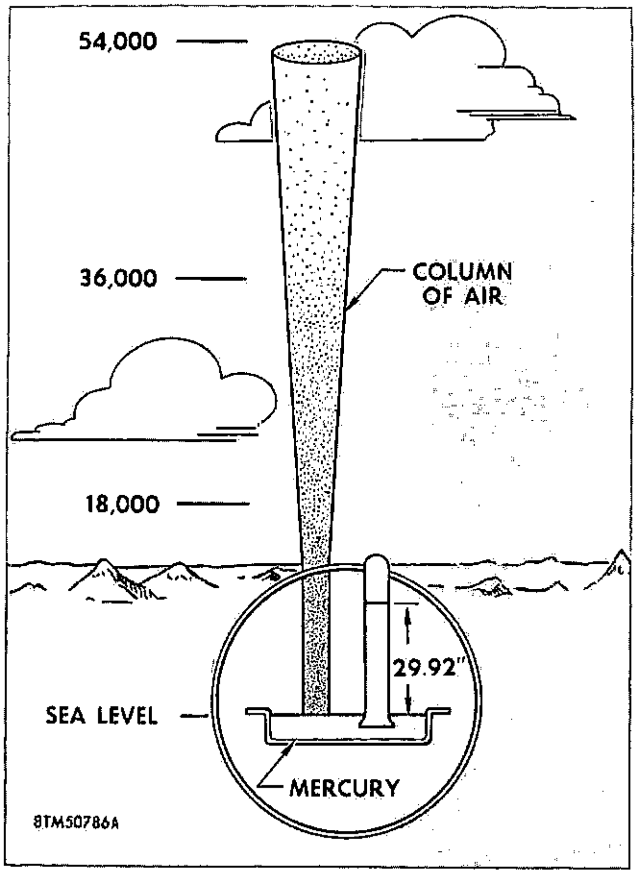


Figure 1-17. Atmospheric Pressure Measurement

Complicated ventilating systems are necessary to keep electronic equipment at proper operating temperatures. All of these subsystems or circuits are grouped together for convenience and called the *low-pressure pneumatic system.*

TYPES OF AIRCRAFT PNEUMATIC SYSTEMS.

Modern high-speed airplanes, like the F-102A, contain many systems and subsystems. Some of them are very complex while others are relatively simple. The major power systems furnish power to operate various other systems. For instance; the hydraulic system furnishes power to operate the flight control system, and the high-pressure pneumatic system powers the armament bay doors.

The electrical system of the F-102A airplane furnishes electricity to power the electronic components and various control circuits. In some airplanes, electric motors are used to move flight control surfaces, to open doors, etc. In the F-102A, all such power functions are performed by either the hydraulic system or the high-pressure pneumatic system.

The F-102A low-pressure pneumatic system is not, strictly speaking, a power system. It furnishes a supply of ram and bleed air to the air-conditioning, ventilating, pressurizing, and anti-icing systems. It differs from the high-pressure system in its source of air, operating pressures and temperatures, rate of flow, and in the type of functions it performs.

CHARACTERISTICS OF THE F-102A PNEUMATIC SYSTEMS.

The F-102A high-pressure pneumatic system uses high-pressure air, from storage flasks, to perform its work functions. Before a flight, the flasks are charged with dry air from a ground compressor. The supply of air is limited, so the number of operating cycles of the armament bay doors or other components is limited. The low-pressure system of the F-102A airplane uses ram air—at atmospheric temperature and slightly above atmospheric pressure—and engine bleed air (up to 427°C (800°F) and 225 psi pressure). The supply of air is unlimited and the rate of flow is very high. Bleed air is available whenever the engine is running, and ram air whenever the airplane is moving.

SUMMARY.

In the preceding pages of this supplement, you have learned in a general way what the F-102A low-pressure pneumatic system is, what it does, and how it operates. You realize that many parts of the system, such

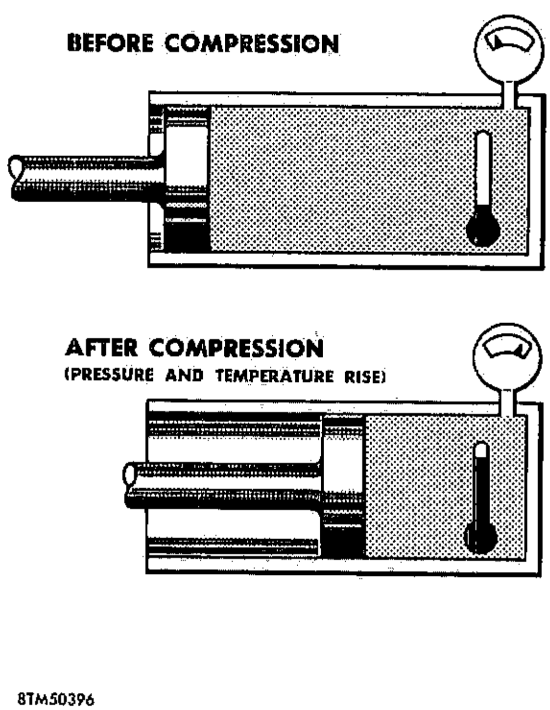


Figure 1-18. Effect of Compressed Air

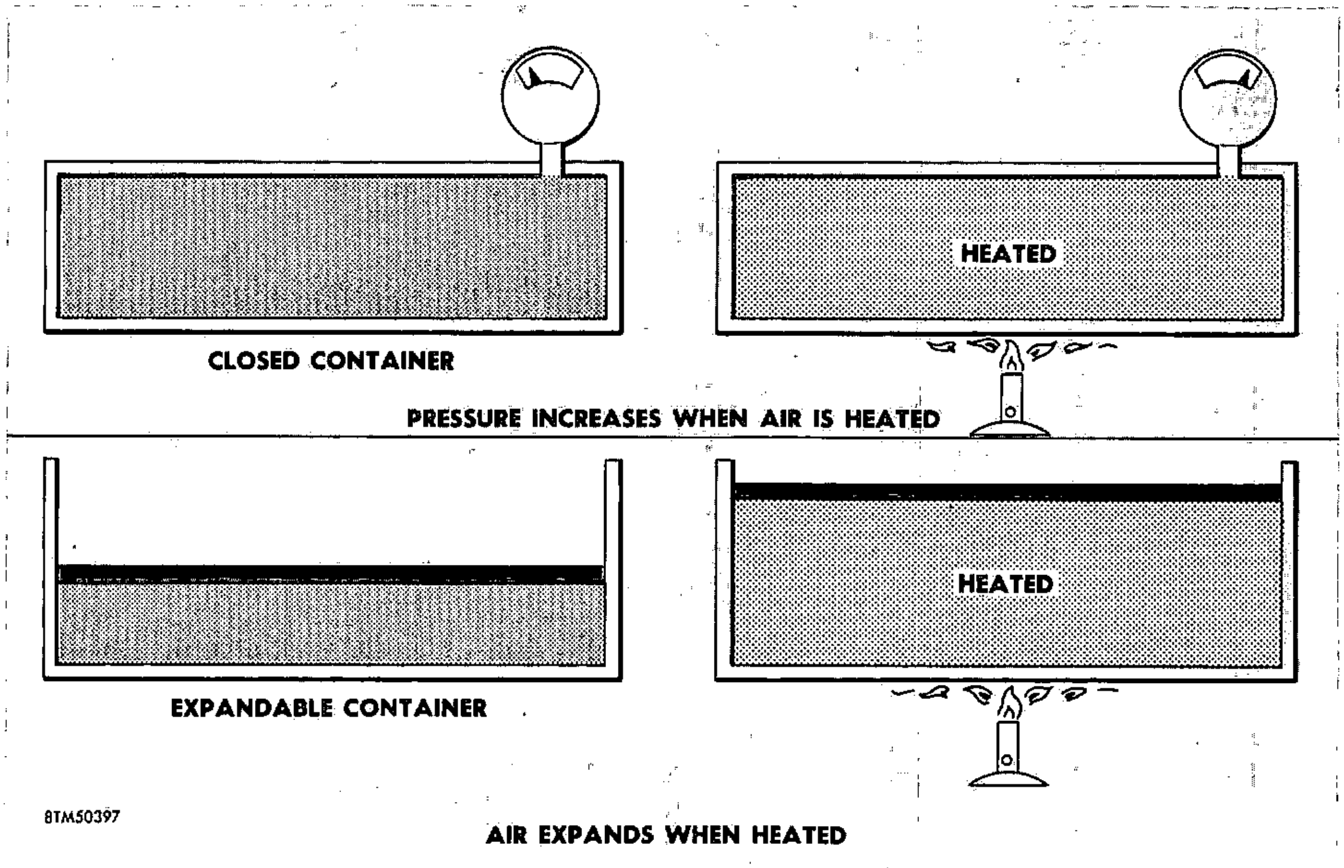


Figure 1-19. Effect of Heating Air

as valves, regulators, and ducts, are subject to extreme temperature and pressure changes. These sudden, drastic changes in temperature cause the metals to expand and contract rapidly. This expansion and contraction over a period of time may cause metal fatigue and failure of an important component of the system.

To properly maintain the system, you must know how to recognize such defects, how to test and inspect the system, and how to correct any faults. In the following chapters you will learn in detail about each part of the system, how to recognize its defects, and what you should do to correct them.

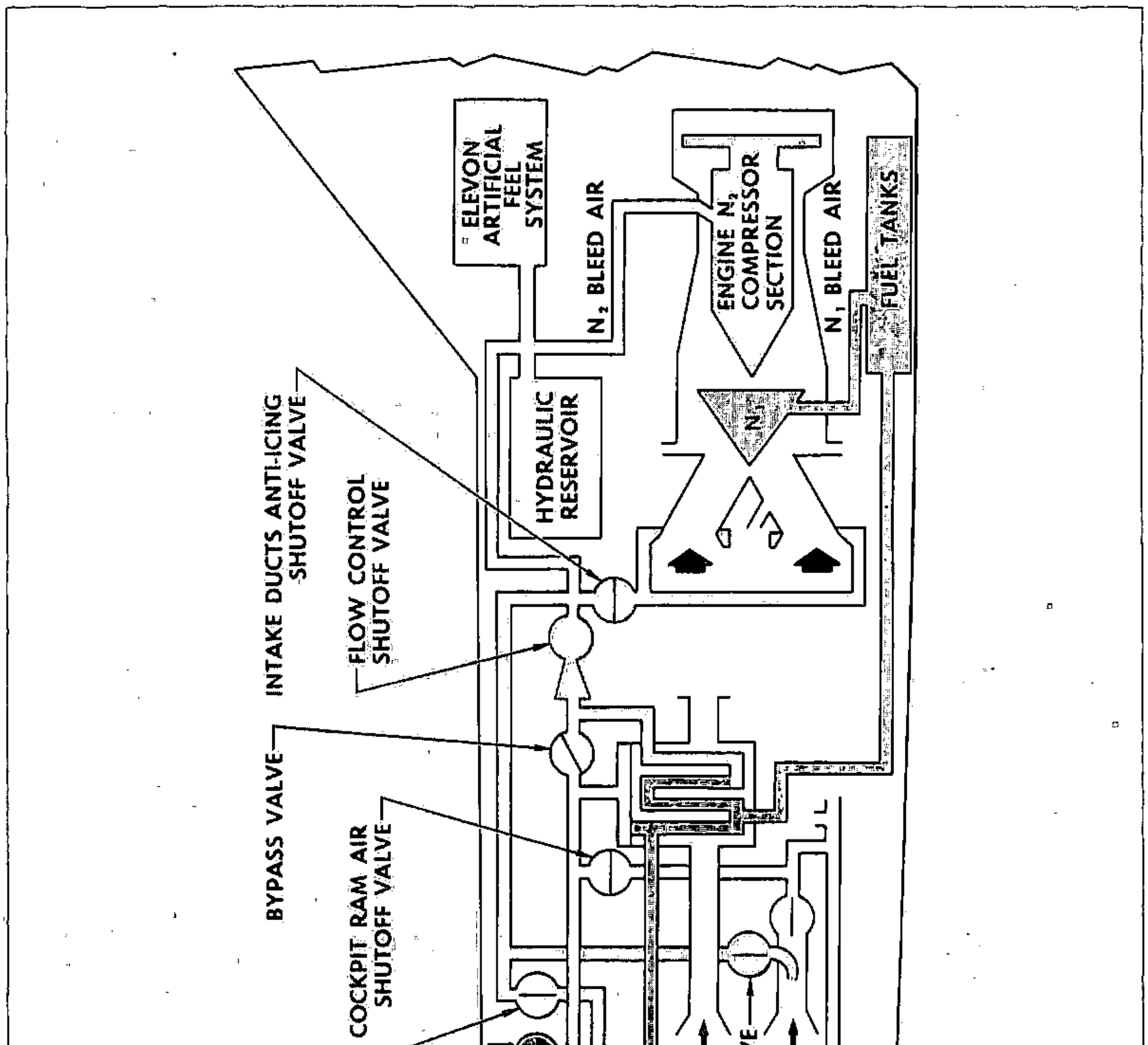


Figure 2-1. Bleed Air Source and Distribution Diagram

Chapter II

AIR SUPPLY SYSTEM

Contents *Page*

The preceding chapter presented an overall view of the F-102A low-pressure pneumatic system—its air sources, supply system, and the various subsystems. Each of the above was described briefly to familiarize you with the various functions of the systems. You also learned what air is, and its reaction to compression and heat.

This chapter describes the air supply system and its two sources of air—engine bleed air and ram air. In this chapter, you will learn how the air supply is distributed and controlled by means of valves and regulators, and the changes that take place in air pressure and temperature as the air passes from one section of the system to another. The various subsystems, such as the air-conditioning, ventilating, and pressurization systems, are covered individually in the succeeding chapters.

ENGINE BLEED AIR SUPPLY.

In Chapter 1, you learned that the low-pressure pneumatic system is supplied hot, compressed air by "bleeding" or tapping air from the engine compressor. The entire engine bleed air system, from its takeoff at the engine compressor to the various subsystems and the heat exchanger, is illustrated in figure 2-1. The lines shown dark in the illustration carry unconditioned bleed air. The lines carrying partially conditioned bleed air are represented by the lighter lines.

You learned in Chapter 1 that the pressurized air for the F-102A low-pressure pneumatic systems is bled from the N_2 section of the engine compressor. The pressure of this bleed air will vary considerably and may be as high as 225 psi. The temperature will also vary, but will not exceed 427°C (800°F). A brief study of the engine and its operation will help you understand why bleed air temperature and pressure is so much higher than atmospheric temperature and pressure.

The F-102A airplane is powered by a J57 turbojet engine with afterburner. The turbojet engine consists essentially of a compressor section, a burner or combustion section, and a turbine section. The afterburner is really a second engine attached to the first and since its operation does not affect the low-pressure pneumatic system, we will not discuss it further in this supplement.

In figure 2-2 note that the compressor section contains two compressors which operate independently of each other. Note also that the turbine section contains a three-stage turbine. The first, or forward, stage of the turbine is connected by a shaft to the aft, or N_2, compressor. The last two turbine stages are connected together and drive the forward, or N_1, compressor by means of a second shaft which is independent of the first and rotates inside of it.

The N_1 compressor has nine stages, or sets of compressor blades, while the N_2 compressor has seven stages, making a total of 16 compressor stages. A fixed "stator" vane is between each set of blades. These fixed "stator" vanes have a definite contour and angle and are designed to receive the air from the preceding compressor blades and deliver it to the next compressor blade (stage) at a workable angle, velocity, and pressure.

Each compressor stage increases the pressure of the air passing through it. The pressure of this compressed air at the sixteenth stage is about 10.5 times the pressure at the engine intake. It is at this point that some air is "bled" off for the low-pressure pneumatic system. After leaving the compressor section, the compressed air enters the combustion section where it mixes with fuel and is ignited. Exhaust gases expand to the rear through the turbine section and

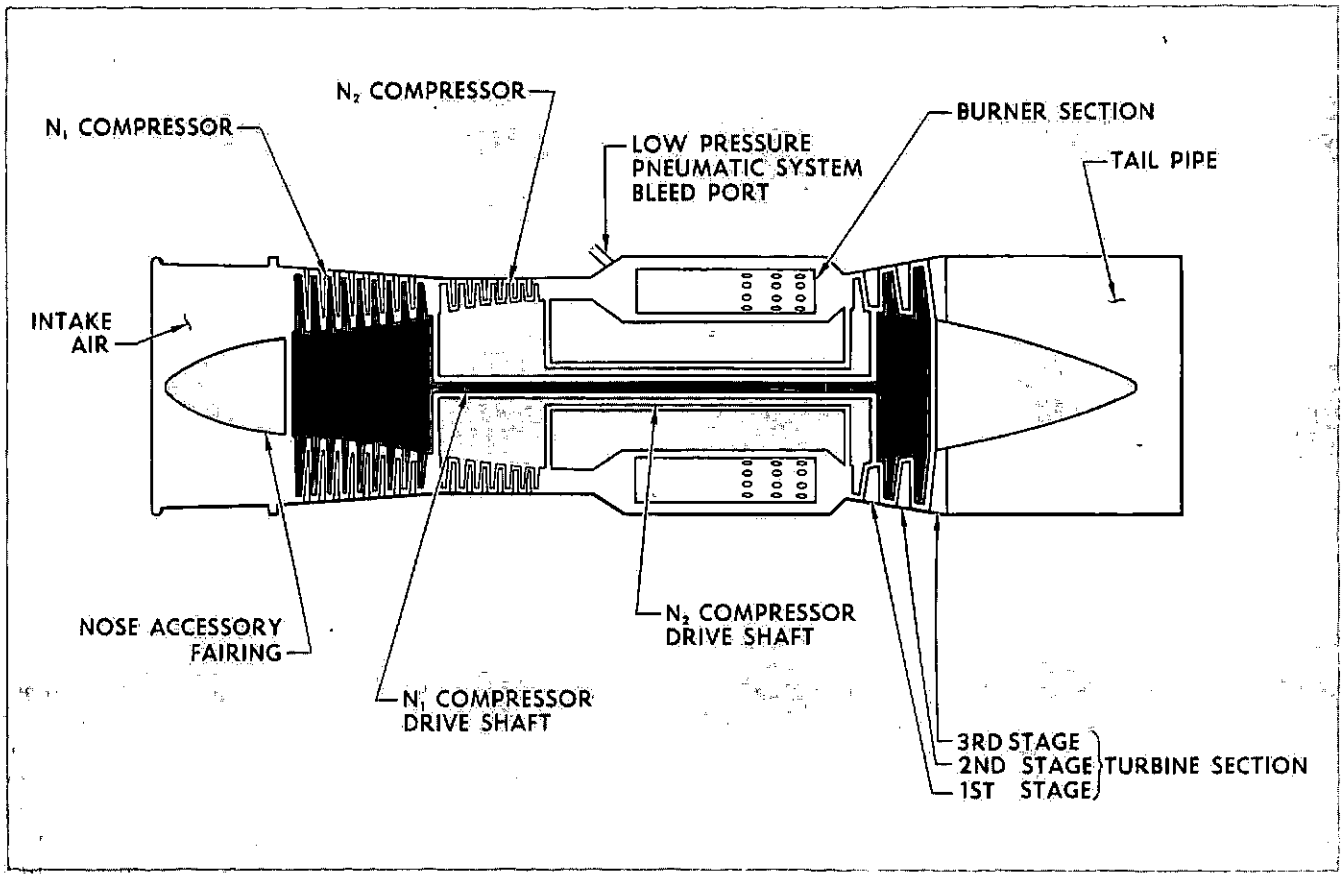


Figure 2-2. Turbojet Engine Diagram

drive the turbine at high speeds. Part of the energy of the exhaust gases drives these turbines, which in turn drive the compressors. The remaining energy of the exhaust gases imparts a forward thrust to the airplane.

As just mentioned, the compression ratio between intake air and sixteenth stage N_2 air is about 10.5 to 1. The *ratio* remains constant. If intake air pressure always stayed the same, N_2 air pressure would stay the same. This, however, is not the case since intake air pressure varies with altitude and speed. Intake pressure is highest at sea level and at maximum speed because atmospheric pressure is highest at sea level, and because the motion of the airplane causes air to be "rammed" into the air intake. Ram air pressure increases with speed.

In Chapter I, you learned that the temperature of air increases as it is compressed. The higher the pressure, the higher the temperature. The temperature of the bleed air will, therefore, vary with pressure, being highest at low altitudes and high speeds.

HOW BLEED AIR IS DISTRIBUTED.

A system of metal ducts distributes the engine bleed air to various sections of the airplane. Distribution lines that carry unconditioned (hot) bleed air are made of stainless steel and are insulated to prevent loss of heat. Bleed air that has passed through the heat exchanger of the refrigeration unit will have a much lower pressure and temperature, so the duct sections carrying bleed air are made of aluminum and are not insulated.

As you might already know, stainless steels have the property of being both strong and corrosion-resistant at high temperatures. Therefore, they are used extensively in exhaust collectors, manifolds, and similar structures subject to much heat. Although aluminum does not possess the same heat-resistant properties that stainless steel does, the areas in which aluminum ducting is used are not required to carry extremely hot air.

As you can see in figure 2-3, several duct sizes are used. The main supply line from the engine compressor is 2½ inches in diameter, with the insulating material increasing the dimension to 3½ inches. The remaining lines that branch off this main supply line vary from ¾ inch to 2 inches in diameter, with the insulation increasing the diameters by ½ inch for the smaller duct sections and one inch for the larger sections.

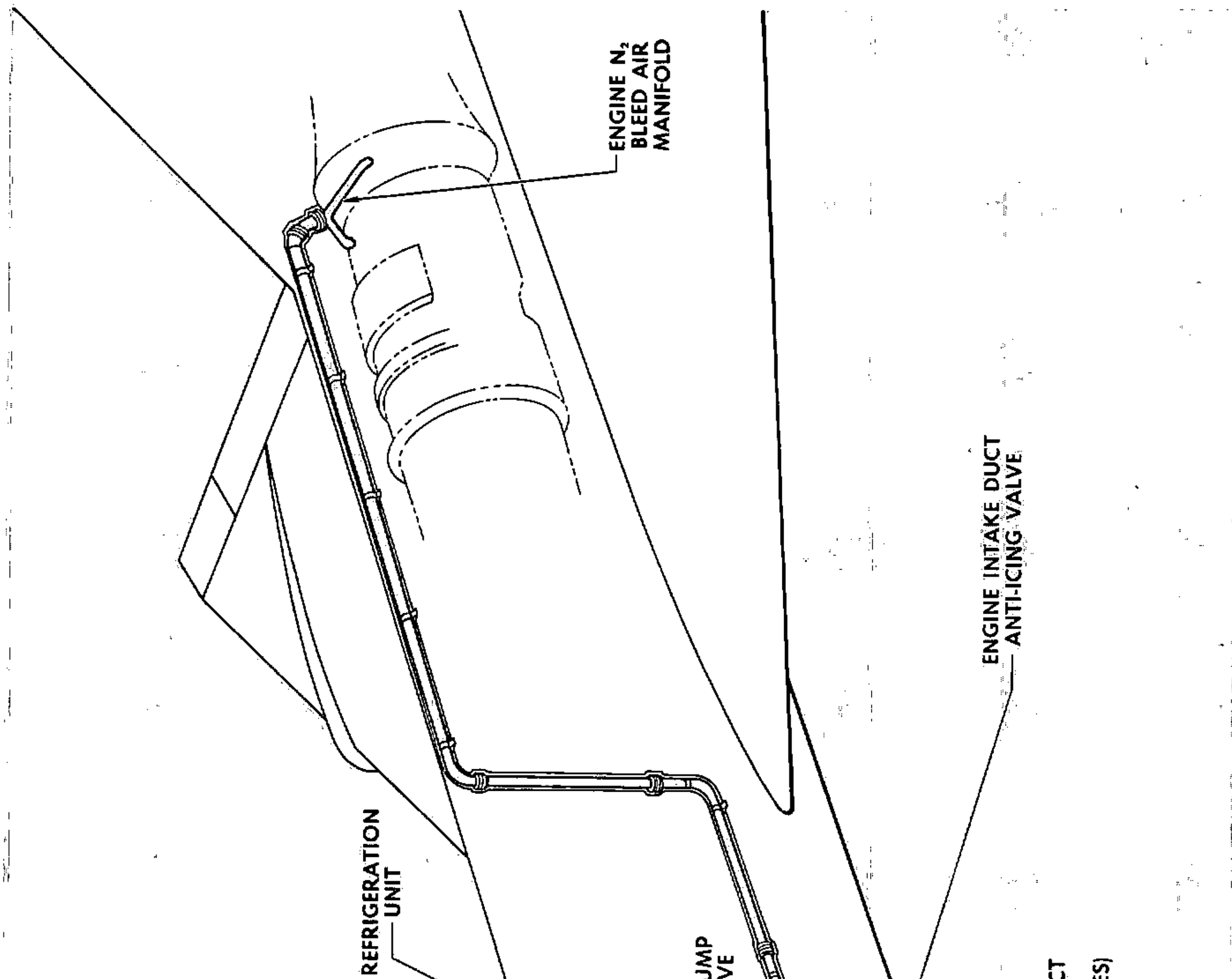


Figure 2-3. Bleed Air Distribution System

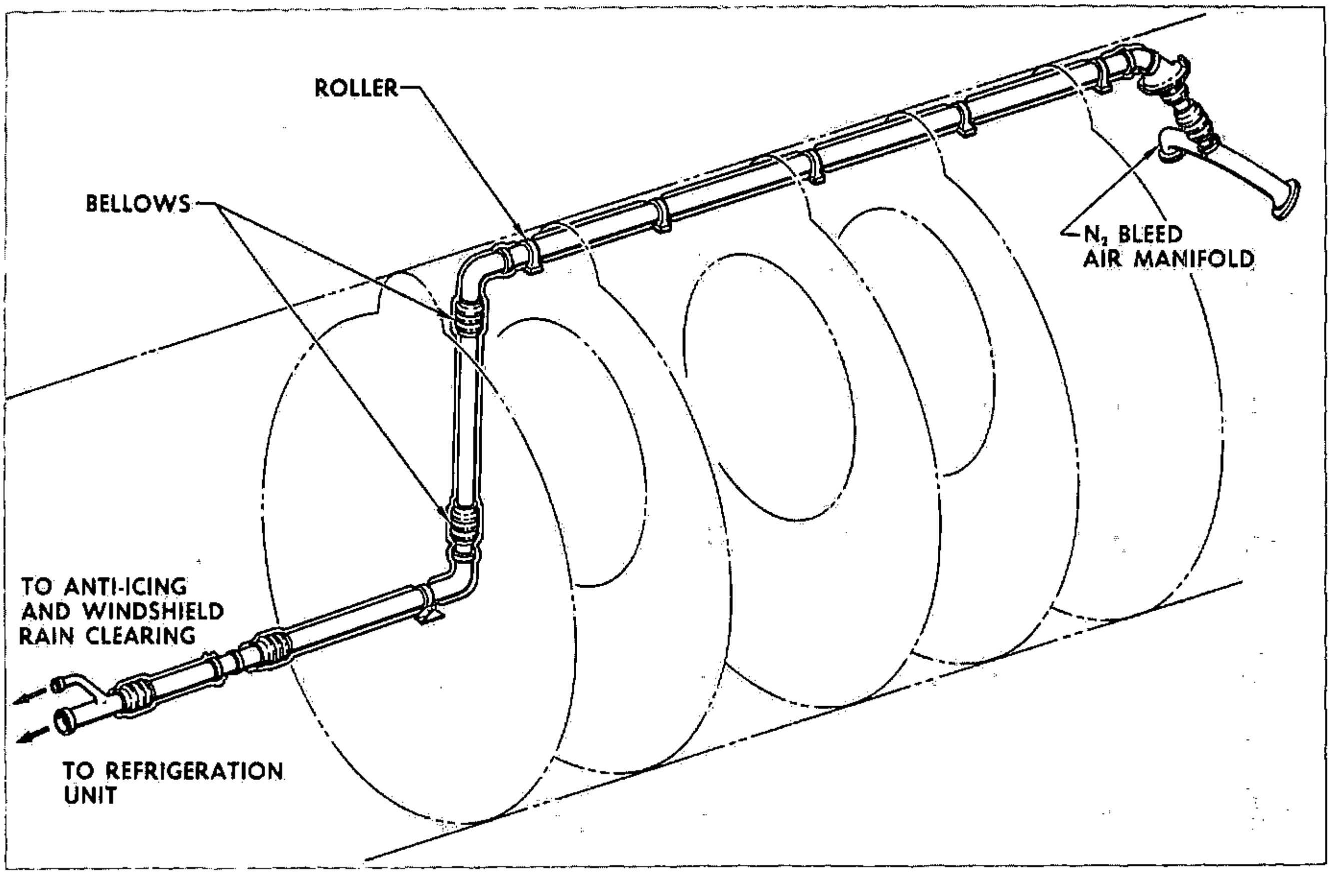


Figure 2-4. Main Bleed Air Supply Line

The main supply line connects to the N_2 bleed air manifold as shown in figure 2-4. Note that it is routed forward through the airplane dorsal fairing on top of the fuselage to the refrigeration unit. In addition to the manifold, the supply line consists of five duct sections. The longest section—the one along the dorsal fairing—is about 16 feet long. The length of this section may vary as much as one inch or more as the duct temperature changes from its coldest non-operating temperature to its maximum possible operating temperature of about 427°C (800°F). Other duct sections of this main supply line expand or contract in the same manner.

This expansion or contraction, as the duct temperatures rise and fall, could cause buckling or other damage. The ducts are therefore mounted on rollers which move back and forth in guide tracks as the duct metal expands or contracts. Figure 2-5 shows one of the duct roller supports and its guide tracks that allow duct expansion and contraction.

In addition to the rollers, flexible "bellows" are built into the duct sections at various points. These bellows expand, contract, or bend as the duct temperature rises and falls. Figure 2-6 shows one of these flexible bellows installed at the bleed air manifold. Detail A shows how a typical bellows fitting is constructed. Note the accordion, or bellows, section which does the expanding and contracting, and the internal structure which limits the overall joint movement.

The smaller ducting that supplies hot bleed air to the windshield rain-clearing, intake-duct anti-icing, and the jet-pump cooling systems connects to the main supply line just aft of the refrigeration unit as shown in figure 2-4. Note that each of the three systems has its own valve and is independent of the refrigeration unit. The ducting of these three systems is similar to, but smaller than, the main supply ducting. Duct sections are joined in the same manner and are insulated in the same way as described below for the main supply line. The ducting for each system is described more fully in the succeeding chapters of this Trainning Supplement, where each system is discussed individually.

Ducting or tubing supplying the canopy seal, G-suit, glycol tank, fuel tank pressurization, and the canopy defogging systems connect to the heat exchanger. (See figures 2-3 and 2-4.) These ducts carry partially conditioned air and are not insulated because of the relatively low temperature of the air carried. Uninsulated aluminum ducting also carries conditioned air from the refrigeration unit to the cockpit air-conditioning system.

How Duct Sections Are Joined.

Any air leak in the engine bleed system, especially in the main supply line, can seriously reduce the efficiency of the bleed system or completely disrupt it. Because of this, all duct systems are joined in such a manner that they are able to withstand the high temperatures and pressures.

Figure 2-7 shows a typical engine bleed air duct connection. Note that the duct sections are held together with a special clamp and are sealed with a gasket. The flanges on the ends of the ducting are welded to the ducting during assembly and are not removable for line maintenance. If a ducting section is new when being installed, there will be a plastic coating on the flanges; this coating must be carefully removed to avoiding damaging the edges of the flanges.

Figure 2-8 shows a typical duct joint assembly. Note that the inside of the clamp is in the shape of a "V". Bring the duct flanges carefully together and align them with the gasket centered between the flanges as shown. Hold the ducts together while the clamp is installed. The nut on the clamp must be tightened with a torque wrench, set at 80 inch-pounds (±10). After the clamp is tightened, tap the clamp around its entire circumference with a small non-metallic type mallet. This procedure distributes any binding around the clamp and insures a tight seal around the entire joint. Retighten the nut to the 80 inch-pounds torque, and then lockwire it to the swivel holding the "T" end of the clamp bolt.

Duct Installation.

Bleed air duct insulation consists of one-half inch of compressed fiberglass sandwiched between layers of very thin stainless steel shells. Note in figure 2-9 that the insulation is in two half sections. These sections of insulation vary in length according to the distance between clamps. The half sections fit around the duct and are laced together with lockwire as shown in the right view of the illustration. Whenever you install or remove this duct insulation, be careful not to damage the metal shell. It is so thin it could easily be punctured with a screwdriver or other tool.

Access Provisions for Bleed Air Ducting.

In performing the necessary maintenance and inspections on the bleed air system, you will have to remove doors and, in some cases, equipment to gain access to the bleed air ducting and connections. Figure 2-10 shows most of the bleed air ducting and its location in the airplane. As you will note, the longest section of ducting is along the top or "dorsal" of the fuselage. To gain access to this section of ducting, you must remove the six sections of dorsal fairing as shown in the illustration.

Note the location of the access doors at the base of the fin. These doors must be removed before you can

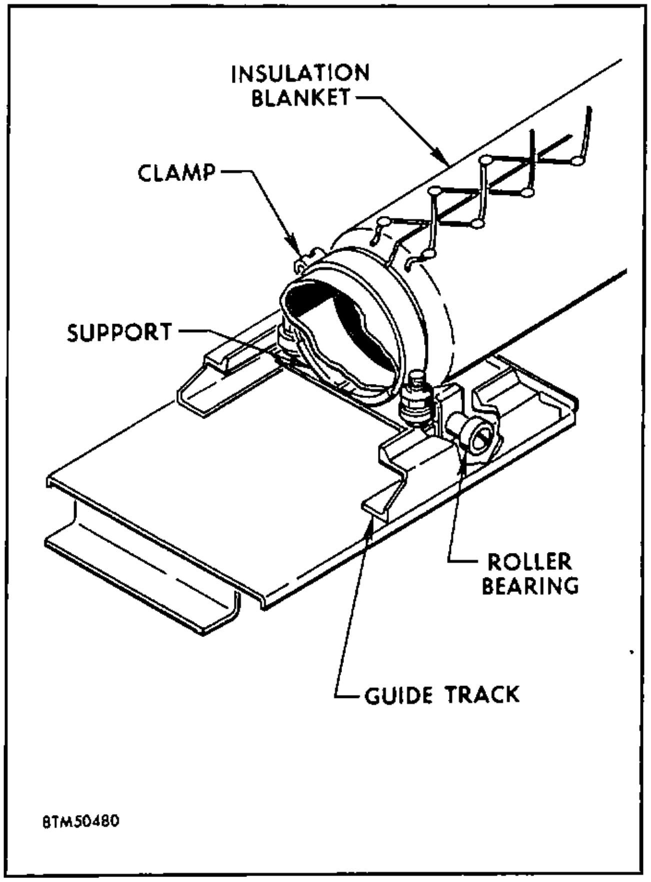


Figure 2-5. Bleed Air Duct Expansion Roller Support

perform maintenance on the joint between the aft duct section and the bleed air manifold. Access to the remaining ducting forward of the long section can be obtained by opening either the upper electronic compartment door or the intermediate electronic compartment door. Some of the bleed air ducting can also be reached through the nose wheel well.

The Refrigeration Unit.

The refrigeration unit converts hot, compressed engine bleed air to conditioned cockpit air. It also furnished partially conditioned air to the various subsystems. Two views of this refrigeration unit are shown in figure 2-11. Note that it consists of a heat exchanger, an expansion turbine and blower, a flow-control valve assembly, and a bypass valve. The oil level sight gage for the expansion turbine and blower can be seen in the top view.

Air entering the refrigeration unit is cooled in two stages; the heat exchanger is the first stage, and the expansion turbine is the second. The flow control assembly regulates the flow of bleed air to the unit. The bypass valve receives signals from a temperature control system and allows a certain proportion of hot bleed air to mix with cold conditioned air in order to deliver air to the cockpit at the temperature selected by the pilot.

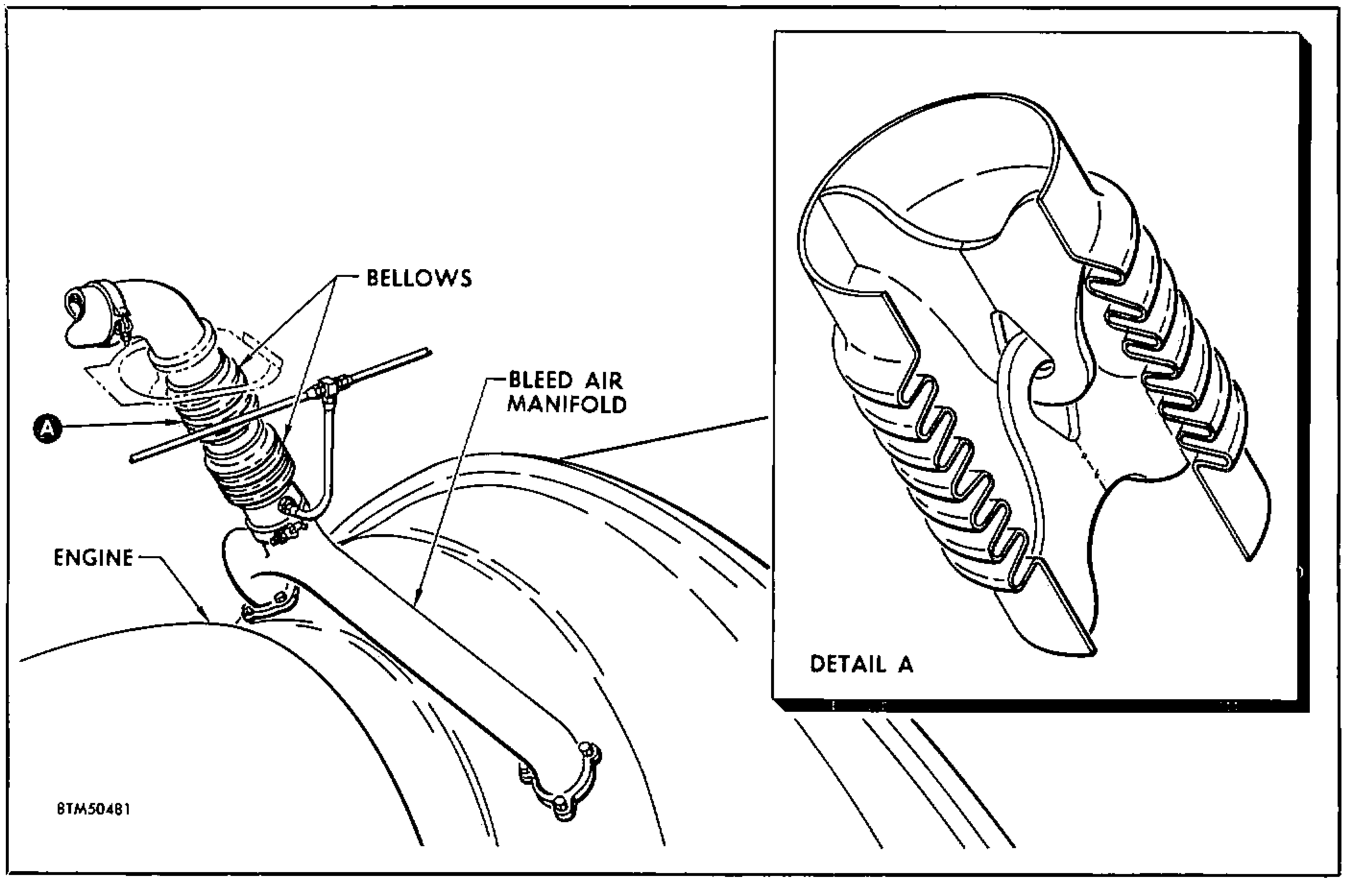


Figure 2-6. Typical Expansion Bellows Fitting

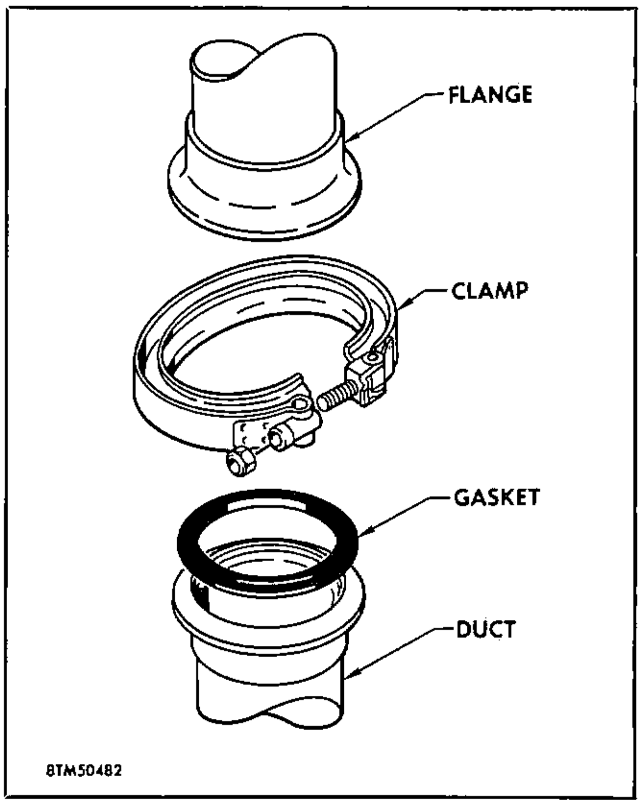


Figure 2-7. Typical Bleed Air Duct Connection

Figure 2-12 shows the location of the refrigeration unit, between the intermediate and upper electronic compartments. Some parts and connections of the refrigeration unit can be reached from the upper compartment while others can be reached only from the intermediate compartment. The unit itself can be removed through either compartment opening.

When working in the electronic compartments, always be very careful not to damage the many electronic components. In some cases, it may be necessary to remove some electronic equipment in order to work on the refrigeration unit.

Heat Exchanger.

The heat exchanger section of the refrigeration unit (figure 2-13) uses ram air from the right boundary-layer intake to cool bleed air passing through the exchanger. The bleed air first passes through the flow control shutoff valve to the heat exchanger. In the heat exchanger, the bleed air passes through a system of tubes before continuing to the expansion turbine. Ram air is conducted across these tubes, absorbs heat from them, and is then vented overboard through the blower portion of the turbine.

The cooling capacity of the heat exchanger depends on the temperature and rate of flow of the ram air.

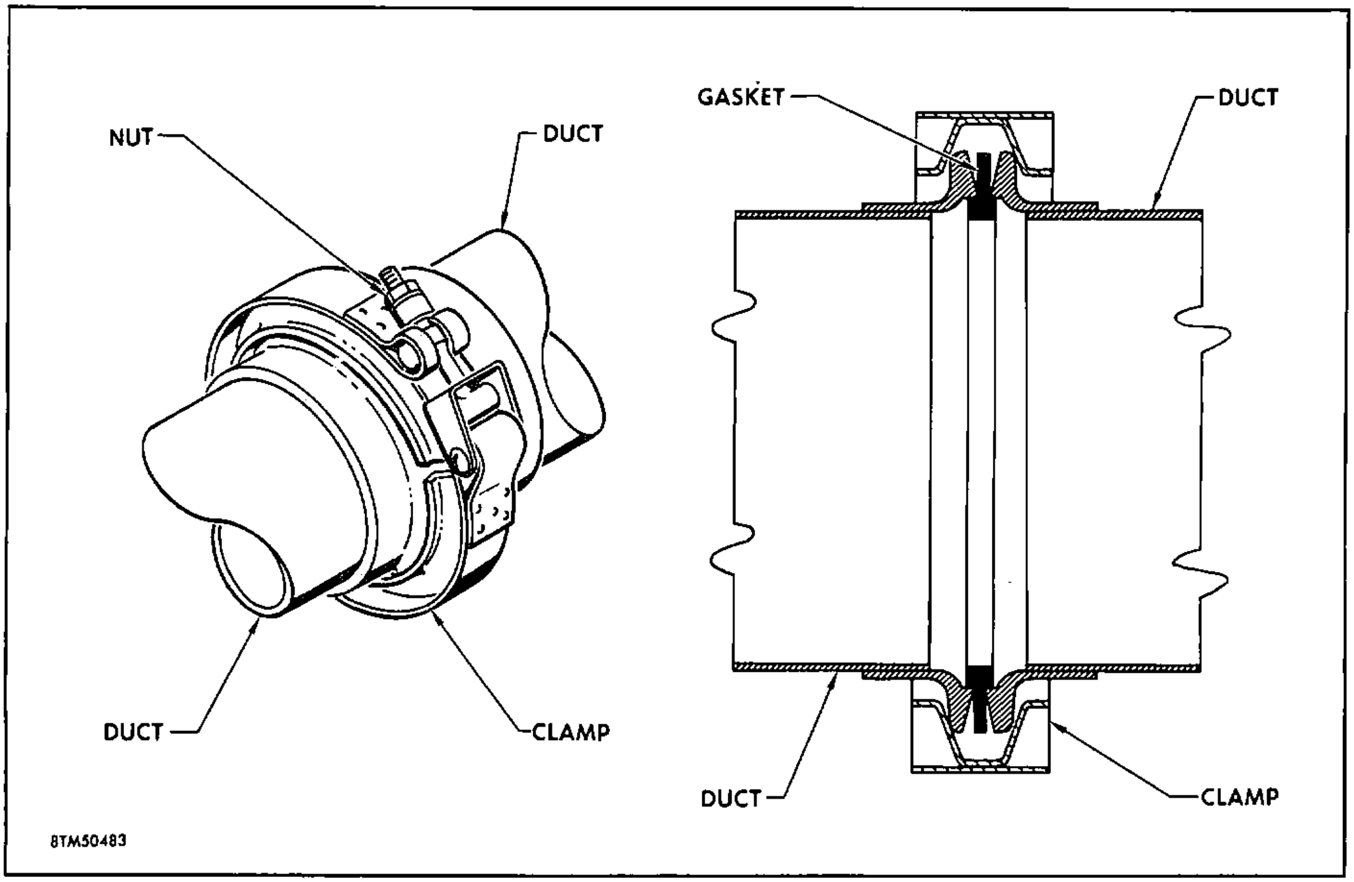


Figure 2-8. Assembled Duct Joint and Clamp

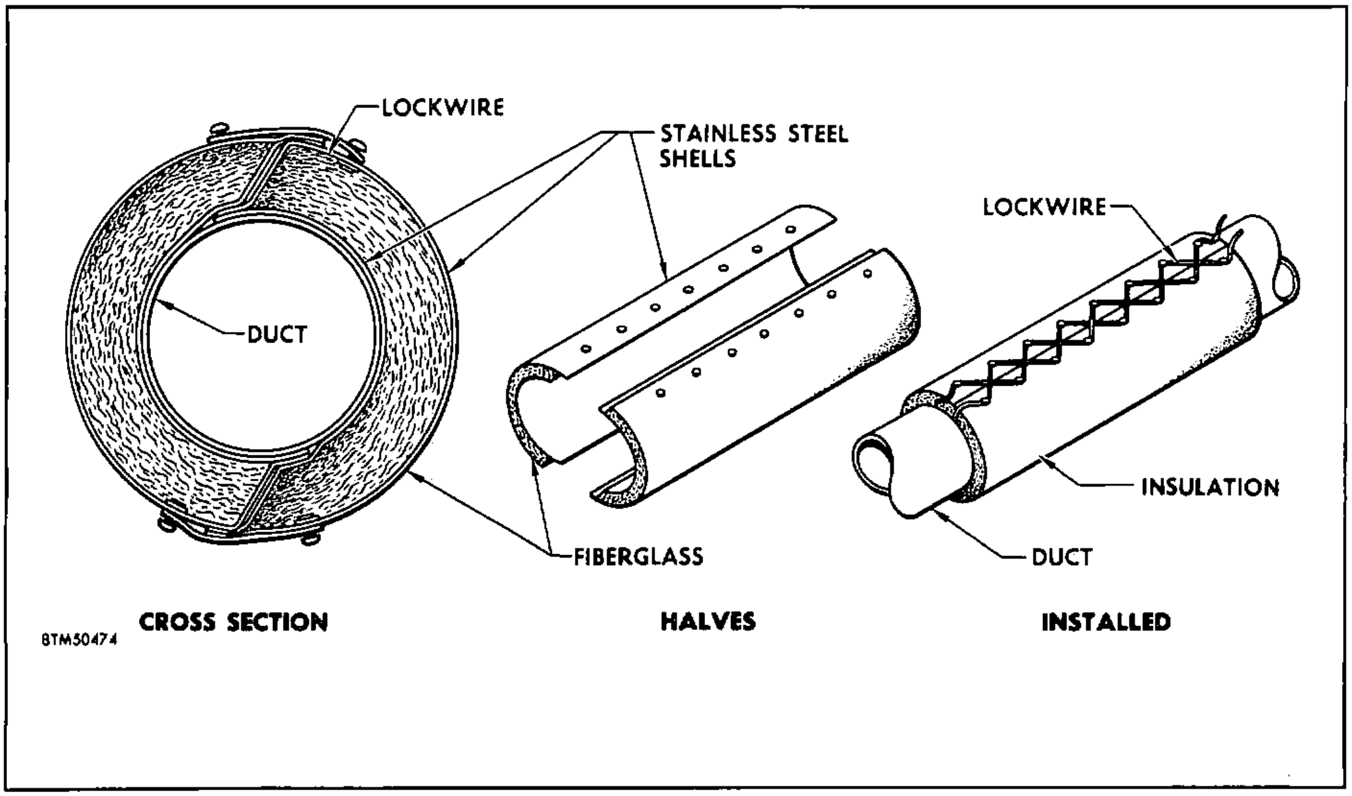


Figure 2-9. Duct Installation

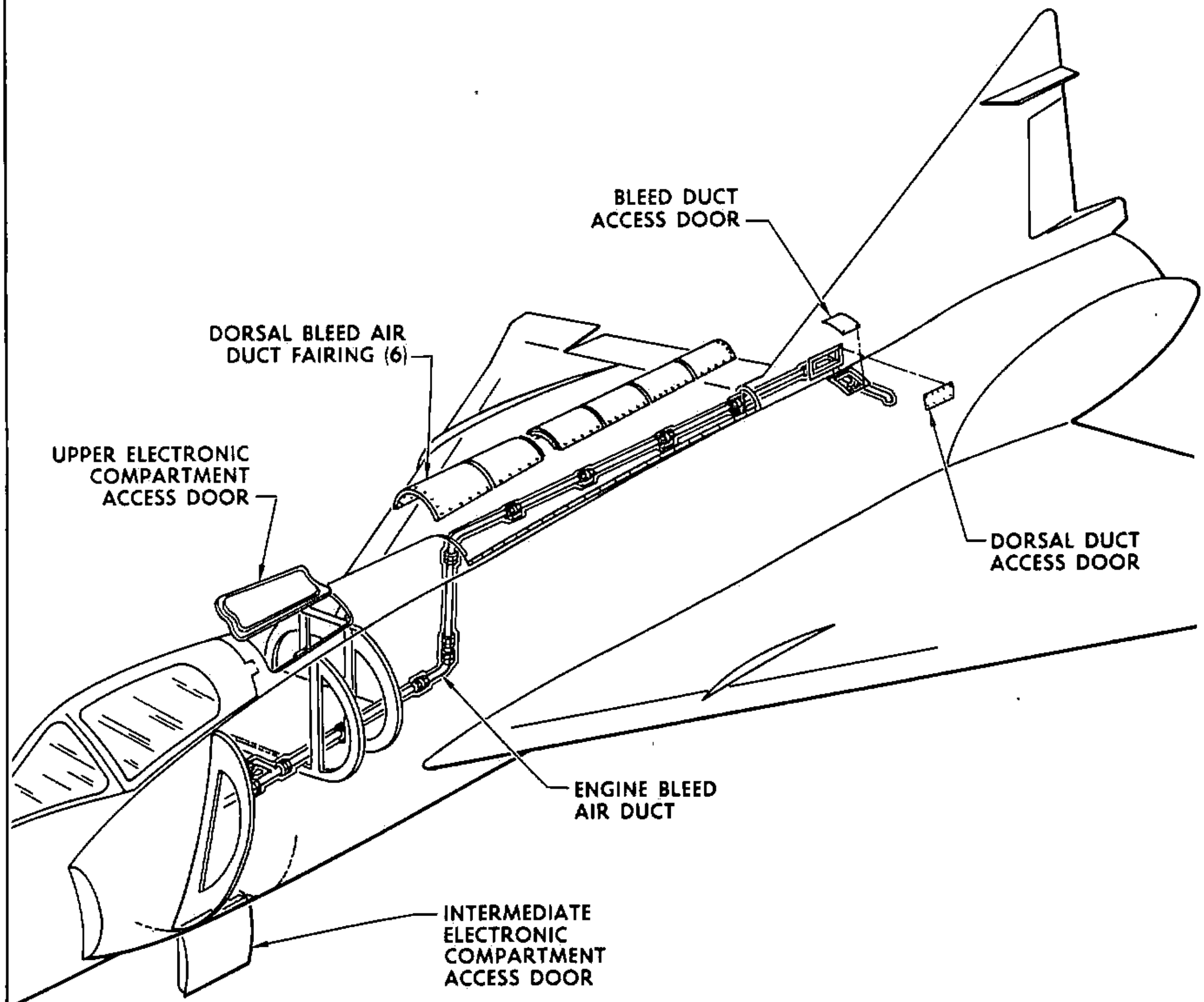


Figure 2-10. Bleed Air Ducting Access Provisions

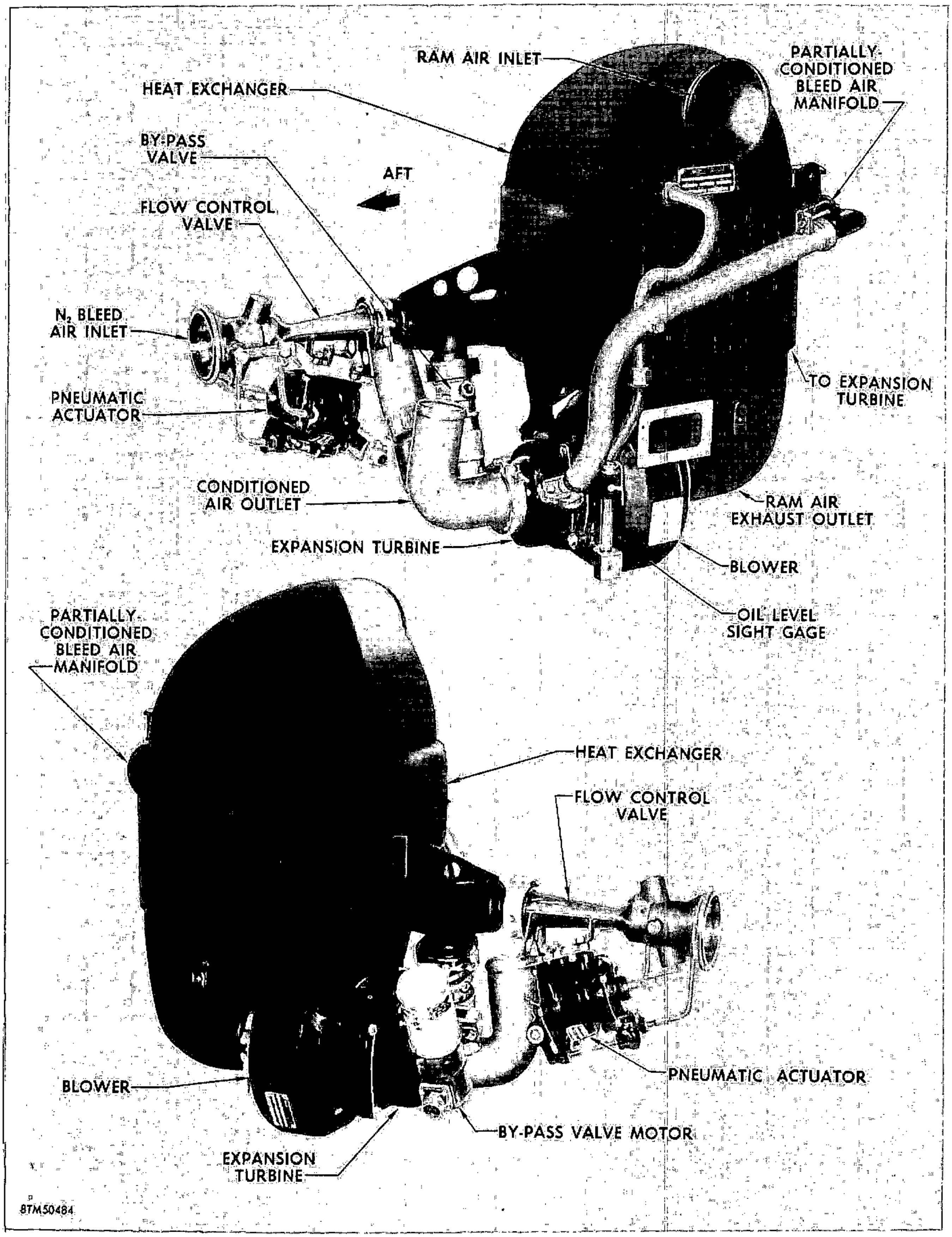


Figure 2-11. Refrigeration Unit

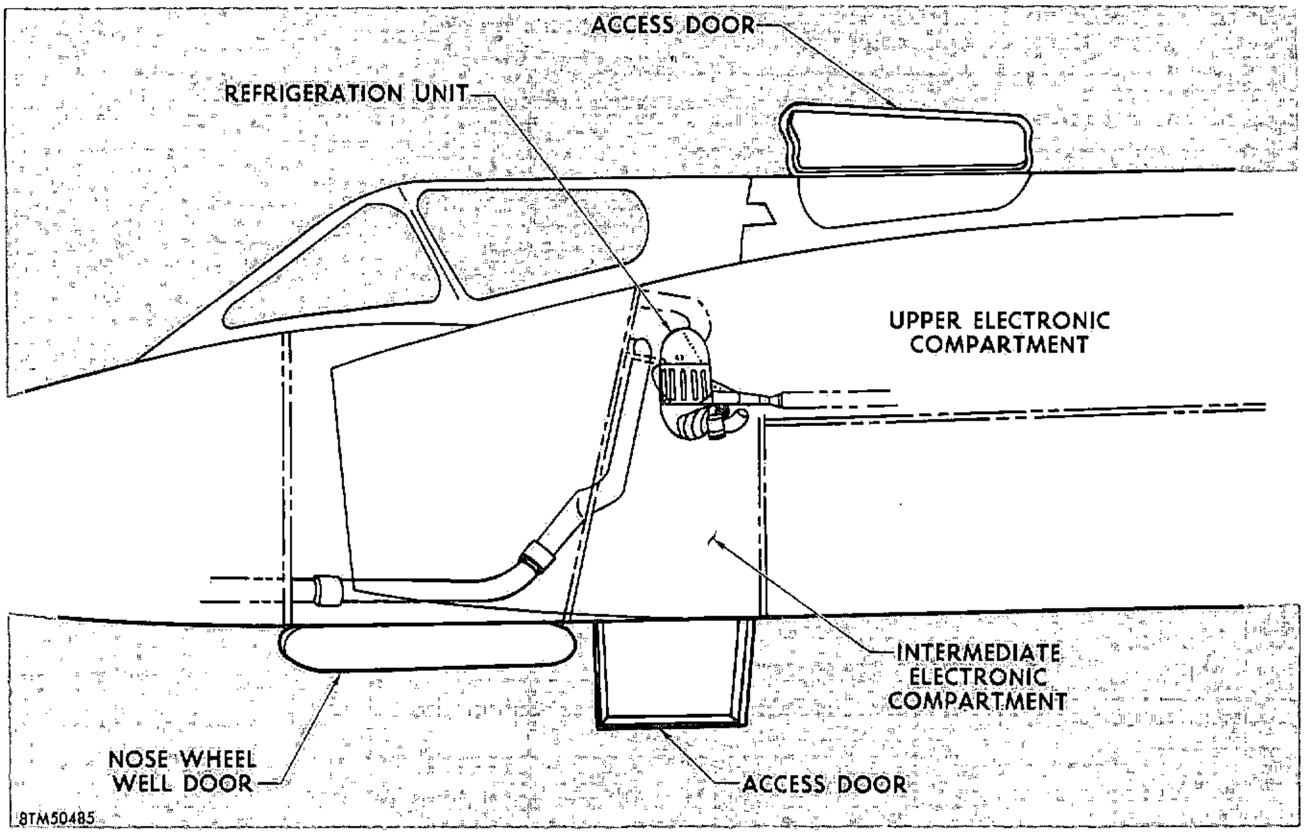


Figure 2-12. Refrigeration Unit Location and Access Provisions

Temperature of the ram air increases with airplane speed and decreases with altitude. It also varies with the weather conditions. Rate of flow of ram air increases as the speed of the airplane increases. The rate of flow, however, does not depend entirely on airplane speed since a blower fan, driven by the expansion turbine, increases the rate of ram air flow. This fan draws cooling air through the heat exchanger even when the airplane is not moving. The cooling capacity of the heat exchanger is greatest at low atmospheric temperatures and at higher airplane speeds.

Cooled engine bleed air leaving the heat exchanger is called partially conditioned air. Note in figure 2-13 that the manifold which distributes this air is positioned across the left end of the heat exchanger. This manifold provides "taps," or fittings, which supply several of the pressurization subsystems with partially conditioned air. These subsystems are described later in Chapter V.

The temperature and pressure of the "exit" air depends not only on the cooling capacity of the heat exchanger, but also on the temperature and pressure of the bleed air as it enters the heat exchanger. As we have seen, bleed air temperature and pressure depend on engine speed, as well as ram air temperature and pressure. With so many variables it is difficult to say what the temperature and pressure would be. Under certain conditions, bleed air entering the heat exchanger with a temperature of 357°C (675°F) and a pressure of 180 to 200 psi might leave the heat exchanger with a temperature of 93°C (200°F) to 107°C (225°F) and a pressure of 75 to 85 psi. Since the heat exchanger has no moving parts, few maintenance problems need be expected.

Expansion Turbine.

The air from the heat exchanger is further reduced in temperature and pressure by allowing it to expand through the expansion turbine. You will recall from Chapter I that the temperature and pressure of air will drop if the air is allowed to expand. The temperature and pressure of the air leaving the turbine depend upon the temperature and pressure of the air leaving the heat exchanger. The temperature of the turbine exit air will normally be in the −18°C (0°F) to 5°C (40°F) range, and the pressure will be around 20 psi.

In figure 2-14, note that the force or energy of the air entering the expansion turbine turns the turbine rotor. Note also that the turbine rotor is mounted on the same shaft with the blower fan which it drives. This blower fan increases the flow of ram air through the

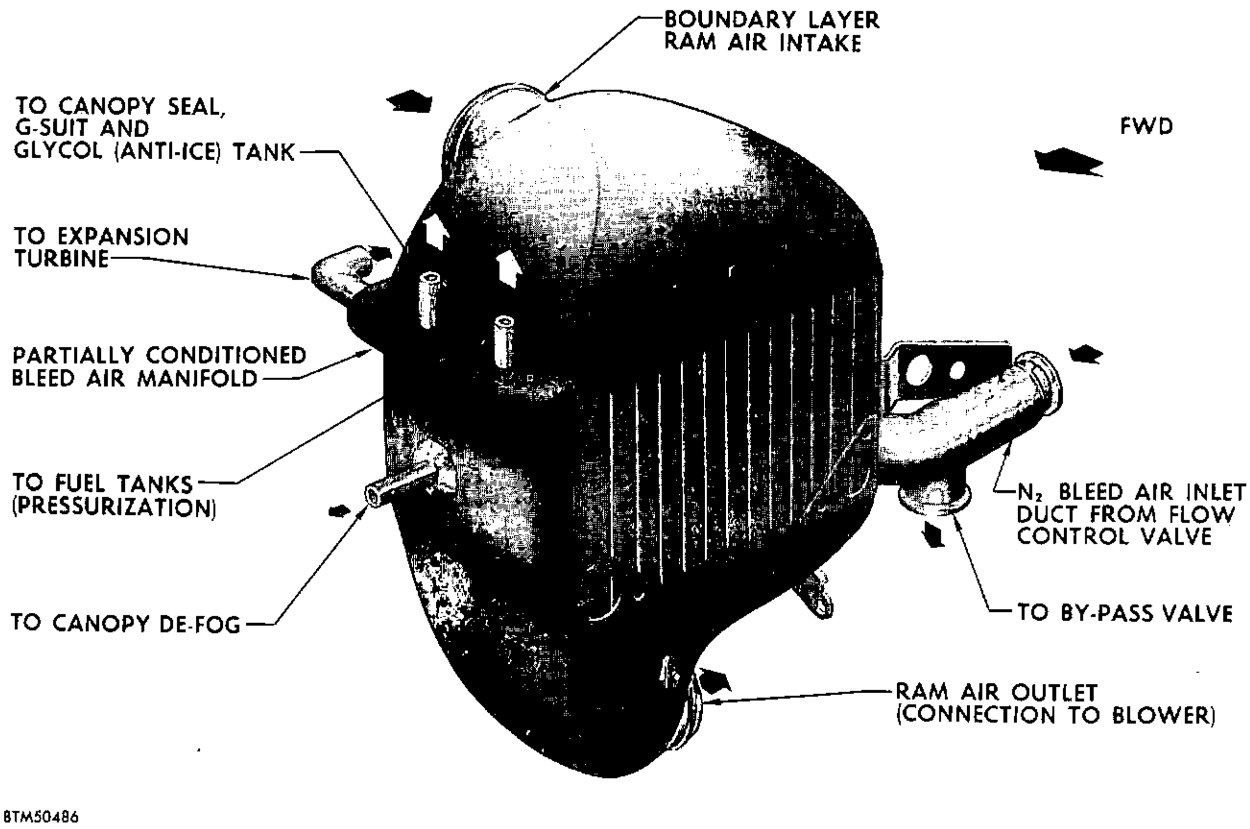


Figure 2-13. Heat Exchanger

heat exchanger. The speed of rotation of the turbine rotor and blower fan will vary with the rate of flow of bleed air through unit. The speed is greatest when the bypass valve is fully closed, and lowest when it is fully opened.

The turbine and blower fan bearings are wick-lubricated from the small oil reservoir shown below the shaft. The turbine and blower assembly is expected to have long service life. If periodic inspections reveal that the assembly is defective, replace it as a complete unit.

Figure 2-15 is a close-up of the expansion turbine oil sight gage. It consists of a glass tube with oil level FULL and ADD markings. When the oil level drops to the ADD mark, it will take about 20cc of oil to bring the level up to the FULL mark. The total capacity of the reservoir and gage is about 55cc. In filling this sight gage and reservoir it is only necessary to remove the filler cap shown and add the oil. When the reservoir is to be drained, remove the drain plug shown at the bottom of the gage.

Flow Control Assembly.

Flow of bleed air to the heat exchanger is controlled by a butterfly shutoff valve, operated by pneumatic pressure and controlled by an electrical solenoid. Note in figure 2-16 that this flow control assembly consists of a shutoff valve and pneumatic actuator with connecting lines and mechanical linkage. You can see this assembly installed on the refrigeration unit in figure 2-11. This assembly is replaced as a unit when-

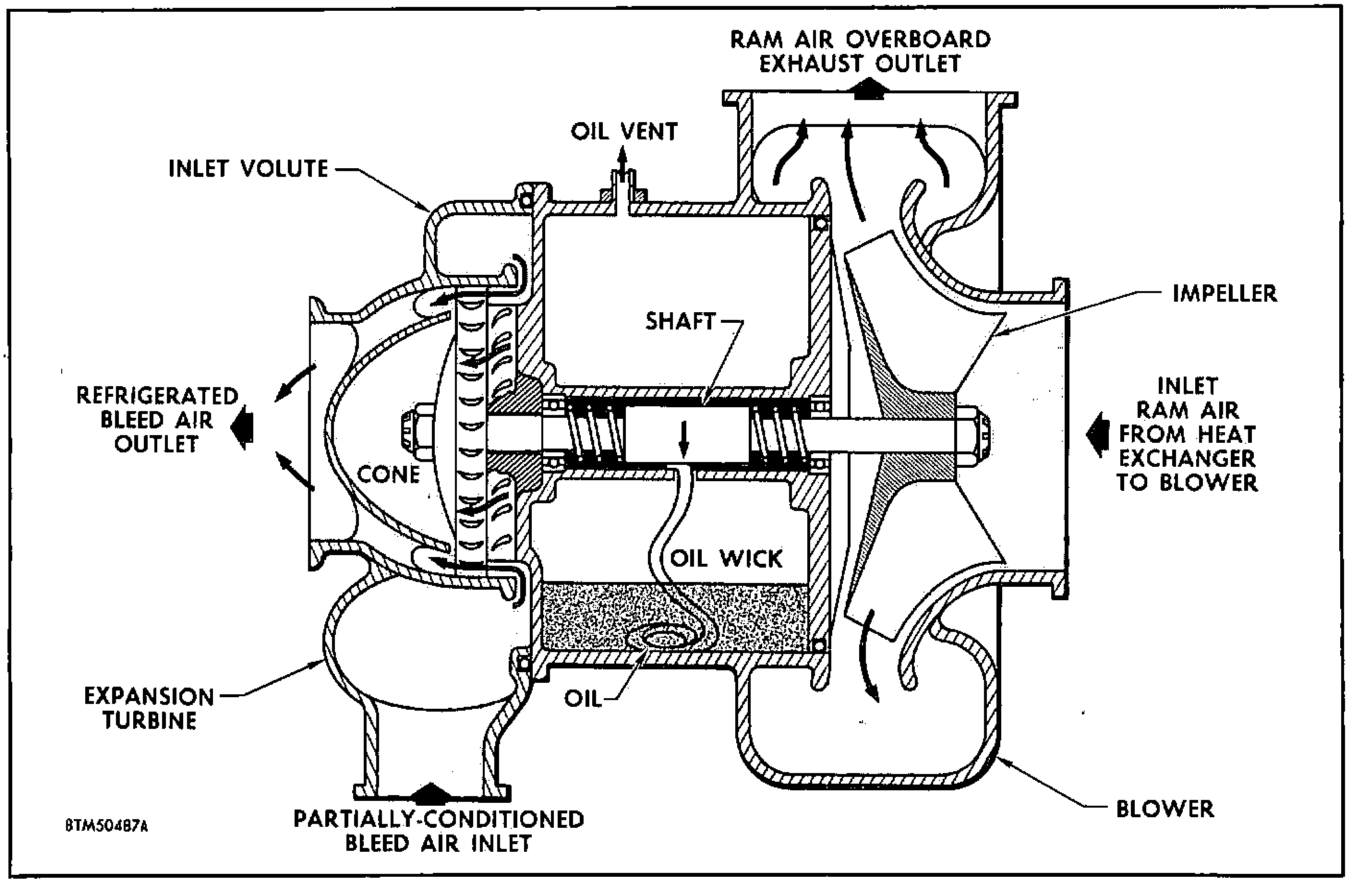


Figure 2-14. Turbine and Blower Schematic

ever it malfunctions or is due for a periodic overhaul check.

In the cutaway illustration (figure 2-17) note that the shutoff valve has a butterfly shutter with a metal sealing ring in a groove around the circumference of the shutter. This ring is similar to the piston rings in a gasoline engine. As the shutter closes through rotation of the shaft, the ring forms a tight seal to prevent the leakage of bleed air.

The air passage in the valve body is much smaller downstream (to the right) of the valve than it is upstream. This narrow passage restricts the flow of air to a maximum of about 40 pounds per minute. This weight of air refers to the total weight of the air molecules. When air is compressed, there will be many more molecules in a given volume than in the same volume of free air. Therefore a given volume of compressed air will be much heavier than the same volume of free air. The valve shaft is connected by mechanical linkage to the piston of the pneumatic actuator.

How the Flow Control Assembly Functions.

Figure 2-18 shows schematically how the actuator and solenoid work together to operate the valve. Note that there are three air chambers in the body of the actuator. These are labeled A, B, and C on the diagram. Pressurized air for operation of the actuator is drawn from the valve body upstream (to the left) of the valve shutter. This air is distributed to various parts of the actuator as shown in the illustration. For purposes of identification in the discussion, various points in the air lines are marked by the numbers 1, 2, 3, etc.

To prevent too sudden a rise in the cockpit pressure, it is necessary to modify the opening of the valve to limit the flow of air to the cockpit. To provide this gradual rise in cockpit pressure, a "rate sensor" unit is installed in the cockpit wall to sense cockpit pressure. This rate sensor controls the opening rate of the flow control valve. This rate sensor is connected to the flow control assembly at the point marked 2 on the illustration.

The upper section of this illustration shows the solenoid in the deenergized position and the shutoff valve closed. The three chambers (A, B, and C) will be pressurized anytime there is bleed air in the supply line. The pressure in chambers B and C tends to extend the piston, while the pressure in chamber A tends to retract it. The pressure forcing the piston to the left (extended position) is greater than the pressure forcing it to the right (retracted position). As a result, the piston will be in its left position and the valve will be closed, since the pressure in chamber A is normally greater than the pressure in chamber B.

As you can see in figure 2-18, this is because there is a small flow of air through the restrictor at point 6. This air returns to the valve body at point 3 instead of going to chamber B, and bleeds off some of the pressure that would otherwise be in chamber B, thus allowing chamber A pressure to exceed chamber B pressure. This difference of pressure tends to force the piston to the right, opening the shutoff valve. However, as long as chamber C is pressurized the valve will remain closed.

Now, by referring to the lower schematic, you can see what happens when the solenoid is energized. In this case, the solenoid piston closes the small valve at the intake port (point 4) to chamber C, thus cutting the chamber off from its source of air pressure. Air in chamber C is then vented out through the port at point 5. Since there is no pressure in chamber C, the pressure difference between chambers A and B forces the actuator piston to the right and opens the shutoff valve. So long as the rate sensor remains out of the picture, pressure A will be greater than pressure B and the valve will stay open.

However, as cockpit pressure increases, the rate sensor senses the increase and drains air from point 2. This air, which is no longer available for chamber A, is vented from the system at the rate sensor. Since its supply pressure is decreased, chamber A pressure drops until chamber B pressure exceeds it. The piston is then forced to the left and the shutoff valve closes. As cockpit pressure stabilizes, the rate sensor stops draining air from point 2 and the pressure in chamber A builds up until it exceeds chamber B pressure and the shutoff valve again opens. This "open-shut-open" cycle continues until the cockpit reaches its regulated pressure (5 psi over atmospheric pressure).

One important point to remember about the flow control assembly is that the shutoff valve is open only when the solenoid is energized. This means that when the current to the solenoid is shut off, chamber C will be pressurized and will cause the shutoff valve to close.

This is a "fail-safe" feature which means that a power failure would automatically close the shutoff valve and prevent possible damage to the air-conditioning system and overheating of the cockpit.

Proper operation of the flow control assembly depends upon very precise adjustments. Never attempt to repair or adjust a defective assembly, but replace it with a new one. One problem that you can handle yourself concerns the manner in which the rate sensor line connects to the actuator. A standard pressure fitting is used; however, you must be very careful to make sure that no leaks are allowed. Rate sensor connections are discussed more fully in Chapter III.

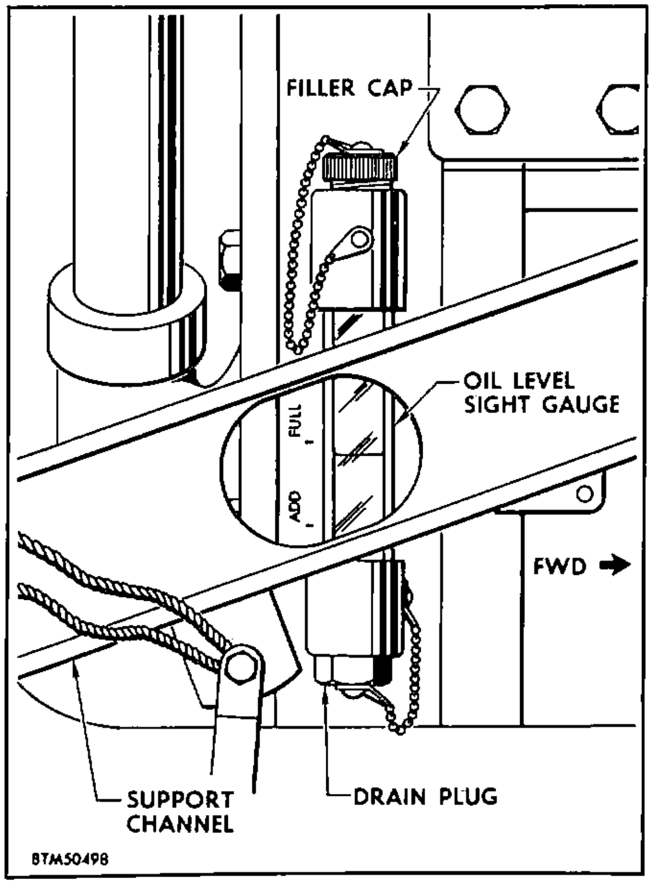


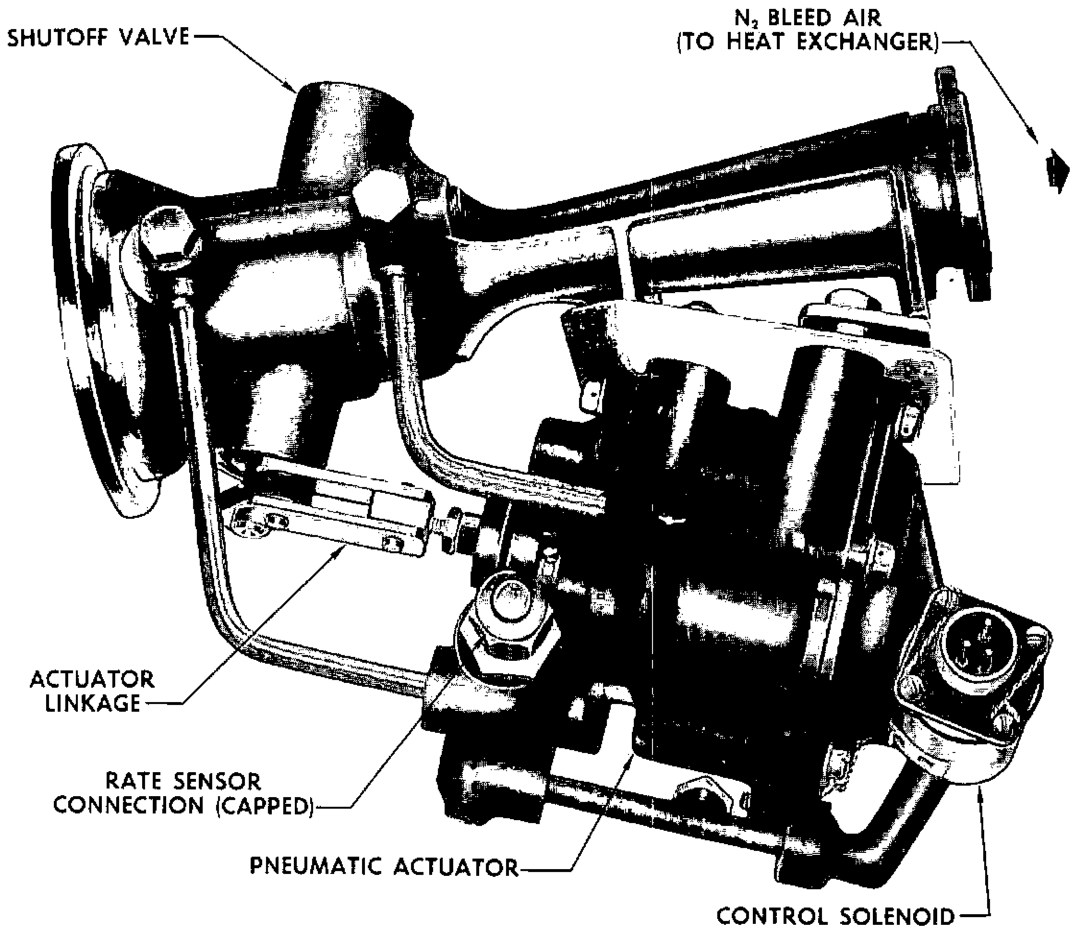
Figure 2-15. Expansion Turbine Oil Sight Gage

The principle of electromagnets is applied to the simple solenoid shown in figure 2-19. If a metal rod or core is partially inserted in the energized solenoid coil, the magnetic field will force the core to the center of the coil. In the deenergized solenoid (switch open), the spring pulls the rod from the coil.

In the flow-control valve solenoid, the piston moves the valve which controls air pressure to chamber C of the pneumatic actuator, when current flows in the coil. The flow of current is controlled by a rather complicated electrical circuit, discussed in the next chapter.

Bypass Valve.

The motor-operated bypass valve is a very important part of the cockpit temperature control system. When it receives a signal from the cockpit temperature control system, the reversible d-c motor moves the valve butterfly shutter to regulate the flow of bleed air through the valve. Since the motor is reversible, it can move the valve in either direction and stop it at any desired position. The rate of air flow through the valve is further controlled by a flow limiter on the downstream side of the valve. This flow limiter has an opening of slightly less than one-half inch. No adjustment of the valve should be attempted. Replace it with a new unit if it is defective.



FLOW CONTROL ASSEMBLY

8TM50488

Figure 2-16. Flow Control Assembly

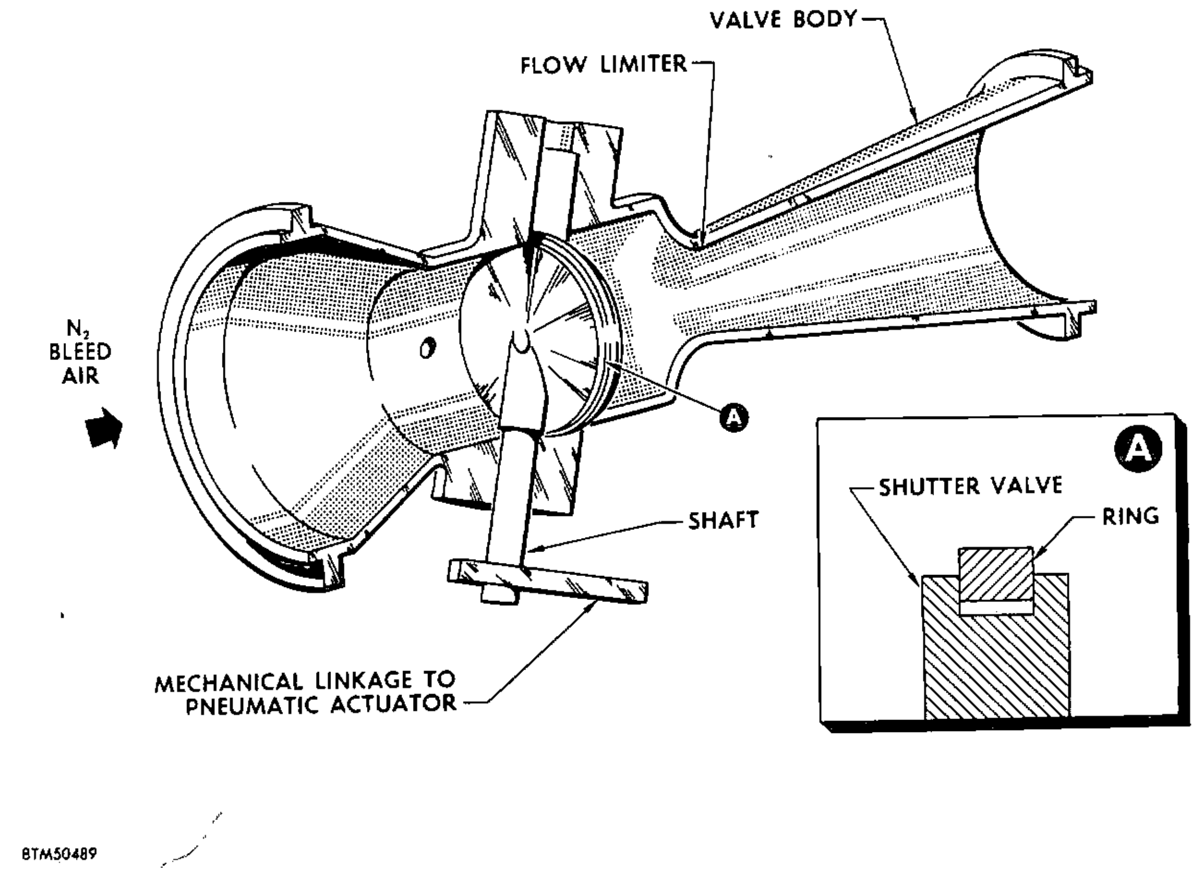


Figure 2-17. Flow Control Shutoff Valve Cutaway

RAM AIR SUPPLY.

As the airplane moves forward in flight, air is scooped or "rammed" into the air intakes and distributed by ducts to various sections of the airplane. Most of this air enters the engine where it is compressed and used for combustion or "bled" off as a supply of hot compressed air for various parts of the low-pressure pneumatic system. The rest of the ram air is used to cool and ventilate various electronic compartments or electrical components.

All ram air enters the airplane through four openings or intakes; two of these openings are engine intakes and the other two are called boundary-layer intakes. The two engine intakes are shown in figure 2-20. These intakes are located on each side of the fuselage near the forward part of the cockpit and are readily visible on the aircraft.

The boundary-layer intakes, however, are not so easily seen. One of these is between each engine intake and the fuselage. The two boundary-layer duct systems are connected through a short duct section and a check valve. You can see in figure 2-20 how the two engine intakes join just forward of the engine to form one large engine intake duct. Some air from this common duct is drawn off to be used for ventilation and cooling, while the rest enters the engine. All air from the two boundary-layer intakes is used for cooling and ventilation.

The temperature, pressure, and density of this ram air varies over a wide range and depends on weather, altitude, and airplane speed. The manner in which ram air temperatures, pressure, and rate of flow varies was discussed earlier

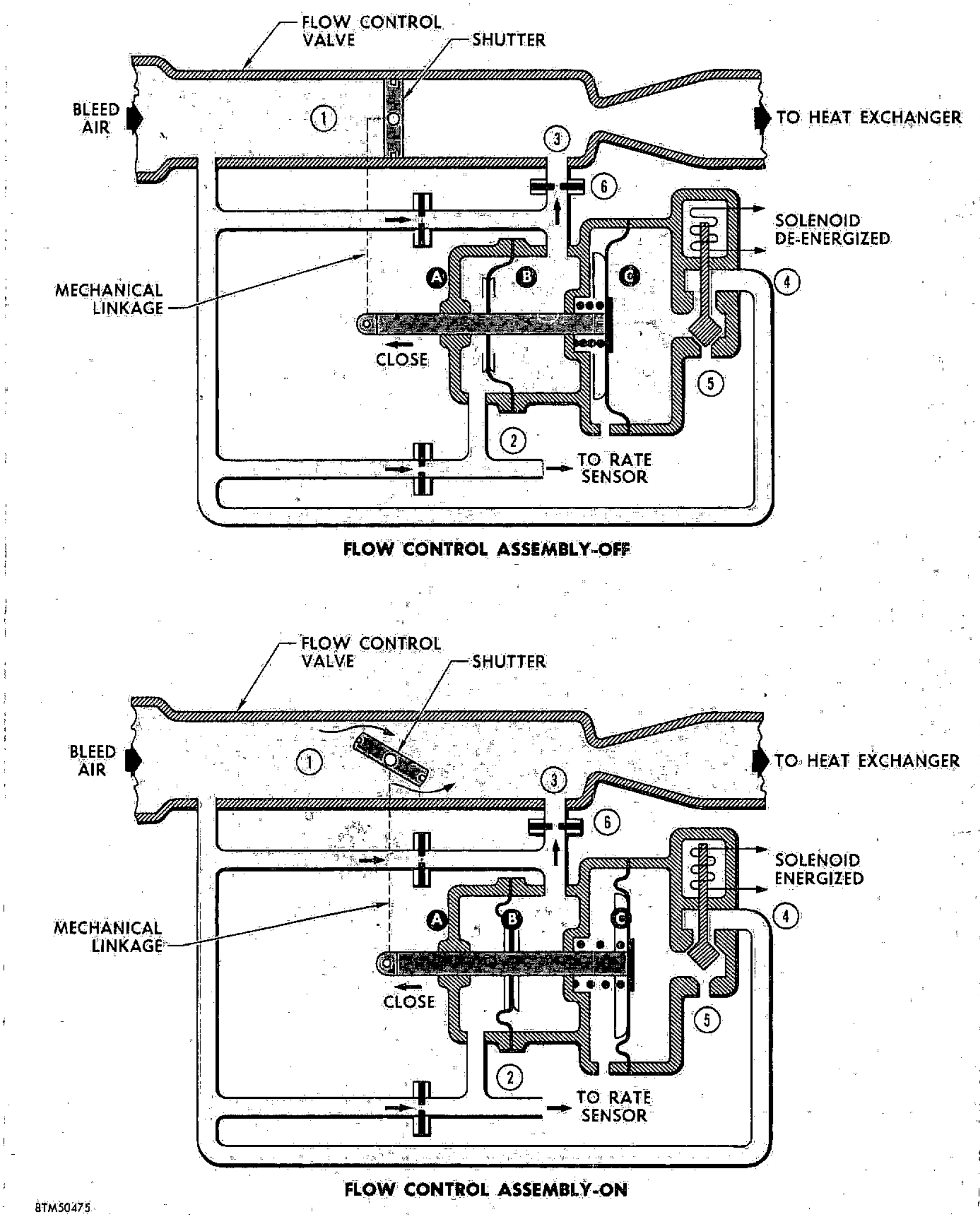


Figure 2-18. Flow Control Assembly Schematic

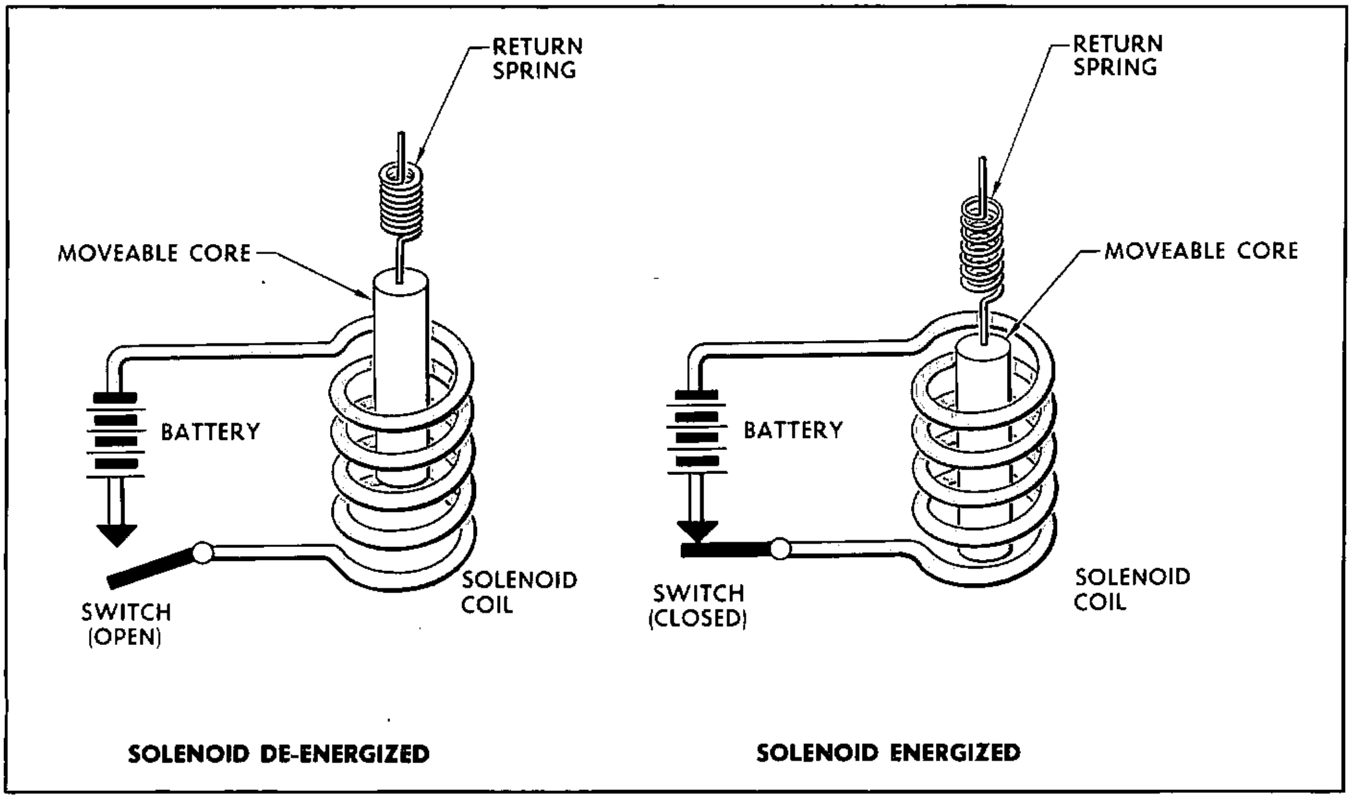


Figure 2-19. A Simple Solenoid

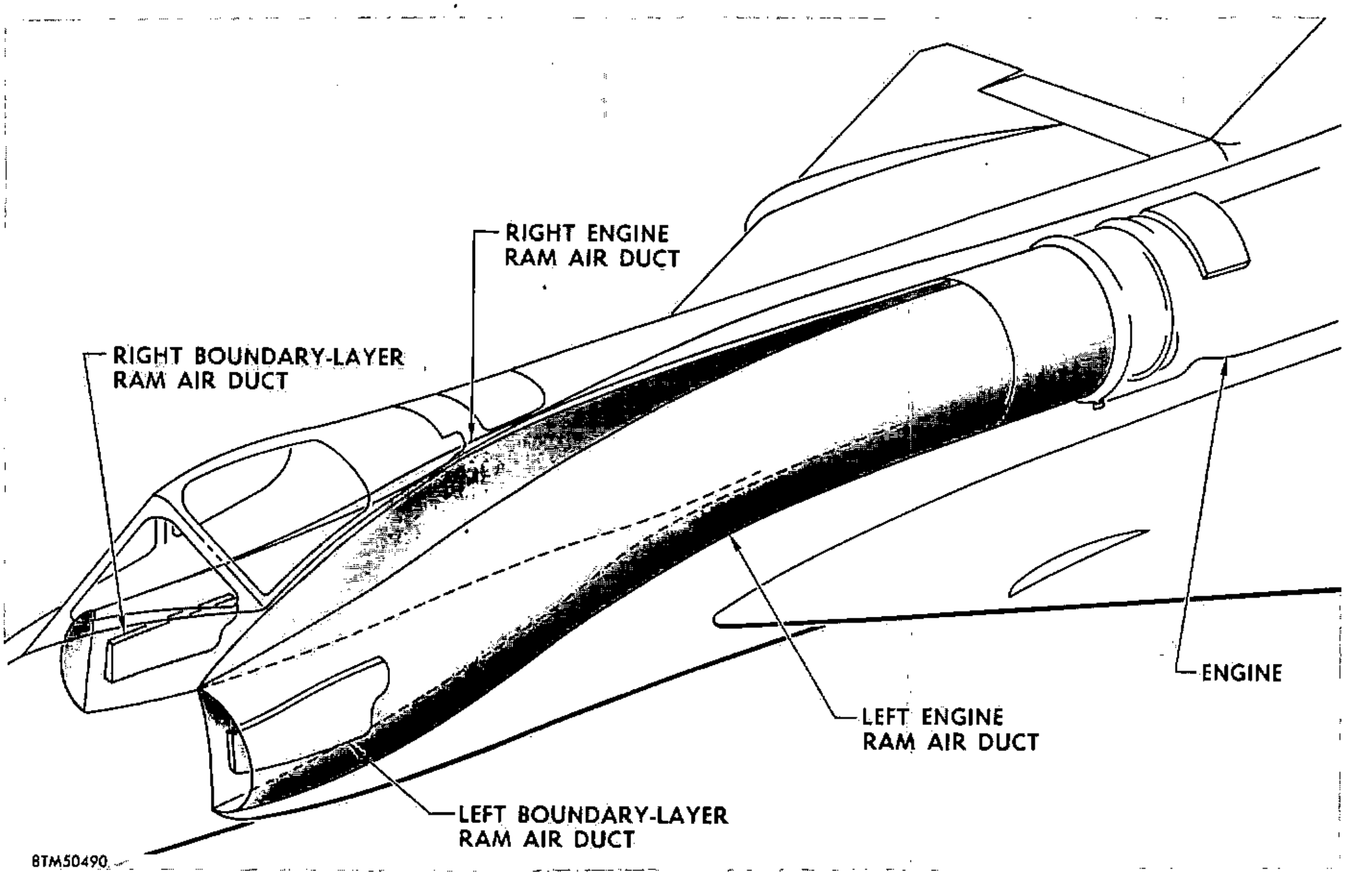


Figure 2-20. Ram Air Intake Sources

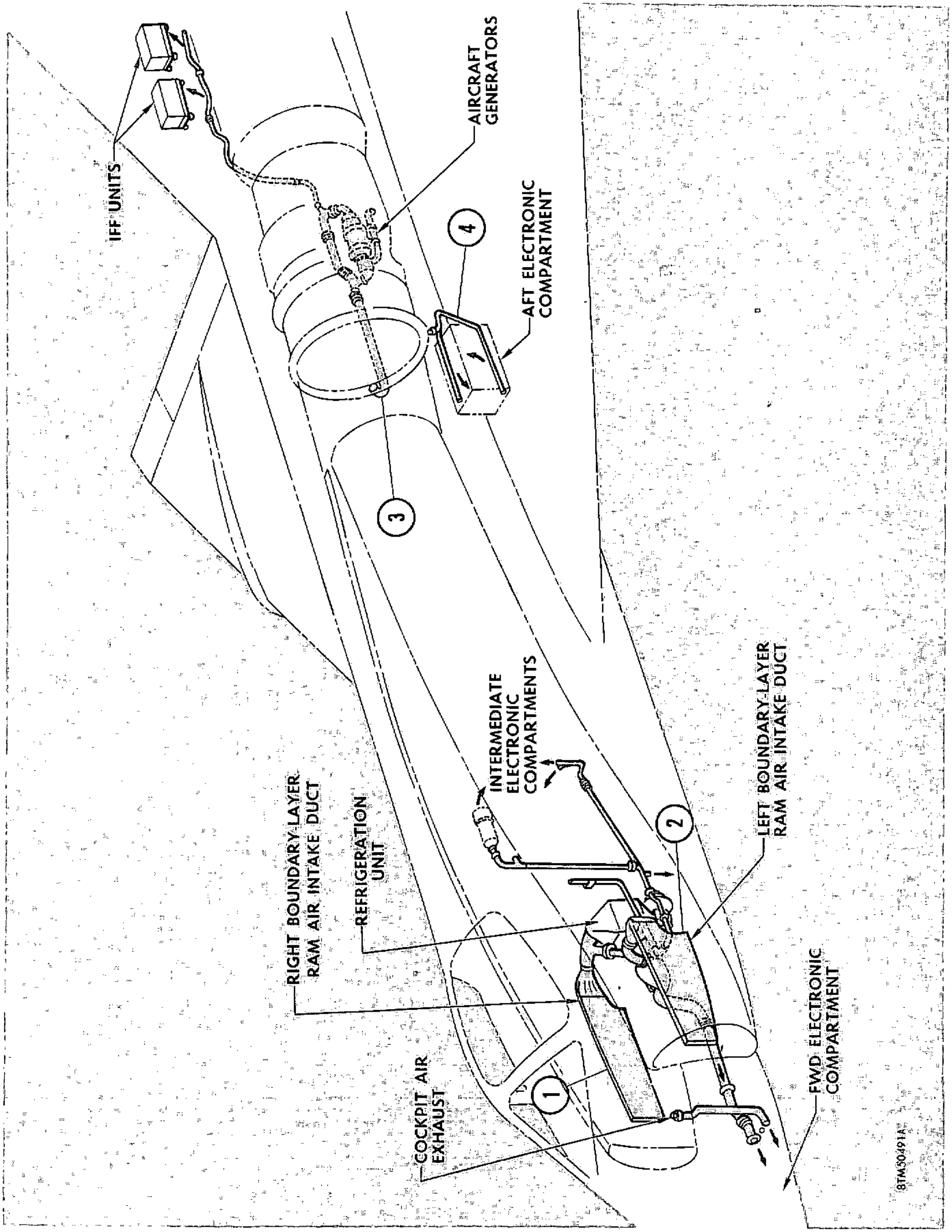


Figure 2-21. Ram Air Distribution Ducts

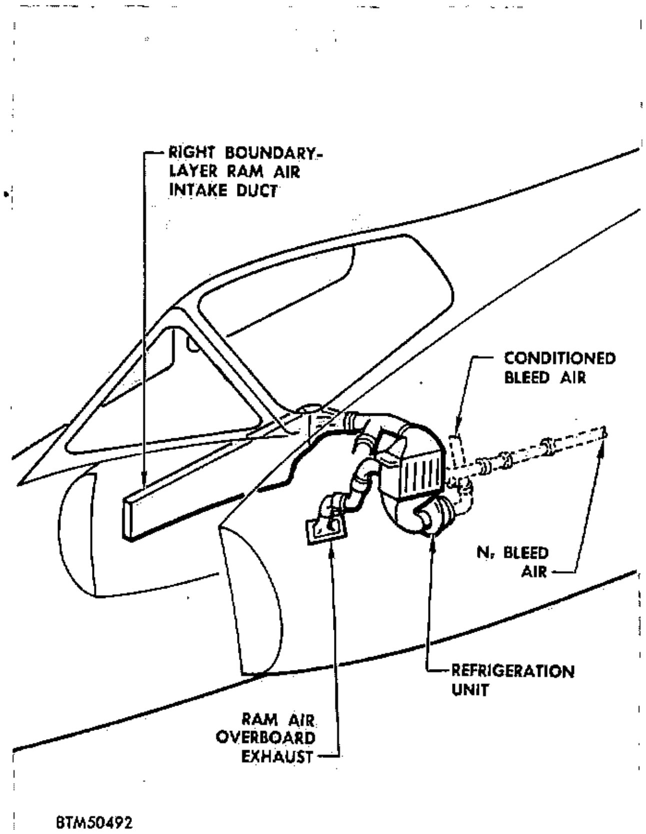


Figure 2-22. Right Boundary-Layer Ram Air Ducting

The supply of cooling air is not entirely dependent on the forward motion of the airplane, since there are several provisions for drawing cooling air into the system through the ram air ducts when the airplane is stationary on the ground. The blower fan in the refrigeration unit draws air in through the right boundary-layer intake to furnish cooling air to the heat exchanger. A jet pump in the left boundary ram air system creates a reverse flow of cooling air through the distribution ducts connected to the left boundary-layer intake.

HOW RAM AIR IS DISTRIBUTED.

Ram air entering the boundary-layer air intakes is distributed to various parts of the airplane by a system of noninsulated aluminum ducting. There are really four separate duct systems besides the large engine air intake ducts, as seen in figure 2-21. Note that two front duct systems receive air from each of the boundary-layer intakes and the two aft systems tap air from the forward edge of the engine just in front of the engine compressor.

In figure 2-22, you can see that the right boundary-layer ducting supplies only the heat exchanger with cooling air. After passing through the heat exchanger and cooling the engine bleed air, this boundary-layer air is then vented overboard through exhaust ducting. The left boundary-layer ducting, shown in figure 2-23, distributes cooling air to the forward, intermediate, and upper electronic compartments and is also connected through a shutoff valve to the cockpit conditioned air ducting.

One aft duct system taps air from the engine intake and conducts it through the a-c and d-c generators, after which most of the air is vented into the engine accessory compartment. The remainder is conducted into the vertical fin to ventilate the IFF unit. The other aft system feeds cooling air into the aft electronic compartment. The two boundary-layer duct systems are connected through a short duct section and a check valve.

Ram air pressure and temperature are only slightly higher than atmospheric pressure and temperature; therefore, ram air ducting presents very few maintenance problems as compared to bleed air ducting. This is because ducts expand very little, and also minor air leaks are not too important; thus the maintenance of duct sections is not a problem.

FLOW CONTROL OF BOUNDARY-LAYER RAM AIR.

In comparison to the bleed air system, flow control of ram air is relatively simple. There are no valves or

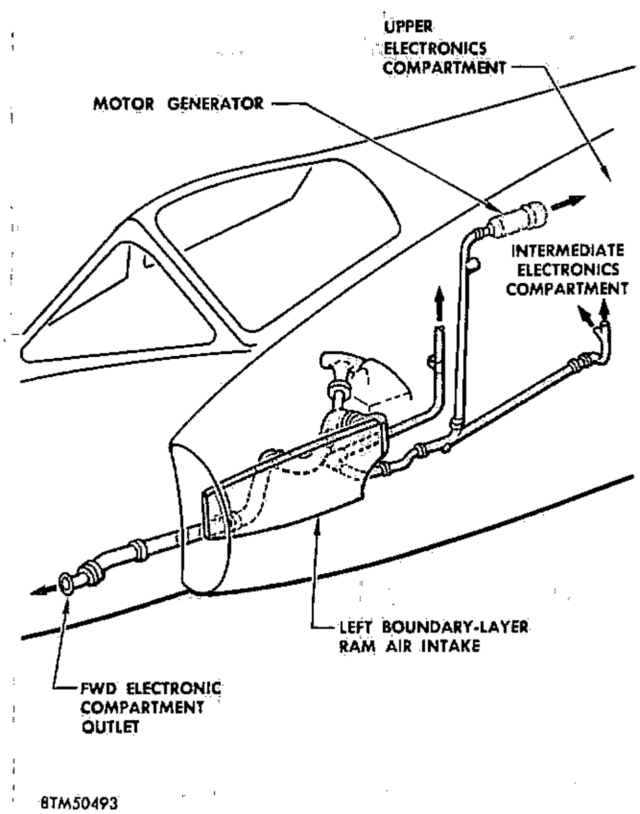


Figure 2-23. Left Boundary-Layer Ram Air Ducting

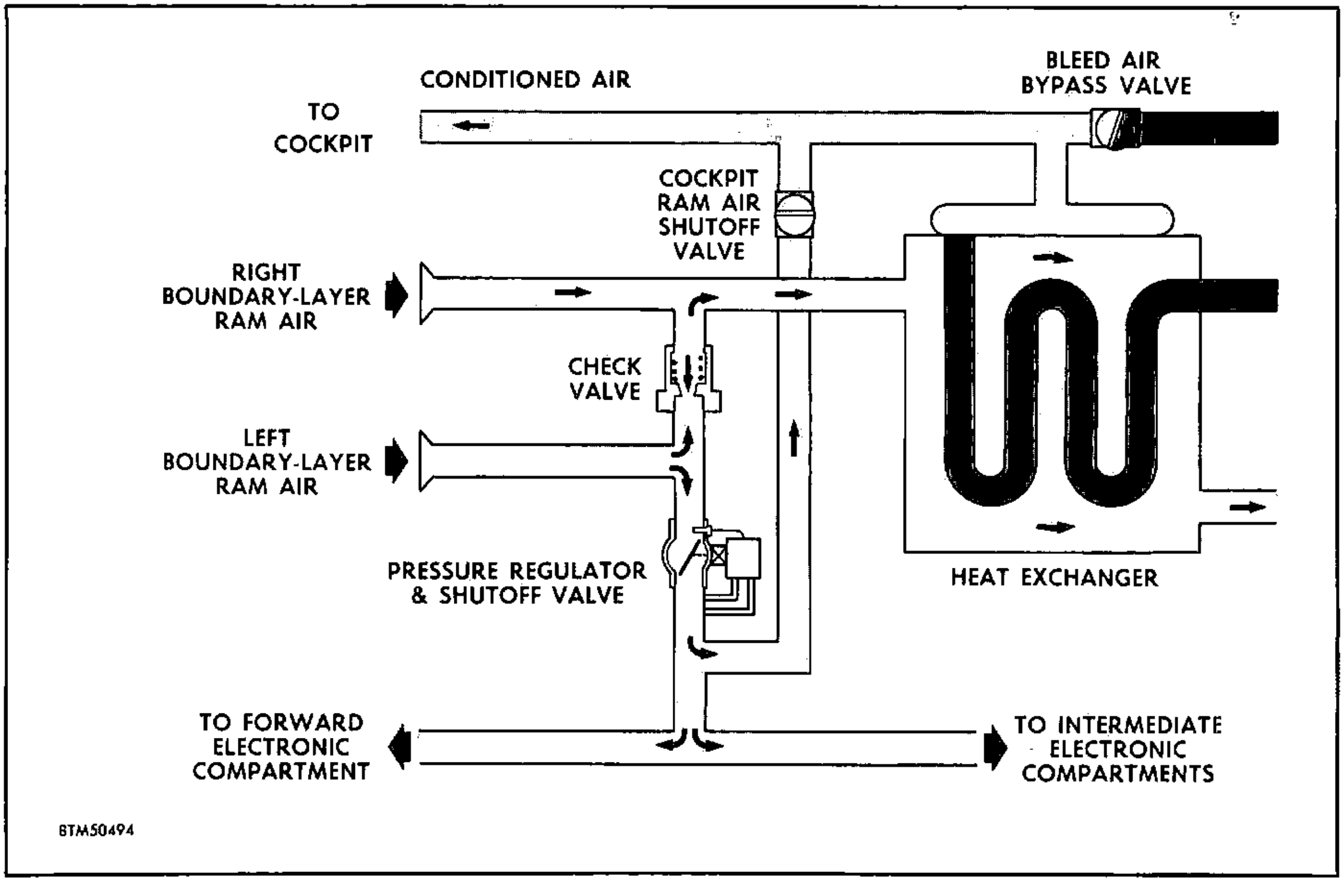


Figure 2-24. Ram Air Flow Control System Schematic

regulators in three of the four ram air systems, and ram air flow depends only on the forward motion of the airplane. The two boundary-layer ram air systems are shown in figure 2-24.

The ducting for the right boundary-layer ram air controls the flow of air through the heat exchanger and exhausts it overboard. The ducting connected to the left boundary-layer intake contains a combination pressure regulator and temperature controlled shutoff valve. In addition, there is a cockpit ram air shutoff valve in the duct leading from the left boundary-layer intake ducting to the cockpit conditioned air ducting. There is also a check valve in the short duct section that connects the left and right boundary-layer ducting.

The combination pressure regulator and temperature controlled shutoff valve shown in the diagram is a very important part of the airplane ventilation system. This butterfly-type valve is motor operated and solenoid controlled. This unit opens and shuts on an electrical signal from an electrical air control system.

A temperature-sensing element in the ducting upstream from the valve causes the valve to close when the intake-air temperature exceeds about 71°C (160°F). Pressure-sensing elements downstream from the valve control the position of the valve to prevent downstream pressure from being more than approximately one psi over atmospheric pressure.

The cockpit ram air shutoff valve leading to the cockpit conditioned-air ducting is part of the cockpit air-conditioning and pressurization system. It is a motor operated valve controlled by a switch in the cockpit. A landing-gear safety switch causes the valve to close automatically whenever the airplane is on the ground.

The check valve in the duct section connecting the right boundary-layer ducting to the left boundary-layer ducting is shown in its installed position (figure 2-25). It is a simple mechanical double-flap type valve as you can see in detail A. The valve opens in one direction only; that is, air is permitted to flow from the left ducting to the right ducting, but not in reverse. When the ram air pressure regulator and shutoff valve shuts off the ram air flow to the left ducting, the flow is diverted through the check valve to the right ducting. This increases the flow of cooling ram air to the heat exchanger.

Even when the shutoff valve is open, the flow across it is usually restricted to keep the downstream pressure from becoming too high. The remaining available air from the left boundary-layer intake passes through the check valve to increase the flow of cooling air to the

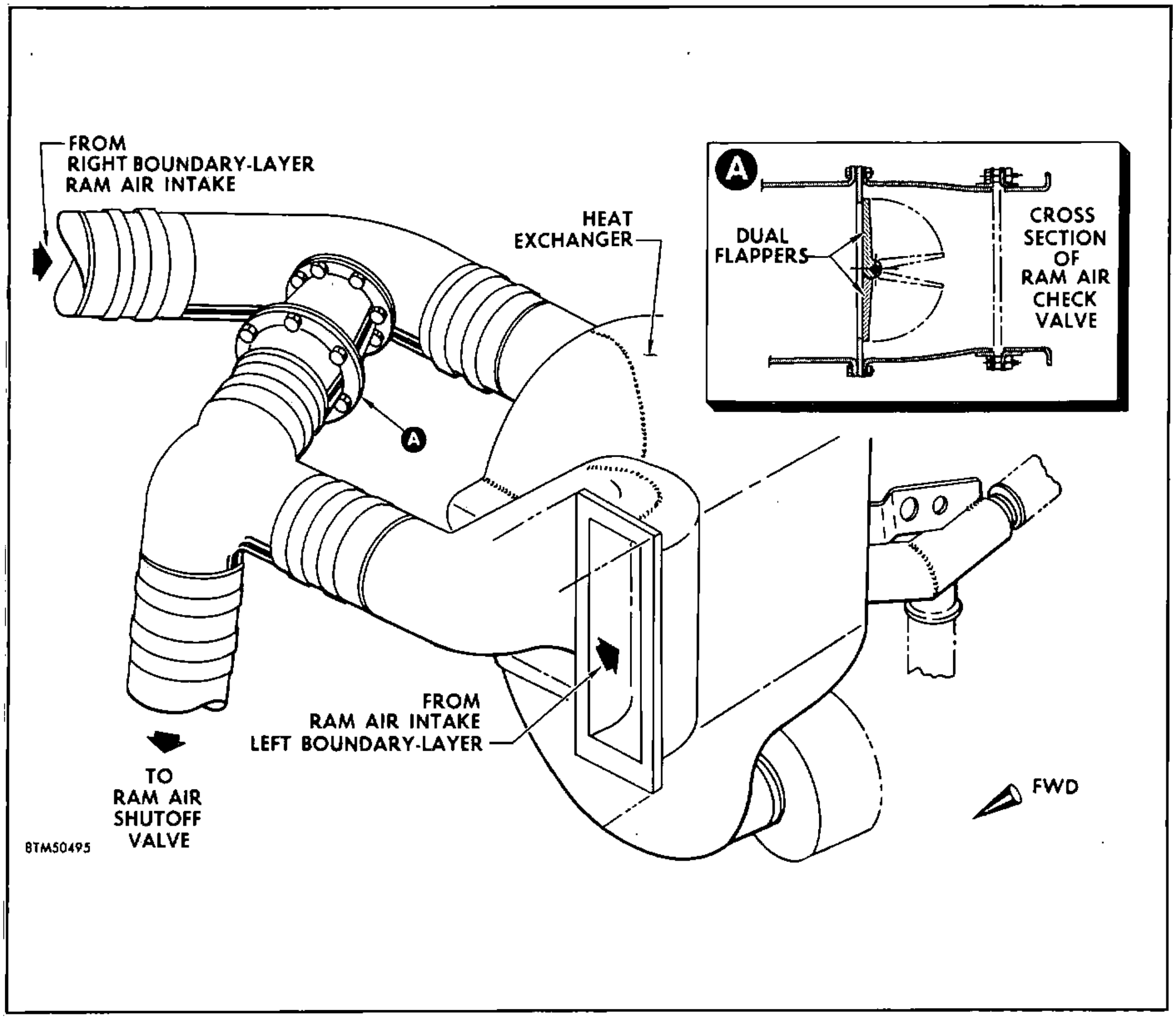


Figure 2-25. Ram Air Check Valve and Ducting

heat exchanger. Since the valve opens in one direction only, air cannot flow from the right intake ducting to the left.

This valve is not likely to cause maintenance difficulties. Even if it should malfunction, it will not greatly affect the operation of the ventilation and cooling systems.

SUMMARY.

In this chapter you have learned the two sources of air used in the low-pressure pneumatic system—engine bleed air and ram air. We have followed the bleed air from the engine compressor through its insulated ducts to the refrigeration unit. We have also seen how ram air from the right boundary-layer ram air duct cools this engine bleed air in the refrigeration unit.

In addition you learned that the source of cooling air for the various electrical and electronic units comes from the left boundary-layer ram air duct and from two "tap-offs" in the engine air intake duct system. In succeeding chapters we will follow the various subsystems that use the straight engine bleed air, partially conditioned air, and ram air which you learned about in this chapter.

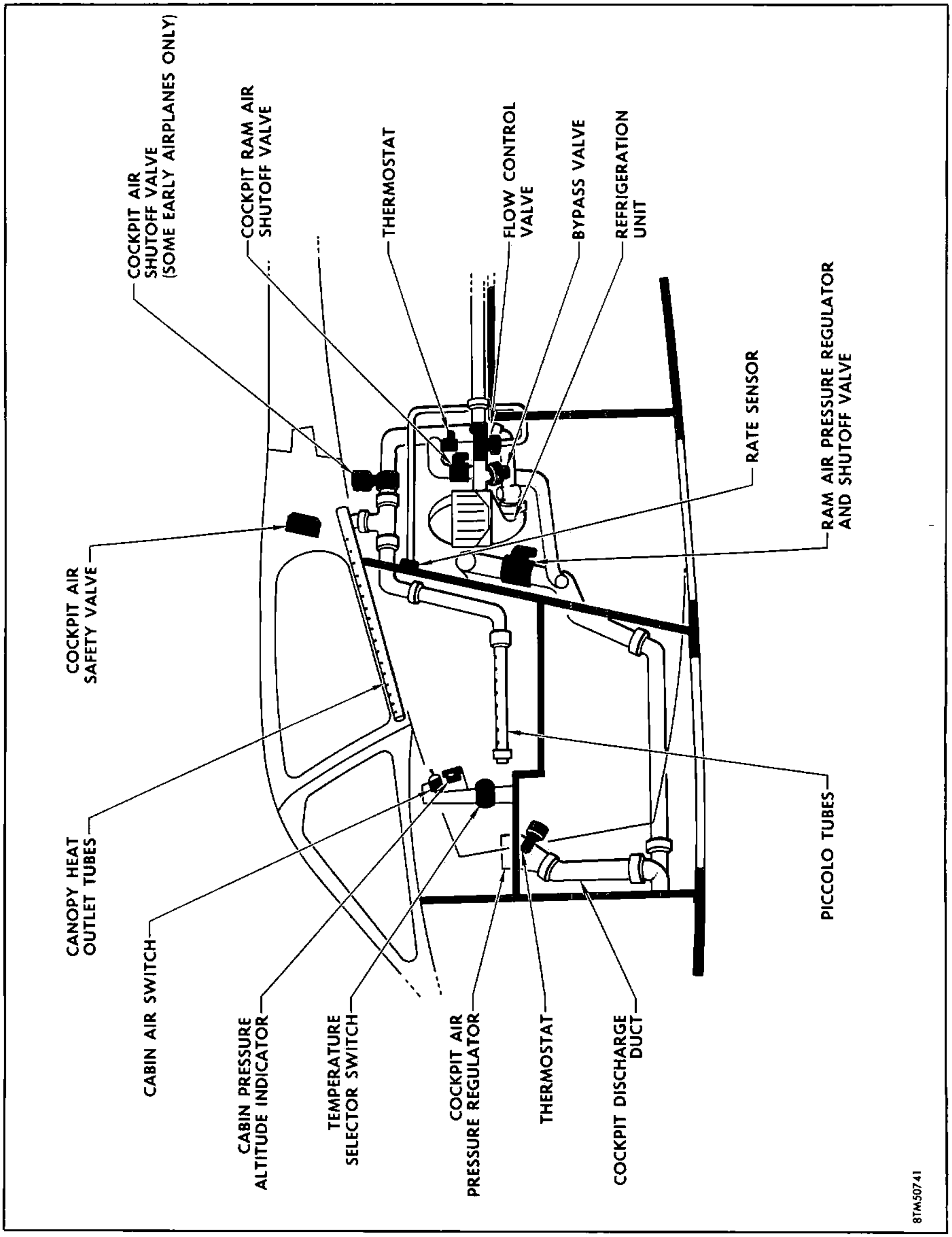


Figure 3-1. Cockpit Air Conditioning and Pressurization System

Chapter III

COCKPIT AIR-CONDITIONING AND PRESSURIZATION SYSTEM

Contents *Page*

In Chapter I of this manual you learned about the cockpit air-conditioning and pressurization system in a general way and without the details. The air sources and distribution systems and the refrigeration unit were discussed in detail in Chapter II. In this chapter, you will learn about cockpit air-conditioning and pressurization, how it operates under various conditions, and what you must do to keep it operating.

Before we get involved in a detailed discussion of the different components and control systems, it is best that we first fix in our minds how the system operates, both while the airplane is on the ground and during normal flight, and under other special conditions. Individual components and controls will be fully discussed in this chapter.

DESCRIPTION OF SYSTEM.

You should remember from previous chapters that the cockpit is supplied with conditioned bleed air from the refrigeration unit whenever the engine is running and the flow control valve is open. You can see in figure 3-1 that conditioned air is conducted to the cockpit by ducting and is vented from two "piccolo" tubes, one on each side of the cockpit floor. Conditioned air also enters the cockpit from two canopy-heat outlet tubes, one along each edge of the canopy. A duct section containing the cockpit ram air shutoff valve connects the left boundary-layer ram-air ducting to the cockpit ducting. When the pilot wishes, ram air can be used in place of conditioned air to ventilate the cockpit.

A cabin air switch (figure 3-2) on the left auxiliary instrument panel controls the air flow. When the switch is in PRESS, the flow control valve is open, the cockpit ram air valve is closed, and conditioned air enters the cockpit. When the switch is in RAM position, the flow control valve is closed, the cockpit ram air valve is open, and ram air enters the cockpit (if the airplane is in flight).

If the switch is OFF, both valves are closed and all cockpit air is shut off. Cockpit air flows from the cockpit through a discharge duct to the forward electronic compartment where it helps to cool and ventilate the electronic equipment. Cockpit pressure is controlled by the cockpit air pressure regulator in the cockpit discharge duct. The cockpit air safety valve relieves cockpit pressure by venting air overboard if the pressure regulator becomes defective.

A cabin pressure altitude indicator near the cabin air switch shows the cockpit pressure in terms of the equivalent altitude in thousands of feet. Cockpit temperature is regulated by an automatic temperature control system consisting of a heat control box, a thermostat in the discharge ducting, and another in the cockpit supply ducting. The thermostats send electric signals to the control box to indicate air temperatures, and the control box controls the position of the bypass valve. The position of the bypass valve determines the amount of hot bleed air to bypass the refrigeration unit and mix with refrigerated air.

The proportion of bypass air to refrigerated air determines the temperature of the air entering the cockpit. A cabin temperature control switch on the utility switch panel (figure 3-3) allows the pilot to select any temperature he desires between 4°C (40°F) and 38°C (100°F).

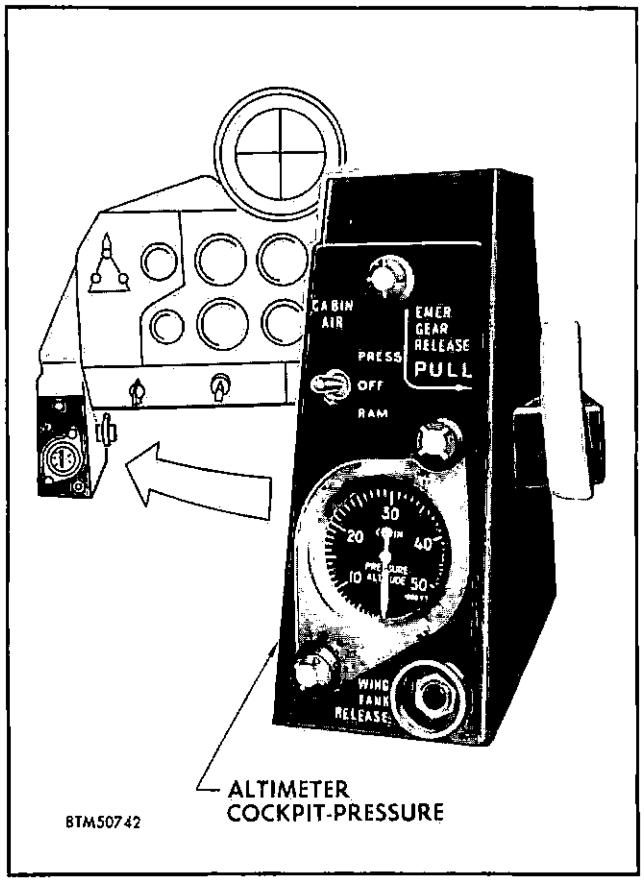


Figure 3-2. Cabin Air Switch and Pressure Altitude Indicator

NORMAL OPERATION IN FLIGHT.

As previously noted, conditioned air will flow into the cockpit any time the engine is running and the flow control valve is open. The temperature of this conditioned air is automatically held by the temperature control system at the temperature selected by the pilot. The pressure regulator in the cockpit discharge duct controls the flow of air from the cockpit to the forward electronic compartment.

Below 10,000 feet altitude the flow is unrestricted; therefore, the cockpit is unpressurized. Above 10,000 feet the flow of cockpit discharge air is restricted in order to maintain proper cockpit pressurization. On the ground, air can flow through the discharge duct in either direction. However, when the airplane is airborne, air can flow in one direction only; that is, from the cockpit to the forward electronics compartment.

The cockpit safety valve will remain closed in flight unless the pressure regulator is defective and the cockpit pressure becomes too high. In that case, the safety valve will open and vent some cockpit air overboard to reduce the pressure to a safe level. The safety valve automatically opens on the ground. Opening of the safety valve on the ground is controlled by the main landing gear safety switch which causes the valve to open when the weight of the airplane is on the landing gear.

RAM AIR VENTILATION.

Ram air can be used in place of conditioned bleed air to ventilate the cockpit whenever the pilot desires. If the refrigeration unit or the temperature control system becomes defective or if some other emergency occurs, the pilot closes the flow control valve and opens the cockpit ram air valve by moving the cabin air switch to RAM. At the same time the air safety valve opens to vent the cockpit which will then be completely unpressurized.

Ram air from the left boundary-layer intake duct enters the cockpit through the cockpit ducting and leaves it through the air safety valve. Little, if any, ram air will leave the cockpit through the cockpit pressure regulator. Ram air is not available for cockpit ventilation on the ground, even when the airplane is moving, since the cockpit ram-air shutoff valve is connected to the main landing gear safety switch. This switch closes the valve whenever the weight of the airplane is on its landing gear.

When the pilot again wishes to use conditioned bleed air for cockpit air conditioning, he moves the cabin air switch to PRESS. The flow control valve then opens and the safety valve closes. If the flow of bleed air were unrestricted, cockpit pressure would increase much too fast for pilot comfort. The rate sensor senses the rate of cockpit pressure buildup and regulates the opening of the flow control valve to keep the rate at a comfortable level. Cockpit pressure must not be allowed to increase faster than 4 psi per minute.

OPERATION DURING ARMAMENT FIRING.

The burning of fuel in the motor of a rocket or missile creates a large cloud of highly corrosive dust and debris. If this dust were allowed to enter the cockpit or the electronic compartments, serious damage could result. Perhaps the most immediate danger is the possibility that dust in the cockpit might reduce the pilot's visibility at a critical moment.

These dangers are avoided in the F-102A by means of an air control timer in the armament system. When the armament bay doors open, a door actuated switch sends a signal to the air control timer. The timer in turn sends signals to both the bleed air flow control valve and the ram air pressure regulator and shutoff valve, causing them both to close. This cuts off all air to both the cockpit and the electronic compartments except the aft electronic compartment and the IFF unit.

After the armament is fired and the doors close, the switch again signals the timer. Three or four seconds

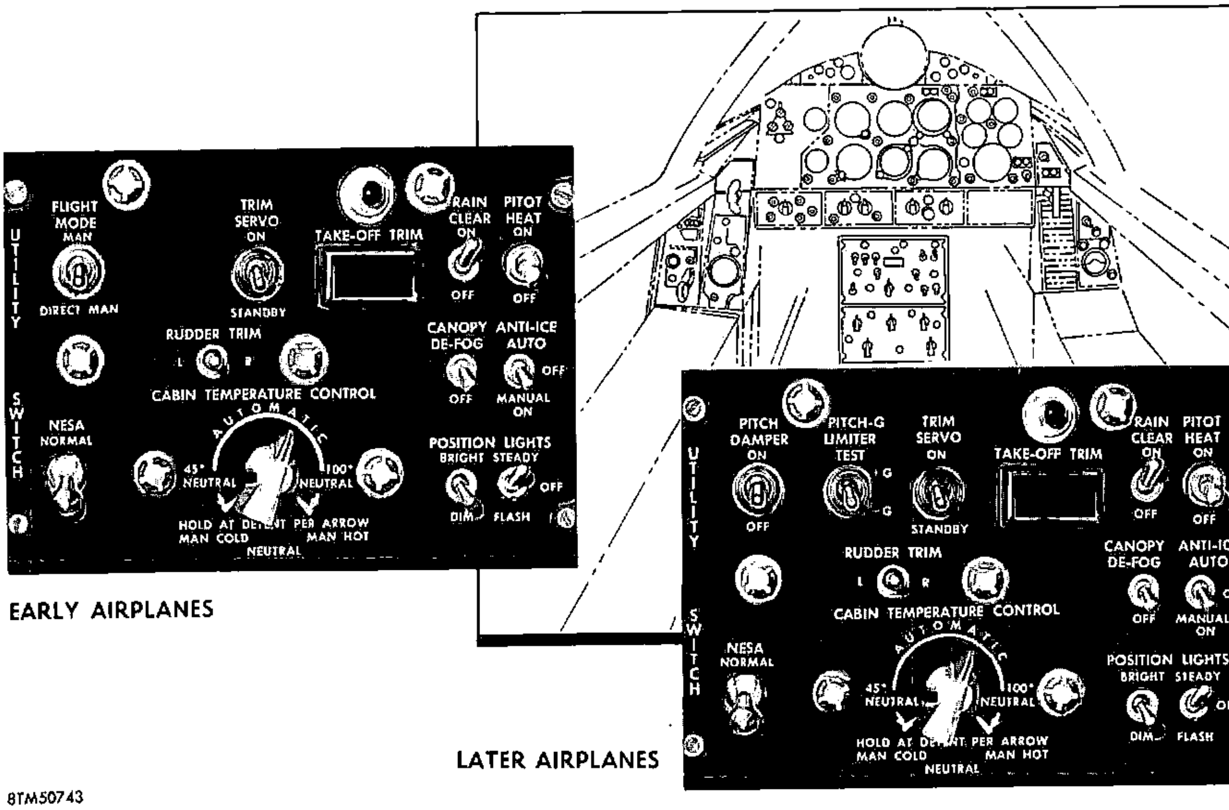


Figure 3-3. Cabin Temperature Control Switch

after receiving the signal, the timer signals the flow control valve and the ram air pressure regulator and shutoff valve. The pressure regulator and shutoff valve always opens; however, the flow control valve opens only if the cabin air switch is at PRESS. Since the flow control valve had shut off all air to the heat exchanger, those subsystems that tap air from the heat exchanger will also be shut off.

In addition, the canopy seal, which also uses air from the heat exchanger, is not pressurized during armament firing and cockpit air leaks overboard, thus causing cockpit pressure to drop. When the flow control valve is again opened, the rate sensor regulates its opening to keep the cockpit pressure from building up too fast.

OPERATION ON THE GROUND.

The cockpit air-conditioning system functions in the same manner on the ground, when the engine is running, as it does when the airplane is airborne. The only difference is that the cooling capacity of the heat exchanger will be lower and a minimum amount of bleed air will mix with conditioned air. The cockpit is not pressurized on the ground.

When the engine is not running, a ground air-conditioning unit can be attached to a fitting in the nose wheel well. This ground unit furnishes conditioned air which is distributed to the forward, intermediate, and upper electronic compartments. Some of this conditioned air flows through the cockpit discharge duct to air-condition the cockpit. If the canopy is closed, this air leaves the cockpit through the safety valve.

COCKPIT AIR DISTRIBUTION SYSTEM.

Conditioned air from the refrigeration unit flows to the cockpit through uninsulated aluminum ducting. The main duct sections are about two inches in diameter and are joined in the same manner as the hot bleed air ducting described in Chapter II. As you can see in figure 3-4, the cockpit air ducting connects to the refrigeration unit at the expansion turbine exhaust. The ducting then travels forward and into the cockpit where air is vented through holes in two "piccolo" tubes.

A piccolo tube is a section of aluminum ducting with a row of outlet holes. One tube is along each side of the cockpit floor. As you can see, there are other cockpit air outlets along each edge of the canopy. These outlets are called canopy-heat outlets and are built into the canopy. The canopy-heat tubing is connected to the cockpit air ducting only when the canopy is closed. Aluminum ducting from the left boundary-layer ram air distribution system connects to the cockpit air ducting near the refrigeration unit.

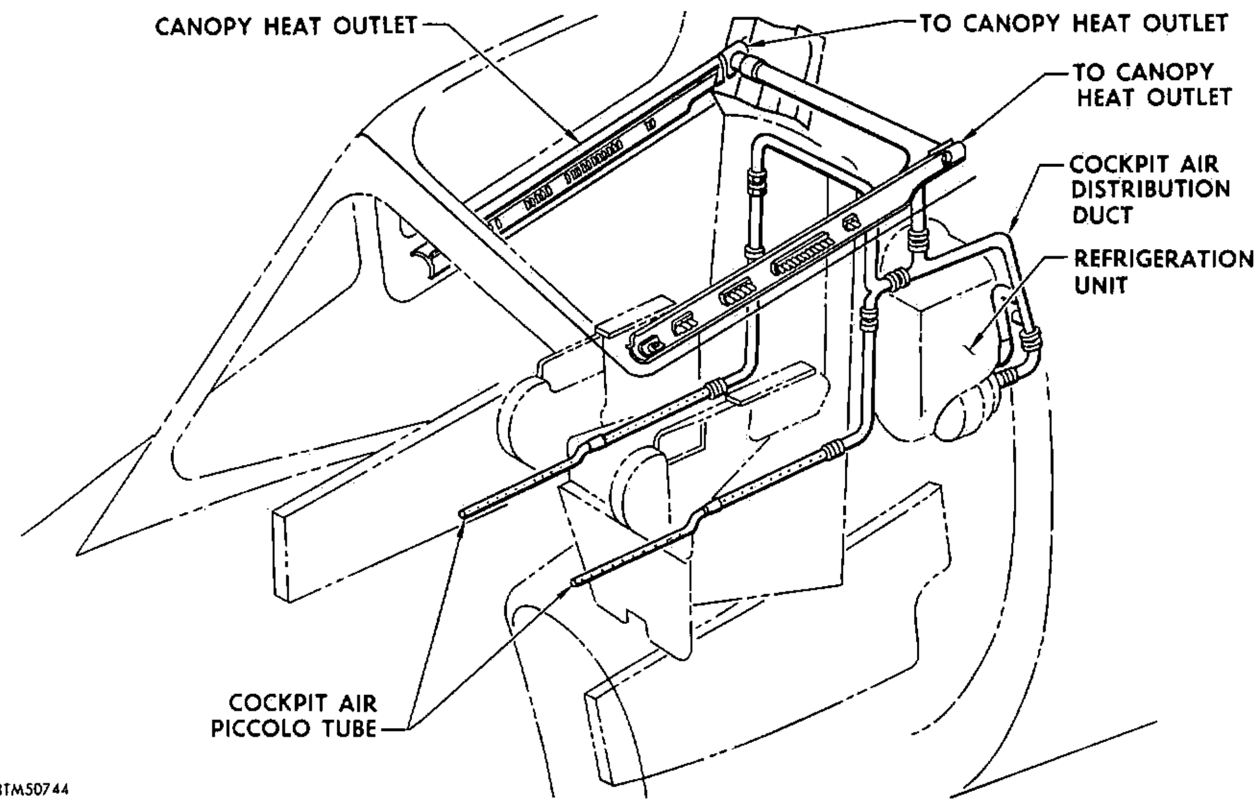


Figure 3-4. Cockpit Air Distribution System

When the pilot desires, ram air can be used in place of conditioned air for cockpit ventilation. Cockpit air ducting should give little trouble and minor leaks between duct sections are unimportant. You must be careful, however, not to block any of the piccolo tube outlets.

CONTROL SYSTEMS.

Three different but intermeshing systems control the operation of the cockpit air-conditioning and pressurization system. They are the flow control system, the temperature control system, and the pressure control system. Since some components affect both air flow and air temperature or pressure, it is sometimes difficult to separate their different functions. In this section we will first completely describe each system and its overall operation, and then we will describe the individual components. Some components will be discussed under more than one control system.

The cockpit air flow is controlled by a system of valves and regulators. To simplify the discussion we will divide the control system into two parts. One part controls the flow of air into the cockpit and the other part controls the air leaving it. Remember throughout the following discussion that the flow control system works in conjunction with the temperature control and pressure control systems, and at the same time.

AIR INPUT FLOW CONTROL SYSTEM.

The flow of air into the cockpit is controlled and regulated by two shutoff valves, one bypass valve, and one combination pressure regulator and shutoff valve. In addition, a cockpit air shutoff valve is installed in some

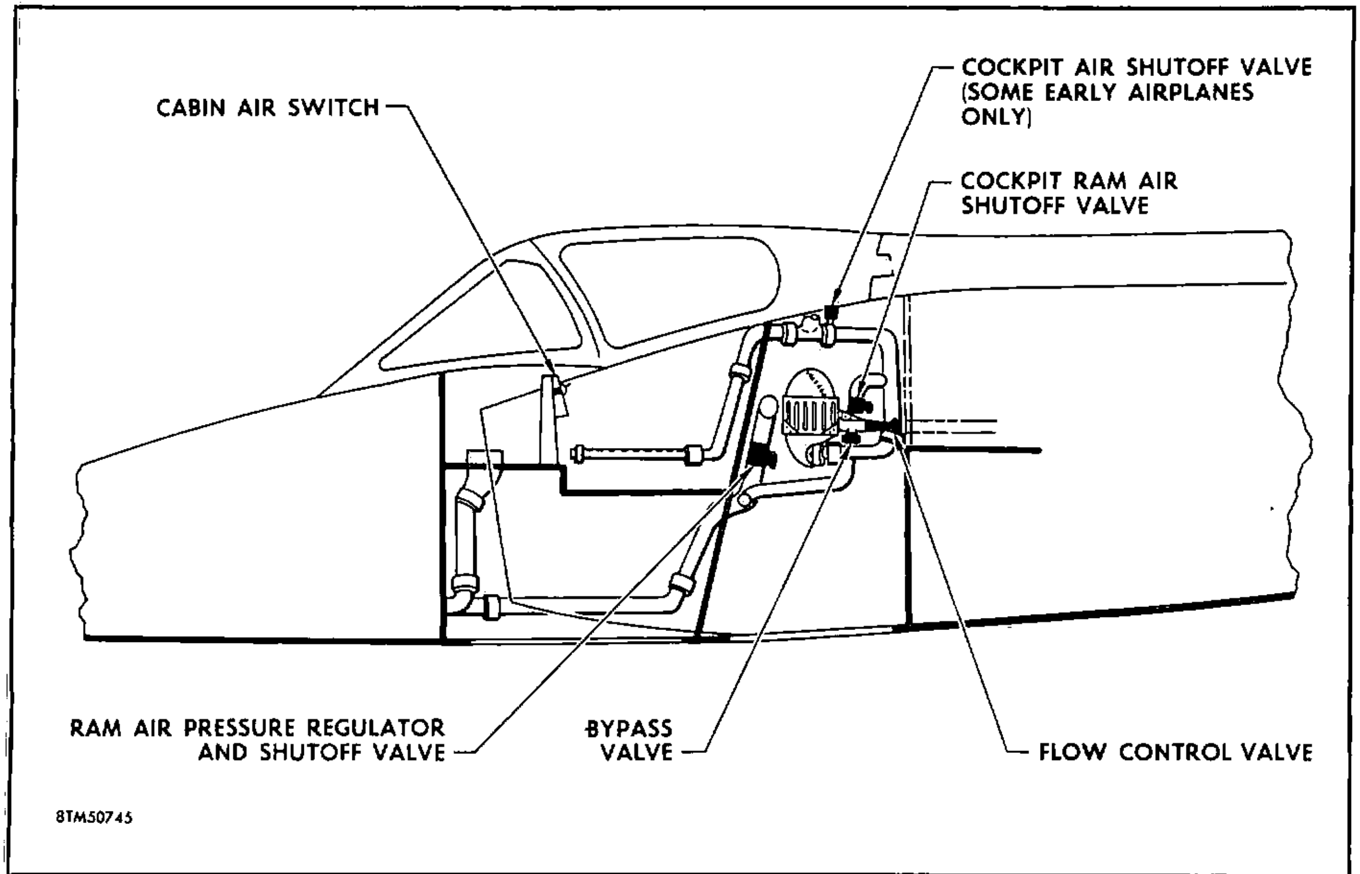


Figure 3-5. Cockpit Air Input Control System

early airplanes. This valve is electrically maintained in the open position on early airplanes, and will not be found on later ones.

The locations of these five components are shown in figure 3-5. The cockpit ram air valve and the bypass valve are controlled electrically. The flow control valve is also controlled electrically, except that the rate sensor varies the opening rate. The pressure regulator and shutoff valve is controlled in several different ways in addition to its electrical controls.

If you look at the schematic wiring diagram, figure 3-6, you will be able to follow the description of the system. The diagram shows all switches, valves, and relays in the position they would be in during normal airborne operations when conditioned bleed air is being used for cockpit air conditioning, and ram air is being used for electronic compartment ventilation. In this condition, the ram air regulator and shutoff valve and the bleed air flow control valve are both open, while the cockpit ram air valve is closed.

Ram Air Pressure Regulator and Shutoff Valve.

The most complicated single component of the control system is the ram air pressure regulator and shutoff valve. This is the main valve that controls the flow of all ram air that enters the airplane through the left boundary-layer intake. It is not only the most complicated single component, but its external controls are also the most involved.

Since ram air is used for electronic ventilation all the time, and for cockpit ventilation only part time, the unit itself will be completely discussed in the following chapter on the electronic compartment ventilation system. As a rule, the valve will be open, or partially open, whenever there is current to its solenoid and motor, and will be closed whenever the current is interrupted.

The unit itself has a complicated pressure and temperature sensing system that will interrupt the current under certain conditions. In this chapter we will cover the system controlling the flow of current to the unit. How the unit is controlled once the current reaches it will be discussed in the next chapter.

Current to the unit comes from the 28-volt d-c essential bus. If you trace the circuit in figure 3-6 you can see that current passes through three separate switches before it reaches the regulator and shutoff valve. The diagram shows all circuits in the closed position. As you can see, the current passes through the cabin air control circuit breaker to contact number 4 on the cabin air relay. It then passes through the switch to

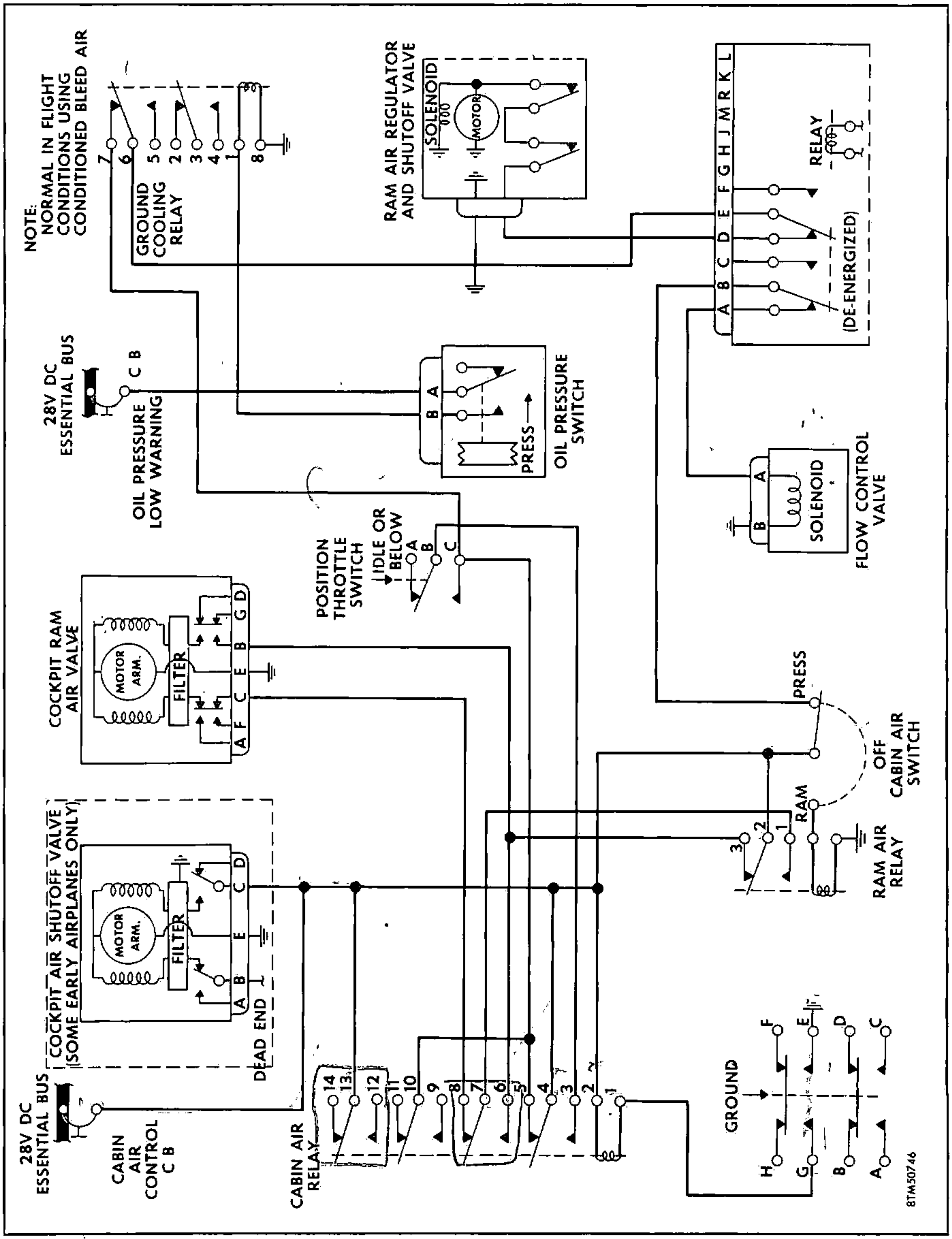


Figure 3-6. Cockpit Air Input Control Schematic

contact 5 and continues to connection C on the throttle position switch. From there it goes to contact 7 and through the switch to contact 6 on the ground cooling relay. It then passes through terminals E and D on the air control timer before continuing to connection A on the ram air regulator and shutoff valve.

The current can be interrupted by opening any one of the three switches, each of which is controlled by a relay. The cabin air relay opens the switch between contacts 4 and 5 whenever the main landing gear safety switch is closed. This safety switch is closed when the weight of the airplane is on its landing gear. The cabin air relay gets its power through the cabin air control circuit breaker when the safety switch closes the relay circuit to ground.

When the ground cooling relay is energized, it breaks the circuit by opening the switch between contacts 6 and 7. The ground cooling relay is energized when the engine oil pressure drops and allows the oil pressure switch to close and connect the relay to the 28-volt d-c essential bus through the oil-pressure-low warning circuit breaker. The third and last control switch is in the air control timer between connections E and D. This switch is opened by a relay which is energized when the armament bay doors open, and is de-energized several seconds after the doors close.

From the above discussion you can see that there will always be power at connector A on the regulator and shutoff valve whenever the airplane is in flight, there is engine oil pressure, and the armament bay doors are closed. Power will be cut off when the airplane is on the ground, when there is no engine oil pressure, or when the armament bay doors are open. There is one exception to this rule. The ram air regulator and shutoff valve must be open on the ground when the jet pump is to be used for ventilating the electronic compartments.

Since the ground safety switch prevents the regulator and valve from receiving its power through the switch between contacts 4 and 5 on the cabin air relay, other arrangements are necessary. If the engine is running, normal oil pressure, and the throttle is at IDLE, current for the ram air regulator and shutoff valve comes through the cabin air control circuit breaker to contact 4 on the cabin air relay, and through the energized switch to contact 3. From there it goes to connection B on the throttle position switch and through the switch to connection C. The current then flows to connection 7 on the ground cooling relay which will be in the position shown if there is engine oil pressure. The current then continues on through the air control timer to the regulator and shutoff valve.

From the above discussion, you can see that the regulator and shutoff valve will be *open* on the ground even though the safety switch is closed, *if* the engine is running and has normal oil pressure, and *if* the throttle is in the IDLE position.

Proper operation of this system depends on many things. The system will not operate if power is not connected to it. If there is no power, you should first check to see that the airplane electrical system is functioning, and that the 28-volt d-c essential bus is energized. Next check the cabin air circuit breaker on the aft cockpit circuit breaker panel on the left side of the cockpit. This is a push-button type of breaker that "pops" out if the circuit becomes overloaded. If you push this button in, the circuit should be energized. You will find procedures and methods used to test electrical control systems at the end of this chapter.

Bleed Air Flow Control Valve.

The bleed air flow control valve is the most important single valve in the low-pressure pneumatic system. It controls the flow of bleed air to the refrigeration unit. When it is closed, partially-conditioned air is not available for the four pressurizations subsystems or for the canopy de-fogging system. Its main effect when closed, however, is to shut off the flow of conditioned air to the cockpit. As you learned in Chapter II, the valve is pneumatically actuated and solenoid controlled. It will operate only when the engine is running and N_2 bleed air is available.

The operation of the valve and its actuator was completely discussed in Chapter II. To put the matter briefly, the valve will be open when the solenoid is energized and closed when it is deenergized. As you can see in the system schematic (figure 3-6) the control of current to the flow control solenoid is much less complicated than the control of current to the ram air regulator and shutoff valve. Current flows from the 28-volt d-c essential bus, through the cabin air circuit breaker, to the cabin air switch. If the switch is at PRESS, current flows through the switch to connection B on the air control timer. From B, current flows through the switch to A, and from there to the solenoid.

The flow of current can be broken at either the cabin air switch or the switch in the air control timer. The relay in the air control timer opens the switch between A and B when the armament bay doors *open* and *close* it several seconds after the doors close. The flow control valve will be open any time the engine is running, the cabin air switch is at PRESS, and the armament bay doors are closed. The rate sensor controls the opening rate of the valve to prevent cockpit pressure from building up too quickly. The rate sensor is discussed later in this chapter in the section on cockpit pressurization control.

Proper operation of the flow control valve and its control system depends upon several things. First, power must be available; if it is not, you must check the power

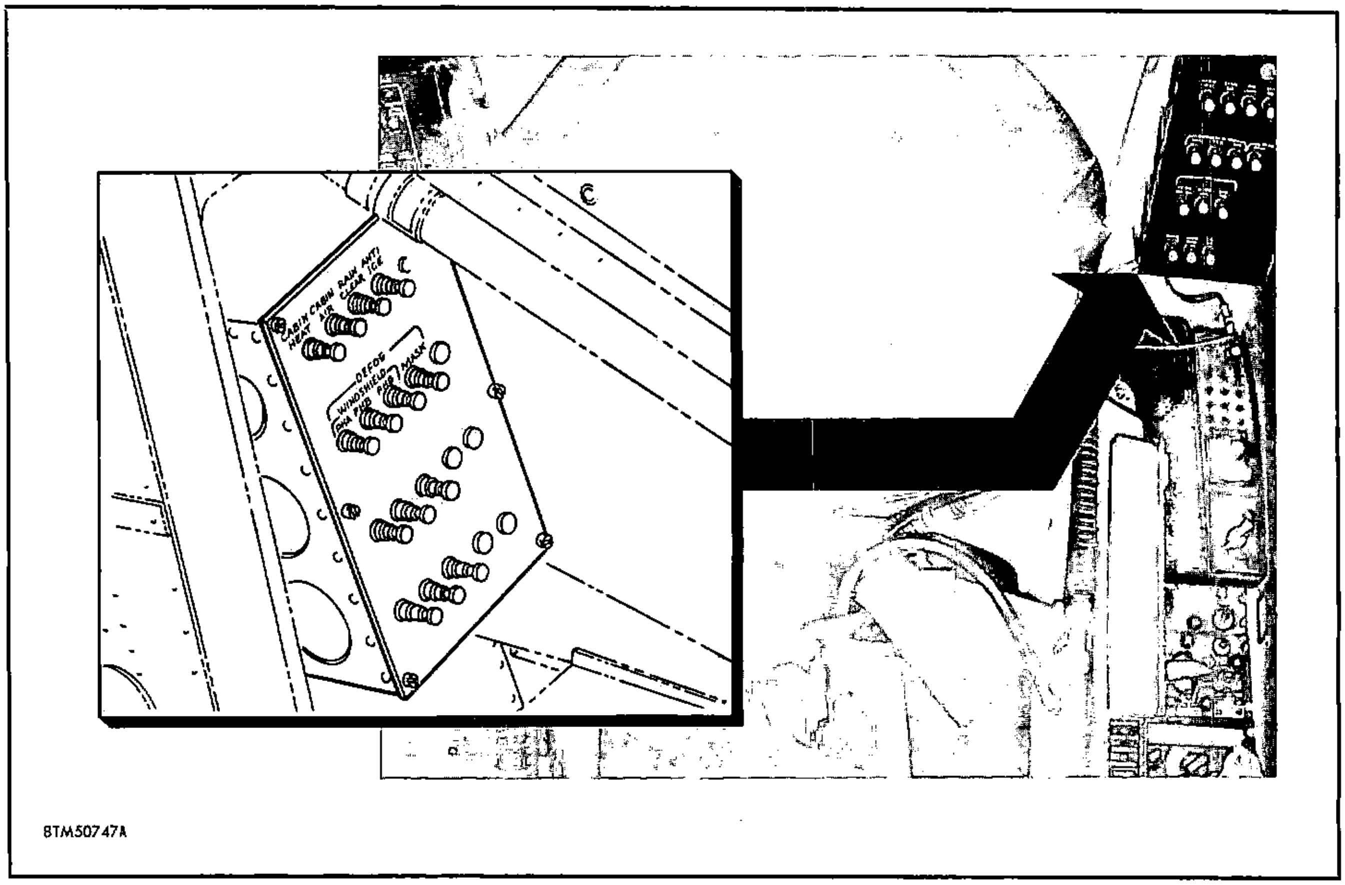


Figure 3-7. Cockpit Circuit Breaker Panel

supply and the position of the circuit breaker. Other procedures for testing electrical control systems will be found at the end of this chapter.

Cockpit Ram Air Valve.

The cockpit ram air valve is a motor-operated valve controlled by the cabin air switch and the main landing gear safety switch. You can see (figure 3-5) that the valve controls the flow of ram air into the cockpit distribution ducting *if* the ram air regulator and shut-off valve is open. Note also in the electrical schematic (figure 3-6) that the cabin air switch has three positions: RAM, PRESS, and OFF. In the RAM position, the cockpit ram air valve is open. In the PRESS position, the bleed air flow control valve is open. In OFF, both valves are closed. Both of these valves *cannot* be open at the same time.

The cockpit ram air valve is also controlled by the landing gear safety switch and will *always* be closed when the airplane is on the ground. As shown in figure 3-8, the valve is a simple butterfly type which is opened and closed by a reversible d-c motor. Note the position indicator on the side of the motor. This will help you when you test the valve. The armature of the motor is connected, through reduction gears, to the valve shutter.

How a Reversible Motor Operates.

By a reversible d-c motor, we mean that the motor can be made to rotate in either direction. The direction of rotation of a d-c motor depends upon the direction of the magnetic field and the direction of the current through the armature. The magnetic field of a d-c motor is created by a current-carrying coil of wire around a metal core or pole piece. When d-c current flows through the coil a magnetic field is created in the core. The direction of the field depends upon the direction of current (flow of electrons) through the coil. If you reverse the current, the field will also reverse. This is the same principle you learned in the previous chapter when we discussed solenoids.

Another electrical principle, figure 3-9, states that if a conductor is placed in a magnetic field and electrons allowed to flow through the conductor, the conductor will be forced from the field. In a motor, many conductors are mounted on an armature, and the sum of all the force exerted on the conductors is the total force exerted on the armature to cause it to rotate. The direction the conductor or armature moves depends upon both the direction of the magnetic field and the direction of electron flow in the conductor. If either the field or the current in the conductor is reversed, the direction the conductor will take when being forced from the field will also reverse. However, no change will take place if the direction of both the field and the current is changed.

In this case, the electric motor has a split field. The wiring of this motor is shown schematically in figure 3-10. You can see that the coil or field winding is divided into two parts. Only one part is used at a time. If the switch is thrown to one contact, current will flow through half of the coil, through the armature to the ground, and thus cause the armature to rotate. If the switch is thrown to the other contact, the other half of the coil will be used. The field will be in another direction but the current through the armature will be in the same direction, so the direction of armature rotation will change.

When the valve is all the way closed, a limit switch inside the motor will open the *close* field coil circuit to prevent a continuous flow of current through the motor. If current is applied to the *open* field coil (You can see this arrangement if you refer to the wiring schematic, figure 3-6.) the motor will turn until the limit switch opens the field coil circuit.

When the cabin air switch is not in RAM, current is supplied to the *close* field at connection B. This current flows through the cabin air control circuit breaker to contact 2 on the ram air relay, and through the switch to contact 3. From 3 the current goes to connection B on the ram air valve motor. When the cabin air switch is in RAM, the control between 2 and 3 on the ram air relay is broken, and 2 is connected to 1. From 1 the current flows to contact 7 on the cabin air relay, through the switch to contact 8 and from there to connection C on the motor. This is the *open* field connection; the valve will move to the full open position, and then the limit switch will break the circuit.

The landing gear safety switch causes the cabin air relay to actuate when the airplane is on the ground. The valve *open* circuit will be broken between contact 7 and 8 current will flow from contact 7 to contact 6 and from there to connection B on the motor. The valve will then close and the limit switch will break the circuit.

Maintaining the Valve.

Proper operation of the cockpit ram air valve and its controls depends on several things. If there is trouble in the circuit, check the power source and the circuit breaker. Procedures for checking an electrical circuit will be found at the end of the chapter. The valve and motor itself can be checked by removing it from the system and applying 28-volt d-c current to the proper contacts and observing the movement of the shutter as shown by the position indicator (figure 3-8). Be sure to connect the negative side of the power supply

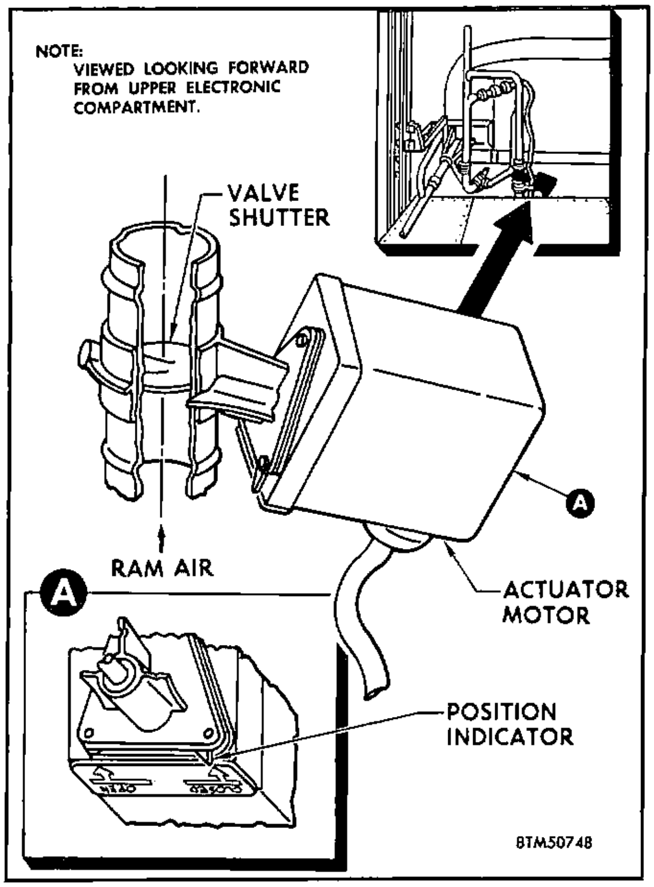


Figure 3-8. Cockpit Ram Air Valve

to connection B or C. Connect the positive side to E. If the unit is defective, it must be replaced.

Proper operation of this valve is especially important since the cockpit cannot be pressurized unless this valve is firmly closed. Any leaks in the closed positions will make it almost impossible to maintain the cockpit at the desired pressure level.

Cockpit Air Valve.

The cockpit air valve is found on only a few early models of the F-102A airplanes. Its location in the cockpit air distribution systems was shown in figure 3-5. The valve is a butterfly-shutter type controlled by a reversible d-c motor, and is almost identical with the cockpit ram air valve. If you find this valve installed in a particular airplane, it should not be operated and must *always* be in the open position. Check the position indicator during inspections to be sure the shutter is open.

The *close* field motor circuit *must not* be connected to power, and the *open* field *must* be connected directly to power to make certain that it will remain open. If it should become fully or even partially closed, it will shut off or seriously interfere with the entire cockpit air conditioning and pressurization system. Make cer-

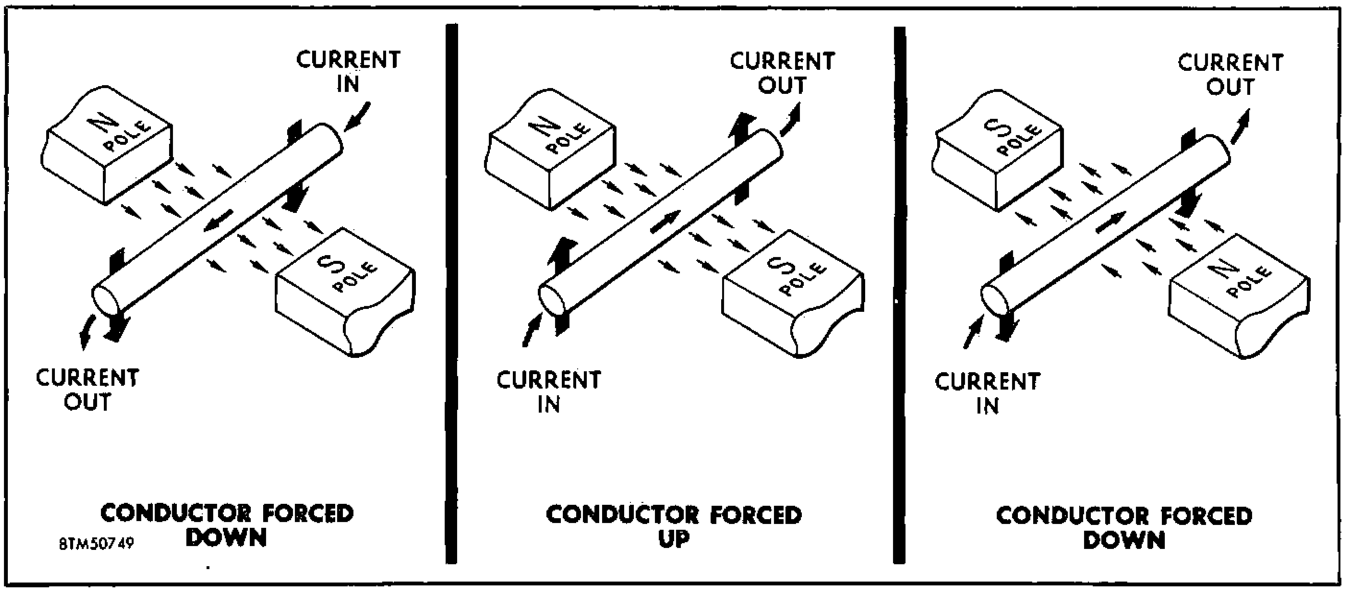


Figure 3-9. Current-Carrying Conductor in a Magnetic Field

tain that the motor terminals are connected as shown in the wiring diagram, figure 3-6. The valve should cause no difficulty, but if it should cause trouble replace it with a short length of plain ducting. Consult your maintenance chief for instructions on how to proceed.

Bypass Valve.

The bypass valve controls the amount of hot bleed air which mixes with the expansion turbine discharge air before entering the cockpit. Its main purpose is to regulate cockpit air temperature. It is a part of the refrigeration unit, and was discussed in Chapter II. This valve and its control system will be covered in the next section of this chapter. It is mentioned here because the total flow of air to the cockpit varies with the valve setting.

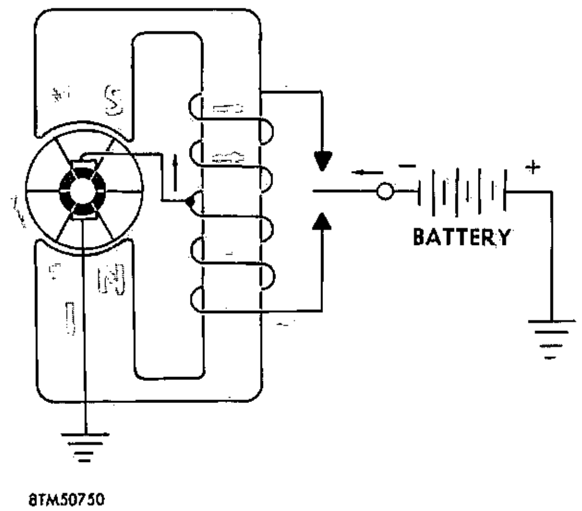


Figure 3-10. Simplified Schematic of a Split-Field Reversible D-C Motor

The maximum flow of air through the heat exchanger is 25 pounds per minute. And, if the bypass valve is closed, that will be the maximum air flow to the cockpit. If the bypass valve is fully open, the maximum flow is increased to 40 ppm which is the maximum permitted through the bleed air flow control valve.

Relays.

There are three relays in the cockpit air input control circuit. Other relays will appear later when we discuss other control systems. A relay is a simple adaptation of an electromagnet, or nothing more than an electromagnetic switch. There are many different types, but they all work on the same principal as the simple relay shown in figure 3-11.

When current flows through the coil in a relay, the iron core becomes magnetized, causing the plate above it to be pulled down. Note that the plate is hinged on one end and suspended by a spring. A strip of iron on the lower surface of the plate contacts the iron core of the magnet. The plate itself is a nonconductive material. At the other end of the plate is a set of contact points.

Note that the relay in figure 3-11 has only one set of points. Some relays have several sets, depending upon the application. When the electromagnet is energized, the contact points close, completing a circuit. This type of relay is a *normally-open* relay. Other relays have contact points which *close* when the current to the electromagnetic coil is off (normally-closed relay) and *open* when the coil is energized. Some relays are a combination of the two types mentioned.

The three relays illustrated in figure 3-6 are all in the intermediate electronic compartment, are well marked, and should be replaced when defective. Do not try to adjust them.

Oil Pressure Switch.

The oil pressure switch is a diaphragm-type that *opens* a contact when the engine oil pressure reaches its proper level, and *closes* the contact when the pressure drops. The switch does two things: first, it causes a warning light in the cockpit to light if oil pressure is low; and second, it controls the flow of current to the ground cooling relay. This second function has already been discussed.

As you can see in figure 3-12, the oil pressure switch is mounted on the left side of the engine. It is completely discussed in another supplement of this series which covers the power plant. When the airplane is on the ground and the engine is not running, there is power to the 28-volt d-c essential bus, and the oil pressure low warning circuit breaker is closed, the warning light will be lit.

COCKPIT AIR DISCHARGE CONTROL SYSTEM.

The flow of air from the cockpit is controlled and regulated by a pressure regulator and a safety valve. As you can see, (figure 3-13), air leaves the cockpit through the cockpit discharge duct and the cockpit safety valve. The pressure regulator, installed on the cockpit floor between the rudder pedals, automatically controls the flow of air from the cockpit to regulate cockpit air pressure. The safety valve is really another regulator with a slightly higher pressure setting. If the pressure regulator becomes defective, the safety valve vents cockpit air overboard to prevent too high a pressure.

The complete operation of these two components will be completely discussed in another section of this chapter on cockpit pressurization control. Each of these components sense cockpit and outside air pressure in order to regulate the air flow. Both the regulator and the safety valve have pressure sensing lines and each has a solenoid-controlled shutoff valve in one of the lines. It is those shutoff valves we will discuss here.

In figure 3-13, note that the pressure regulator has a sensing line connected to the discharge ducting. There is a check valve and a solenoid-controlled shutoff valve in this line. The check valve allows air to flow from the ducting to the regulator control chamber, but prevents a reverse flow. When air flows to the control chamber through this line, the regulator will close and prevent air from flowing from the forward electronic compartment to the cockpit when the pressure in the ducting is higher than cockpit pressure.

The shutoff valve is open whenever the airplane is in the air. When the airplane is on the ground, the landing gear safety switch causes the shutoff valve to be energized, thus closing the valve. This allows air to flow from the forward electronics compartment to the cockpit when a ground air conditioner unit is ventilating the electronic compartments. This control circuit is shown in figure 3-14.

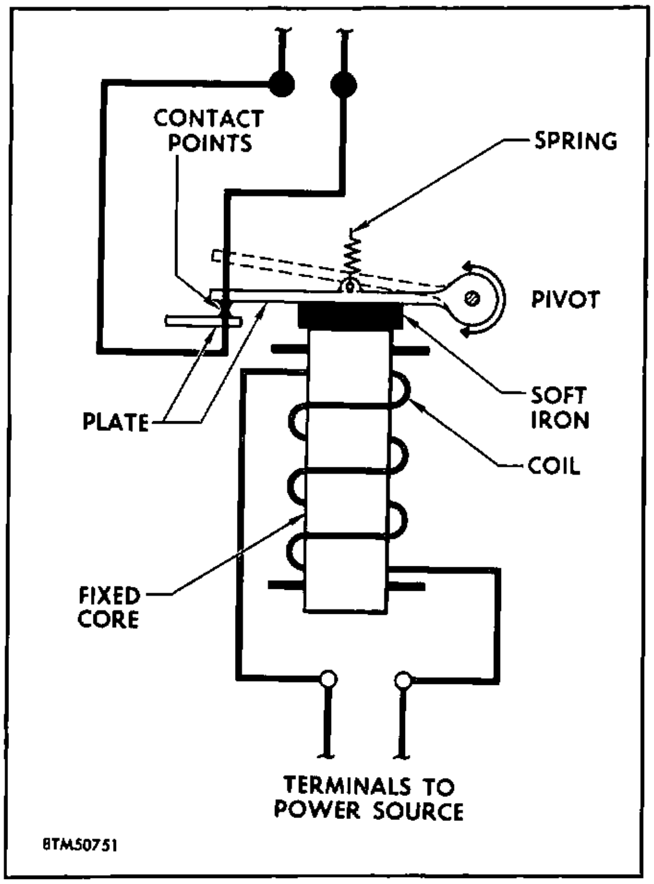


Figure 3-11. A Simple Relay

Note that the solenoid is not connected to power unless the landing gear safety switch is closed (G-E), thus energizing the cabin air relay and closing the switch between contacts 12 and 13. The solenoid will deenergize and the valve will open any time electrical power is cut off, either at the relay or at the source. The solenoid-controlled valve is a very simple type in which the valve is connected directly to the solenoid core or actuating piston. If it causes difficulty, replace it.

Operation of the shutoff valve in the safety valve sensing line is very similar to that of the valve described above. This is a solenoid-controlled valve that is closed when the soleneid is deenergized. When the shutoff valve is open, the safety valve control chamber is vented to atmosphere and the safety valve will open to relieve cockpit pressure. The solenoid is controlled by both the cabin air switch and the landing gear safety switch. When the airplane is in the air and the cabin air switch

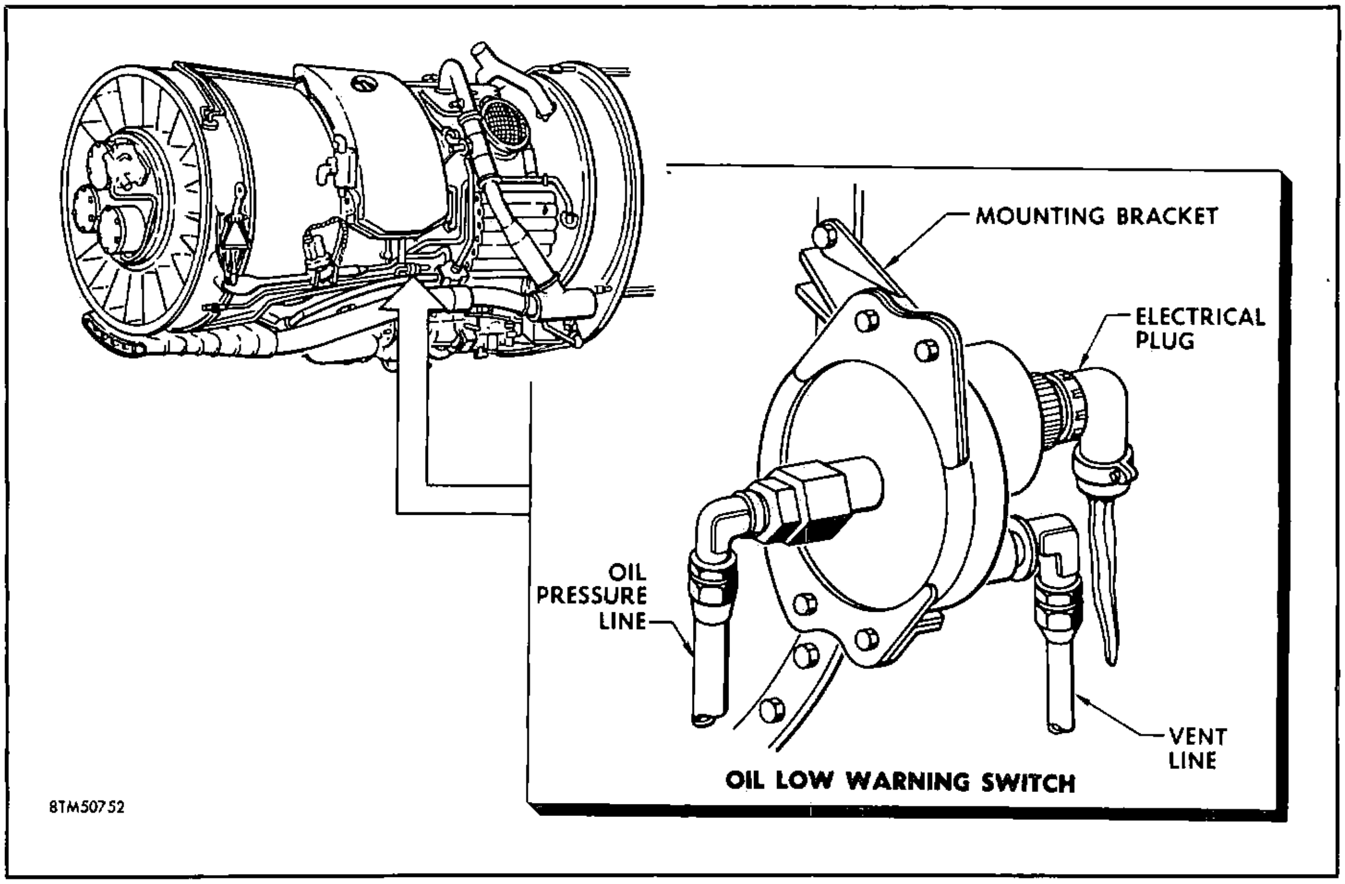


Figure 3-12. Oil Pressure Switch

is not in RAM, the valve is closed. In figure 3-14, you can see how the solenoid valve is energized.

If the cabin air switch is in RAM, the ram air relay will energize and close the switch between contacts 1 and 2. Power can then flow from the 28-volt d-c essential bus, through contacts 1 and 2 and on to the solenoid. On the ground the solenoid will be energized and the valve will be closed regardless of the position of the cabin air switch. When the cabin air relay is energized on the ground by the safety switch, the switch between contacts 6 and 7 will be closed. If the cabin air switch is not at RAM, current will flow through contacts 2 and 3 on the ram air relay and then continue through contacts 6 and 7 on the cabin air relay before reaching the shutoff valve solenoid. This valve and its control solenoid should also be replaced if it is defective. Tests required to locate defects in this system are discussed later in this chapter.

COCKPIT TEMPERATURE CONTROL SYSTEM.

As previously shown, cockpit temperature is regulated by an automatic temperature control system consisting of a heat control box, a thermostat in the discharge ducting and another in the supply ducting. The thermostats send signals to the control box to indicate air temperatures, and the control box in turn controls the flow of current to the reversible d-c motor that actuates the bypass valve.

The position of the bypass valve determines the amount of hot bleed air that bypasses the refrigeration unit and mixes with refrigerated air. As you know, the proportion of bypass air to refrigerated air determines the temperature of air entering the cockpit. Figure 3-15 shows the components of the system in their relative locations in the airplane.

HOW THE SYSTEM OPERATES.

You know from our previous discussions that the function of the temperature control system is to regulate the flow of hot bleed air through the bypass valve. It does this by automatically sensing the temperature of cockpit inlet and discharge air, and energizing one field winding or the other of the bypass valve d-c motor. The d-c motor is a reversible, split-field type similar to the cockpit ram air valve motor discussed earlier. A brake on the motor locks the valve in position when the motor is deenergized. The electrical schematic of the temperature control system is shown in figure 3-16.

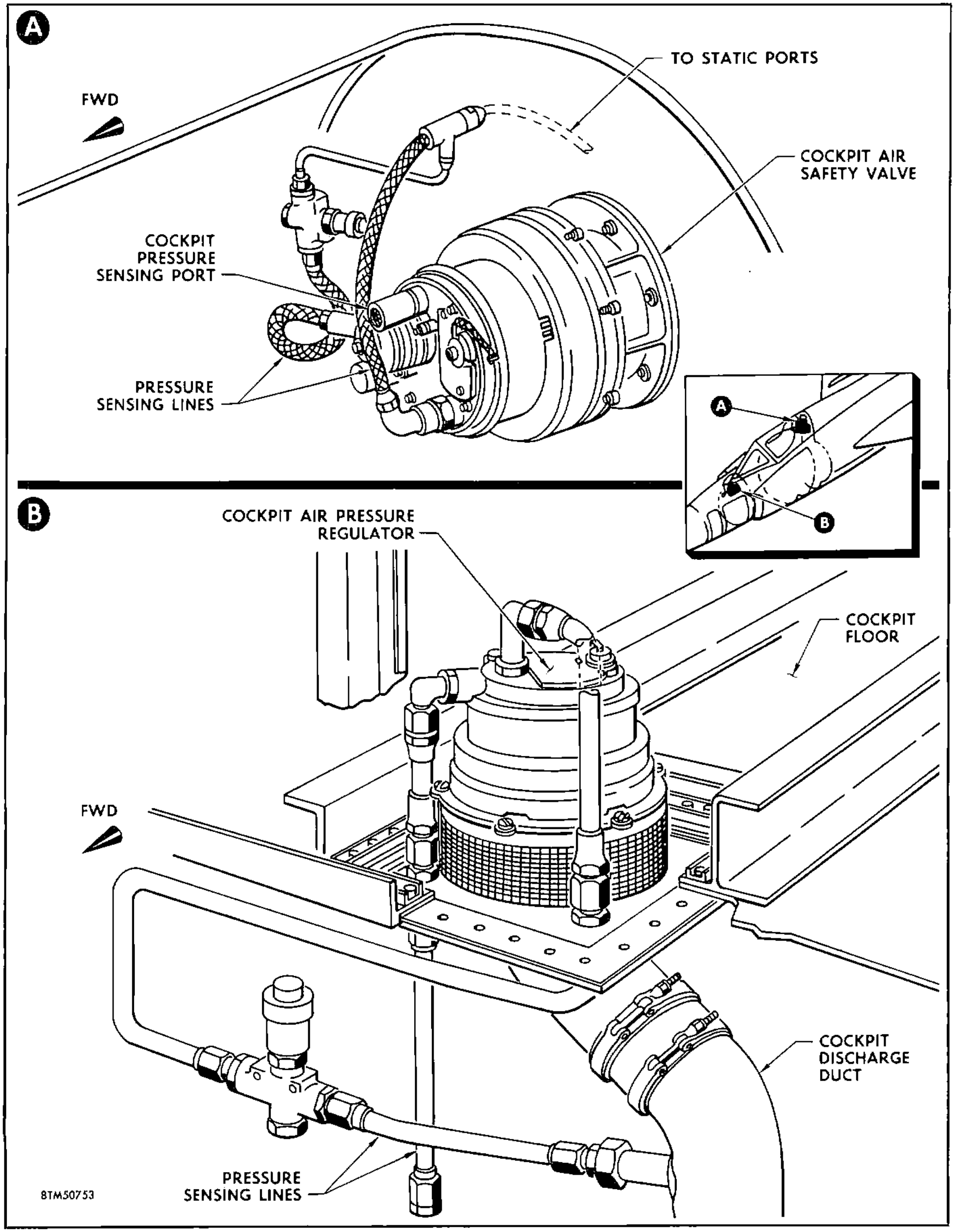


Figure 3-13. Cockpit Air-Discharge Control

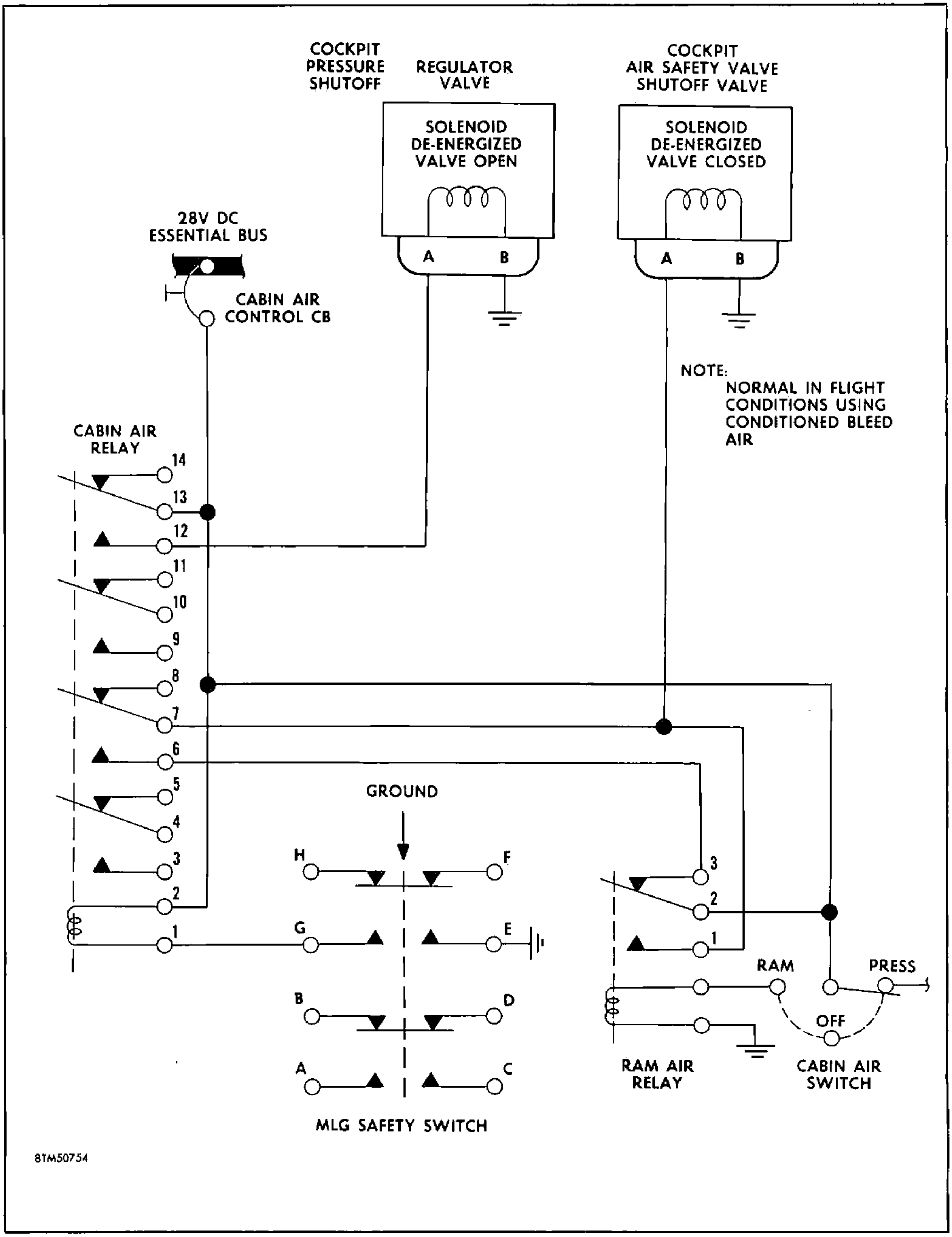


Figure 3-14. Cockpit Air Discharge Control Electrical Schematic

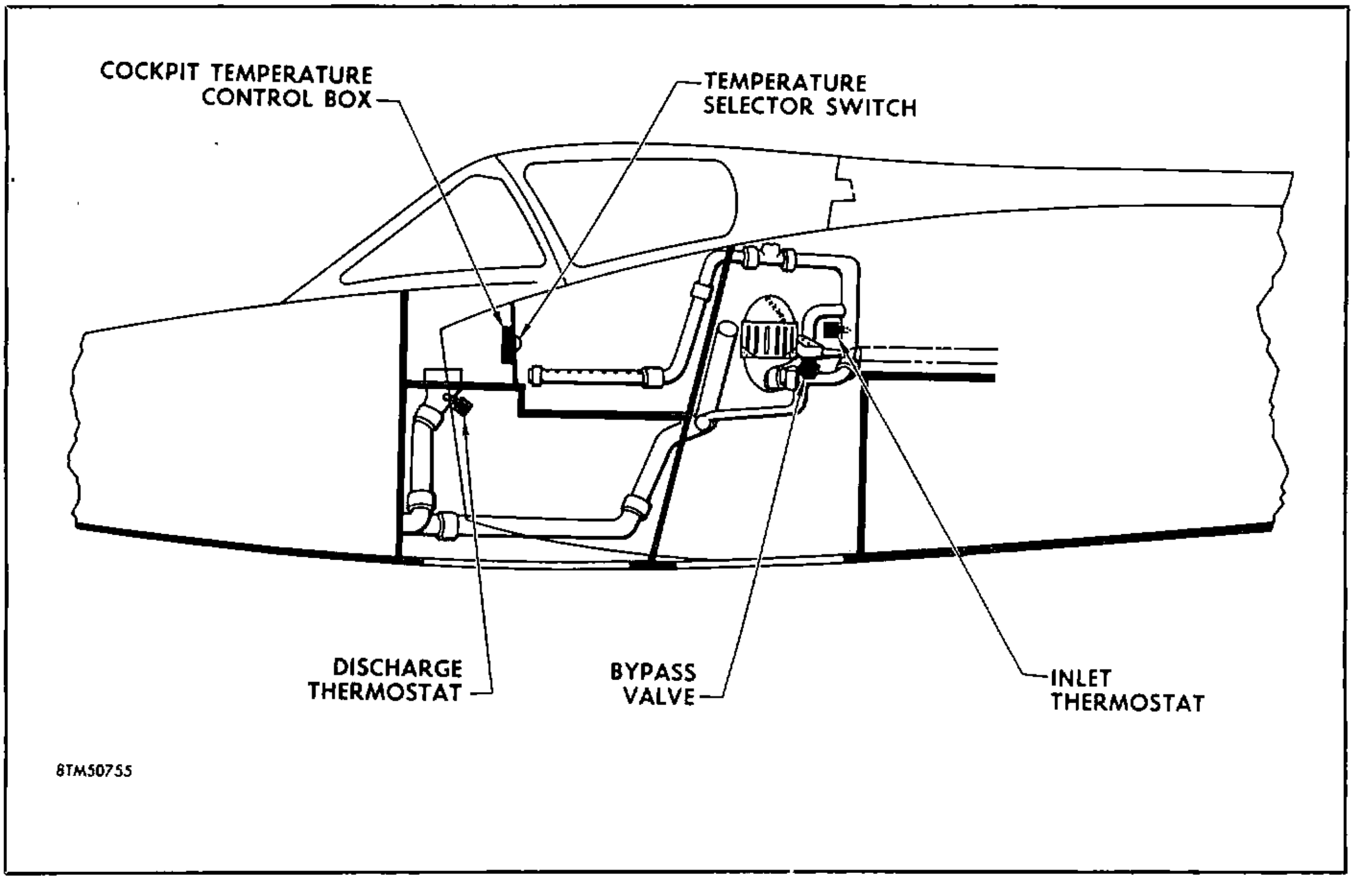


Figure 3-15. Cockpit Temperature Control System Components

The schematic has been simplified by removing some of its resistors. Some of these resistors are adjustable so that minor adjustments can be made. Never attempt to adjust these resistors in the airplane. For the exact location of these resistors and the procedure to be followed, refer to the maintenance manual, T.O. 1F-102A-2-6.

The four main units of the system are the valve and its d-c reversible actuating motor, the two mercury-column type thermostats, and the heat control box. The heat control box contains three relays, a manual temperature selector switch, and various resistors. The control knob, marked CABIN TEMPERATURE CONTROL, on the selector switch is on the pilot's utility switch panel. The heat control box is mounted directly behind this panel. The knob is connected to the three switches and the temperature selector rheostat.

When the knob is in the automatic range, the automatic temperature control system is operating and will regulate the temperature according to the position of the knob within the range. When the knob is at MANUAL HOT or MANUAL COLD, the automatic part of the system is disconnected and the MORE HEAT or LESS HEAT field winding of the motor is connected directly to power. The manual positions are spring-loaded, and the switch will move to a position between AUTO and MANUAL when it is released. All power to the motor is cut off in this position and the valve will remain where it was when the switch was released.

Since it takes about 10 seconds for the valve to go from full *open* to full *closed,* the manual switch may be rapidly turned on and off to make small adjustments in the valve position.

When the selector switch is in the automatic range, either of the two field windings on the motor can be connected to power through a series of switches on the three relays. You can see in figure 3-17 that current to the MORE HEAT winding must pass through four switches. One switch is controlled by each of the three relays, while the other is controlled manually by the temperature selector switch. Current to the LESS HEAT winding passes through two switches, one controlled by the LESS HEAT relay and the other by the manual temperature selector switch. Only one of the field windings is energized at a time. The flow of current to the field windings is controlled by the position of the switches which are in turn controlled by the relays. The relays are controlled by the level of the mercury column in the thermostats which depends in turn on the temperature of the air in the ducts and the amount of current in the thermostat heater windings.

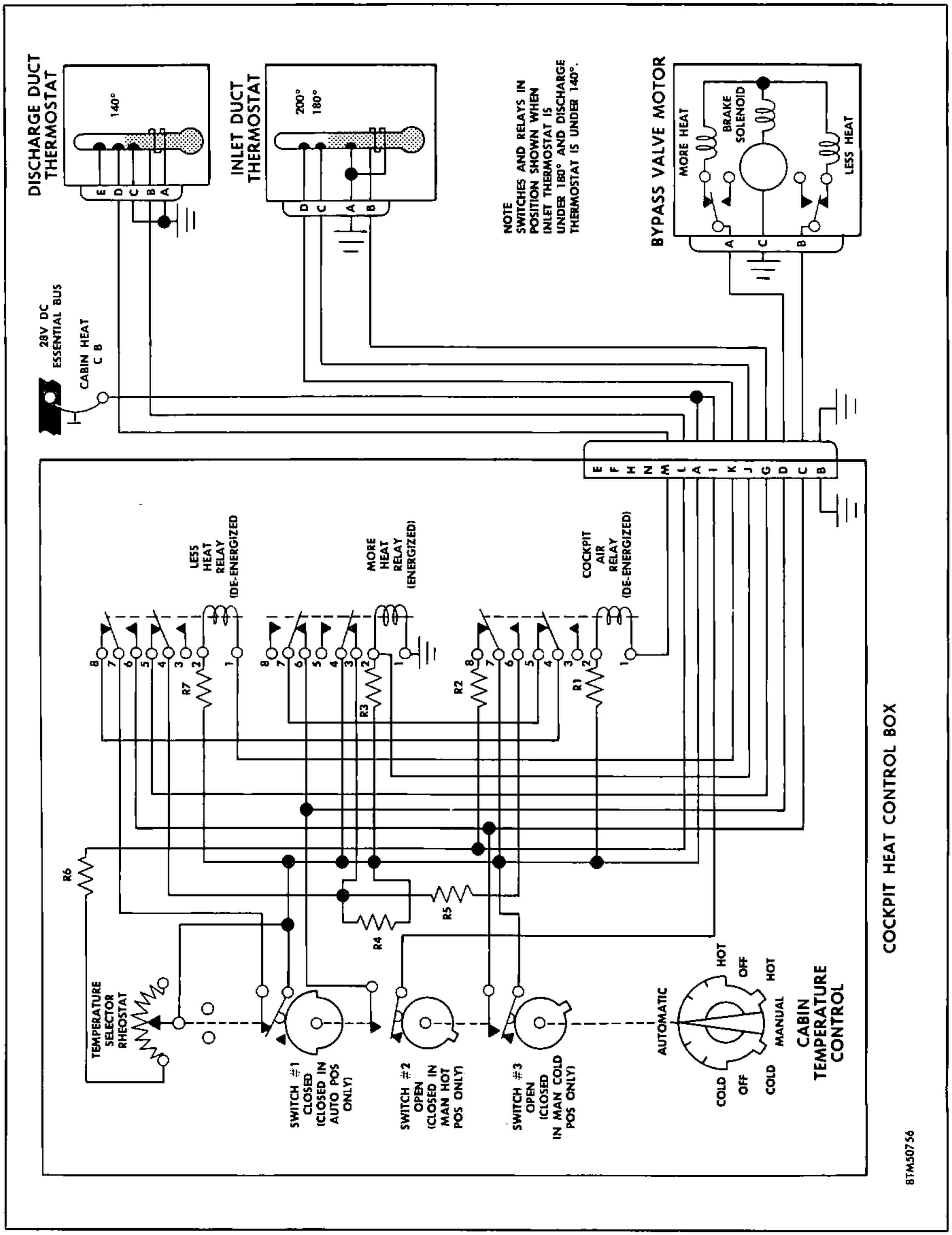


Figure 3-16. Cockpit Temperature Control System Electrical Schematic

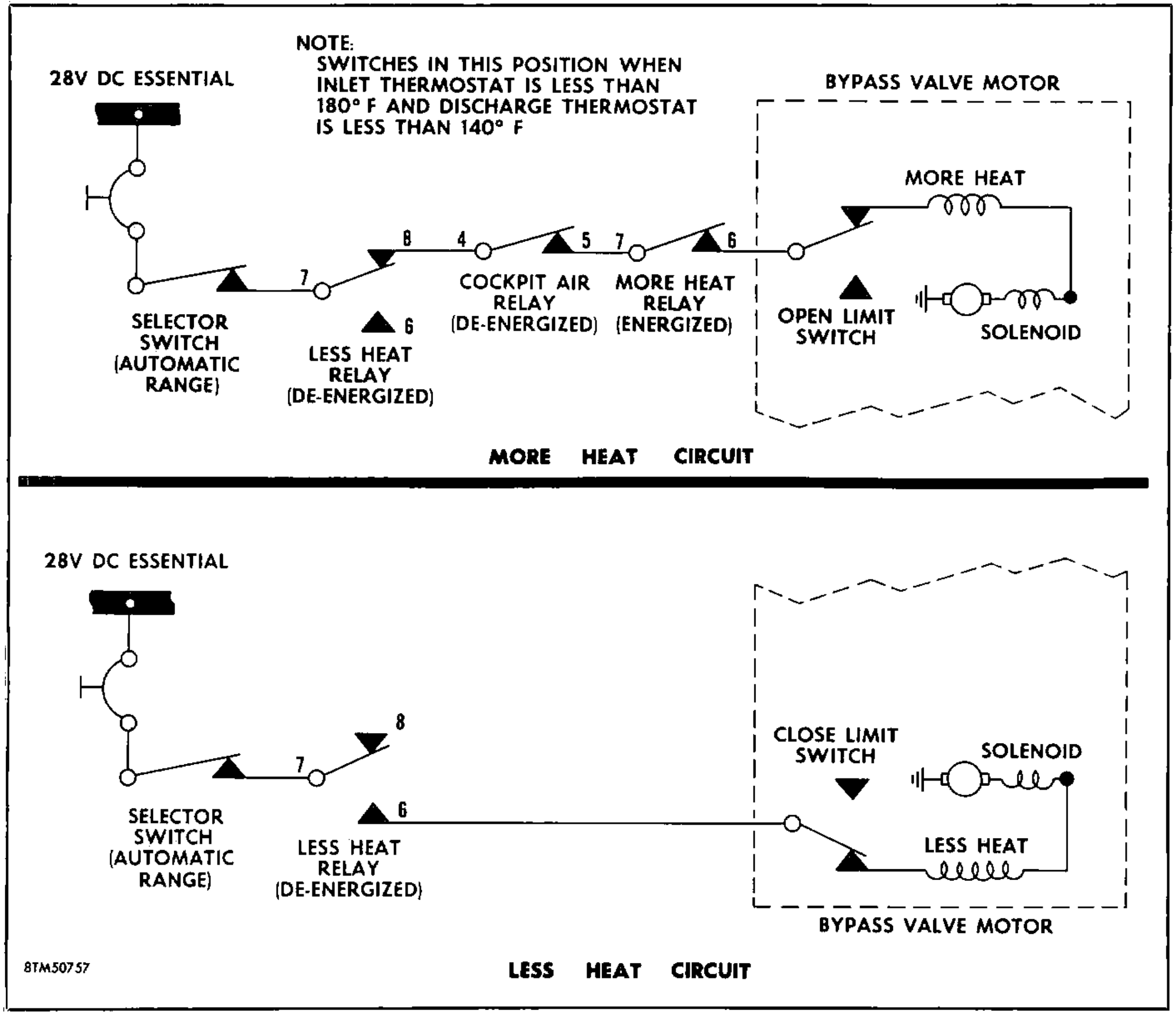


Figure 3-17. Field Winding Circuits of a Bypass Motor

The heater windings are used to increase the speed at which the mercury columns respond. Current to the heater windings depends upon switches controlled by the three relays and upon the position of the temperature selector rheostat. The heater winding circuits are illustrated in figure 3-18. These circuits, and the ones in figure 3-17, are taken from the main electrical schematic, figure 3-16.

The switches and relays will be in the positions shown when the selector switch is in the automatic range, the inlet thermostat registers under 82°C (180°F), and the discharge thermostat registers under 60°C (140°F). The relay coil for the cockpit air relay is energized when the mercury column in the discharge duct thermostat reaches 82°C (180°F), thus completing the relay circuit, through the mercury column, to ground. The relay and its switches are shown in the deenergized position.

The MORE HEAT relay is always energized when the inlet duct thermostat registers less than 82°C (180°F). When the column of mercury reaches 180°F, the MORE HEAT relay is shorted to ground and the relay is de-energized. When the inlet thermostat registers 93°C (200°F), the LESS HEAT relay will be energized.

When conditions are stabilized the system modulates, or fluctuates, around the 140°F mark on the discharge duct thermostat and the 180°F mark on the inlet duct thermostat. The bypass valve will move toward the *open* or MORE HEAT position only when the discharge thermostat is below 140°F at the same time the inlet thermostat is below 180°F. The bypass valve will move toward the *closed* or LESS HEAT position *only* when the inlet thermostat reaches 200°F.

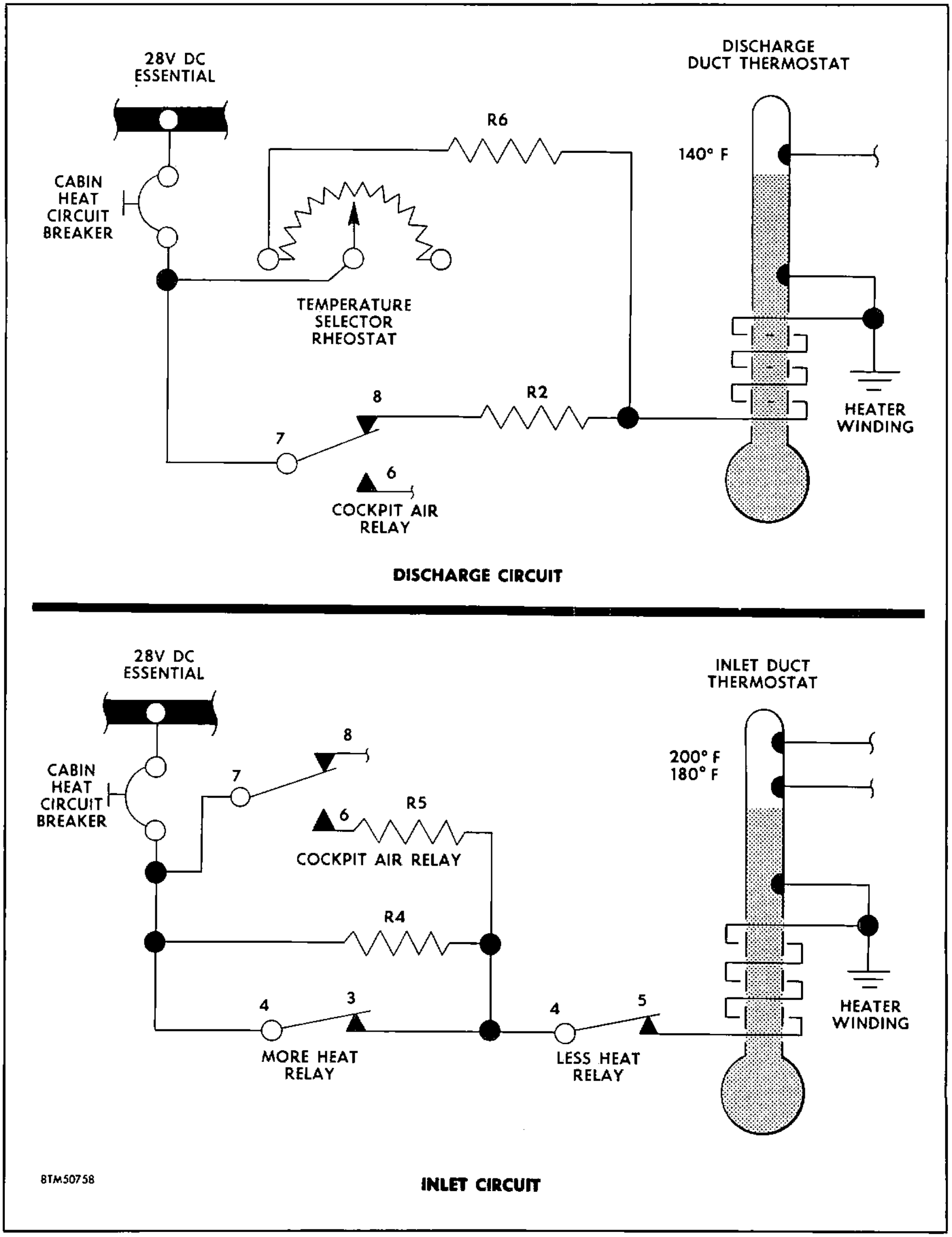


Figure 3-18. Thermostat Heater Circuits

Even when the air temperatures are constant, the thermostat readings will vary because the thermostats are heated not only by hot air but by the electric current through the heater windings. This current is not constant so the thermostat readings will not be constant. Figure 3-18 is a schematic of the circuit that controls current to the discharge thermostat heater winding. When the thermostat is below 140°F, the switch across 7 and 8 is closed and the resistors are in parallel; that is, R_2 is in parallel with R_6 and the rheostat (variable resistor). In this case the effective resistance of the circuit is low and the current to the winding is high; therefore, the mercury will rise faster.

The exact rate and extent of the mercury rise is dependent on the position of the rheostat. When the mercury reaches 140°F, the cockpit air relay is energized, the switch across 7 and 8 is broken, resistor R_2 is removed from the circuit and the effective resistance of the circuit is increased. This sharply reduces the current in the heater winding, the mercury will fall below 140°F, the switch across 7 and 8 closes, the effective resistance decreases and the mercury rises again. The mercury will fluctuate continually around 140°F.

If the rheostat is turned to the COLD side (removing resistance) the mercury will climb faster and will go further past the 140°F mark. The mercury will spend more time at 140°F, or higher, than it does when the rheostat is centered or at the HOT side (adding resistance).

The heating circuit for the inlet thermostat is shown at the lower section of figure 3-18. When the inlet thermostat is under 180°F and the discharge thermostat is under 140°F, the switch across contacts 4 and 3 on the MORE HEAT relay and the switch across 4 and 5 on the LESS HEAT relay will be closed. The switch across 7 and 6 on the cockpit air relay will be open.

At this time, the effective resistance will be very low and the heater current will be highest; thus, the mercury will be driven up. When it reaches 180°F, the switch across 4 and 3 will open and resistor R_4 will be the sole path for current flow. The effective resistance is increased; therefore, the heating current is reduced. The mercury will fall below 180°F, the current will increase driving it back up to 180°F, and the cycle will be repeated. This means that the mercury level will fluctuate around 180°F. However, if the discharge thermostat reaches 140°F, the switch across 7 and 6 on the cockpit air relay will close, and resistors R_4 and R_5 will be in parallel. The effective resistance of these two resistors in parallel is much less than the resistance of R_4 by itself.

This reduction in resistance will allow an increase of heating current, between 180° and 200°F, driving the mercury further toward the 200°F mark. When the 200°F mark is reached, the switch across 4 and 5 on the LESS HEAT relay will open and remove all current from the winding until the mercury again falls below 200°F. When the discharge thermostat is at 140°F or higher, the inlet thermostat will tend to fluctuate around the 200°F mark. When the discharge thermostat is below 140°F, the inlet thermostat tends to fluctuate around the 180°F mark.

The heater circuit never furnishes enough heat by itself to drive the mercury to 180° or 200°F; the air itself furnishes most of the heat. If the air in the inlet duct stays at a low temperature, the inlet thermostat will never reach 200°F and possibly will not reach 180°F.

It still may not be clear just how a movement of the temperature selector switch causes the bypass valve to move. Referring to figure 3-6 as you read the following discussion, should better enable you to understand the system. First, assume that conditions are stable and the selector switch (and rheostat) have been in one position for some time. The valve may not be at a fixed position, but will probably be making small fluctuations or movements in both directions about a control point. Effectively, the valve is in a steady condition. If the cockpit air temperature should fall, allowing the inlet thermostat to fall below 180°F and the discharge thermostat to fall below 140°F, the MORE HEAT winding will be energized to move the valve toward the open position. When the thermostats again reach 180°F and 140°F, the valve will fluctuate around a new control point. Now let us assume that the air temperature is constant, but the pilot decides he is too hot and wants less heat. He will turn the selector switch to the left. This reduces the rheostat resistance in the circuit. More current will flow to the discharge thermostat heater winding and the mercury will hit 140°F and stay there for a longer period of time. At 140°F, the cockpit air relay opens and breaks the switch at contact 4 and 5. This positively removes power from the MORE HEAT motor winding.

To make the valve move toward the closed or LESS HEAT position, the inlet thermostat must reach 200°F. At this position, the less heat relay will energize and close the switch across contacts 7 and 6 to energize the less heat field winding. To make the thermostat reach 200°F, more current must be applied to the heater winding. Ordinarily, there is no difficulty in reaching 180°F, since there is a full current flow in the heater winding. Above 180°F, the current is reduced.

The current in the winding, when the thermostat is above 180°F, will be increased if the discharge thermostat is above 140°F. This will drive the mercury to 200°F, thus causing the valve to move toward the closed position. Since moving the selector switch to the left, or COLD, side causes the discharge thermostat to rise to 140°F, it follows then that moving the switch to the COLD side will cause the valve to move toward the closed position. As soon as this results in cooler air

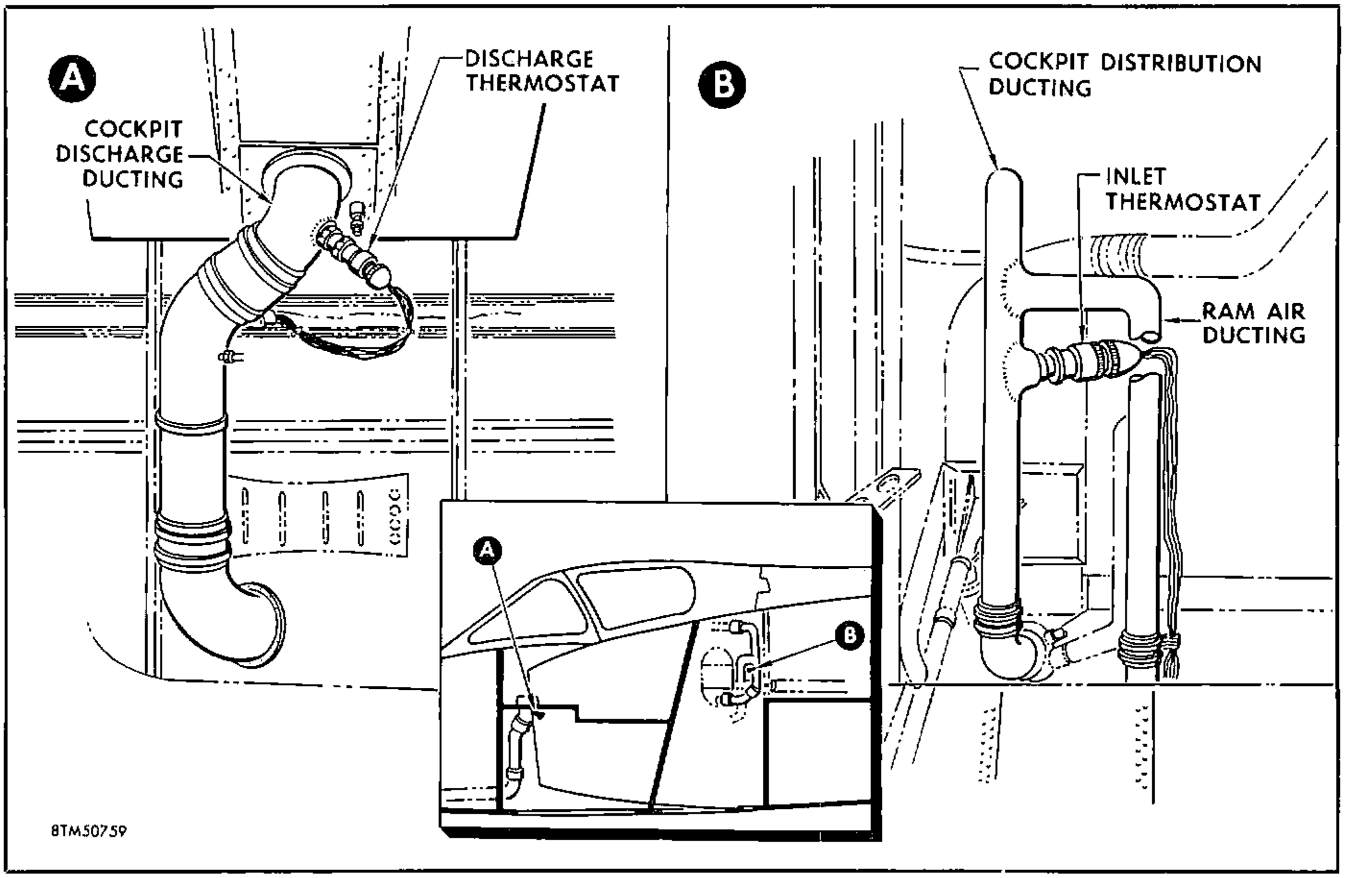


Figure 3-19. Cockpit Temperature Control Thermostats

going through the system, both thermostat readings will drop and the system will stabilize around a new control point.

Thermostats.

The two mercury thermostats are essentially mercury thermometers. The main difference is the platinum wire contacts inserted into the center of the thermostat at two or more places. These contacts provide a current-carrying junction in the electrical circuit. The junction is closed and the circuit is completed only when the mercury touches both contacts. Since the height of the mercury column varies with the temperature, the thermostats act as thermal *off-on* switches. The only difference between the two thermostats is in the number and position of the platinum contacts. Little trouble should be expected from these components.

When testing a thermostat, be careful not to send too much electrical current through the mercury column. Large currents tend to separate the column so that it will no longer measure temperatures accurately. Use the smallest amount of current that will still give good test results. If tests show the unit to be defective, replace it, since it cannot be repaired.

Cockpit Heat Control Box.

The heat control box has already been discussed in detail. Most defects or malfunctions of the temperature control system will be found in this unit. The most common defect will be a stuck relay or a relay that operates only part time. The switches operated by the manual temperature-selector knob may also become defective. See T.O. 1F-102A-2-6 for the tests you should perform to locate defects in the system. If the tests show that some component in the box is defective, the whole box must be replaced.

Bypass Valve.

The bypass valve has already been mentioned in several other sections of this manual. It is a butterfly-type valve actuated by a reversible d-c motor and controlled by the automatic temperature control system. The motor is a split-field type similar to the motor of the cockpit ram air valve. This valve and motor were discussed previously. The main difference in the two valves is the fact that the cockpit ram air valve is either full *open* or full *closed,* while the bypass valve can be stopped in any position. The addition of a brake to the bypass valve motor makes the difference.

The brake is the mechanical type that holds the motor armature in a fixed position when there is no current through the motor. When the motor is energized, current flows through a solenoid coil which causes the brake to be released by magnetic action. The control

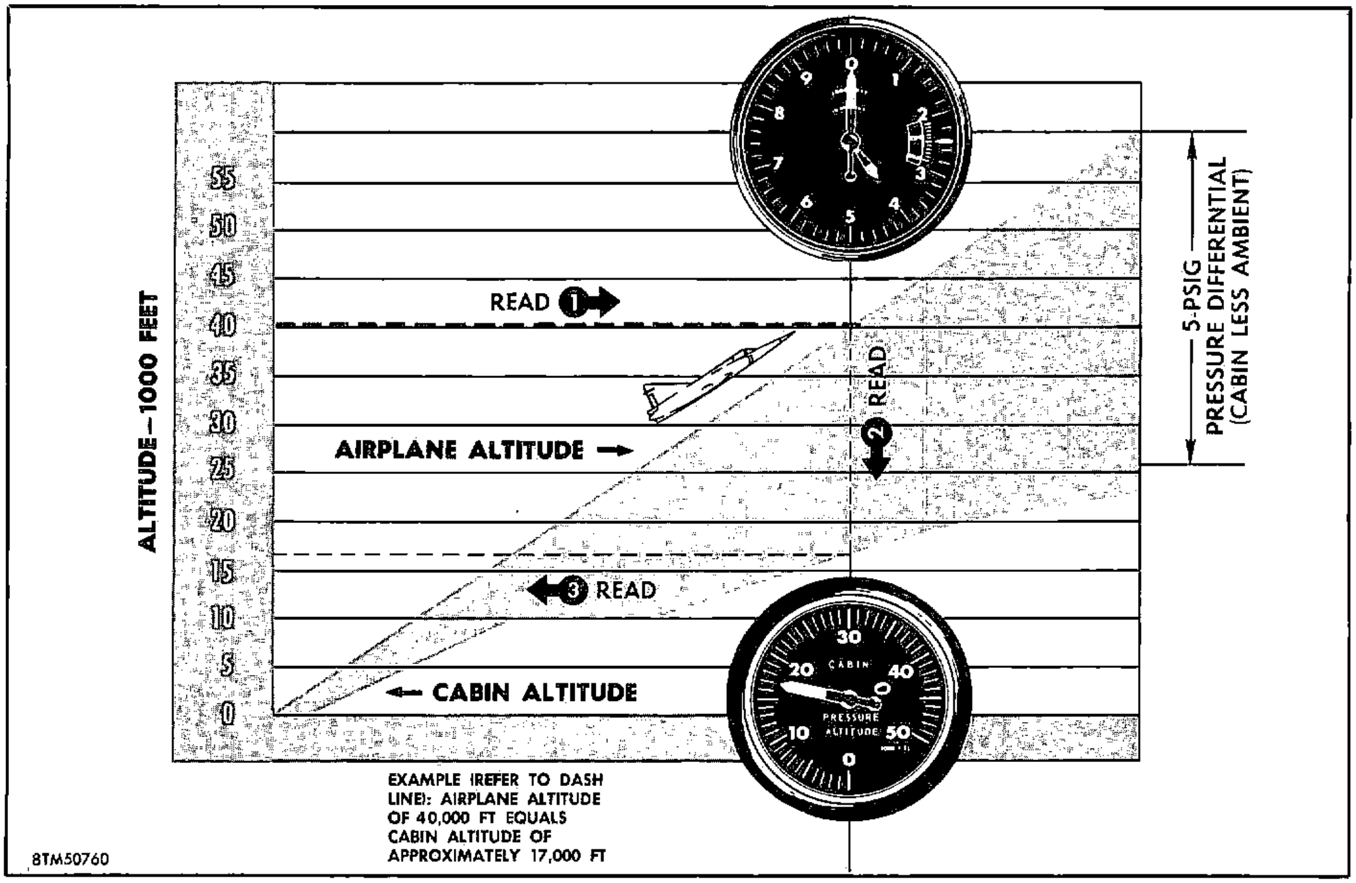


Figure 3-20. Cockpit Pressure Altitude Schedule

system feeds current into one or the other of the field windings until the valve reaches the desired position. Current is then removed, and the brake locks the armature and the valve in place. The valve and its motor is a part of the refrigeration unit and was discussed and illustrated in Chapter II.

THE COCKPIT PRESSURE CONTROL SYSTEM.

The cockpit air pressure is regulated automatically by a control system consisting of a pressure regulator, a safety valve, and a rate sensor. The pressure regulator, installed on the cockpit floor between the rudder pedals, controls the cockpit pressure by regulating the flow of air from the cockpit. The safety valve is similar to the regulator, except that it has a higher pressure setting. Both of these components regulate the air flow by balancing cockpit pressure against atmospheric static air pressure.

The system is entirely non-electrical, except for a solenoid-operated shutoff valve in a pressure-sensing line to the regulator, and a similar valve in another pressure-sensing line connected to the safety valve. The first valve is normally open, but is closed on the ground by a signal from the landing gear safety switch. The second shutoff valve is normally closed; however, it opens on the ground upon a signal from the safety switch. It can also be opened in the air by manually placing the cabin air switch in the RAM position. These valves and their control were discussed earlier in this chapter under Cockpit Air Flow Control.

The pressure regulator maintains cockpit pressure at a level slightly higher than atmospheric pressure at all altitudes up to 10,000 feet. From 10,000 feet up to 26,000 feet, cockpit pressure is maintained at a constant value of about 10.2 psi, which is equivalent to atmospheric pressure at about 10,000 feet. From 26,000 feet to the maximum altitude of the airplane, cockpit pressure is maintained at a value approximately 5 psi higher than atmospheric pressure at the altitude at which the plane is flying.

A cabin pressure altitude indicator on the left auxiliary instrument panel indicates cockpit pressure in terms of the equivalent altitude in thousands of feet as shown in the chart in figure 3-20. For instance, at 40,000 feet of airplane altitude, the indicator would show a cockpit pressure altitude of 17,000 feet. If the cockpit regulator should become defective, the safety valve regulates the pressure at a level slightly higher than the pressure normally maintained by the pressure regulator.

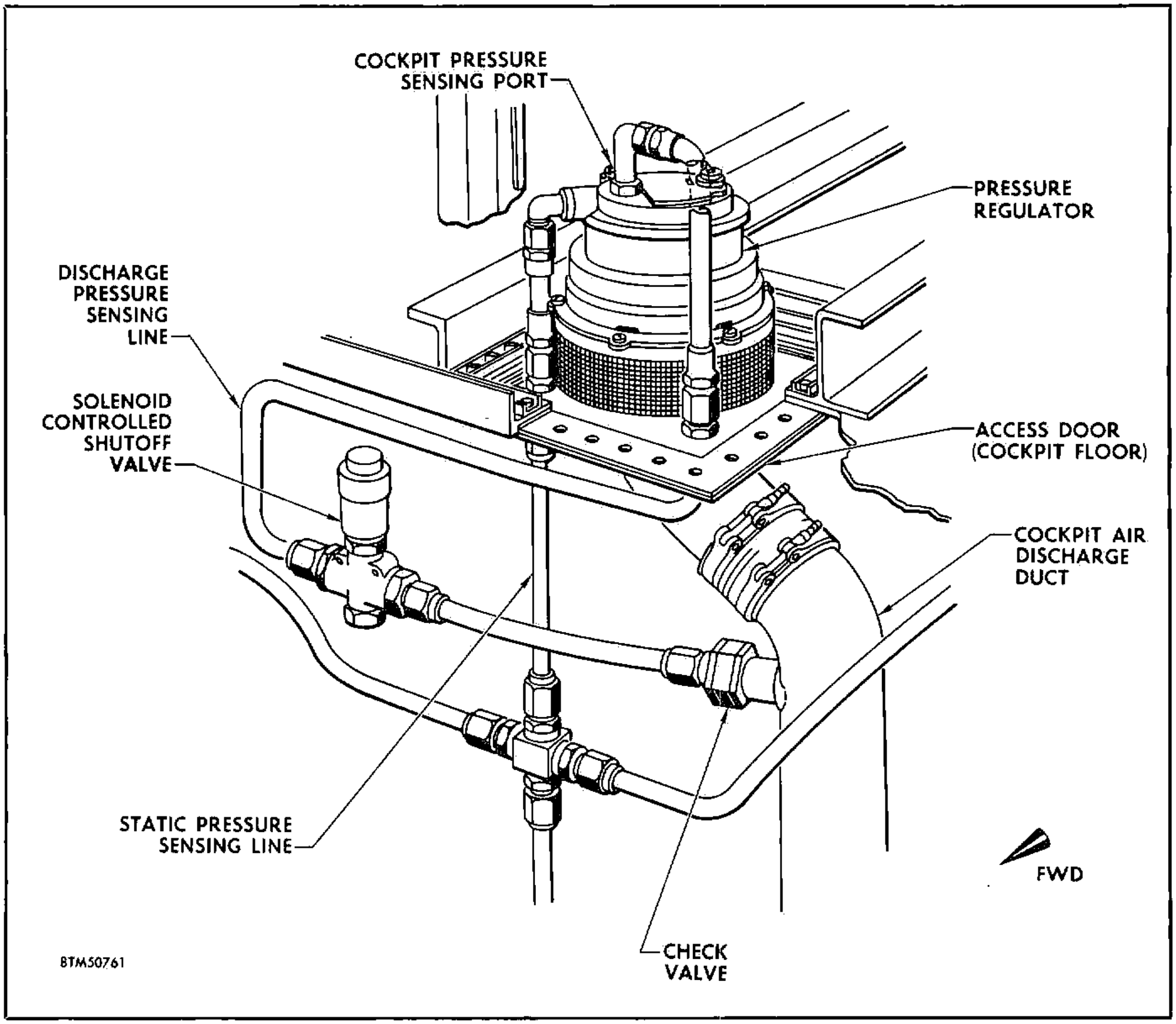


Figure 3-21. Cockpit Pressure Regulator

When the cabin air switch is at RAM or when the airplane is on the ground, the safety valve allows an unrestricted flow of air overboard from the cockpit. This results in a completely unpressurized cockpit. The rate sensor prevents cockpit pressure from building up too fast for the pilot's comfort or safety. This is important after ram air has been used for ventilation and after armament has been fired. The cockpit is unpressurized during ram air ventilation since the safety valve is open.

During armament firing, the flow of conditioned air to the cockpit is stopped and cockpit air pressure gradually leaks off. In any case, when the safety valve is closed and the flow of conditioned air is turned back on, the pressure rise would be very fast if the flow control valve were to be opened all the way. The rate sensor senses the sudden build-up of cockpit pressure and drains air flow from a control chamber in the flow control valve pneumatic actuator. This causes the valve to close and restrict the air flow and to slow down the rate of pressure build-up. The flow control valve and its actuator were completely covered in Chapter II.

Cockpit Pressure Regulator.

The cockpit pressure regulator controls the flow of air from the cockpit by sensing and comparing static air pressure (atmospheric pressure, not ram air pressure) with cockpit pressure. Note that two sensing lines connect to the top of the regulator in figure 3-21. One line connects static air pressure from pressure sensing ports on the airplane fuselage to the regulator control chamber. The second line connects the control chamber to the cockpit discharge duct in the nose wheel well. There is also a check valve and a

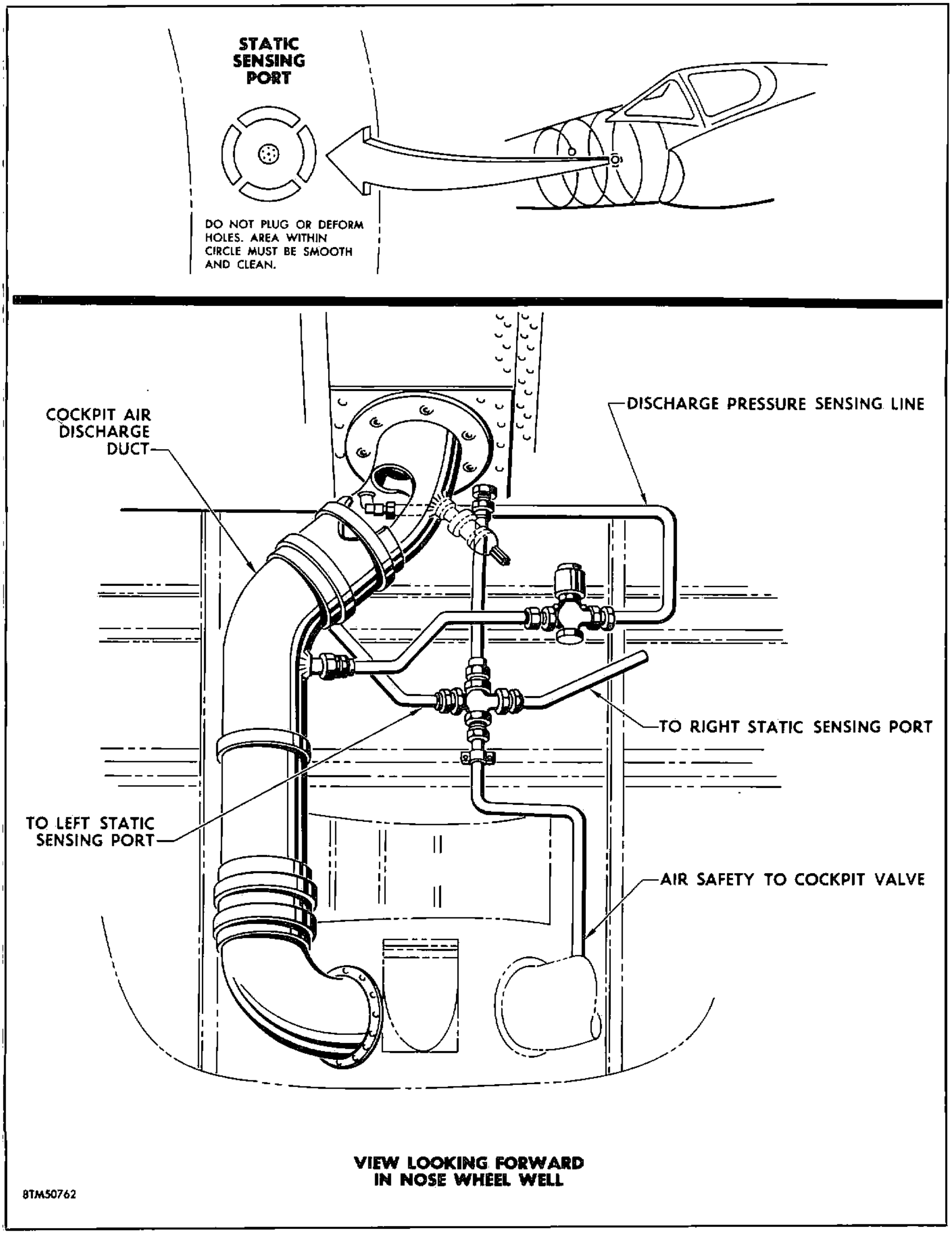


Figure 3-22. Cockpit Regulator Sensing Lines

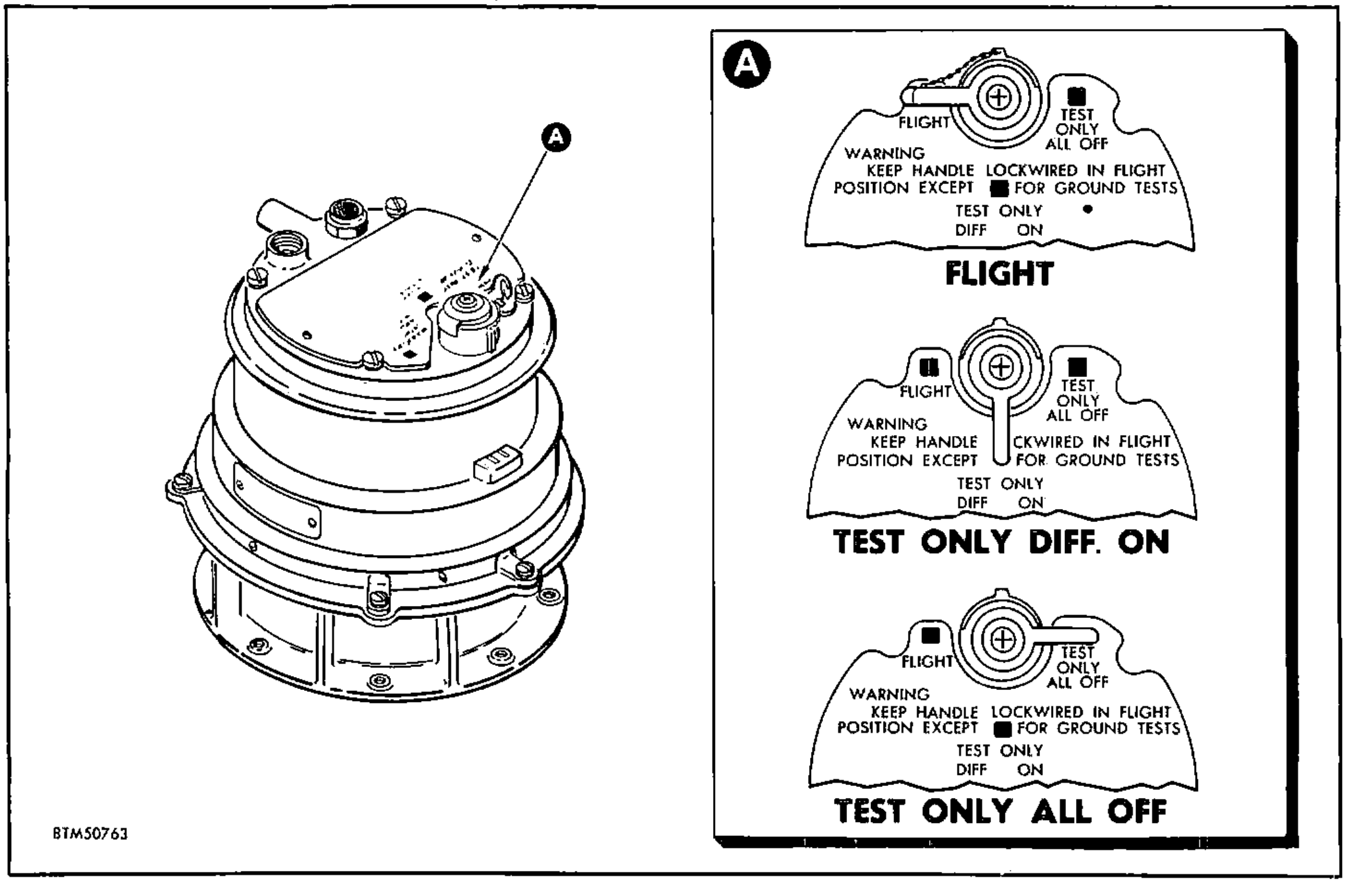


Figure 3-23. Cockpit Pressure Regulator Test Handle

solenoid-controlled shutoff valve in this line. The manner in which the two lines are connected to the static lines and the discharge duct is shown in figure 3-22.

Cockpit pressure is sensed through a filtered opening next to the static pressure connection. A test handle on the regulator is normally lockwired in the FLIGHT position; however, as shown in figure 3-23, it has two other positions which are used when ground testing the cockpit pressurization system. They are marked TEST ONLY-DIFF ON and TEST ONLY-ALL OFF.

The regulator actually controls cockpit pressure in three stages. The first stage can be called "Unpressurized Operation," although cockpit pressure will actually be slightly above atmospheric pressure in this stage. This stage will last from take-off time until the airplane reaches an altitude of 10,000 feet. The second stage, called "Isobaric Operation," is in operation from an altitude of 10,000 feet to about 26,500 feet. Within this altitude range the regulator maintains the cockpit pressure constant at about 10,000 feet pressure altitude (about 10.2 psi).

The third stage is called "Differential Operation." The regulator is in this stage from an altitude of about 26,500 feet up to the maximum operating altitude of the airplane. In this stage the regulator maintains the cockpit at a pressure that is always about five psi higher than atmospheric pressure.

Regulator Operation.

Figure 3-24 is a schematic of the cockpit pressure regulator. The assembly consists essentially of a base casting and an outflow valve. The outflow valve is suspended from the base casting by a diaphragm and is mechanically guided by a guide post attached to the casting. The space above the diaphragm is a control chamber. Cockpit air pressure bears against the bottom of the diaphragm, while control chamber pressure bears against the top. The valve is balanced so that only a slight difference in pressure will cause the valve to move toward the side with the least pressure. If the pressures are equal, the diaphragm spring will keep the valve closed.

When cockpit pressure is higher than control chamber pressure, the valve will move up and air will leave the cockpit through the passages in the base casting. Note in the schematic that there are three pressure sensing openings to the control chamber. Cockpit air enters through a filtered orifice, while the main port opens the chamber to atmospheric pressure. Control chamber air is vented overboard through this port

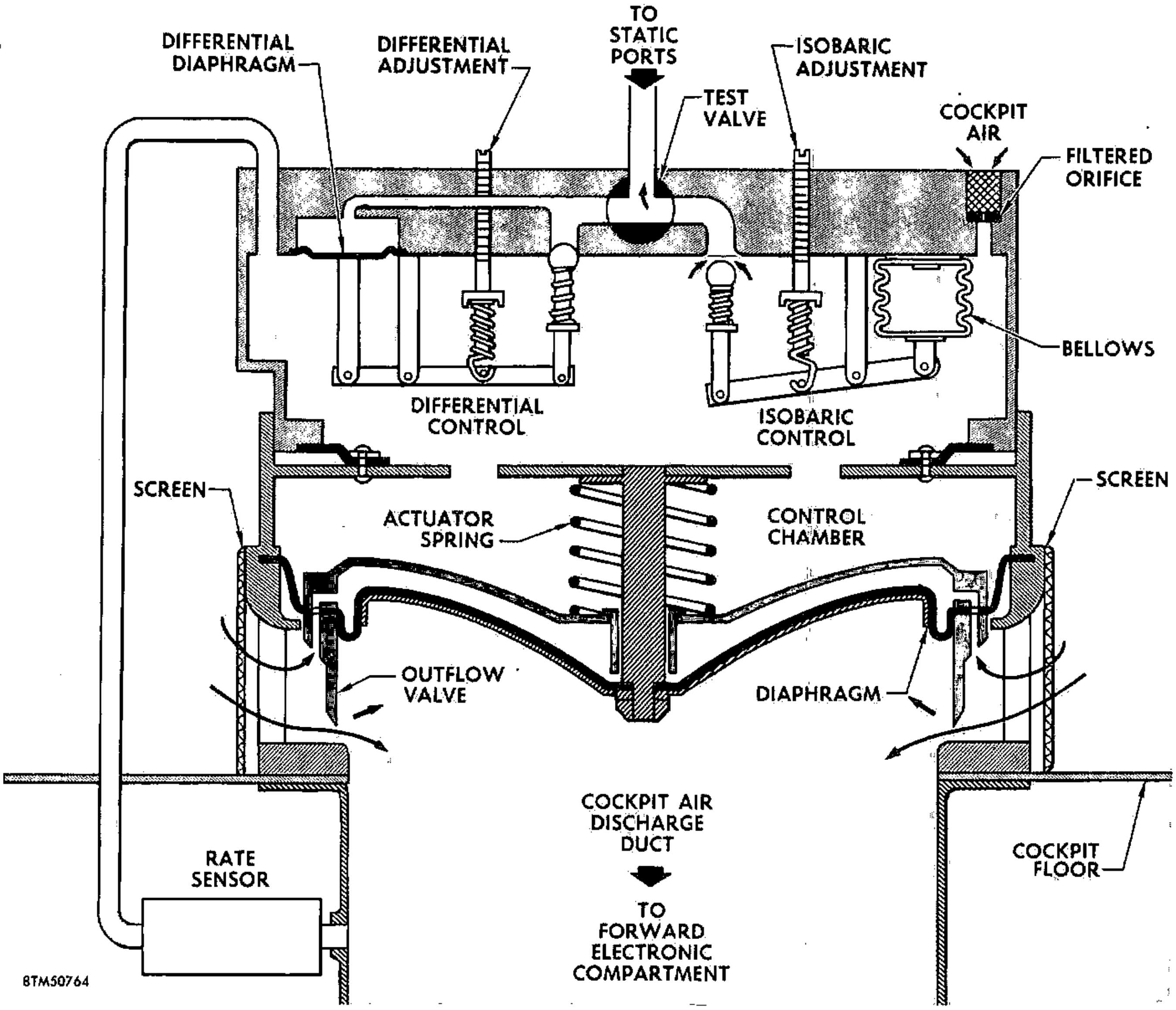


Figure 3-24. Cockpit Pressure Regulator Schematic

when atmospheric pressure is less than control chamber pressure. The third port connects to the discharge ducting through a sensing line.

A check valve keeps this line closed except when pressure in the ducting exceeds cockpit pressure. In this case, the higher duct pressure enters the control chamber and closes the valve to prevent a reverse flow of air from the forward electronic compartment to the cockpit. A shutoff valve closes this line when the airplane is on the ground so that air from a ground air conditioning unit can flow from the electronic compartment to the cockpit.

The control chamber contains two control systems, the "isobaric control" system and the "differential control" system. The isobaric system includes an evacuated bellows, a metering valve and spring, a rocker arm, and a tension spring. The differential system consists of a diaphragm, a metering valve and spring, a rocker arm, and a tension spring. The control chamber housing encloses a three-way selector valve.

This valve is controlled by the test handle on the top of the regulator. It controls the vent to atmosphere from the isobaric metering valve and the vent from the differential metering valve. When these vents are closed, the regulator is inoperative and the cockpit can be pressure tested. In the FLIGHT position, both vents are open; while in the TEST ONLY-ALL OFF position, both vents are closed. In the TEST ONLY-DIFF ON position, the Isobaric Control System is inoperative and the Differential Control System can be ground tested.

At sea level, cockpit pressure entering the control chamber through the filtered orifice keeps the bellows contracted, and the isobaric metering valve stays open. Cockpit air enters through the orifice and is vented to atmosphere through the metering valve. Since the orifice is smaller than the metering valve opening, pressure in the control chamber becomes less than cockpit pressure and the outflow valve opens to permit cockpit air to flow to the discharge duct. As the airplane climbs toward 10,000 feet, the pressure in the control chamber becomes less and less until at 10,000 feet it is no longer enough to keep the bellows in the collapsed or contracted position.

When the bellows start to expand, the isobaric control stage of operation begins. As the bellows start to expand, it begins to rotate the rocker arm to move the metering valve toward its seat and restrict the flow of air from the chamber. Restricting the flow of air from the control chamber causes the control chamber pressure to increase in relation to cockpit pressure and the outflow valve moves toward the closed position. The higher the airplane goes, the more the bellows will expand. The more the bellows expands, the smaller the metering valve vent will become.

This results in a constant closing movement of the outflow valve as the airplane climbs from 10,000 feet to about 26,500 feet. Within this range, the cockpit pressure will be at a constant level of about 10.2 psi, which is about equal to atmospheric pressure at 10,000 feet. At about 26,500 feet the isobaric metering valve will be firmly closed and the isobaric control system can no longer operate.

From sea level to about 10,000 feet, the outflow valve will be wide open. From 10,000 feet to about 26,500 feet of altitude, the valve will gradually move toward the closed position. At about 26,500 feet the Differential Control System begins to operate and the outflow valve will gradually open as the airplane climbs.

As the airplane reaches 26,500 feet, the isobaric metering valve closes and the control chamber pressure becomes sufficiently higher than atmospheric pressure to move the differential diaphragm. The diaphragm rotates the differential rocker arm and the differential metering valve opens. This reduces control chamber pressure and allows the outflow valve to open wider under pressure from cockpit air. The higher the airplane goes, the farther the diaphragm will move and the larger the metering valve opening will get. This results in a progressive reduction of control chamber pressure with altitude, and a gradual opening movement of the outflow valve.

Maintenance.

Adjusting screws on the face of the regulator can be used to adjust the two rocker arm tension springs. Turning these springs changes the pressure setting of the regulator. This is a job for skilled maintenance personnel, and should not be attempted while the regulator is installed in the airplane. A regulator that is not operating properly should be replaced. Limit your maintenance work on this regulator to removing and cleaning the filter in the cockpit air port. The cockpit floor around the large filter screen, and the screen itself, should be kept clean at all times.

Do not use compressed air to clean the screen since the air pressure might damage the diaphragm. If a vacuum cleaner with a hose and brush attachment is available, use it to clean around the screen. The vacuum also should not be too powerful, since it too could damage the diaphragm. Make certain the test handle is in the FLIGHT position whenever you inspect the regulator.

Cockpit Safety Valve.

The cockpit safety valve is mounted on the aft bulkhead of the canopy. When it is open, it vents cockpit air overboard through the bulkhead and through dump louvres on the canopy skin. Note that this valve is very similar to the cockpit pressure regulator in appearance and function. It senses and compares cockpit and atmospheric pressures, and opens to permit cockpit air to vent overboard when cockpit pressure reaches certain limits. Since these regulated pressure limits are higher than those of the cockpit pressure regulator, the safety valve will not ordinarily function unless the regulator is defective. It has three sensing ports, one of which allows cockpit air to enter the control chamber through a filter screen.

The other two openings are connected to atmospheric pressure through sensing lines to the nose wheel well sensing port connections. The sensing lines in the nose wheel well were shown in figure 3-22. The main sensing port is connected to atmosphere directly, but the other port, the "dump" port, is connected to atmosphere only when the shutoff valve in its sensing line is open. When the shutoff valve is open, the safety valve opens wide to dump cockpit air overboard.

The shutoff valve is normally closed, but is automatically opened when the airplane is on the ground and when the cabin air switch is in the RAM position. A test handle on the safety valve is normally lock-wired in the FLIGHT position. It also has two other positions. One is marked TEST ONLY-SEC. DIFF ON, and is used for ground testing. The last position is the extreme counterclockwise position and is not marked. When the handle is in this position, the safety valve will remain shut and the air pressure could build up and damage the cockpit. NEVER use this position.

Valve Operation.

The safety valve controls cockpit pressure in three stages, as does the pressure regulator. The first stage is

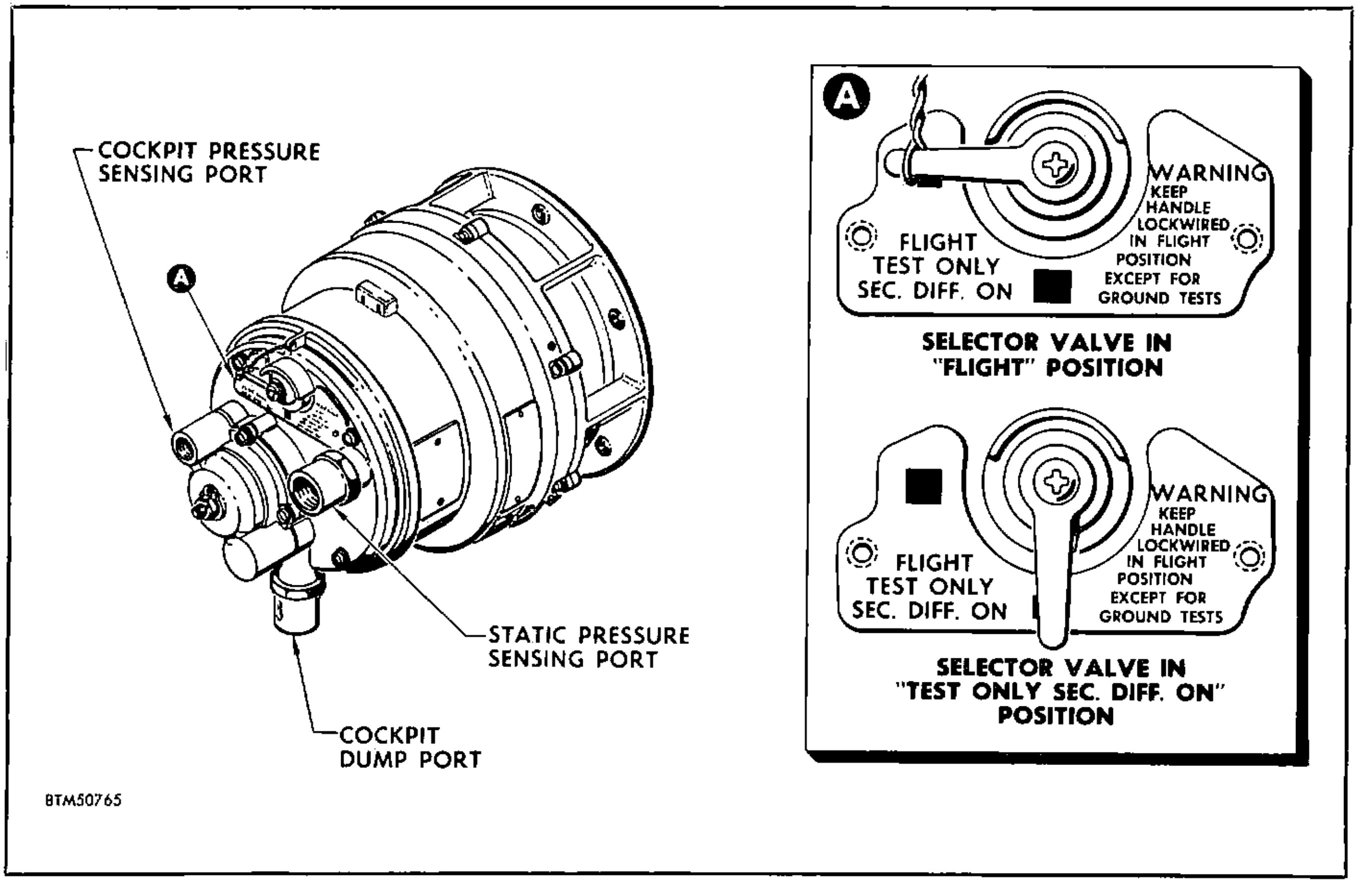


Figure 3-25. Cockpit Safety Valve

called "Primary Differential Operation" and is comparable to "Unpressurized Operation" in the pressure regulator. This stage lasts from takeoff up to about 13,000 feet of altitude. Within this altitude range the cockpit pressure will stay about 1.5 psi over atmospheric pressure.

The second stage is called "Isobaric Operation" and is exactly the same as the second stage of the pressure regulator, except that it begins at about 13,000 feet and maintains the cockpit pressure constant from that altitude to about 26,500 feet. The third stage is called "Secondary Differential Operation" and is exactly the same as the third stage of the pressure regulator. The only difference is that the cockpit pressure would be maintained at about 5.25 psi over atmospheric pressure, instead of 5 psi as for the pressure regulator.

The simplified cross section of the safety valve, figure 3-26, shows how closely the safety valve resembles the pressure regulator. The "Isobaric Control" systems in both components are identical. The "Secondary Differential Control" system of the safety valve is identical with the "Differential Control" system of the pressure regulator. These two systems in the safety valve operate exactly as the comparable systems in the pressure regulator.

There is also a dump port that is open to atmosphere when the shutoff valve is open. When that happens, control chamber pressure becomes much less than cockpit pressure and the outflow valve opens to permit cockpit air to flow to atmosphere.

The main difference between the regulator and the safety valve is that a third control system is added to the safety valve. This system is called "Primary Differential Control" and operates from sea level to 13,000 feet altitude range. This system is composed of a poppet valve, a diaphragm, and two springs. At sea level and unpressurized operation, the diaphragm spring will keep the poppet valve open. As the airplane climbs, cockit pressure will exceed atmospheric pressure, thus causing the poppet spring to move the poppet valve toward the closed position. This reduces the flow of cockpit air to the control chamber and thus reduces chamber pressure.

The isobaric metering valve remains wide open during this stage. As the chamber pressure is reduced, cockpit pressure against the outflow valve diaphragm will cause the outflow valve to move toward the wide open position. As the airplane continues to climb, the poppet valve continues to move toward the closed position, thus reducing control chamber pressure and causing

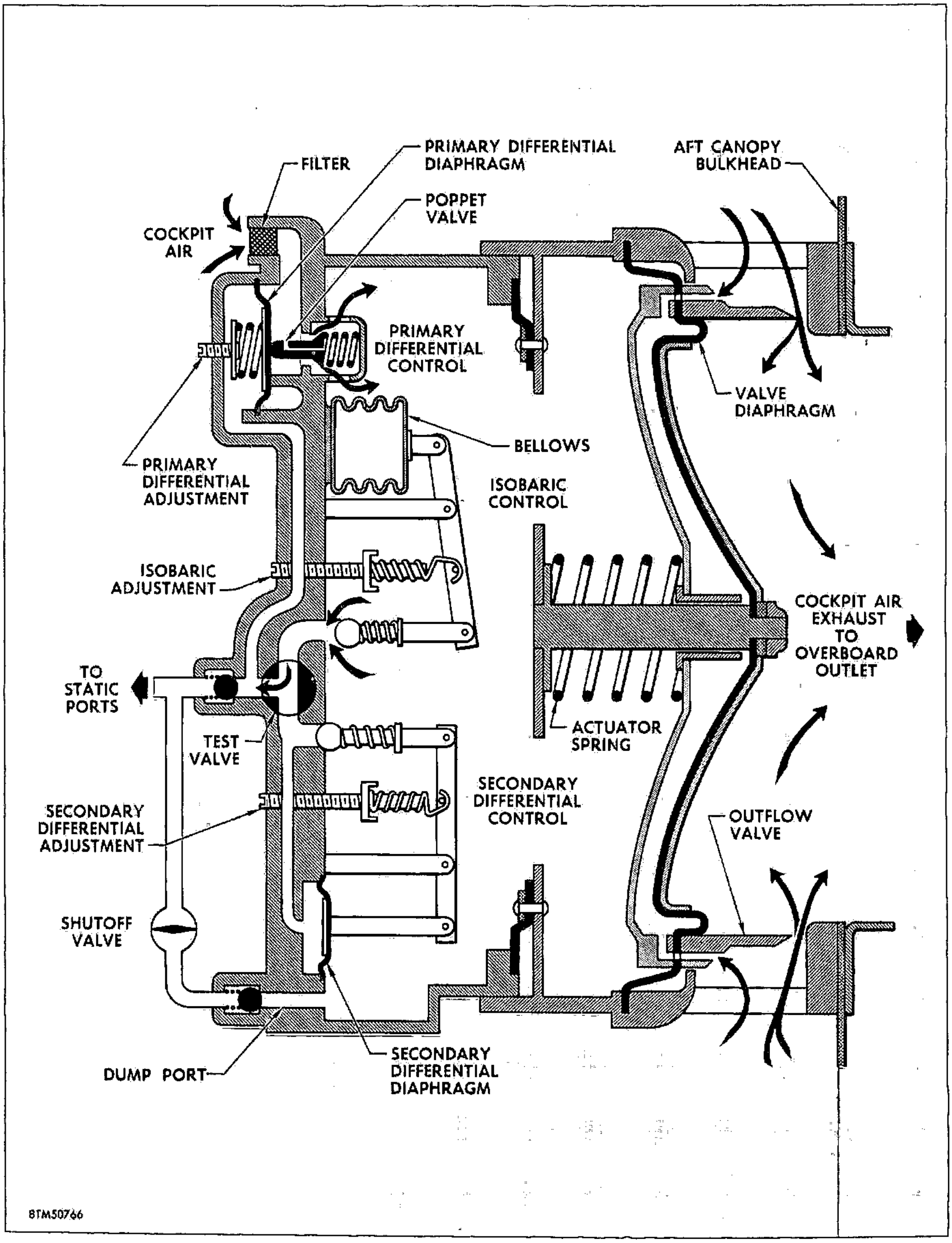


Figure 3-26. Cockpit Safety Valve Schematic

the outflow valve to open wide. This action maintains cockpit pressure at about 1.5 psi over atmospheric pressure.

At 13,000 feet the poppet valve will be firmly closed and cockpit air can enter the chamber only through the orifice in the poppet valve. As the airplane continues to climb, the isobaric bellows begins to expand and the Isobaric Control system operates in the same manner as for the pressure regulator. At about 26,500 feet, the Secondary Differential Control system begins to function in the same manner as the Differential Control system of the pressure regulator.

After 13,000 feet, the only difference in operation between the safety valve and the pressure regulator is the fact that the safety valve would maintain a slightly higher cockpit pressure. If the pressure regulator were to remain closed, the safety valve would open gradually as the airplane climbed from sea level to about 13,000 feet. From that altitude to about 26,500 feet the valve would slowly close. As the airplane climbs from 26,500 feet to its maximum altitude, the valve would slowly open.

Maintenance.

Adjusting screws on the face of the safety valve can be used to adjust the pressure settings of the valve. NEVER attempt to adjust the safety valve. Replace a valve that is not operating correctly. Limit your maintenance work on the safety valve to removing and cleaning the filter in the cockpit air port. Keep the area around the outflow valve clean, but do not use compressed air or you might damage the diaphragm. Use a low-powered, household-type vacuum cleaner, with a hose attachment, to clean around the valve. Be sure that the test handle is in the FLIGHT position before the airplane takes off, or the airplane structure will be severely damaged if the pressure regulator should fail.

The Rate Sensor.

The cockpit rate sensor is mounted on the aft side of the cockpit canted bulkhead. A small hole in the bulkhead allows the unit to sense cockpit pressure.

The rate sensor controls the opening rate of the flow control valve to prevent too rapid a build-up of cockpit pressure after a period of unpressurized operation. As the name implies, the rate sensor, which is shown in figure 3-27, senses the change in cockpit pressure. Not only does it sense the change, but it causes the flow control valve to close momentarily to reduce the rate at which the pressure increases.

The valve actuator will move the valve toward the closed position when air pressure in one of the actuator control chambers is reduced. The rate sensor drains air from this chamber to reduce the control chamber pressure any time there is a sudden increase in cockpit pressure.

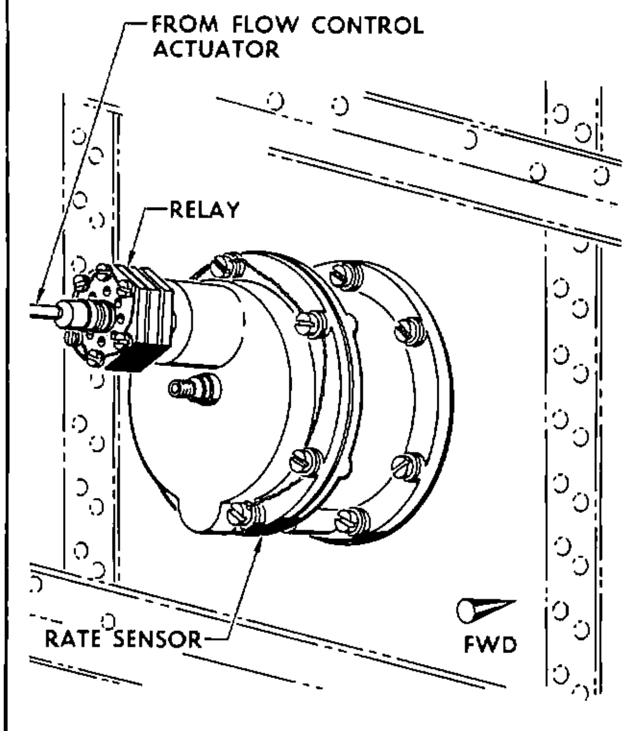


Figure 3-27. Rate Sensor

The rate sensor assembly consists essentially of two pressure chambers and a relay valve (figure 3-28). The relay valve is really a separate unit that is attached to the rate sensor. It is through the relay valve that air from the control chamber of the flow control valve is drawn and vented from the system.

Rate Sensor Operation.

The rate sensor itself consists of two control chambers separated by a diaphragm. The smaller chamber, marked "A," is connected to cockpit air, and its pressure will change at the same rate that cockpit pressure changes. The other chamber, marked "B," is connected to cockpit air through very small openings that restrict the air flow to such a low rate that there is a time lag between a cockpit pressure change and the same change in chamber B pressure.

When cockpit pressure increases, the pressure in chamber "A" will exceed that in chamber "B" and the diaphragm will move toward chamber "B." A small valve that normally blocks the small orifice will move to unblock the orifice when the diaphragm moves. The

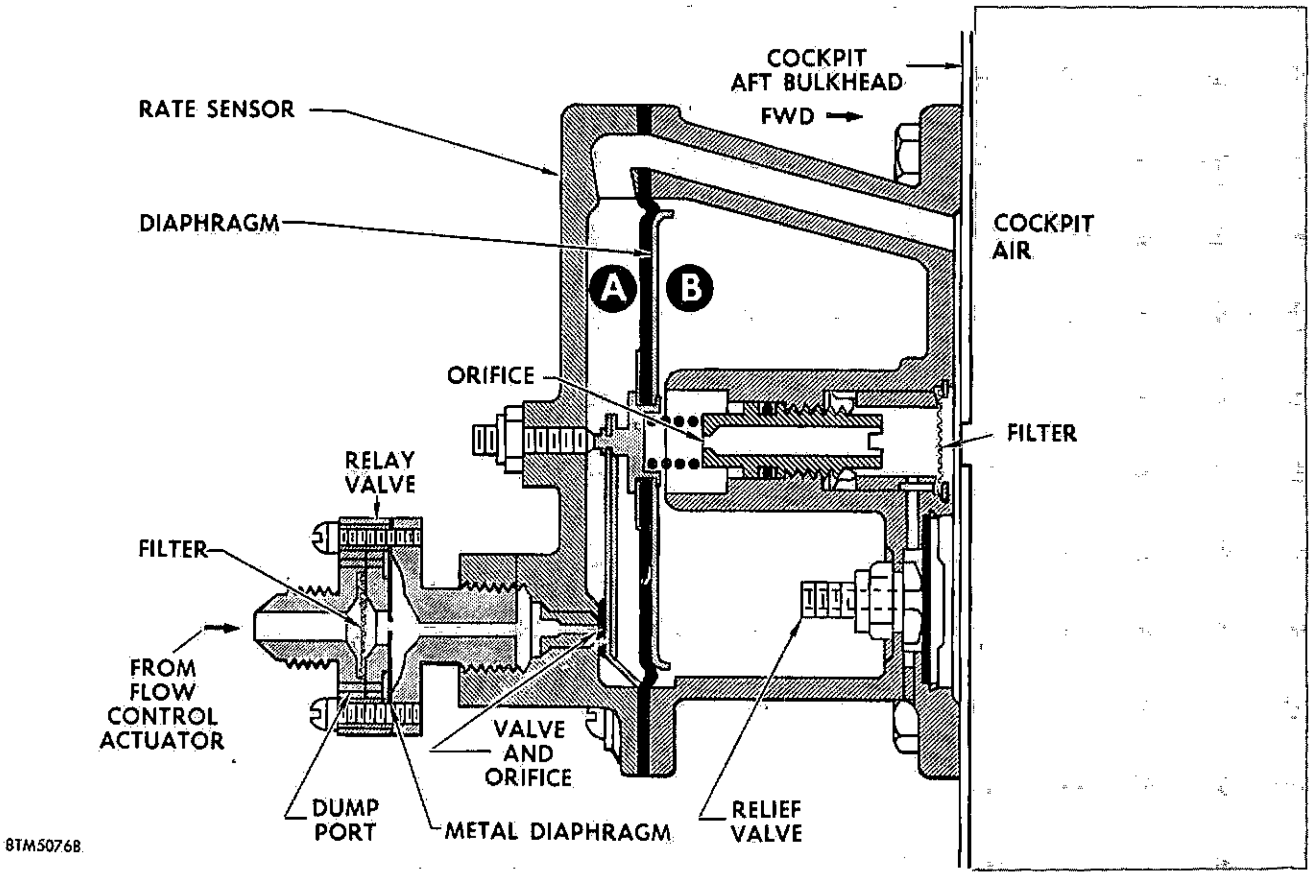


Figure 3-28. Rate Sensor Schematic

orifice is connected to another small pressure chamber in the relay valve. The relay valve has a small metal pressure diaphragm which blocks a circular ring of small dump ports. Air pressure from the flow control valve actuator bears against one side of the diaphragm.

Since there is a small hole in the diaphragm, the air leaks into the small pressure chamber until the pressure on both sides of the diaphragm is equal. Since the diaphragm has more area exposed to the pressure chamber than it does to the upstream pressure, the diaphragm will be pressed away from the chamber and will block the ring of dump ports. When the rate sensor diaphragm moves toward chamber "B," and the small valve unblocks the orifice, air from the pressure chamber in the relay valve is vented into the rate sensor. The relay valve diaphragm is then forced into the pressure chamber and air from the flow control valve actuator is allowed to vent from the system through the dump ports.

The flow control valve closes, cockpit pressure stops building up, the rate sensor diaphragm moves toward chamber "A," and the small valve blocks the orifice. Pressure in the relay valve pressure chamber then builds up until it forces the relay valve diaphragm to block the dump ports. The flow control valve will again open and the cycle will be repeated until a further increase of cockpit pressure is prevented by the pressure regulator or safety valve.

Maintenance.

The control of the rate of cockpit pressure increase by the rate sensor depends upon some very precise adjustments. The one thing most destructive to its proper operation is an air leak. A leak any place in the system —from the pressure connection at the flow control valve actuator to the rate sensor itself—can cause the flow control valve to close or to fluctuate. If the pilot reports cockpit pressure surges, you should first check carefully for leaks by using a soapy solution.

The relay valve itself is the part most likely to leak. It is connected to the rate sensor by pipe threads. The connection must be tight. The relay valve has three circular parts which are held together by screws. There are no gaskets between these parts; however, the steel diaphragm acts as a gasket.

A filter is found at the other joint. If these screws are loose, air leaks may develop around the joints; so you should make certain the screws are tight. Other than checking for leaks, do not attempt repairs on a defective rate sensor; instead remove it and install a new one.

Pressure Fittings.

One of the chief causes of difficulty in the pressure regulation system is air leakage at pressure sensing line connections. This applies to all sensing lines; however, the pressure line between the flow control valve actuator and the rate sensor relay valve is the most sensitive. This line is under high pressure and even small leaks could cause the flow control valve to close at the wrong time.

The atmospheric pressure sensing lines and the sensing line between the cockpit pressure regulator and the cockpit discharge ducting are also important.

All pressure sensing lines in the pressure regulation systems are made of one-fourth inch aluminum tubing joined by standard AN pressure tubing connections. Figure 3-29 shows a typical connection.

There are two chronic troubles encountered with the installation of AN couplings: overtorqued connections and stripped threads. You will find it best to connect AN couplings by using hand pressure only; then if you do have the threads crossed, you can detect it before any damage is done. The worst problem, however, is overtorquing.

Sooner or later all mechanics must become familiar with torquing procedures and values. Torquing is the process of tightening bolts, nuts, screws, and tubing connections to the correct point without damaging the threads by using excessive pressure. Torque values are merely the measurement of how much force you must apply to the bolt or nut to give a secure connection without stripping the threads.

The force varies with the material and with thread sizes. Since all mechanics do not have the same strength, it is necessary to have a standard method whereby each mechanic can determine "how tight is tight." Your F-102A Maintenance Technical Orders (See T.O. 1F-102A-2-3) specify the amount of force or torque to be applied to a particular part to insure a tight connection without causing damage.

This amount of force will be given in either inch-pounds or foot-pounds. For example, ten inch-pounds of torque means one pound of force applied at the end of a lever ten inches long. In a similar manner, ten foot-pounds means ten pounds of force applied at the end of a lever one foot long. A torque wrench may be calibrated in either inch-pounds or foot-pounds. The length of the handle will be constant; therefore a spring and ratchet arrangement in some wrenches will determine the amount of force that can be applied before the ratchet will slip. The amount of force that can be applied before slippage can be varied in many torque wrenches. Other wrenches have a dial which registers the torque being applied.

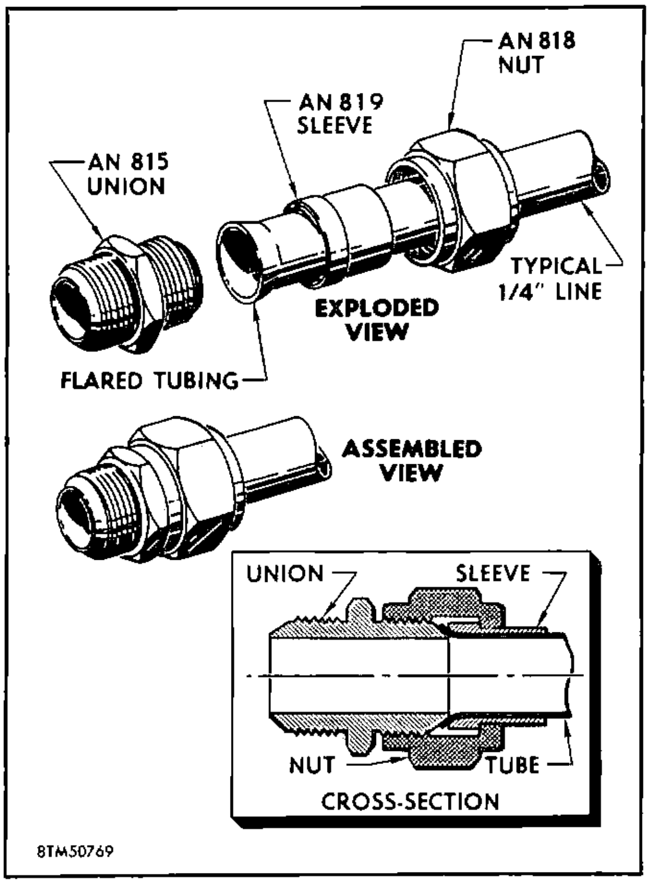


Figure 3-29. AN Pressure Coupling

MAINTAINING THE SYSTEM.

The failure of any part of the cockpit air conditioning and pressurization control systems will produce certain trouble symptoms. In this respect, your job as a maintenance man is similar to that of a doctor. It will be your responsibility to examine the system and discover the symptoms. You will then have to diagnose the cause and "operate" to correct it.

There are several ways in which you can detect trouble symptoms. You can check the different components visually; the valves, regulators, ducting, pressure lines, etc., for evidence of air leakage, ruptures, or other damage. The best source of information concerning trouble symptoms is the airplane Form 1A after a flight or pre-flight "runup." All troubles which the pilot or the crew chief have noted are entered on this form. Some troubles in the system are indicated to the pilot by the refusal of the circuit breakers in the different control systems to stay in the closed position.

This could indicate that some part of the control wiring is shorted or grounded in such a manner as to overload the circuit. Warning lights on the panel indicate electrical power failures that would affect the cockpit temperature of air control systems. The more common troubles noted by the pilot would be a failure of the system to maintain the cockpit at a comfortable temperature or to prevent pressure surges. Failure of the system to change the air flow as the cabin air switch is manipulated may also be reported.

The pilot might also note that the pressurization control system does not regulate cockpit pressure to the proper level for the particular altitude he is flying. All such defects will be entered on the form and it is your job to find the specific trouble and correct it. From the description of the condition, you must determine the actual cause of the trouble, perform an operational check to make certain that your diagnosis is correct, and then remedy the defect.

INSPECTIONS.

The F-102A Inspection Technical Order (T.O. 1F-102A-6) describes the inspections that must be performed on the airplane. In general, inspections are divided into three categories: Preflight, Postflight, and Periodic. In the section on preflight inspections you should become familiar with all of the items dealing with the Low-Pressure Pneumatic System. As far as the Cockpit Air Conditioning and Pressurization System is concerned, perhaps the most important preflight inspection concerns the test handles on the cockpit pressure regulator and the safety valve. *Both* test handles *must* be lockwired in the FLIGHT position, or the cockpit pressurization will be erratic, or the cockpit may overpressurize resulting in serious structural damage.

Certain postflight inspections must be performed after each flight. Study the items listed in the Inspection T.O. under Postflight Inspections and check each item carefully for any indication of a defect or malfunction. Many times minor defects can be spotted and corrected before they can become serious.

In addition to the regular preflight and postflight inspections, certain components or systems are to be inspected at regular intervals of flight time. For instance, a certain group of items must be inspected every 50 hours of flight time, while others are inspected every 100 hours or every 1000 hours. The Inspection Manual (T.O. 1F-102A-6) tells you what items to inspect and what to look for; however, the Maintenance Manual (T.O. 1F-102A-2-6) will give more detailed instructions. Study the instructions carefully and consult your crew chief if you have any doubt about how to proceed.

OPERATIONAL TESTS.

Whenever a certain system or component of a system is suspected of being defective, it is often a good idea to perform an operational test to pinpoint the trouble in a particular component. This is especially true of the pressurization control system. It is easy to assume that one or the other of the components of a system is the cause of the trouble. However, if you do not perform an operational test to make certain, you might replace a perfectly good component only to find the system will still not operate properly.

These operational tests are described in all necessary detail in T.O. 1F-102A-2-6. The tests that concern the Cockpit Pressurization and Control System are the Cockpit Pressurization Test and the Cockpit Temperature Control Test. These tests must be performed whenever a component of the system to be tested has been removed, replaced, or adjusted.

Cockpit Pressurization Test.

This test includes the functional testing of the cockpit safety valve, pressure regulator, and rate sensor. The test will also indicate how well the cockpit is pressure sealed. Before performing the test, refer to Chapter V for instructions on checking and testing the canopy seal. Make certain that the canopy seal is functioning properly before proceeding. The test itself is described in T.O. 1F-102A-2-6.

The airplane must be resting on the ground while you perform this test. If it is jacked up, there will be a certain amount of warping that will change the rate of air leakage through the structure. The system is designed so that a certain small amount of leakage will not have a serious effect on the level of cockpit air pressure.

This test is designed to test the operation of the system under flight conditions. The main landing gear safety switch must be manually placed in the airborne position or the shutoff valve in the safety valve sensing line will be open. This shutoff valve must stay closed or the safety valve will remain open and cockpit pressure cannot be built up. The safety switch is mounted on the left main landing gear. Figure 3-30 indicates the ground position and airborne position of the switch, and shows how to manually place it in the airborne position.

During these tests the engine will not be operating and the flow control valve and the cockpit ram air valve will be closed. Therefore, a source of external air pressure must be connected to the cabin pressure test fitting beneath the cockpit floor in the nose wheel well. This fitting and the other two test fittings we are concerned with here are shown in figure 3-31.

Another source of pressure must be connected to the canopy seal test line fitting so that the canopy may be pressure sealed. A pressure measuring instrument must be connected to the fitting marked STATIC PRESSURE CONNECTION. Make certain that you make these three connections as tight as possible, or the test may not give you reliable test data.

It is very important that you observe all safety precautions specified in the technical order. The cockpit will be under pressure and a structural failure could cause severe injury to you or any bystanders.

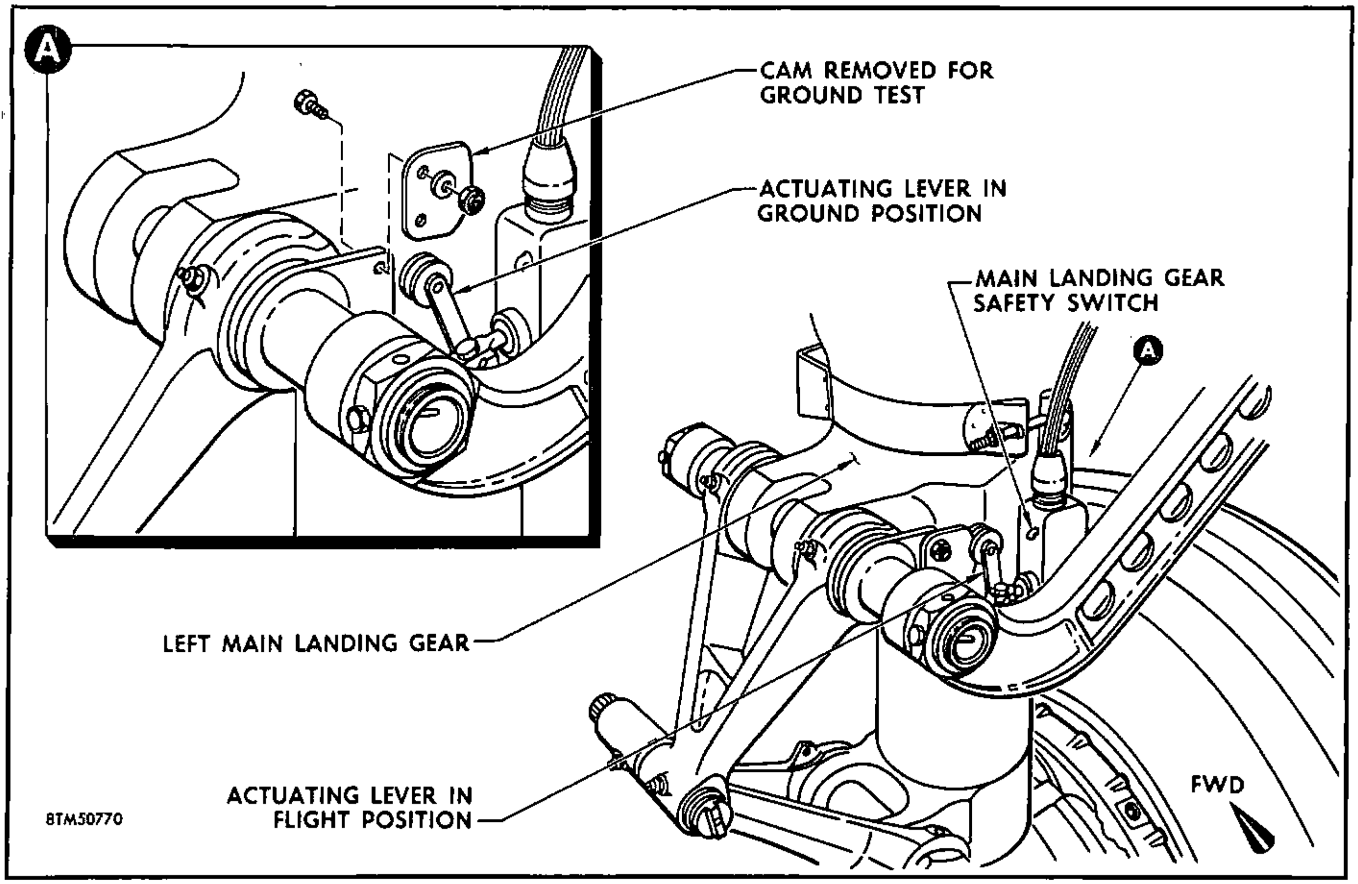


Figure 3-30. Main Landing Gear Safety Switch

A safety valve set to relieve at 8 psi must be installed at the fuselage fitting inside the small access door below the right windshield panel. This will prevent damage to the structure if the cockpit safety valve malfunctions and fails to relieve the pressure at 5 psi.

The test is conducted in several stages. At each stage, the test handles on the safety valve and the cockpit pressure regulator must be in certain specified positions. After conducting the tests, be very careful to lockwire these handles in the FLIGHT positions, or the system will not operate correctly in the air and the system might overpressurize the cockpit and damage it severely. It could even cause the loss of both the airplane and the pilot.

If the cockpit fails to pressurize, check all ducting leading to the cockpit for leaks. Remember that the cockpit cannot be pressurized unless the cockpit ram air valve is firmly closed, so check the position of this valve if the pressure will not build up.

Be sure to move the ground safety switch to the ground position before opening the cockpit after a pressure test. This will cause the safety valve to open and dump the cockpit pressure. If this is not done, some damage to equipment or injury to personnel could result when the canopy is unlatched. Be certain to leave the safety

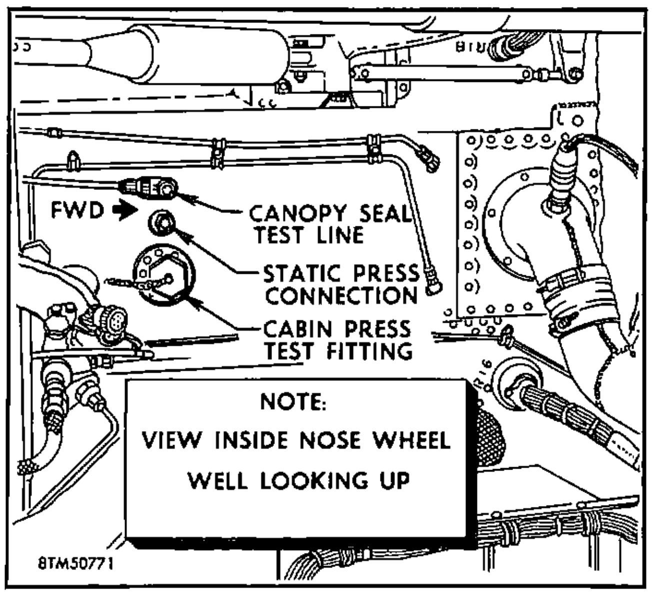


Figure 3-31. Cockpit Pressure Test Fittings

switch in an operating condition after these tests, or the cockpit may not pressurize in the air.

After performing these tests, disconnect external pressure lines and the pressure indicator from the fittings

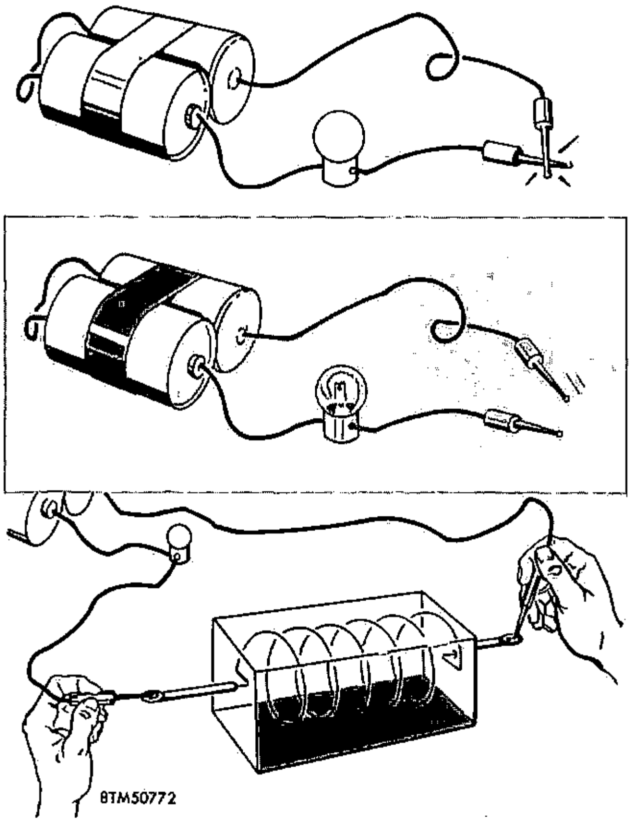


Figure 3-32. Continuity Light

in the nose wheel well. Be *certain* that the fittings are tightly capped after completing these tests.

Cockpit Temperature Control Test.

This test is an electrical test of the cockpit heat control box and the duct thermostats. Follow the instructions in T.O. 1F-102A-2-6 carefully and replace those components found defective. Do not attempt to repair or adjust the control box, but instead replace it as a unit. Retest the system after replacing a component. If there are any opens, shorts, or other defects in the wiring, follow the procedures given below for checking electrical wiring for continuity and shorts.

Testing Electrical Circuits.

The continuity light and voltmeter are the items that you will use most often in checking electrical systems. Continuity lights are generally used to check for open circuits. As shown in figure 3-32, a continuity light consists of two flashlight batteries, a bulb, and two leads. Each lead has a contact probe. If you touch the contact probes together, the light will come on. When the contact probes are apart, the light will be off. Whenever you desire to check a circuit for *opens,* touch one end of the circuit with one probe and the other end with the other probe. If the circuit is continuous, the light will come on. Make certain that there is no current in the circuit before using the continuity light. If you must test the circuit while the power or current is on you will need a voltmeter.

Voltmeters are designed to measure potential differences; that is, they indicate the voltage drop across certain portions of the circuit. Figure 3-33 shows a typical voltmeter used in checking circuits. The voltmeter has two leads with contact probes similar to those of the continuity light. When using a voltmeter, always connect it in parallel with the electrical circuit being tested. If you should connect the voltmeter in series no harm will result to the voltmeter since the current will be limited by the high internal resistance of the meter. However, since this is true, the components of the circuit will not operate.

Always be careful to use the right voltage scale for the voltage range you expect in the circuit. The voltmeter will have several scales and a switch with which to select the desired scale. It is always a good idea to place the switch at the highest voltage position at first to avoid damage to the voltmeter. If the voltage indicated is less than the range of the next smaller scale, then you can switch to that scale for maximum sensitivity.

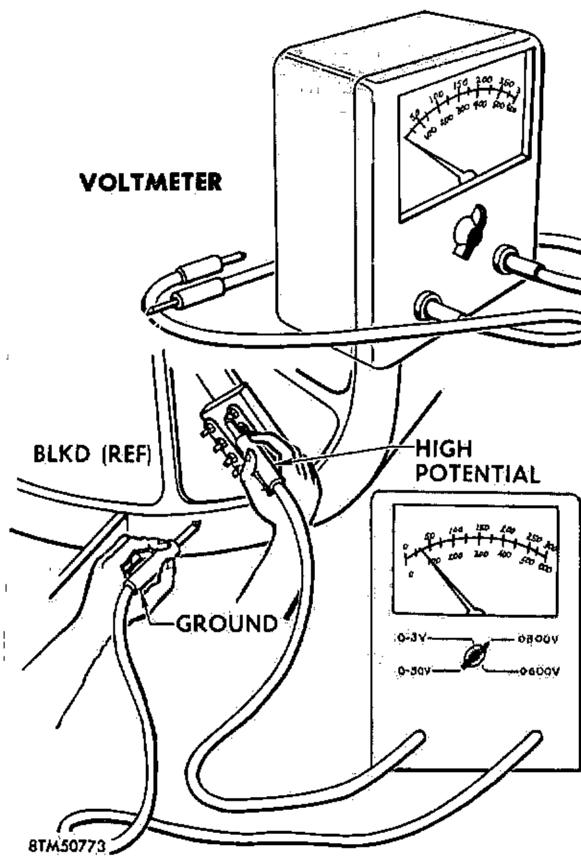


Figure 3-33. Voltmeter

Chapter IV

ELECTRONIC COMPARTMENTS VENTILATION SYSTEM

Contents *Page*

In previous chapters of this manual you learned about the air supply systems of the F-102A airplane. You also learned about the cockpit air conditioning and pressurization system. In this chapter, we will discuss the ventilation systems of the various electronic compartments. As you should well know, electronic equipment generates quite a bit of heat. In some of the electronic compartments, particularly the forward compartment, the entire space is almost completely occupied by various electronic components.

When all of these components are operating at once, the compartment could become so hot that the equipment would overheat and function improperly or not at all. An airplane, such as the F-102A, cannot perform its mission unless its complicated electronic systems function properly. You can see, therefore, how important it is that a continuous flow of cooling air be maintained through the electronic compartments.

Operation of the electronic compartments ventilation system is very simple compared to the cockpit air conditioning and pressurization system. Ram air is conducted through ducting to the various compartments and its flow control is entirely automatic. There is no manual switch or other control for the pilot to worry about, and defects in the system may not be apparent to him. In detecting and analyzing troubles in the electronic ventilation system, you will receive much less help from the pilot than is the case with other systems.

You will need a very thorough knowledge of the system and its operation in order to maintain it. Before getting involved in details of the different components we will first describe the system as a whole and its operation under various conditions.

DESCRIPTION OF ELECTRONIC COMPARTMENTS VENTILATION SYSTEMS.

As you will recall from Chapter II, all air used to cool and ventilate the various F-102A electronic compartments enters the airplane through one of three openings. The most important of the three openings or intakes, as far as electronic cooling is concerned, is the left boundary-layer ram air intake. The other two openings are the large engine air intakes. Both engine air intakes join just forward of the engine to form one large intake. One system of distribution ducts connects to the boundary-layer intake and two systems connect to the engine intake just forward of the engine compressor.

In figure 4-1, note that these three systems are entirely separate from each other. Note that the boundary-layer system conducts air to the forward, intermediate, and upper electronic compartments. One of the two aft distribution systems conducts air to the aft electronic compartment, and the third system distributes air to the a-c and d-c generators and the IFF units. Neither of the aft systems has valves or regulators, and cooling air always flows through them as long as the airplane is moving or the engine is running.

The left boundary-layer distribution system contains a combination pressure regulator and shutoff valve that controls the flow of air through the system. This unit has a complicated external control system as well as

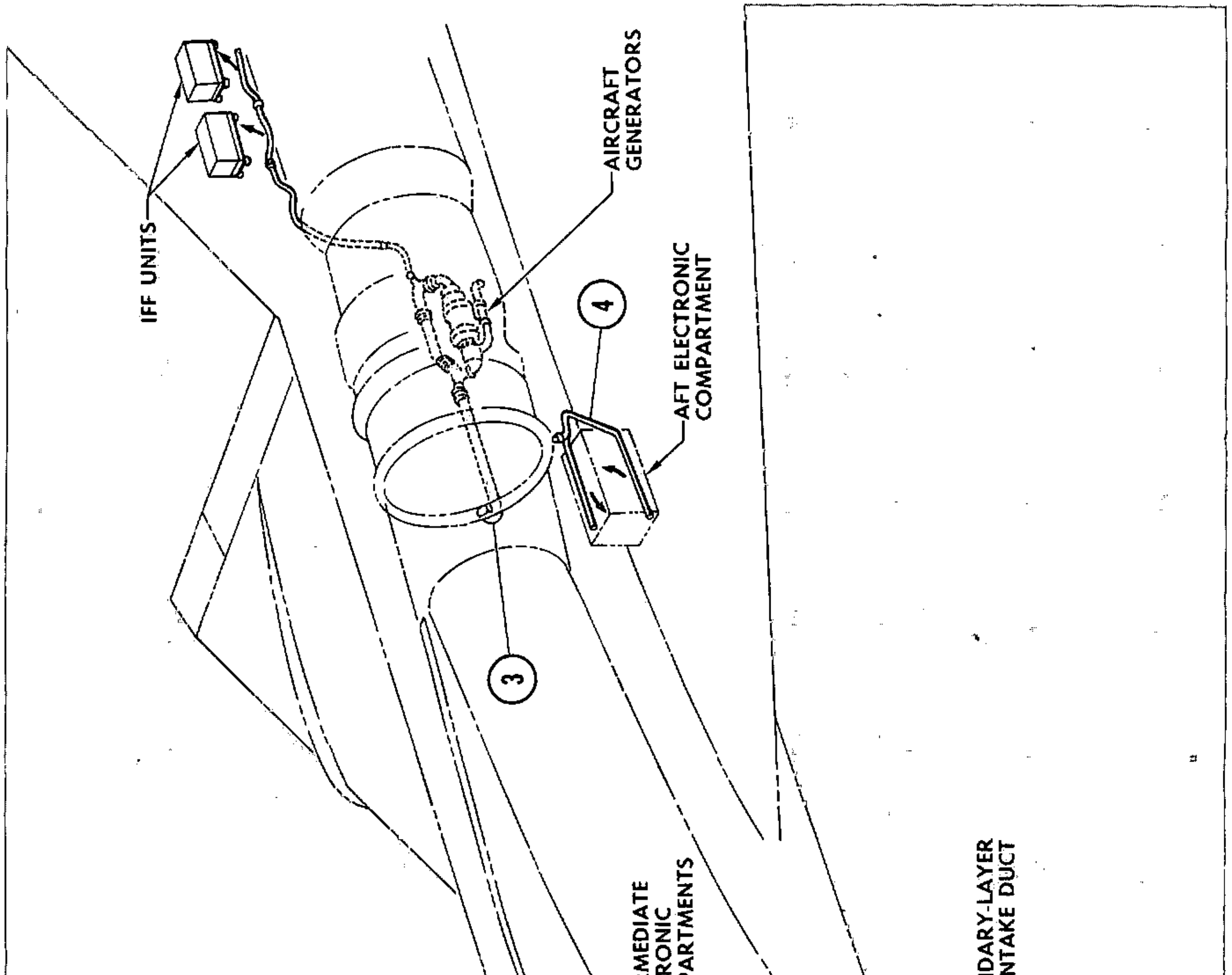


Figure 4-1. Electronic Compartment Cooling Air Distribution

integral, or "built-in," pressure and temperature control features. The valve shutter is normally open when the airplane is in flight; only during armament firing is it closed. Ordinarily, it is also closed when the airplane is on the ground.

When the engine is running at the IDLE throttle position on the ground, the valve is open to permit the jet pump ventilation system to operate. The position of the valve shutter is controlled automatically to regulate the pressure in the distribution ducting to about one psi over atmospheric pressure. The valve closes automatically when the intake air temperature reaches 71°C (160°F) to 74°C (165°F). In figure 4-2, note that the boundary-layer system has four "branches". One branch conducts air forward to the forward electronic compartment, and another vents air into various ports of the intermediate and upper electronic compartments.

Note that one part of the second branch carries cooling air to, and through, a motor-generator in the upper compartment. (This motor generator is part of the armament fire control system and may not be found installed in all F-102A airplanes.) A third branch contains the cockpit ram air shutoff valve which was discussed in the previous chapter. This branch connects to the cockpit air distribution ducting. The fourth branch (see figure 4-1) connects to the left boundary-layer ducting upstream of the pressure regulator and shutoff valve. It contains a check valve which permits air to flow from the left boundary-layer to the right boundary-layer ducting when the pressure in the left ducting exceeds the pressure in the right ducting. This check valve and duct section were described in Chapter II.

The forward electronic compartment contains two chambers or plenums. The upper plenum (light shading) contains electronic equipment, the lower plenum (shaded portion) acts as a manifold for even distribution of air to the upper chamber. Note that the cockpit discharge duct connects to the lower plenum. Air from the cockpit normally flows into the lower plenum (when the cabin air switch is at PRESS) to increase the amount of available cooling air. A fitting in the nose wheel well permits a ground air conditioner unit to be connected to the lower plenum. A number of openings in the upper plenum permit air to flow laterally from one compartment across to the other. Note also, in figure 4-3, that the upper plenum is connected to the nose wheel well through an opening in the bulkhead between them. Openings in the aft bulkhead of the nose wheel well allow a free flow of air to the intermediate and upper electronic compartment.

A relief valve connects the upper electronic compartment to the armament bay (see figure 4-4). All air from

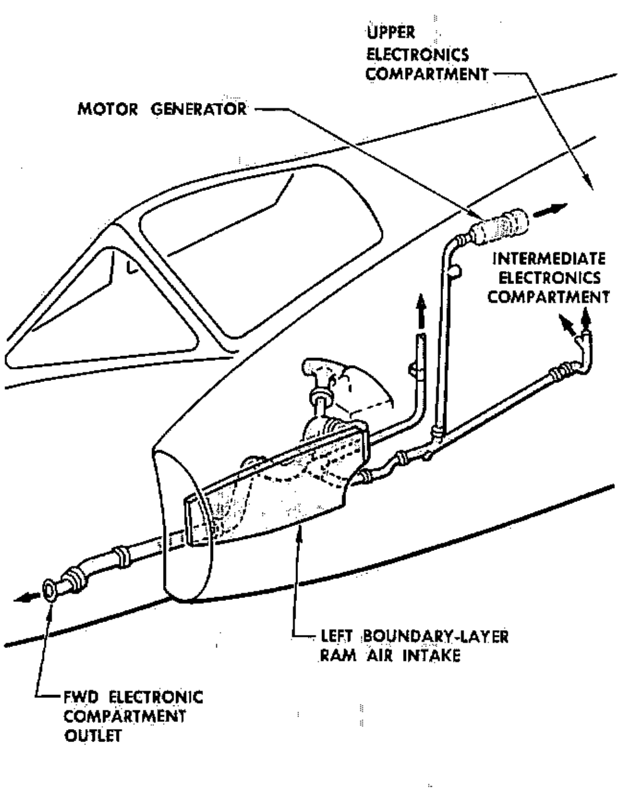


Figure 4-2. Left Boundary-Layer Ram Air Ducting

the forward, intermediate, and upper electronic compartments must leave the airplane through this valve (when the airplane is in flight) and through any normal leaks in the armament bay door hinges. The valve is open whenever the electronic compartment pressure is slightly higher than atmospheric pressure.

A check valve connects the upper electronic compartment to the left engine intake duct. This valve is the double-flapper type that is closed when the duct pressure is greater than upper electronic compartment pressure. It opens downward when the duct pressure drops below the electronic compartment pressure.

A jet pump is connected to the left boundary-layer intake. In figure 4-6, note that the jet pump consists of six nozzles which are connected through a shutoff valve to the bleed air supply line. The jet pump valve is open only on the ground when the engine is running at the IDLE throttle position. When hot bleed air flows out the nozzles, a suction effect is created in the distribution ducts and air is drawn from the forward, intermediate, and upper electronic compartments. Temperature sensing probes (not shown in the illustration) near the nozzles cause the jet pump valve to close when the airplane structure becomes too hot.

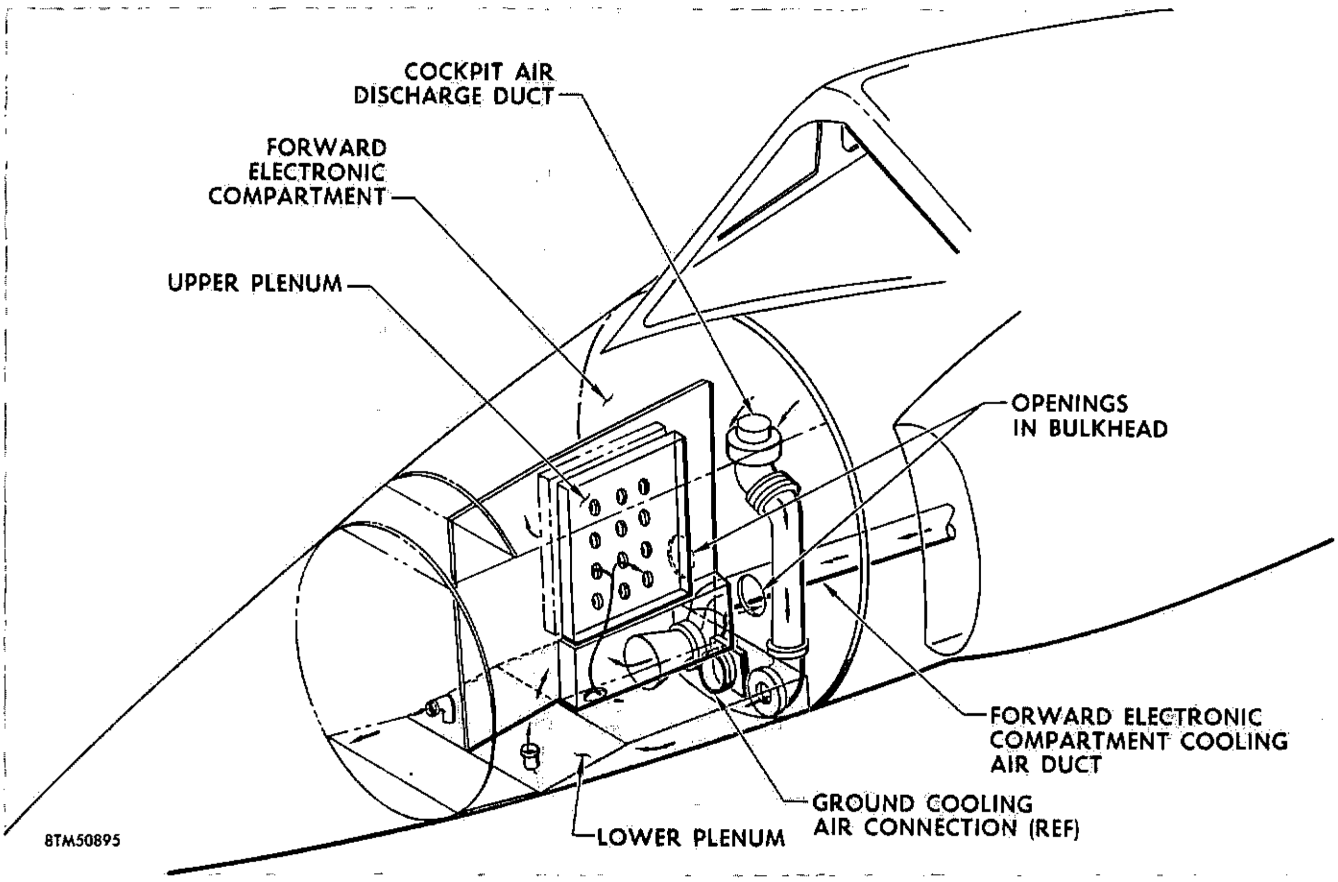


Figure 4-3. Forward Electronic Compartment

NORMAL OPERATION IN FLIGHT.

During normal flight, the ram air pressure regulator and shutoff valve opens and ram air flows through all three of the electronic-cooling distribution systems. When the cabin air switch is at PRESS, there is a flow of cockpit discharge air into the forward electronic compartment. This cockpit discharge air combines with ram air from the left boundary-layer intake to cool the forward, intermediate, and upper electronic compartments. All air from these three compartments leaves the airplane through the relief valve in the upper compartment and the armament bay door hinges. The relief valve maintains the air pressure in the upper electronic compartment at a pressure slightly above atmospheric.

Ram air from the engine intake flows through a small distribution system to the aft electronic compartment. This air is vented into the main wheel well, and then leaks overboard through the wheel well door hinges. Another distribution system conducts ram air from the engine intake "scroll" to the ac and dc generators (see figure 4-1). After passing through the housing of the two generators, most of the air in this system enters the engine accessory compartment and helps cool it. The rest of the air leaving the generators is conducted into the vertical fin section to cool the IFF units. This air is vented overboard through the fin by normal leakage.

In Chapter III, you learned that the cockpit is not pressurized when the airplane is at an altitude of less than 10,000 feet. In that case, if airplane speed and ram air pressure are fairly high, the pressure in the lower plenums is higher than cockpit pressure. The cockpit pressure regulator senses this condition and shuts off all air flow through the cockpit discharge duct. During this condition, the electronic compartments are cooled and ventilated solely by ram air.

The ram air pressure regulator and shutoff valve may close during flight for one of several reasons. The ram air may be too hot, the electrical current to the unit may be interrupted, or the regulator may become defective and cause the valve to close. When the valve is closed there is no ram air flow in the left boundary-layer distribution system and cockpit discharge air will flow through the system (when the cabin air switch is at PRESS and armament is *not* being fired) to maintain a minimum flow of cooling air to the forward, intermediate, and upper electronic compartments.

OPERATION DURING ARMAMENT FIRING.

The burning of fuel in the motor of a rocket or missile creates a cloud of highly corrosive dust and debris. If this dust were allowed to enter the electronic compartments, the electronic equipment could be seriously damaged. To avoid this danger, an air control timer in

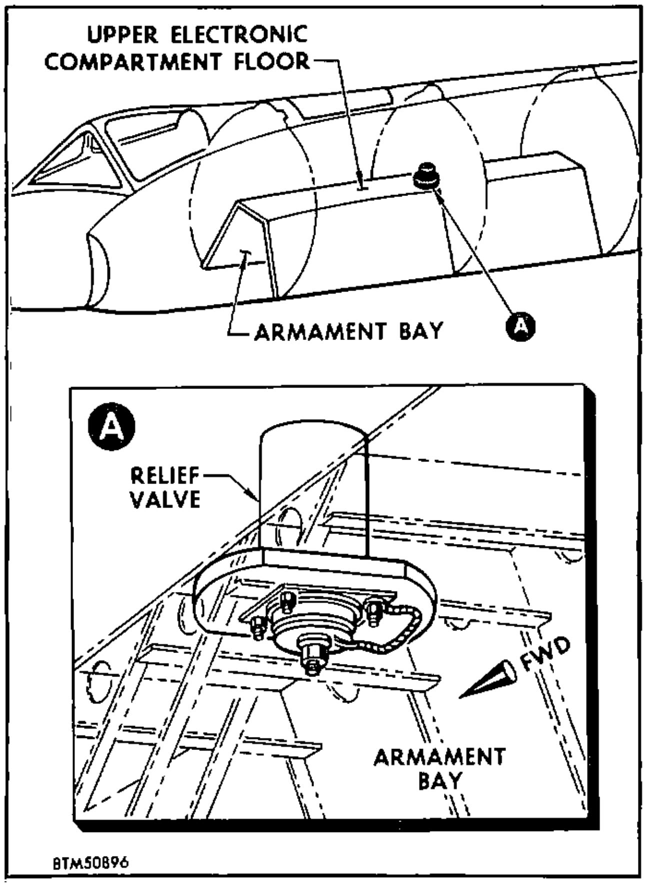


Figure 4-4. Intermediate and Upper Electronic Compartment Relief Valve

the armament system closes both the ram air pressure regulator and shutoff valve and the bleed air flow control valve. When the armament bay doors open, a switch sends an electrical signal to the air control timer. The timer in turn signals the flow control valve and the pressure regulator and shutoff valve, causing them both to close. This cuts off all air to both the cockpit and all electronics compartments, except the aft electronic and the IFF unit compartments.

After the armament is fired and the doors close, an electrical signal is again sent to the timer. Three or four seconds after receiving this signal, the timer signals the flow control valve and the pressure regulator and shutoff valve, causing them both to open again. The cabin air switch must be at PRESS before the flow control valve will open. However, the pressure regulator and shutoff valve will normally open as soon as it receives the signal. The flow of ram air to the aft electronic compartment, the a-c and d-c generators, and the IFF units is not interrupted during armament firing.

OPERATION ON THE GROUND.

Remember that the electronic equipment in the airplane will overheat quickly if it is operated without proper ventilation. When the airplane is in the air the electronic compartments are cooled by ram air. On the ground, however, the electronic equipment must often be operated for testing purposes. It may also be kept on "standby" operation before takeoff so that it can begin operating immediately without the need of a "warm up" period. At any rate, when electronic equipment is operated on the ground, special arrangements are necessary to prevent overheating.

On the ground, with the engine *off*, the forward, intermediate, and upper electronic compartments may be ventilated by a ground air conditioning unit. There are no provisions for ventilating the aft electronic compartment or the IFF units on the ground with the engine shut down; however, overheating can be prevented by opening their access doors before operating the equipment. When electronic equipment in the forward part of the airplane is to be operated, the ground air conditioning unit is connected to a fitting on the forward bulkhead of the nose wheel well. The ground unit then pumps conditioned air into the lower plenum of the forward electronic compartment (figure 4-3).

From the lower plenum some of the air flows into the upper plenum and overboard through an opening in the forward bulkhead of the wheel well. The nose wheel well door is, of course, open when the airplane is on the ground. Part of the remaining air flows into the

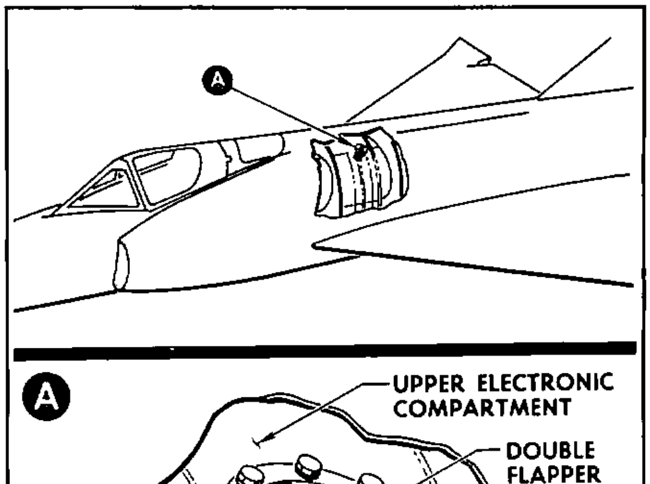


Figure 4-5. Upper Electronic Compartment Check Valve

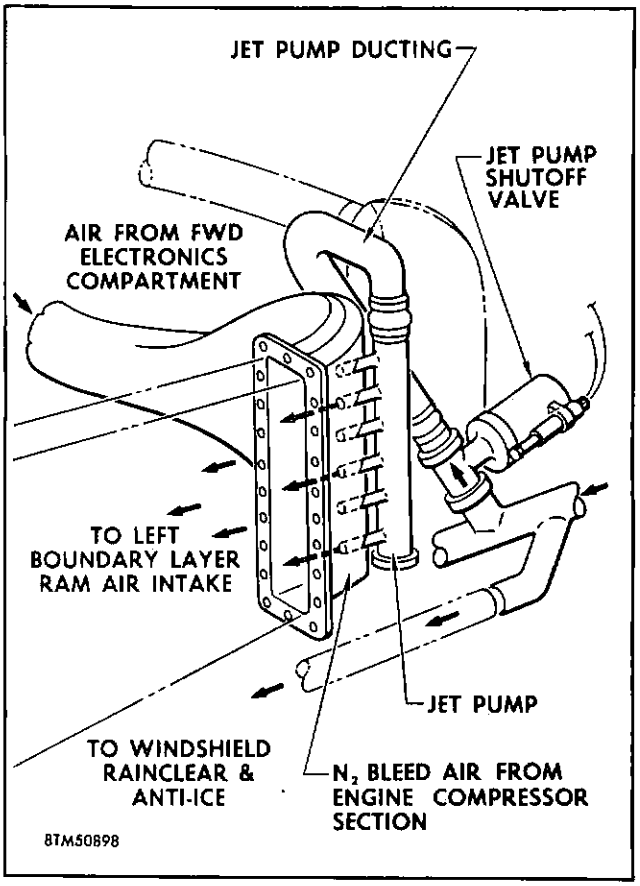


Figure 4-6. The Jet Pump

cockpit through the cockpit discharge duct and the rest enters the left boundary-layer distribution ducting; then the air flows through this ducting and is distributed to various parts of the intermediate and upper electronic compartments.

The pressure regulator and shutoff valve and the cockpit ram air valve are shut off when the engine is not running and the airplane is on the ground. This prevents the conditioned air from the ground unit from escaping overboard through the boundary-layer intake or into the cockpit through the cockpit air distribution ducting. This air flowing into the intermediate and upper compartments is vented overboard through openings in the aft bulkhead of the nose wheel well. The access doors to the intermediate and upper compartments can also be left open if additional ventilation is necessary.

When the airplane is on the ground with its engine *running,* the ground air conditioner cannot be used to ventilate the electronic compartments and all access doors must be kept closed. Since the engine may often run for some time before the airplane takes off (such as waiting for takeoff), some system for providing electronic compartment ventilation is necessary. Normally, the compartments are ventilated by air *forced* through the air distribution ducting by the forward motion of the airplane, or by the ground air conditioner. However, when the engine is running with the airplane parked or standing still, the electronic compartments are ventilated by *suction;* that is, air is drawn from the compartments and is replaced by more air from outside the airplane. This creates a continuous flow of cooling air.

The forward, intermediate, and upper electronic compartments are ventilated by suction created by the jet pump. As hot bleed air flows from the jet pump nozzles and leaves the airplane by the left boundary-layer intake (see figure 4-6), a suction is created in the distribution ducting and air is drawn from the three electronic compartments. This air is replaced by air drawn from the open nose wheel well. (The pressure regulator and shutoff valve will always be open when the jet pump is operating.) The aft electronic compartments, the IFF units, and the a-c and d-c generators are ventilated by suction in the engine air intake duct—in fact, air flow is a reversal of that shown in figure 4-1.

Normally, when the airplane is in flight, there is a positive pressure in the engine intake ducting which is caused by the ramming effect of the airplane's forward motion. However, when the airplane is on the ground, this positive pressure is replaced by a negative pressure, or suction, caused by the engine compressor. Instead of air being forced or rammed through the two aft air distribution systems, air is drawn or sucked *from* the distribution ducting and into the engine. This air is drawn from outside the airplane and flows through the aft electronic compartment, the a-c and d-c generators, and the IFF units. This creates a continuous flow of cooling air around these units for ground cooling. The negative pressure, or suction, in the engine intake also draws air from the upper electronic compartment through the one-way check valve. This valve opens only when the duct pressure is less than the electronic compartment pressure. Thus, air is sucked from the upper and intermediate compartments by both the jet pump — through the boundary-layer air distribution system, and the engine compressor—through the check valve.

Occasionally, the hot bleed air from the jet pump nozzles may cause the airplane structure to become too hot. When this happens, temperature sensing probes cause the jet pump shutoff valve to close and stop the flow of air through the forward electronic compartment. The intermediate and upper electronic compartments will still be ventilated by air drawn through the check valve, but the total air flow through these compartments is greatly reduced. Since the engine is running, the cockpit air conditioning system should be turned on by placing the cabin air switch at PRESS.

If the canopy is open on the ground, all of the cockpit air will go overboard and none will flow to the forward electronic compartment. If the canopy is closed, however, cockpit pressure will be slightly higher than atmospheric pressure and the cockpit pressure regulator opens. Thus, the system is designed for the canopy to be closed during this operation. Some cockpit air then flows through the discharge ducting to the lower plenum of the forward electronic compartment. Some of this air flows through the upper plenum and the rest is distributed through the distribution ducting to the intermediate and upper electronic compartments. You must understand that this is an emergency procedure since the air flow in the upper plenum is too low for sustained operation.

AIR DISTRIBUTION DUCTING.

The air distribution ducting for the electronic compartment's cooling system consists of uninsulated aluminum duct sections. Some "branches" of the various duct systems are as much as 4 inches in diameter. Other duct sections are smaller with the smallest branch having a diameter of only 1½ inches. The cooling air ducting, which was discussed in Chapter II, presents very few maintenance problems in comparison to the bleed air ducting. The pressure and temperature of cooling air in the ducting is much lower than bleed air pressure and temperature. Normally, the pressure will never be more than perhaps 3 or 4 psi, and the temperature will not exceed 74°C (165°F). The sudden expansion of metal parts because of rapid temperature changes, which is a serious problem with bleed air ducting, causes no difficulties in the electronic cooling system. Minor air leaks are not too important, so clamping of the duct sections together is not such a problem as it is for bleed air ducting. When inspecting the low pressure pneumatic system, check the ducting for damage and replace defective sections in the manner described in T.O. 1F-102A-2-3.

ELECTRONIC COOLING AIR FLOW CONTROL.

The control of cooling air flow to the electronic compartments is relatively simple. The flow of air to the aft electronic compartment, a-c and d-c generators, and the IFF units depends only on the speed of the airplane and the suction created by the engine compressor. There are no valves or regulators that control or shut off the air flow. On the other hand, the flow of air to the forward, intermediate, and upper electronic compartments is more complicated. The left boundary-layer air distribution ducting contains the pressure regulator and shutoff valve. There is a shutoff valve in the jet pump ducting (see figure 4-4). In addition to these control valves in the ducting, there are two relief valves that help to regulate the air pressure in the electronic compartments. The operation of all of these valves and regulators is entirely automatic. The pressure regulator and shutoff valve and the jet pump valve are controlled electrically; the relief valves are entirely mechanical in operation.

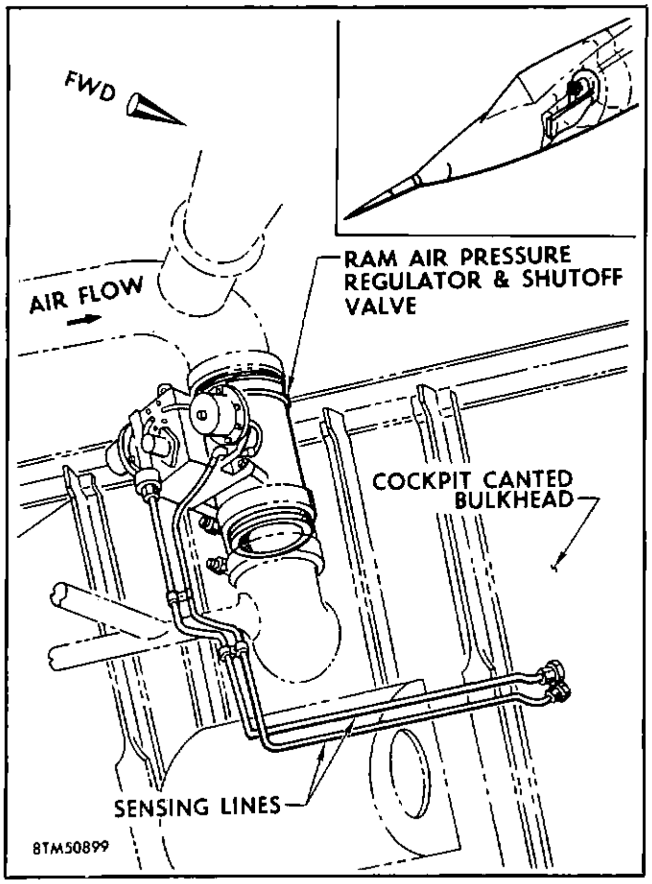


Figure 4-7. Ram Air Pressure Regulator and Shutoff Valve

RAM AIR PRESSURE REGULATOR AND SHUTOFF VALVE.

The ram air pressure regulator and shutoff valve is a very important, and the most complicated, component of the low-pressure pneumatic system. The unit is actuated electrically and has a complicated external control system which was completely described in Chapter III. In addition to the external controls, the unit has built-in features that regulate the pressure downstream and shut off the air flow whenever the downstream pressure or the upstream temperature exceeds certain limits. The regulator and shutoff valve controls the flow of ram air, through the left boundary-layer ram air distribution system, to the forward, intermediate, and upper electronic compartments. It must also be open for ram air to be available for cockpit ventilation. In this section the unit itself will first be described in detail, and then we will discuss its operation under various conditions.

As you can see in figure 4-7, the regulator and shutoff valve is installed in the left boundary-layer distribution ducting and is located in the intermediate electronic compartment just aft of the cockpit bulkhead. Note the two pressure-sensing lines that connect the unit to the downstream ducting. There is also a thermostat in the valve body itself just upstream of the shutter.

As we have previously learned, the valve is actuated electrically. This sounds simple, but it is somewhat more complicated than other electrically operated valves. The valve shutter is opened or closed by a reversible d-c motor that is very similar to the reversible motor that operates cockpit ram air valve. As you will remember from the discussion in Chapter III, the direction of rotation of the motor armature depends on which of the two motor field windings are energized. In the cockpit ram air valve, the motor armature is connected directly to the shaft of the valve shutter. In the pressure regulator and shutoff valve, the armature is connected to the shaft by a mechanical clutch. This clutch is engaged *only* when a solenoid is energized. This rather interesting arrangement is shown in figure 4-8.

How the Regulator and Shutoff Valve Operates.

Note the worm teeth cut into the extension of the armature shaft. This worm is permanently engaged to a worm gear. This worm gear, however, is *not* permanently connected to the shaft of the valve shutter. Note also the three steel balls and their cage (only two show in the illustration). The cage is permanently secured to the shutter shaft. Now note the three holes, or detents, in the side of the worm gear (only two show). When the cage is pressed against the worm gear by solenoid action, the balls will engage the detents and the cage and shutter shaft will be connected to the worm gear. When the solenoid is de-energized, the spring will force the cap away from the cage and the shutter shaft will no longer be connected to the worm gear.

Now look at the stop collar secured to the cage. Notice the spring secured to the collar; when the shutter shaft is no longer engaged, this spring will cause the shaft to rotate to the closed position and all ram air to the distribution ducting will be cut off. This is an important "fail-safe" feature, for whenever electrical current to the solenoid is cut off for any reason, the valve shutter will automatically close. When the shutter moves to the full *open* or full *closed* position, the stop screws on the collar will contact limit switches to cut off current to the motor field windings.

At this stage, you may ask two questions. The first one: "How is the electrical current to the solenoid and motor controlled?" and the second is probably, "How is the downstream pressure regulated?" The simple diagram, figure 4-9, will help you understand the discussion. Remember we are talking about the *built-in* control system. The external control system was discussed in Chapter III.

How Electrical Current to the Solenoid and Motor Is Controlled.

To refresh your memory, the current to the regulator and shutoff valve is cut off during armament firing, when the airplane is on the ground, when the engine oil pressure is low, or when there is an electrical supply failure. Remember also that current is restored to the unit, even when the airplane is on the ground, when the jet pump is operated. Now, assuming that the external circuit permits current to be available, how is the current controlled internally?

In figure 4-9 you can see that current enters the unit at connection A. From there it flows through the pressure switch to the relay switch and on to point 1. From this point current flows to the solenoid clutch and the reversible motor. At this point the solenoid is energized, the clutch is engaged, and current is *available* for the reversible motor. The current may or may not be connected to one or the other of the motor field windings. The pressure regulation portion of the pressure regulator and shutoff valve controls the motor current at this point. You can see that the current from connection A to point 1 passes through two switches.

The first switch is the pressure switch, and will normally be closed. This pressure switch is a safety feature and will actuate to break the circuit when the downstream pressure reaches about 2 psi. Since the pressure is normally regulated to about 1 psi, you can see that the pressure switch will actuate only when there is some failure in the pressure regulator. The second switch is normally closed and is controlled by a normally de-energized relay. This relay is energized and the switch is opened when the temperature upstream of the valve shutter reachs 71°C (160°F) to 74°C (165°F). This causes the thermostatic switch to close and connect the relay to power. To sum up, the current to the solenoid clutch and the motor is cut off when either the downstream pressure or the upstream temperature becomes too high.

How Downstream Pressure Is Regulated.

As we have just learned. the downstream pressure is regulated to a value of about 1 psi higher than atmospheric pressure. Actually, it is regulated to about 1 psi higher than the pressure in the intermediate electronic compartment. This pressure will normally be very slightly higher than atmospheric pressure, but will not exceed a differential of about 0.3 psi. Notice in the diagram (figure 4-9) that power from point 1 is not connected directly to the motor field windings, but is connected to two microswitches, one for each field winding. Notice also that power is connected to contact C on each of the switches. The *open* field winding of the motor is connected to contact B on one microswitch, and the *close* field winding is connected to contact B on the second microswitch. You will note also that both field windings are connected to ground. This is the normal condition when the downstream pressure is within the required limits.

You can see the arm attached to the piston and diaphragm in a control chamber which balances downstream pressure against a spring and atmospheric, or

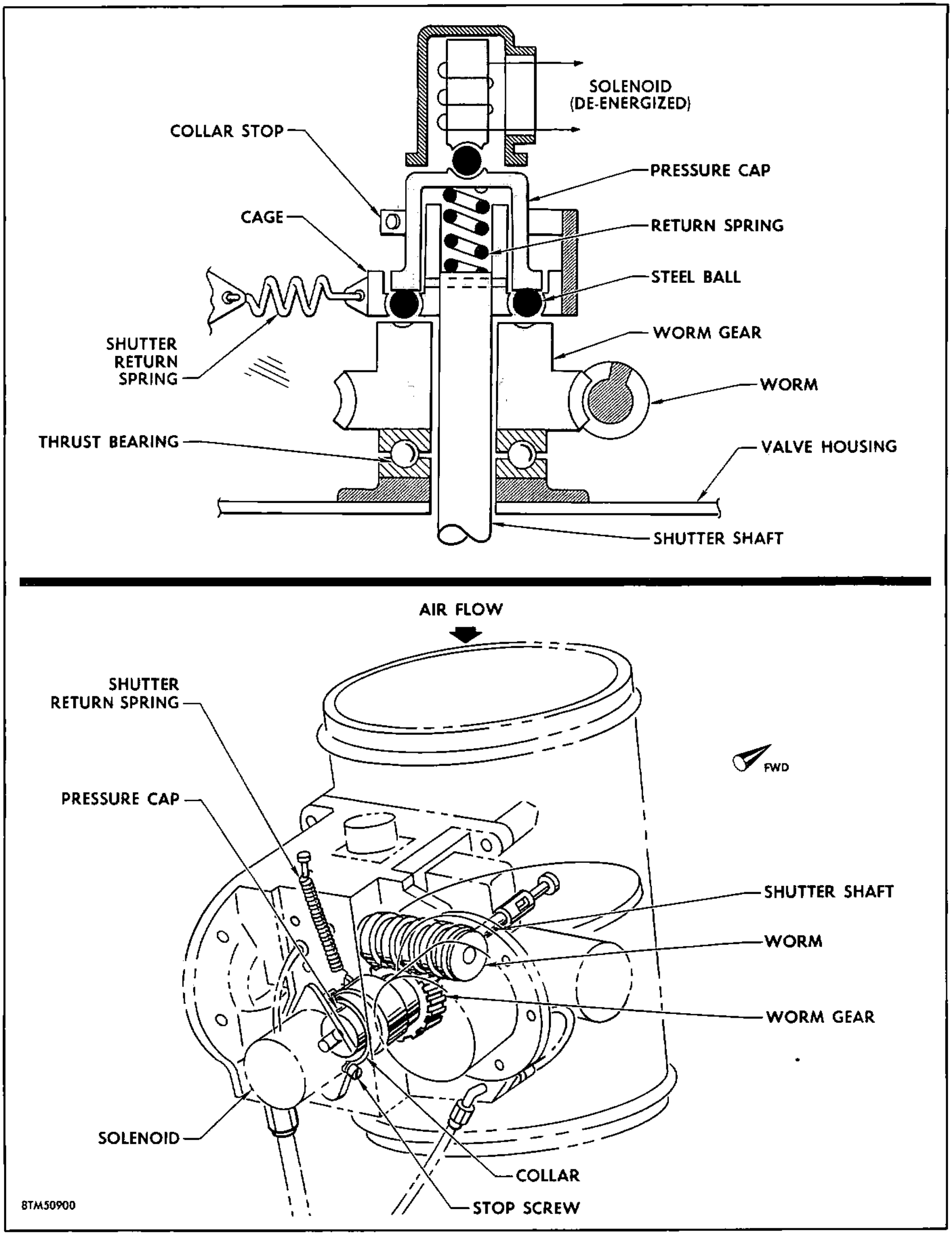


Figure 4-8. Solenoid Controlled Clutch

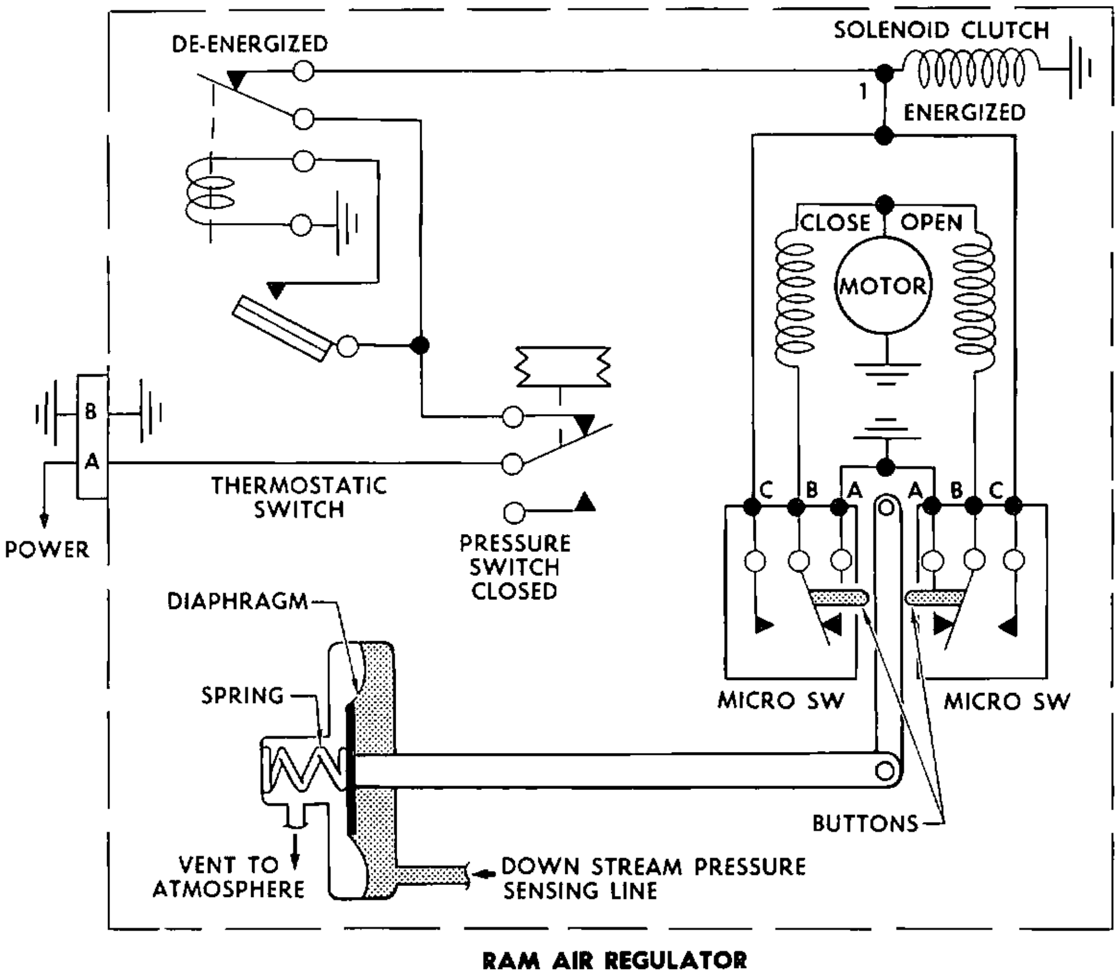


Figure 4-9. Pressure Regulator and Shutoff Valve Electrical Schematic

electronic compartment, pressure. The pressure sensing line connects one side of the diaphragm to the downstream ducting. When air is not flowing in the ducting, downstream pressure will equal atmospheric pressure and the diaphragm spring will hold the diaphragm, the piston, and the arm at the *open* microswitch. The arm will depress the button and connect the *open* field winding to power through contacts B and C. The motor will then turn to move the valve shutter toward the open position, and air will flow through the ducting if the airplane is moving.

As the downstream pressure builds up, the diaphragm will move to compress the diaphragm spring. When the downstream pressure becomes about 1 psi above atmospheric pressure, the diaphragm will move its piston and the arm away from the button on the *open* microswitch. The spring in the microswitch will then break the circuit to the *open* winding, and the motor will stop. The solenoid clutch will hold the shutter in the position it is in when the motor stops. As long as the pressure remains constant, the motor and shutter will not move. If the pressure increases too much, the downstream pressure will compress the diaphragm until the *close* circuit is energized.

The motor will then move the shutter toward the closed position. This OPEN, OFF, CLOSE action is continuous and the position of the shutter changes continuously in a "modulating" action to maintain a constant downstream pressure. The simplified cross section (figure 4-10) shows the actual physical arrangement of the pressure regulation portion of the regulator and shutoff valve. Notice that the diaphragm travel, and therefore the piston and arm travel, is very limited. Only a slight change in pressure is sufficient to energize either of the motor field windings.

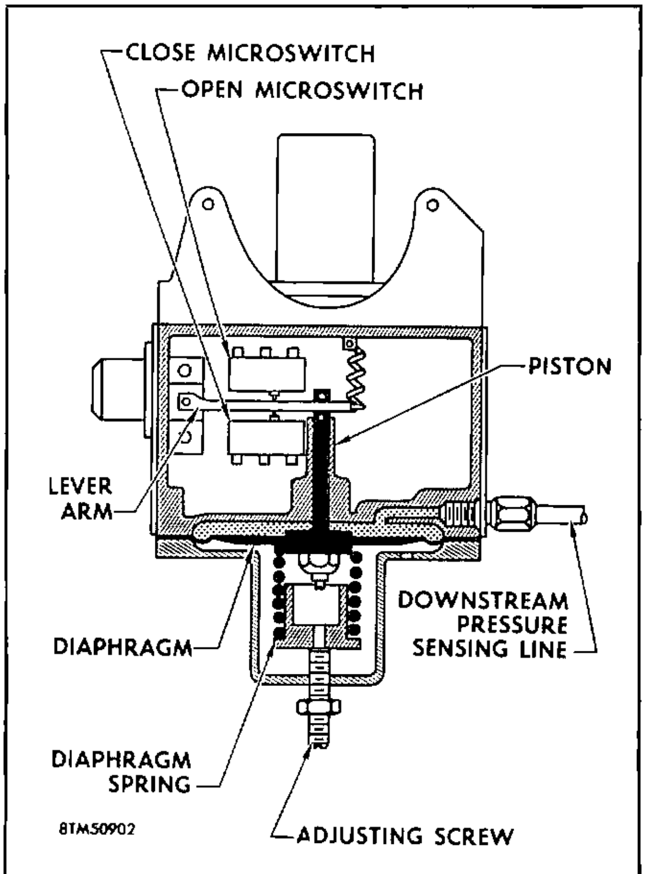


Figure 4-10. Pressure Regulation Section of Pressure Regulator and Shutoff Valve

Thermostatic and Pressure Switches.

Now that you understand how the downstream pressure is regulated we will return to the two "built-in" electrical control features, the thermostatic switch and the pressure switch. Actually these controls are safety features that cause the valve to close when certain unwanted conditions exist. As you can see, the thermostatic switch is mounted in the valve body just upstream of the valve shutter. The thermostatic switch makes use of the different expansion properties of different metals, one piece brass and the other iron. When they are cold, the pieces will be straight; but if they are heated, the brass will expand more than the iron and the iron will tend to bend away from the brass side. This principle can be used in a switch by positioning a contact so that a circuit will be completed when the bimetallic elements bend to a certain position. The elements will reach this position at a certain definite temperature, therefore the arrangement is called a thermostatic switch.

In the pressure regulator and shutoff valve, the thermostatic switch will close when the air temperature just upstream of the valve shutter reaches 71°C (160°F) to 74°C (165°F). This will energize the relay and break the circuit to the clutch solenoid and the electric motor (see figure 4-9). The thermostatic switch will not reopen, de-energizing the relay, when the temperature drops about 3°C (5°F) lower than the closing temperature. For example, if the switch closes at 72°C (162°F), it will reopen at 69°C (157°F).

The pressure switch is the second of the safety control features. The upstream pressure varies from 1 to 9 psi above atmospheric pressure depending upon the speed of the airplane. The pressure switch senses downstream pressure from the smaller of the two sensing lines and opens the electrical circuit to the clutch solenoid and the electric motor when the downstream pressure reaches about 2 psi. (See figure 4-9.)

If the pressure regulating diaphragm should rupture, the diaphragm spring would cause the *open* field circuit of the motor to energize and the shutter would move to the wide open position. Little harm would result at lower airplane speeds, but at higher speeds the excessive pressure could rupture the comparatively light aluminum ducting.

Now notice the indicator on the face of the pressure switch. When the pressure switch has *not* been activated, the indicator will point to NORMAL. If the indicator points to OVER PRESSURE, the switch has been activated and the electrical circuit has been broken.

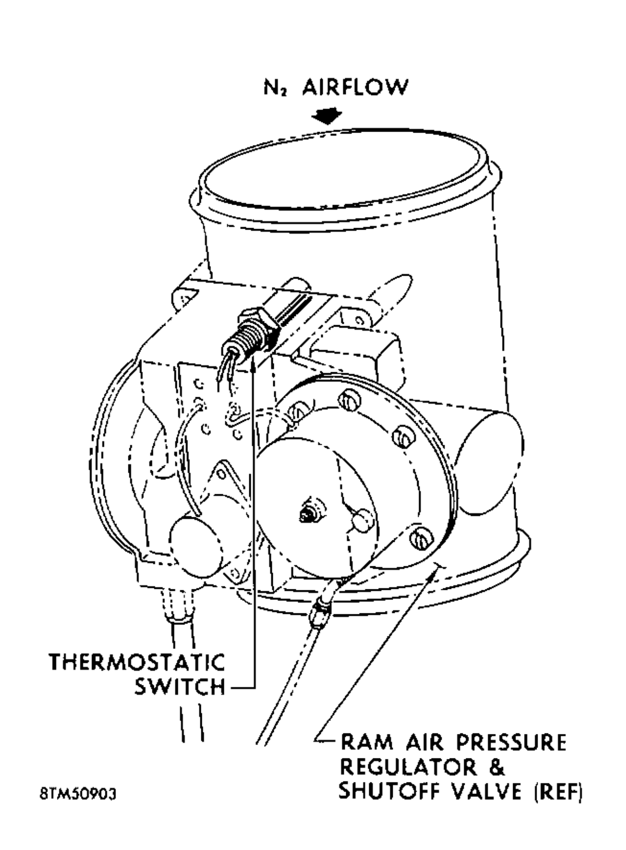


Figure 4-11. Thermostatic Switch

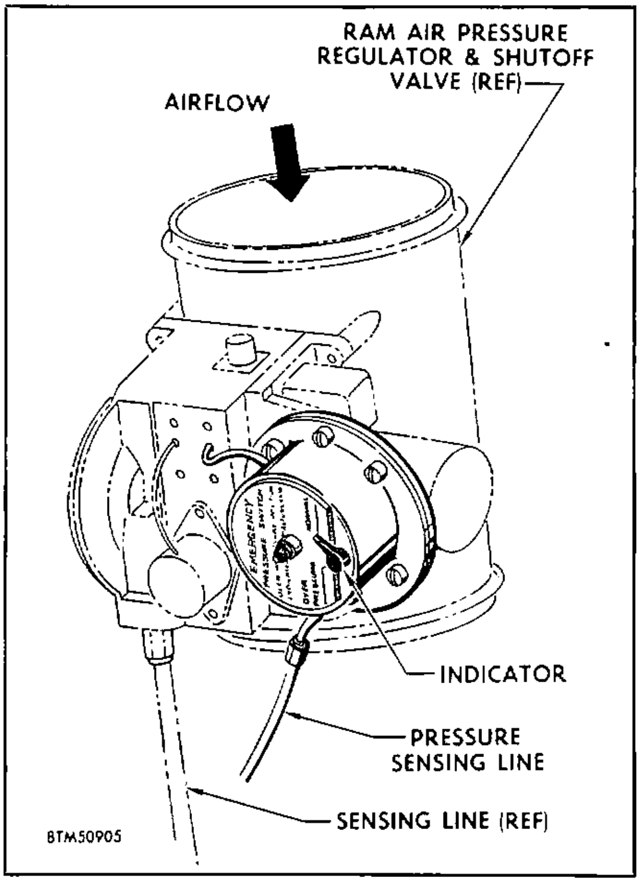


Figure 4-12. Pressure Switch and Indicator

Once the switch has been activated, the pointer will stay at OVER PRESSURE, the switch will be inoperative, and the valve will stay closed until the pointer is manually turned to NORMAL. Since the indicator pointing at OVER PRESSURE usually means that the pressure regulating diaphragm is ruptured, you should always replace the entire unit rather than move the pointer. Your F-102A technical order maintenance manual T.O. 1F-102A-2-6, has the instructions for replacing this component.

Valve Shutter Operation.

So far we have discussed the various controls and mechanisms that control or activate the valve shutter, but we have not discussed the shutter itself. You can see, in figure 4-13, that the shutter is a butterfly type and that there is another smaller butterfly shutter built in the main butterfly. The small one is called the "pilot" butterfly. Note that the shutter shaft passes through the main shutter but is *not* directly connected to it, but rather by linkage to the pilot butterfly. The use of a pilot butterfly allows the main butterfly shutter to be operated with much less force than would otherwise be required. This idea will be easier to understand if you look at the illustration (figure 4-13).

It is obvious that the air upstream of the shutter will exert an equal pressure on all parts of the shutter. If the pilot butterfly is closed, there is an equal force against both the top and the bottom halves of the shutter. Since the forces are equal, there will be no tendency for the air pressure to open the shutter, and the entire force necessary to turn the shutter must be supplied by the motor. If a portion of the shutter surface were removed the total force on one half of the shutter would be less than the force on the other half. This unbalance of force would cause the shutter to open. The amount of opening would depend on the difference of the total forces. In this case, the force on one half of the shutter is reduced by opening the pilot butterfly. The larger the pilot opening the further the main shutter will open.

In figure 4-15, note that the shutter is not at right angles to the valve body and the entire movement from full close to full open is only 75° instead of 90°. The first 50° of this movement is controlled only by the opening of the pilot butterfly and the resulting unbalanced force is exerted by the air pressure on the shutter. After the full 50° movement, the linkage, from the shutter shaft to the pilot butterfly, can no longer turn the pilot butterfly and the linkage will bear directly against the main shutter. The last 25° of shutter movement is controlled by the force of the motor applied to the shutter through the linkage.

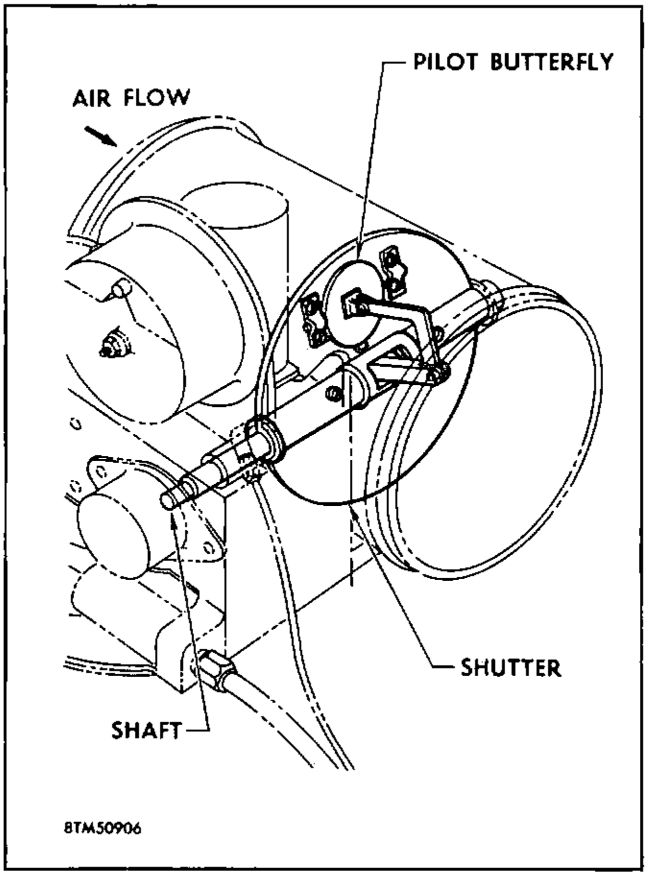


Figure 4-13. Pressure Regulator and Shutoff Valve Shutter

Operation of the Ram Air Pressure Regulator and Shutoff Valve During a Typical Mission.

After having thoroughly covered the pressure regulator and shutoff valve and its various interlocking controls, perhaps a description of its operation during a typical mission would help. Suppose that the airplane is on the ground with its engine off. In that case there would be no power to the unit because there would be no oil pressure. When the pilot starts the engine, the oil pressure builds up, but there will still be no power to the unit because the landing gear safety switch breaks the circuit. The pilot places the throttle at IDLE for a brief warm-up. This bypasses the ground safety switch and power is supplied to the regulator and shutoff valve. The valve opens and at the same time the jet pump shutoff valve opens, permitting a reverse flow of air through the left boundary-layer ducting. Since the pressure will be low, the butterfly shutter of the regulator and shutoff valve will be wide open. Now as the pilot prepares to take off, he advances the throttle past the IDLE position. The jet pump shutoff valve and the pressure regulator and shutoff valve will close since neither can be open on the ground when the throttle is not at IDLE.

As the airplane leaves the ground and its weight is removed from the landing gear, the safety switch opens, and current flows to the pressure regulator and shutoff valve causing the shutter to move to the wide open position. As the airplane speed increases, the duct pressure increases above 1 psi and the pressure regulator will cause the shutter to close enough so that the pressure downstream stays at about 1 psi.

Now if the armament is to be fired, the power to the unit is cut off when the armament bay doors open. The butterfly shutter will close. After armament is fired the doors will close and power will be restored after a three or four second delay. Of course, the airplane will be flying very fast, so ram air pressure will be high. The pressure regulator will react very fast and prevent the shutter from opening very wide. As the pilot returns for a landing, he reduces his speed; the ram air pressure drops and the shutter opens wider.

At about the moment the airplane wheels touch the ground, the pilot places the throttle at IDLE and the jet pump shutoff valve opens. If the weight of the airplane should be on the landing gear *before* the throttle is placed at IDLE, the pressure regulator and shutoff valve would close and then reopen as soon as the throttle is placed at IDLE. It will close immediately when the engine is turned off.

We have just described a normal mission. Now, suppose it is a very hot day. When the airplane is flying extremely fast at low altitudes, the ram air temperature might reach 71°C (160°F). At about that temperature, the thermostatic switch would close and remove power to the solenoid and motor, and the shutter would close.

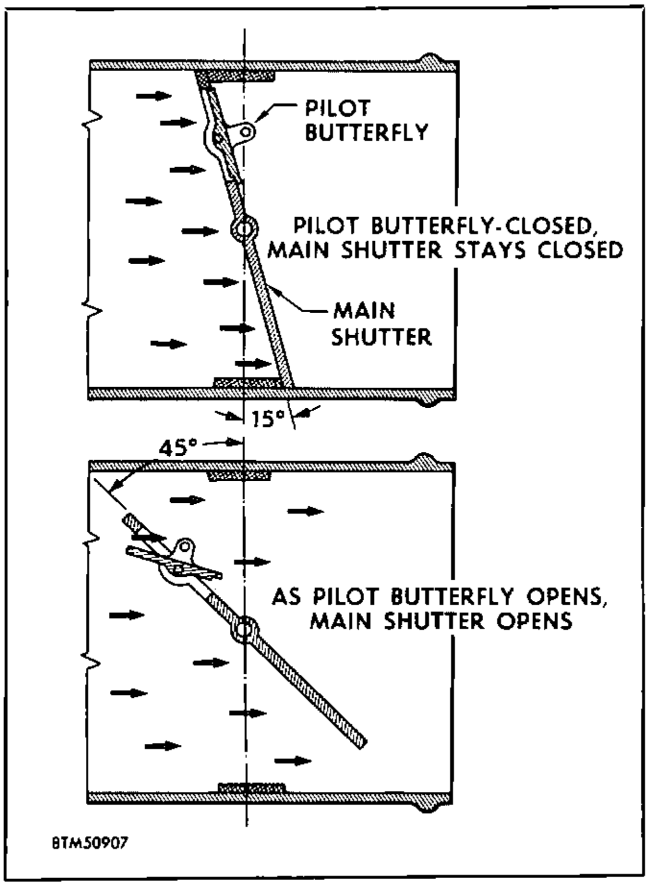


Figure 4-14. Cross Section of Shutter with the Pilot Butterfly

The switch would not reopen to restore power until the ram air temperature drops 3°C (5°F) from the closing temperature. Since air temperature normally drops as the airplane climbs and since the F-102A performs its mission at high altitudes, the valve shutter will be closed for a very short time only because of high ram air temperatures. The rupture of a regulator diaphragm is much more serious since the pressure switch would open and cut the power to the solenoid clutch as soon as the ram air pressure reaches about 2 psi. Once closed, because of a ruptured diaphragm, the valve shutter will stay closed until the unit can be replaced on the ground. All cooling air for the electronic compartments must then come from the cockpit discharge duct.

MAINTAINING THE RAM AIR PRESSURE REGULATOR AND SHUTOFF VALVE.

Maintaining the ram air pressure regulator and shutoff valve in the F-102A airplane is fairly simple. During your periodic tests and inspections, always check the indicator on the pressure switch. If it points to OVER PRESSURE you must replace the whole unit. Instructions for replacing the switch are given in T.O. 1F-102A-2-6. Be sure that the pressure sensing lines are tightened properly and do not leak. Refer back to Chapter III for a discussion on maintaining pressure

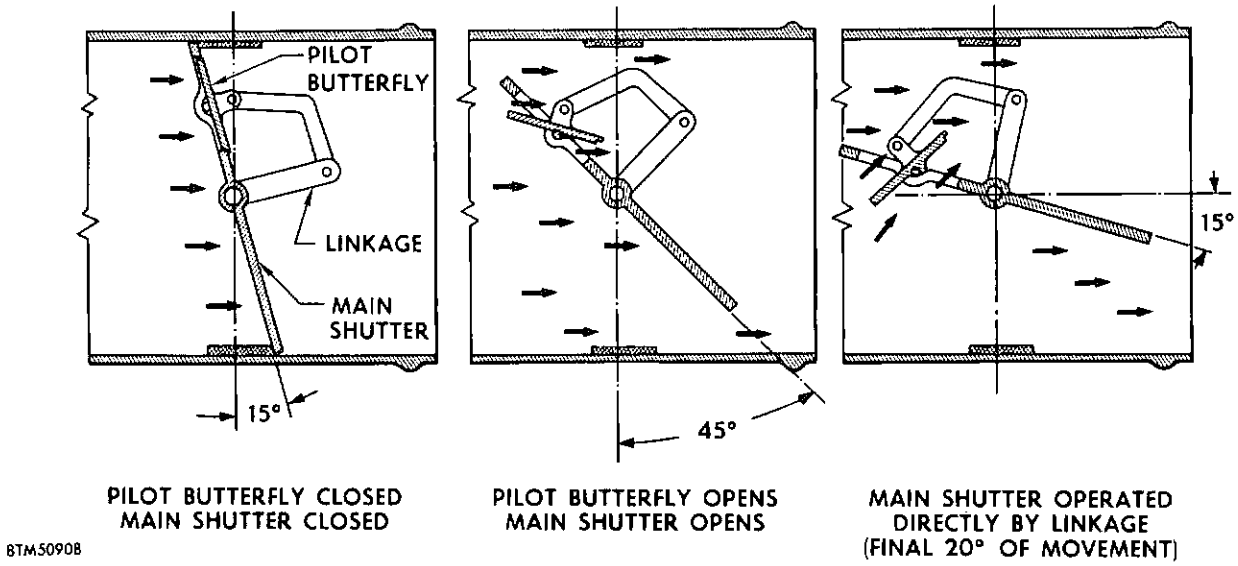


Figure 4-15. Shutter Operation

sensing lines. Although certain parts of the unit, such as the diaphragm, thermostatic switch, and pressure switch, can be replaced or adjusted, do not do this on the flight line. If the unit does not operate properly, replace it and turn the old one in for repairs.

THE JET PUMP.

The jet pump consists of six nozzles which are connected through a shutoff valve to the bleed air supply line. As shown in figure 4-6, the nozzles are mounted near the left boundary-layer ram air intake and point forward and out of the intake. When the jet pump valve is *open,* hot bleed air leaves these nozzles in a high-velocity jet stream. This jet stream creates a suction in the distribution ducting and results in a flow of "cooling" air through the forward, intermediate, and upper electronic compartments. In this section, we will discuss the jet pump control system, the jet pump ducting, and the shutoff valve.

Jet Pump Ducting.

The jet pump ducting consists of several short sections of stainless steel ducting. Since hot air will normally flow through this ducting only for short periods of time, this ducting is not insulated. The ducting is two inches in diameter and the sections are clamped together in the manner as described in Chapter II for all bleed air ducting.

The Jet Pump Control System.

Operation of the jet pump is controlled by an electrical system that monitors the flow of current to the solenoid-controlled, pneumatically-actuated shutoff valve. The electrical control system is shown in figure 4-16. This control system should look familiar because certain components also appeared on the wiring diagrams which you studied in Chapter III under the cockpit air flow control system. You should remember the main landing gear safety switch, the cabin air relay, and the throttle position switch. As you learned, the shutoff valve is solenoid-controlled. The flow of current to the solenoid depends on the position of the cabin air relay switches (which depend on the ground safety switch) and the throttle position switch. The overheat detection system breaks the circuit to the solenoid whenever the airplane structure near the jet pump nozzles becomes too hot.

All switches and relays are shown in the position they assume when the airplane is on the ground, the throttle is at IDLE, and the shutoff valve is *open.* The structure overheat relay is shown de-energized; that is, the structure temperature is below the maximum allowable temperature. Note the parts of the diagram within the dotted box. Those parts make up the warning system that operates if the control system becomes defective.

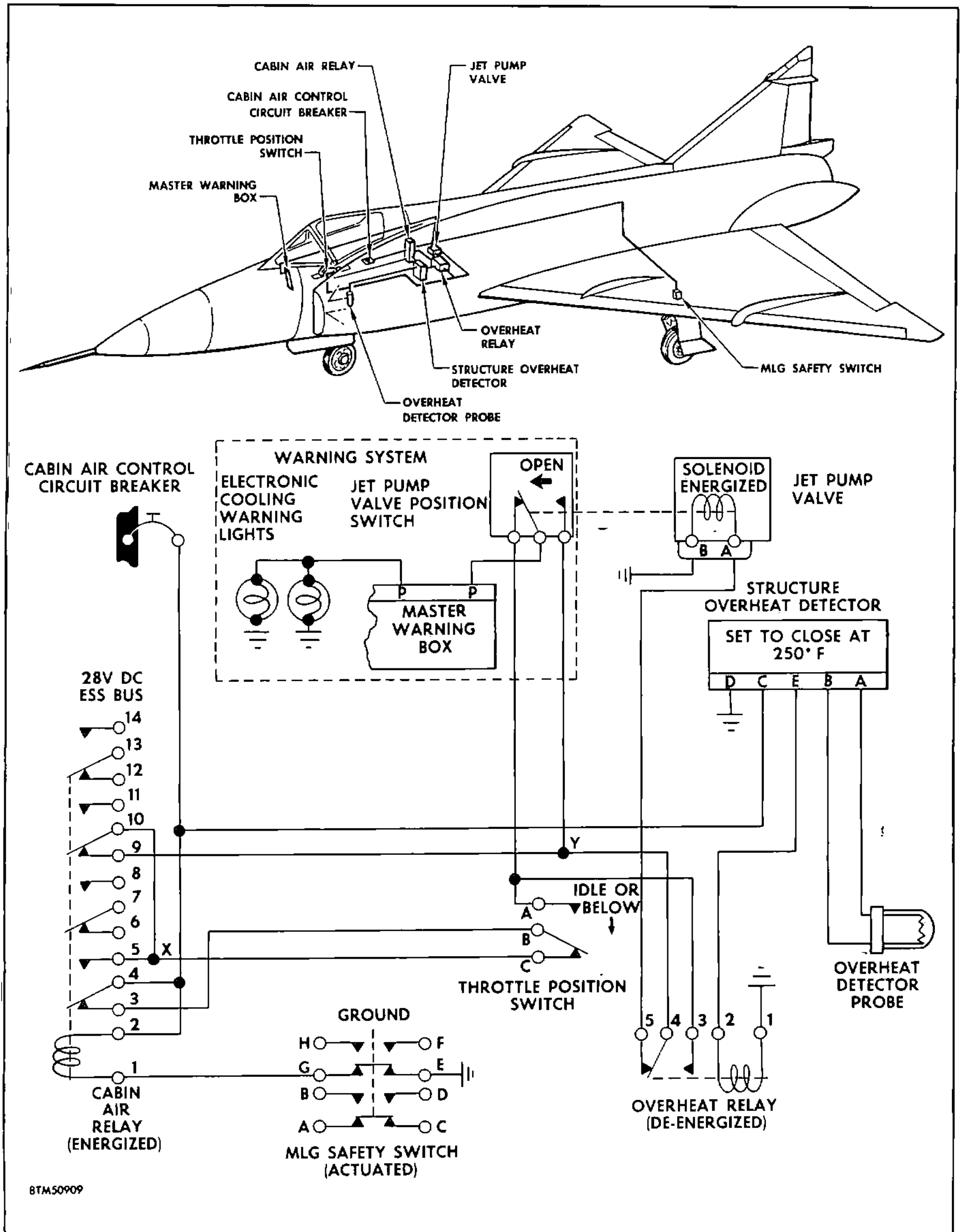


Figure 4-16. Jet Pump Electrical Control System Diagram

The shutoff valve will open if its solenoid is energized, and will close when it is de-energized. Follow the diagram step by step while we "guide the current" to the solenoid. Note that power comes from the 28-volt d-c essential bus to contact 4 on the cabin air relay. From there current flows through the switch to contact 3 and then to connection B on the throttle position switch. If the throttle is at IDLE, current will flow through the switch to connection C, and then through point X to contact 10 on the cabin air relay. From contact 10, the current passes through the switch to contact 9 and on to contact 4 on the overheat relay. It continues through the switch to contact 5 and on to connection A on the shutoff valve solenoid. The current has passed through two switches on the cabin air relay, the switch in the overheat relay, and the throttle position switch.

All of these switches must be in the position shown for the current to reach the solenoid. The ground safety switch is actuated to provide a path to ground for the cabin air relay. The throttle must be at IDLE and the overheat relay must not be energized. The structure overheat system is completely described in another supplement of this training series which covers the J57 Power Plant Installation.

To summarize the control system, the solenoid should be energized when the airplane is on the ground, the throttle is at IDLE and the structure near the nozzles is not too hot. The cabin air relay solenoid de-energizes as the airplane leaves the ground, when the throttle is advanced, or when the airframe structure overheats. The ram air pressure regulator *must* be open when the jet pump is operating, and will be open on the ground if the throttle is at IDLE and the engine is running.

Since the pilot cannot observe the jet pump control system directly, a warning system is provided to warn him of malfunctions in the system. A warning light labelled ELECTRONIC COOLING is installed in the cockpit and will light when any one of the following three conditions of malfunctions exists:

1. Valve *closed* when airplane is on ground and throttle is at IDLE.
2. Valve *open* when airplane is on ground and throttle is advanced beyond IDLE.
3. Valve open when overheat relay is energized.

These are, of course, only three of the defects that might possibly occur in the jet pump control system. The warning system to be found in later production airplanes will indicate some additional defects. In the diagram, notice the master warning box and the warning lights. Whenever current reaches the box, the warning lights light. Current to the master warning box is controlled by position switches actuated by the shutter shaft of the shutoff valve. The switch will always be in the *open* or *closed* position. If it is in a particular position at the wrong time, current will be fed into the warning box to light the lights.

To understand more fully how this warning system operates, let's examine what happens when we have the first of the defects listed above. When the airplane is on the ground and the throttle is at IDLE, the shutoff valve should be open. If it is closed, the position switch on the diagram will be at CLOSED and current will pass to the warning box through the switch across contacts 9 and 10 on the cabin air relay and point Y. The warning lights will light. In this event, the electronic equipment should be turned off immediately to prevent overheating, and the defect must be remedied.

If we have the second defect—that is, the valve is open, the airplane is on the ground, and the throttle is beyond the IDLE position—the position switch will be at the OPEN position. Power will then reach the warning lights through contacts 4 and 3 on the cabin air relay and connection B and A on the throttle position switch.

If we have the third defect—that is, the valve is open when the overheat relay is actuated, we have a longer route to trace the current path. Current will reach the warning lights through the position switch at OPEN and contacts 4 and 3 on the cabin air relay, connections B and C on the throttle position switch, point X, contacts 10 and 9 on the cabin air relay, and contacts 4 and 3 on the overheat relay.

The Jet Pump Shutoff Valve.

The jet pump shutoff valve is a pneumatically-actuated, solenoid-controlled valve of rugged construction. You can understand why it must be rugged if you consider that it controls the flow of bleed air with a temperature as high as 385°C (725°F) and a pressure as high as 220 psi. In figure 4-17, note that the shutoff valve consists of a valve body, a pneumatic actuator, and a control solenoid. Note also the position indicator switches on the shutter shaft extension and the arrow on the valve body. The valve must *always* be installed with this arrow pointing downstream. The shutoff valve will not open if the valve is installed with the arrow pointing upstream.

In the simplified schematic of the assembly shown in figure 4-18, note that the shutter shaft is connected to the actuator shaft through two gears. The actuator shaft rotates to open or close the shutter. Compressed air furnishes the pressure or force required to open the valve shutter. This air pressure is supplied by the hot bleed air through a port just upstream of the valve shutter. The air is then conducted through the actuator housing and is vented into a "balancing chamber" in the control solenoid. From the balancing chamber, air pressure is fed into either the CLOSE or the OPEN actuator pressure chamber. The position of the balance cylinder (which depends on whether the solenoid is energized or not) determines which chamber is pressurized. Note that the two pressure chambers are separated by a piston assembly.

When the upper, or OPEN, chamber is pressurized, the piston assembly is forced down and the shutter opens. When the CLOSE chamber is pressurized, the piston assembly is forced up and the shutter closes. The piston spring will hold the assembly in the *closed* position if neither chamber is pressurized. Now, you are probably wondering just how a movement of the piston assembly causes the shutter to move. Note in figure 4-19 that the piston is hollow inside and that the inner surfaces are rifled like a rifle bore. Also note the "lands" or raised portions of the rifling.

You can also see that two large rollers are on the end of the actuator shaft. The shaft and rollers fit inside the piston. When the piston is forced up or down, the lands engage the rollers and cause the actuator shaft to rotate. This rotation is transmitted to the shutter shaft by the two gears, and the shutter is opened or closed. Note the sealing ring around the edge of the shutter. This ring and shutter are very similar to the ring and shutter described in Chapter II when we discussed the flow control valve. Refer back to figure 4-18. Note the piston guide at the bottom of the piston and the stop screw at the top. The guide not only guides the piston movement but acts as a stop to prevent further pressure to the actuator shaft after the shutter is fully open. The adjustable stop screw at the top prevents further pressure after the shutter is closed.

As mentioned, the position of the shutter depends on which actuator chamber is pressurized and that depends on whether the solenoid is energized or not. Note in figure 4-20 that the solenoid works on the principle of an electromagnet. In the first part of the illustration note that when the solenoid is de-energized, the return spring will hold the balance cylinder in the *close* position, thus pressurizing the CLOSE pressure chamber. You can see that the OPEN chamber is vented through the solenoid assembly. When the solenoid is energized (shown in the lower half of the illustration) the balance cylinder is forced down and the OPEN chamber is pressurized. The CLOSE chamber is vented through the hollow balance cylinder and the solenoid assembly.

The actuator and solenoid assembly has two "fail-safe" features. First, the solenoid return spring will hold the balance cylinder in the CLOSE position to pressurize the CLOSE pressure chamber whenever current to the solenoid is cut off. Second, the piston spring will cause the piston and the shutter to move to the closed position whenever all pressure is removed from the actuator.

The position indicator mounted on the jet pump valve is actuated by the shutter shaft. Note in figure 4-21 that a pointer attaches to the extension of the shutter shaft. You can also see the microswitch on the bracket at the end of the shaft. The pointer actuates the microswitch when the shutter is in the CLOSED position. The microswitch then signals the master warning box as shown in figure 4-16. Notice also that the bracket is

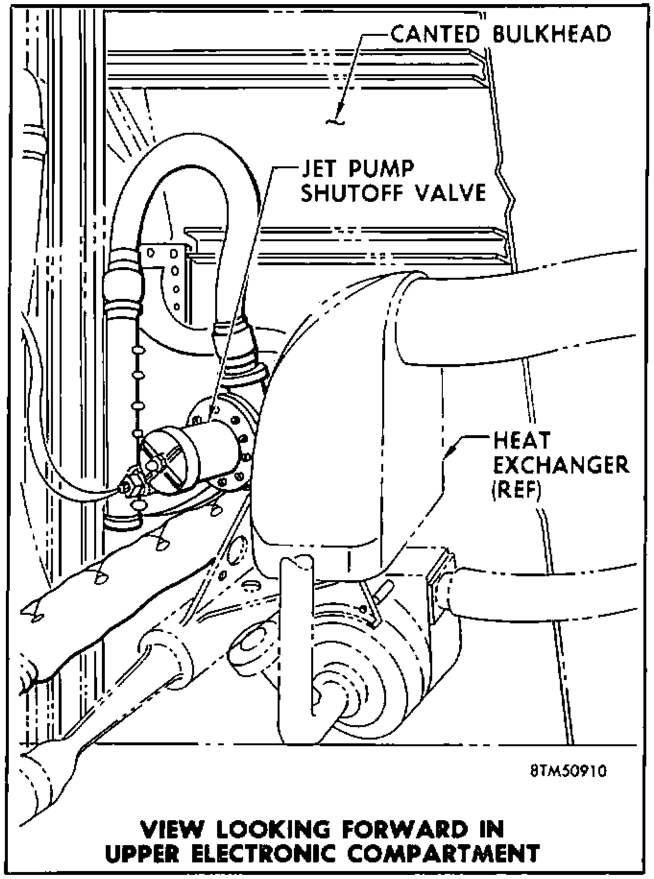


Figure 4-17. Jet Pump Shutoff Valve

marked CLOSED and OPEN to enable you to visually check the position of the shutter.

There is very little that you can do with the shutoff valve assembly. You can check its position under the various operating conditions already described. The warning system can also give you indications of trouble. If the valve does not operate properly, replace it as described in T.O. 1F-102A-2-6. You should also check the control system in the same manner as described for electrical control systems in Chapter III. Access to the jet pump valve can be obtained through either the intermediate or the upper electronic compartment.

GROUND AIR CONDITIONER FITTING.

As mentioned earlier in this chapter, a ground air-conditioner is connected to a fitting of the forward bulkhead of the nose wheel well whenever electronic equipment is being operated on the ground. The fitting consists of a square plate hinged at the top and normally fastened to the bulkhead at the bottom. When attaching the ground air-conditioner, simply unfasten the plate at the bottom, then swing it up and insert the ground tube through the round hole in the bulkhead and into the lower plenum of the forward electronic compartment. Be sure this plate is secured before the airplane takes off. As you will note in figure 4-22, a small flap on the plate acts as a relief valve.

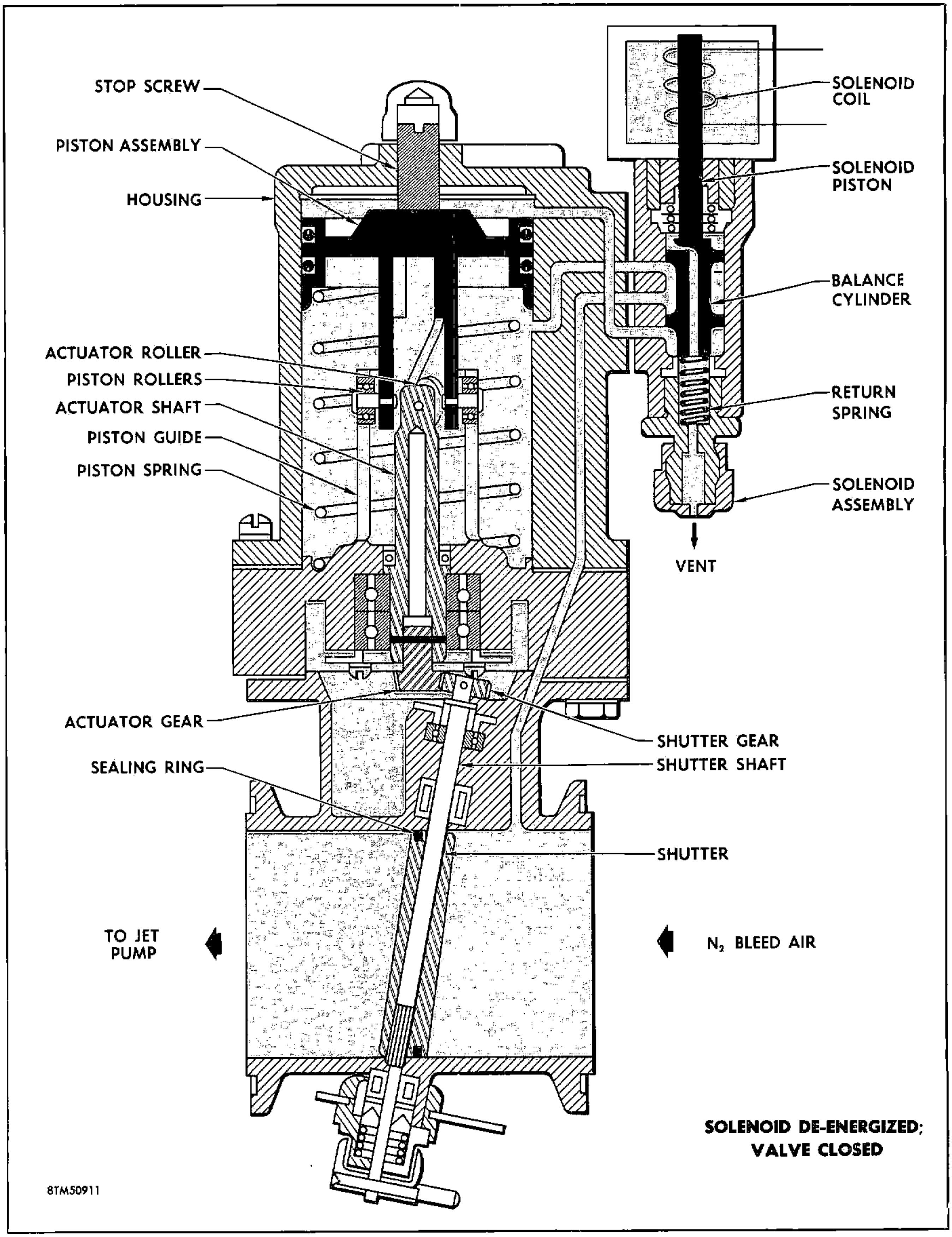


Figure 4-18. Jet Pump Valve and Actuator Schematic

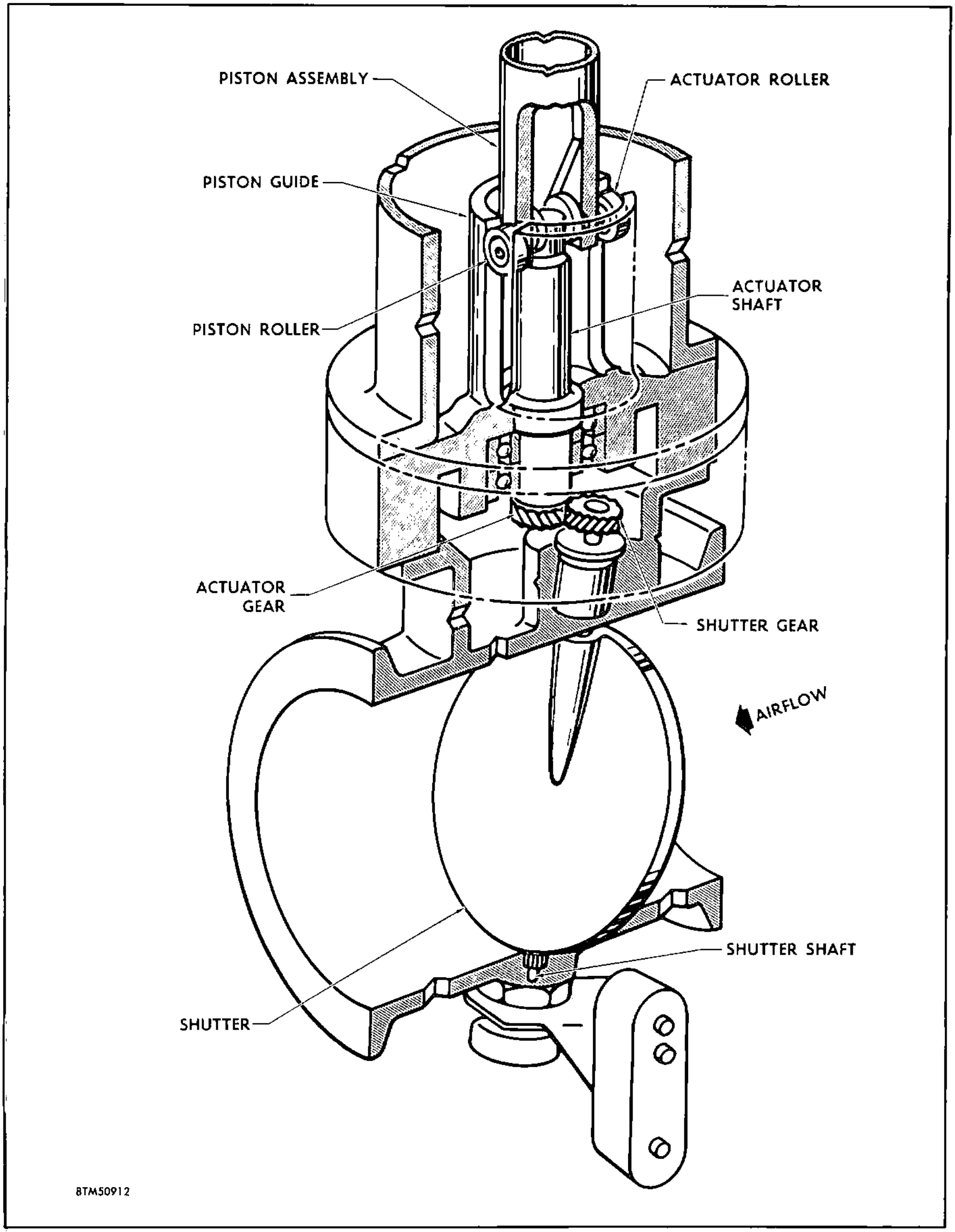


Figure 4-19. Jet Pump Shutter Drive Mechanism

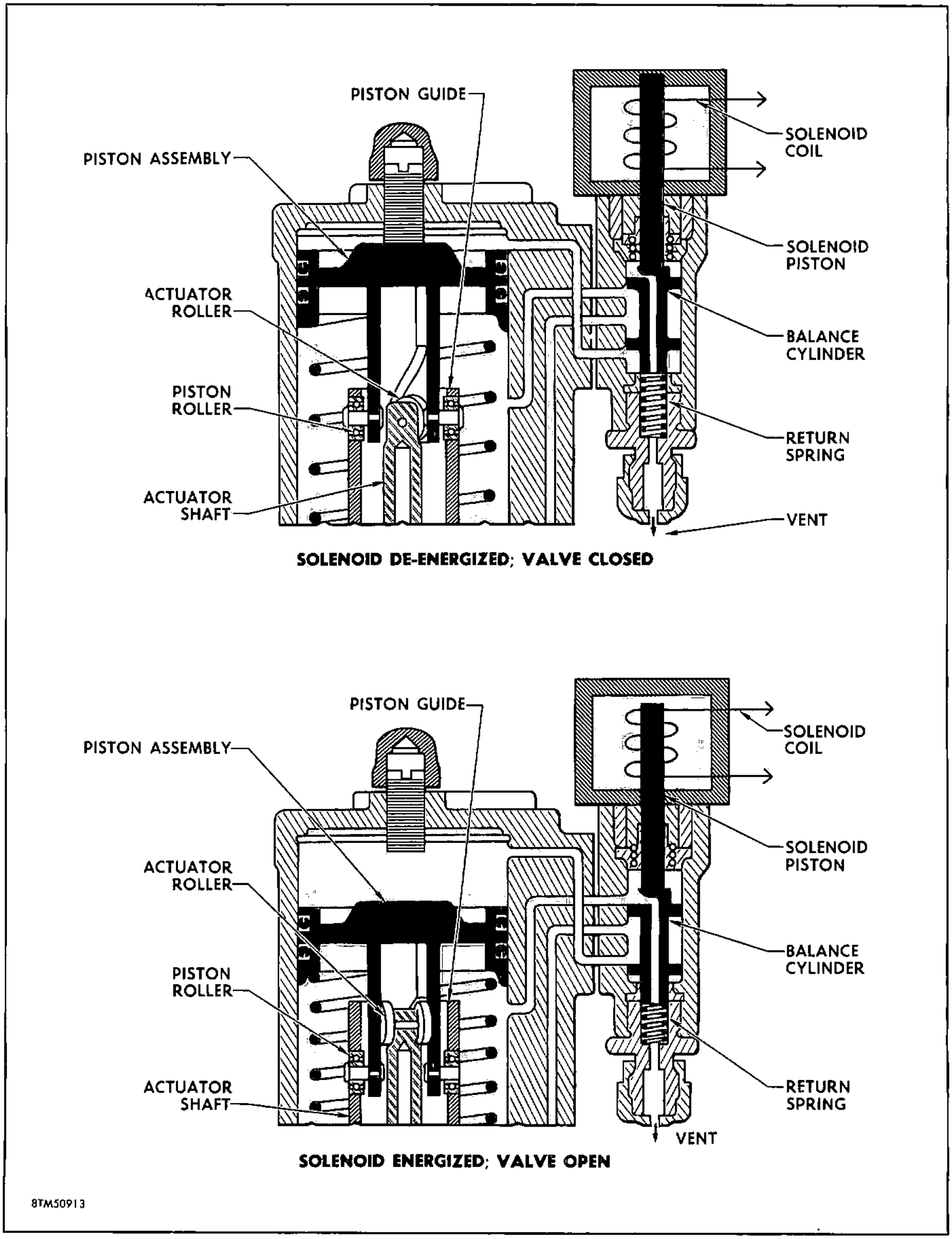


Figure 4-20. Solenoid Control of the Actuator

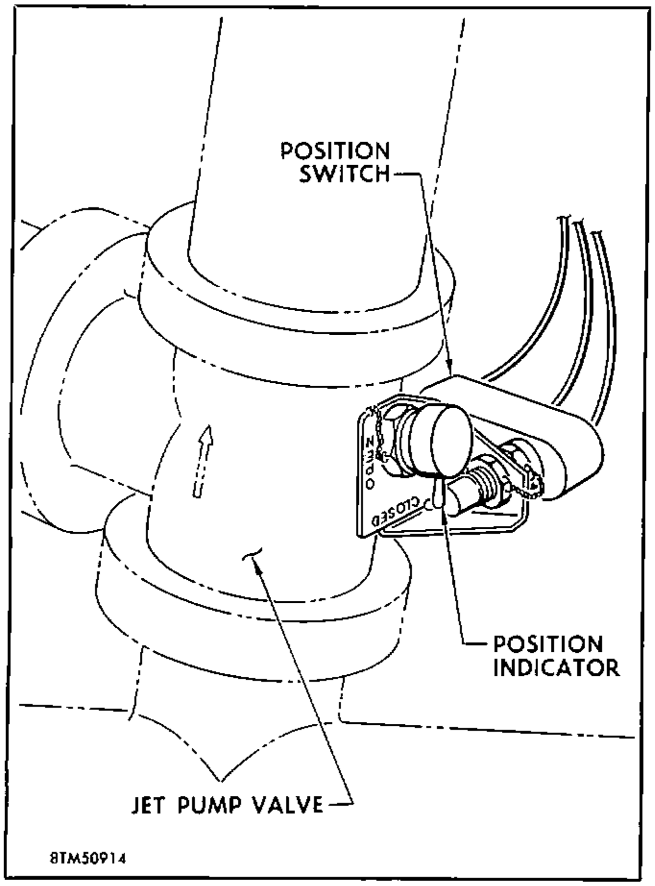


Figure 4-21. Jet Pump Valve Position Switch and Indicator

FORWARD ELECTRONIC COMPARTMENT RELIEF VALVE.

We have learned that the forward electronic compartment contains two chambers or plenums and that the lower plenum acts as a manifold for distributing air to the upper plenum which contains the electronic equipment. During normal operations, air enters this plenum from both the cockpit discharge duct and the boundary-layer distributing duct. The air pressure in the boundary-layer duct is normally regulated at about 1 psi over atmospheric pressure, and a pressure switch will shut off the air flow through the boundary-layer duct when the pressure reaches about 2 psi. Two psi is a higher pressure than is desired in the lower plenum because this plenum is constructed of rather light material. For this reason, a spring-loaded flap, or relief valve, is installed in an opening on the plate over the ground air conditioner fitting. This flap, shown in figure 4-22, opens when the lower plenum air pressure reaches about 1.5 psi. Air flows into the nose wheel well and helps cool the intermediate and upper electronic compartment.

INTERMEDIATE AND UPPER ELECTRONIC COMPARTMENTS RELIEF VALVE.

Earlier in this chapter, you learned that a relief valve is installed on the floor of the upper electronic compartment. All air that enters the intermediate and upper electronic compartments leaves through this valve and flows into the armament bay where it discharges overboard through the door hinges. Because of the large flow of air, a large opening is necessary. An open hole in the electronic floor will not suffice for the purpose. An open hole is not possible for two reasons: first, exhaust fumes from armament firing must not "backflow" into the electronic compartment; and second, the electronic compartment must be slightly pressurized to help the electronic equipment function properly at high altitudes. A relief valve is therefore installed in the opening. This valve maintains the electronic compartment pressure at a pressure altitude of 50,000 feet when the airplane is at 55,000 feet.

The relief valve is a simple, automatic-type that should give little maintenance difficulties. Notice in figure 4-23 that the valve is spring-loaded. Whenever the pressure in the upper compartment exceeds the pressure in the armament bay, the valve will open. When the pressures are equal, the spring will keep the valve closed. The valve will not allow the pressure differential to exceed about 0.3 psi. As the electronic compartment pressure starts to increase, the valve opens wider to keep the pressure constant. Since there is no fabric or rubber diaphragm in this valve, it should have a long service life and require no maintenance.

UPPER ELECTRONIC COMPARTMENT CHECK VALVE.

A check valve connects the upper electronic compartment to the left engine intake duct. This valve has double flappers that open in one direction only; that is the flappers open away from the upper electronic compartment. When the pressure in the electronic compartment is greater than the left engine intake duct pressure, the flappers open and air is drawn from the compartment. The valve is of very simple mechanical construction and should cause no difficulties in maintenance.

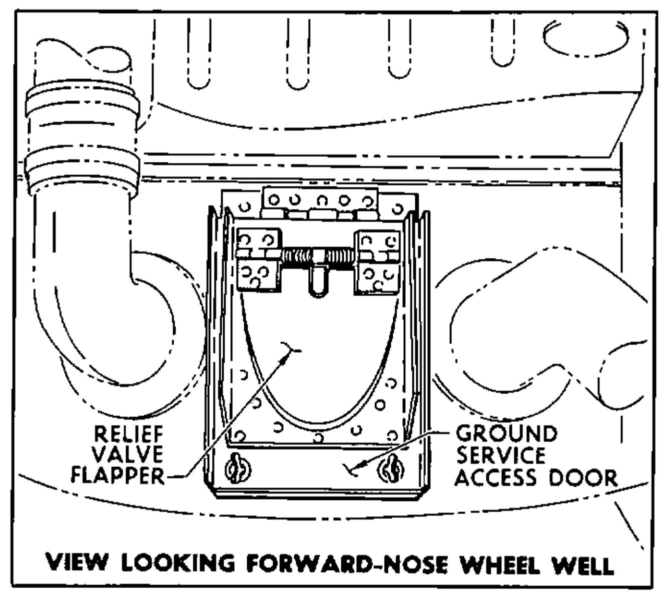


Figure 4-22. Ground Air Conditioner Fitting and Forward Electronic Compartment Relief Valve

GENERATOR COOLING.

The a-c and d-c generators are cooled by air from the engine intake. These generators are mounted in the engine accessory section on the right side of the fuselage. They are driven by a constant-speed drive unit that is connected to the engine. Note in figure 4-24 that a generator is mounted on each end of the drive unit. Note also the aluminum ducting connected to the engine intake just forward of the engine compressor which conducts air through both generator housings. When the airplane is flying at speeds over 150 knots IAS (indicated air speed), the ram air pressure in the air intake forces air through the generator housings and into the engine accessory compartment through the flapper door assembly. This air helps prevent the engine accessory compartment from overheating.

Note also that air is tapped from the ducting near the aft end and is conducted to the rudder assembly to cool the IFF units. When the airplane is on the ground or is flying at a speed less than about 150 knots IAS, the compression or suction of the engine will be greater than the ram air pressure. This results in a negative pressure, or suction, in the engine intake. In this case, the flapper door to the accessory compartment will close and another one will open to atmosphere. Air will then be drawn from outside the airplane, through the generators and into the engine intake where it is drawn into the engine. Air is also drawn from the fin to cool the IFF units. At a speed of about 150 knots IAS, the ram air pressure will balance the compressor suction and there will be no flow of cooling air through the generators. The closer to 150 knots IAS the airplane flies, the less air will flow through the generators.

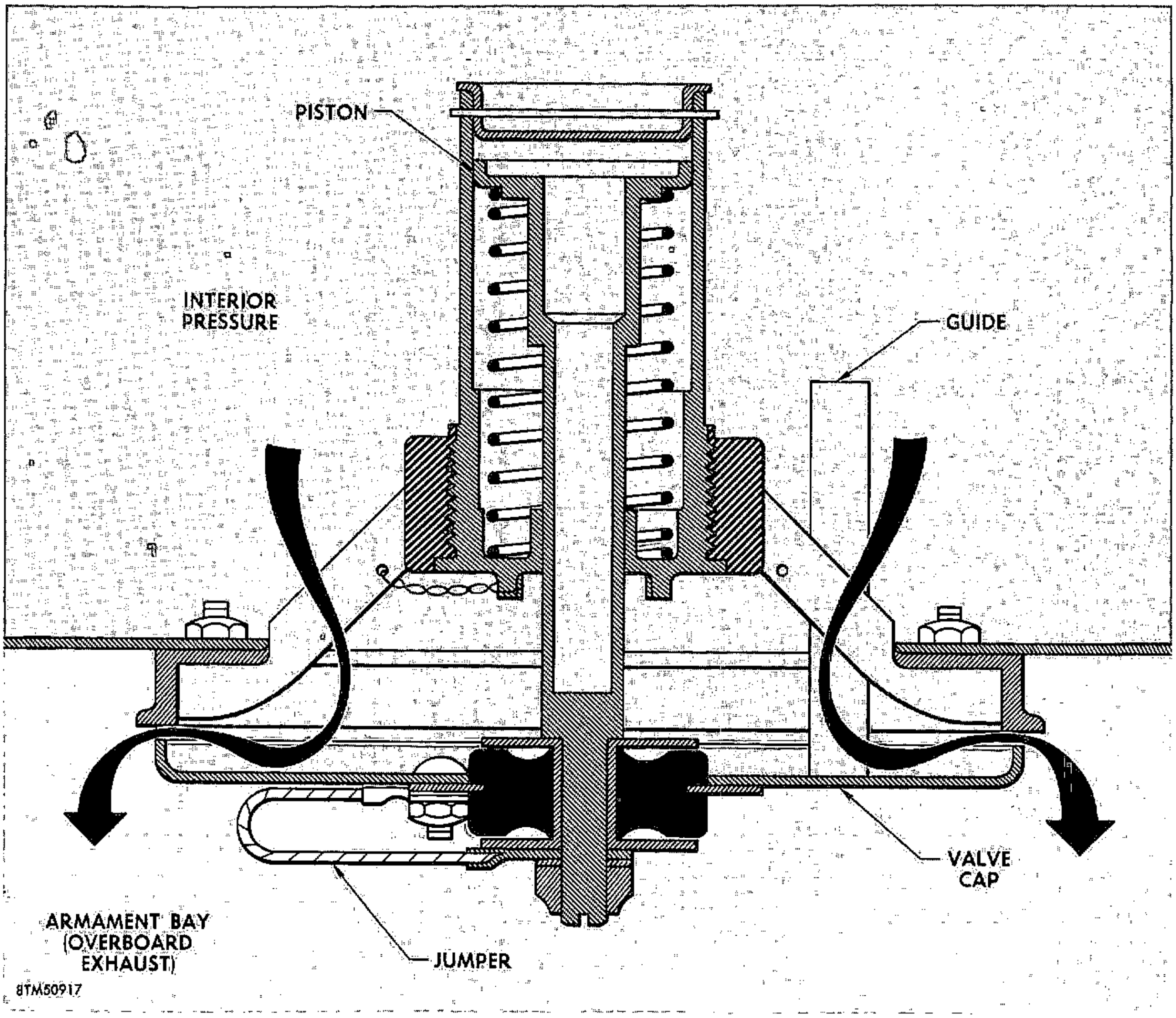


Figure 4-23. Intermediate and Upper Electronic Compartment Relief Valve Schematic

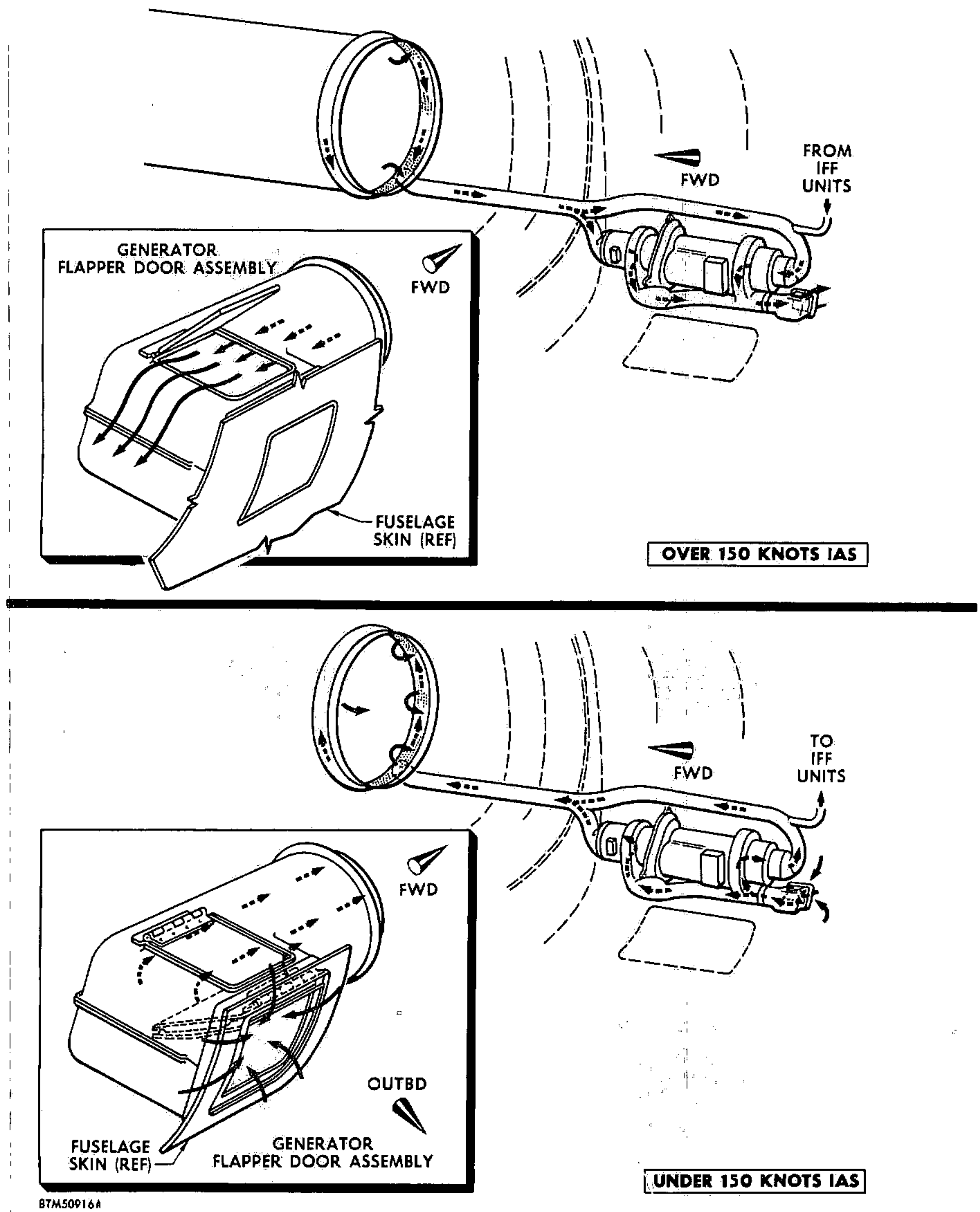


Figure 4-24. Aircraft Generator Cooling

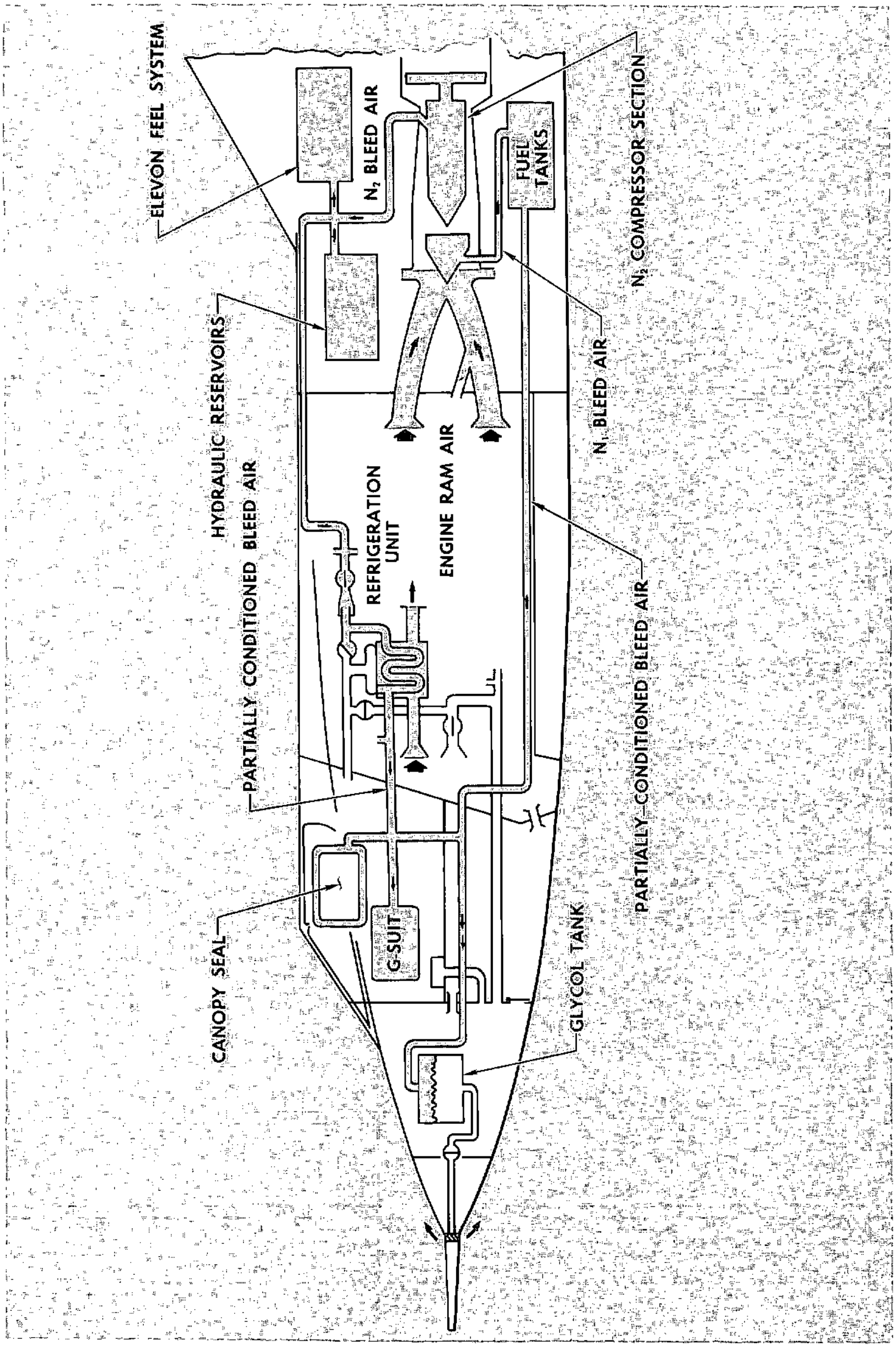


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Figure 5-1. Pressurization of Subsystems

Chapter V

PRESSURIZATION OF OPERATING SUBSYSTEMS

Contents *Page*

So far in this manual you have learned of the most important functions of the Low-Pressure Pneumatic System. These functions are cockpit air-conditioning and pressurization, and the electronic compartment ventilation. You also learned about the air supply systems. In this chapter we will discuss those systems, or subsystems, that are not operated by low-pressure air but require a supply of compressed air to function properly. Most of these sub-systems are fully covered in other manuals of this series. Some of them will be covered completely only in this manual. Those subsystems discussed in other manuals will be described briefly in this chapter while the remaining subsystems will be covered completely.

WHERE THE COMPRESSED AIR COMES FROM.

From Chapters I and II you should recall that the low-pressure pneumatic system furnishes several different kinds of air for various purposes. Ram air is normally used for electronic compartment ventilation and can also be used for cockpit ventilation. Unconditioned bleed air, tapped from the second or N_2 compressor of the J57 engine, is used in the anti-icing system, rain clearing system, and in the jet pump cooling system. Air which has been conditioned by the refrigeration unit is used for cockpit air conditioning and pressurization. Bleed air that has passed through the heat exchanger, but not through the expansion turbine, is called partially-conditioned bleed air.

It is unconditioned N_2 bleed and partially-conditioned N_2 bleed air that we are concerned with in this chapter. Notice in figure 5-1 that two subsystems are pressurized by unconditioned N_2 bleed air and four are pressurized by partially-conditioned N_2 bleed air from the heat exchanger. You can see also that one subsystem, the fuel tank pressurization system, uses not only partially-conditioned N_2 bleed air but also unconditioned N_1 bleed air. As a brief review: N_1 air is tapped, or bled, from the last stage of the forward, or N_1, engine compressor; while N_2 air is bled from the last stage of the second, or N_2, compressor. Unconditioned N_1 air always has a lower temperature and pressure than unconditioned N_2 air.

TAPPING UNCONDITIONED N_2 BLEED AIR.

Unconditioned N_2 bleed air for two of the subsystems, hydraulic reservoir pressurization and the elevon artificial fuel system, is tapped from the aft duct section of the main bleed air supply line. In figure 5-2 you can see that one pressure fitting is set into the duct section near the N_2 bleed air manifold. Notice the standard AN "TEE" connected to the pressure fitting. Two ¼-inch, stainless-steel, pressure lines connect to the "TEE" and carry pressurized air to the two subsystems. The air pressure and temperature in the pressure lines will vary but the pressure may be as high as 225 psi and the temperature may be as high as 427°C (800°F) under certain conditions.

TAPPING PARTIALLY-CONDITIONED N_2 BLEED AIR.

Partially-conditioned N_2 bleed air for four of the subsystems is tapped from the heat exchanger. You should remember from our previous discussions, that the heat exchanger is part of the refrigeration unit. Notice, in figure 5-3 that there is a manifold across the forward end of the heat exchanger. You can see the two pressure fittings on the manifold. Notice that one of the fittings is connected to a "TEE." Two ¼-inch, stainless-steel, pressure lines connect to the "TEE." One of these lines carries partially-conditioned air to the pilot's G-suit system, another carries partially-conditioned air to both the canopy seal and the glycol tank in the radome anti-icing system. A third pressure line carries partially-conditioned N_2 air to the fuel tank pressurization system. The pressure and temperature of this partially-

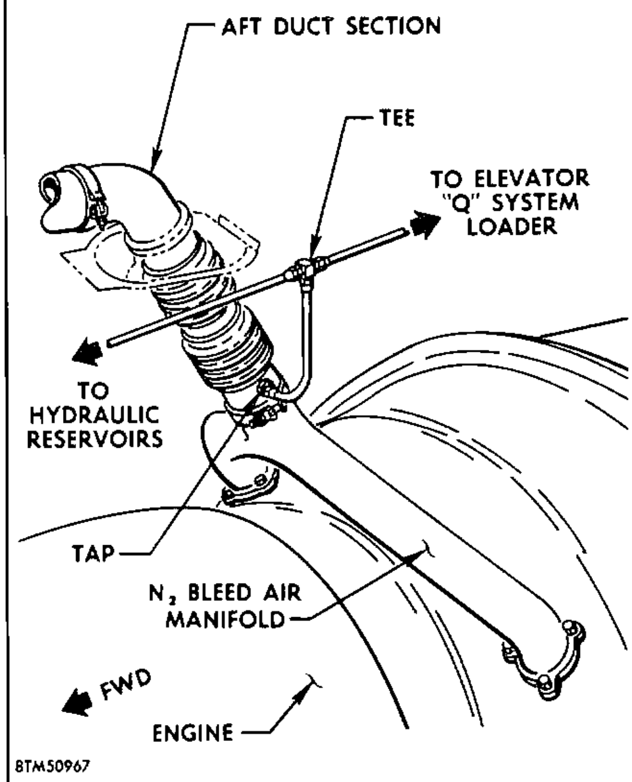

Figure 5-2. Unconditioned N_2 Bleed Air

conditioned N_2 air varies with operating conditions but may be as high as 80 psi and 93°C (200°F). Remember that no partially-conditioned N_2 bleed air is available when the main flow control valve is closed.

TAPPING UNCONDITIONED N_1 BLEED AIR.

Unconditioned N_1 bleed air is tapped from the alternate-cooling bleed air duct. Notice in figure 5-4 that this duct connects the last stage of the engine N_1 compressor section to the engine-shroud cooling air duct. The engine cooling system is covered in another training manual of this series. To put it very briefly, and by no means completely, ram air is drawn from the engine intake and passed through an engine oil cooler. After that the cooling ram air is conducted to the engine shroud cooling manifold. It then passes aft between the engine and its shroud. This layer of air helps keep the shroud, and the airplane fuselage, from becoming too hot.

Now, below 150 knots IAS (indicated air speed), the compressor creates a vacuum in the engine inlet and a check valve in the cooling duct prevents a reverse flow of air through the duct. During this condition, the alternate cooling air shutoff valve opens and N_1 air is forced into the cooling manifold in place of ram air. Notice the pressure fitting in the alternate cooling duct-

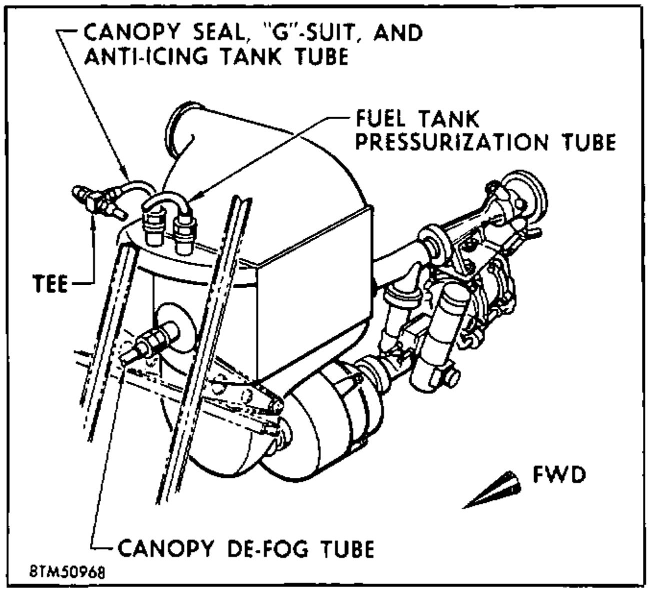

Figure 5-3. Heat Exchanger Pressure Manifold

ing just upstream from the shutoff valve. A ¼-inch pressure line connects to this pressure fitting and conducts N_1 air pressure to the fuel tanks. Remember that this is only one source of air pressure for the fuel tanks. They also are pressurized by partially-conditioned N_2 air from the heat exchanger. The pressure and temperature of unconditioned N_1 air will vary but under some conditions it may be as high as 65 psi and 205°C (400°F).

PRESSURIZED SUBSYSTEMS.

In this section we will very briefly discuss those systems that are pressurized by air from one of the three sources discussed above. Those systems that are completely covered in other training manuals of this series, or in other chapters of this manual, will be covered only as far as it is necessary to show how they are pressurized and why they are pressurized. The canopy seal is very important to the proper operation of the cockpit pressurization system; therefore, we will discuss it more completely.

PRESSURIZATION OF HYDRAULIC RESERVOIRS.

Both the primary and secondary hydraulic systems have a pressurized fluid reservoir located in the hydraulic accessory compartment. In addition to other duties, the reservoirs are storage containers for the hydraulic fluid not at work in the system. They filter the fluid and aid the hydraulic pumps in drawing fluid through the suction line without cavitation, or vapor locks, at the pumps. Both pumps have filters and are pressurized with unconditioned N_2 bleed air. Since the N_2 pressure will vary, each reservoir has its own air pressure regulator that holds the air pressure in the reservoir constant at about 50 psi. This pressure gives a pressure "head" to the fluid in the pump suction line and helps the pump operate more efficiently.

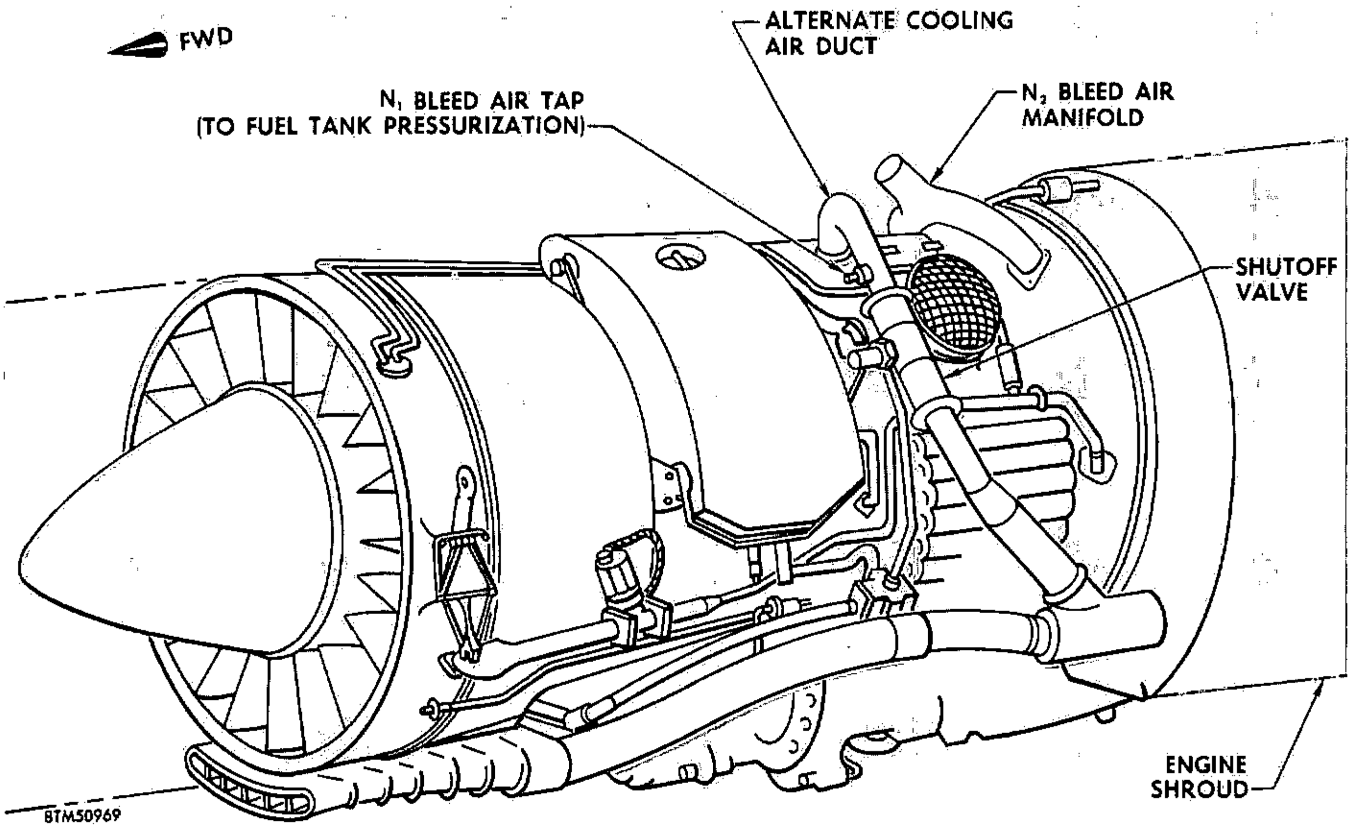

Figure 5-4. N_1 Bleed Air Pressure Tap

In figure 5-5 note that the pneumatic pressure line, connected to the "TEE" at the aft duct section, is secured to the reservoir pressurization system by a quick-disconnect coupling near the reservoirs. Notice that the pressure line then goes to a "TEE" from which separate lines go to each air pressure regulator. The regulators are the diaphragm type and are considered part of the hydraulic system; therefore, we will not discuss them further.

Quick-Disconnect Couplings.

As we learned above, the pressure line from the aft duct section is secured to the reservoir pressurization system by a quick-disconnect coupling. You will encounter this type of coupling in various parts of the pressurized subsystems. This coupling is a self-sealing type and consists of two self-sealing halves. Each half is connected to the end of a tube section. When uncoupled, both halves automatically seal themselves to prevent leakage that would ordinarily pour out through an opening.

Figure 5-6 shows a typical quick-disconnect coupling used in the F-102A. Notice that in the upper part of the illustration the two self-sealing sections are shown disconnected, while in the lower view they are shown correctly coupled. No tools are required for this connection. Simply hand-tighten the union nut until the lock teeth engage the lock spring assembly. When the lock teeth and spring assembly are correctly installed, the back side of the union nut will be flush with the flange on the valve. This type of coupling simplifies ground tests since the pressure supply line can be quickly disconnected and pressurized air from a ground source substituted.

ELEVON ARTIFICIAL FEEL SYSTEM.

The elevon artificial feel system is a part of the airplane flight controls and is discussed in another manual of this series. However, since an important component of the system is supplied with unconditioned N_2 bleed air tapped from the aft duct section, we will discuss it briefly.

In conventional, low-performance airplanes, the different control surfaces are moved by cables connected to the control stick and rudder pedals. The pilot is able to feel the force of the airstream striking the control surfaces. The faster the airplane flies and the more extreme the position of the control surfaces, the greater is the force resisting further movement. This feel, similar to "road sense" in an automobile, is very important to the pilot.

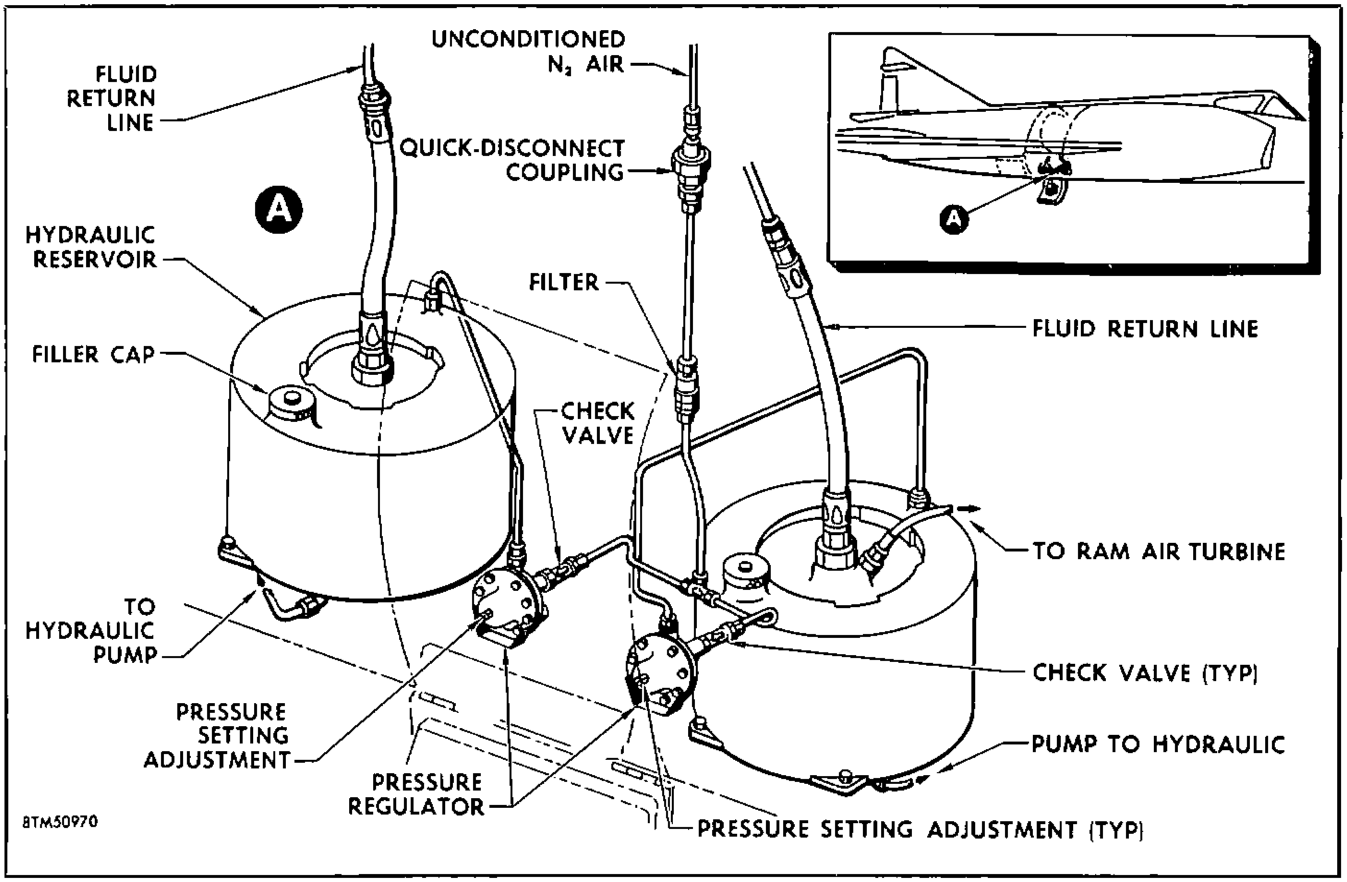

Figure 5-5. Hydraulic Reservoir Pressurization

In modern, high-performance airplanes the control stick and rudder pedals are not connected directly to the control surface. The force needed to move control surfaces at high speed is so great that a "power steering" system is necessary. In the F-102A airplane, control cables connect the control stick and rudder pedals to the hydraulic actuating systems that actually do the work. Just as the power steering system in an automobile insulates the driver from the "feel of the road," the pilot cannot feel the force of the airstream against the control surfaces. This can result in dangerous "over controlling." Thus, artificial feel systems are necessary to provide resistance to control movement. This resistance must vary with the speed and altitude of the airplane and the position of the control surfaces.

In most airplanes there are three types of control surfaces: the rudder, the ailerons, and the elevators. You probably know that the rudder controls movement of the airplane to the right or left, the elevators can cause the airplane to climb or dive, and the ailerons can cause it to bank or roll. In the F-102A, the elevators and ailerons are combined into one set of control surfaces called "elevons." The elevons can cause both elevator or aileron movement or a combination of the two. Each of the three types of movement has its own artificial-feel system. The rudder and the aileron artificial-feel systems do not use low-pressure pneumatic air so we will not discuss them further. The elevator artificial-feel system uses unconditioned N_2 bleed air and is discussed below only so far as its use of N_2 air is concerned.

The artificial-feel system that provides resistance to an elevator movement is illustrated in a simplified diagram, figure 5-7. As you can see, the movement of the control stick, in fore and aft directions only, is resisted by a centering spring and the air pressure in the feel force cylinder. The pressure of the air in the cylinder, and therefore the resistance, is controlled by the elevator "Q" system loader. This rather complicated component consists of a system of bellows and pressure chambers. The loader senses several different air pressures, compares and balances them, and varies the pressure in the feel force cylinder as the airplane speed and altitude varies.

In figure 5-8, note that the loader senses pressure from both the rudder and elevator "Q" intakes. These intakes measure ram air pressure which increases with airplane speed and decreases with altitude. The loader also receives N_2 bleed air pressure from the aft duct section. In addition the loader is vented to atmosphere. You can see that the "Q" system loader is installed in the engine accessory compartment beneath the engine and can be reached through the left engine accessory compartment door.

As we have already learned, the temperature and pressure of unconditioned N_2 air will vary but the temperature may be as high as 427°C (800°F) and the pressure may reach 225 psi. A regulator built into the "Q" system loader controls the pressure of the N_2 air that enters the control system inside the loader. The regulator maintains this N_2 air at a level that is always about 10 psi higher than atmospheric pressure. The "Q" system loader and the artificial feel system are discussed in another Training Supplement of this series which covers the F-102A Flight Controls System.

FUEL TANK PRESSURIZATION SYSTEM.

The fuel supply for the F-102A airplane is carried within the delta wings. There are no separate fuel tanks since the wings are of a "wet wing" type of construction. Each wing is divided into three separate fuel tanks that are pressurized to prevent excessive "sloshing," to help sequence the order in which the tanks are emptied, and to help the fuel pumps draw fuel from the tanks without cavitation or vapor locks. The airplane fuel system is discussed in the Fuel System Supplement of this series.

As shown in figure 5-1, the fuel tanks are pressurized with air from two sources: partially-conditioned N_2 bleed air from the heat exchanger, and unconditioned N_1 air. N_1 air is available whenever the engine is running, but partially-conditioned N_2 air is available only when the main bleed-air flow control valve is open. You should remember from Chapters II and III that this valve is closed during armament firing and when the cabin air switch is at RAM or OFF. The pressure lines from each of the two sources are connected to a common "TEE" fitting.

Notice in figure 5-9 that there is a check valve in each supply line. These check valves prevent "back surging" (reverse flow) whenever the downstream pressure is greater than upstream pressure. This also allows either N_1 or N_2 air to be used as the sole source of pressurization should one source be cut off. You can see in the illustratiton that air flows from the "TEE" fitting, through a pressure regulator, and into the fuel tanks where it cools as it expands. The regulator maintains the air pressure in the fuel tanks constant at about 4.5 psi over atmospheric pressure. Remember that partially-conditioned N_2 air may have a pressure as high as 80 psi and a temperature of about 93°C (200°F). N_1 air pressure may reach 65 psi with a temperature of around 205°C (400°F). The air pressure regulator is located in the forward left side of the main wheel well.

THE CANOPY SEAL SYSTEM.

The canopy of the F-102A airplane is sealed against rain or dust leakage and against loss of cockpit pressurization by three different types of seals, one of which is a pressure type supplied with partially-conditioned N_2 air from the heat exchanger. The canopy, its latching

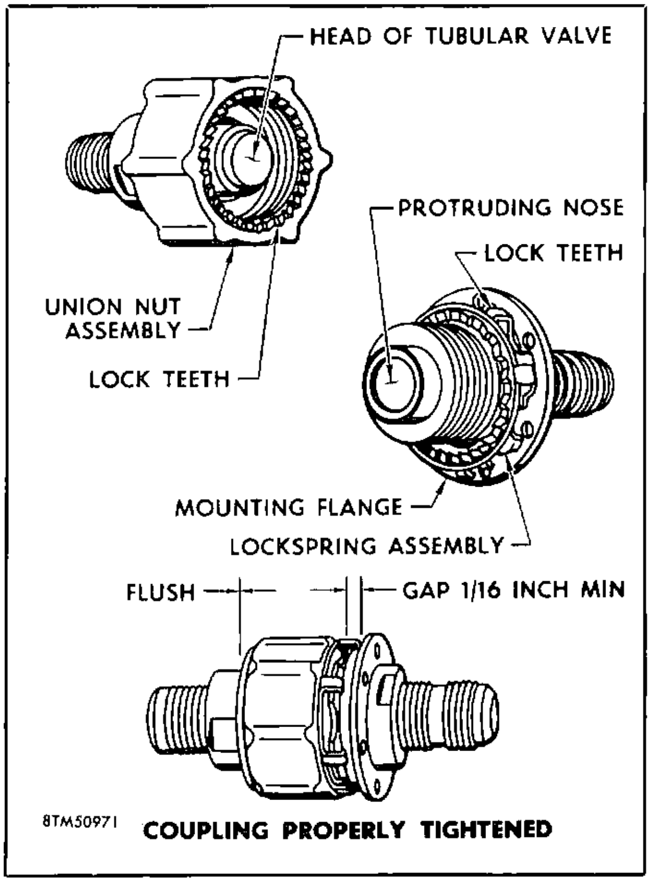

Figure 5-6. Quick-Disconnect Coupling

mechanism, and the canopy seal system are discussed in the Airplane General Supplement of this series. In this chapter we will mention the other two types of canopy seals, but we will study only the pressure seal.

The first seal is a diaphragm type that prevents leakage or loss of pressure between the canopy window panels and the canopy frame. This seal extends around the top, bottom, and aft edges of each window panel. The open ends of the seal are closed with rubber plugs that are sealed in place with sealer compound. The forward edge of each window panel is sealed with a bead of sealer compound.

The second seal is called the canopy rain-and-dust seal. This seal is tubular in shape and extends around the aft edge of the windshield frame. It ends at the lower forward corners of the canopy. This seal is attached to the windshield frame by integral tabs pulled through holes in the frame. When the canopy is closed this seal is wedged tightly between the canopy and the windshield.

The pressurized canopy seal consists of a continuous, molded, synthetic rubber tube installed in a channel around the pressurized portion of the canopy. As you

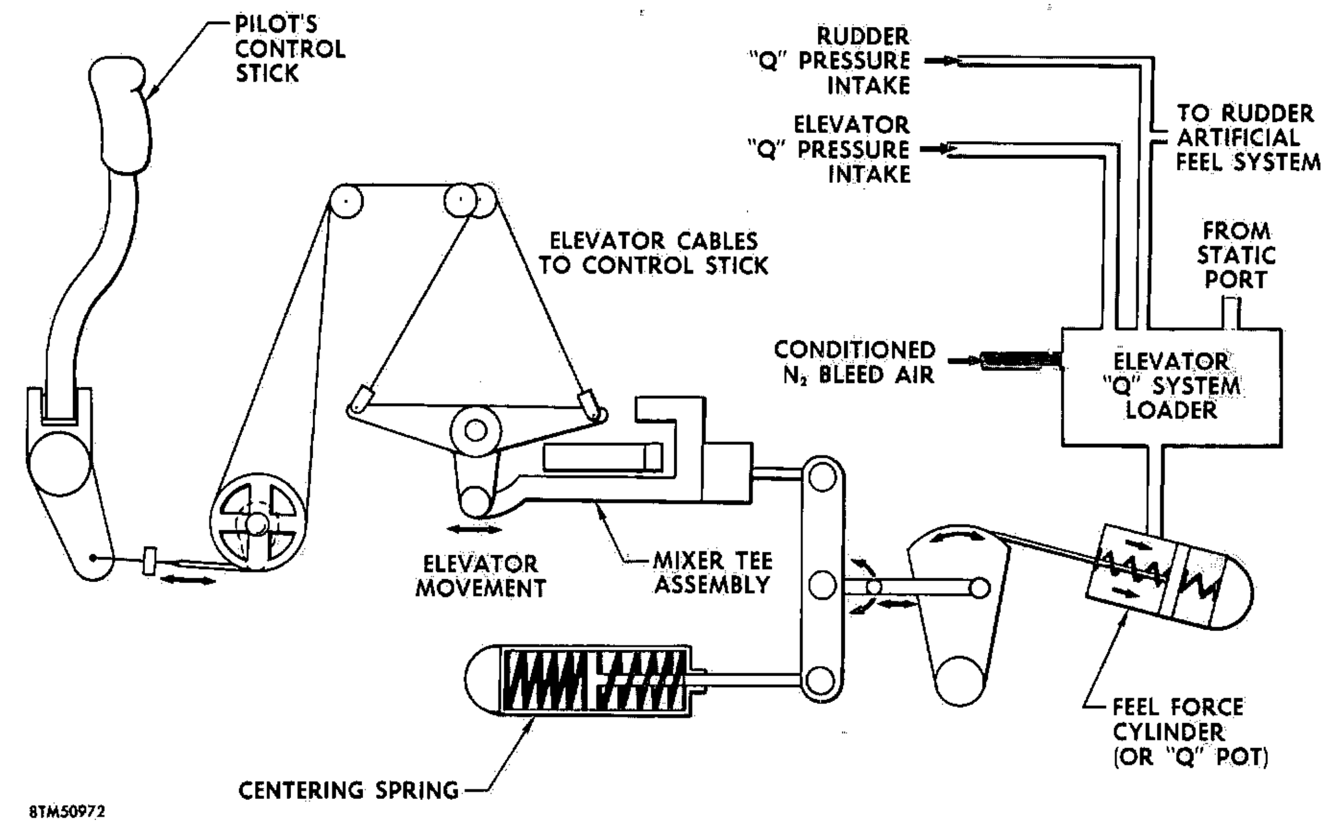

Figure 5-7. Elevator Artificial Feel System Schematic

can see in figure 5-10, the tube is supplied with partially-conditioned N_2 bleed air from the heat exchanger. Notice that the pressure line from the heat exchanger is connected to the canopy seal selector valve. This valve opens to route air pressure to the seal when the canopy is latched, and closes to depressurize the seal when the canopy is unlatched. Notice also the relief valve and the safety valve. The relief valve opens when the seal pressure reaches about 16 psi and vents the excess air from the airplane through the dump louvers aft of the canopy.

There will always be at least a small amount of flow through this relief valve. This continuous flow prevents the rubber seal from becoming cold and brittle. As you know, N_2 air from the heat exchanger may have a temperature of 93°C (200°F). You can also see the safety valve in the seal line. This valve is exactly like the relief valve except for pressure setting. If the relief valve should fail to operate, the safety valve will open at about 30 psi pressure and will dump the excess air into the intermediate electronic compartment. You should remember from Chapter III that there is a canopy seal test fitting in the nose wheel well. When the canopy seal is to be ground tested, you must connect a source of compressed air to this fitting. Notice the check valve in the test line. This allows air to flow in one direction only.

The canopy seal selector valve also acts as a check valve. Only one of the check valves is open at a time depending on which pressure source is used. If N_2 air is used, the test line check valve is closed, but if another source of air is attached to the test fitting, the selector valve closes. Notice also the two filtered orifices, one in the N_2 supply line and the other in the test line. These orifices contain bronze filters and also act as pressure reducers to help prevent excessive canopy seal pressure. The seal itself is installed in a retainer channel around the edge of the canopy. The pressure line to the seal is a flexible rubberized tube that is vulcanized to the seal as shown in figure 5-10.

The Canopy Seal Selector Valve.

As you can see in figure 5-10, the canopy seal selector valve is mounted on the aft side of the cockpit canted bulkhead in the top right corner. It can be seen from the upper electronic compartment but you can probably reach it easier from the intermediate electronic compartment. You can see that two pressure lines connect to the valve; one line from the heat exchanger and the other to the canopy seal. The valve is operated by a cam which in turn is operated by a flexible cable connected to the canopy latch handle at the forward right side of the cockpit. When the handle is pulled, the valve will close to cut off pressure to the seal. When it is pushed in, the valve will open.

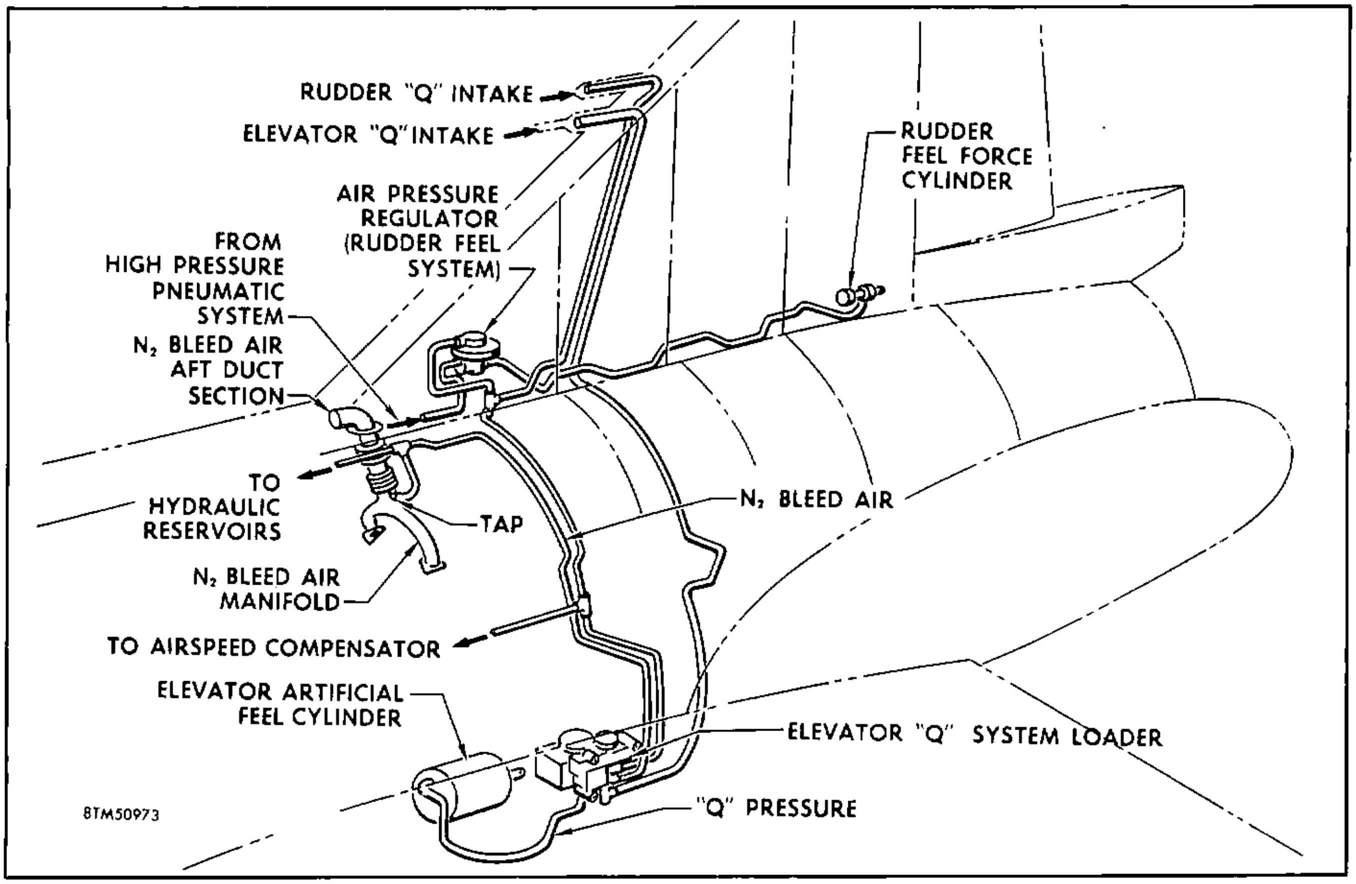

Figure 5-8. Elevator Feel Force Control System

Now look at the schematic cross section of the valve in figure 5-11. Notice the three ports; one connects to the N_2 supply line, another connects to the canopy seal line, and the third is a dump port. As you can see, a spring and ball blocks an opening from the N_2 port and acts as a check valve to prevent any reverse flow through the valve. It takes about 14 psi of pressure (on the intake side) to force the check valve open to allow N_2 air to flow through the selector valve.

Now notice the shaft of the double poppet. The shaft is connected to a clevis and the clevis in turn rides in a slot in the cam. When the canopy latch handle is pulled or pushed, a cable moves the cam and the slot causes the clevis and shaft to travel in or out. When the shaft is pushed in, N_2 air can travel through the valve to the canopy seal. When it is pulled out, N_2 air is cut off and the pressure in the seal is vented through the vent port. The canopy seal selector valve should cause little trouble. When the seal system is tested and any defects in the valve show up, you must replace it as described in T.O. 1F-102A-2-2.

Relief Valve and Safety Valve.

The canopy seal relief valve and safety valve are almost exactly the same except for the pressure setting. The relief valve opens at about 16 psi and the safety valve opens at about 30 psi if the relief valve should fail. Notice in the top half of figure 5-12 that each valve consists of a spring-loaded poppet which opens when the air pressure exceeds the spring strength. Since these valves are so identical in appearance, you must be very careful to install the correct valve whenever replacing one of them. Always check the valve number and pressure setting as stated on the valve nameplate. Notice also the IN and OUT markings on the valve body. Always install the valve with the end marked IN facing downstream. This is very important since too much pressure could rupture the seal. The relief valve is installed on the canopy bulkhead and the safety valve is mounted on the aft side of the cockpit canted bulkhead.

Filtered Orifices.

The two filtered orifices are identical and interchangeable. You can see in the second half of figure 5-12 that each orifice contains a porous bronze filter and a restrictor plate. The orifices prevent dirt from entering the system and restrict the flow of air to help reduce the pressure. Never attempt to replace a filter but always replace the unit if the filter becomes clogged. The orifices can be installed in either position. Both of them are positioned at the aft side of the cockpit canted bulkhead.

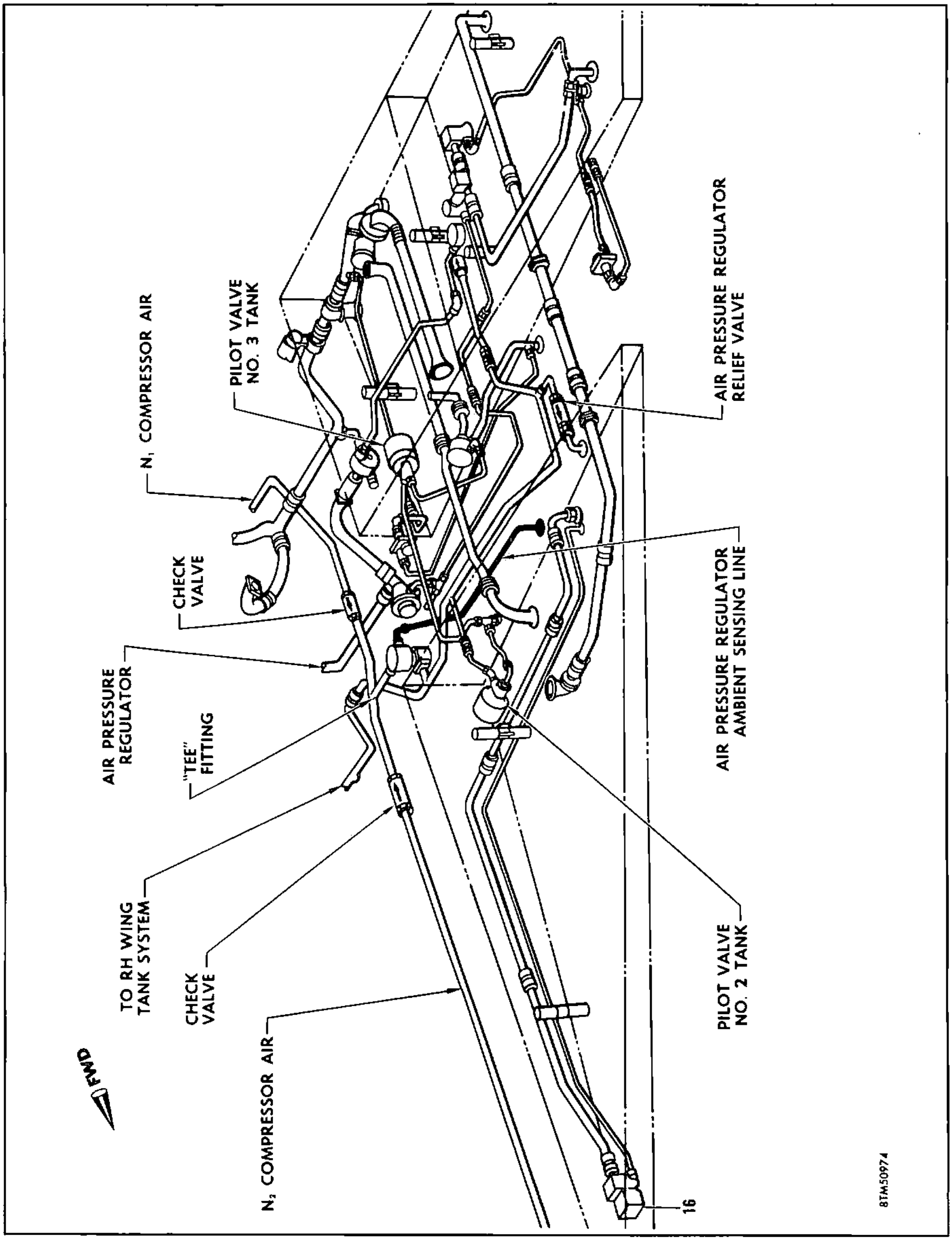

Figure 5-9. Fuel Tank Pressurization System

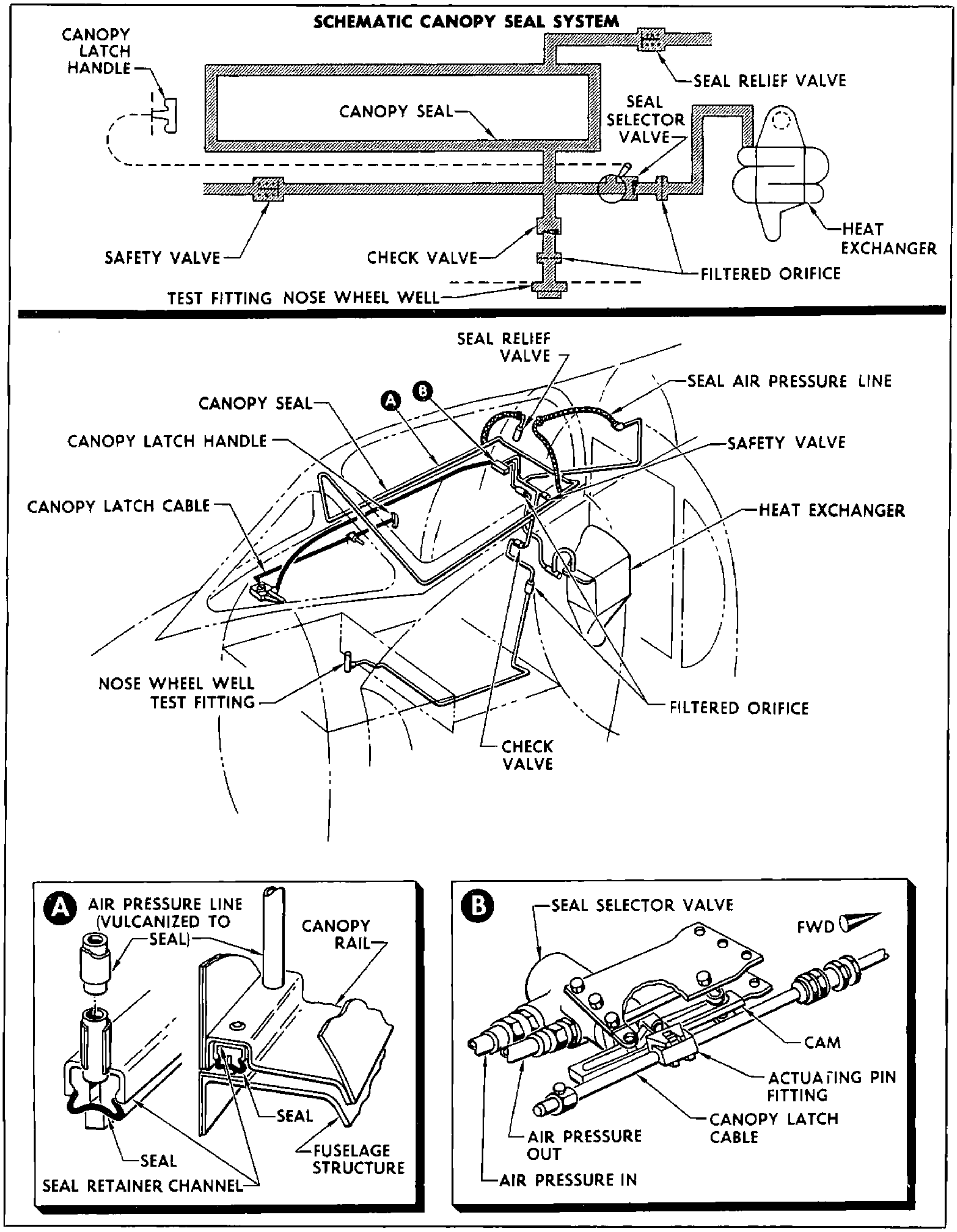

Figure 5-10. Canopy Seal System

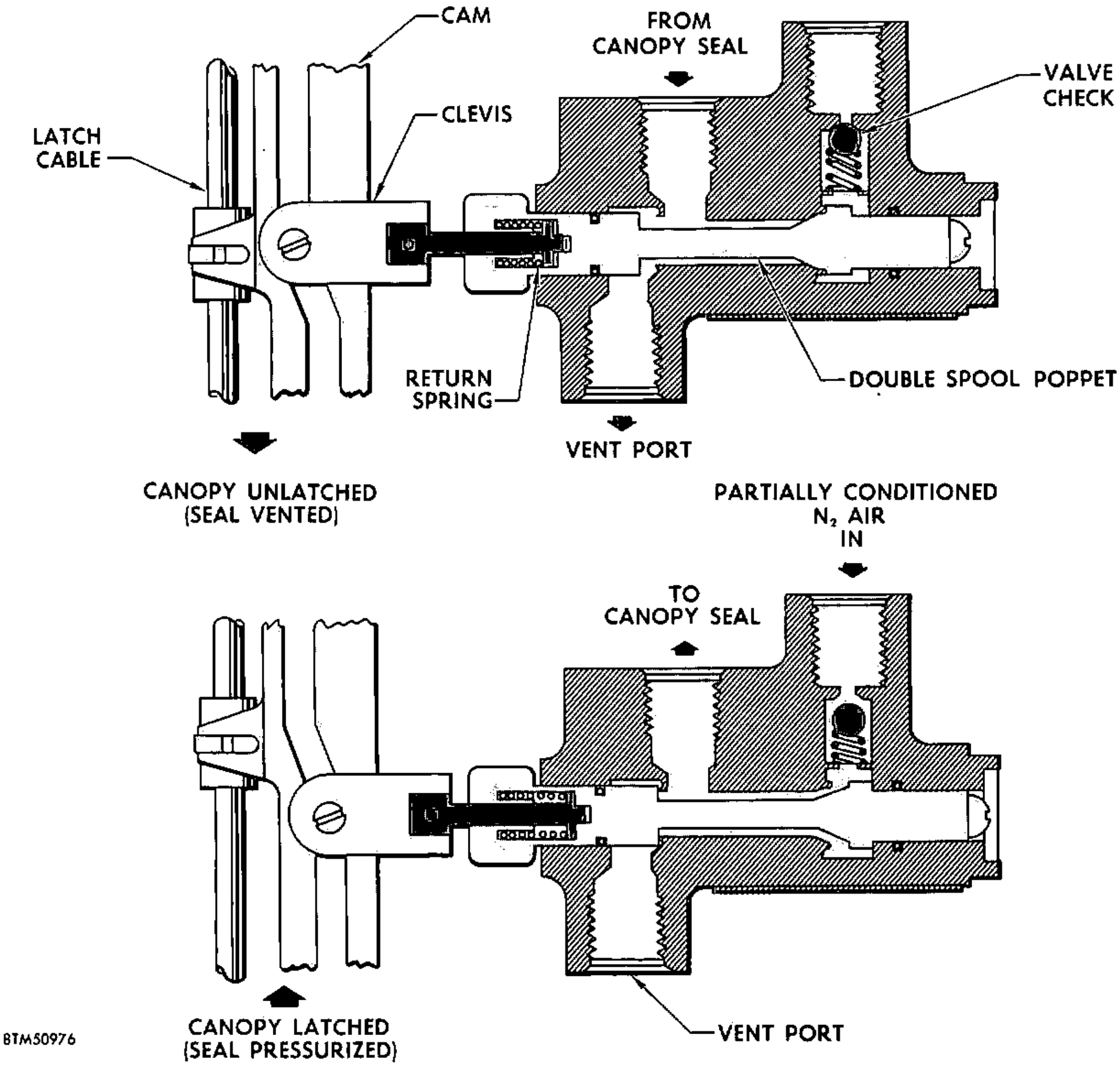

Figure 5-11. Canopy Seal Selector Valve Schematic

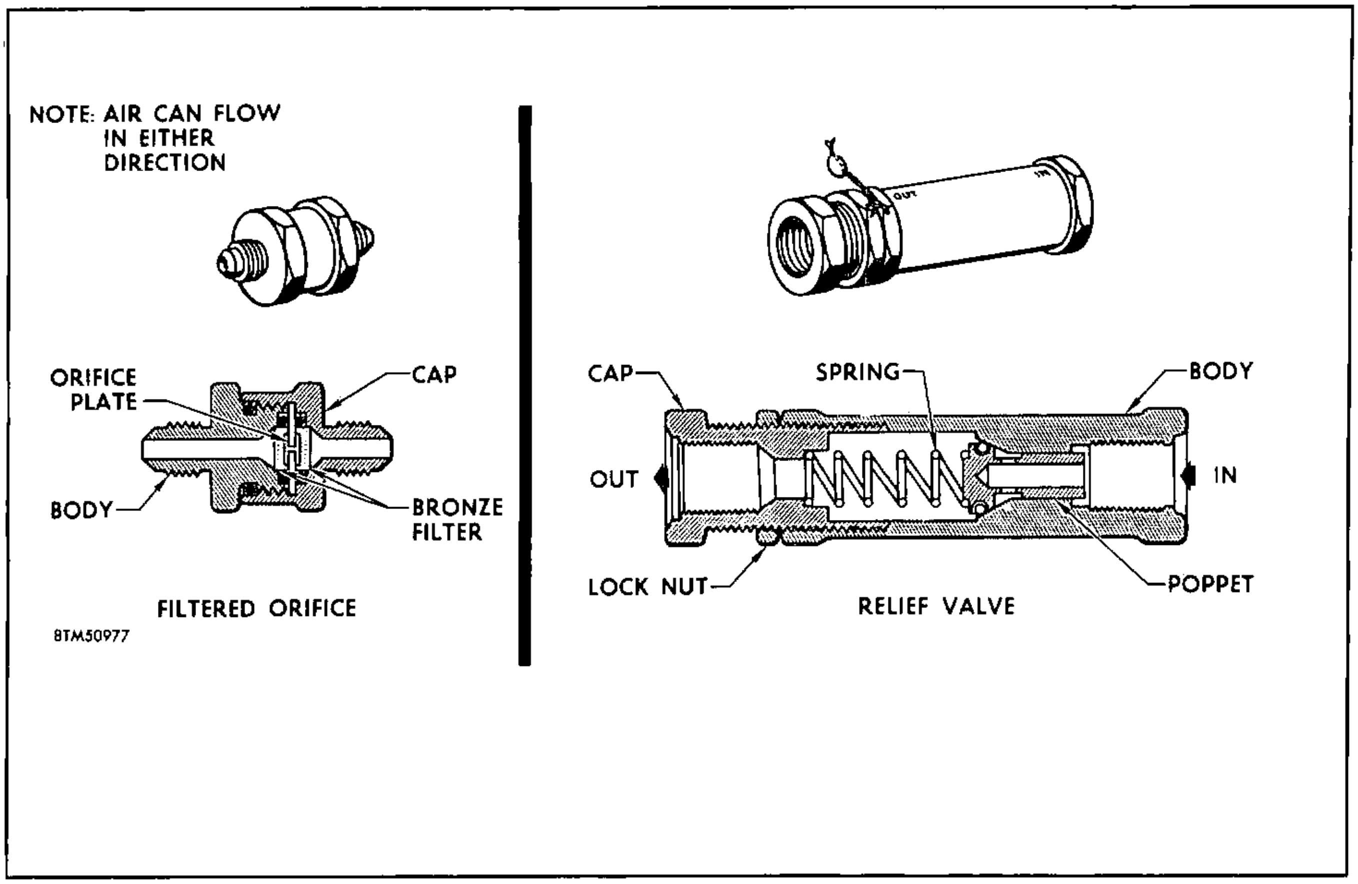

Figure 5-12. Canopy Seal Relief Valve and Filtered Orifice

Check Valve.

The check valve in the canopy seal test line is very similar to the relief valve or safety valve in appearance and construction. It contains a spring-loaded poppet that will open when the pressure on the IN side of the valve reaches about 14 psi. It is a one-way valve that allows a flow in one direction only. It is also mounted at the aft side of the cockpit canted bulkhead and is reached from the intermediate electronic compartment.

Testing the Canopy Seal System.

In your F-102A Maintenance Manual (T.O. 1F-102A-2-2) you will find a complete description of the functional tests that must be performed periodically on the canopy seal system. Follow the directions carefully and make sure you use the proper test equipment as described in the Technical Order. Replace all items found defective without attempting any adjustment. Be sure to tightly cap the test fitting in the nose wheel well after completing the tests.

THE GLYCOL TANK PRESSURIZATION SYSTEM.

The formation of ice on the radome of the F-102A airplane is prevented by an anti-ice system that spreads a film of glycol fluid over the surface of the radome when anti-icing conditions exist. The glycol is contained in the pressurized reservoir, or tank, located in the nose of the airplane. The glycol tank is pressurized by partially-conditioned N_2 air from the heat exchanger. The radome anti-icing system and the glycol tank pressurization will be completely covered in the next chapter of this supplement; however, let us take a quick "preview" here.

Note that the glycol tank is connected by tubing to a porous metal ring at the tip of the radome immediately aft of the pitot-static boom. When the radome anti-ice valve is opened, either manually or by an automatic ice detection system, glycol is forced under pressure through the porous ring and overboard to form a protective film on the radome. Notice that there are two filtered orifices, a relief valve, a safety valve, and a check valve in the pressure line. These items are either identical with, or similar to, the equivalent items in the canopy seal system. The relief valve maintains the air pressure at about 5 psi while the safety valve opens at about 16 psi if the relief valve fails. When the anti-ice valve is shut off, there will be a small continuous flow of air through one of the filtered orifices and through the glycol line and overboard. This small flow keeps the lines clear and prevents icing on the porous metal ring. Remember that no N_2 air is available for glycol pressurization when the refrigeration unit flow control valve is closed.

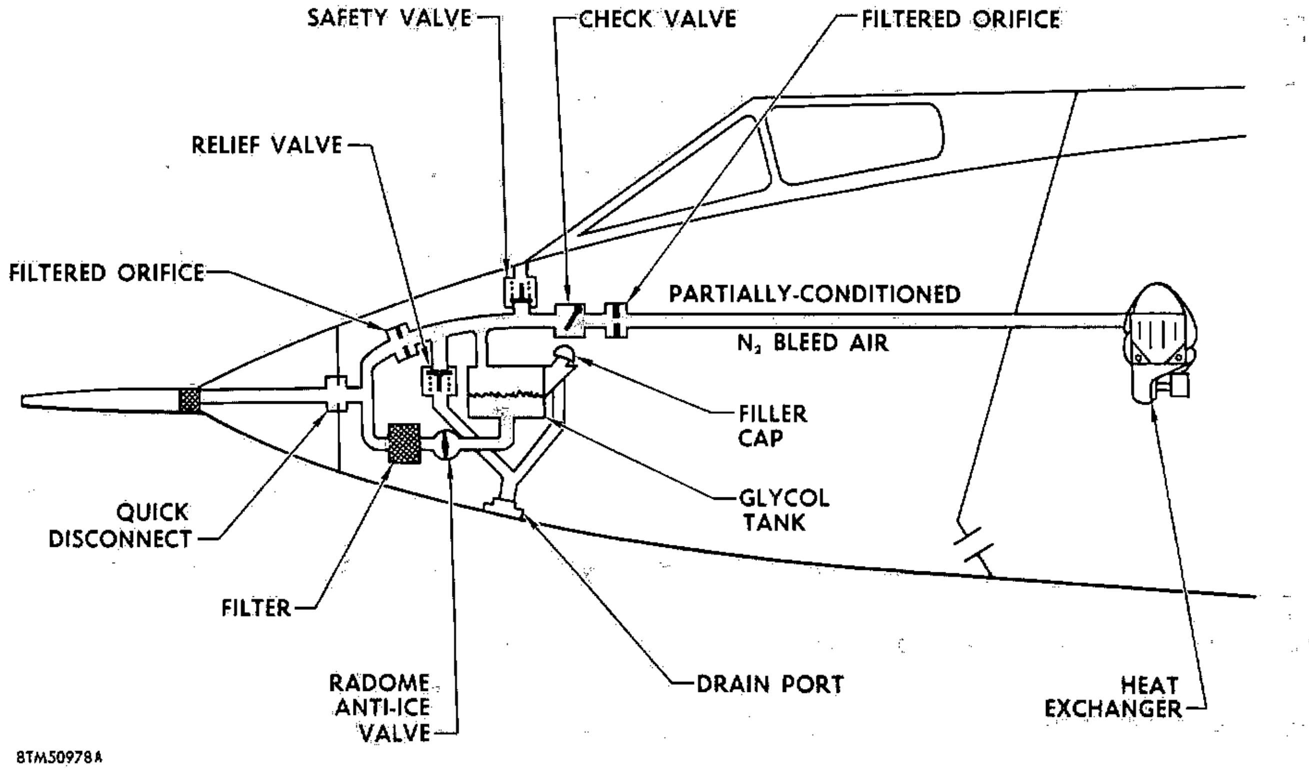

Figure 5-13. Glycol Tank Pressurization Schematic

THE PILOT'S G-SUIT PRESSURIZATION SYSTEM.

The pilot of the F-102A airplane is often subjected to violent changes of directions or sudden accelerations. These sudden changes are very hard on the human body. This is easy to understand if we think about one of the fundamental laws of physics: the one about inertia. To quote the classical form of the law, "Inertia is the tendency of a body when at rest to remain at rest, and when in motion to remain in motion." When an airplane dives at high speeds every particle of the pilot's body is also going at high speed. If the airplane should "pull out" suddenly, all of the individual parts of the body would tend to continue in the old direction. The main effect of this would be a "blackout"; that is, blood would rush from the pilot's head to his feet. The brain, starved for blood, would cause a momentary blackout until conditions stabilized again.

This force will vary according to the violence of the maneuver and is measured in "G's." A force of one "G" is equal to the force of gravity. When at rest, we are subject to a one-G force. The more violent the maneuver, the higher will be the G-load on the pilot. To protect himself from the effect of G loads, the pilot wears a G-suit. This suit, which is an item of individual issue, is pressurized with partially-conditioned N_2 air from the heat exchanger. The higher the G load, the more the suit must be pressurized. This suit applies pressure at certain parts of the body and helps prevent the blood from rushing from the head and causing a blackout. The suit is also connected to the pilot's bail-out oxygen bottle and under certain conditions oxygen is used to pressurize the suit. The suit itself and the oxygen bottle will not be discussed further in this manual. However, the supply and control of partially-conditioned N_2 air for the G suit is discussed here.

In figure 5-14, note that air from the heat exchanger is conducted to a pressure regulator valve on the pilot's left console. The regulator opens automatically when the G loads reach 1½ to 2 G's. The regulator controls the G-suit pressure and increases it with the G load. The maximum pressure in the G-suit is about 10 psi at the higher G loads. The regulator is an inertia type that senses the G loads and increases G suit pressure as the G loads increase. Notice the control knob with the HI and LO positions.

This knob varies the tensions of a spring inside the regulator to allow the pilot to increase or decrease the pressure in the suit to agree with his personal preference. Except for this control knob, the action of the regulator is entirely automatic. The regulator should be removed from the airplane from time to time and bench tested as described in T.O. 1F-102A-2-3. This regulator

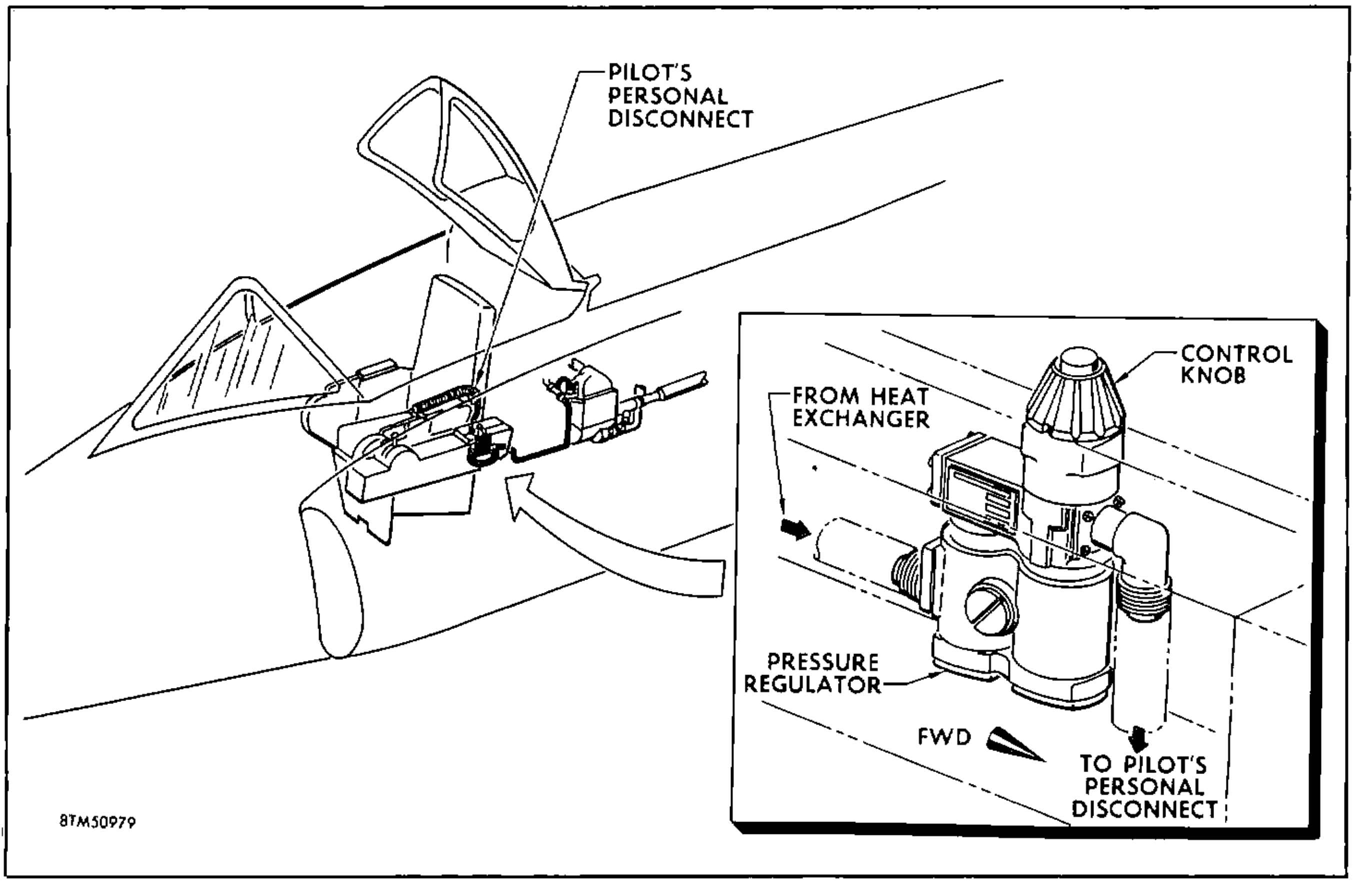

Figure 5-14. Pilot's G-Suit Pressurization System

must be replaced when defective. Do not attempt to repair it without studying the detailed instructions in the technical order.

The pressure line from the regulator does not connect directly to the G-suit. Instead, it connects to a personal-leads disconnect unit at the aft end of the left arm of the pilot's seat. The pressure line from the G-suit also connects to this unit. This disconnect unit, completely described in another training supplement and in your Maintenance Manual, T.O. 1F-102A-2-3, provides a separation point during emergency seat ejection.

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Chapter VI

ANTI-ICING AND DE-FOGGING SYSTEMS

Contents *Page*

In this final chapter we will learn about those systems that prevent ice from forming on critical areas of the airplane and those systems that keep the windshield, canopy, and pilot's mask free from fog or ice which might obscure the pilot's vision. These systems use either hot N_2 bleed air, electrical current, or glycol fluid to perform their functions. Some of the systems are controlled by an automatic, electrical, ice detection and control system, while others are pilot controlled by means of switches in the cockpit. Although some of the individual electrical anti-icing or de-fogging systems have nothing to do with the low pressure pneumatic system, they are included in this chapter so that a reader interested in anti-icing or de-fogging can find all of his information in one place. Remember also that anti-icing and de-fogging are very important functions. You find them discussed in this last chapter only because it is more convenient to discuss them here, not because they are of any lesser importance.

WHY WE NEED ANTI-ICING AND DE-FOGGING SYSTEM.

Anti-icing and de-fogging systems are needed for many reasons. First of all, many of the flight instruments, such as the altimeter and the air speed indicator, need to sense atmospheric pressure (both ram and static) in order to give an accurate reading. These instruments sense pressure through pressure-sensing ports on the pitot static tube on the nose of the airplane. If these ports are clogged by ice, the instrument reading would be false. The ram air pressure sensing intakes (Q intakes) are on the leading edge of the airplane fin and supply ram air pressure to the flight control artificial feel systems and pitch and yaw damper system. These systems may operate erratically if the intakes are clogged with ice. Ice in the engine air intakes can reduce the supply of air to the engine and cause it to stall because of air starvation. Ice on the plastic radome can interfere with radar reception and reduce the efficiency of the airborne radar.

The defogging systems are very important for adequate pilot vision. The airplane may fly blind on instruments or under radar control, but it has a poor psychological effect on the pilot to do this continuously. Trying to land on an emergency strip while peering through a glazed windshield is hazardous. The canopy and pilot's mask can also "mist up" and interfere with the pilot's vision. The anti-icing and de-fogging systems of the F-102A are designed to reduce or remove the dangers mentioned above. A malfunction of any one of these systems will reduce the efficiency of the pilot, interfere with the proper operation of control systems, give false instrument readings, or prevent the performance of the primary mission; that is, the interception of an enemy air attack.

ANTI-ICE AND DE-FOG CONTROLS.

As we noted at the beginning of this chapter, some of the individual anti-icing or de-fogging systems are always pilot controlled by means of switches in the cockpit. The remaining systems are normally controlled

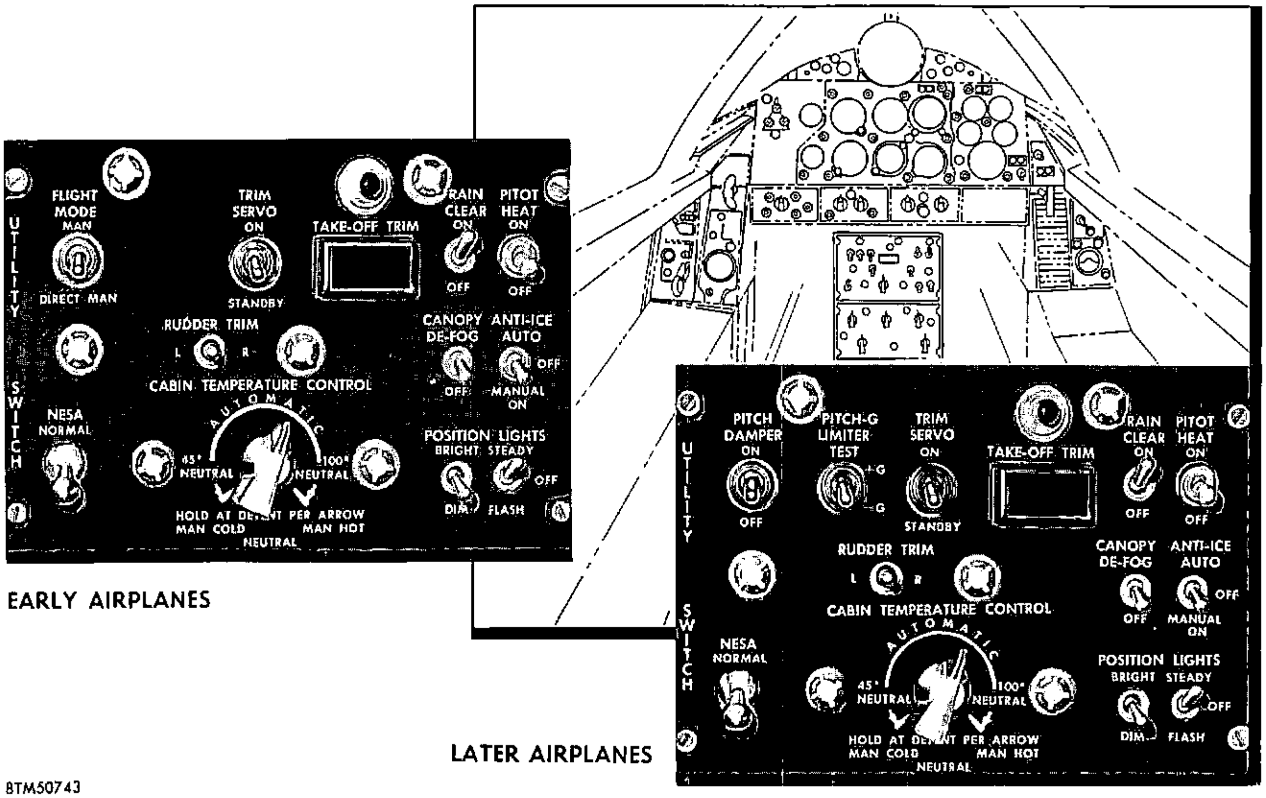

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Figure 6-1. Anti-Ice and De-Fog Controls

by an automatic anti-ice detection and control system, but the automatic system can be overridden manually by means of a selector switch in the cockpit. Therefore, *all* of the anti-icing and de-fogging systems *can* be manually controlled when necessary.

MANUAL CONTROLS.

All but one of the manual anti-ice and de-fog controls are mounted on the utility switch panel, which is just forward of the pilot's control stick. These controls are shown in figure 6-1. Notice that all of the anti-icing and de-fog switches are two-position toggle switches, except the anti-ice switch which has three positions. The pilot's mask de-fog control is mounted on the pilot's left console and is therefore not shown in the illustration.

THE ANTI-ICE DETECTION AND CONTROL SYSTEM.

The anti-ice detection and control system detects the formation of ice in the engine intake duct and automatically actuates four of the anti-ice subsystems. These four subsystems are the engine, the inlet duct, the Q-intake, and the radome anti-ice systems. The anti-ice detection and control system consists of an ice detector assembly, an interpreter assembly, an anti-ice control relay, and an ignition power relay. An ice detector indicator relay causes a warning light in the cockpit to illuminate if the control system malfunctions. An anti-ice switch allows the four anti-ice subsystems to be manually operated if the control system malfunctions.

By following the wiring schematic (figure 6-2) carefully, you will obtain a better understanding of how this system operates. Note the ice detector assembly and the interpreter assembly. These two assemblies are the "heart" of the control system. The detector assembly detects the presence of ice in the engine intake duct and signals the interpreter assembly. The interpreter assembly then energizes the anti-ice control relay which in turn actuates the four anti-ice systems. This action is rather complicated, but if you follow the schematic carefully, you should be able to "follow the electrons" through the circuit.

First let us trace the current flow when the anti-ice switch is in AUTO position. Note the 10 amp anti-ice power circuit breaker. All power for the detector and interpreter assemblies comes from the 28-volt d-c essential bus through this circuit breaker. The current flows from the circuit breaker to contact 2 on the ignition power relay and through the switch to contact 3, then on to connection A_1 on the interpreter assembly. The

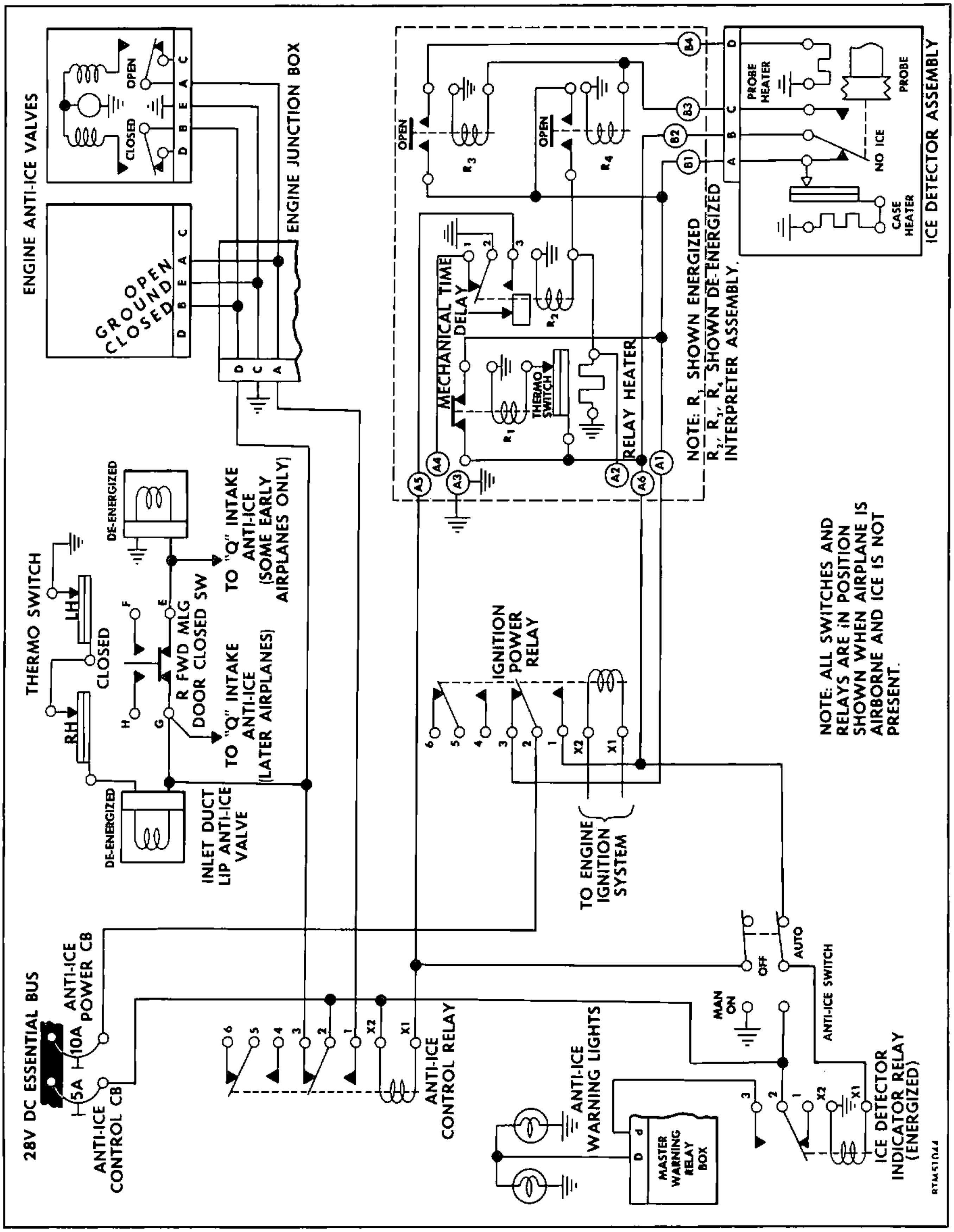

Figure 6-2. Anti-Ice Detection and Control System Electrical Schematic

ignition power relay opens the switch across 2 and 3 when the engine is being started and connects 2 to 1. Connection A_1 is always connected directly to power except during an engine start.

During an engine start, power is connected directly to A_6 through the switch at connections 1 and 2 on the ignition power relay, instead of A_1. Note that power from connection A_1 goes to B_1 and on through the switch—in the *no ice* position—across A and B on the ice detector assembly to B_2 on the interpreter, and then back to A_6. When an icing condition does not exist, both A_1 and A_6 *are* connected to power. Relay R_1 is connected to A_6 and also grounded. You can see, therefore, that the relay is energized in the *no-ice* condition. When relay R_1 is energized, its switch is closed, which provides another "hot" line across A_6 and A_1. You can see that relays R_3 and R_4 are connected to B_3 and contact C on the detector, and are de-energized in the *no-ice* condition. Since R_2 is energized only when R_4 is energized, it is also de-energized when there is no ice.

Now you may ask "What happens when the detector probe ices up?" First of all, the detector causes its switch to break contact between A and B, and connects B to C. A_6 is connected to A_1 through the switch in the A_1 relay. Therefore, B_3 is now "hot" since it is connected to A_6 through C, B, and B_2. Relays A_3 and R_4 now energize. This results in the probe heater being connected to power through B_4, the R_3 relay switch, and A_1. The R_2 relay moves its switch across its contacts 2 and 3 to furnish a ground to the anti-ice control relay which is then energized by current from the 28-volt essential bus through the 5-amp anti-ice control circuit breaker. The anti-ice control relay contacts now close. This completes the circuit across contacts 1 and 2. When the probe heater heats up, it melts the ice on the probe and the detector switch returns to the *no-ice* position.

Relays R_2, R_3, and R_4 de-energize, and the probe heater and the R_1 relay heater are disconnected from power. Note the mechanical time delay at relay R_2. This is a clock mechanism that keeps the switch across 2 and 3 closed until 60 seconds after the relay de-energizes. This 60-second delay feature assures that the anti-ice systems will be *on* long enough to do some good, even if the detector switch returns to the *no ice* position after only a few seconds. Normally, the probe heater will melt the ice at the probe and return the switch to the *no ice* position within 17 to 20 seconds. If it does not, it indicates that the detector is malfunctioning. It is the function of the thermoswitch at relay R_1 to keep the probe heater from overheating the probe and to de-energize the relays during any malfunction.

About 17 to 20 seconds after the detector switch moves to the *ice* position—across B and C the R_1 relay heater will cause the thermoswitch to heat up sufficiently to break the circuit to R_1 relay. The R_1 switch will then open, and A_6 and B_2 will no longer be connected to power. Since B_3 is connected to B_2 across the detector switch—at B and C—the R_2, R_3, and R_4 relays will de-energize, and the probe heater and the R_1 relay heater will be disconnected. Sixty seconds later the switch at the R_2 relay will move across 1 and 2, the anti-ice control relay will de-energize, and the four anti-ice subsystems will shut down.

Malfunction of the detector destroys the effectiveness of the automatic control system, and the anti-ice systems will not operate. If the pilot is informed of the malfunction, he can move the anti-ice switch to the MAN ON position, thus grounding the anti-ice control relay. The four anti-ice subsystems will then operate until he moves the switch to OFF. The warning lights on the right hand auxiliary instrument panel illuminate to warn the pilot of the malfunction. These lights will come on when the master warning box is energized. The warning box is connected to contact 3 on the ice detector indicator relay. When the relay is energized, contact 3 will not be connected to power. Note that the relay is conneced to point A_6 on the interpreter assembly through the anti-ice switch when it is at the AUTO position. A_1 is normally powered and A_6 is normally connected to A_1; therefore, the detector interpreter relay is normally energized. When the ignition power relay is energized, A_6 will always be connected to power across contacts 1 and 2.

When a malfunction occurs, the connection between A_1 and A_6 is broken, across the detector switch and also across the R_1 relay switch, and the ice detector indicator relay is de-energized *if* the ignition power relay is de-energized and the warning lights will come on. The warning lights are on the right side of the instrument in the column of warning lights for other systems. It is marked anti-ice and is easily distinguished from the others when it is on. Since the ice detector relay is disconnected from power when the anti-ice switch is OFF, the warning lights will be on. You can see that the relay is always energized when the switch is at MAN ON. The warning light will therefore be on during a malfunction of the detector or interpreter assembly, or when the anti-ice switch is at OFF.

THE ICE DETECTOR ASSEMBLY.

The ice detector assembly is mounted in the upper part of the engine air intake duct. As you will note in figure 6-3, the assembly consists of a main housing, or case, and a probe. Notice the pressure diaphragm in the case. The chamber on one side of the diaphragm is connected to openings on the upstream side of the probe, while the chamber on the other side senses pressure from openings on the downstream side of the probe. When the airplane is moving, the upstream pressure will exceed the downstream pressure.

The diaphragm controls the ice detector switch. When the airstream is moving at less than 40 knots, the pressure difference across the diaphragm is not sufficient to

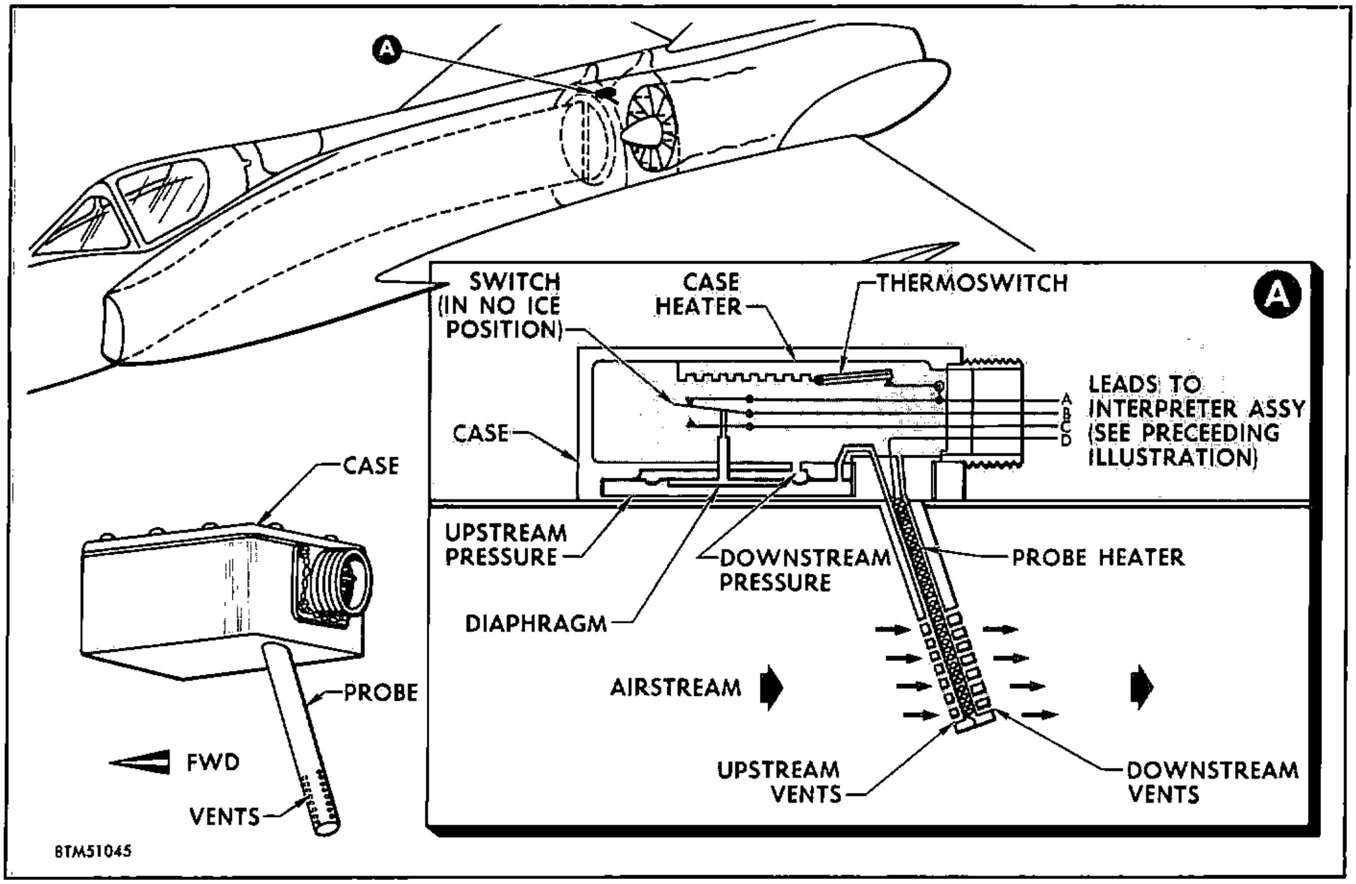

Figure 6-3. Ice-Detector Assembly

move the switch to the *no ice* position. If the anti-ice switch is at AUTO at this time, the anti-ice warning lights will illuminate. This does not necessarily indicate a malfunction. The lights should go out as soon as the airstream exceeds 40 knots. When the airstream exceeds 40 knots, the switch will be in the *no ice* position. However, if ice blocks the probe intakes, the pressure difference across the diaphragm will drop and the switch will move to the *ice* position.

The heater in the probe should melt the ice in 17 to 20 seconds, and the switch should then move to the *no ice* position. If the probe intakes should become clogged and not open after 17 to 20 seconds, the interpreter will turn off the probe heater and the detector will be inoperative until the probe openings are no longer blocked. Now notice the heater in the case. This heater keeps the case warm to prevent condensation of moisture. The thermoswitch in the heater line interrupts the current to the heater when the temperature reaches a preset point. This heater will cycle off and on continuously.

The detector assembly is a rather sensitive item and it is better not to attempt any repairs without first having complete overhaul information. If the probe becomes clogged or it otherwise becomes defective, replace the assembly with a new unit. If you must repair it, be sure to consult T.O. 1F-102A-2-6, and proceed carefully.

ANTI-ICING AND DE-FOGGING WITH HOT BLEED AIR.

Four of the anti-icing or de-fogging subsystems use hot N_2 bleed air. In figure 6-4, note that the three anti-icing subsystems are the engine anti-icing system, the intake duct lip anti-icing system, and the windshield rain-clearing and anti-icing system. The canopy de-fogging system uses partially-conditioned, but still hot N_2 air from the heat exchanger. The engine and intake duct lip systems are normally controlled by the automatic anti-ice detection and control system while the other two systems are always pilot controlled.

THE ENGINE ANTI-ICING SYSTEM.

The engine anti-icing system is an integral part of the engine and is covered in the Power Plant Installation Supplement of this series. The system consists of two bleed air transfer lines, one on each side of the engine. These lines tap N_2 compressor discharge air from the engine diffuser section and route it forward to the engine air inlet quide vanes. Note in figure 6-5, that the N_2 air flows from the guide vanes to the accessory section fairing where it exits—to be sucked back into the

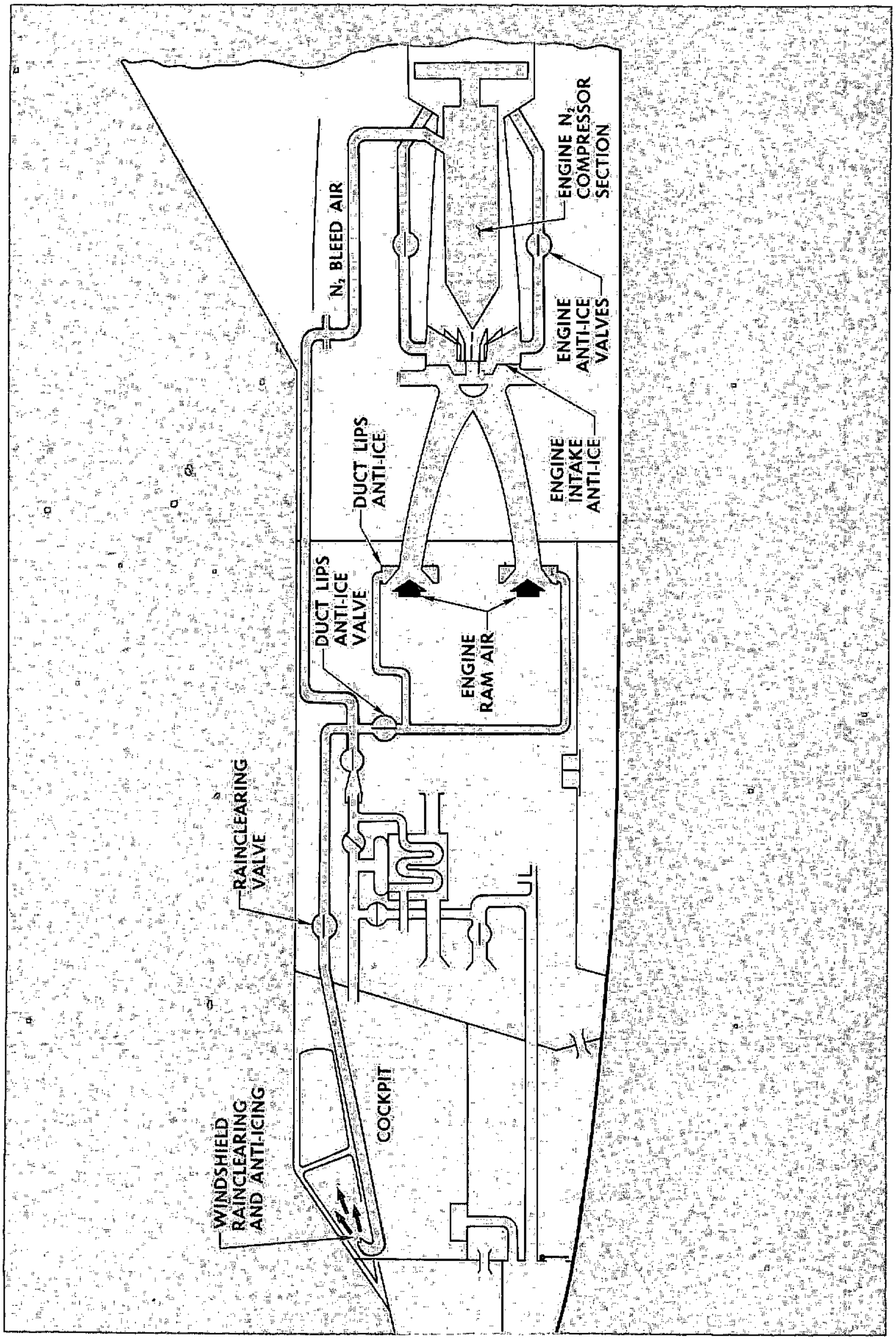

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Figure 6-4. Anti-Icing with Hot Bleed Air

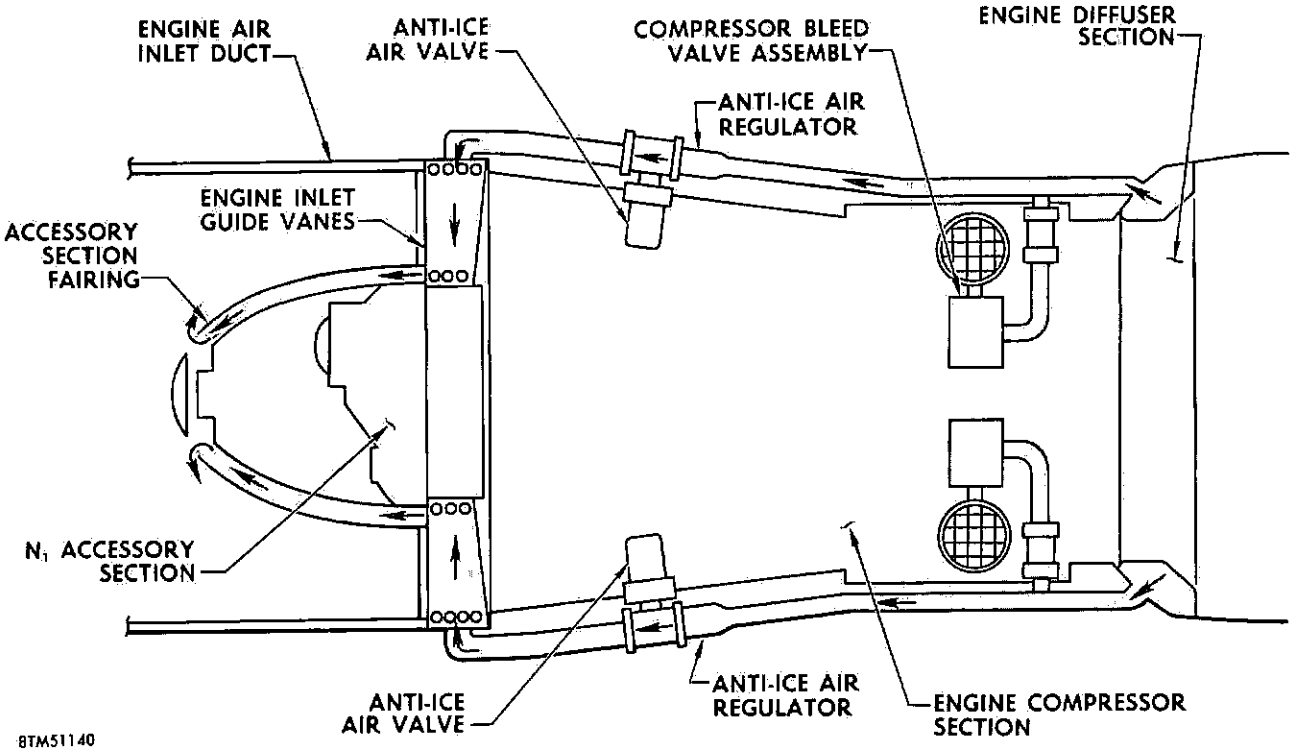


Figure 6-5. Engine Anti-Icing System Schematic

engine intake. Note also the anti-ice shutoff valve in each transfer line. Each valve is controlled by a reversible d-c electric motor. These motors were shown schematically in the electrical diagram (figure 6-2), and are controlled by the automatic ice detection and control systems. They can be manually operated by the pilot by means of the MAN ON position of the anti-ice switch. Note the air regulator in the transfer line next to each shutoff valve. These regulators automatically regulate the flow of N_2 air as the temperature of the air changes. The regulators reduce the flow of air as its temperature increases, and increases the flow as its temperature decreases.

THE INTAKE DUCT LIP ANTI-ICING SYSTEM.

The forward edges, or lips, of the engine air intake ducts are of double-skin construction. When the automatic anti-ice detection and control system detects ice, hot N_2 bleed air flows to the hollow, forward edges of the ducts to melt the ice. This air is then exhausted overboard through small slots along the inboard edges of the ducts. In figure 6-6, note that the bleed air is tapped from the N_2 bleed air distribution duct and conducted to both duct lips through steel or titanium tubes. Notice the shutoff valve and regulator in the ducting. This component is actuated by pneumatic pressure controlled by an electrical solenoid.

A pressure regulation feature of the unit regulates the downstream pressure at a constant value. Thermostatic switches around both intake duct lips will close the shutoff valve if the intake duct structure should become too hot. Hot air will flow to the duct lips when the anti-ice detection and control system detects ice, *if* the engine is running and *if* the structure around the duct lips does not overheat.

Distribution Ducting.

The two ducts sections which connect to the shutoff valve, one upstream and one downstream, are made of stainless steel and have a diameter of 1¼ inch. Notice the "fork" in the duct section below the shutoff valve. The ¾-inch tubes connect to each end of the fork. One tube leads to the duct lip on the left intake and the other leads to the right duct lip. You will probably find stainless steel tubes in some F-102A airplanes while others will have titanium tubes. All of the tubes and duct sections are insulated with a wrap-around blanket of fiberglas cloth and an insulating compound. This insulation increases the tube or duct diameters by about ½-inch. The tubes or duct sections are clamped together in the same manner as described in Chapter II for the main supply ducting.

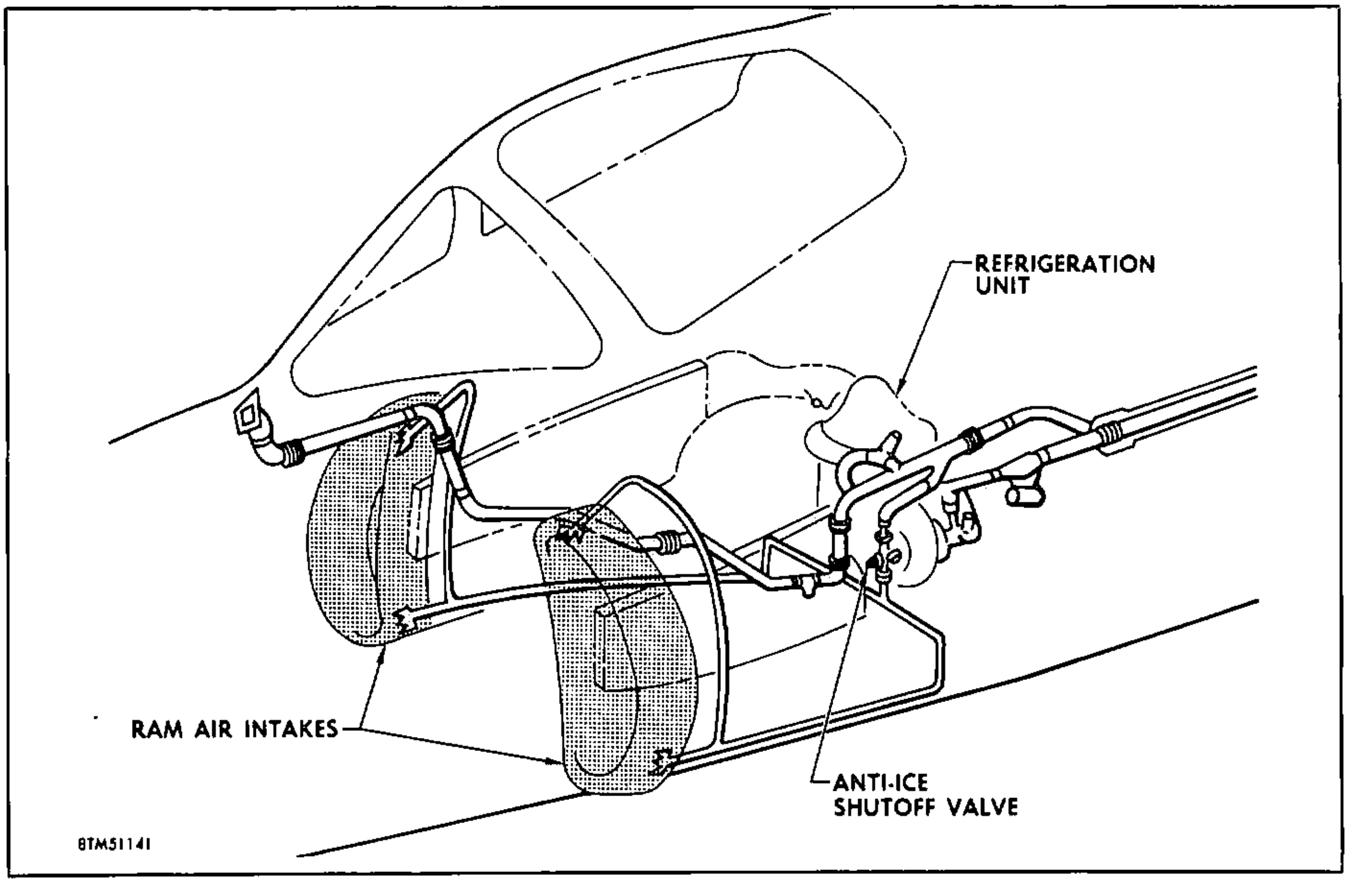


Figure 6-6. Duct Lip Anti-Icing Systems

The Shutoff Valve and Pressure Regulator.

The shutoff valve and pressure regulator is a pneumatically-operated, solenoid-controlled unit that controls the flow of hot N^2 bleed air to the intake duct lips. The unit is located in the intermediate, electronic compartment on the left side.

Notice in figure 6-7, the two pressure lines, the one connected to the valve body upstream of the butterfly shutter and the other connected downstream. One other important point to notice is the arrow on the valve body. This valve will permit air to flow in one direction only, and must be installed with the arrow pointing downstream.

How the Valve and Regulator Operates.

In figure 6-8, note that the shutter is connected by linkage to an actuator diaphragm. When the solenoid is energized, it opens a valve which allows chamber "A" on one side of the diaphragm to be pressurized with air tapped from the upstream side of the shutter. Chamber B on the other side of the diaphragm is also connected to upstream pressure through the small orifice. When chamber A pressure exceeds chamber B pressure, the shutter will open. If the pressures are equal, the spring will close the shutter.

Notice that chamber B is connected to a check valve in the pressure switch assembly. This check valve opens at a set pressure to vent air from chamber B, and to prevent its pressure from exceeding a set value. Chamber A is connected to one side of a piston which connects to the check valve. When the solenoid valve is open, upstream pressure against the piston will open the check valve and dump chamber B pressure. The relief valve connected to chamber A opens at a set pressure to regulate chamber A pressure at a set value.

When the shutter is open, downstream pressure bears against one side, the close side, of the actuator diaphragm and regulated chamber A pressure bears against the other, or open side. If downstream pressure increases above a desired point, the pressure against the close side of the diaphragm, aided by the spring, will move the diaphragm and shutter toward the close side. This means that the position of the shutter will change whenever downstream pressure changes. If the downstream pressure increases, the shutter will close slightly until the downstream pressure returns to the desired level.

Operational Cycle of Valve and Regulator.

Let us follow a complete operational cycle. As a starting point, the airplane is on the ground and the engine is off. The actuator spring keeps the diaphragm and shutter in the *closed* position. As the engine starts, N_2 bleed air fills the upstream side of the shutter. This N_2 pressure may be as high as 200 to 220 psi. Upstream air will enter chamber B through the orifice until chamber B pressure reaches a certain value. The check valve in the pressure switch assembly then opens to prevent chamber B pressure from becoming too high. As long as the solenoid remains de-energized, the pressure in chamber B will keep the shutter closed. When the anti-ice detection and control system detects ice, the solenoid energizes. (This is illustrated in figure 6-2.)

When the solenoid is energized, it opens the valve to chamber A, and upstream pressure bears against the *open* side of the actuator diaphragm. At the same time, the pressure against the piston in the pressure switch assembly causes the check valve to open wide and dump the pressure in chamber B. Since the pressure on the *open* side of the diaphragm now exceeds the pressure on the close side, the actuator opens the shutter. As the downstream pressure builds up, chamber B pressure senses downstream pressure through the wide open check valve. The downstream pressure will be almost equal to the upstream pressure initially, so the actuator diaphragm, acted upon both by the downstream pressure and the spring, moves toward the *close* position. As the shutter moves toward its closed position, downstream pressure reduces.

The actuator spring will also be under less compression and exerts less force, so the combined force trying to move the diaphragm to the *close* position lessens as the diaphragm moves. When the forces on both sides of the diaphragm are equal, the diaphragm and the shutter stop in that position. As downstream pressure varies, the shutter continually adjusts its position to maintain a constant downstream pressure.

The supply pressure varies with operating conditions, but the pressure in chamber A must be held constant as a reference pressure if the downstream pressure is to be held constant. The relief valve that is shown vented to atmosphere will bleed off sufficient air to maintain chamber A pressure at a definite value above outside static pressure. This regulated value can be changed by altering the spring tension in the check valve. The spring tension is altered by turning the adjusting screw shown at the base of the relief valve spring. This will alter the regulated downstream pressure. Removing the tension from the spring will lower the regulated pressure, while adding tension will increase it.

After a certain length of time, normally about one minute, the solenoid will de-energize and cut off upstream pressure to chamber A. The relief valve will close, and chamber A pressure will vent to atmosphere

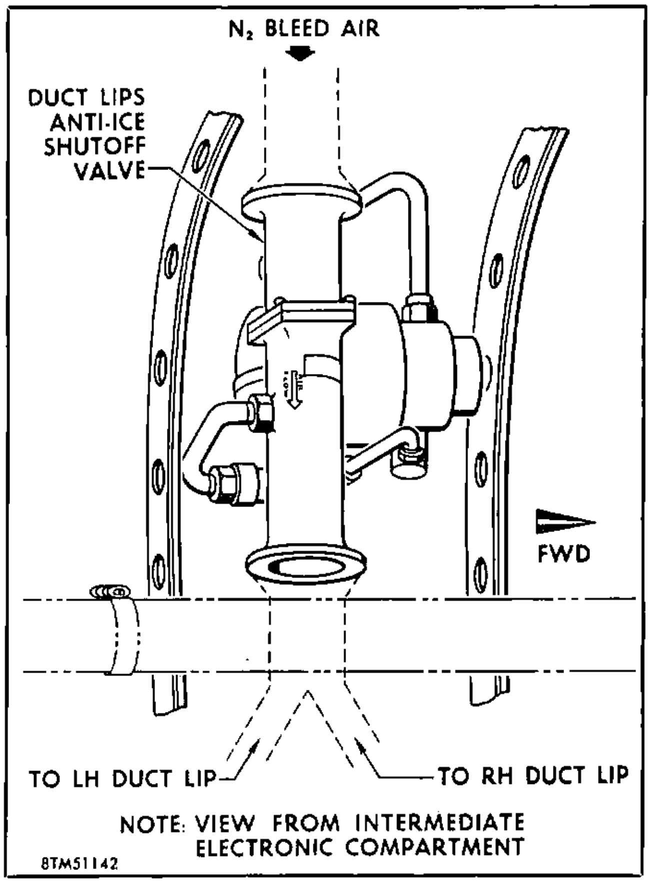


Figure 6-7. Duct Lip Anti-Icing Shutoff Valve and Pressure Regulator

(the intermediate electronic compartment) through the bleed orifice. As the pressure reduces, the piston in the pressure switch assembly allows the check valve to close, and upstream pressure builds up in chamber B and holds the shutter firmly closed. As the pressure in chamber B reaches a certain level, the check valve opens enough to keep it from rising above that level. The shutter will now remain closed until the solenoid is again energized.

The anti-ice switch has a MAN ON position that allows the pilot to override the automatic anti-ice detection and control system, and to turn on the four anti-ice subsystems. If the pilot leaves the systems on too long, the structure around the duct lips might become too hot. To prevent this, thermoswitches are situated around both intakes to sense the heat and interrupt the current to the solenoid to de-energize it.

Maintaining the Valve and Regulator.

The shutoff valve and pressure regulator has been factory-adjusted to regulate downstream pressure at a predetermined value. This value is stamped on the unit's nameplate. On most airplanes, the downstream pressure is regulated at about 26.5 psi. Always check your Maintenance Manual, T.O. 1F-102A-2-6, to determine the correct pressure and to make sure you install the correct unit when replacing parts.

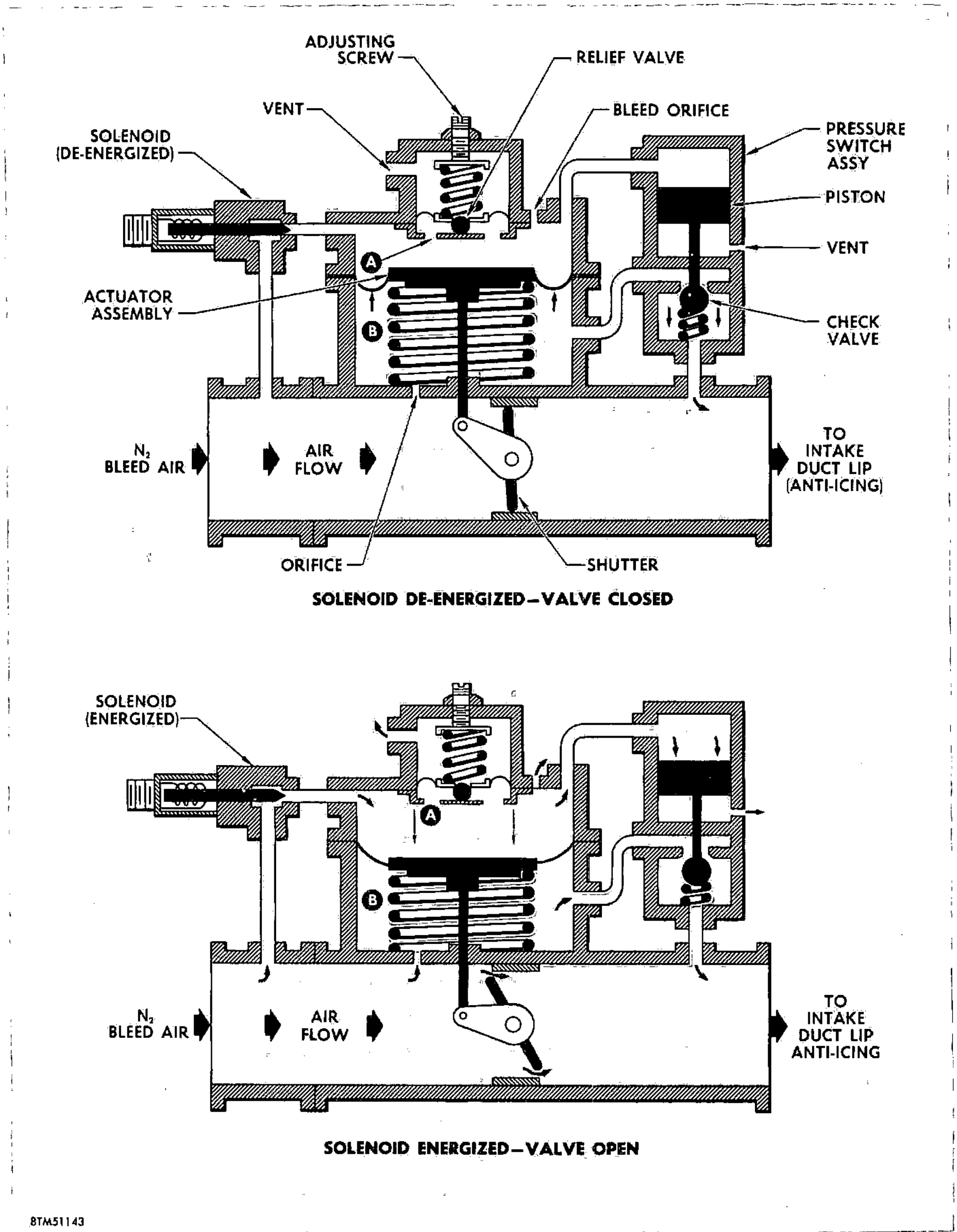


Figure 6-8. Shutoff Valve and Pressure Regulator Schematic

THE WINDSHIELD RAINCLEARING AND ANTI-ICING SYSTEM.

In operating airplanes in all kinds of weather good visibility must be maintained. If the airplane windshield is obstructed by rain, snow, or ice, the pilot's visibility will be severely handicapped. Rain encountered at low altitude may freeze on the windshield as the airplane climbs. On slower airplanes, windshield wipers similar to those on automobiles will keep the windshield clear. On the F-102A, the high speeds involved make windshield wipers unsuitable, so hot N_2 air is used.

In figure 6-9, note that the windshield rainclearing and anti-icing system taps hot N_2 air from the distribution system near the heat exchanger and conducts it through stainless steel, insulated ducting to a nozzle at the forward apex of the *left* windshield. When the pilot throws the rain clear switch on the utility switch panel to ON, the shutoff valve will open and N_2 air will flow through the ducting and exit through the nozzle. It will then flow aft along the outside of the left windshield. This hot air forms a thin sheet or layer of air, over the surface of the windshield, but it also helps remove any ice that might already be formed.

The N_2 air may have a temperature as high as 425°C (800°F) at the bleed air manifold, but there will be considerable heat loss by the time it reaches the windshield nozzle. Just before it exits from the discharge nozzle, the temperature of the bleed air will probably be between 205°C (400°F) and 260°C (500°F). A warning circuit causes the windshield overheat warning lights to illuminate if the hot air causes the windshield to become too hot. Never operate the rainclearing system on the ground unless you keep the warning lights in sight. Windshield panels can be severely damaged by overheating.

The Rainclearing and Anti-Icing Shutoff Valve.

The shutoff valve in the windshield rainclearing and anti-icing system is a pneumatically actuated, solenoid-controlled valve. This valve is installed in the intermediate electronic compartment and is almost exactly identical to the jet pump shutoff valve described in Chapter IV. The differences are very minor. However, be very careful not to interchange these valves when installing one of these units. The main difference is that the limit switch found on the jet pump valve is not on the rainclearing valve. The limit switch is necessary for the functioning of the electronic cooling warning system. Be sure that the valve is installed with the arrow on the valve body pointing downstream. The solenoid of the rainclearing valve is pilot controlled by means of the rainclear switch, while the jet pump solenoid is controlled by an automatic circuit. The internal construction and operation of the two units are identical.

The position indicator will indicate the position of the valve shutter at any time. The shutter should always be closed if the engine is not running or the rainclear switch is OFF. The windshield overheat warning system can also indicate trouble. If the warning light should stay on for some time after the valve is closed, check the rainclearing system and also the warning system. If the valve does not operate properly, replace it as described in T.O. 1F-102A-2-6. If the valve should fail to open on a signal from the rainclear switch, make sure that there is power to the 28-volt d-c bus and that the rainclear circuit breaker (located on the aft left cockpit circuit breaker panel) is energized before replacing the valve.

Windshield Overheat Warning System.

As previously stated, there is a good possibility of damage to the windshield from too much heat. The windshield is of sandwich type construction, with a layer of plastic between two layers of glass. The glass can withstand considerable heat, but the plastic cannot. To prevent damage to the windshield, the pilot must know when it is getting too hot. Then he can shut off the rainclearing air until the windshield again cools. The next to the last light on the warning indicator panel, placarded WINDSHIELD OVHT, tells him of an overheat condition.

The windshield air overheat warning system consists of a *thermistor,* a *bridge amplifier,* and the usual warning light circuit. The thermistor is a resistance bulb which is sensitive only at its tip. It is mounted at the forward apex of the left windshield, and can be removed from inside the cockpit. The bridge amplifier (sometimes called the thermistor indicator) is a small unit using transistors instead of vacuum tubes. An adjusting screw is provided for temperature calibration; aside from that, no change or repairs are permitted on this unit. The amplifier, thermistor, and other components of the warning system are shown in figure 6-10.

The thermistor is the component of the overheat warning system that senses windshield temperature. The tip of the thermistor is spring-loaded to maintain a good contact with the windshield glass. Whereas most temperature bulbs increase their resistance when temperature increases, this one does just the opposite. When the windshield temperature increases, the thermistor resistance drops. For that reason it is sometimes called a negative temperature bulb.

As you will note in the overheat schematic (figure 6-10), the source of power to this warning system is the airplane's 28-volt d-c essential bus. Current flows through the circuit breaker to contact C of the amplifier. The amplifier reduces the voltage to about 5 volts in the thermistor circuit. Since the resistance of the bulb decreases when temperature increases, the current flow

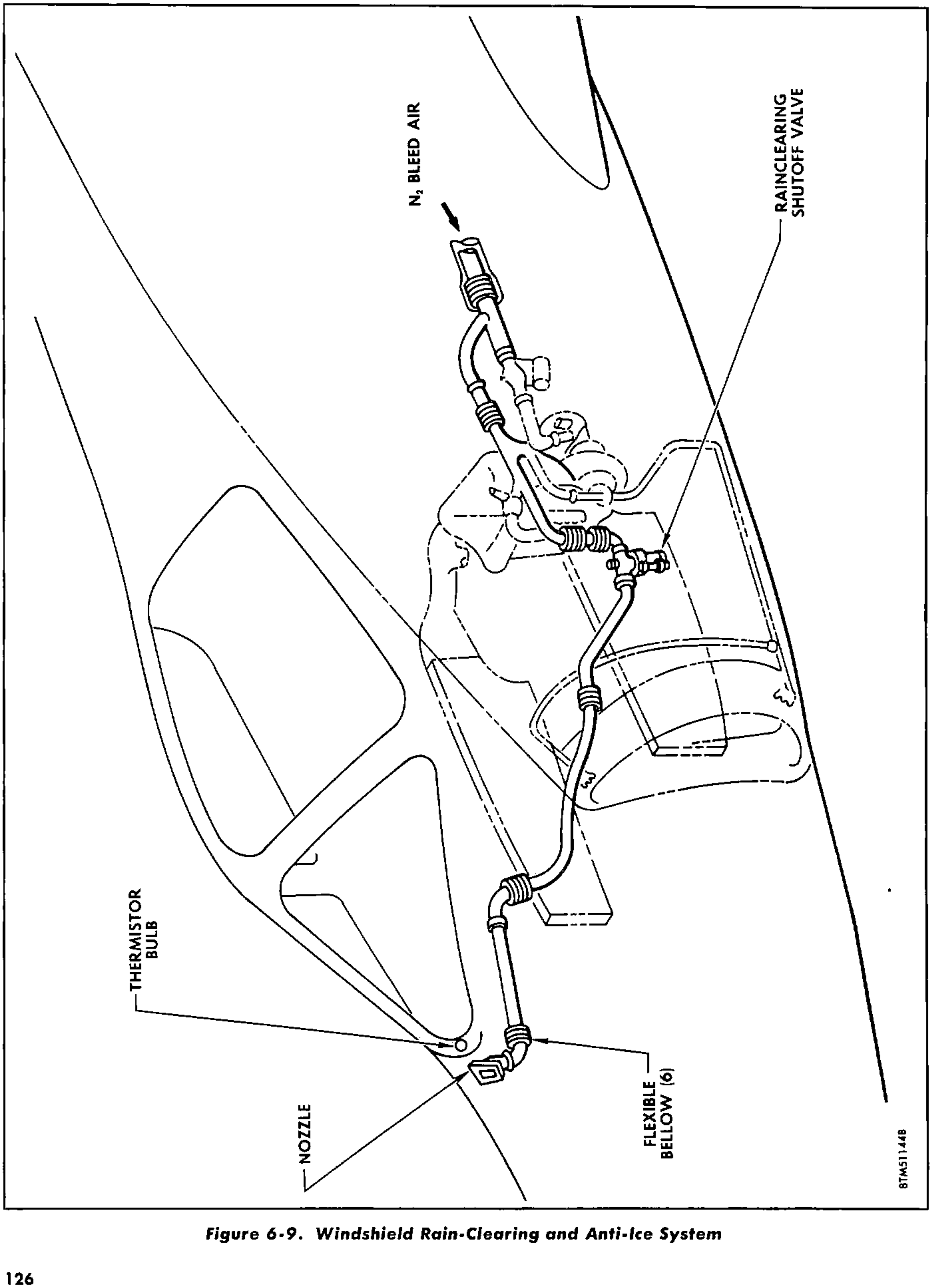


Figure 6-9. Windshield Rain-Clearing and Anti-Ice System

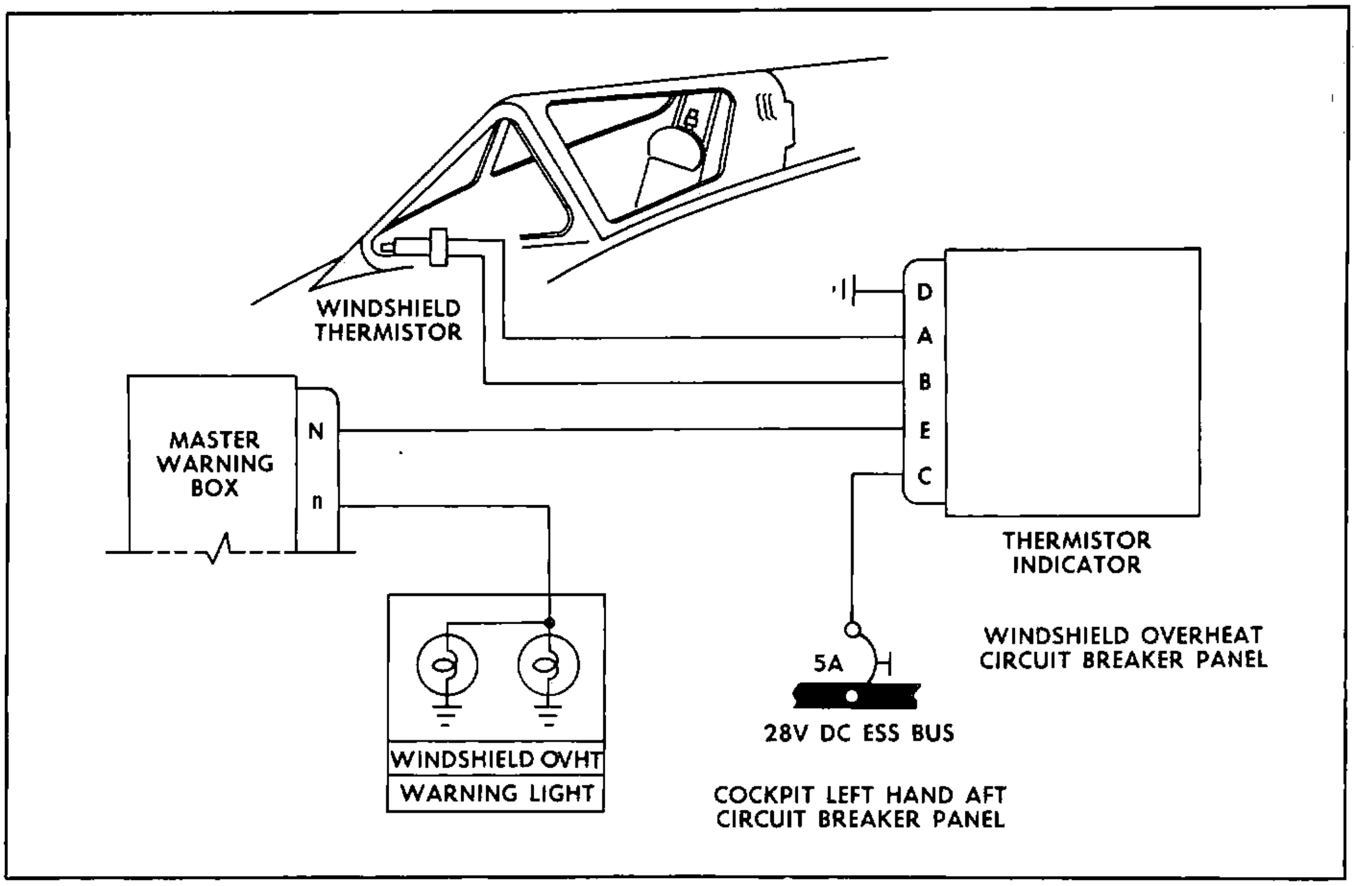


Figure 6-10. Windshield Overheat Warning Circuit Schematic

through the thermistor circuit increases. This acts to unbalance the bridge circuit in the amplifier, thus producing an electrical signal proportional to the temperature of the bulb. After amplification, the signal goes to a relay. When the current flow becomes great enough due to low resistance of the thermistor bulb, the relay actuates, letting current flow through terminal E to the master warning box and the warning light. The light will illuminate and remain on until the windshield temperature decreases below a certain level.

The windshield temperature at which the windshield air overheat warning light comes on is about 200°F. You should always consult your Maintenance Manual, T.O. 1F-102A-2-6, for calibration data.

THE CANOPY DE-FOGGING SYSTEM.

When the air in the cockpit contains moisture and is much warmer than outside air, there is a tendency for the moisture to condense on the inside of the glass areas.

The canopy de-fogging system uses a mixture of partially-conditioned but still hot N_2 air from the heat exchanger and cockpit air to keep the canopy clear of fog. Note in figure 6-11, partially-conditioned air is tapped from the heat exchanger and conducted to the de-fog ducting by means of ¾-inch steel tubing. Notice the shutoff valve and the pressure regulator in the tubing. The valve is controlled by the canopy de-fog switch on the utility switch panel. When it is open, N_2 air, regulated at about 11 psi over cockpit pressure, enters the de-fog ducting through a sonic nozzle similar to the jet pump nozzles described in Chapter IV.

This jet of air acts as a jet pump. It creates a negative pressure behind it, and cockpit air is sucked into the de-fog ducting through the air intake shown on the canted bulkhead in the illustration. The resulting mixture of cockpit air and partially-conditioned N_2 air flows through the de-fog ducting and exits from corrugated outlets along the forward edges of the canopy. This air flows along the inner surfaces of the canopy and evaporates the moisture to improve visibility. This addition of partially-conditioned N_2 air may add some heat to the cockpit air, but the temperature control system will compensate for it almost immediately.

The Canopy De-fog Shutoff Valve.

The canopy de-fog shutoff valve is a motor-operated, butterfly-shutter type with which you should now be familiar. The motor is a reversible-field 28-volt d-c unit.

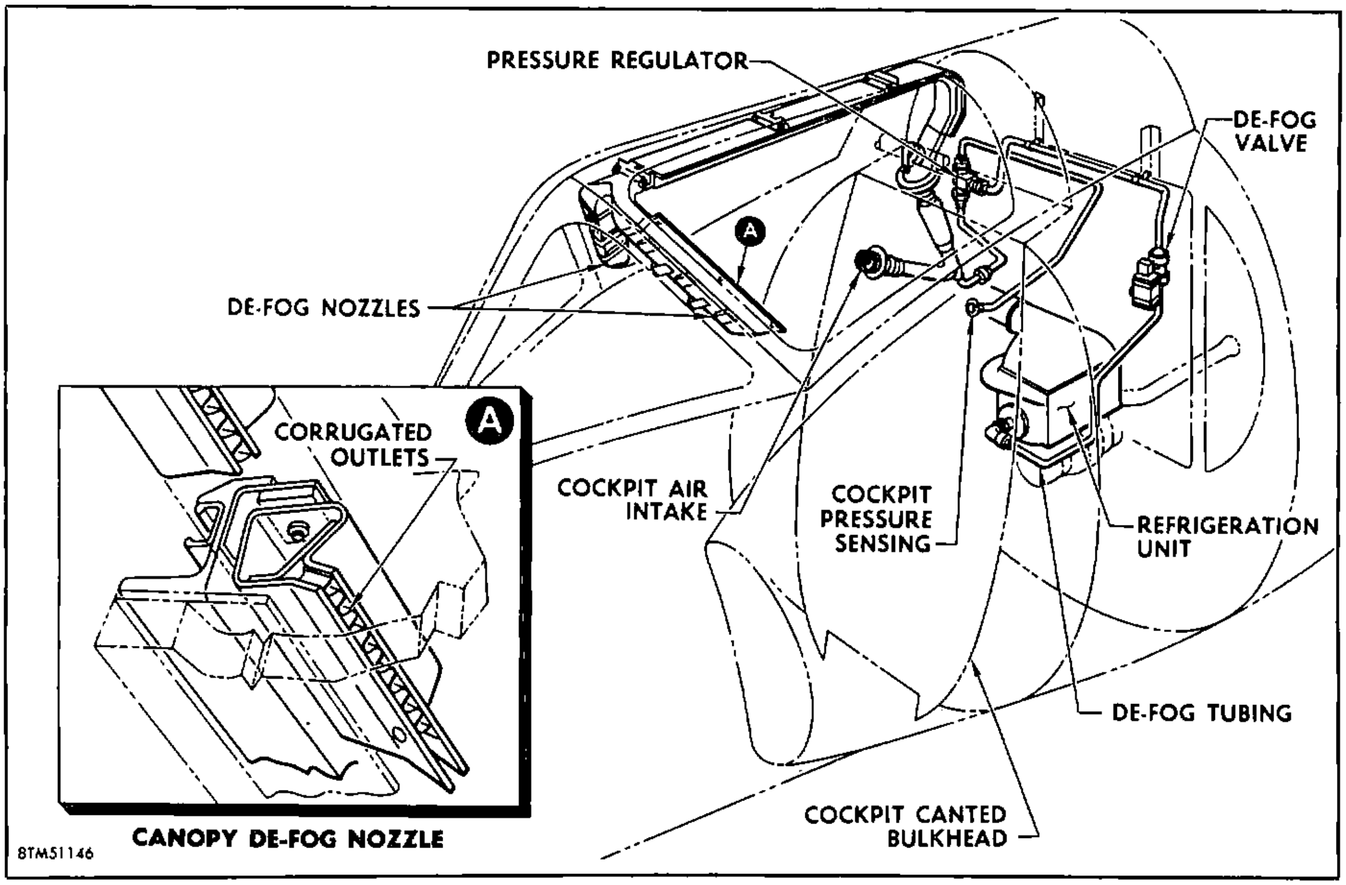


Figure 6-11. Canopy De-Fogging System

When the canopy de-fog switch is at OFF, the *close* side of the motor field winding is connected to power. This prevents the shutoff valve from opening accidentally. When the switch is turned ON, the *open* field winding is energized and the valve shutter opens. The only really distinctive aspect of this unit is the small size of the valve housing. The internal diameter is less than one inch. Due to its distinctive size, there is little danger of installing the wrong valve if the old one should require replacement. If the valve does not respond to movements of the canopy de-fog switch, make sure that the 28-volt d-c bus is energized and that the canopy de-fog circuit breaker, on the aft cockpit circuit breaker panel, is closed before placing the blame on the valve.

The Canopy De-fog Pressure Regulator.

The pressure regulator in the de-fogging system tubing regulates the downstream pressure to a value which does not exceed cockpit air pressure by more than 11 psi. Notice on figure 6-11 that the regulator is connected to the cockpit by a ¼-inch pressure sensing line. The pressure regulator compares cockpit pressure with the pressure in the downstream portion of the regulator, and restricts the flow to keep the downstream pressure from becoming greater than desired.

Note in figure 6-12 that the partially-conditioned N_2 air enters one port, passes through openings in a piston assembly, and exits through the second port. Also note the spring on one side of the piston, which holds the piston assembly in the wide-open position when there is no air pressure. Note also that cockpit pressure bears against one side of the piston assembly, and that downstream pressure bears against the other side. As long as downstream pressure does not exceed cockpit pressure by more than 11 psi, the regulator will stay wide open. As downstream pressure exceeds 11 psi, the piston assembly will be forced toward the closed position until the downstream pressure reduces to around 11 psi. This operation is entirely automatic and should give little trouble. Be sure to check the cockpit pressure sensing connections for leaks if the regulator does not operate properly. Always replace a defective regulator; do not attempt to repair it.

ELECTRICAL ANTI-ICING AND DE-FOGGING SYSTEMS.

Four electrical systems anti-ice or de-fog different parts of the airplane. None of these systems are directly connected to the low-pressure pneumatic system. However,

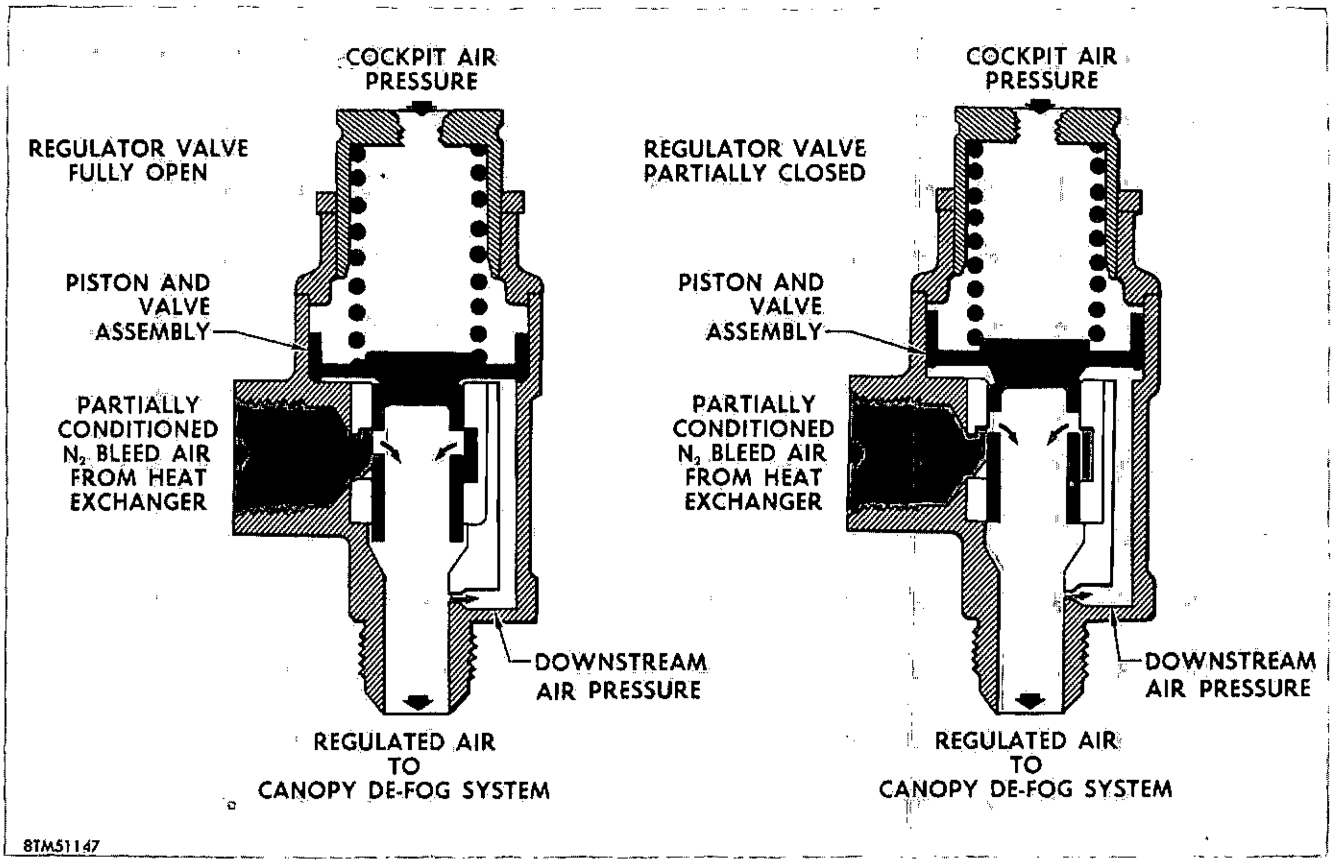


Figure 6-12. Canopy De-Fog Pressure Regulator

they are included here so that you may understand how the entire anti-icing system works. These four systems are shown in figure 6-13. Notice that only the Q-intake anti-icing system is controlled by the automatic anti-ice detection and control system. The other three systems are pilot-controlled by means of switches in the cockpit. All four systems can be operated on the ground, whether or not the engine is running. The only exception is the Q-intake system which, on some early airplanes, is shut off when the right main landing gear door is open.

ANTI-ICING OF THE Q-INTAKE.

In Chapter V of this manual, we discussed the flight control artificial-feel systems. As you will recall, both the rudder and the elevator feel systems regulate the artificial feel by means of air pressure regulators or loaders. The elevator Q-system loader senses ram air pressure from both of the two "Q" intakes on the leading edge of the fin, while the rudder feel system regulator senses ram air pressure from the small Q-intake, called the rudder Q-intake, only. The air speed compensator in the pitch and yaw damper system is also connected to the rudder Q-intake. These three flight control systems, the elevator and rudder artificial feel systems and the pitch and yaw damper system, are all very important to the pilot in controlling the airplane. If the Q-intakes are clogged, these systems will not function properly. The Q-intake anti-icing system prevents the formation of ice which would often clog the intakes if allowed to form.

As we have noted, the two Q-intakes are mounted on the leading edge of the fin. In figure 6-14 you can see that the elevator Q-intake assembly is much larger than the rudder Q-intake. Both of these assemblies have electrical heating elements around the intake tubes. In the electrical schematic (figure 6-15) note that the elevator Q-intake tube has two heating elements. The rudder tube has only one element. The element in the rudder tube and the forward element of the elevator tube is powered by the 28-volt d-c non-essential bus. In case of power failure during flight, the non-essential bus is disconnected and the essential bus is powered by the battery.

Note in the schematic that the elements are connected to power through switches controlled by the Q-intake heater relay. The relay is controlled by the automatic ice detection and control system which was described earlier in this chapter. The heater elements "heat-up" whenever the relay is energized. Remember that the relay can be energized any time the pilot wishes by

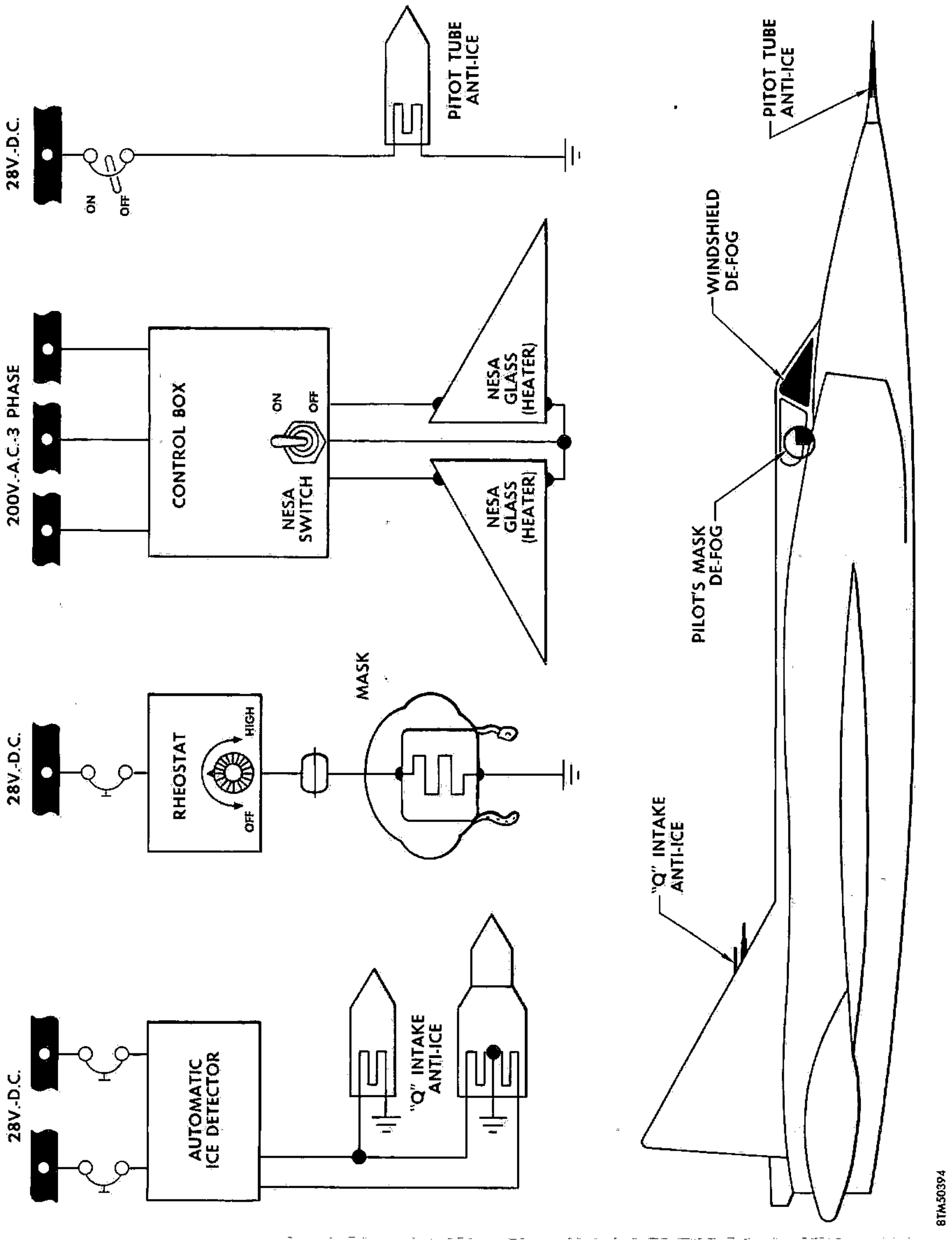


Figure 6-13. Electrical Anti-Icing and De-Fogging Systems

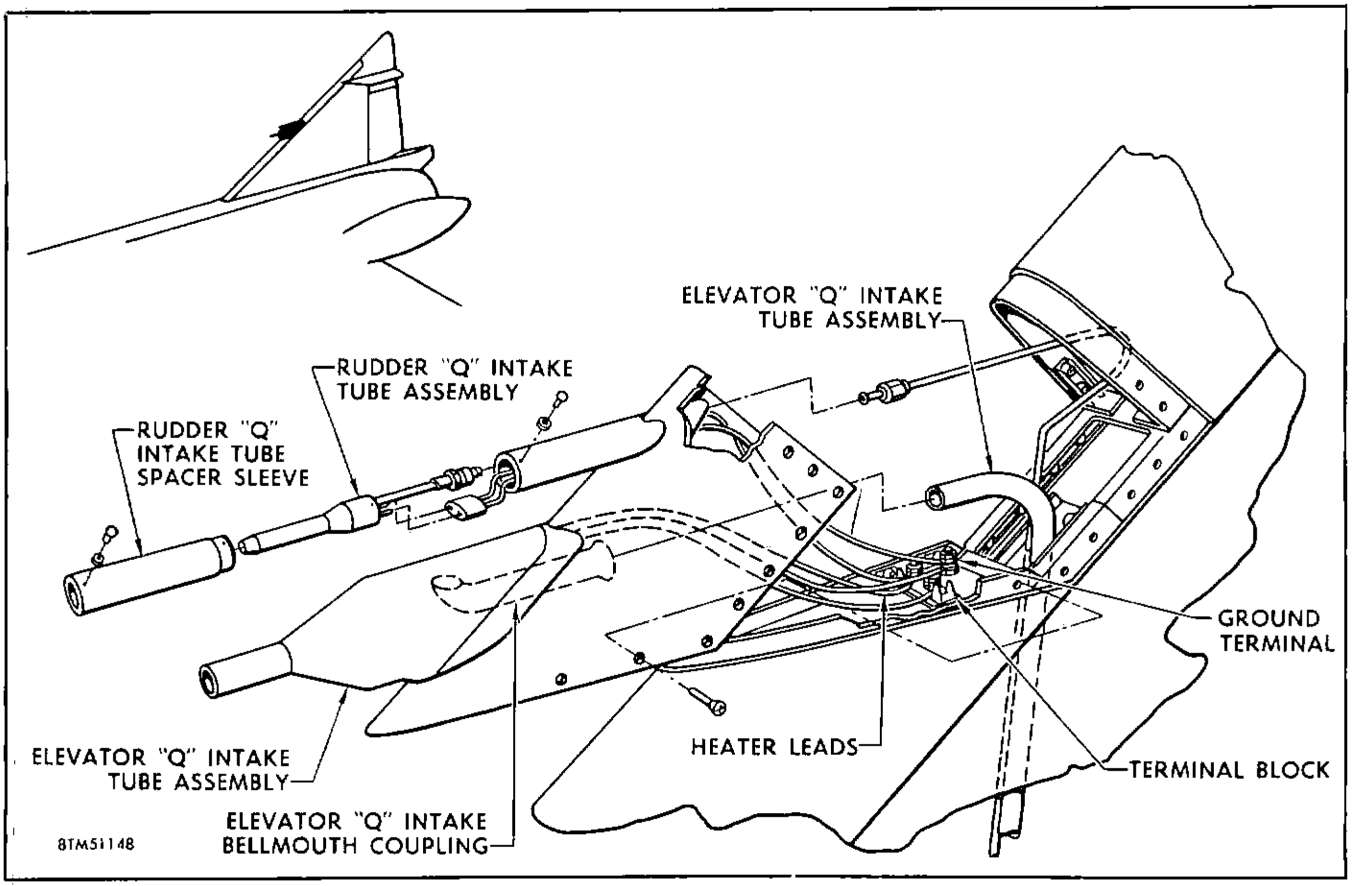


Figure 6-14. Q-Intake Anti-Icing

placing the anti-ice switch at MAN ON. On some early airplanes, the relay control could not be energized when the right forward main landing gear door *closed* switch was in the *open* position. This means that the Q-intake anti-ice system in these early airplanes cannot be energized while the airplane is on the ground.

In later airplanes, the system can be energized at any time. Remember that the heaters become very hot when there is no airstream to remove heat, so be careful not to grasp them when making a ground test or you may be badly burned. Your maintenance manual, T.O. 1F-102A-2-6, describes tests to be conducted on this system. Perform these tests carefully, and replace any defective components. If the elements fail to heat when tested, check the Q-intake anti-ice circuit breakers in the main well before replacing any component.

PITOT-STATIC TUBE ANTI-ICING SYSTEM.

The pitot-static tube is installed on the nose of the airplane. Note in figure 6-16, that there is an air pressure inlet at the very tip of the tube. This inlet picks up ram air pressure which is also called *impact* or *pitot* pressure. Smaller inlets for picking up *static* air pressure are found farther back on the tube. Both pitot and static pressure are fed to various important flight instruments, such as the altimeter and the air speed indicator. Since the inlets are small, they could easily become clogged with ice, resulting in failure of the instrument to read correctly.

The instrument that the pilot observes most closely is the airspeed indicator. When he notices that its readings do not seem to be correct, he places the pitot heat switch, on the utility switch panel, at ON. An electrical heating element in the pitot-static tube will then be connected to the 28-volt d-c essential bus, the ice will melt, and the inlets will be freed of the ice obstruction.

There is an opening on the underside of the tube which allows the water to drain overboard. The heating element becomes extremely hot and can burn itself out if the pitot-static and anti-icing system is operated too long on the ground. Be careful that you do not burn yourself when testing the system. If the heater element does not warm up when the control switch is turned on, replace the element as described in your Maintenance Manual, T.O. 1F-102A-2-6.

The pitot heat switch is the circuit breaker-type and there is no separate circuit breaker; the switch will move to OFF if the current should exceed 10 amps. If it is absolutely necessary to have the pitot-static anti-icing system on, the switch can be manually held in the

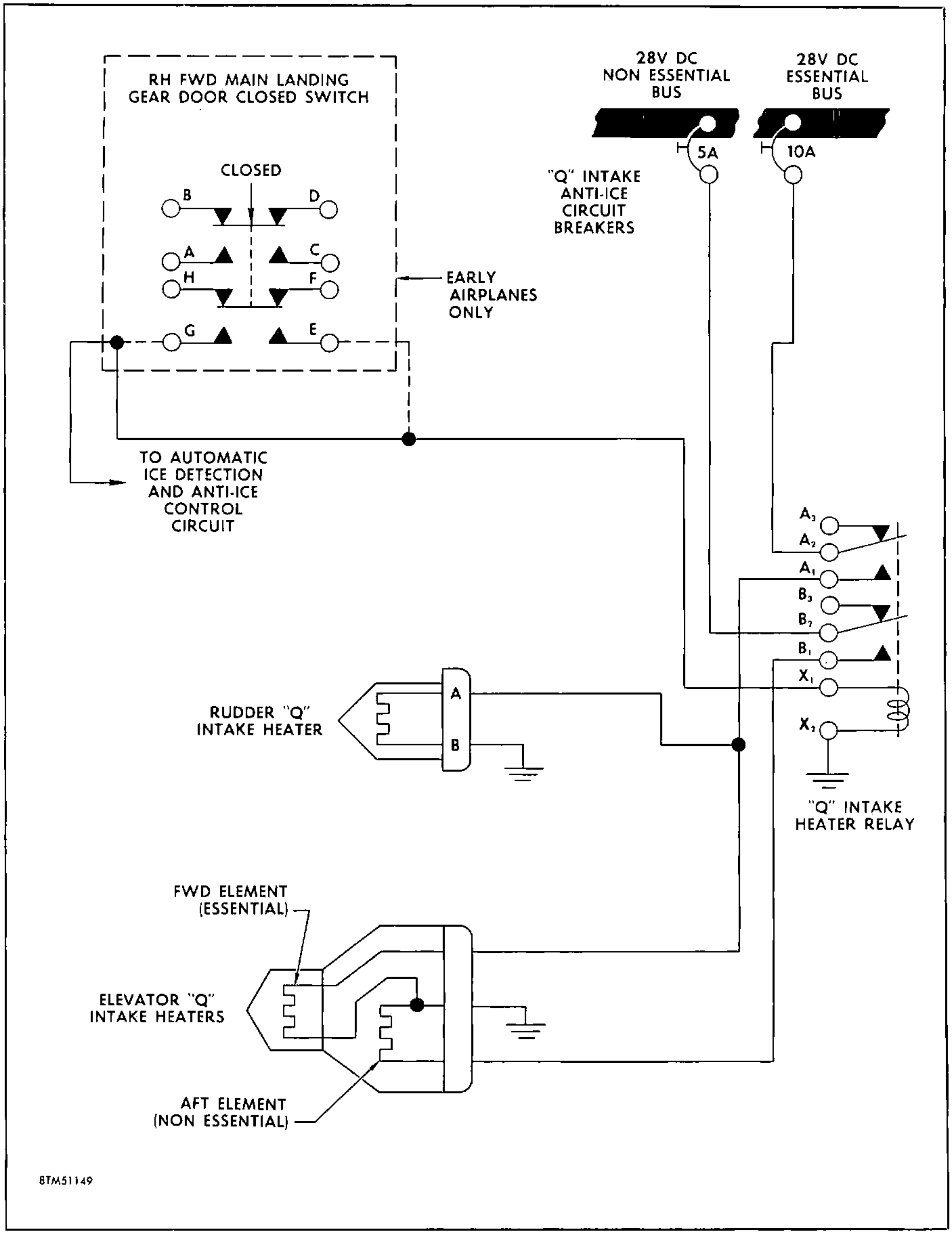


Figure 6-15. Q-Intake Anti-Icing System Electrical Schematic

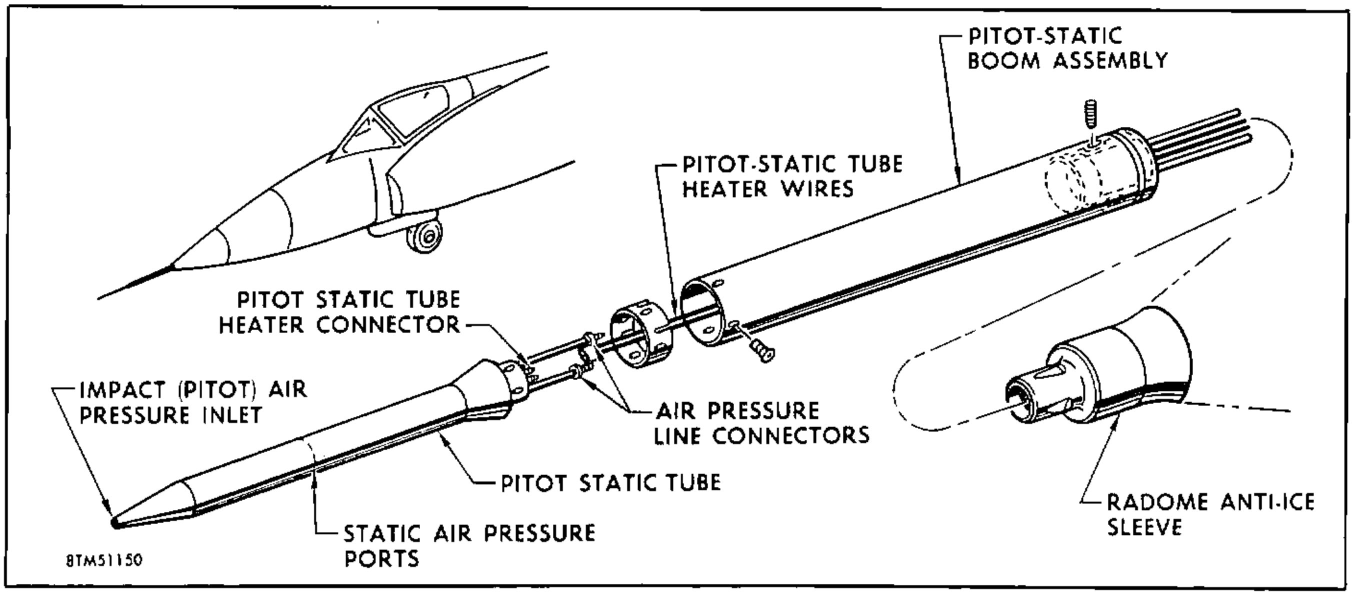


Figure 6-16. Pitot-Static Tube

ON position. This will probably result in a burned out element which must then be replaced as soon as the airplane returns to its base.

THE WINDSHIELD DE-FOGGING SYSTEM.

When we discussed the canopy de-fogging system earlier in this chapter we learned that condensation of moisture on the inside surfaces of the windshield and canopy "fogs" up the glass panels and interferes with vision. The canopy is de-fogged by means of warm air which is blown across the canopy glass to evaporate this moisture. The two glass windshield panels are de-fogged electrically by means of heating elements built into the panels. This glass, with built-in heating elements, is called *Nesa* glass. The heater elements are automatically controlled by an automatic control system in the Nesa control box. This control system is energized when the pilot places the Nesa switch on the utility switch panel at NORMAL.

Note in figure 6-17, each of the windshield panels has its own heating element and temperature-sensing element. The sensing elements are made of a material whose resistance increases greatly as its temperature changes. This change of resistance is *sensed* by the circuits in the Nesa control box. This control box will disconnect power to the heating elements as the windshield temperature reaches a certain level. The control box can be adjusted to maintain the windshield temperature constant at any level from 24°C (75°F) to 71°C (160°F) and within a range of 3°C (5°F). Note that the windshield de-fogging system is powered by the 200 volt, 400 cycle, non-essential a-c bus. This means that the windshield de-fogging system will not operate if the normal a-c power system shuts down.

The windshield overheat warning system, described earlier in this chapter, is connected to the left windshield only. Ordinarily if the warning lights should come on, it would indicate that the windshield is too hot due to operation of the rainclearing and anti-icing system. However, if the lights should come on when the rainclearing system is not operating, it would indicate that the Nesa control box is defective and that the windshield de-fogging system should be shut down.

The Nesa control box is the heart of this system. This unit contains rectifiers, transformers, capacitors (condensers), and relays. It is beyond the scope of this manual to describe it here. There is an Air Force Technical Order published concerning it (T.O. 1F-102A-2-6) which completely describes its maintenance, testing, and calibration procedures. Do not attempt to adjust or repair this unit without detailed instructions. If the control box seems to be defective, simply replace it. Never attempt to adjust a unit of this type, unless you have received special training.

DE-FOGGING THE PILOT'S OXYGEN MASK.

The pilot's oxygen mask can "fog-up" for the same reason that the windshield and the canopy fog-up. As we learned earlier in this chapter, this is due to condensation of moisture on the surfaces of the glass. The oxygen mask is de-fogged in the same manner as the windshield. The glass in the mask contains built-in heater elements that evaporate the condensation. The electrical current to the mask is controlled manually by the pilot by means of a control rheostat mounted on the pilot's left-hand console, and adjacent to the G-suit pressurization control. The pilot's personal electrical lead from the mask is connected to a quick-disconnect

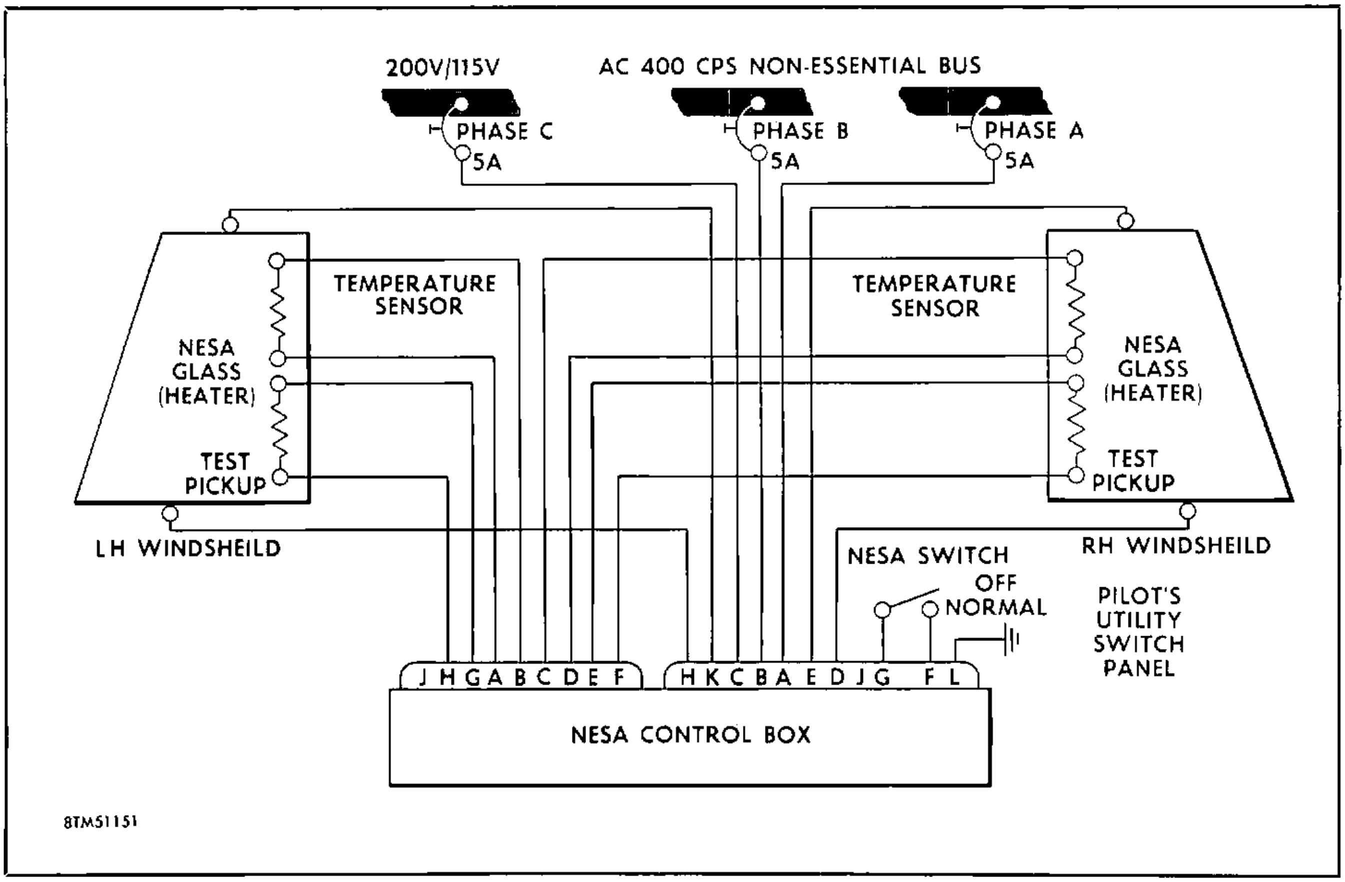


Figure 6-17. Windshield De-Fogging System Schematic

fitting on the left side of his seat. This electrical disconnect pulls apart during a seat ejection in the same manner as the G-suit pressure line and the oxygen line.

If the pilot desires more heat to the mask, he rotates the control knob clockwise. This reduces the control resistance and allows more current to flow to the mask. There is a circuit breaker in this line which you will find on the aft cockpit circuit breaker panel on the left-hand side. This system should give you little trouble. The heating element in the mask itself is the item most likely to burn out, and that is an item of personal equipment.

THE RADOME ANTI-ICING SYSTEM.

The radome at the nose of the airplane is made of a plastic material that will not interfere with the reception of radar signals by the antenna scanner enclosed in the radome. Any ice forming on this radome will interfere with radar reception enough to completely ruin the effectiveness of the radar system. The radome anti-icing system prevents the formation of ice by forcing glycol fluid overboard through a porous metal ring at the tip of the radome. The airstream distributes the fluid evenly over the surface of the radome to form an anti-ice film. This system is controlled by the automatic anti-ice detection and control system, but cannot be operated when the main landing gear doors are open.

As we learned in Chapter V, the glycol fluid is contained in a tank or reservoir which is pressurized by partially-conditioned N_2 air from the heat exchanger. Note in figure 6-18 that the N_2 air is connected to the radome anti-icing system through a filtered orifice and a check valve. You can also see the relief valve and the safety valve in the line. The relief valve holds the pressure constant at about 5 psi over atmospheric pressure. The safety valve opens at 16 psi, if the relief valve should fail. The shutoff valve in the fluid line is solenoid-actuated and opens when the solenoid is energized. The solenoid is energized when the anti-ice detection and control system detects ice, or when the anti-ice switch is placed at MAN ON.

The solenoid cannot be energized when the main landing gear doors are open. When the shutoff valve is closed, there will be a continuous flow of pressurized air through the tubing forward of the valve. This air passes through the forward filtered orifice and overboard through the porous metal ring. This continuous flow keeps the tubes clear and the ring free of ice when the valve is closed. The drain line you see connected to the tank vents spilled fluid overboard.

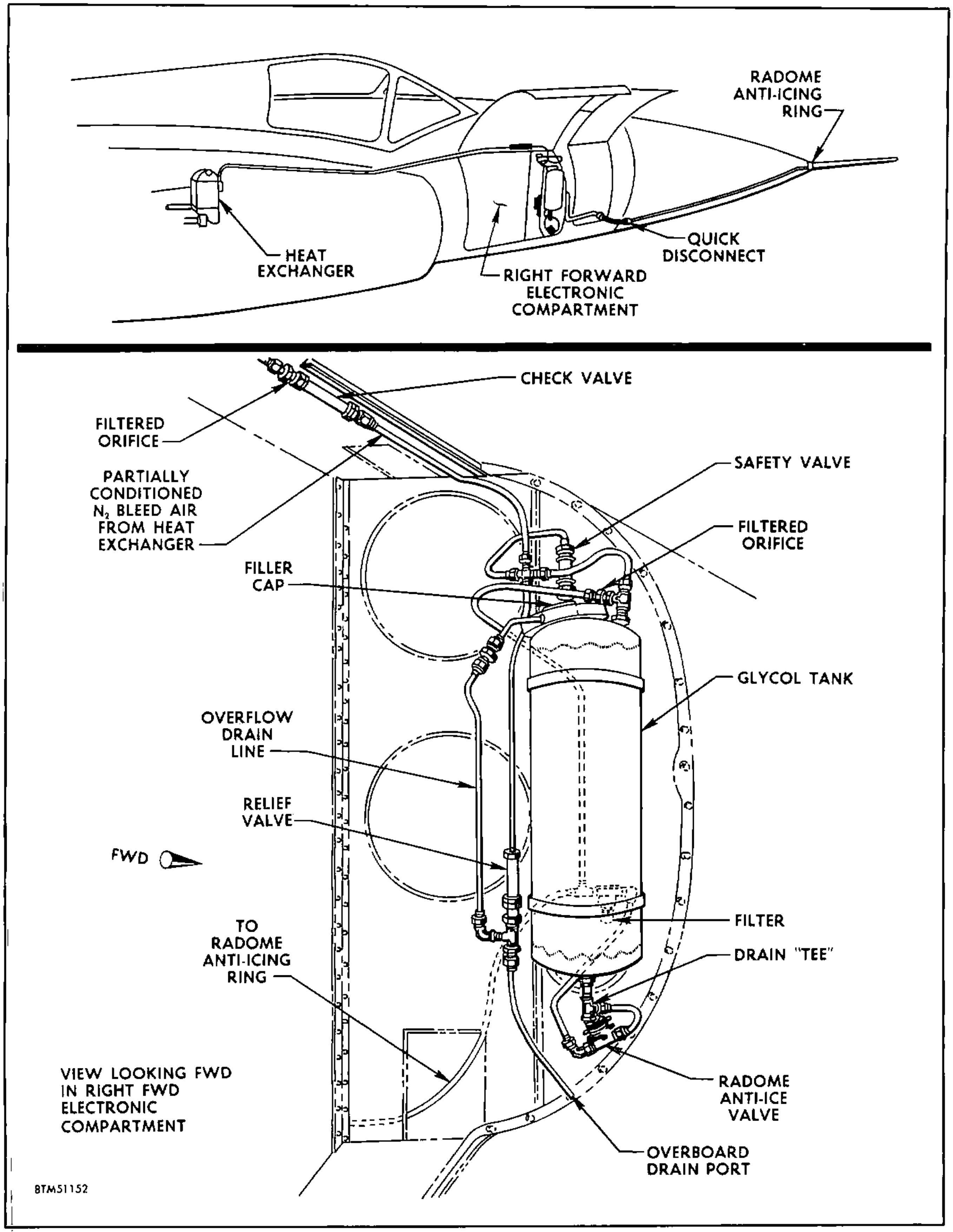


Figure 6-18. Radome Anti-Icing System

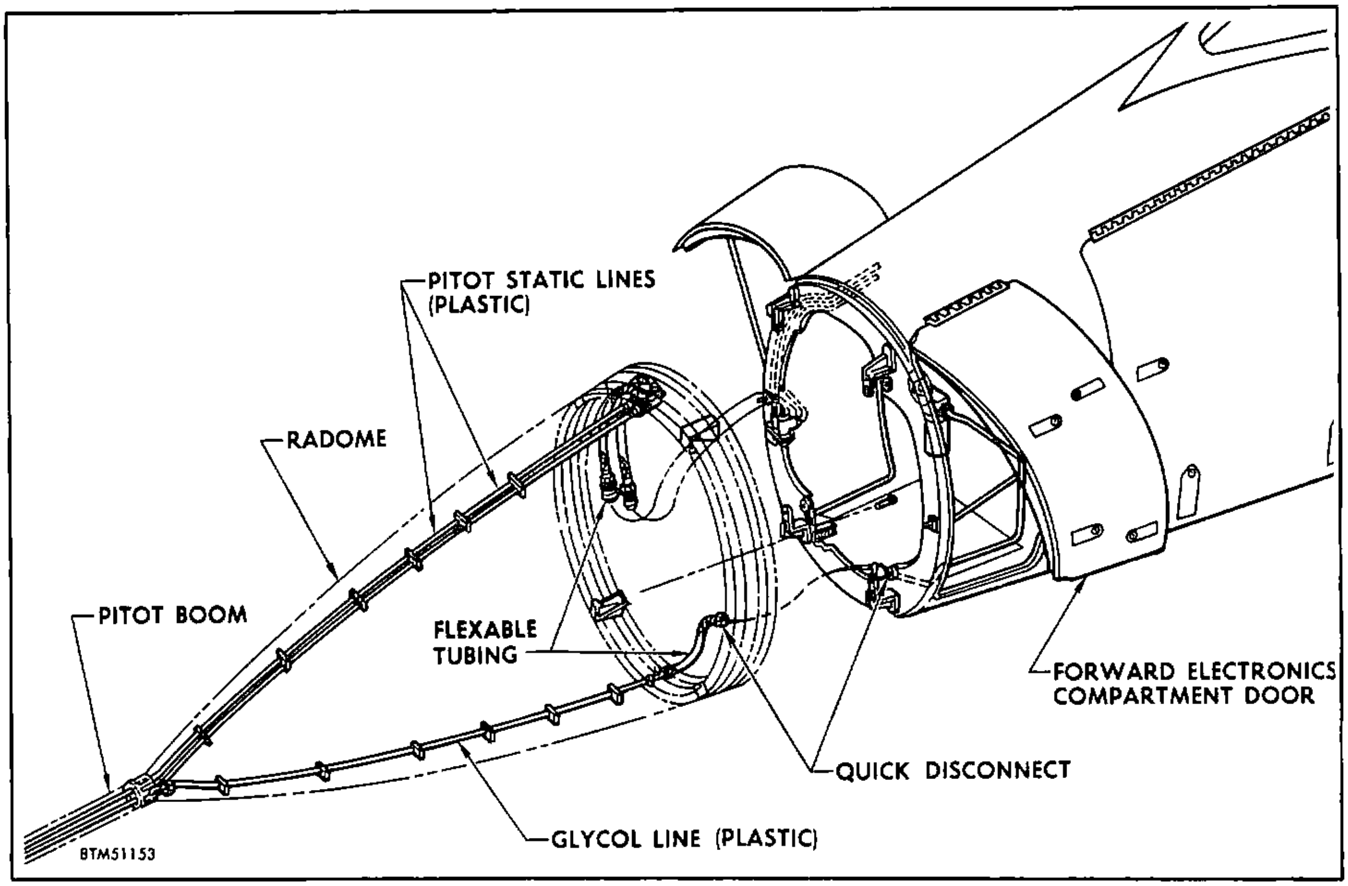


Figure 6-19. Non-Metallic Tubing in Radome

The air pressure lines, the drain line, and the fluid line aft of the radome itself are all made of ¼-inch aluminum tubing. The different tube sections are connected with standard AN pressure couplings. The fluid line in the radome itself is made of ⅜-inch plastic tubing. The pitot-static lines shown in figure 6-19 are also made of a non-metallic material to prevent interference with radar reception. A short length of ⅜-inch flexible hose connects the plastic tubing to the aluminum tubing.

THE GLYCOL TANK.

All of the glycol fluid for the radome anti-icing system is stored in a welded aluminum tank. This tank is pressurized to a regulated 5-psi air pressure, and contains integral baffles to help prevent "sloshing." In figure 6-20, note that the tank is installed in the right, forward side of the nose and is accessible through the right hand aft door on the forward electronic compartment. Note the built-in relief valve in the filler cap. This valve will open to relieve air pressure at a maximum of about 16 psi. Note also the small button in the center of the valve. Before opening the cap, always depress this button to allow the air pressure to bleed off. Hold a cloth around the cap to prevent being sprayed with the fluid. Any spilled fluid will then be caught in the *scupper* and drained overboard through the drain line. The air pressure line connects to the top of the tank. The input pressure is normally regulated by the relief valve in the line to about 5 psi. The *Tee* fitting that you can see attached to the bottom of the tank has two functions. The fluid line connects one end of the *Tee* to the anti-icing shutoff valve. The bottom end of the *Tee* is normally capped. When you wish to drain the tank, remove this cap. The tank is easier to drain if you do it while the tank is pressurized. An outside source of compressed air should be used. Do not use a pressure higher than about 10 psi for pressure draining. The tank should give you little trouble. Check all connections from time to time for evidence of leaks. The tank contains two gallons of fluid which is enough for about 15 minutes of continuous operation.

THE RADOME ANTI-ICING SHUTOFF VALVE.

The shutoff valve in the radome anti-icing glycol line is a solenoid-actuated valve of a type normally found in fuel systems or systems that use other hydrocarbon fluids. In figure 6-21, note that the valve is a simple type that is *closed* when it energizes. The valve is the *fail-safe* type since the solenoid return spring will keep it closed when power to the solenoid is removed. The valve solenoid is controlled by the automatic anti-icing detection and control system, and will normally be energized when ice is detected. The circuit to the solenoid is interrupted by the right main landing gear door

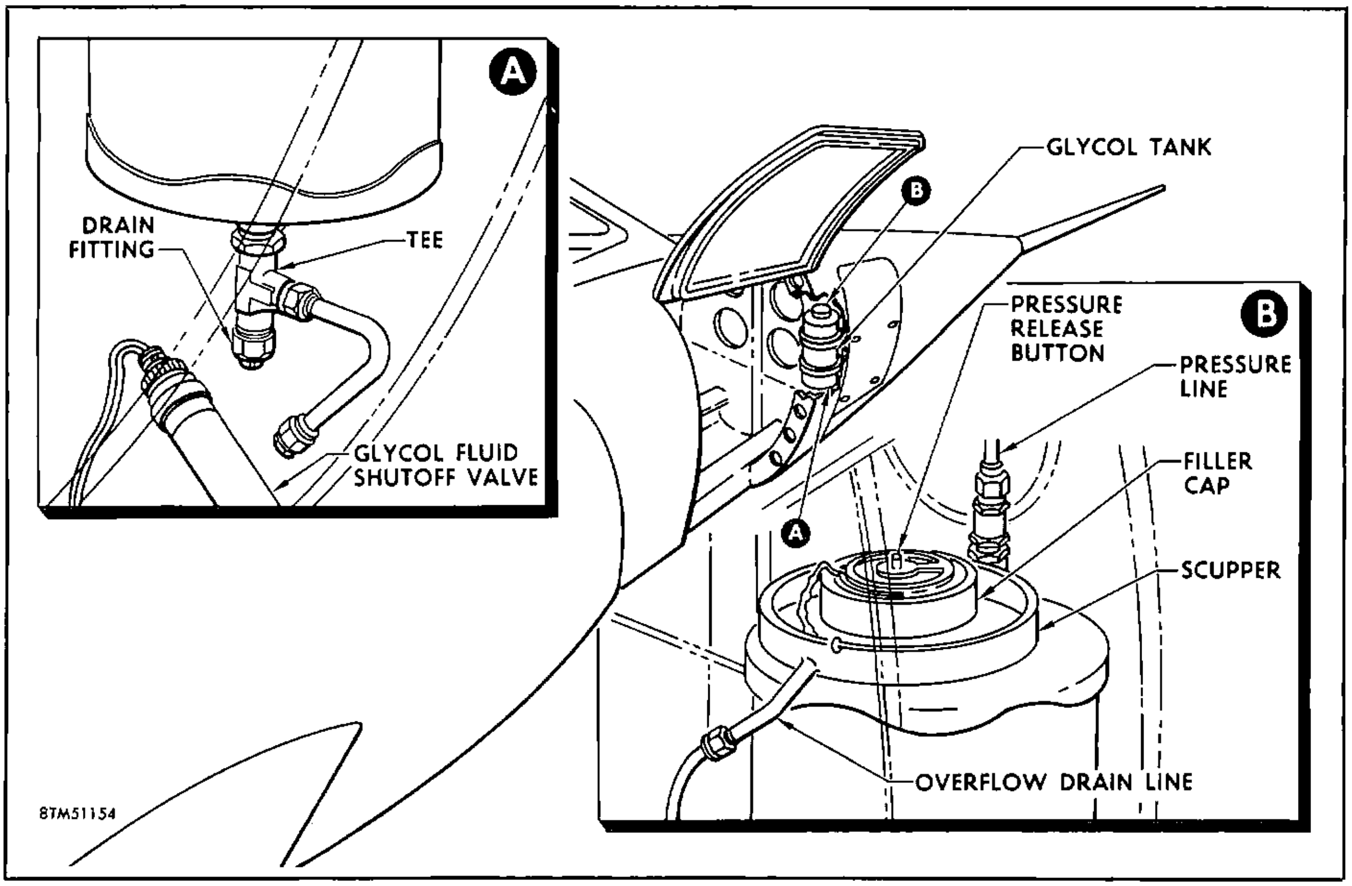


Figure 6-20. Glycol Tank

switch when the landing gear doors are open. The valve will also be open in the air if the anti-ice switch is placed at MAN ON. Replace a defective valve; do not attempt to repair it.

THE RELIEF VALVE AND THE SAFETY VALVE.

The relief valve and the safety valve in the glycol tank pressurization line are not only identical to each other, except for pressure setting, but they are also identical to the relief and safety valves in the canopy seal pressurization system. They are all made by the same manufacturer, so you must be very careful to be sure you install the correct unit when replacing these items. The canopy seal relief and safety valves were described and illustrated in Chapter V. The relief valve in the canopy seal system opens at 16 psi, while the safety valve opens at 30 psi. However, in the glycol tank pressurization line, the relief valve opens at 5 psi, and the safety valve opens at 16 psi. Where the units have the same pressure setting, they are interchangeable. All of these valves are one-way valves, and must be installed with the end marked IN facing upstream.

THE CHECK VALVE.

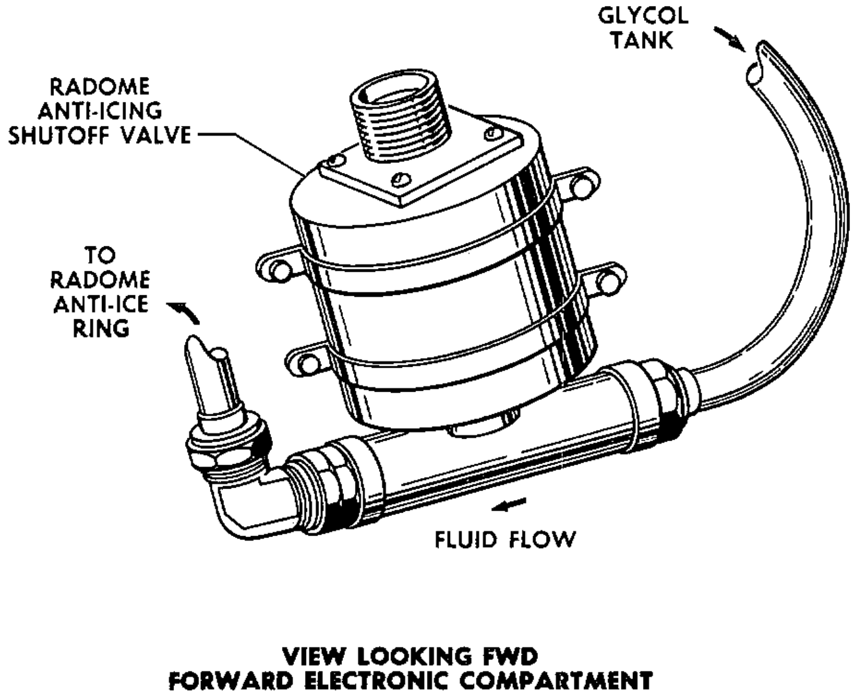
The check valve in the pressurization line is very similar to the relief valve or safety valve in construction and appearance. It contains a spring-loaded poppet that will open when the pressure on the IN side of the valve reaches about 3 psi. It is also a one-way valve that allows a flow in one direction only.

FILTERED ORIFICES.

The two filtered orifices are identical in appearance; however, the forward orifice has a smaller opening and allows a smaller air flow than the aft unit. The aft orifice is interchangeable with the orifices in the canopy seal system described in Chapter V. Each orifice contains a porous bronze filter and a restrictor plate. The orifices prevent dirt from entering the system and restrict the flow of air to help reduce the pressure. Never attempt to replace a clogged filter; but, instead, always replace the entire unit. The orifices can be installed in either position.

SUMMARY.

If you have carefully studied this supplement, you may feel that you are now an "expert" on the Low-Pressure Pneumatic System in the F-102A airplane. No one will blame you if you feel that way. You should remember, however, that it takes a lot of practice, actual doing, to make an expert. Remember too, that this supplement describes the system as it exists at a particular time.



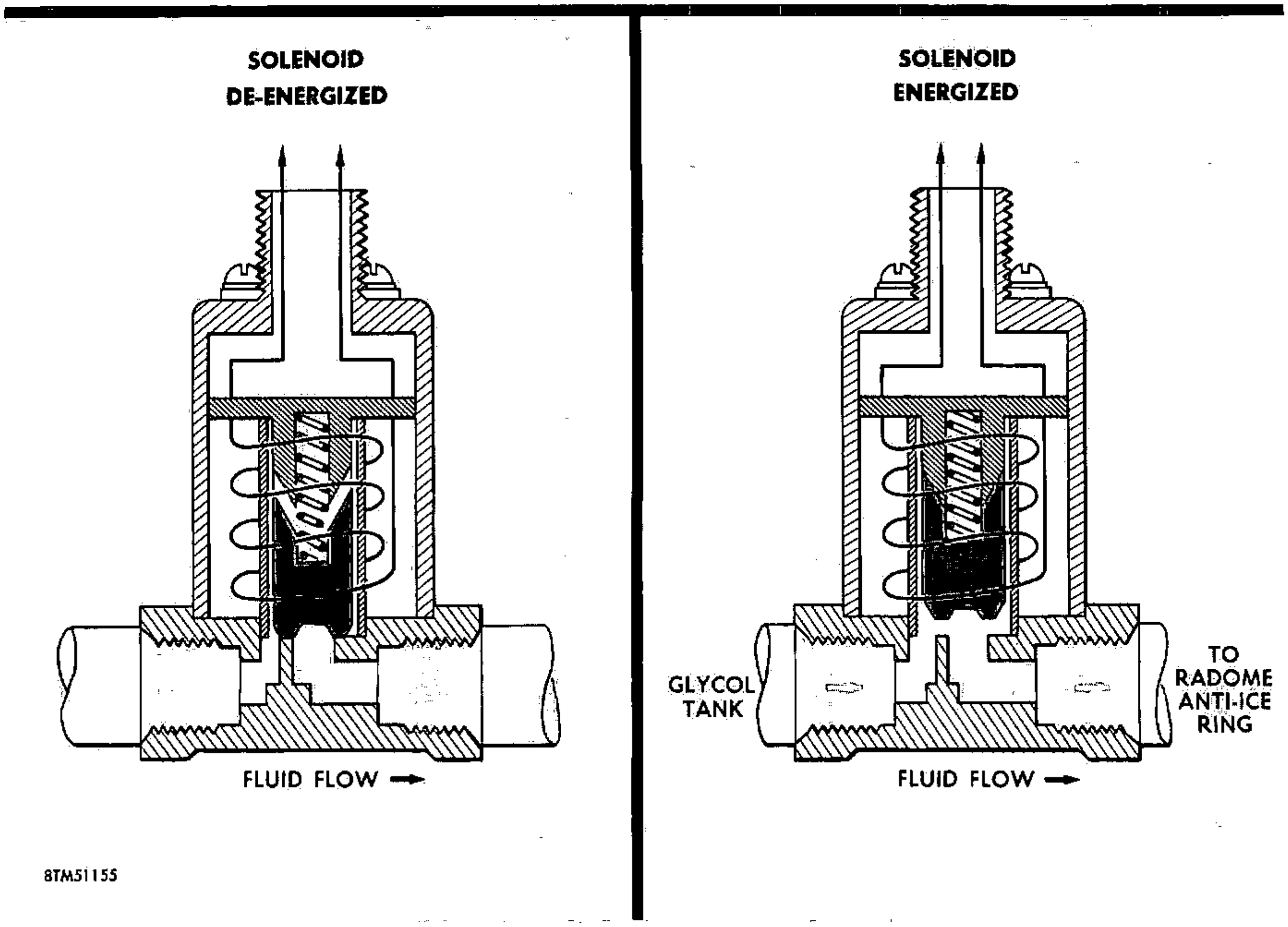


Figure 6-21. Radome Anti-Icing Shutoff Valve

Some of the components or systems may change from time to time. You are in a "jet age" Air Force that must always be dynamic and changing to keep ahead of the "other team."

As new and advanced equipment becomes available you may find some of it installed in the F-102A you are working on. The F-102A's produced in the future may have many changes. For that reason you must always check your F-102A maintenance manual before working on an airplane. These manuals contain the latest information and list the airplanes that have a particular configuration by serial numbers.

Remember also, that this series of F-102A maintenance training supplements and the F-102A maintenance manuals are not the only sources of information concerning the various components. Many Air Force Technical Orders have been issued, or soon will be issued, covering many of the individual components found in the airplane. These "T.O.'s" should be on file at your base. If they are not, you should inquire about them; they are useful sources of information.

When trouble shooting any one of the systems, bear in mind that the trouble may not always be in that system. Few of the systems are independent, for they intermesh with other systems.

Remember when working on the F-102A, that human lives and extremely expensive "hardware" are dependent on your skill, knowledge, and sense of responsibility. Always obtain the latest information available before proceeding with repairs. If you can't find the answer, don't be afraid to ask. It is much better to display a little ignorance than to be the cause of an incomplete mission or an accident.

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ALPHABETICAL INDEX

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*The Symbol * Indicates An Illustration*

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ALPHABETICAL INDEX (CONT)

F-102A

MAINTENANCE TRAINING SUPPLEMENT

(EXPERIMENTAL)

To Be Used in Conjunction With

T.O. 1F-102A-2-9

INSTRUMENTS

CONVAIR
F-102A

MAINTENANCE TRAINING SUPPLEMENT

INSTRUMENTS

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE

AIR FORCE—AFPS—SAN ANTONIO, TEXAS—5-57-2600

1 JULY 1956

Foreword

The F-102A Training Supplements have been prepared by Convair, A Division of General Dynamics Corporation, to provide you, the system mechanic or maintenance technician, with basic information concerning the F-102A airplane and its operating systems. There are ten of these supplements, each covering an airplane operating system or major component. The text is direct and informal and is supplemented with illustrations and diagrams to aid you in understanding how the systems operate. The following is a list of the Training Supplements on the F-102A airplane:

Title

Flight Control System
Hydraulic System
High-Pressure Pneumatic System
Low-Pressure Pneumatic System
Airplane General
Airframe Fuel System
Power Plant Installation
Airframe Armament System
Electrical System
Instruments

Each supplement describes an operating system or major component, how and why it operates, and maintenance problems that you may encounter. The entire major system is further simplified by describing each of its subsystems separately. Thus, when you understand how all of the subsystems operate, you will be familiar with the functions of the major system. This knowledge of the airplane systems will enable you to use other operational, service, and maintenance instructions.

Since the purpose of these publications is to familiarize and acquaint you with the airplane systems and components, specific values, measurements, and tolerances are not given. Any values that appear in these supplements are approximate only and are used to emphasize more clearly a certain operation or condition. You must refer to your 1F-102A-2-9 Technical Order and other pertinent handbooks for specific values when adjusting or checking a system and its components. These Technical Orders are revised periodically to include the latest maintenance procedures and data on the equipment installed in the F-102A.

The F-102A Maintenance Technical Orders consist of the following handbooks:

T.O. 1F-102A-2-1	**General Airplane**
T.O. 1F-102A-2-2	**Ground Handling, Servicing, and Airframe Group Maintenance**
T.O. 1F-102A-2-3	**Hydraulic and Pneumatic Power Systems**
T.O. 1F-102A-2-4	**Power Plant**
T.O. 1F-102A-2-5	**Fuel Supply System**
T.O. 1F-102A-2-6	**Air Conditioning, Pressurization, and Anti-Icing Systems**
T.O. 1F-102A-2-7	**Flight Control Systems**
T.O. 1F-102A-2-8	**Landing Gear**
T.O. 1F-102A-2-9	**Instruments**
T.O. 1F-102A-2-10	**Electrical Systems**
T.O. 1F-102A-2-11	**Radio-Communication and Navigation Systems**
T.O. 1F-102A-2-12	**Armament and Armament Electronics**
T.O. 1F-102A-2-13	**Wiring Data (F-102A)**
T.O. 1F-102A-2-13A	**Wiring Data (TF-102A)**

The Air Force is vitally interested in seeing that its equipment functions properly and is maintained as efficiently as possible. The Air Force, like a manufacturer of an automobile or commercial appliance, provides printed instructions for use with its equipment. The printed instructions you will use most frequently in maintaining the F-102A are the dash-2 Technical Orders listed above. They are available for your reference at the Maintenance Office on your base. You must refer to them frequently so that you will always have the latest maintenance infromation. The Training Supplements will help you to use these Technical Orders by answering many of the "hows" and "whys" that may puzzle you from time to time.

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Introduction

This supplement consists of six chapters which cover the F-102A Instruments and Warning Systems. Chapter I introduces the instruments used in the F-102A and reviews the principles of the basic instruments. Chapter II describes the flight and navigation instruments. Chapter III explains the engine and fuel system instruments. In Chapter IV you learn about the airplane power supply and pressure indicating instruments. Chapter V explains the master and fire warning systems. Chapter VI gives the miscellaneous instruments that are not included in the above groups. This supplement concludes with a descriptive summary of the F-102A instruments.

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Chapter I

AIRCRAFT INSTRUMENTS AND PRINCIPLES OF OPERATION

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Aircraft instruments are quite bewildering to anyone who has never had an opportunity to learn about them. The mere fact that there are so many of them is enough to cause confusion in anyone's mind when he views the average airplane instrument panel. You will find that the F-102A is no exception in this respect—it has lots of complicated-looking instruments on its panels. But the fact that they look complicated doesn't necessarily mean that they are; even if you are new to the Air Force you are already familiar with at least one of those instruments—the clock. And you will probably decide after reading this manual that many of the other instruments are less complicated than the clock. By learning about the instruments used in the F-102A, you will become familiar with some which are also used in other types of aircraft.

WHY INSTRUMENTS ARE NEEDED.

Human limitations make it impossible for a pilot, using his senses alone, to cope with all the factors involved in the flight of an airplane. Instruments give him invaluable assistance by indicating many conditions which influence the successful operation of his aircraft. The dials of various instruments, located conveniently before him, record variations in temperature, pressure, speed, altitude, direction, drift, and attitude. They also give him information regarding the mechanical functioning of his equipment. Even when it is impossible for the pilot to see the ground, reliable instruments provide him with all the information he needs to maintain flight.

BASIC CHARACTERISTICS OF AIRCRAFT INSTRUMENTS.

Although there are many types of instruments, the demands of modern aviation make it necessary that all instruments meet certain requirements. The individual features of the various instruments are discussed in the next chapter of this manual. The features which are common to all instruments will be discussed here.

All instruments are small and light in weight. They must be small to fit in the limited space available for them in the airplane. It is essential that they weigh as little as possible because excessive weight will retard aircraft speed and effectiveness. The "payload", whether armament, cargo, or passengers, should not be sacrificed for unnecessary equipment weight.

Instruments are precision made to indicate accurately under all conditions encountered in normal flight, and yet rugged enough to require little maintenance. Each one is isolated by its own case, as much as is practical, to reduce the effects of magnetic fields created by electrical devices surrounding it. Every instrument must be easy to read. Special luminous paint is applied to the dials and pointers, and some form of lighting is used to make the instruments clearly readable under all conditions. Either individual lighting or complete panel lighting is always provided.

Each instrument mechanism is inclosed in a protective case to keep out dust and moisture and prevent corrosion. Cases are made of plastic or light-weight metal. The glass in front of the dials is sealed, usually by the pressure of a snap ring which holds it against a rubber gasket. Some instruments are entirely sealed and filled with a gas to allow maximum protection and consistent operating environment for the mechanism.

F-102A INSTRUMENT PANELS.

The instrument panels in the F-102A are shown in figures 1-1 and 1-2. Notice that there are actually several different panels—the main instrument panel, the right and left auxiliary panels, the pedestal, and the right and left consoles. Most of the instruments used in the F-102A are located on these panels. A few of the instruments are primarily installed for your convenience in ground operations and are not located in the cockpit area; their locations are explained in later chapters. You will also see that the instruments in the cockpit area are not all independent in their operation —most of them are connected to systems and devices in other parts of the airplane.

The instruments are discussed separately throughout the manual according to their group function. You will find, for example, that the main instrument panel contains instruments from several groups, but only one particular group is discussed at a time.

THE MAIN INSTRUMENT PANEL.

The entire center (the area from the glare shield down to the lower radar and armament controls) is the main instrument panel. Radar scope controls occupy the space on each side of the scope at the top of the panel. Below that, in the center of the panel, are the flight and navigation instruments. These instruments tell the pilot his position in relation to the air and the ground.

Engine instruments are to the right of the flight and navigation group. These keep the pilot posted on the operation of his power plant. On the left side of the main panel, you can see the landing gear position indicators and compass slaving switch. The three rectangular bars on the top and right side of the panel are warning lights. All of the controls on the lower strip of the main panel are armament and radar controls—they will not be discussed in this manual.

The main instrument panel is clamped to a tie rod near the bottom. By removing a screw on each top corner, you can tilt the top of the panel aft, toward the seat, for replacement of any equipment above the armament and radar strip. To free the top of the panel, you must first remove the glare shield, one radar control panel, and the scope.

THE AUXILIARY INSTRUMENT PANELS.

Extending down from the lower corners on each side of the main panel are the left and the right auxiliary panels. The left panel contains the landing gear control panel, drag chute control panel, landing and taxi light switch, and the cockpit pressure altimeter. On the right auxiliary panel you will find the canopy latch and warning mechanism, the warning indication panel and test switch, and the a-c voltmeter.

THE PEDESTAL.

The pedestal is the center panel, below the main panel. The pilot straddles the pedestal when he is seated in the cockpit. As you can see in figure 1-1, there are two sections to this panel. The upper section is called the utility switch panel. It contains various controls for cabin temperature regulation, anti-icing, and trimming the airplane. The lower section, called the lighting control panel, has switches to turn on and adjust the brightness of the lights. The crank at the base of the pedestal is used for rudder pedal fore and aft adjustment. With this crank the pilot can set the pedals to his most comfortable position.

THE CONSOLES.

The consoles are the panels which extend along the sides of the cockpit. On the left console you can see the oxygen regulator controls, throttle quadrant, emergency hydraulic control, communications controls, fuel control panel, and equipment to operate the pilot's mask and G-suit. The right console includes an electric power switch panel, the radio navigation controls, hydraulic pressure and free air temperature gages, and the controls for various electronics and radio gear. The circuit breaker panel is located at the aft end of this console.

The instrument panels of the F-102A are lighted by small light caps which screw into sockets in clear plastic panels. These plastic panels are held in place over the instrument panels by the light caps. The outer surfaces of the panels are painted so that the light will reflect through the cutouts—for the instruments and controls—and through the printing and numbering on the panels.

BASIC TYPES OF INSTRUMENTS AND PRINCIPLES OF OPERATION.

Each instrument which appears on the panels of the F-102A has a very definite purpose. Sometimes the information given by one indicator can be obtained from another and you may be sure that both instruments taken either individually or in combination, are important to the safety or efficiency of the airplane. Several instruments operate on the same principles and

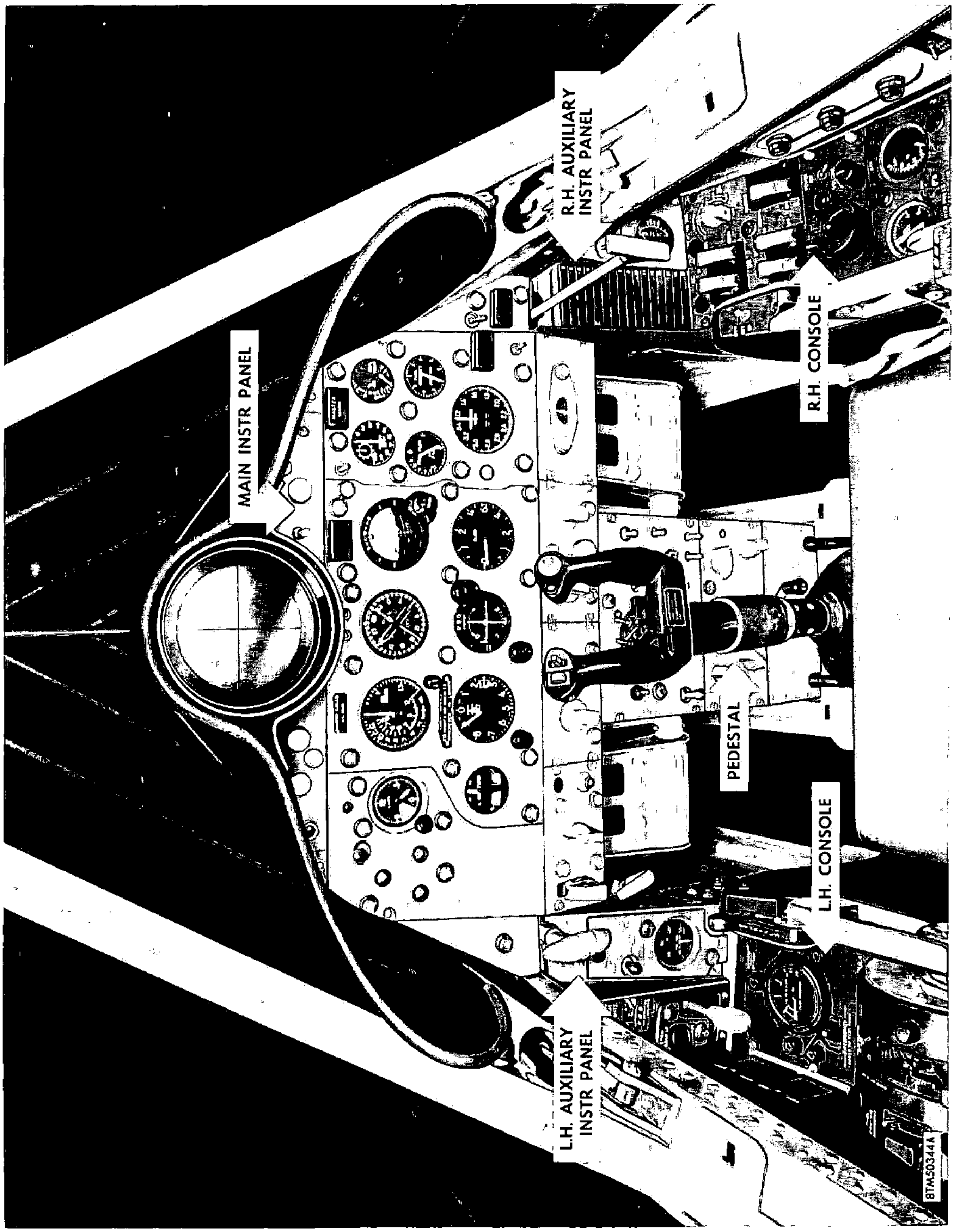

Figure 1-1. F-102A Instrument Panel

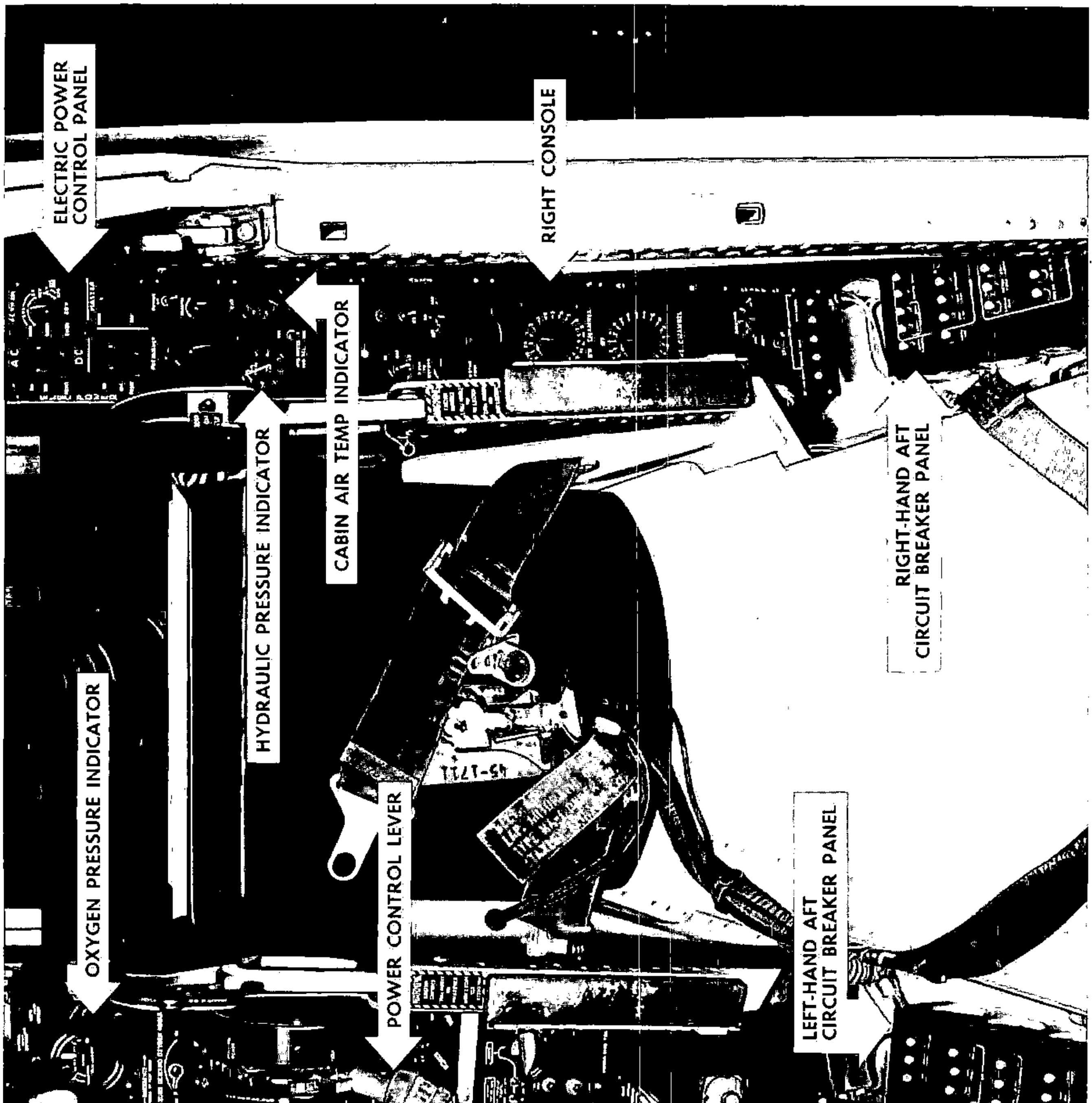

Figure 1-2. F-102A Cockpit Console Panels

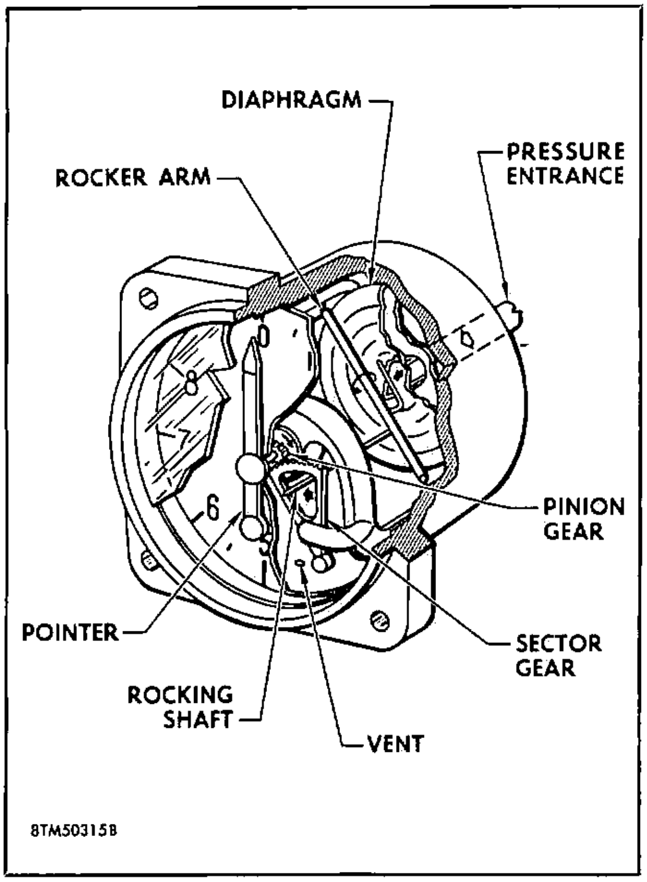

Figure 1-3. Diaphragm-Type Pressure Gage

use very similar mechanisms. It is therefore convenient to classify some of them according to functional operation. An explanation of these basic kinds of instruments will help you to understand the various instruments discussed in detail in the next chapter.

DIAPHRAGM INSTRUMENTS.

Some aircraft instruments measure differential pressure by the use of one or more metallic diaphragms. A simple diaphragm mechanism is shown in figure 1-3. The diaphragm is a hollow, circular disk made from very thin metal, usually copper or brass alloy which is springy and will resist corrosion. The sides of the disk are corrugated to allow for expansion and contraction. Sometimes a spring is installed inside of the disk to damp its action or prevent damage from excessive pressures.

A fitting in the case of the instrument admits the pressure which is to be measured. It enters the inside of the diaphragm where the diaphragm is attached to the fitting. An opposing pressure, such as atmospheric pressure, is admitted to the case through another opening. Since the walls of the diaphragm are very thin and flexible, any increase in the pressure on its inner surface causes it to expand. The amount that the diaphragm expands is proportional to the pressure differential. Notice that a rocking shaft positions the pointer of the indicator through a gear and pinion arrangement.

ANEROID INSTRUMENTS.

The aneroid mechanisms found in many instruments are very similar to ordinary diaphragms in both appearance and mechanical operation, in fact, the aneroid unit is a type of diaphragm. It is different from the regular diaphragm in one very important respect—it is a sealed, evacuated chamber, but the regular diaphragm type as you will recall, had a chamber that received the varying pressures. Notice in figure 1-4 that the chamber does not have an opening to admit pressure as was the case in the diaphragm type. The aneroid unit used in aircraft instruments is the same as the unit which operates the ordinary aneroid barometer. You will find that most aneroid instruments use a more elaborate system of linkage from the diaphragm to the pointer than is shown here, but the principle of operation is the same.

You have learned that the diaphragm unit expands when the pressure on the inside surface becomes greater than the pressure on the outside surface. But, the aneroid instrument is sealed and will operate differently. The only pressure which can change is the

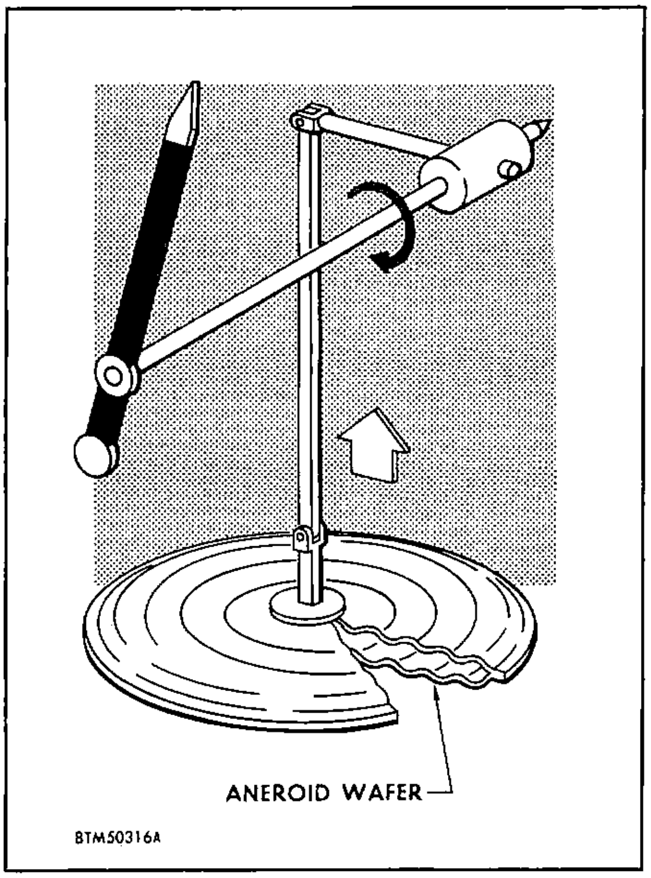

Figure 1-4. Simple Aneroid Unit

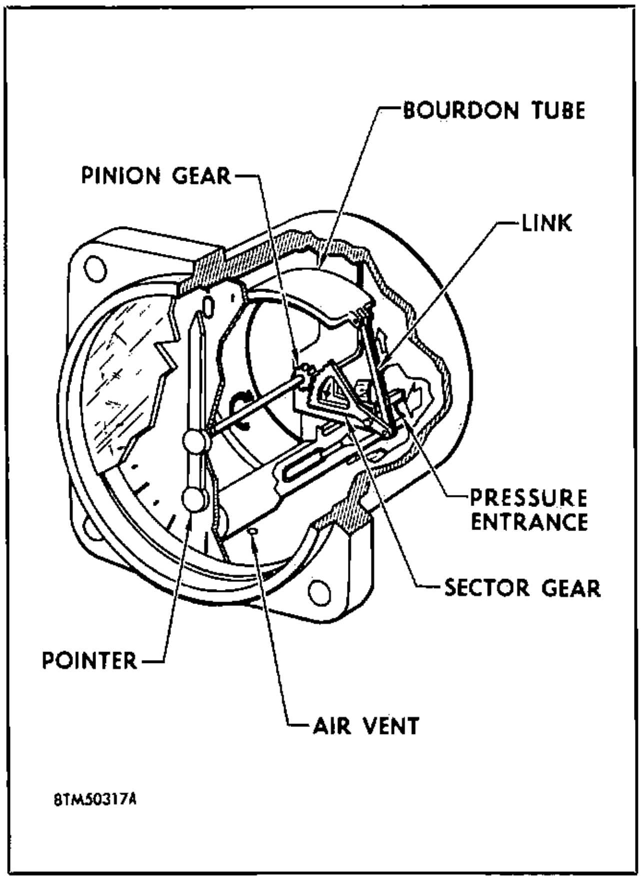

Figure 1-5. Bourdon-Tube Instrument

pressure on its outer surface, which is the pressure that is admitted to the case surrounding the chamber. The strength of the metal, sometimes aided by a spring within the unit, is all that prevents it from being totally collapsed. The aneroid instrument is not, therefore, a differenial pressure measuring device, since the pressure within the aneroid unit is constant. Rather, it might be called an absolute pressure indicator.

BOURDON-TUBE INSTRUMENTS.

Bourdon-tube instruments are used to measure the pressure of either gases or liquids. A simple Bourdon-tube instrument (figure 1-5) contains a single curved tube. The tube is eliptical in shape and is made of bronze or copper alloy. One end of the tube is open and is attached to a fitting in the instrument case. The other end is sealed and is free to move to a limited degree, but it is connected by a mechanical linkage system to the pointer of the instrument.

The principle involved in the Bourdon tube can be easily demonstrated by the novelty "snakes" of Halloween and New Year's parties. As you blow into the mouthpiece, the "snake" unwinds and will straighten due to the air pressure in its body overcoming a spring. When you release the air, the "snake" will coil up once again because the spring is tempered in a coil. A Bourdon tube is quite similar, except that it does not have many coils. The short metal tube is made from a spring type metal and tends to coil when there is no pressure in the tube.

Pressure to be measured by a Bourdon-tube instrument is piped to the back of the instrument case. The fluid or gas, whichever is used in the system, enters the tube through the end which is attached to the fitting. When the pressure in the tube exceeds the normal air pressure in the case surrounding it, the tube tends to straighten. The greater this pressure differential becomes, the more the tube extends to give a greater pointer deflection. Releasing the pressure in the Bourdon tube permits it to spring back to its neutral position and the pointer returns to zero.

GYROSCOPIC INSTRUMENTS.

Many pilots learned the hard way that the old "seat-of-your-pants" technique of flying is inadequate for bad weather operations. The experience gained by their flights through the overcast proved that the human senses are unreliable under such conditions. When the pilot of an airplane loses sight of the ground, some other fixed reference is needed. Gyroscopic instruments provide that reference.

The primary component of any gyro is its rotor—a finely balanced wheel. The rotor is very heavy for its size (a high weight-to-mass ratio) and it spins at a high rate of speed. The axle of the rotor is called its spin axis. Low-friction bearings hold the spin axis in mountings which are known as gimbals. A simple model gyroscope is shown in figure 1-6. You may have played with one just like it when you were a child. Chances are you never fully understood its antics, nor realized that it could serve any practical purpose.

There are two general types of mountings for gyros; the type used depends on which property of the gyro is being utilized. The model shown in figure 1-6 is a freely or universally mounted gyro, that is, no matter which way you would tilt the base, the rotor would be able to hold its original (in this case, vertical) position. Such a gyro is said to have three planes of freedom, because it can rotate in any plane in relation to its base. Restricted or semi-rigidly mounted gyros are those mounted so that one of the planes is held fixed in relation to the base. If the outer gimbal of the model were welded to the frame instead of mounted in bearings, you would have a restricted gyro.

Now consider what it is that makes the gyro so useful in aircraft instruments. There are two rules, or principles of operation, which are common to all gyros—*rigidity in space* and *precession.* All practical applications of the gyro are based on these two fundamental properties of gyroscopic action.

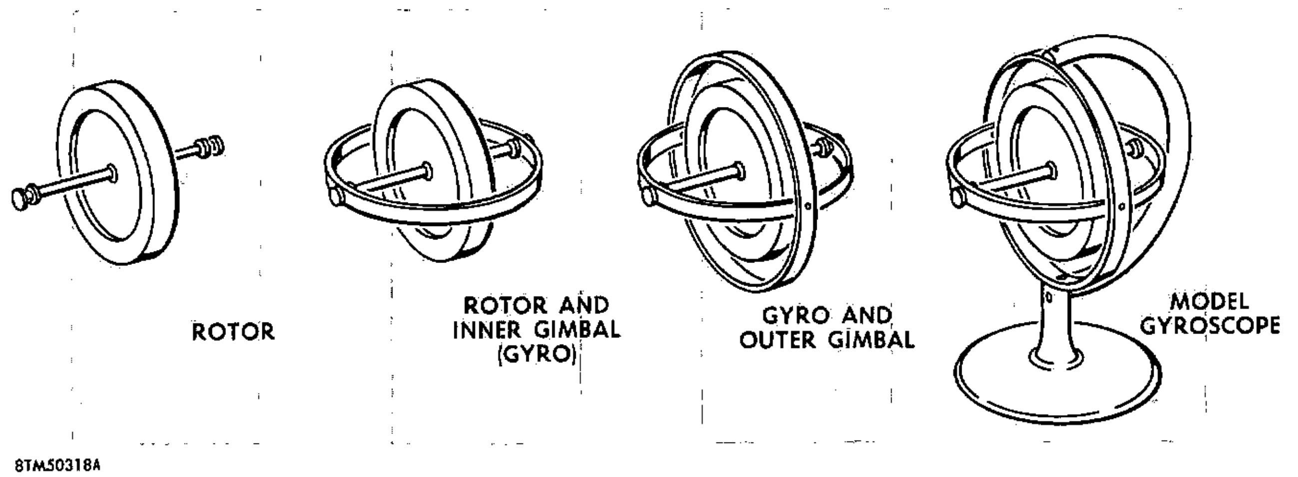

Figure 1-6. Elements of a Simple Gyroscope

Rigidity In Space.

A universally mounted gyro is free to remain in any set position. If the rotor of that gyro was turning at a high rate of speed, you could hold the base in your hand and tip it upside down in any direction without disturbing the attitude of the rotor. Similarly, you could attach the base to the aircraft structure and the spinning rotor would remain in the same plane of rotation during all flight maneuvers. A restricted gyro has the same characteristic of rigidity but is not free to rotate in all three planes.

Precession.

The principle of gyroscopic precession is difficult to explain in words, but it is easy to demonstrate. If you study figure 1-7 as you read the text, you may be able to understand it readily. Precession is the resultant action of a spinning wheel when a deflective force is applied to its rim. When a force is applied to the rim, the resulting force is 90° ahead of the point where the force is applied—in the direction of rotation and in the same direction as the applied force. Now that rule can be broken down into three simple steps and you can always determine how a gyro will precess by these steps.

First, determine at what point on the rim the deflective force is applied. In figure 1-7, the pressure applied at the bottom of the bearing (e) is—in effect—a force applied at the rim at point A and in the direction indicated. Second, determine where the deflective force actually takes effect. This is at point B—90° ahead of point A in the direction of rotation. Third, remember that the gyro precesses in the same direction as the applied force (which is perpendicular to X, or to A). The result is the same as if you pushed directly at point B (perpendicular against the side of the rim) when the rotor is not spinning.

Precession of a gyro is caused by mounting the rotor so that one of the three planes of rotation is fixed; that is, by using a restricted gyro. Any movement which causes a deflective force against that plane results in precession. The amount of precession is proportional to the deflective force—this can be measured and shown by an indicating mechanism. Some precession is caused by bearing drag and the rotation of the earth; this precession is undesirable. Aircraft instruments always provide some means—either manual or automatic—to return the gyro to its proper position.

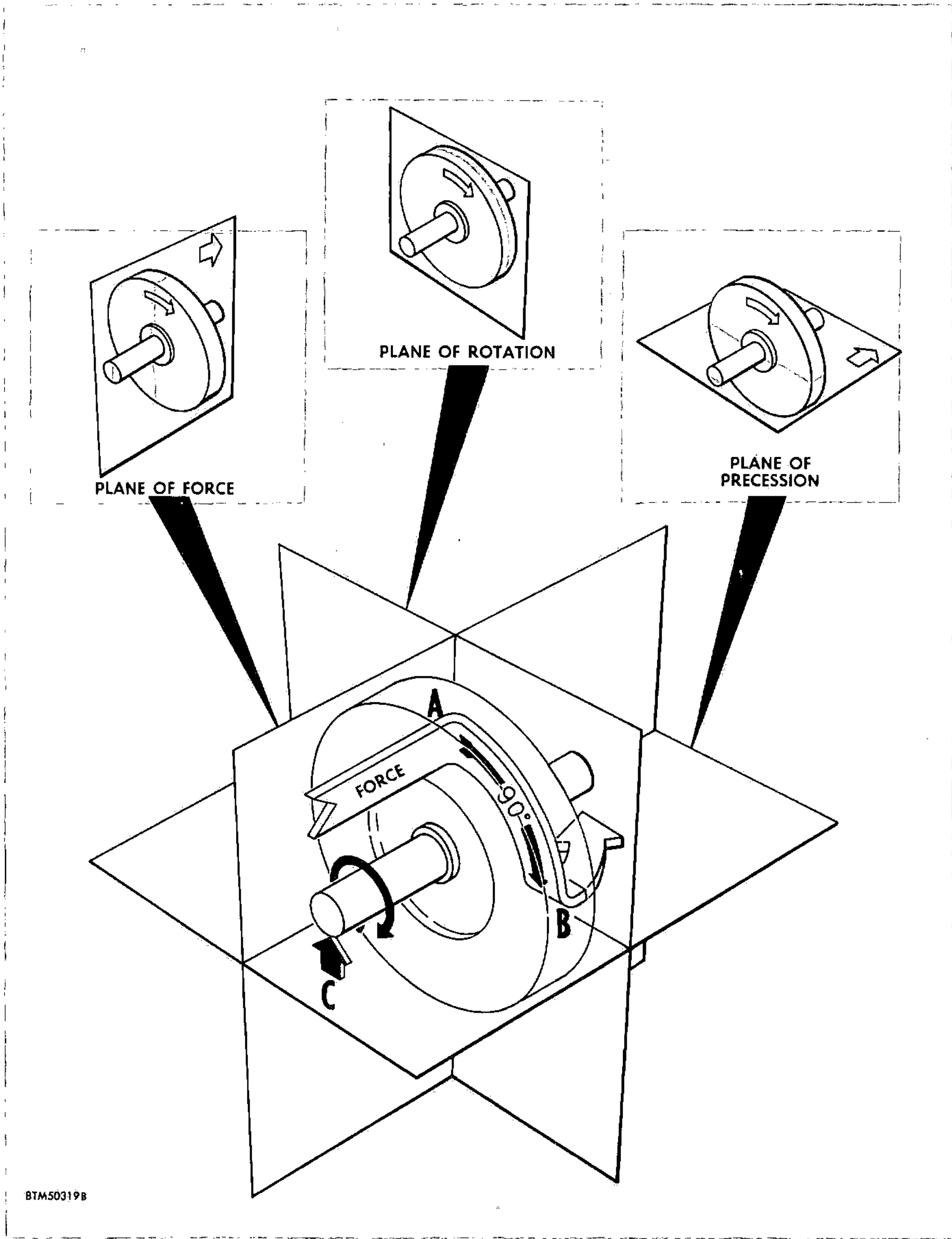

Figure 1-7. Gyroscopic Precession

MAGNETIC INSTRUMENTS.

Perhaps you have had some experience with magnetism. It is quite likely that a rusty magnet from a telephone or Model "T" Ford magneto was once one of your prized possessions. Some of the basic principles of magnetism may not be familiar to you, so we will review these principles that apply to instruments in the F-102A airplanes.

There are two types of magnetic instruments in the F-102A. One type is used for directional guidance and operates by sensing the magnetic lines of force of the earth—these instruments are frequently called earth magnetic instruments. The other type uses magnetic fields that are created by electrical means within the indicator; these are known as electromagnetic devices.

Earth Magnetic Instruments.

The ancient Greeks discovered that a certain kind of rock (known as magnetite) would attract bits of iron. Later, it was learned that when a piece of this material was suspended on a string, it would turn so that one end would always point north. This material became known as lodestone (leading stone) and was used for navigation. Such was the beginning of magnetic instruments.

Every magnet has a north pole and a south pole. Invisible lines of force leave the magnet from the north pole and spread out in all directions. These lines (magnetic flux) curve around and return to the magnet through the south pole. From the south to the north pole, they move within the magnet, but never intersect. Where the lines are nearest together—within the magnet and at its ends—the force is the most concentrated. Magnets are frequently curved like horseshoes to provide a concentrated area of force between the two ends. This magnetic force is called magnetomotive force.

If you place two magnets together, the two north poles will repel each other and the two south poles will repel each other. The north pole of one will attract the south pole of the other. This represents the fundamental law of magnets: like poles repel, unlike poles attract.

Metals other than magnetite can be magnetized by induction. This is accomplished by stroking the metal with a strong magnet or by use of electricity. The "reluctance" of the metal (similar to resistance in electricity) determines how difficult it is to magnetize. You can see an example of induced magnetism by dipping a magnet into a box of tacks. Not only do the tacks cling to the magnet, they cling to each other as well. What actually happens is that the molecules in the tacks assume a regular pattern—each molecule lines up as though it were a separate little magnet. After you remove the magnet, some of the tacks will retain their magnetism for a time—this is called residual magnetism.

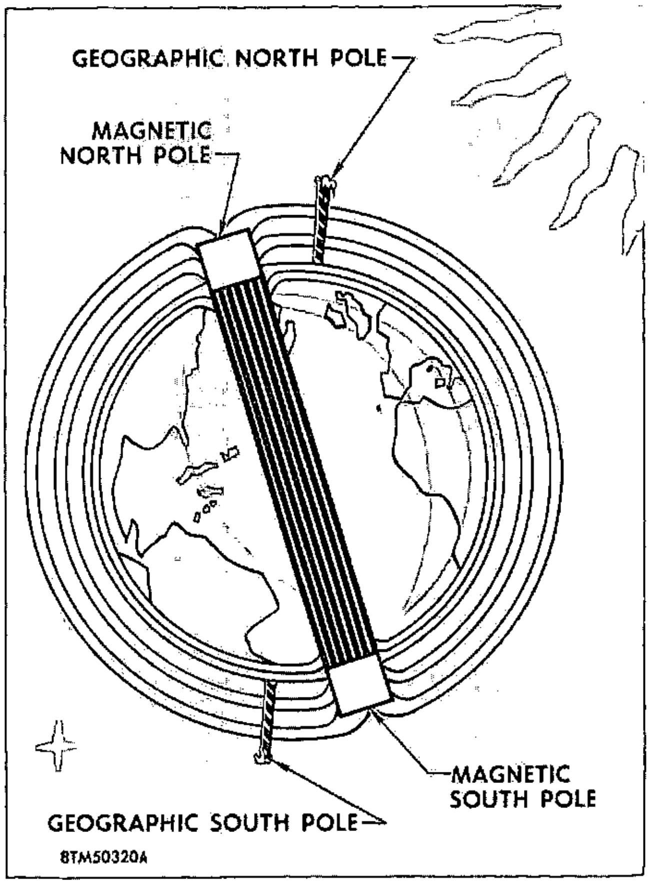

Figure 1-8. The Earth as a Magnet

Certain high-reluctance metals have the ability to retain magnetism for a long period of time. Hard steel is frequently used to make "permanent" magnets because it has this characteristic of retentivity. These man-made magnets are called artificial magnets. By observing a freely mounted magnetic needle (a compass), it was proved that a magnetic field surrounded the entire earth. It is as though there were a huge bar magnet running through the earth, with its ends several hundred miles below the surface of the earth.

Figure 1-8 shows how the lines of force flow from the magnetic core. Any freely mounted magnet will tend to align itself with the lines of force, as shown in the illustration—the north pole of the magnet will then point toward the earth's magnetic north pole. Now, this statement may seem contradictory, but it really isn't. The earth's magnetic north pole is so called because it is in the northern hemisphere, near the geographical north pole. In magnetic terms, it is a south-seeking pole, because it attracts the north-seeking pole of a freely mounted magnet.

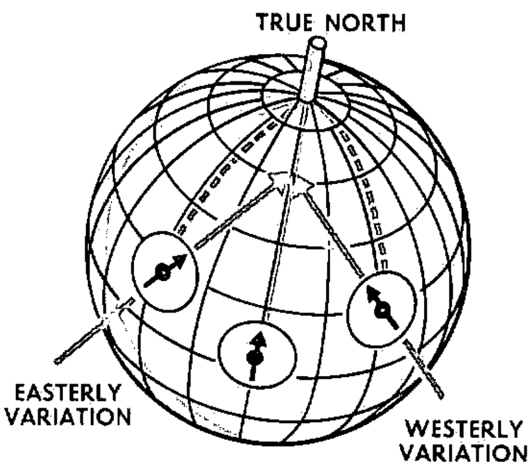

Figure 1-9. Effect of Variation on a Magnetic Needle

OPERATION OF THE EARTH MAGNETIC INSTRUMENTS. You have learned that the earth is surrounded by a magnetic field, and that a magnetized needle reacts to this field. This is the principle behind most magnetic compass instruments. It is also possible to get magnetic directional information from a sensing element that shows at what angle it is moving across the earth's lines of force. Regardless of the magnetic instrument used, there are certain factors which must be considered during its use; variation, deviation, and dip. We will discuss these factors now, but you must keep them in mind in later discussions of specific instruments and systems.

The earth's magnetic poles are not in the same locations as are its geographical poles. Consequently, at most places on the earth's surface, a needle that is lined up with the magnetic field will not point to true north. The difference between true north and the position taken by a freely suspended magnetic needle is called variation. Notice in figure 1-9 that the variation is called *easterly* if the needle points to the right of true north and *westerly* if the needle points to the left of true north. Areas of equal variation are shown on charts by isogonic lines. The area of no variation—where the needle points to both the magnetic north pole and the geographical north pole—is indicated by the agonic line. To determine true north from magnetic north, a pilot must subtract easterly variation or add westerly variation for the area in which he is flying.

As you can see on figure 1-10, the lines of equal variation are neither straight nor parallel. The reason is that local magnetic fields—resulting from mineral deposits and other conditions—distort the earth's magnetic field, thereby causing additional error in the position of a magnetic needle with reference to true north.

All of the earth magnetic aircraft instruments are subject to deviation—the error caused by disturbances within the airplane itself. As you will learn in the discussion of electromagnetism, operating electrical equipment sets up magnetic lines of force which tend to affect the needle in an earth magnetic instrument. In addition, various metal parts of the airplane also cause deviation; remember, metal contains residual magnetism. Therefore, each magnetic compass is provided with some means of compensation. Usually this takes the form of a system of adjustable magnets within the case of the instrument. They can be adjusted to remove some of the errors by "splitting" the error on one heading with the error present at the reciprocal heading. Any remaining error is shown on a deviation card, adjacent to the instrument, that tells the pilot what course to steer to obtain a certain magnetic heading.

Referring to figure 1-8 which shows the magnetic lines of force around the globe, notice that these lines run approximately parallel to the surface of the earth at the equator. In fact, they are absolutely parallel to the earth's surface at the magnetic equator—the area halfway between the magnetic poles. This represents the "horizontal component" of the magnetic field. At the upper latitudes, near the magnetic poles, the lines of force turn towards the center of the earth as though the ends of our imaginary bar magnet were buried beneath the surface of the earth. This is referred to as the "vertical component" of the magnetic field, and is responsible for what is called "dip."

Any freely mounted magnetic needle will tend to dip toward the earth in these areas, causing it to oscillate and read erroneously. Directly over the poles—where the vertical component is 90° in relation to the surface of the earth—the ordinary magnetic needle becomes totally unreliable as a source of directional information. Magnetic sensing elements which use gravitational or gyroscopic stabilization are less affected by magnetic dip, and are used successfully in these areas.

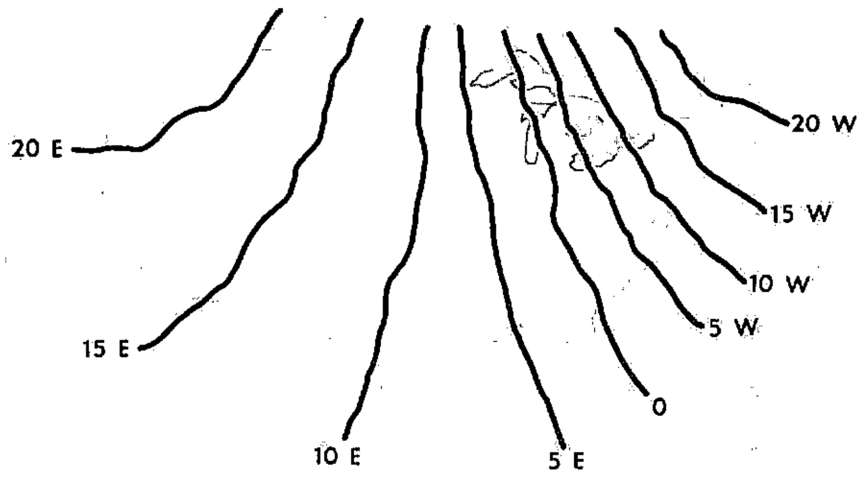

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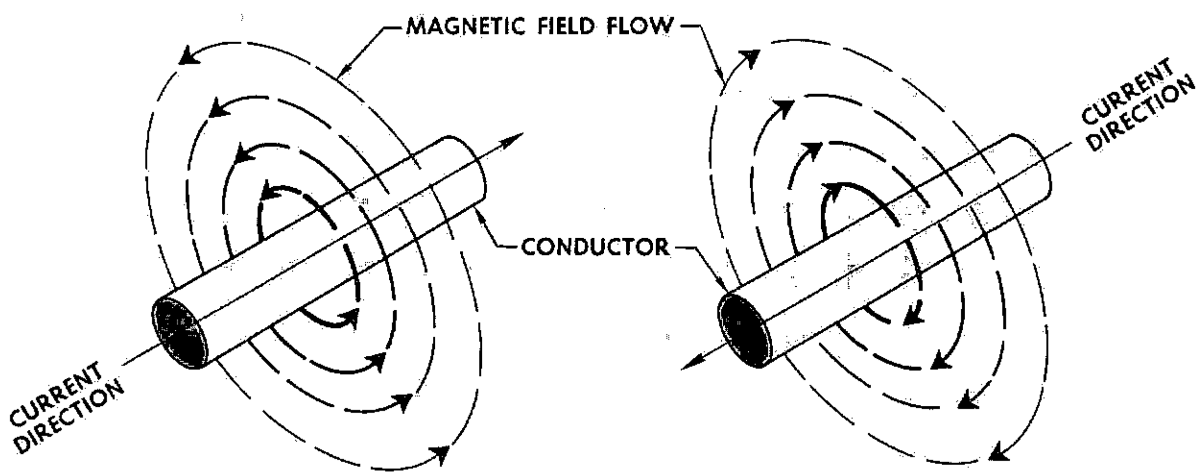
Figure 1-10. Isogonic Lines on a Map of the United States

Electromagnetic Instruments.

Electromagnetism, in contrast to natural magnetism, is the magnetic field of force set up around a conductor when current passes through it. Every electric current produces a magnetic field—the strength of this field is proportional to the strength of the current flowing in the conductor. You can prove the existence of a magnetic field around a current-carrying conductor by holding a compass close to a "hot" wire. The compass needle will tend to align itself at right angles to the wire. This proves that magnetic lines of force are traveling in concentric circles around the wire. The intensity of this magnetic field varies inversely with the distance from the conductor; that is, the magnetic force of the field is strongest close to the wire and decreases with distance from the wire. Figure 1-11 shows how some of these lines of force would appear if they were visible.

A simple rule will always show you the direction taken by the lines of force around a current-carrying conductor. If you take the conductor in your left hand (it might be a good idea to disconnect the power supply first) with your thumb pointing in the direction of the current flow, your fingers will be pointing in the direction of the magnetic force. This is commonly referred to as the left-hand rule. Remember, then, that the direction of the magnetic field is directly dependent on the direction of the flow of electricity. If the



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MAGNETIC FIELDS ABOUT A CONDUCTOR

Figure 1-11. Lines of Force Around an Electrical Conductor

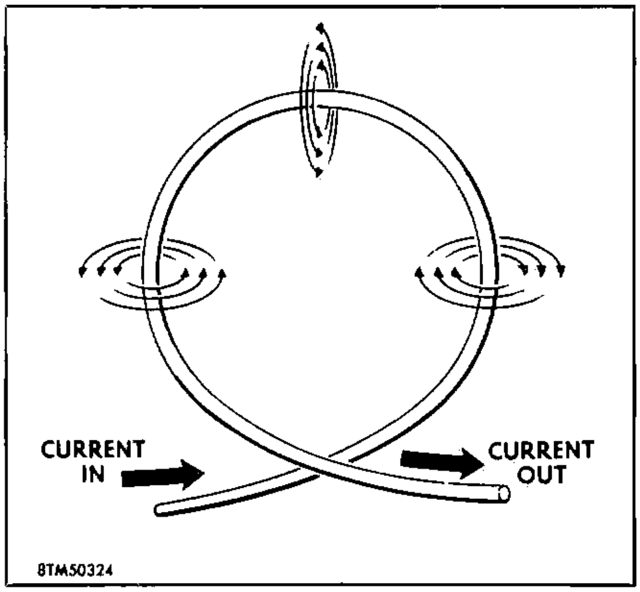

Figure 1-12. Concentrating the Magnetic Force

electrical flow is reversed, the magnetic lines of force also change direction.

Now let's loop the wire once, as shown in figure 1-12. Notice that the magnetic flux now becomes more concentrated in the center of the loop. You will recall from the discussion of natural magnetism that the force of the field is strongest where the lines are nearest together. Bending the wire into a loop does not reduce the number of lines of force; it only brings them closer together within the inside of the loop. A coil of wire is simply a succession of loops.

When you connect electric power to a coil, the magnetic force of the individual loops tends to be cumulative; that is, each loop adds its lines of force to the total field. Notice in figure 1-13 how the lines around the individual loops conflict because they are moving in opposite directions at their closest points. Therefore, most of the lines of force go through the entire coil (the path of least reluctance) and then circulate around to the other end. Regardless of where they leave the center of the coil, the lines of force always return again—since any line of force must close.

You could use an ordinary "air core" coil of wire as an electromagnet, but it is desirable to concentrate the magnetic lines of force even more. To do this, the coil is wound around a soft iron core—the wire in the coil is insulated to prevent it from "shorting" on the core. This iron core is much more permeable to magnetic flux than air, in other words, it offers less reluctance. Consequently, the iron core becomes magnetized by induction. When the electrical current is shut off, the magnetic field collapses and the soft iron loses its magnetism because it has very low retentivity. This is basically how every electromagnetic device works.

OPERATION OF ELECTROMAGNETIC INSTRUMENTS. The most common applications of electromagnetism in aircraft instruments are in solenoids, relays, and synchros. We will discuss synchros separately in the next section. Solenoids and relays are rather simple adaptations of the basic electromagnet you just finished reading about. Although instruments do not operate solely by the use of relays, these relays are an important part of some instruments. Because they are so commonly used in aircraft and are mentioned later on in the discussion of certain instrument systems, let's learn how they operate.

A relay is nothing more than an electromagnetic switch which you have probably seen in telephones or in automobile voltage regulators. There are many different types, but they all work on the same principle as the one shown in figure 1-14. When current flows through the coil in a relay, the iron core becomes magnetized and pulls the plate above it down. Notice in the illustration that the plate is hinged on one end and is suspended by a spring. The plate itself is of a nonconductive material with a strip of iron on its lower surface that contacts the iron core of the magnet.

At the other end of the plate is a set of contact points. Notice that the relay in figure 1-14 has just one set of

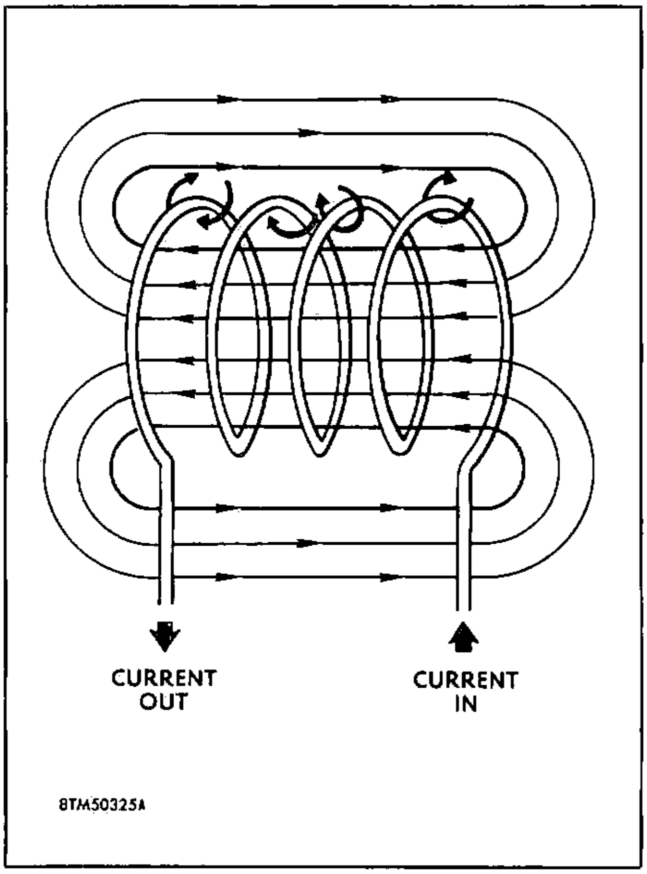

Figure 1-13. Magnetic Flux Around a Coil

contact points—some relays have more than one set, depending on the application. When the electromagnet is energized, the contact points close and the circuit is complete. This type of relay is known as a normally open relay. A normally closed relay is designed so that that contact points are closed when the current to the electromagnet is off, and they open when the coil pulls the iron strip down to its core.

A solenoid is actually an electromagnet whose core—or part of its core—is free to move. You may have heard it called a "sucking coil." Figure 1-15 shows you a typical solenoid in which only part of the core is movable—the armature or plunger. A spring within the coil holds the two sections of the core apart until the coil is energized. Notice that this solenoid has several layers of tightly wound wire or coils around its core. This coil winding serves two purposes: it conserves space and it assures that the lines of force will circulate through the entire coil rather than around each individual loop of wire.

Solenoids are used in many different ways, in aircraft instruments the most common application is in position indicators. You will learn about some of these indicators in the following chapters of this supplement.

SYNCHRONOUS INSTRUMENT SYSTEMS.

Self-synchronous instrument systems are used to measure position, pressure, temperature, or quantity and to show the results of these measurements on remote indicators in the cockpit. Each synchro system consists of an indicator and one or more transmitters. Transmitter positioning is accomplished by many devices—diaphragms, Bourdon-tubes, liquidometers, or a variety of other devices—depending on the function. The indicator synchros, in turn, are positioned by electrical signals from the transmitters, and assume positions identical to the positions of the transmitting synchros.

Synchro systems are used where the distance from the cockpit to the units being controlled or measured is too great for other methods. The need for piping dangerous gases or fluids into the cockpit area is also eliminated by the use of synchro systems.

OPERATION OF THE SYNCHRONOUS SYSTEMS.

There are several different types of synchro systems—some operate on a-c and others on d-c. You will hear the systems called different names, depending on the company which manufactures them—some of the common trade names are: Autosyn, (Eclipse-Pioneer), Selsyn, (General Electric), Learsyn, (Lear), Synchrotel, (Kollsman) and Magnesyn, (Eclipse-Pioneer). We will discuss a typical a-c synchro in some detail, and mention a few features of a common d-c synchro.

A typical a-c synchro system has three units—a gage mechanism, a transmitter, and an indicator—with a

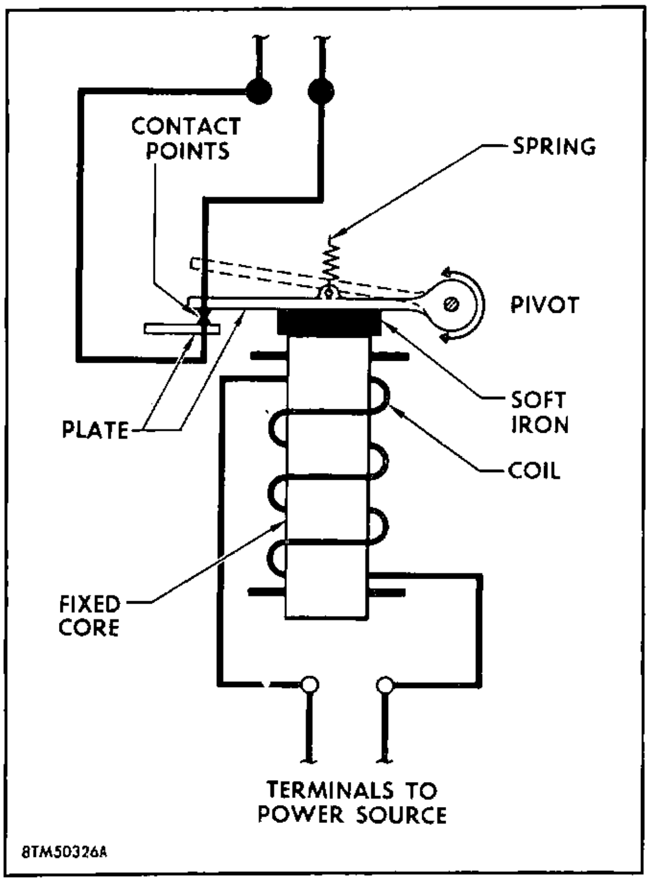

Figure 1-14. A Simple Relay

power supply that is usually a common current source used for other devices as well. The gage mechanism, which measures the movement, flow, or whatever is to be measured, is coupled directly to the transmitter synchro. The indicating synchro, sometimes called the *receiver synchro,* is coupled directly to the indicator pointer.

The transmitting and the receiving synchros are essentially the same. Both contain a stator and a rotor (armature) as shown in figure 1-16. You may recognize a similarity between these units and an ordinary synchronous electrical motor, in fact, the synchro transmitter and receiver frequently are called motors. They are not true motors, however, because they do not rotate continuously or furnish power.

Notice that the rotors are connected in parallel and receive power from the a-c power supply. The stators are also connected in parallel with each other. When current flows through the circuit, identical magnetic fields are set up by the current in the rotors of both units. These magnetic fields are at right angles to the flow of current through the windings of the rotors. When the gage mechanism of the transmitter positions the transmitter rotor, definite voltages are induced in each of the three windings of the stator—the voltage

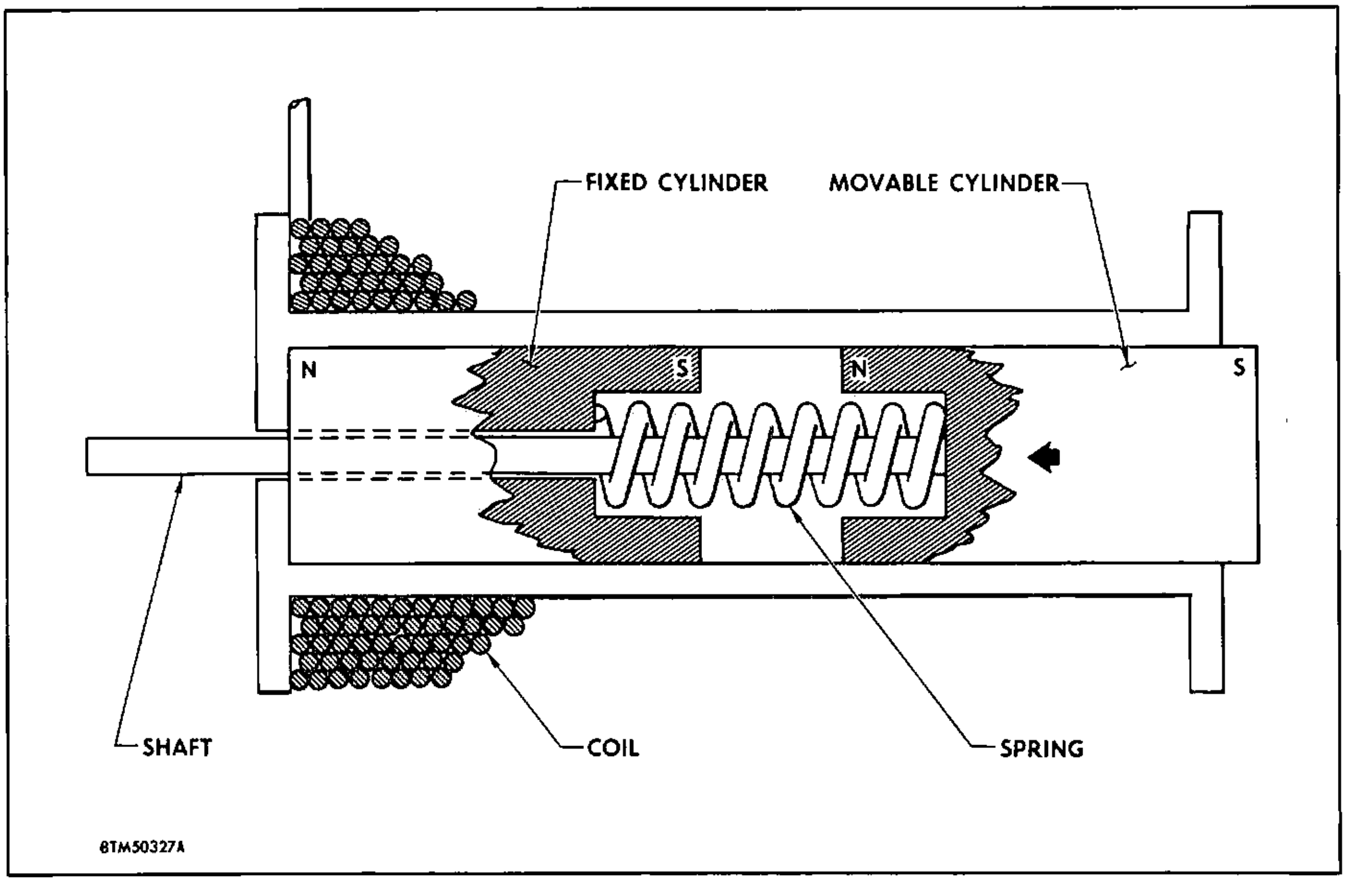

Figure 1-15. Magnetism and the Solenoid

depends on the number of magnetic lines of force cutting each winding. These fixed values correspond to only one position of the rotor. Any other position of the rotor will have another fixed set of voltage values.

The induced voltages of the stator windings of the transmitter will be duplicated in the stator windings of the indicator, because the three stator windings of both units are connected in parallel. The same induced voltage values will also be found in the indicator rotor. Since the indicator rotor is free to rotate, it will take the same position ("null" position) as the transmitter rotor. It remains in that position until the movement of the transmitter rotor changes the values of the voltages in the stator windings. Through this method, the mechanical movement at the place of measurement is transmitted electrically to the pointer of the indicator mounted on the instrument panel in the cockpit. Thus, the synchro system is a simple method of indicating in the cockpit what a direct reading instrument would show at the source of the measurement. A-C synchros are usually operated by 26-volt, 400-cycle current, although some use other voltages and frequencies.

Now take a look at figure 1-17, showing the circuit of a d-c system. Instead of an armature or rotor, this synchro transmitter uses contact arms. The positioning mechanism rotates these arms over a resistance coil that is being supplied current from the airplane d-c power source (battery and generator system). The indicator unit of the system contains three windings and a magnetized armature or rotor which has a north and south pole. Note that the windings receive current from three equally-spaced taps from the transmitter resistance; therefore, the magnitude and direction of the current in the indicator windings depend on the position of the transmitter contacts.

You will recall from our discussion of electromagnetism that current flowing through a coil of wire produces a magnetic field having strength and direction. It is apparent, then, how the polarized rotor is turned to correspond with the position of the transmitter contact. Synchro systems, whatever the type or function, are not necessarily limited to a single transmitter. You will learn about some multiple synchro instruments which are used in the F-102A.

PITOT-STATIC SYSTEM.

The pitot-static system is a tubing system that brings air pressure from the outside of the airplane into various instruments and controls. It furnishes two kinds of air pressure: pitot pressure (which is the impact pressure of the air against the moving airplane) and

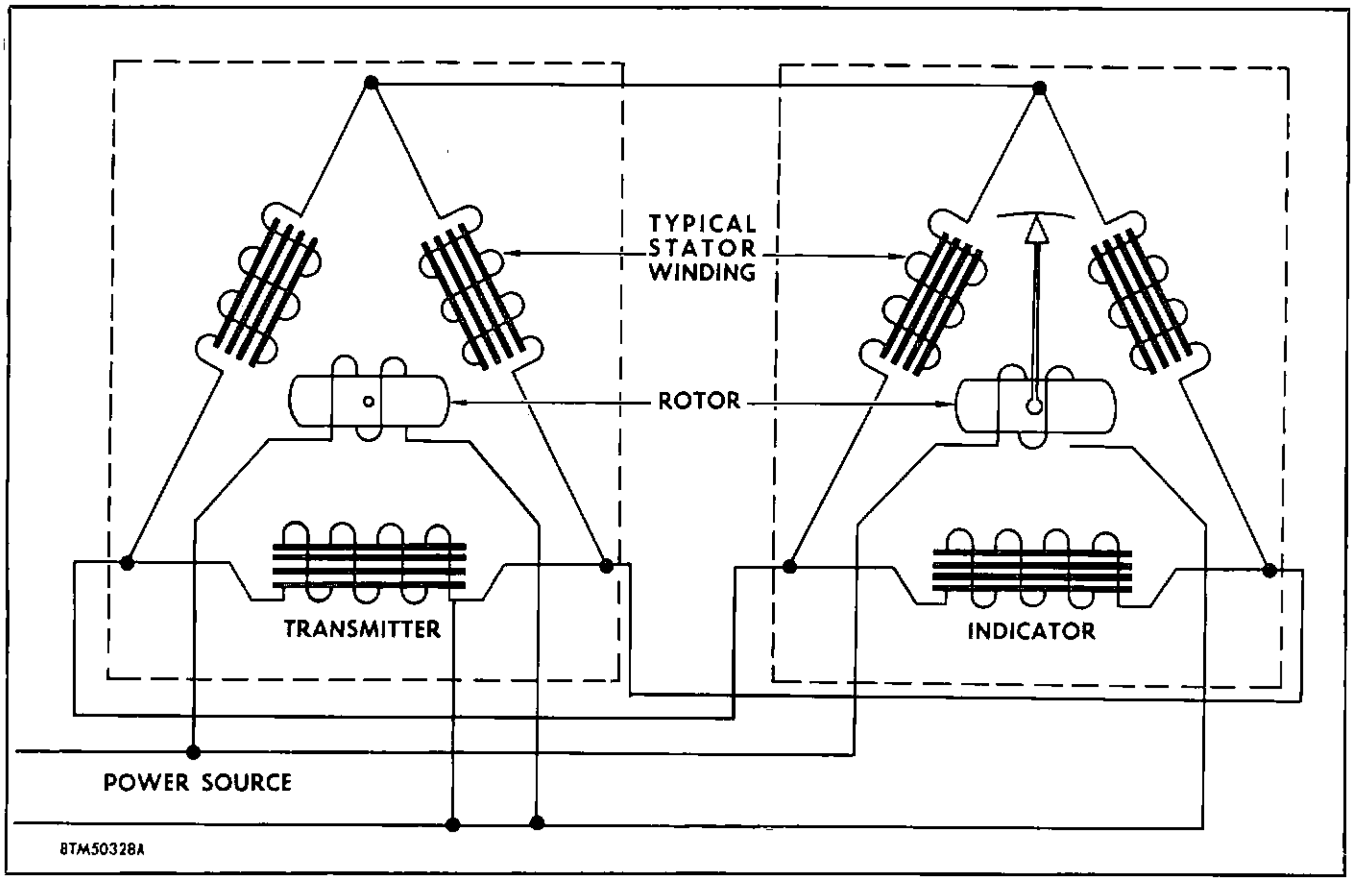

Figure 1-16. Circuit of an A-C Synchro

static pressure (which is the normal atmospheric pressure surrounding the airplane).

Pitot and static pressures are taken into the system through the pitot-static tube (14). In the F-102A airplanes, this tube is mounted on the end of the nose probe (airspeed boom). Figure 1-18 shows how it connects to the airplane. The tubing (11) which carries the air from the boom through the radome is made of nylon material. Aft of the radome, aluminum and rubber tubes are used.

Impact (pitot) pressure enters the pitot-static tube through an opening in its tip, (1) and, as you know, this pressure increases as the speed of the airplane increases. Static pressure enters the system through small parts (3) in the outer surface of the pitot-static tube. This pressure is unaffected by changes of speed—it is actually still air pressure—but varies with altitude changes; you will see why in the next chapter.

Since the pitot-static tube (14) is mounted outside the airplane, it is subject to icing conditions. To keep it warm and prevent ice from forming on it, the tube is equipped with a heating element of resistance wire, operated by a switch in the cockpit. This switch should not be turned on while the airplane is on the ground, except for very brief periods for testing. The heater is connected to the airplane 28-volt d-c system. You can see the male electrical plug (5) projecting from the tube in figure 1-18—the wires that connect to it (8) are carried within the boom (11).

Icing is not the only condition which affects the operation of the pitot-static system. It is obvious that any open tube mounted on the front of a moving airplane will pick up moisture, dust, and possibly other foreign matter. To prevent moisture from entering the instruments connected to the pitot-static system, you will find several drain points (2) in the system. One drain hole is located in the bottom of the pitot-static tube, and drain taps are included in each of the two lines leading from the tube. If the lines become constricted by dirt or other foreign matter, it is necessary to "purge" them in accordance with the appropriate technical orders.

Now let's see what the pitot-static system consists of. Figure 1-19 shows a schematic that includes the pressure or differential pressure measuring instruments operated by this system. These instruments operate on the diaphragm or aneroid principles which you read about earlier in this chapter. Notice the instruments are the airspeed and Mach number indicator, the altimeter, the rate-of-climb indicator, the engine pressure ratio indicator, and the airspeed switch of the

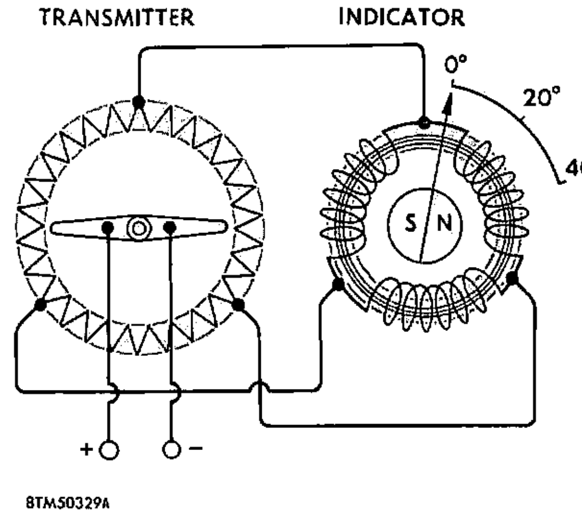

Figure 1-17. Circuit of a D-C Synchro

landing gear warning system. In addition to these instruments, a pressure transmitter, differential pressure transmitter, altitude barometric control, up elevator signal transducer, and elevator automatic trim transducer are also connected to the pitot-static lines.

The pitot-static system is a simple system that is usually trouble free. However, keep in mind that it carries air pressures which are measured; thus there can be no leaks in the system. If there are any leaks, all or part of the instruments connected to the system will indicate incorrectly. Whenever a pilot reports malfunctioning of more than one of these instruments at the same time, you should test the system according to the procedures outlined in the applicable technical orders.

SLAVED GYRO COMPASS SYSTEM.

The primary source of magnetic directional information for the F-102A pilot is the slaved gyro compass system. This system detects the horizontal component of the earth's lines of force. How the airplane is heading in relation to these lines of force determines the indications given by the system. To provide stable and reliable indications during normal maneuvers and rough air conditions, a gyro is "slaved" to magnetic north. Movement of the airplane around the gyro is then transmitted electrically to the indicating mechanism.

Indications from the slaved gyro magnetic compass system are combined with navigational information

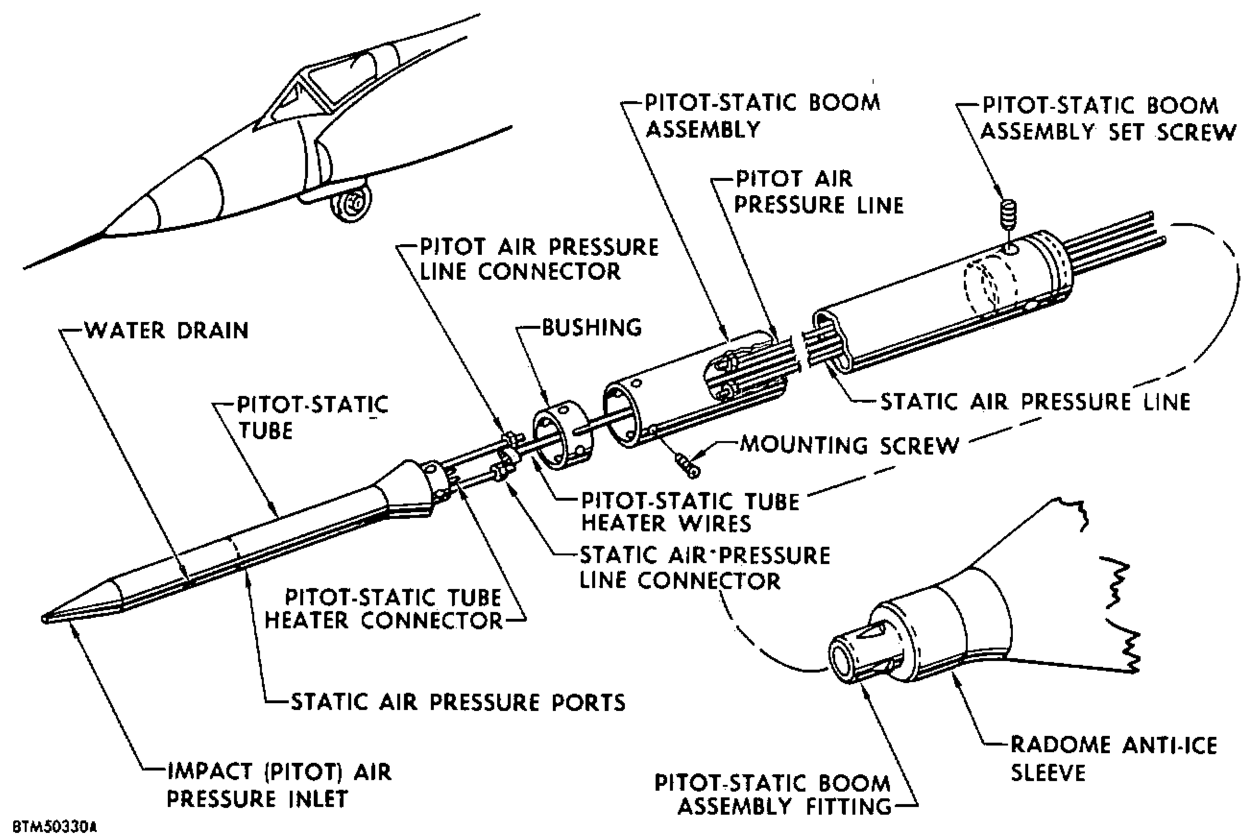

Figure 1-18. Pitot-Static Tube Installation

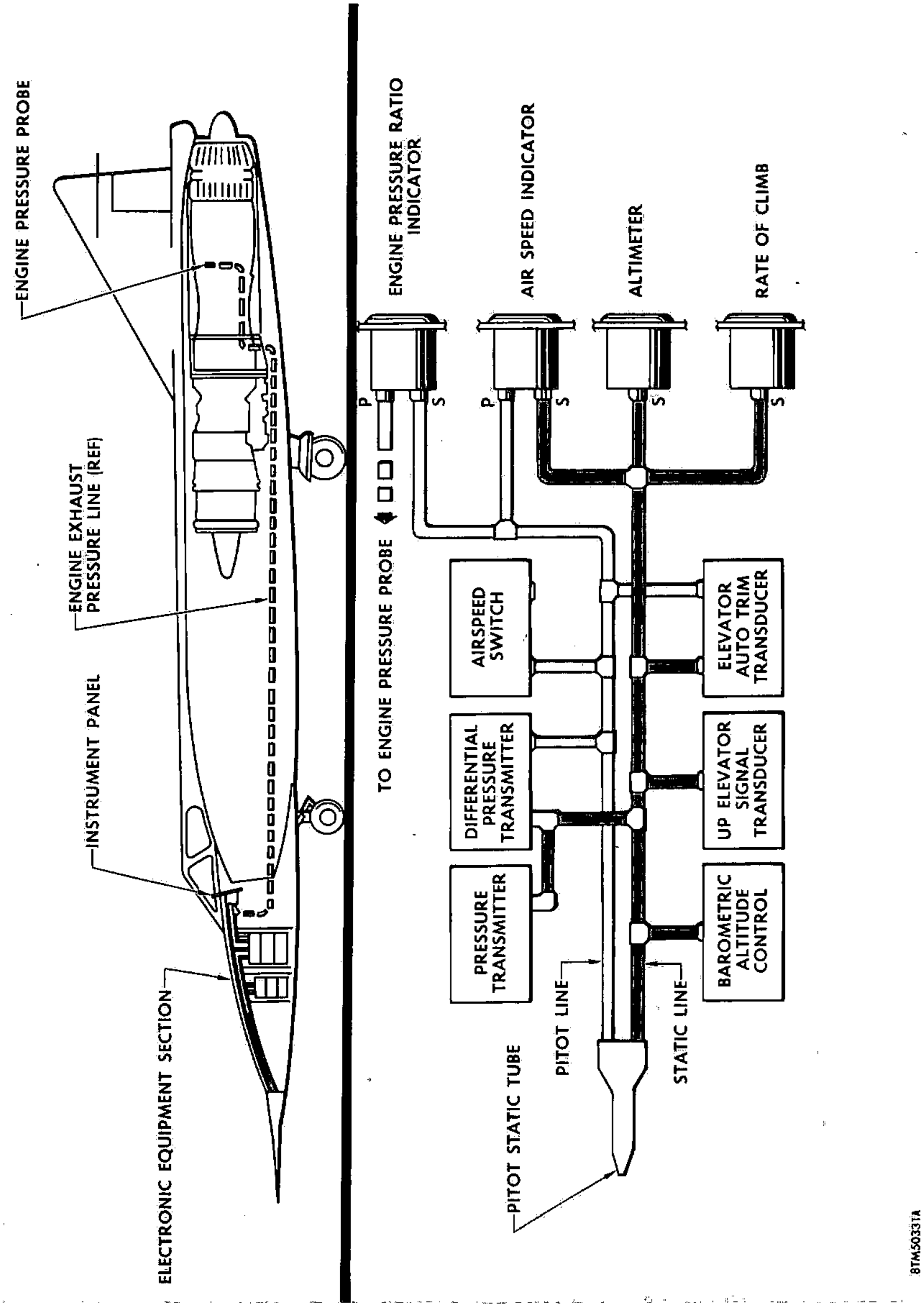

Figure 1-19. Pitot-Static System, Schematic

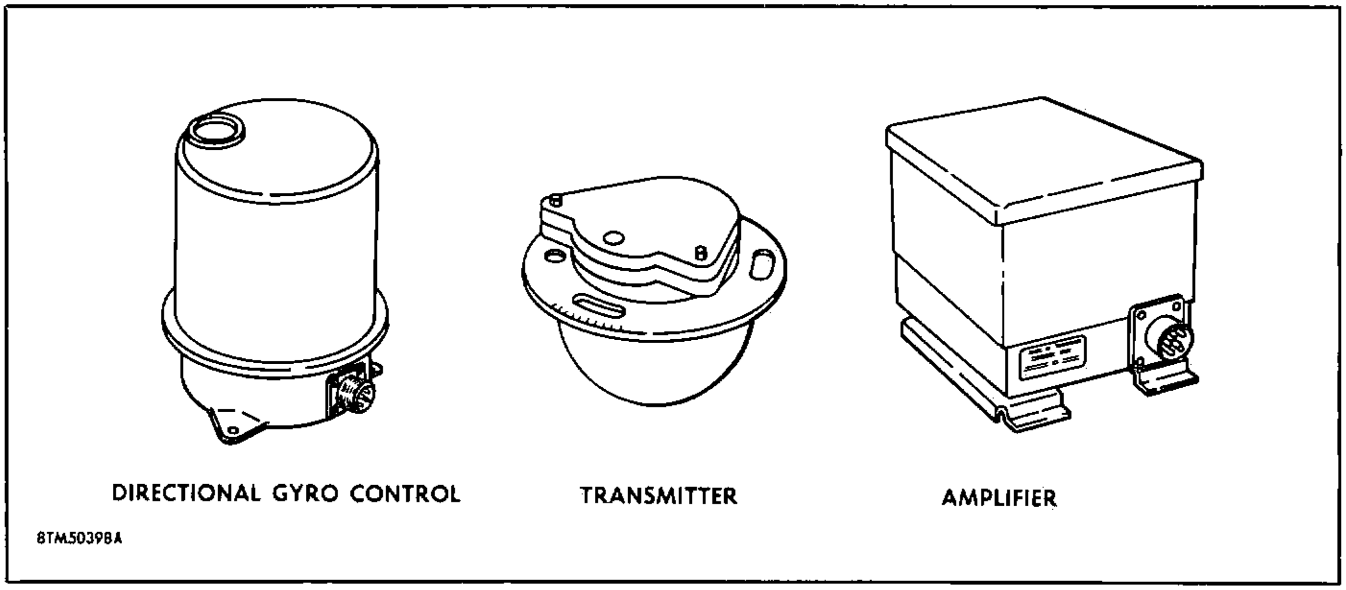

Figure 1-20. Components of Slaved Gyro Magnetic Compass System

from other systems. Therefore, we will discuss the actual indicating mechanism in the next chapter along with the other navigational instruments. The rest of the components of the system which provide information to the indicator are discussed here — they are the transmitter, the directional gyro control, and the amplifier. Figure 1-20 shows these units as they appear in the airplane.

THE REMOTE COMPASS TRANSMITTER.

Direction sensing for the slaved gyro compass system is accomplished by the transmitter. It is mounted in the left wing of the F-102A — just forward of the elevon — where magnetic disturbance is at a minimum. Markings on the mounting flange serve as a reference for installation. The transmitter access door is labeled so you won't have any difficulty in locating it.

The remote compass transmitter consists primarily of a fundamental flux valve unit suspended on a universal joint and surrounded by damping fluid. The universal joint permits the unit to hang like a pendulum, thus remaining in a horizontal position within its limits of movement (about 30° of pitch or roll). By surrounding the flux valve with damping fluid, excessive oscillation is prevented.

The fundamental flux valve unit is mainly a three-spoke wheel or spider made of laminated metal. Notice in figure 1-21 that an excitation coil (5) is placed in the center of the spider (10). This is the primary (exciting) coil. When the unit is in operation, this coil is energized with a 23.5-volt, 400-cycle alternating current. A secondary (pick-up) coil (8) encircles each of the spider legs (10) to pick up induced current, and each leg terminates in a slot between laminated strips called flux collectors.

To understand the flux valve you must remember the basic principles of magnetism which you read about earlier in this chapter. Both electromagnetism and the natural magnetism of the earth's flux are involved in its operation. Remember also that alternating current flows first in one direction and then in the other, reversing its polarity in the process. For example, a 400-cycle alternating current builds up to a maximum positive value, decreases to zero, builds up to a maximum negative value, and decreases to zero—all this happens at the rate of 400 times per second. Here's how these principles apply to the flux valve.

Because the laminated metal used in the flux valve is highly permeable (offers low reluctance to magnetic lines of force), the earth's flux enters the spider legs. As the positive half-cycle of the exciting current builds up in the primary coil, (5) it sets up a magnetic field which expels the earth's flux. The expelled flux cuts the pickup coils (8) on the spider legs (10), inducing current in each one. As the positive half-cycle of the exciting current decays, the earth's flux—because of the higher permeability of the legs compared to the surrounding air—returns to the legs. Thus, it again cuts the pickup coils and again induces current, but of the opposite polarity.

You can see, then, that a half-cycle of alternating current produces one full cycle of induced current. The negative half-cycle of the exciting current reproduces the action of the positive half. Since there are 400 cycles to the exciting current, the induced current is 800 cycles per second.

The magnitude (strength) of the induced current in each pickup coil varies according to the number of magnetic lines cutting it. Now, notice in figure 1-22 that the number of lines cutting each leg varies

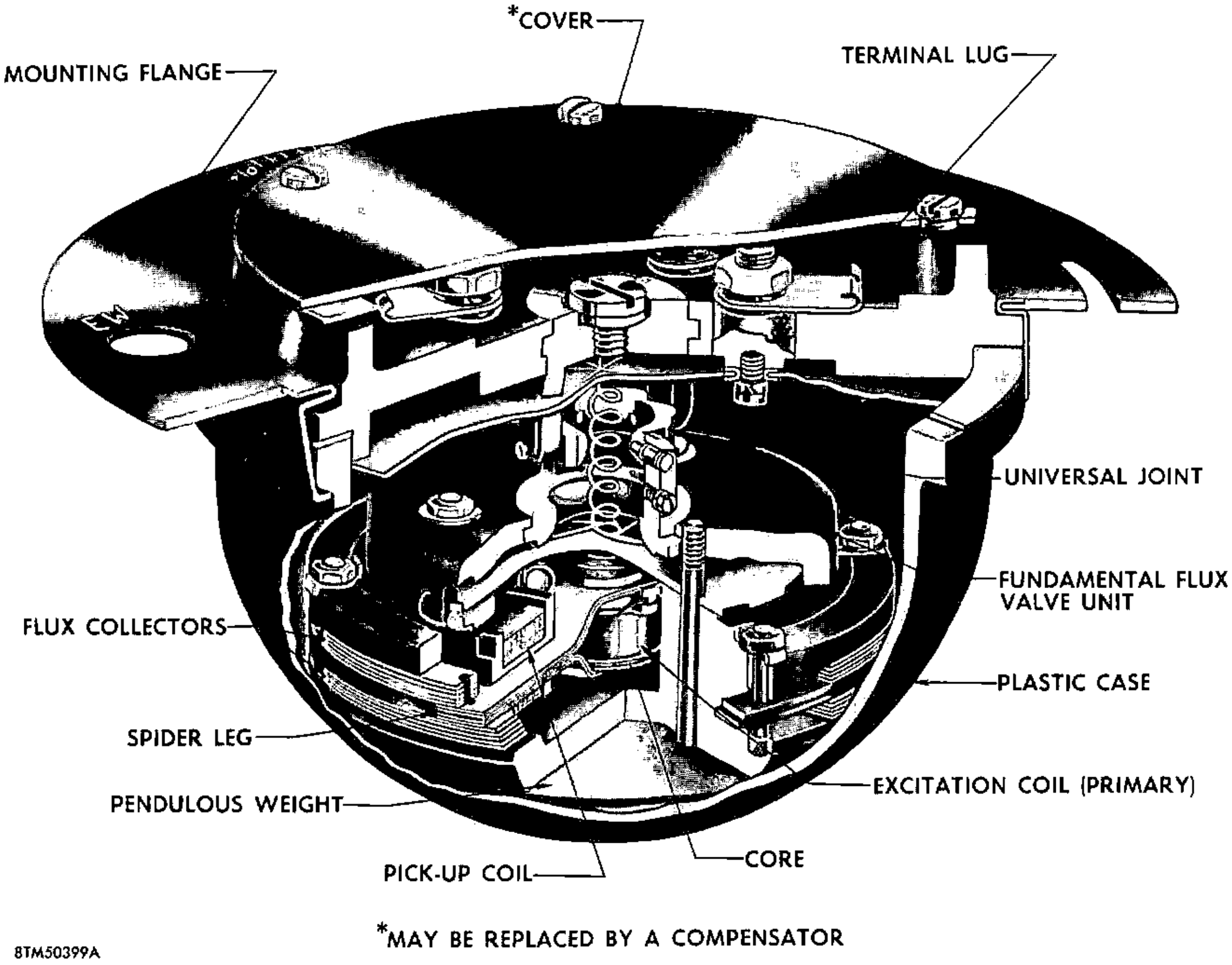

Figure 1-21. Remote Compass Transmitter with Compensator Removed

according to the position of the leg with regard to the earth's magnetic lines of force. Thus, the magnitude of the current in each pickup coil is determined by the heading of the airplane. From these coils, the induced currents are carried to the stator of a flux valve synchro in the directional gyro control. You will learn what function this synchro performs later in the chapter.

Earlier in this chapter we discussed the need for some means of compensating magnetic compass instruments. The compensating mechanism for the slaved gyro compass system, used in the F-102A, is mounted on top of the transmitter. Four small permanent magnets within the compensator are adjustable by the two screws protruding from the cover. You must never attempt to compensate the system without consulting the applicable technical orders.

THE DIRECTIONAL GYRO CONTROL.

This mechanism provides the stabilized reference for the compass system. The entire unit is enclosed in a sealed case and is kept dry by a silica-gel capsule. A window in the top of the case permits you to check the scale which shows the compass heading. This scale can be used only for ground checking of the system—the pilot can't see it because the unit is mounted in the nose wheel well. Figure 1-23 shows the directional gyro control with the cover removed.

The directional gyro control is built around a horizontal gyro—that is, one which is mounted with the spin axis in a horizontal position relative to the earth's surface. The gyro is electrically driven by 115-volt, 400-cycle, 3-phase alternating current, and spins clockwise as viewed from the housing cover. The rotor is

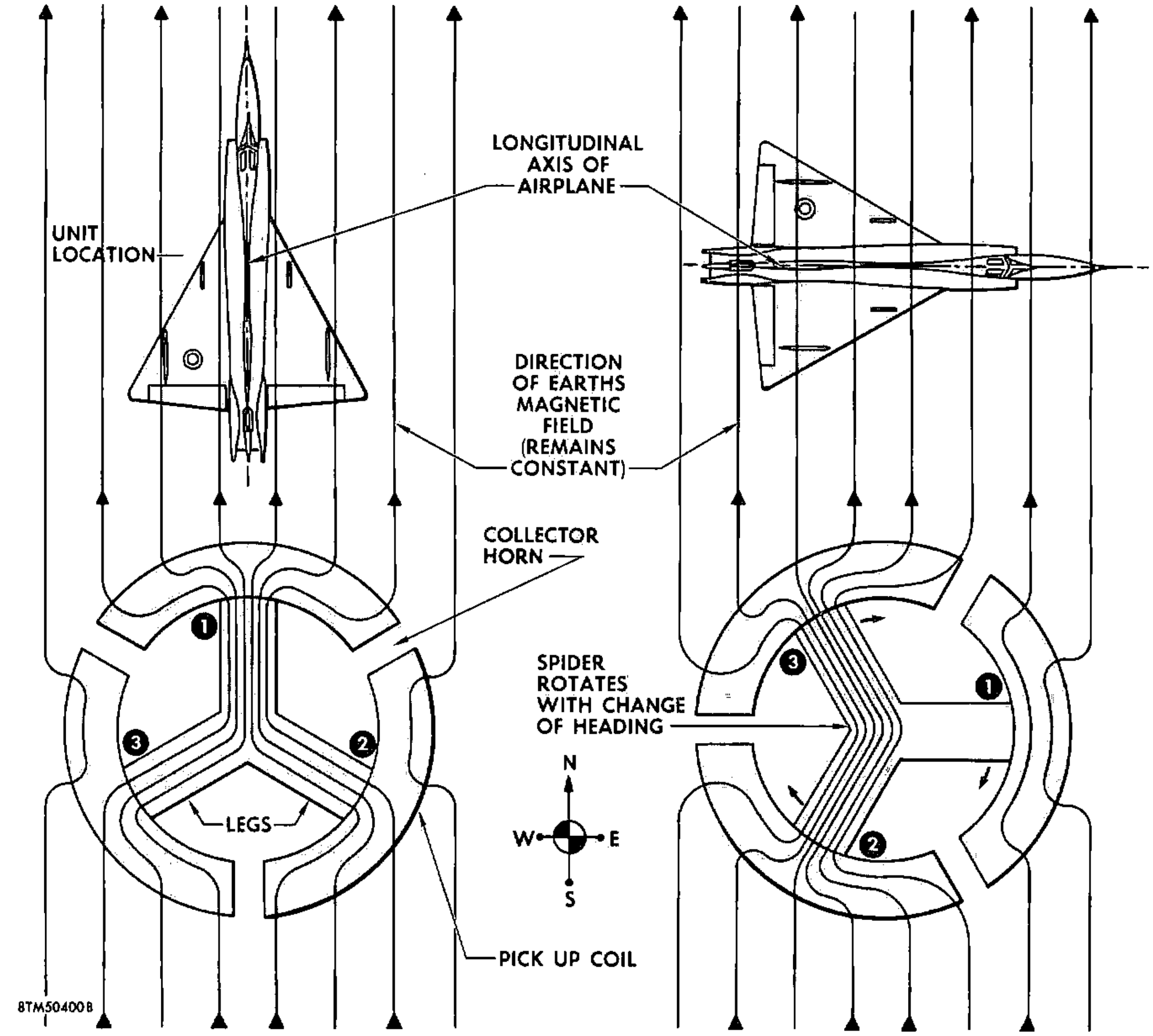

Figure 1-22. Effect of Aircraft Heading on Magnitude of Induced Current

the "squirrel cage" of an induction motor. In other words, it rotates around the fixed, current-carrying stator because of induced current from the stator.

The gyro housing is the inner (horizontal) gimbal (7), and is free to rotate ±80° on its bearings in the outer (vertical) gimbal ring (14). The vertical ring is supported by upper and lower pivots in the frame (19), and is free to rotate (or the frame can rotate about the ring) a full 360°. Thus, the spin axis of the gyro tends to remain in a horizontal position and the vertical gimbal is stabilized in azimuth.

Now take a look at the operational schematic in figure 1-24, notice that a leveling torque motor stator, slaving torque motor rotor, and two synchro rotors are attached to the vertical ring. The scale (not shown in this illustration) is also fixed to the vertical ring. The stator of the slaving torque motor attaches to the gyro housing and is free to tilt with the housing, and the rotor of the leveling torque motor attaches to the upper part of the frame. It will help you to understand how all these devices operate if you keep in mind two things: everything connected to the vertical gimbal ring can rotate all the way around within the case and everything connected to the inner gimbal (gyro housing) can also tilt up or down 80° from its level position.

In our discussion of the transmitter, we mentioned how varying the positions of the spider legs resulted in different signals to the flux valve synchro stator in the gyro control. Let's see how these signals keep the gyro "slaved" to the earth's magnetic meridian. You will recall from our discussion of synchros that any set of voltage values will result in a certain positional relationship between the rotor and stator. The flux valve synchro is in its null position any time the gyro spin axis is aligned with the lines of force of the earth. In other words, the rotor is properly aligned with the magnetic field set up by the current in the stator.

Suppose now that something has caused the gyro to become misaligned with the earth's field, as detected by the flux valve in the transmitter. The flux valve synchro rotor is no longer in the proper position in relation to the magnetic lines of force from the stator. Thus, voltage is generated in the rotor. This voltage energizes the slaving torque motor, causing the gyro to precess about its vertical axis. When the gyro is correctly aligned, the rotor of the flux valve synchro is again in its null position and the "error signal" ceases.

In actual operation, once the gyro is erected and aligned it tends to remain that way by its own properties of rigidity. When the airplane turns, the case of the directional gyro control actually turns around the gyro assembly. The changing position of the flux valve alters the voltages in the three windings of the synchro stator, but the stator—connected to the directional gyro control case—turns around the stabilized rotor at the same rate. Hence, the synchro tends to remain in the null condition throughout the turn.

In order for the action of the slaving torque motor to result in the desired precession, it is necessary that the gyro be properly erected. A liquid level switch, attached to the gyro housing, operates the leveling torque motor. This motor serves to keep the gyro spin axis in the horizontal position unless the airplane exceeds the 80° limits during maneuvers. If these limits are exceeded, the torque motor will level the gyro again in a few minutes.

Like the motor which spins the gyro rotor, the torque motors of the directional gyro control are induction-type motors. At first glance you might think that figure 1-24 shows the positions of these two torque motors reversed. It looks as though the slaving motor tilts the gyro on its lateral axis, and that the leveling motor turns the vertical axis. If the gyro rotor was motionless, thats how it would work. Actually, the torque motors don't really turn the gyro. They only apply a precessional force. If you will think back about the principles of gyroscopic precession which we discussed earlier in this chapter, you will understand why they are located as they are.

A switch located on the upper left corner of the main instrument panel permits the F-102A pilot to remove all power from the slaving motor. It is used when flying in polar regions where the "vertical component" of the magnetic flux exceeds 84°. You will recall from our discussion of magnetism that a compass will be affected by "dip" in these regions. When the slaving

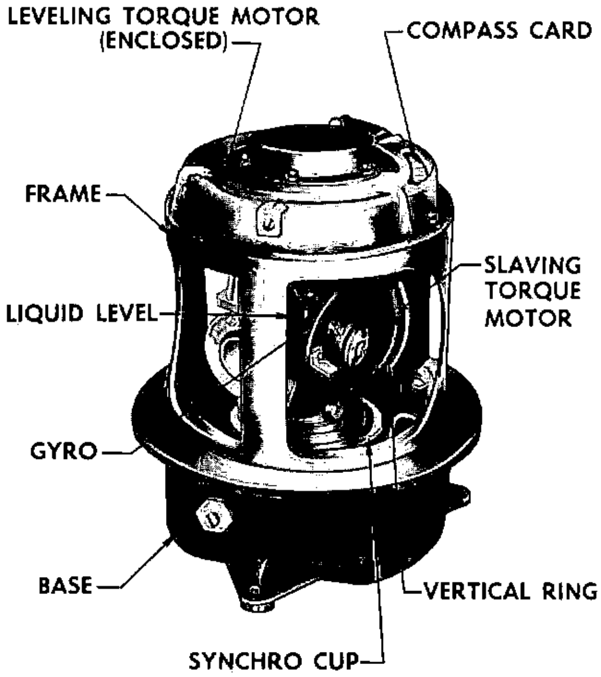

Figure 1-23. Directional Gyro Control—Cover Removed

switch is in "deslave," the system may be used as a directional gyro, and is also reliable as a compass until the "natural" precessive forces—bearing drag and the rotation of the earth—have had time to precess the gyro.

We have discussed the manner in which the heading of the F-102A is sensed by the transmitter, and how the gyro is "slaved" to the earth's magnetic field. Now let's see what happens within the directional gyro control to make use of this stable reference.

Refer back to figure 1-24 again, and notice that the heading synchro is mounted in the same way as the flux valve synchro. Thus, the rotor of the heading synchro turns when the vertical ring rotates, and therefore, is "slaved" along with the gyro. The stator —being attached to the control case—turns with the airplane. An alternating current of 115-volts, 400-cycles, energizes the rotor which in turn causes an induced current in the coils of the stator. The relative positions of the rotor and stator—established by the heading of the airplane in relation to magnetic north —determines the current flow in the three stator windings. You can see, then, that this synchro is a transmitting synchro.

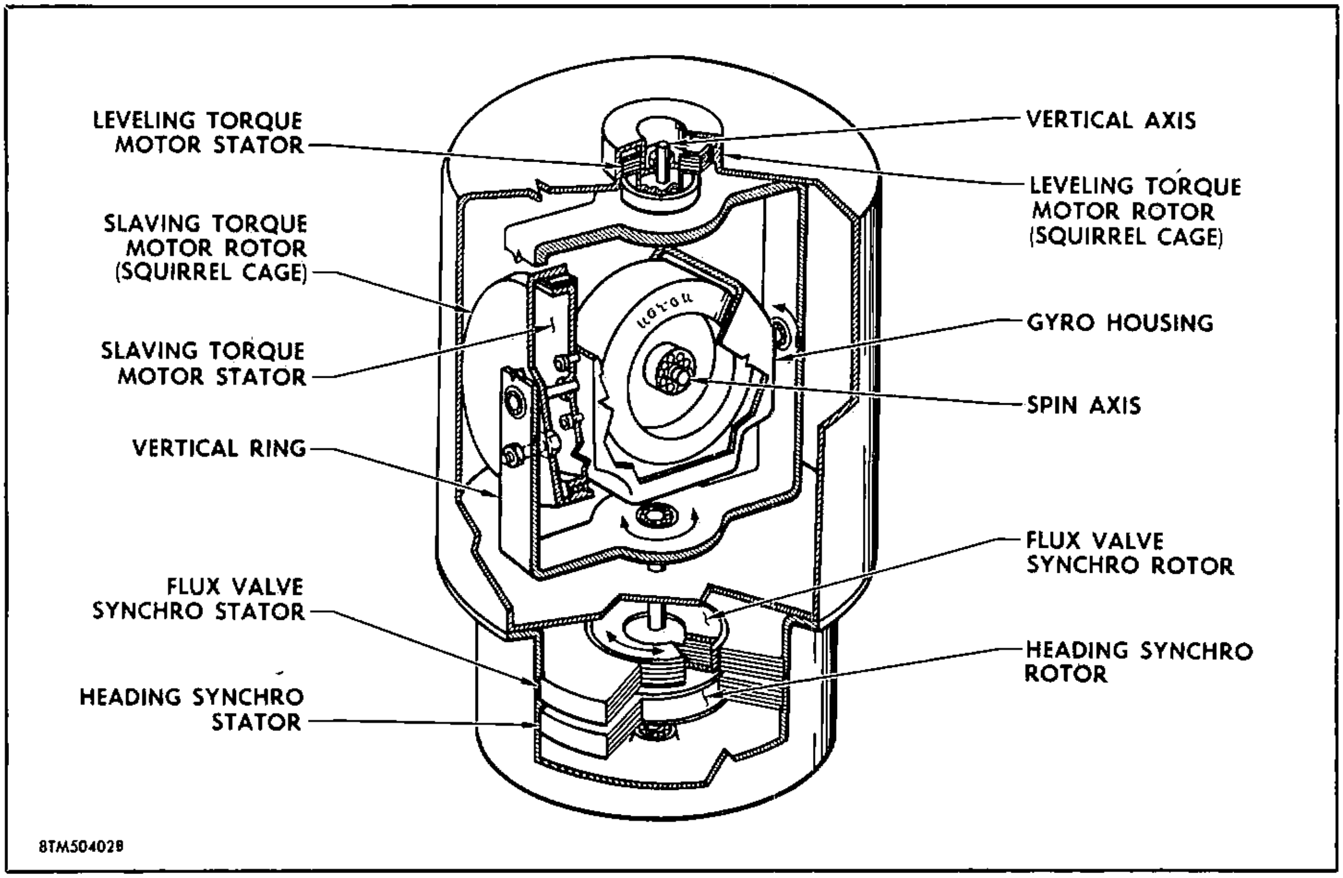

Figure 1-24. Operational Schematic of Directional Gyro Control

As was mentioned earlier, the indicating mechanism which uses the information provided by the slaved gyro magnetic compass system is a combination instrument. The signals from the heading synchro are transmitted to the radio indicator control (bearing converter indicator) in the air-conditioning compartment, and from there to navigation instruments in the cockpit. You will learn about those instruments in the next chapter.

THE AMPLIFIER.

The electrical power requirements of the slaved magnetic compass system are supplied through the amplifier. In the F-102A airplanes the amplifier is located in the nose wheel well compartment, beside the gyro control unit. Figure 1-25 shows this unit with the cover removed. We will discuss the function of this unit without going into detail on its electronic operation.

Two major functions are performed by the amplifier. First, this unit amplifies and detects the phase of the error signals from the flux valve synchro rotor in the directional gyro control. Thus, it determines the direction of the error, and controls the amount and direction of torque produced by the slaving torque motor.

The second major function of the amplifier is to increase the amount of voltage applied to both the slaving torque motor and the leveling torque motor when the system is first turned on. When power is first applied to the system, the increased voltage to the torque motors will cause the gyro to erect at the rate of about 60° per minute, until the gyro is within 5° of its erected and slaved position. As the gyro comes within 5°, the torque is reduced steadily, and ceases completely when the gyro is properly positioned.

After two or three minutes, from the time the power is first applied to the slaved gyro system, a time delay relay in the amplifier causes the voltage to be reduced. The gyro then erects at the rate of 3 to 6° per minute. A light on the upper corner of the main instrument panel, near the deslave switch, glows when the fast erection system is in operation. If this light does not glow when power is first applied to the airplane (both a-c and d-c), you should check the lamp. If the lamp is not burned out, the fault probably is in the time delay mechanism in the amplifier. You should consult T.O. 1F-102A-2-9 before attempting any repairs.

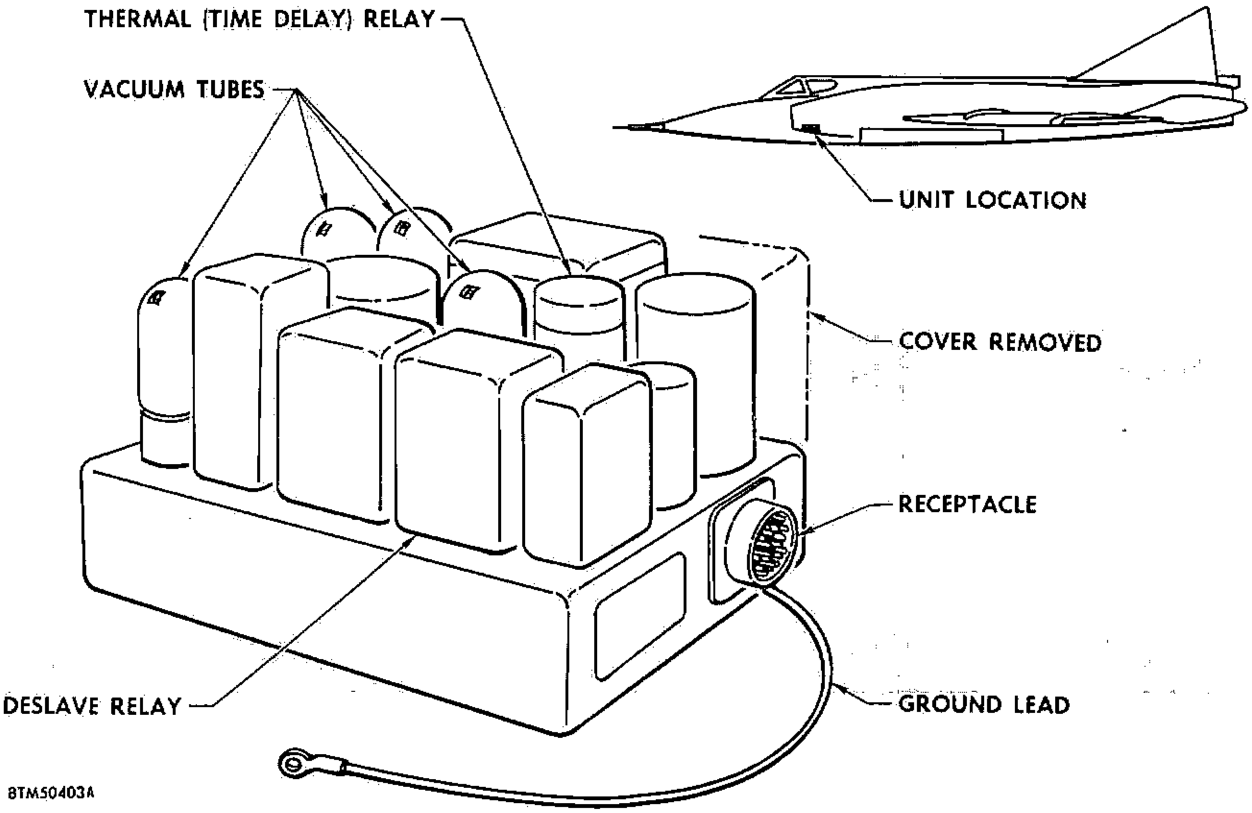

Figure 1-25. Slaved Gyro Magnetic Compass Amplifier

SUMMARY.

It is difficult to say that certain instruments are more important than others, so the design of an instrument panel always involves some compromises. However, those instruments which perform similar functions are placed close together in logical groups. The major groupings which you will read about in the succeeding chapters are: the flight and navigation instruments; the engine instruments; the power supply instruments; and master and fire warning indicators. An additional group is covered in the last chapter. It includes the instruments which may not logically be considered part of the other groups.

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Chapter II

FLIGHT AND NAVIGATION INSTRUMENTS

Contents *Page*

The instruments most frequently referred to on any pilot's panel are those in the flight and navigation group. In the F-102A, most of these instruments are grouped directly in the center of the panel. Figure 2-1 emphasizes this arrangement.

Flight instruments indicate the ariplane movement in relation to the air, and the airplane attitude in relation to the ground. The flight instruments used in the F-102A are: the *airspeed and Mach number indicator*, the *sensitive pressure altimeter*, the *attitude indicator* or vertical gyro indicating system, the *rate-of-climb indicator*, and the *turn-and-slip indicator*. You can identify all of these instruments in figure 2-1.

Navigation instruments give the pilot information regarding the airplane position and direction of movement. The airplane position indication is in relation to the navigational facilities on the ground. Its direction of flight indication is relative to navigational facilities and magnetic north. The following navigation instruments are used in the F-102A: the *clock*, the *deviation indicator*, the *magnetic standby compass*, and the *radio magnetic course indicator*.

You will notice that the magnetic standby compass and the radio indicator control (which operates in conjunction with the radio magnetic course indicator) are not shown on the pilot's instrument panel. Since the pilot does not refer to the radio indicator control, it is installed in the upper electronics compartment behind the cockpit. The standby compass is located in the peak of the canopy.

The order in which the flight and navigation instruments are discussed does not imply any difference in importance—they are all important to the pilots who fly the F-102A. However, to make it easier for you to learn and remember these instruments, they are arranged according to similarity in function and mechanical operation.

AIRSPEED AND MACH NUMBER INDICATOR.

The speed of the F-102A airplane is indicated both by *airspeed* in knots and by *Mach number*. These two types of speed indication are combined in a single instrument, the airspeed and Mach number indicator,

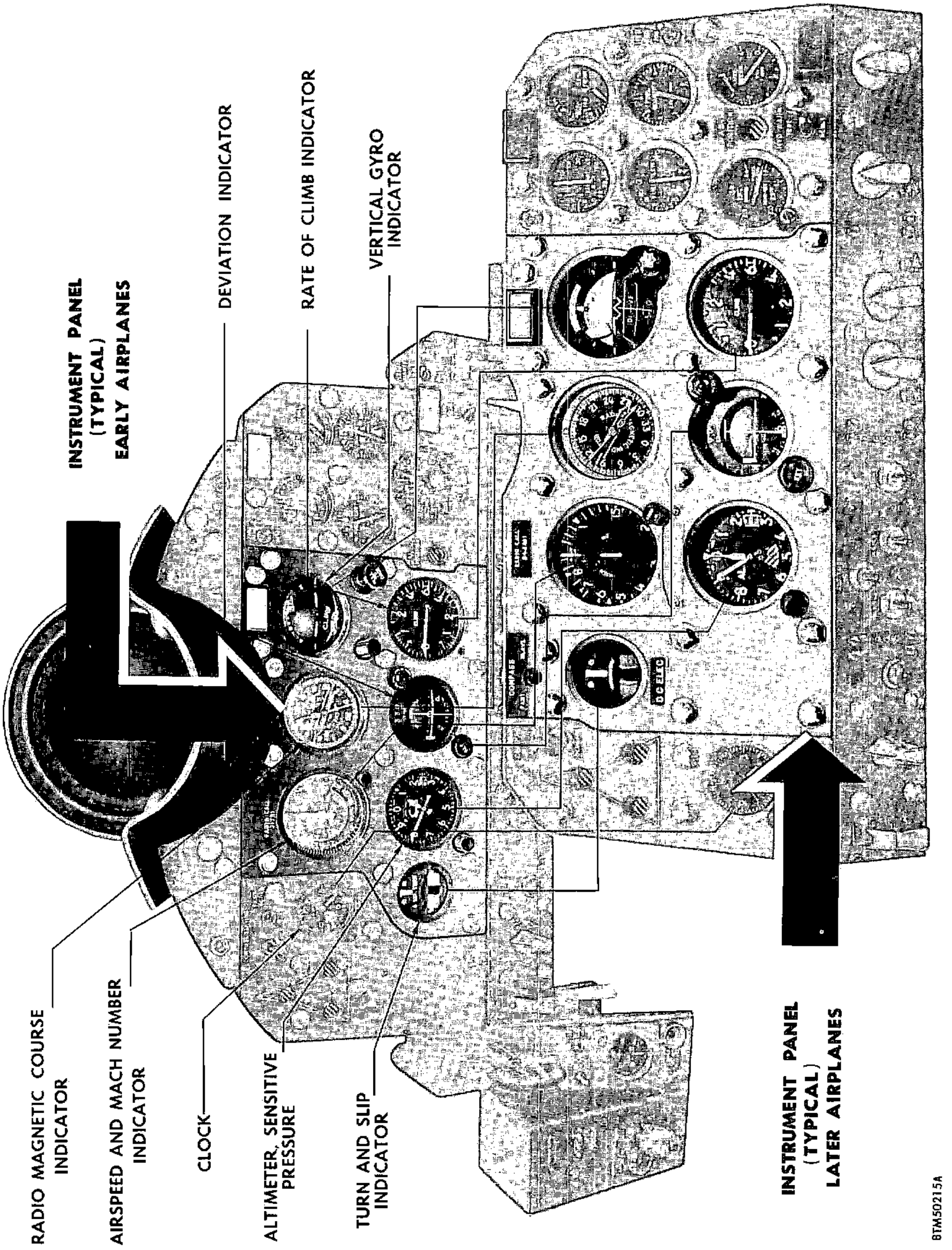


Figure 2-1. Flight and Navigation Instruments, Typical

which is shown in figure 2-2. This indicator is a cylindrical, airtight instrument that is vented to atmospheric pressure through the static section of the pitot-static system. It consists of a fixed airspeed dial, a rotating Mach number dial, three pointers, three diaphragms and actuating linkage, and two fittings on the aft side of the case for connections to the pitot-static system lines. The knob you see on the front of the instrument is the only control which the pilot has over its operation.

This indicator is actually a sensitive differential-pressure instrument. It measures the difference between pitot (impact) pressure and static (atmospheric) pressure, and indicates this difference as airspeed in knots. It also utilizes the change in air pressure that is caused by increasing or decreasing altitude. This change in pressure rotates the Mach scale to indicate the proper Mach number. Before we discuss how this instrument works, a brief review of airspeed and Mach number is given to refresh your memory on these two types of speed indication.

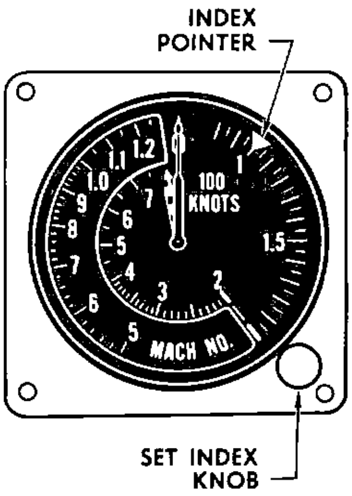


Figure 2-2. Airspeed and Mach Number Indicator

AIRSPEED.

Speed is the rate of motion or distance traveled per unit of time. Airspeed is speed measured in relation to the air mass. Each type of aircraft has specific airspeed requirements for takeoff, landing, and various aerial maneuvers. In addition, there are maximum speed limitations imposed on every airplane for certain conditions of flight, such as maximum safe dive speeds and maximum safe speeds with landing gear and flaps extended. You can easily see why a pilot must know his airspeed for the safe operation of his aircraft.

The rate of travel over the surface of the earth is known as groundspeed. An automobile speedometer indicates groundspeed because the tires are in contact with the pavement. To determine the groundspeed of an airplane isn't such a simple matter. Frequently a pilot computes his groundspeed to determine how far he can go in a given amount of time or with a certain amount of fuel. If he knows his airspeed, he can figure his groundspeed by allowing for the effect of wind on his aircraft. Another method is to clock the flight time between points on the ground which are a known distance apart.

The relation between airspeed and groundspeed is shown in figure 2-3. Note that the "wind component" is added to, or subtracted from, the airspeed—depending on whether it is a headwind or a tailwind—to obtain the groundspeed. In the case of a direct headwind (or tailwind), the total speed of the wind is added or subtracted. This is not true when the air mass is moving at an angle to the line of flight. The difference between airspeed and groundspeed will then depend on the angle at which the wind is blowing as well as its velocity. The point to remember from all this discussion is that the actual flight of an airplane—the lift, thrust, and drag factors involved—depends on its relation to the air. The arplane speed over the ground is important in the final accomplishment of its mission, rather than in its flight characteristics.

Airspeed indicators are calibrated in miles-per-hour or in knots. A knot is a unit of speed equivalent to one international nautical mile an hour. An international nautical mile is equivalent to 1.15079 statute miles. The airspeed indicator in the F-102A is calibrated in knots.

MACH NUMBER AND SONIC SPEED.

Aircraft that fly at speeds approaching or exceeding the speed of sound (sonic speed) also use another type of speed indication known as Mach number. Mach number is the ratio of the speed of an object to the speed of sound under identical atmospheric conditions. Mach number 1.0 is the unit of measurement of the speed at which sound travels. The speed of sound decreases with altitude. It varies from approximately 660 knots (761 mph) at sea level to 573 knots (660 mph) at 35,000 feet, and is correspondingly less at higher altitudes.

HOW THE INDICATOR WORKS.

In figure 2-2, airspeed is shown on the fixed dial (100 knots) by the larger of the two needles. The same needle indicates the corresponding Mach number on the rotating dial (Mach number) which is visible through a cutout in the fixed dial. As altitude increases, the Mach dial revolves counterclockwise. Thus, regardless of the airplane's altitude, Mach number 1.0

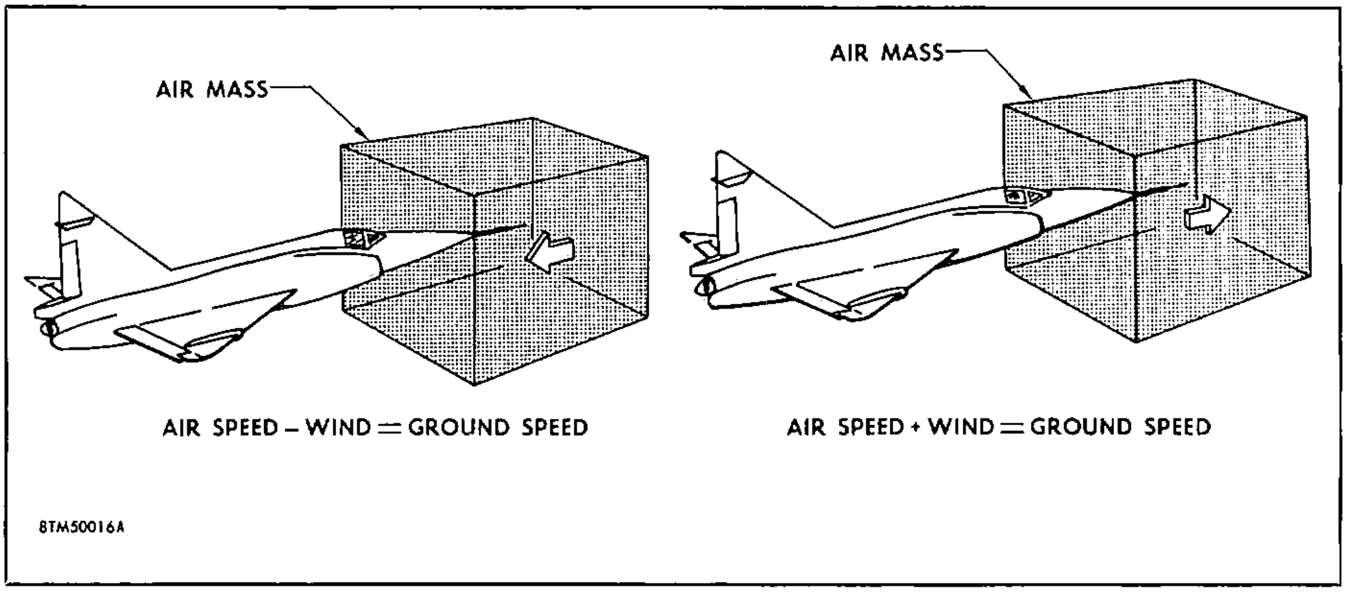


Figure 2-3. Airspeed versus Ground Speed

is always directly opposite the number on the knots scale that corresponds to the speed of sound.

The pilot may set the index pointer on the outer rim of the dial to show some special condition, such as take-off or landing speed. It is adjustable between 100 and 200 knots by the SET INDEX knob on the front of the instrument, and remains in the set position regardless of the airplane speed or altitude. The small needle is adjusted from the back of the case, and operates from an altitude mechanism to show a maximum allowable airspeed. However, in the F-102A this needle is not used and is therefore preset to its maximum position. Instead, the maximum permissible dive speed is indicated by a red line on the dial. A yellow line on the dial indicates the maximum speed at which the pilot can extend the landing gear.

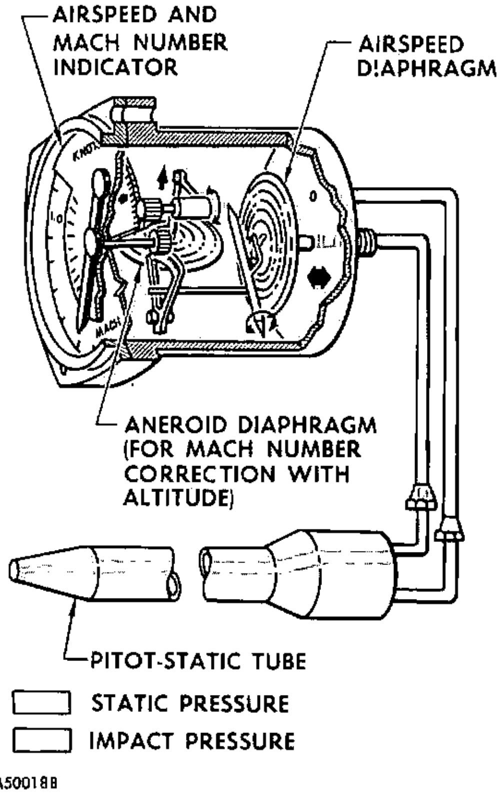
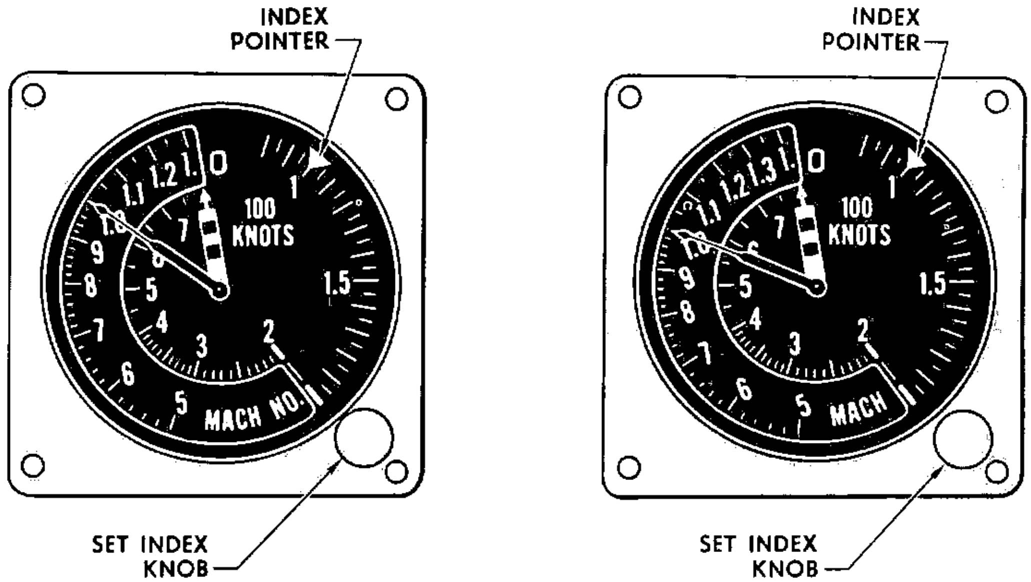


Figure 2-4. Airspeed and Mach Number Indicator Operational Diagram

Now let's take a closer look at the internal mechanism of the indicator. Figure 2-4 is simplified, but it shows the basic principles of operation. Impact pressure from the pitot tube enters the airspeed diaphragm. Atmospheric pressure from the static line is conveyed to the inside of the air-tight case. When the plane is motionless on the ground (and not headed into the wind), the pressure in the diaphragm is equal to the static pressure of the air within the case, and no reading appears. However, when the airplane is moving through the air, the impact pressure becomes greater than the static pressure. This pressure differential increases as the forward motion of the airplane increases and the diaphragm expands proportionately. The lever and gear mechanism transmits the diaphragm distortion to the larger pointer which moves clockwise around the dial. Thus, the distortion of the diaphragm, being a measure of the differential pressure, is also a measure of airspeed.

Pressure poitions the Mach number scale in a somewhat different manner. An aneroid-type diaphragm (similar to that which operates the altimeter) is used. The aneroid-type chamber is evacuated and sealed—



ALTITUDE: 15,000'
MACH NO.: 1
AIR SPEED: 625

ALTITUDE: 30,000'
MACH NO.: 1
AIR SPEED: 588

8TM51217A

Figure 2-5. Airspeed and Mach Number Indicator Readings at Various Altitudes

the airspeed diaphragm receives pressure from an external source. The aneroid-type chamber is surrounded by static air pressure—the same pressure acting on the outer surface of the airspeed diaphragm. Since atmospheric pressure reduces as altitude increases, the sealed Mach number diaphragm expands in a manner similar to a rising balloon. The mechanical linkage rotates the Mach number scale according to the amount of diaphragm expansion. The scale rotates counterclockwise so that Mach 1.0 lines up with the knots scale to indicate the speed of sound at that particular altitude. Thus, the long pointer indicates the airplane speed by Mach number and knots.

You learned previously that this instrument had three diaphragms. The third diaphragm, which is not shown on the illustration, is connected to the smaller needle on the dial. Since this needle is not used on the F-102A and is preset to its maximum position, we will not discuss it or how the third diaphragm affects its movement.

Note, in figure 2-5, the dial readings for airspeed and Mach number at 15,000 and 30,000 feet. You can see that Mach 1.0 is equal to 625 knots airspeed at 15,000 feet altitude. As you know from the previous description of the Mach indicator, the Mach scale moves automatically to align Mach 1.0 (the speed of sound) to the appropriate airspeed reading. Therefore, at 15,000 feet the pilot can see that Mach 1.0 will be 625 knots and at 30,000 feet, Mach 1.0 will be 588 knots.

Since the operation of the airspeed and Mach number indicator depends on the pressures in and around its diaphragm, you can see why there must be no leaks or restrictions in the systems which provide these pressures. If the pitot-static lines become plugged or develop leaks, the airspeed indication will be too low. Even a cracked instrument glass or other leaks in the case can cause serious inaccuracies in the instrument readings. These leaks will allow the pressurized air in the cockpit to replace the true static air pressure surrounding the diaphragm.

In the preceding chapter you were advised to follow the instructions in your F-102A Maintenance Technical Order (T.O. 1F-102A-2-9) when checking out the pitot-static system, but this point is worth repeating! More airspeed indicators (and other instruments on the pitot-static system) are damaged by persons blowing into the pitot-static tube than by normal wear and tear.

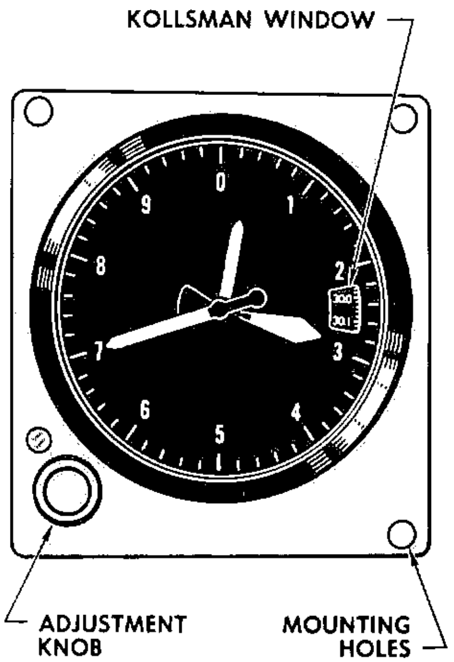


Figure 2-6. Sensitive Pressure Altimeter

SENSITIVE PRESSURE ALTIMETER.

To understand the operation of an altimeter you must first know the purpose for which the indicator is intended and the factors that are involved in its operation. It might be well, then, to define altitude and review some of the principles upon which the altimeter operates.

ALTITUDE.

Altitude is the height of the aircraft above some level used as a reference point. Since there are several possible reference points, there is more than one kind of altitude. You can understand the basic principles of altitude measurement by considering the types of measurement: true altitude, which is the height above sea level; absolute altitude, which is the height above the terrain directly under the aircraft; and pressure altitude, which is the height above the standard datum plane.

The standard datum plane is based on what is called the *standard atmosphere.* Standard atmosphere refers to a condition in which the sea level pressure is 29.92 inches of mercury at +15°C, and in which the pressure and temperature decrease at a standard lapse rate for any increase in altitude. The rate of decrease of temperature and pressure as altitude increases (or standard lapse rate) is not too important at low altitudes. There is quite a distinct difference between pressures and temperatures below 10,000 feet, but as the altitude increases above this point, the variations become more critical. For this reason, only pilots flying at high altitudes need to worry about the standard lapse rate and you should not concern yourself too much with this relationship. The 0-foot level (sea level) of the standard atmosphere is the standard datum plane.

Standard atmosphere is theoretical and exists only on paper. The actual atmosphere varies greatly from this standard. However, since an altimeter has not been designed to indicate the true altitude under all atmospheric conditions, this average has been set up as the standard, and is called the standard atmosphere.

DESCRIPTION OF THE ALTIMETER.

The sensitive pressure altimeter has three pointers which rotate over a fixed dial. As you can see in figure 2-6, the dial is numbered from 0 through 9 and graduated in intervals of 20 feet. The largest pointer makes one complete revolution for every 1,000 feet of altitude. A shorter pointer makes one revolution for every 10,000 feet of altitude. The third pointer, the shortest and thickest of the three, turns at 1/10 the rate of the second pointer up to the maximum limits of the instrument.

An additional scale is used to indicate barometric pressure. As you can see, this scale shows through an opening known as the Kollsman window in the main dial. The opening is called the Kollsman window because the Kollsman Instrument Company was the first to incorporate the barometric scale on the altimeter in this fashion. You can adjust this scale for the correct barometric pressure by turning the knob on the front of the instrument. This knob also moves the three pointers along the altitude scale.

The altitude sensing unit consists of two metal aneroid diaphragms linked mechanically to the pointers and the setting mechanism. These aneroid diaphragms are the sealed, evacuated boxes of thin metal construction which you learned about in Chapter I, and which expand and contract with the changes of pressure on their outer surfaces. The entire sensing and indicating gear is enclosed in an air-tight case that is vented to atmosphere through a static air connection at the aft end.

HOW THE ALTIMETER OPERATES.

Essentially, the altimeter is an aneroid barometer calibrated to show altitude in feet rather than barometric pressure. To understand the functioning of the altimeter, you must first understand the principle of the barometer. The atmosphere is an ocean of air surrounding the earth. Air has weight and density (weight per unit volume) just as water has, and this weight produces pressure. The pressure is heaviest at the bottom of this "ocean" of air, or near the surface of the earth, just as the pressure of water is greatest at the bottom of the sea. Atmospheric pressure at sea level is approximately 14.7 psi (pounds per square inch).

Air differs from water in one very important respect: namely, compressibility. Water is practically incompressible, so pressure decreases upward from the bottom of the ocean at a uniform rate. Air can be compressed into a smaller volume, however, causing its density to increase. Therefore, a given volume of air at sea level contains more air and weighs more than a similar volume of air above sea level. As you can see from figure 2-7, a cubic foot of air at the surface of the earth is compressed by the weight of the column of air extending directly above it to the upper limits of the atmosphere. When a unit of air is raised higher above the earth, the column of air above it becomes shorter; consequently, the air is not compressed as much and its density is lower. Thus, both pressure and density decrease as altitude increases.

The mercurial barometer is a device which can measure atmospheric pressure. It consists of a tube of mercury sealed at the top and inverted in an open dish of mercury. Atmospheric pressure supports the column of mercury in the tube at a level where the weight of the column is equal to the weight of a column of air of equal diameter extending to the upper limits of the atmosphere. Any change in atmospheric pressure causes a corresponding change in the height of the column of mercury.

The pressure exerted by a column of liquid is equal to the density of the liquid times the height of the column. Therefore, atmospheric presssure can be calculatd by multiplying the density of mercury by the height of the column in the barometer. However, it is simpler to express atmospheric pressure in terms of the height of the column. Although the pressure at sea level varies, 29.92 inches of mercury (in. Hg) is taken as the standard value.

An aneroid barometer measures atmospheric pressure in a different manner, but reflects the value in inches of mercury. The aneroid barometer consists of a sealed, evacuated cell (or aneroid unit) which expands and contracts with changes in pressure on its outer surface. Mechanical linkage transmits this expansion and contraction to the dial needle. By changing the dial to read altitude in feet instead of inches of mercury, you have an altimeter. Regardless of the dial calibrations you must remember that the aneroid unit actually measures pressure.

When 29.92 is set in the Kollsman window, the reading of the altimeter is called pressure altitude. Pressure altitude does not correspond with true altitude

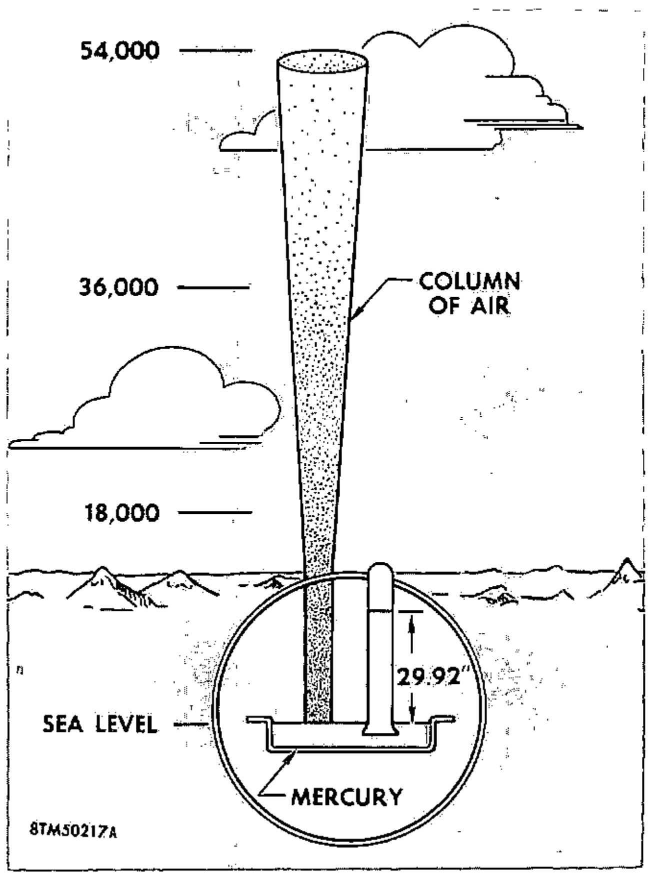


Figure 2-7. Atmospheric Pressure Measurement

unless the pressure flight level corresponds to the standard atmosphere altitude. Actually, it is an exact expression of flight-level pressure.

When the pressure lapse rate is standard, the altimeter is designed to show true altitude. The altimeter setting system enables you to correct for the difference between actual pressure and standard pressure. This involves turning the knob on the front of the instrument until the actual sea level atmospheric pressure appears on the barometric scale. Then the altimeter will show the true altitude at ground level, and the correct pressure altitude above that level. If the pressure lapse rate above the ground is standard, the altimeter reads true altitudes at other levels as well.

Ordinarily, the pressure lapse rate is not standard. Air expands as the temperature increases, so warm air is less dense than cold air. As a result, the pressure lapse rate is less in a column of warm air than in a column of cold air. Therefore, pressure decreases less rapidly with height in warm air than in cold air, and the altimeter tends to read too high in cold air and too low in warm air.

Naturally the pilot of an aircraft is interested in his absolute altitude — his height above the terrain —

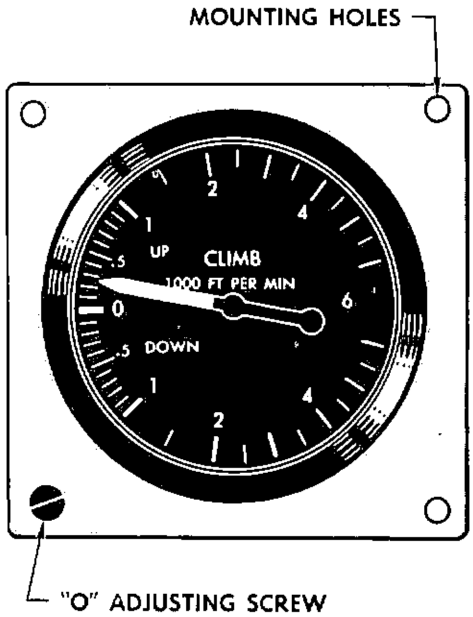


Figure 2-8. Rate of Climb Indicator

otherwise he would always be in danger of flying into mountain peaks or other obstructions. Therefore, he computes his true altitude from his indicated altitude and then consults an aeronautical chart to determine the height above sea level of the terrain where he is flying. Of course, before attempting to compute his true altitude the pilot must make sure that his altimeter setting is correct. If the indicated altitude does not show the correct pressure level above sea level pressure in that area and at that time, his true altitude figures will also be in error.

Frequent weather reports are broadcast over aircraft radio stations. Each station computes its altimeter setting by taking a barometric reading and converting it to sea level pressure by applying the standard pressure lapse rate. Thus, a station 500 feet above sea level reports its altimeter setting as the barometric reading it would get if taken at the bottom of a 500 foot well. A pilot planning to land at that station would set the station reading into his Kollsman window, and upon landing his altimeter would indicate the actual field elevation.

If the altimeter does not show the correct field elevation (plus the approximate height of the instrument above the runway) when the pressure setting is correct, the relationship between the diaphragm and pointers must be changed. This is accomplished by loosening the screw next to the setting knob. Then you can make the pointers and pressure scale agree by rotating the setting knob. Normally, this operation is performed in the instrument shop, and does not concern the line mechanic.

Since the altimeter is connected to the static system, it is subject to error from some of the same causes as the airspeed and Mach number indicator. There is no need to repeat these causes, but remember — don't blow into the static lines! Diaphragms are easily damaged.

RATE OF CLIMB INDICATOR.

Often the pilot of an aircraft wants to maintain his flight at a constant altitude, or change his altitude at a certain rate.When the horizon is clearly visible, this may not present any problem, however, when weather conditions obscure visibility, he must rely on his instruments. There are two ways in which the pilot can determine his rate of ascent or descent. One way is to clock the movement of the altimeter hands. An easier and quicker way is to have a separate instrument to measure vertical speed — this is precisely *what the rate of climb indicator does.*

The rate of climb indicator (sometimes called dive-and-climb or vertical-speed indicator) is an airtight instrument vented to atmospheric pressure through the static system. Figure 2-8 shows you the face of this indicator as you see it in the cockpit. This instrument contains a sensitive diaphragm, temperature compensating device, restrictor valve, single pointer and dial, and connecting tubing and mechanical linkage. The adjustment screw shown at the lower left of the indicator case is used to center the pointer on zero.

Although this indicator receives pressure from only one source, it operates as a differential pressure measuring device. It measures the difference between static pressure at different altitude levels. The difference is shown on the dial which is calibrated from 0 to 6000 feet-per-minute. Rate of ascent is shown on the upper sector of the dial; rate of descent is shown on the lower sector.

HOW THE INDICATOR WORKS.

In the explanation of the altimeter, in this chapter, you saw how atmospheric pressure decreases with increasing altitude. By measuring the rate of change of the atmospheric pressure surrounding the airplane, it is possible to measure the rate of ascent or descent. The rate of climb indicator operates on this principle. Figure 2-9 shows the schematic operation of this instrument in three conditions: level flight, dive, and climb. In the following paragraphs, refer to these three operational schematics as you learn how this indicator operates.

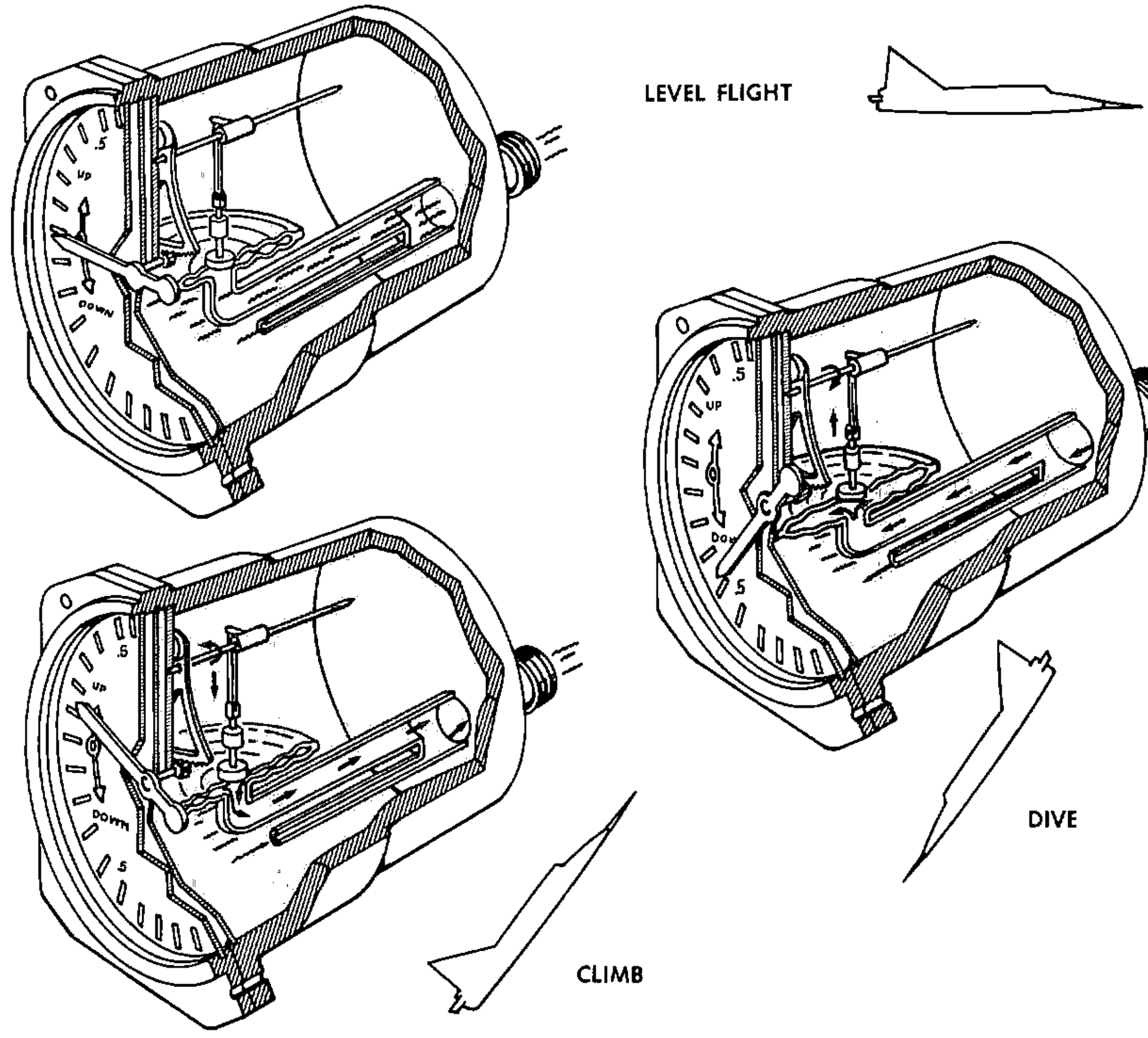


Figure 2-9. Rate of Climb Operational Schematic

Static air pressure is admitted to the instrument from the fitting at the aft end. This air pressure flows freely through the larger tube into (or out of) the flexible metal diaphragm. The smaller tube admits the same static pressure from the fitting through a calibrated restrictor valve. The pressure which passes through this retarding mechanism fills the airtight case around the diaphragm. Any difference between the pressure in the diaphragm and the pressure that surrounds it causes the diaphragm to expand or contract. This movement is transmitted by levers and gears to the pointer.

A pressure differential is created whenever the airplane changes altitude. When the airplane climbs, the pressure inside the diaphragm reduces as the altitude increases. Because of the restrictor valve, the pressure of the air surrounding the diaphragm reduces more slowly. This pressure differential increases at a rate that corresponds to the airplane's rate of climb. The diaphragm contracts and the pointer moves on the upper scale of the dial to a point corresponding to the difference in pressure. Conversely, when the airplane dives, the diaphragm receives the increased atmospheric pressure more rapidly. It expands and deflects the pointer downward on the dive scale an amount corresponding to the rate of descent. Whenever the airplane returns to level flight, the pressure on both surfaces of the diaphragm equalizes and the pointer returns to zero. It takes several seconds for this to occur due to the action of the restrictor valve. The same characteristic prevents excessive oscillation of the pointer that might result from air "bumps" and resulting sudden changes in pressure.

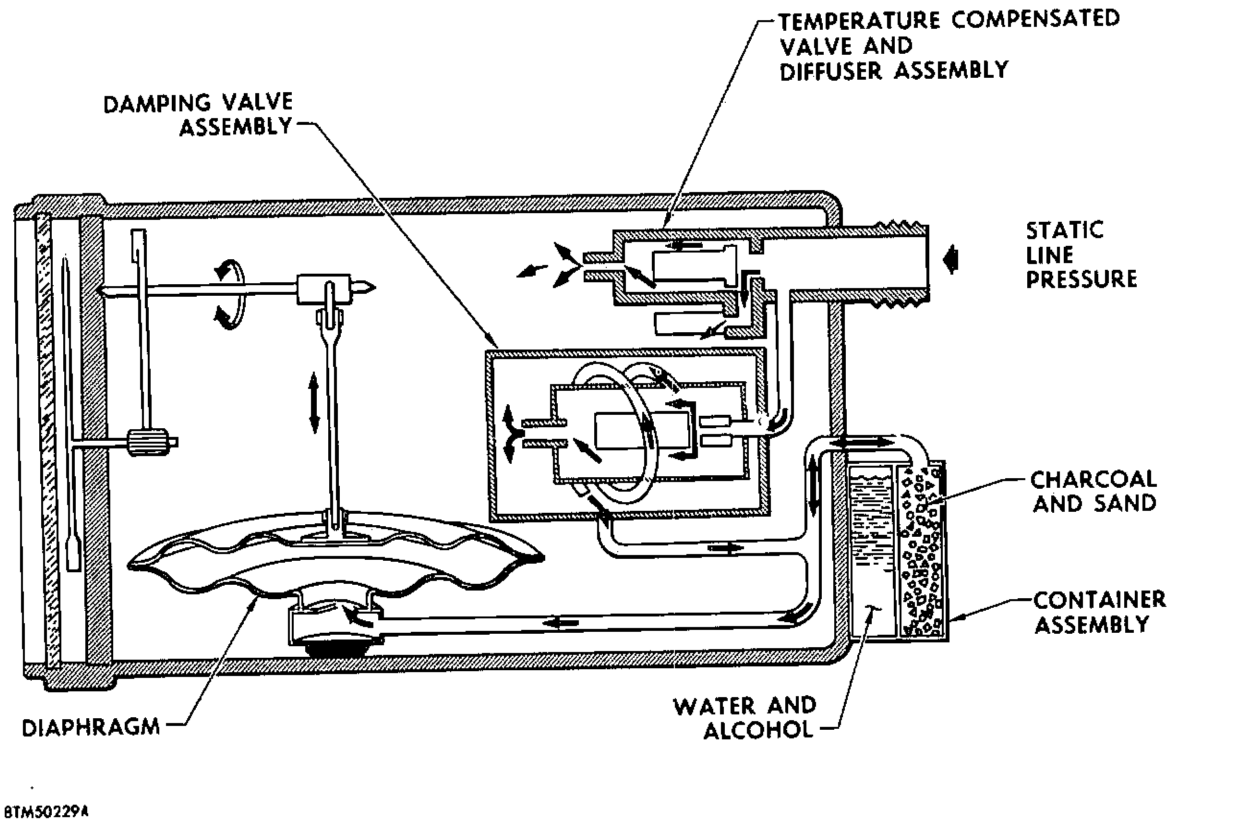


Figure 2-10. Rate of Climb Cutaway

Temperature compensation occurs automatically within the instrument by means of the chemical container assembly and damping valve shown in figure 2-10. The damping valve compensates for temperature changes by modifying the flow of air to the container assembly and diaphragm. The container assembly has two compartments. The *charcoal and sand compartment* absorbs air that is decreasing in temperature and releases air when its temperature rises. The *alcohol and water compartment* delays the rate at which the temperature of the charcoal changes.

Operation of the rate of climb indicator is independent of any external control, except for the zero adjusting screw. This screw offers a simple and convenient means of making minor adjustments before flight if, for any reason, the pointer may not be centered.

TURN AND SLIP INDICATOR.

The turn and slip indicator (also called the turn and bank indicator) combines two instruments in one. The slip indicator is a sealed glass tube situated on the lower part of the dial, as shown in figure 2-11. All other parts of the instrument make up the turn indicator. Since the two indicators function independently, we will discuss them separately; then you will understand why they are combined into one instrument.

THE SLIP INDICATOR.

This indicator is nothing more than a simple inclinometer, that is, a device that indicates when it is inclined or tilted, and it operates much like a pendulum. The curved glass tube contains a ball and a damping liquid. The fluid merely provides for smooth movement of the ball within the tube. Since the tube is mounted horizontally, gravity tends to keep the ball in the center (the lowest point) when the airplane is flying with the wings level. Now suppose that the airplane is banked for a turn to the left. Gravity is still pulling the ball toward the lowest point, which then becomes the left end of the tube. But now centrifugal force enters the picture, it tends to throw the ball outward from the center of the turn which is the high side of the tube. The actual position taken by the ball is therefore determined by the relative pull of these conflicting forces.

Indications Presented by the Slip Indicator.

Figure 2-11 shows how the slip indicator appears under three typical flight conditions. In the left view you can see the slip indicator in a perfectly coordinated turn in which the degree of bank is correct for the rate of turn—the centrifugal force and gravity are balanced. This balancing of forces centers the ball between the reference markers. The upper right view shows the

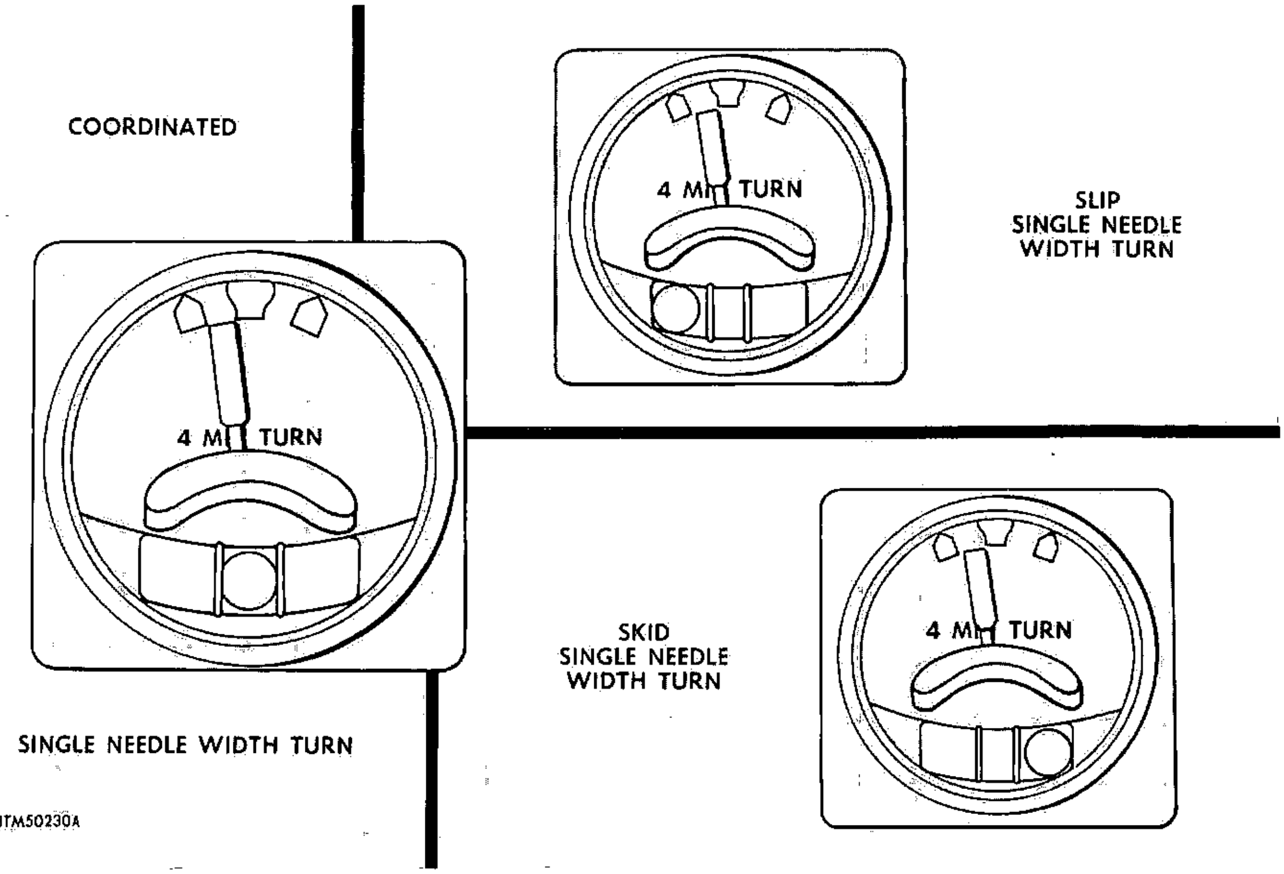


Figure 2-11. Three Typical Indications on the Turn and Slip Indicator

indicator when the airplane is banked without a sufficient turn; the airplane slips, that is, it slides off on the low wing and loses altitude. The airplane slips because the centrifugal force is less than the pull of gravity. The ball, being affected by the same forces, moves to the lower side of the tube. On the other hand, when a turn is made with too little bank, the airplane skids —slides to the outside of the turn—and the ball does the same as shown in the lower right view.

THE TURN INDICATOR.

The turn indicator needle, which is shown above the slip indicator in figure 2-12, points in the direction the airplane is turning. This needle is actuated by the movement of a restricted (not universally mounted) gyro that is driven by the 28-volt d-c electrical system. The gyro is mounted so that the rotor spin axis is normally held by spring tension parallel to the lateral axis of the airplane. You will recall from the discussion on gyros in Chapter I that precession is one of the basic properties of all gyros. The gyro assembly in the turn-indicator always precesses about its longitudinal axis when a force or torque is applied about the vertical axis. To understand how this precession is transmitted to the pointer, study figure 2-12.

Suppose that you are looking at an actual operating turn indicator from the same view as shown in the illustration. The rotor will be spinning clockwise—up and away from the dial. Now, if you take the entire instrument and turn it to the left, the gyro assembly will tilt to the right on its longitudinal axis. You are actually pivoting the gyro about its vertical axis by applying a force at the bearings of the longitudinal axis. Remember, that the precession always takes place 90° ahead of where the force is applied and in the direction of rotation. The amount of precession will correspond to the rate at which the instrument is turned, but will not exceed 45° in either direction because of the mechanical stops. Precession will cease the instant you stop turning the instrument, and the spring tension will return the gyro assembly to its normal position.

Now follow the train of action from the gyro assembly to the pointer. When you rotated the instrument to the left, as in a left turn, the gyro assembly tilted to the right. To make the pointer show the actual direction of the turn, a reversing mechanism is used. It consists of a plate attached to the gyro assembly with a projecting rod on the lower end. This rod engages between the two vertical pins on the pointer shaft.

Thus, when the gyro assembly and reversing mechanism rotate clockwise, the pointer moves counterclockwise a proportionate amount—it works exactly like two meshed gears.

Excessive oscillation of the turn indicator pointer is prevented by a damping unit. Referring again to figure 2-12, notice that a piston is linked to the gyro assembly. Any precession of the gyro moves the piston within the cylinder which is attached to the instrument case. An adjustable opening, near the top of the cylinder, permits the damping effect to be changed in the instrument shop. Remember that the arrows in figure 2-12 show the action that occurs in a left turn only. In a right turn, the movement of the turn mechanism is reversed. Of course, the gyro rotor always spins in the same direction.

You have learned, in the previous paragraphs, how the gyro assembly—and therefore the pointer—deflects to correspond with the rate the airplane is turned around its vertical (yaw) axis. It is not affected by movements about the lateral (pitch) or longitudinal (roll) axis. The pilot of an airplane uses this information during many maneuvers, such as flying a holding pattern under instrument (blind flying) conditions. It also helps him to keep a desired heading. In the F-102A, as well as other high speed aircraft, the turn indicator is calibrated for a four minute needle-width turn. In other words, when the pointer deflects its own width from the center, the pilot knows he will make a complete 360° turn in four minutes.

THE TURN AND SLIP INDICATOR AS A UNIT.

Everything that a pilot needs to know to make a coordinated turn at a given rate is provided by the turn and slip indicator. It was the first genuine *blind flight* instrument developed, and is the first instrument a pilot consults when attempting to recover from an unusual flight attitude. The phrase "center the needle —center the ball" is drilled into every student pilot.

You should have very few maintenance problems involving the turn and slip indicator. If a pilot reports that the turn needle does not respond smoothly, replace the instrument. Since proper functioning of the instrument depends on correct gyro speed, check the power supply first. When the airplane is stationary, the pointer must indicate zero whether power is ON or OFF. The ball inclinometer is practically trouble-free. If any discoloration occurs in the damping fluid or luminous backing and center lines, replace the instrument.

ATTITUDE INDICATOR.

You will recall in the discussion of gyro instruments in Chapter I that a pilot must have some reference to the horizon to maintain safe flying attitudes. If the horizon is obscured for any reason, artificial references must be provided. This artificial reference is made possible by using an attitude indicator. The attitude indicator is a gyro instrument which provides an artificial reference by showing the attitude of an aircraft in relation to the earth's horizontal plane. It functions during any maneuver, giving the pilot an indication of angular displacement (tilting) throughout 360° of pitch or roll.

The attitude of the aircraft is shown on this instrument pictorially by the position of a miniature airplane on a sphere, a horizon bar, and graduation marks. Figure 2-13 provides you with a cutaway view of this instrument. The knobs you see on the front of the instrument are the only control which the pilot has over its operation. The caging knob on the right is used to return the horizon bar to a plane parallel with the horizontal axis and perpendicular to the vertical axis of the aircraft. The other knob that you see on the left of the indicator is used to raise or lower the miniature airplane. This is necessary to maintain a level flight condition for varying speeds and load conditions. The pilot can also use this knob to set the indicator for a certain climbing or gliding attitude. The OFF flag shown at the top of the instrument face disappears whenever current is supplied to the instrument.

CONSTRUCTION OF THE ATTITUDE INDICATOR.

As you can see in figure 2-14, the gyro assembly is mounted in a pipe which is cast as part of the case. The shaft extension from the outer gimbal fits into the pipe and is free to rotate on ball bearings within the pipe. The inner gimbal is free to rotate within the outer gimbal and supports the gyro rotor in a vertical position on ball bearings. Thus, the rotor is able to maintain a fixed position when the airplane climbs, dives, or rolls.

The attitude indicator is operated by 115-volts, 400-cycle alternating current. This power enters through a connector on the back of the case and leads to both the gyro assembly and the low inertia OFF flag motor. The low inertia motor serves only to move the OFF flag from sight when power is supplied to the instruments. Five low-friction contact points carry the current from the gimbals to the gyro rotor.

Erection Mechanism.

Any gyro, even when universally mounted, is subject to precession from bearing drag and external forces. To keep the gyro erect, some mechanism is always provided which will return it to the desired position. The spin axis of the gyro in the attitude indicator is automatically erected to a vertical position by the weight of steel balls. Figure 2-15 shows you how this works in the erection mechanism.

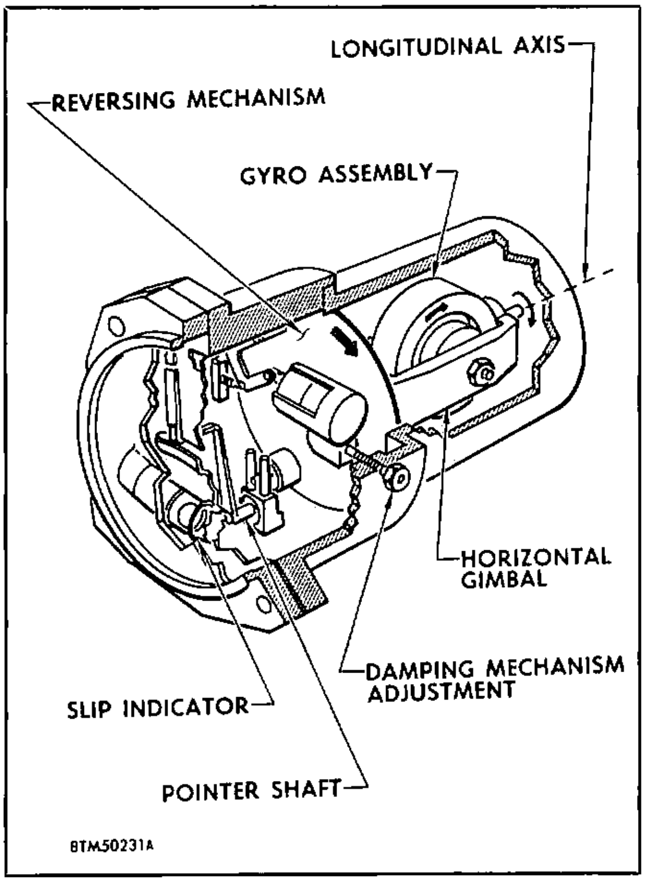


Figure 2-12. Turn and Slip Indicator Cutaway View

As you can see, a round magnet is attached to the upper end of the gyro spin axis. It spins at the same rate as the gyro rotor, about 21,000 rpm. Surrounding this magnet is a plate that is supported by ball bearings to prevent it from touching the spinning part of the gyro or magnet; however, the magnetic attraction from the spinning magnet rotates the plate at about 45 rpm. Note that recesses on top of the plate hold two steel balls. When the gyro is erect, the balls rotate at a steady speed. If the gyro tilts, the balls slow down at some positions in their rotation and speed up at others even though the plate turns at a fairly constant speed. This is possible because the balls are free to move forward or backward within their slots. As the balls pass the high point of the tilted gyro, gravity rolls them forward rapidly and they spend very little time on the "downhill" side of the gyro. When the balls reach the "uphill" side of the gyro, gravity causes them to return to the trailing ends of their slots; consequently, they spend more time on the "uphill" side of the gyro.

Since the balls spend more time on the "uphill" side of the gyro, more weight is exerted in this position. More weight means more precessional force. The precession takes place 90° ahead of the force—in the direction of rotation—so the high side will precess downward. This precession will continue until the spin axis is vertical.

Caging Mechanism.

When power is applied to the attitude indicator, the gyro will attain normal operating speed and position in about three minutes. Sometimes that is too long, as in the case of a "scramble" for takeoff. The manual caging mechanism permits a pilot to erect the gyro after power has been on for only 30 seconds. The gyro will have attained enough speed in that time to remain erect. Thus, it is not necessary to keep ground power connected during an alert just to keep the gyro erect.

Quick erection by caging is also useful in the air. Certain errors in indication, which we will explain later, are induced by high speed maneuvers. The pilot can erect the gyro and eliminate these errors quickly. You should remember, however, that caging the gyro erects it to a vertical position in the case. The pilot must have other gyro instruments or visual reference to the ground to be sure he is flying straight and level when he cages this instrument, otherwise it will not be vertical with reference to the ground.

Caging the attitude indicator is accomplished with two projecting arms which are actuated by the PULL TO CAGE knob. The gyro will center first on the roll axis and then on the pitch axis. To cage the gyro, you must pull the knob smoothly and release it quickly. The knob is spring-loaded to return to the uncaged position. To make sure that the spring-loaded mechanism has returned the caging knob to the uncaged position, push on the knob immediately after you release it. If it is not all the way in or if the gyro

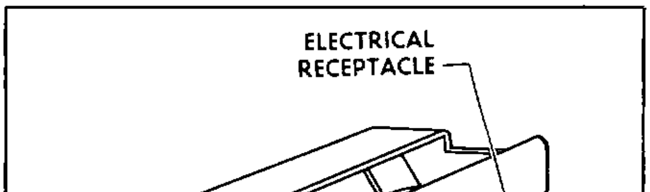


Figure 2-13. Attitude Indicator

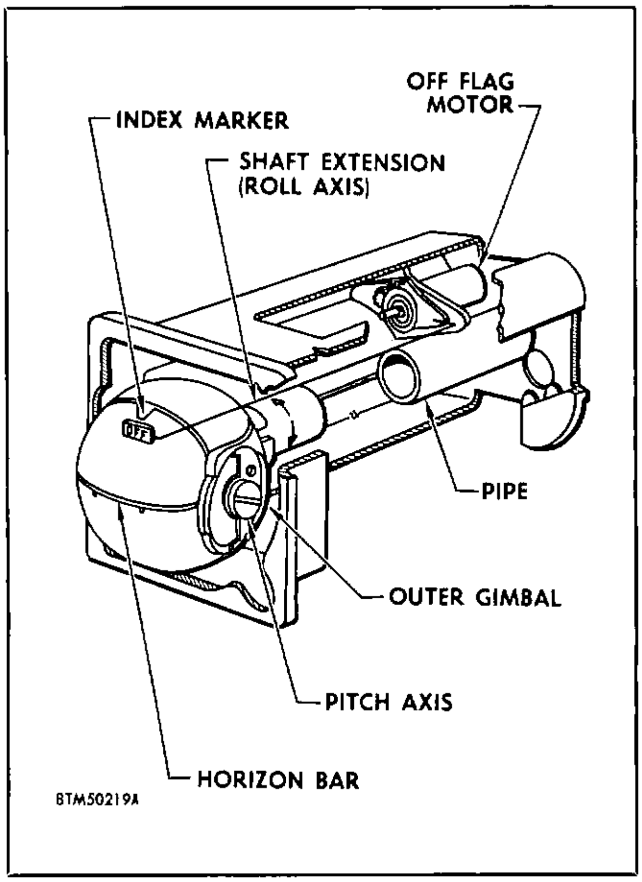


Figure 2-14. Gyro Housing and Major Components

immediately precesses, the caging mechanism is not operating properly and you must replace the instrument.

PRESENTATION OF INDICATIONS.

To understand how the attitude indicator works, you must remember that the gyro assembly—and consequently the spherical dial—actually remains in a fixed position while the case moves around it. The horizon bar is hinged to the gyro mounting and connected to the gyro assembly by a fork and pin arrangement. The pin engages the fork so that it operates like a gear train.

In figure 2-16 you can see the various indications these units present on the face of the instrument. Several attitudes of flight and the corresponding indication for each are shown. You may never see the attitude indicator in these positions, but you will understand the instrument better if you study them. When the airplane climbs, the horizon bar drops below the miniature airplane. When it dives, the bar rises above the miniature airplane. This bar will show the airplane attitude up to a 27° climb or dive. Beyond these limits, the pin slides out of the fork and the horizon bar stops moving. The sphere itself, which is not moving, then becomes the new reference. A continued increase in pitch angle, approaching the vertical attitude, is indicated by the word CLIMB or DIVE on the sphere. Notice that the word DIVE is on the top of the sphere and the word CLIMB is on the bottom. This is because the sphere remains stationary and the instrument case tilts around it.

When the aircraft approaches a 90° pitch (as it does during a loop) a controlled precession takes place. The gyro strikes a mechanical stop on one side of the rotor gimbal. Since the stop is only on one side, a 180° precession results. Thus, the horizon bar turns completely over. In the case of a loop, controlled precession occurs twice—once when it is going straight up and once when it is going straight down. Therefore, when the aircraft attitude is again within 27° of level flight, the bar is completely operable.

Errors in Indication.

The usual errors in a vertical gyro instrument are considerably exaggerated in high speed flight. The greatest of these is the turn error. Remember that the gyro in the attitude indicator remains vertical when the airplane rolls, as in a turn. As a result, centrifugal force tends to keep the balls in the erection mechanism toward the outside of the turn too long. The gyro precesses, and when level flight is resumed, the horizon bar is not centered. A similar situation exists during rapid acceleration, because the balls tend to slow down at the aft end of their plane of rotation. The force is the same as you feel when you floor the accelerator of a "hot" car. These errors are usually referred to as sluggishness or instrument lag. Normal erection will take place once true gravitational forces are sensed by the erection mechanism.

The instrument panel in the F-102A is slanted somewhat to permit the best possible view from the pilot's seat. As a result, the attitude indicator is mounted with a wedge along the bottom half of the bezel. This is necessary to keep the instrument perpendicular to the earth's horizontal plane. Without the wedge, the instrument would indicate a dive attitude when the aircraft was in level flight. A new type of attitude indicator—called the vertical gyro indicating system—is installed in later models of the F-102A. In this system, a rate gyro and torque motor are used to eliminate the errors mentioned above. This system is the next to be discussed.

VERTICAL GYRO INDICATOR SYSTEM.

The vertical gyro indicating system is an improved attitude indicating system. Presentation of the information is accomplished in a different manner, and the system eliminates some of the errors inherent in the old indicator. Some of the theory of operation is identical for the two systems and is not duplicated here. The most obvious difference between the two

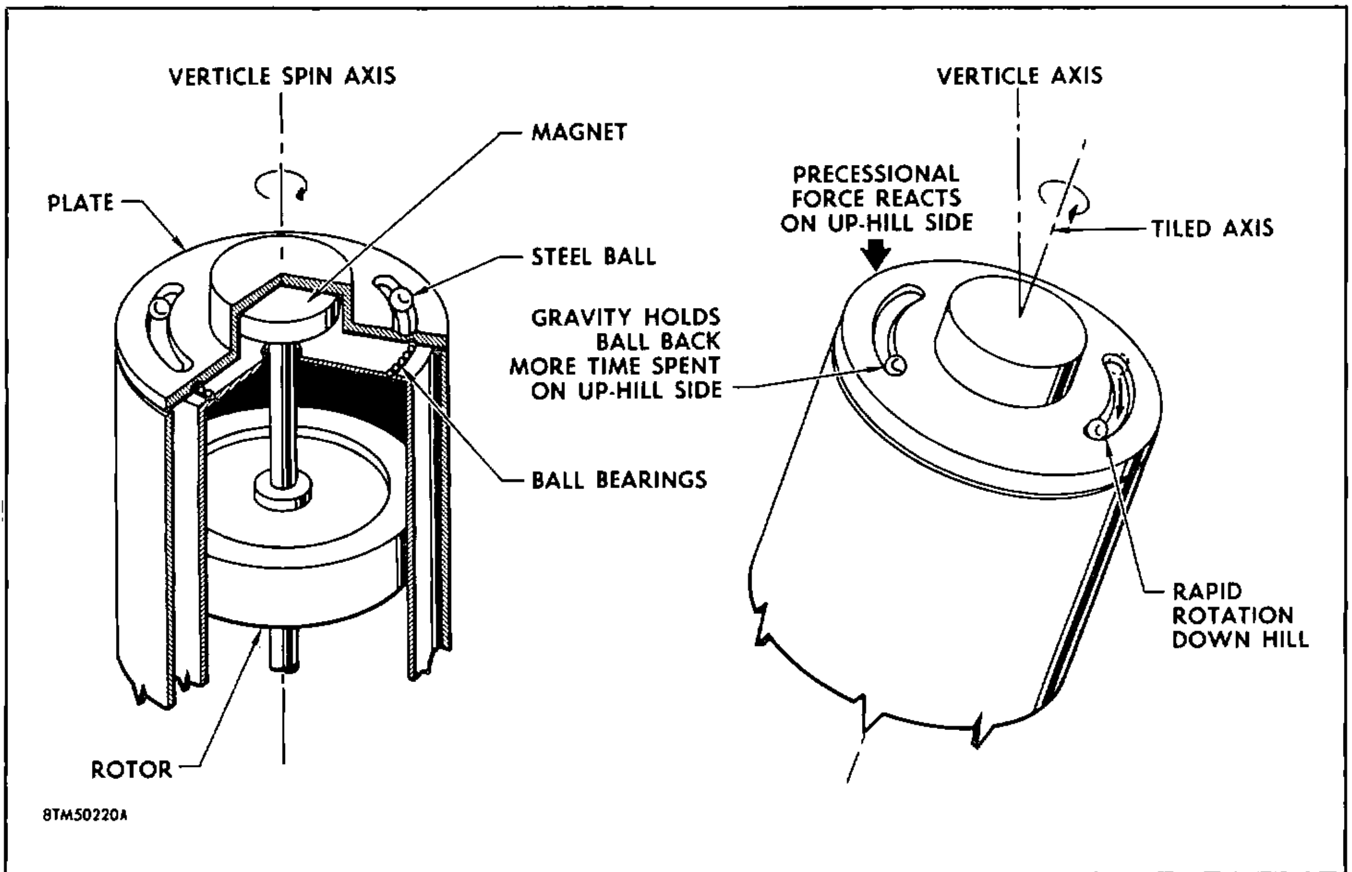


Figure 2-15. Attitude Indicator Erection Mechanism

systems is that the attitude indicator is a self-contained unit, whereas the vertical gyro indicating system is made up of two separate components: a remote indicator and a control unit. Figure 2-17 shows you both units as they appear in the aircraft.

Note in figure 2-17 that the markings on the indicator are similar to those on the attitude indicator of figure 2-16. This indicator shows the lines representing the angles of dive and climb, and the placarding CLIMB and DIVE at about the 30° angle. The miniature airplane, which shows the airplane attitude, is also very similar in appearance. On the control unit cover you can see the DIRECTION OF FLIGHT arrow. Although it would be difficult for you to install this unit backwards, you should always check that the unit is installed correctly. Reverse installation would give reverse indications on the indicator.

CONTROL ASSEMBLY.

The vertical Gyro control assembly is located in the upper electronics compartment and is the "brain" of the vertical gyro indicator system. Inside this control unit are several related assemblies that work together to present accurate information about the airplane's attitude. The vertical gyro assembly, rate gyro assembly, relay box assembly, time delay assembly, two channel amplifier assemblies, and the power supply assembly are located inside the case. These units and how they function are discussed in the following paragraphs.

Vertical Gyro Assembly.

By now you are quite familiar with the principles of vertical gyro operation. The one used in this system is not too different from the one used in the self-contained attitude indicator. It spins slightly faster (about 22,000 rpm) but the power requirements are the same. The gimbal assembly is mounted in the frame so that it has a freedom of rotation for a full 360° along the roll axis. Of course, the gimbal assembly actually doesn't do the rotating. It stands still while the airplane and the frame rotate. The gyro housing is the inner gimbal, and is free to move on its pitch axis up to 82° of dive or climb. Internal springs restrict the pitch displacement to these limits of dive and climb. Slip rings carry the electrical current to the gyro.

The Erection System.

To avoid the centrifugal force turn errors of the ball-type erection mechanism, erection of this vertical gyro is accomplished by torque motors. These motors apply a force, or torque, to torque rings attached to the ygro gimbals; they do not actually turn the rings. They can

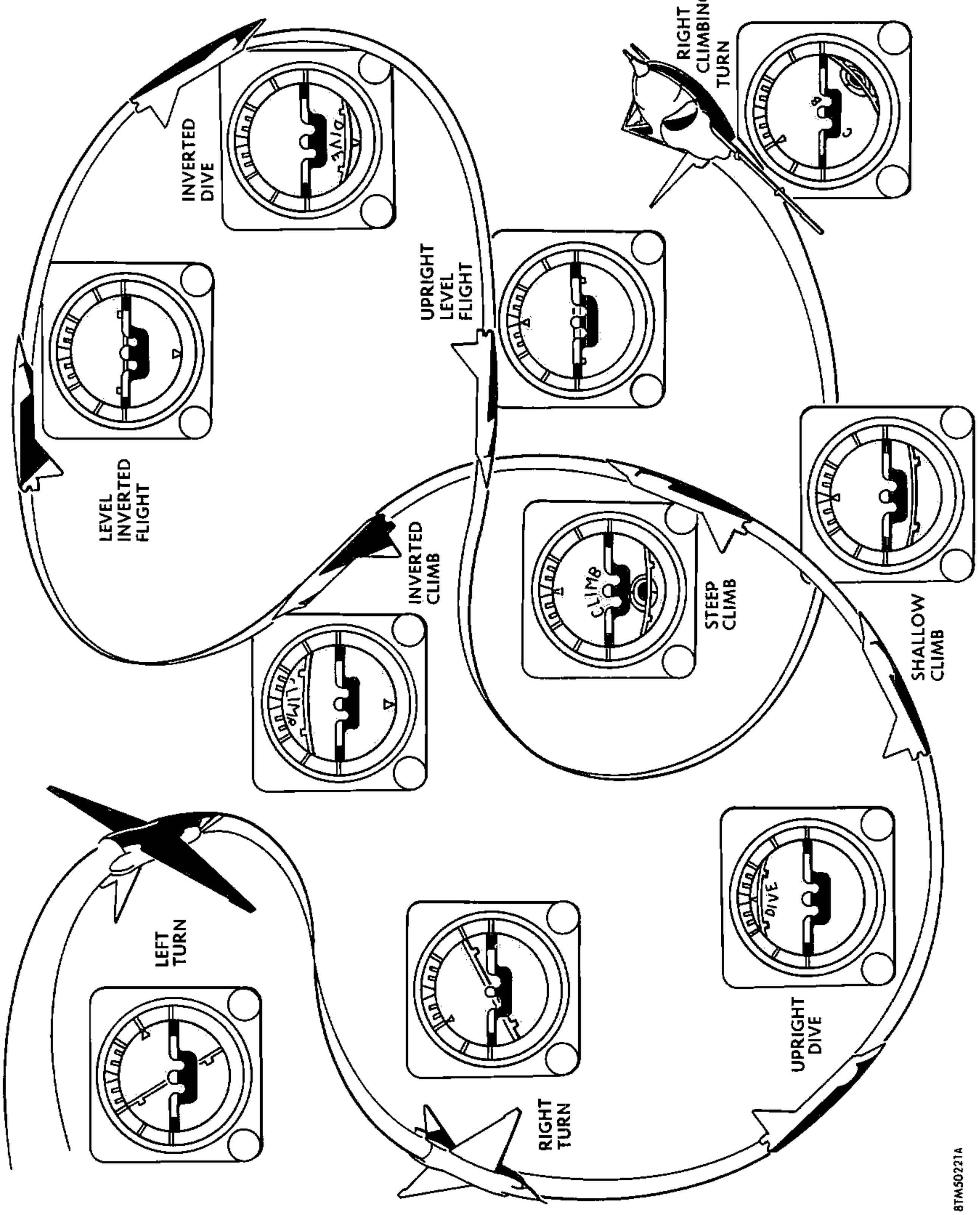


Figure 2-16. Attitude Indicator Positions

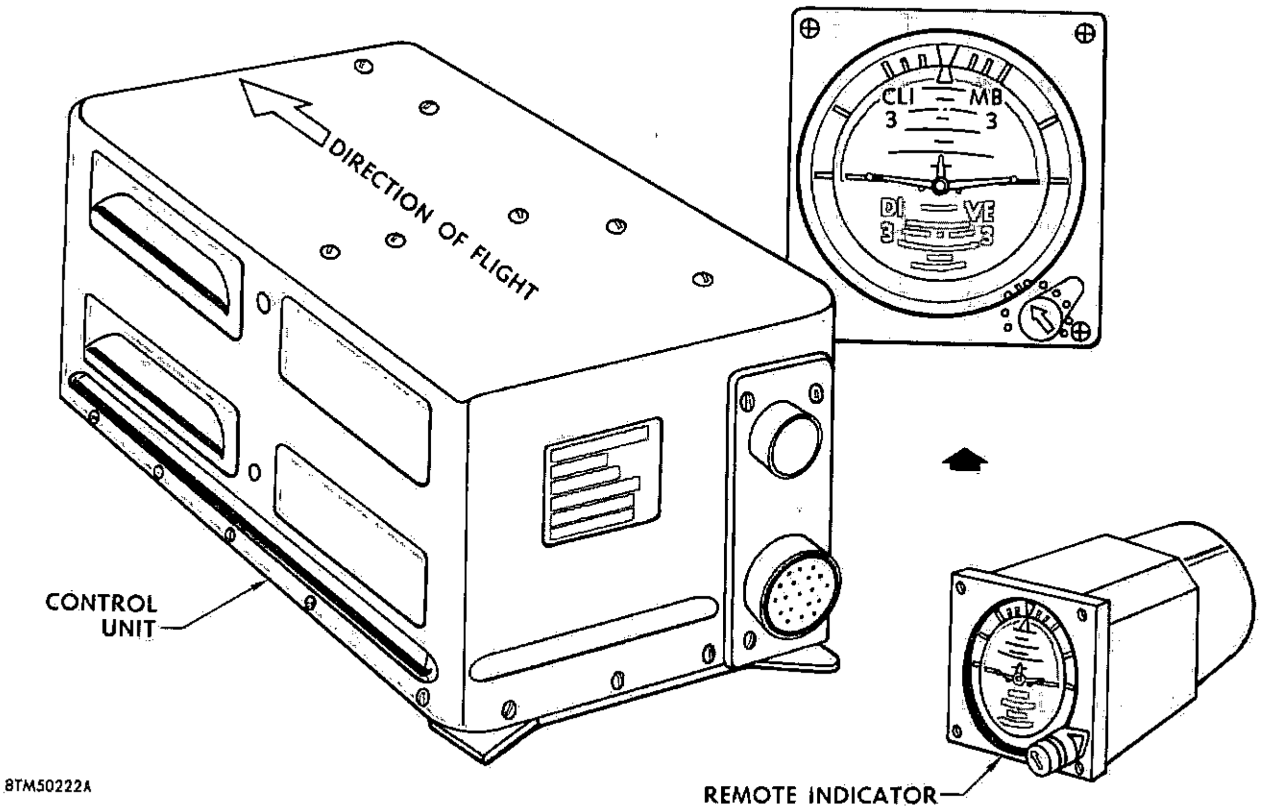


Figure 2-17. Vertical Gyro Indicator System Components

only apply force which causes the gyro to precess about an axis at right angles to the axis of the particular torque ring. That was quite a lengthy and involved statement, but you will get the idea if you examine figure 2-18.

Notice that the roll erection torque ring is attached to the gyro housing (inner gimbal) and the pitch erection torque ring is attached to the outer ring. Applying the rules of precession, which you learned in Chapter I, you can see why a force on the roll ring will cause the gyro spin axis to tilt right or left. Similarly, any torque applied to the pitch ring causes the gyro spin axis to tilt along the fore and aft axis of the airplane. The direction of the precession is, of course, dependent upon whether the torque motors apply torque in clockwise or counter-clockwise direction. For example, suppose that the roll torque motor in the upper diagram is applying torque in the clockwise direction. With the gyro rotor spinning clockwise, precession will cause the top of the gyro to tilt toward you. Conversely, if the torque is applied in a counterclockwise direction, gyro will tilt in the opposite direction.

The erection motors are energized by alternating current from an electrolytic erection switch. This switch consists of a metallic cup with four contact points molded in an insulating material. The cup is partially filled with a low-resistance electrolyte and is attached to the underside of the gyro rotor housing. Its purpose is to keep the spin axis of the gyro perpendicular to the earth's surface.

Note in figure 2-19, that the erection switch contains two pitch contacts along the fore and aft axis of the airplane, and two roll contacts along the lateral axis. A bubble of nonconductive gas covers part of the contacts. The electrolyte grounds the pitch and roll torque motors through the sides of the erection switch cup. When the gyro rotor is properly erected, that is, when the spin axis is perfectly vertical with relation to the earth, the bubble is centered and the electrolyte serves as a ground (with equal resistance) connection to the case for all four contacts. However, any tilting of the gyro causes variations in the resistance to ground one of these contacts, and the torque motors will apply precessive force until the spin axis is again vertical.

The electrolytic erection switch is a gravity device. The fluid tends to cover the contacts on the low side of the tilted cup more than those on the high side. Consequently, it is also subject to the same centrifugal force that affects the steel ball erection mechanism. In other words, the electrolyte in the

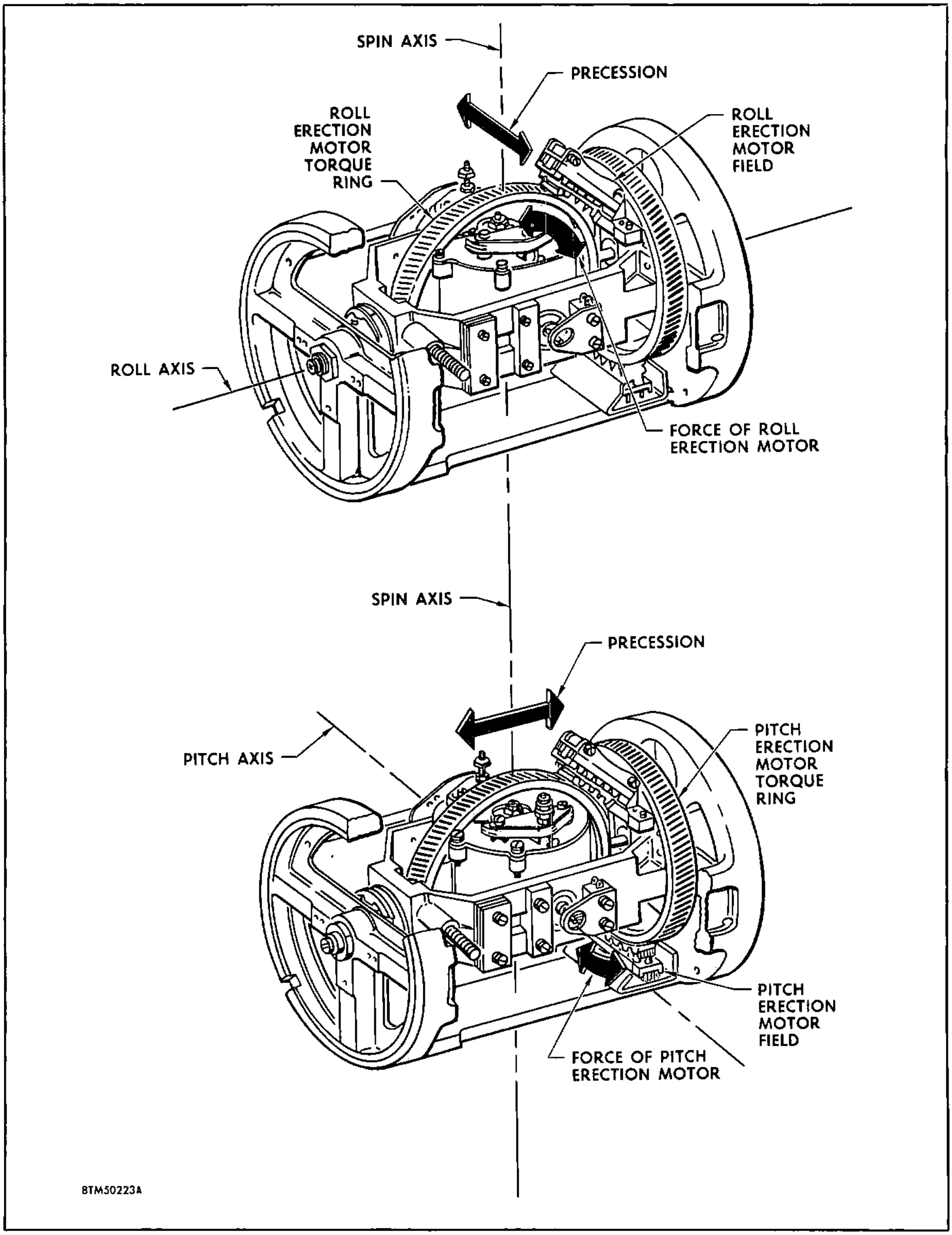


Figure 2-18. Vertical Gyro Indicator System Erection Mechanism

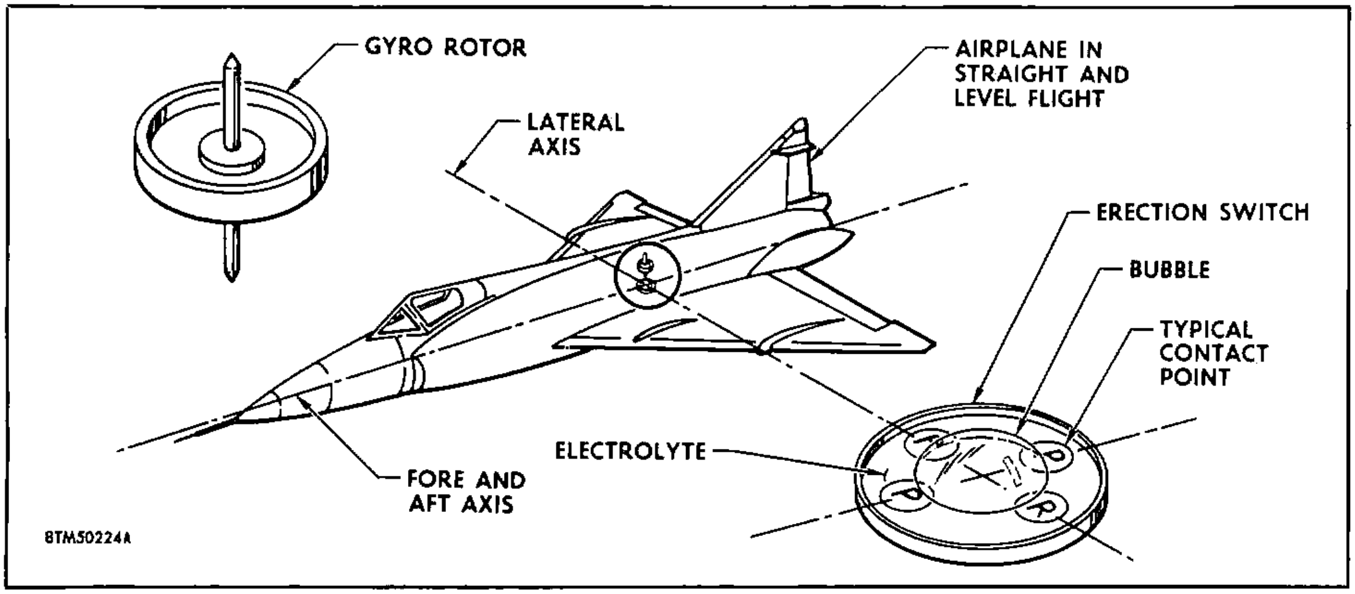


Figure 2-19. Vertical Gyro Indicator System Erection Switch

erection switch tends to be thrown toward the outside of a turn, causing "false erection" of the vertical gyro. Here's where the rate gyro and relay assemblies come in.

As you can see in figure 2-20, the rate gyro is a small, horizontally mounted gyro. Spring tension normally holds this gyro with its spin axis parallel to the lateral axis of the airplane, like the gyro in the turn and slip indicator. When the airplane turns, the rate gyro deflects and trips the switching relays. The relays disconnect the roll contacts in the erection switch and cause both the pitch and the roll torque motors to receive signals from the pitch contacts of the erection switch. A "pitch-bank" erection system is then in effect.

Since the roll contacts are no longer connected to the system; they cannot cause a false correction. However, there has to be some method of roll erection, if not, the vertical gyro could tilt—from some cause besides centrifugal force, such as bearing drag—and a correction would not be made until the aircraft resumed straight and level flight. This is why the switching relays connect the roll torque motor to the pitch contacts. You see, during a turn, any tilting on the roll axis eventually becomes a tilting along the pitch axis.

If that last remark threw you, look at it this way. Suppose a large vertical gyro is spinning on top of an F-102A while it is sitting on the ground, and the gyro has tilted 10° toward the right wing tip—what caused it to tilt is not important here. Now if you hitch a tractor to the nose wheel and swing it 90° to the left, that 10° tilt of the gyro is toward the tail of the airplane. What was a 10° tilt along the roll axis is now a 10° tilt along the pitch axis. Also, when you were in the process of turning the airplane, that 10° error was between the wing tip and the tail. A combination of roll and pitch correction is required to bring this gyro back to vertical.

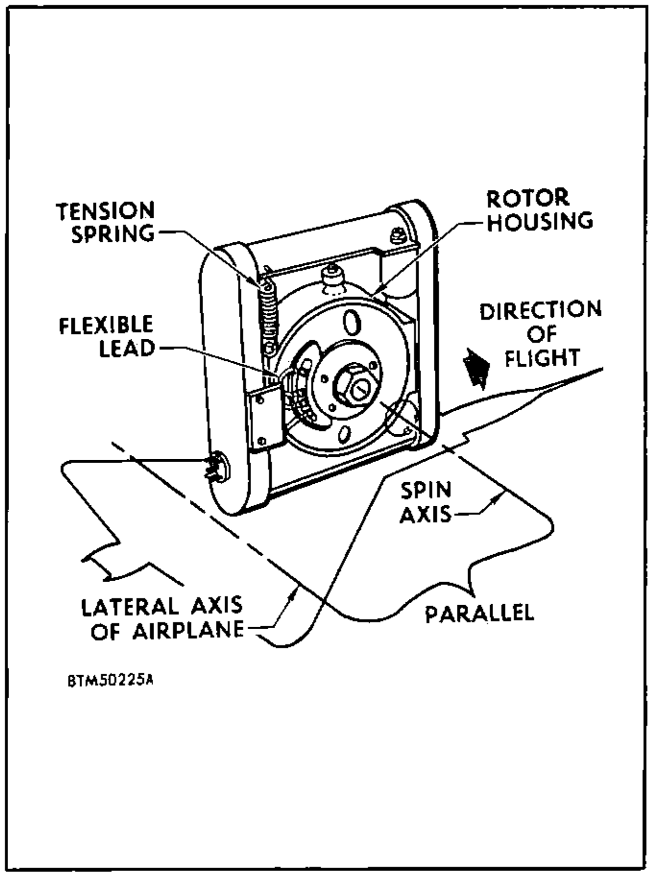


Figure 2-20. Vertical Gyro Indicator System Rate Gyro Assembly

It is undesirable to have the rate-gyro react and switch the erection circuits to the pitch-bank system for every little yaw deviation. The time delay assembly prevents this from happening. Unless the rate-gyro deflects at least 20° for about six seconds, the switching relays will remain in the pitch-roll position.

The time delay relays serve other purposes too. When the master switch in the airplane is first turned on, several things take place to put the system in operation as quickly as possible. A friction-brake solenoid energizes and holds the roll torque ring for about 15 seconds. Its purpose is to prevent the gyro from "tumbling" when power is first applied. For the first two minutes after the power is turned on, the a-c power supply to the torque motors is boosted to expedite the erection process. This same time delay keeps the OFF signal on the indicator in view for the first two minutes. The vertical gyro attains operating speed and is properly erected at the end of that time.

Deviation Measurement.

Information about pitch and roll of the airplane is sensed by the vertical gyro assembly, but it is of no value to the pilot until it is measured and indicated in a convenient manner. Measuring of the gyro displacement is accomplished by two a-c synchro transmitters. To indicate pitch, the primary coil of one synchro is attached directly to the gyro housing while the secondary coil is attached to the gimbal on the pitch axis. The primary coil is fixed in relation to the stabilized vertical reference line (spin axis). The secondary coil, attached to the gimbal, rotates about the primary coil as the airplane noses up or down. To measure roll deviation, the primary coil of the other synchro is attached to the gimbal, and the secondary coil is attached to the gyro frame on the roll axis. The secondary coil rotates around the stabilized primary coil whenever the airplane rolls. Signals from these transmitters serve to position the receiver synchros in the indicator.

THE REMOTE ATTITUDE INDICATOR ASSEMBLY.

The indicating component of the vertical gyro indicator system on later model F-102A's is located in the same position on the instrument panel that the self-contained attitude indicator occupied on early model airplanes. The indicator is similar in appearance, except that the miniature airplane is not adjustable; the spherical dial is graduated, and the indicator does not have a horizon bar. The dial is divided into two sections by the horizon line. The area above the line, or sky sector, is painted gray with black markings for every 5° of climb. The ground sector (dive), below the horizon line, is painted black with gray markings.

You will recall that the other type of attitude indicator has the dive sector on top and the climb sector below. That is necessary because the dial is held rigidly by the gyro and the *case moves around it.* In this indicator the dial is positioned by motors, and is *turned within the case* in the same direction the airplane is rotating on its pitch axis. In figure 2-21, you see two views of the indicating mechanism. The roll axis mounting pin to which the yoke assembly is attached fits into a housing in the case.

A roll servo motor, not shown in figure 2-21, can rotate the entire yoke and sphere assembly a full 360°. The degree of roll is then shown by the pointer which appears directly beneath the marks on the outer fixed dial, and by the pitch lines behind the miniature airplane. In the inverted position you can see the pitch servo motor. It turns the drive shaft of the sphere to show pitch attitude in relation to the miniature airplane. The roll and pitch servo motors get their signals from receiver synchros in the rear of the indicator. These signals are picked up from the synchro transmitters in the vertical gyro assembly, then boosted by the channel amplifiers in the control assembly, and finally sent to the servo motors.

It was mentioned earlier that the miniature airplane in the remote attitude indicator—unlike the one in the self-contained unit—is not movable. Instead, the pilot can change the position of the dial to show a desired pitch angle. The knob on the front of the indicator electrically changes the relationship between the spherical dial and the erected position of the gyro.

Power Failure Flag.

The power failure indicating mechanism consists of an a-c motor geared to the flag shaft. A spring on the shaft of the a-c motor returns the flag into view whenever a-c or d-c power is lost. Relay assembly failure will also cause the OFF flag to appear.

Special mounting provisions are not required for the remote attitude indicator. It is mounted in the slanted F-102A panel without the wedge that was required for the other attitude indicator. In fact, the remote indicator will read correctly in any position, as long as the control assembly is mounted properly for level flight.

The entire remote attitude indicator and the two gyro assemblies in the control unit are hermetically sealed and gas filled to minimize environmental disturbances. These units must be opened only in the instrument overhaul shop. Always consult the appropriate technical orders on the vertical gyro indicating system before you work on this system.

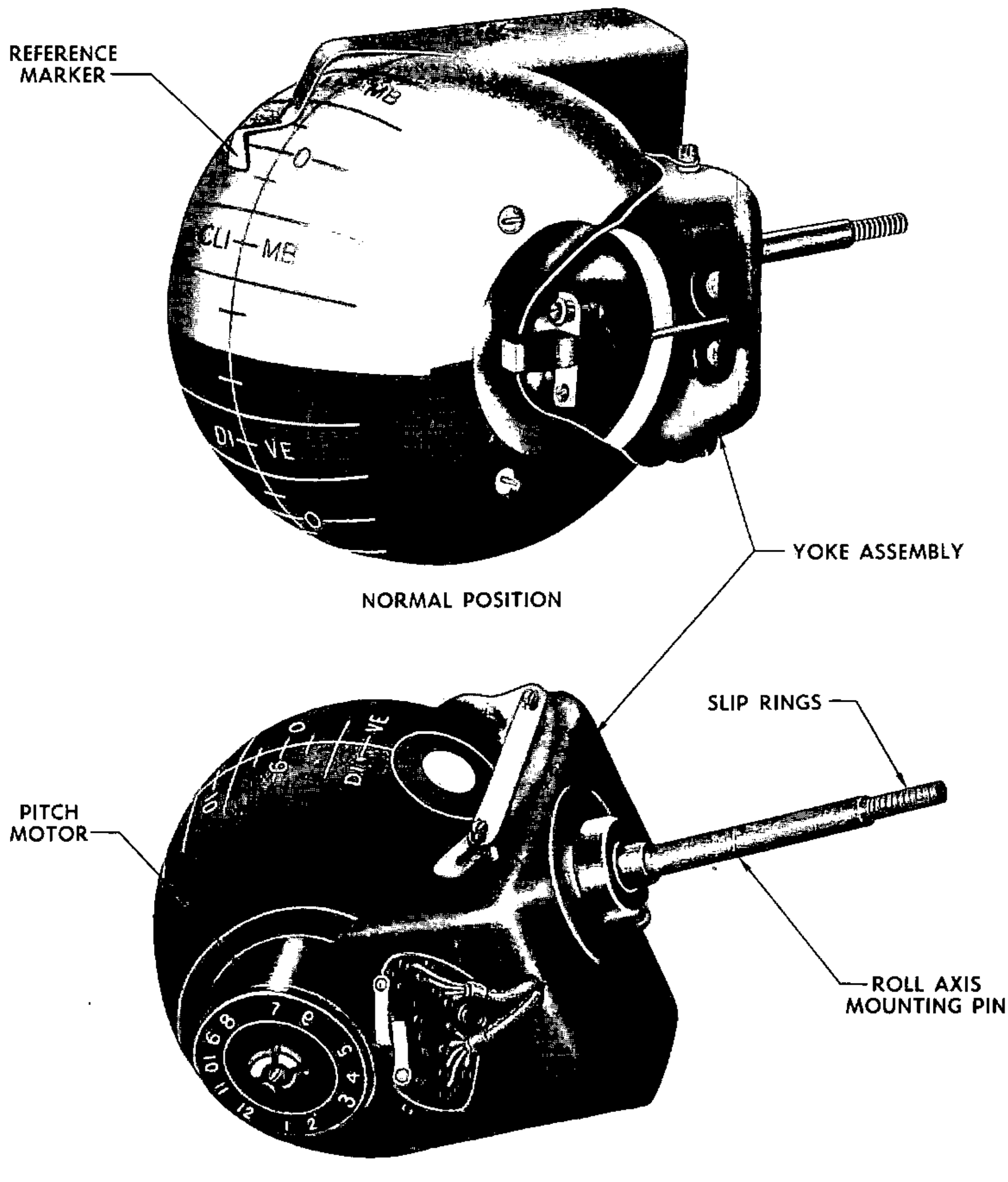


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Figure 2-21. Remote Attitude Indicator Sphere and Yoke Assembly

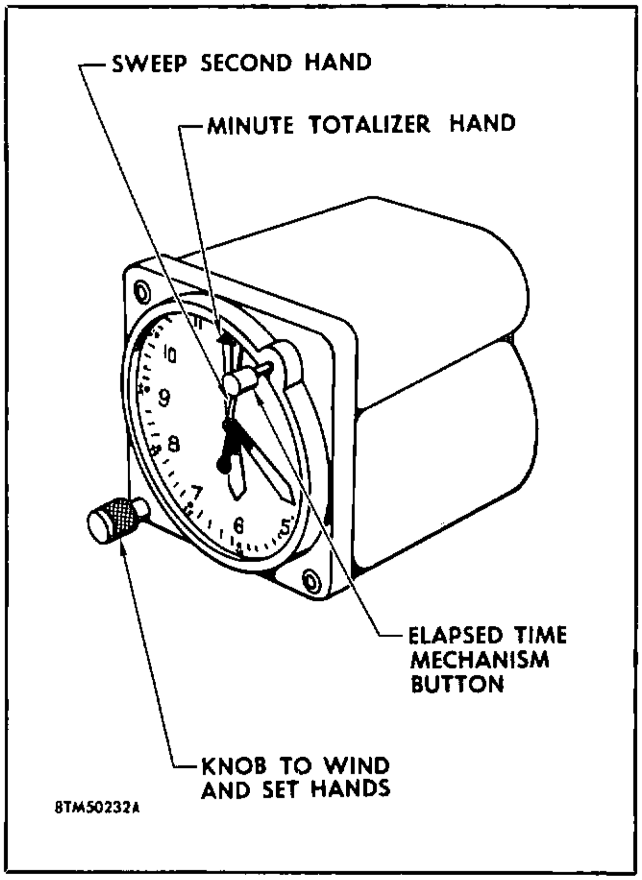

Figure 2-22. Clock

THE CLOCK.

If pilots were concerned with the time of day only for the reasons the rest of us are, the Air Force would be content to let them rely on their wrist watches. Valuable space would be saved on the instrument panel. Actually, the clock is an important instrument for a number of reasons. A brief discussion of some of them will help you to appreciate why it is important.

Everytime an airplane leaves the ground the pilot knows how much fuel he has on board. He also knows approximately how long he can stay aloft on that fuel supply. Accurate time checks enable him to know at any given moment just how long he has been flying, what his fuel consumption has been, and how much longer he can stay up. In a modern jet airplane, like the F-102A, the fuel supply is very limited and the consumption rate is high. Consequently, time in flight and the remaining fuel become critically important.

The clock is also vital to navigation. You may recall from the discussion on the airspeed and Mach number indicator how time in flight and distance covered are used to compute groundspeed—an important factor in aerial navigation. In addition, a pilot can determine his location if he knows how long he has been flying and his location in relation to his point of take-off. All of you understand the basic operation of a clock, so we will just take up the special features of the one that is used in the F-102A.

SPECIAL FEATURES.

The clock on the instrument panel of the F-102A has a mechanical eight-day movement. It is wound and set manually by the knob on the lower left corner of the instrument, as shown in figure 2-22. In addition to the hour and minute hands, there is a sweep second hand and a minute totalizer hand. These are part of the elapsed time mechanism which is controlled by the button on the upper right corner. The button consecutively starts, stops, and returns the sweep second and minute totalizer hands to zero (the 12-hour position). The minute totalizer hand counts the revolutions of the second hand up to 60 minutes, and then repeats. Thus, a pilot can easily time all or part of his flight just as you would time an athletic event with a stop watch. No special maintenance is required on the clock, except to see that it is wound, set, and running before each flight, and that it is keeping accurate time. The "fast" and "slow" adjustment on this clock is inside the case. You will not be able to make any adjustments in regulating the clock speed.

THE MAGNETIC STANDBY COMPASS.

You read in Chapter I about the earth's magnetic field and the basic principles of compasses. The pilot's compass, as the magnetic compass is frequently called, is one of the instruments that operates on these principles. Figure 2-23 shows its internal arrangement and face view. You will recall that there are certain known errors inherent in the operation of the magnetic compass for which the pilot must make allowances. For that reason it is only used as a *standby* instrument in the F-102A. If the radio magnetic course indicator malfunctions, the pilot can look at his standby compass in the peak of the canopy and get his directional information.

CONSTRUCTION AND OPERATION.

Two magnetized steel needles, which are mounted in a float that carries the compass card, make up the sensing element of the standby compass. A jeweled bearing supports the entire mechanism on a pivot assembly and uses a spring suspension to absorb shock. The inside of the case (compass bowl) is filled with a clear, acid-free kerosene which serves several purposes: it dampens the movement of the compass card to reduce oscillation; it provides buoyancy to lessen the weight of the float mechanism on the pivot bearing; and it keeps the moving parts constantly lubricated. A special chamber allows for

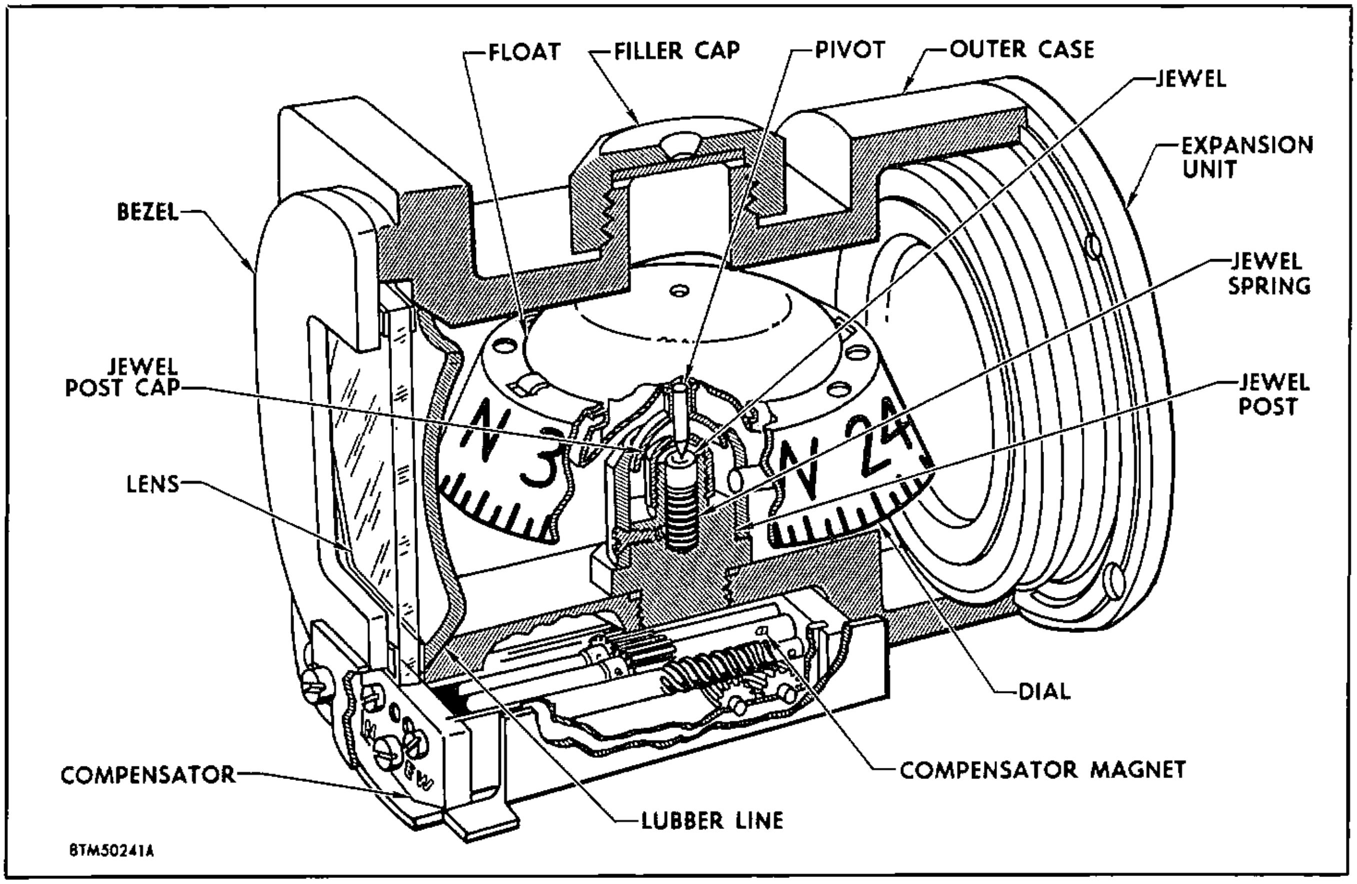

Figure 2-23. Magnetic Standby Compass Cutaway

expansion or contraction of the fluid due to temperature changes. If you ever detect bubbles in the fluid or find that it is leaking from the case, replace the instrument.

The compass card is graduated in five degree increments and is numbered every 30 degrees. Note that the last zero is omitted—number six indicates 60°, number 30 is 300°, and so forth. The 360° position is merely lettered "N," for north, and is 180°—directly opposite—from the north-seeking ends of the magnetized needles. This is necessary because you read the instrument from the end opposite the direction of flight. The lubber line is the reference line behind which the direction of flight is shown.

Above the compass bowl you will find a small lamp—you can remove it from the front of the instrument. A compensating mechanism is located below the bowl, behind a cover plate. It consists of two screws which move tiny magnets to compensate for magnetic disturbances within the airplane. These screws must never be tampered with unless the compass is being "swung" according to the procedures outlined in the appropriate Technical Orders, (see T.O. 1F-102A-2-9). At that time a deviation card is prepared to tell the pilot of any remaining errors that could not be removed by this compensation.

RADIO MAGNETIC COURSE INDICATOR.

The F-102A pilot has two instruments on which he can rely for basic directional information. We discussed one of them—the magnetic standby compass—earlier in this chapter. The other one is the radio magnetic course indicator. As the name implies, this instrument is both a radio compass indicator and a magnetic compass indicator. The dial rotates to show the magnetic heading directly under the fixed index pointer in the top center of the instrument face. The two needles, as shown in figure 2-24, indicate the magnetic direction of the VHF (Very High Frequency) station to which the radio is tuned. In the F-102A both needles point together.

The course indicator is an a-c synchro instrument and operates on the principles described in Chapter 1 in the discussion of synchronous instruments. The internal mechanisms of the indicator are shown in figure 2-25. The three synchro indicators within the instrument (one for each pointer and one for the dial) rotate so that they are always in the same relative positions as their transmitters.

The magnetic heading of the aircraft, shown by the rotating dial and index pointer, is received from the slaved gyro magnetic compass system. You learned

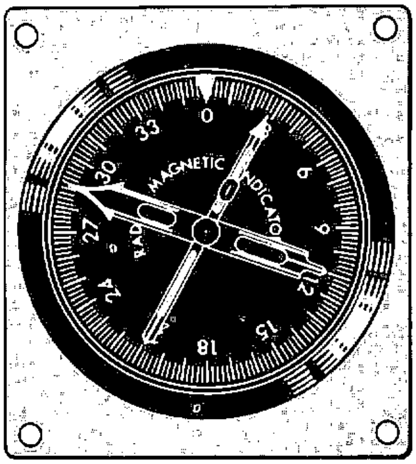

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Figure 2-24. Radio Magnetic Course Indicator

about this system in the preceding chapter. Both synchros which drive the pointers on the course indicator are positioned by signals from a transmitter in the bearing converter indicator known as the radio indicator control.

RADIO INDICATOR CONTROL.

The radio indicator control is located in the upper electronics compartment, just aft of the cockpit. Figure 2-26 shows you the face view of this control. The control contains a differential synchro generator that adds the station bearing to which the radio receiver is tuned to the magnetic heading from the slaved gyro magnetic compass. The resulting signal that positions the radio magentic indicator pointers is said to be "heading sensitive," because the pointers will indicate the correct station bearing on the dial even when the dial rotates.

You probably wonder why the radio indicator control is placed where the pilot can't see the dial. Actually, this device is only used as a control in the F-102A and the pilot does not need to see it. It is essentially a part of the radio equipment so we will not go into further detail on its mechanical characteristics.

USE OF THE RADIO MAGNETIC COURSE INDICATOR.

Past experience—and your study of earlier sections of this supplement—has taught you the use of the magnetic features of this indicator. A few words about the radio compass information will make you more familiar with its use.

An omnidirectional range (called ODR, VOR, or omnirange) is a VHF navigational facility that transmits signals in all directions. These directional signals are called radials of the "omnirange" and are expressed in terms of their magnetic directions from the transmitter. When an F-102A pilot tunes his navigation receiver to a particular omnirange station, the needles of his course indicator show the magnetic heading to that station. If he wants to fly to that point he merely turns the airplane until the pointers turn straight up to the top index marker. Then, the magnetic heading of the aircraft and the heading to the station will coincide. The pilot can fly directly to the station by holding this indication,

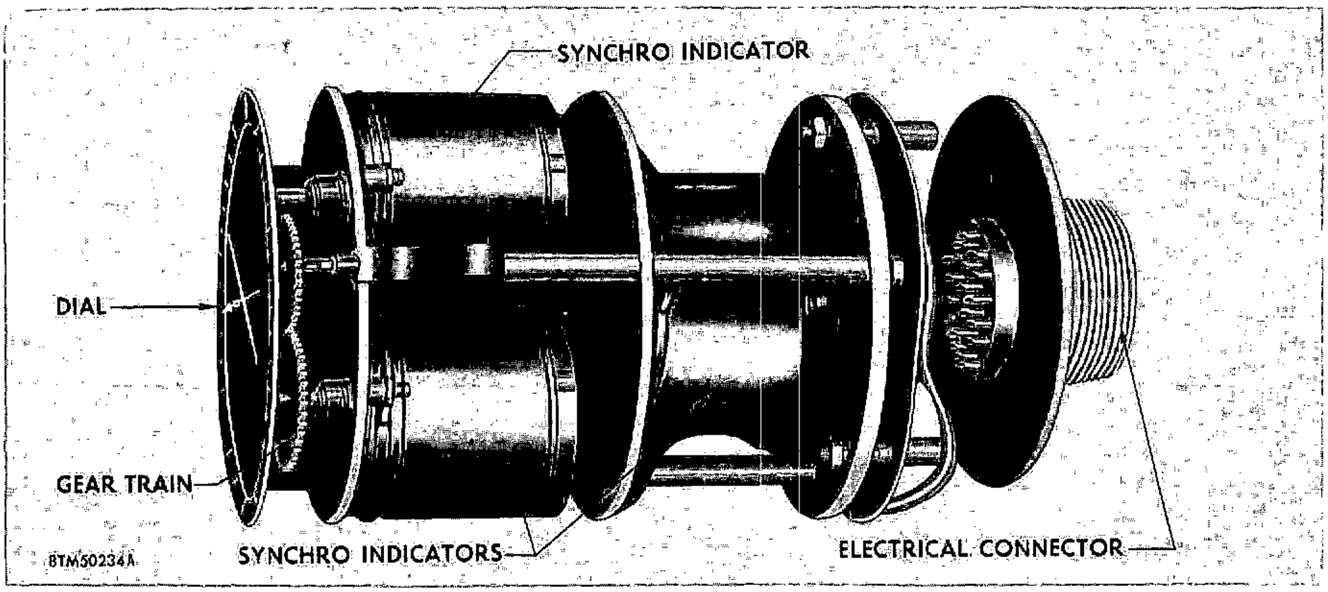

Figure 2-25. Course Indicator Internal Mechanism

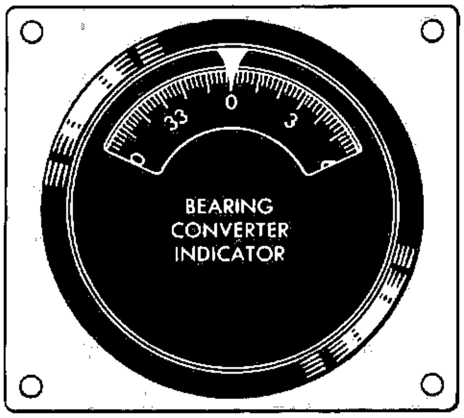

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Figure 2-26. Radio Indicator Control

(if a wind drift does not occur). If the pointers move off center while the magnetic heading remains constant, he knows he must correct for drift.

Another instrument found in the F-102A, the deviation indicator, is usually used to follow an omnirange radial once the desired radial has been intercepted. This indicator is discussed in the next paragraph.

DEVIATION INDICATOR.

It is often desirable to display several different types of related information with one instrument. To do so saves space on the instrument panel, and it is often convenient for the pilot using the information. The deviation indicator—sometimes called the course indicator—is an example of such a multipurpose instrument.

The deviation indicator (figure 2-27) provides the facilities of a cross-pointer indicator, a magnetic heading indicator, a course selector, and a marker beacon indicator. It will help you to understand these functions which are described later if you study the different indications on the indicator in the illustration.

The vertical pointer in the center of the indicator is the localizer or range pointer. It indicates lateral positional deviation when the plane is off to one side of a selected omnirange course, VHF-VAR (Very High Frequency—Visual Aural Range), or ILS (Instrument Landing System) runway localizer. The horizontal (glide path) pointer, also in the center, shows the position of the aircraft with respect to an ILS approach glide angle signal. Note that an OFF flag appears on each of the cross-pointers. These flags are visible when the appropriate signal is not strong enough to be reliable.

A short vertical pointer, shown deflected to the left in figure 2-27, indicates the magnetic heading of the aircraft relative to the selected course. The selected course is set in the upper window marked COURSE by the SET knob. Below and to the left of the course dial you see the TO-FROM indicator. It shows whether the airplane is flying on a "TO" or "FROM" radial. The heading of the aircraft has nothing to do with the TO-FROM indication. It merely tells whether the airplane will go toward or away from the station if the selected course is flown. The pilot can fly to a station on a FROM indication by flying the reciprocal ($\pm 180°$) of the selected heading, and reversing his normal procedures.

The light that you see on the upper right hand corner of the indicator is the marker beacon indicator. As the plane crosses a marker transmitter, the light flashes the coded identification signal of the marker. By now you may be somewhat puzzled as to just how this instrument works. You are probably wishing that it didn't combine so much information in its indications. To eliminate some of this confusion, let's consider a few examples where the various indications are used.

OMNIRANGE FEATURES.

Suppose an F-102A pilot wants to fly a course where an omnirange navigation facility is in operation. He first tunes his VHF navigation receiver to the frequency shown on his charts for that station. An aural signal will come through his earphones, identifying the station. Assume that he left the air base control zone on a heading of 40°, as shown at Position No. 1 in figure 2-28, and planned his course as 315°. You will note that the illustration shows the deviation indicator and the radio magnetic course

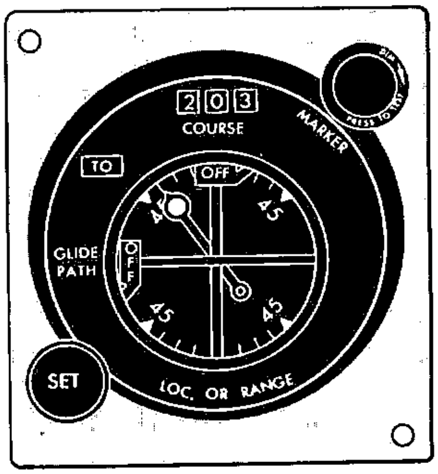

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Figure 2-27. Deviation Indicator

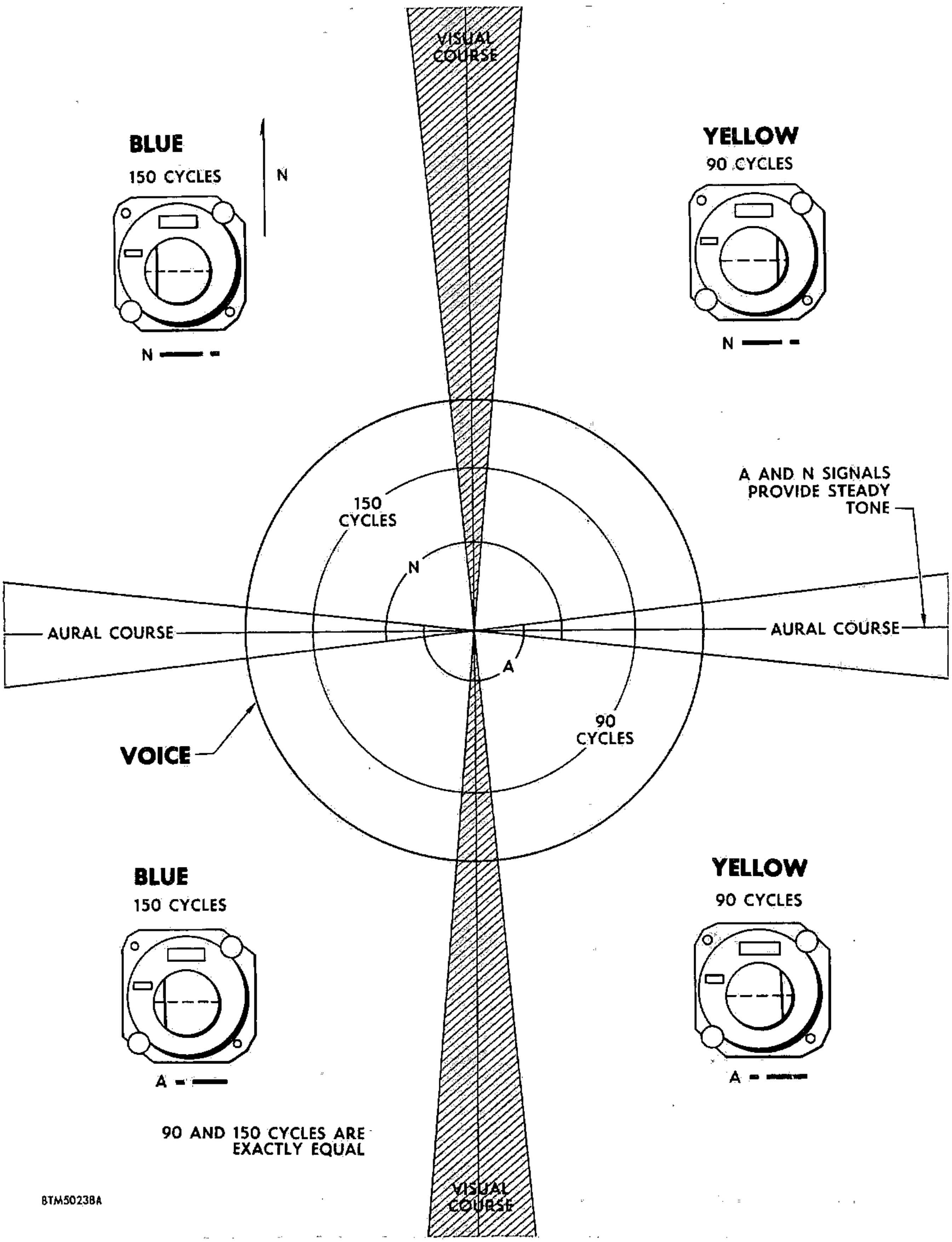

Figure 2-29. Typical Indicator Presentations Using VHF-VAR Facilities

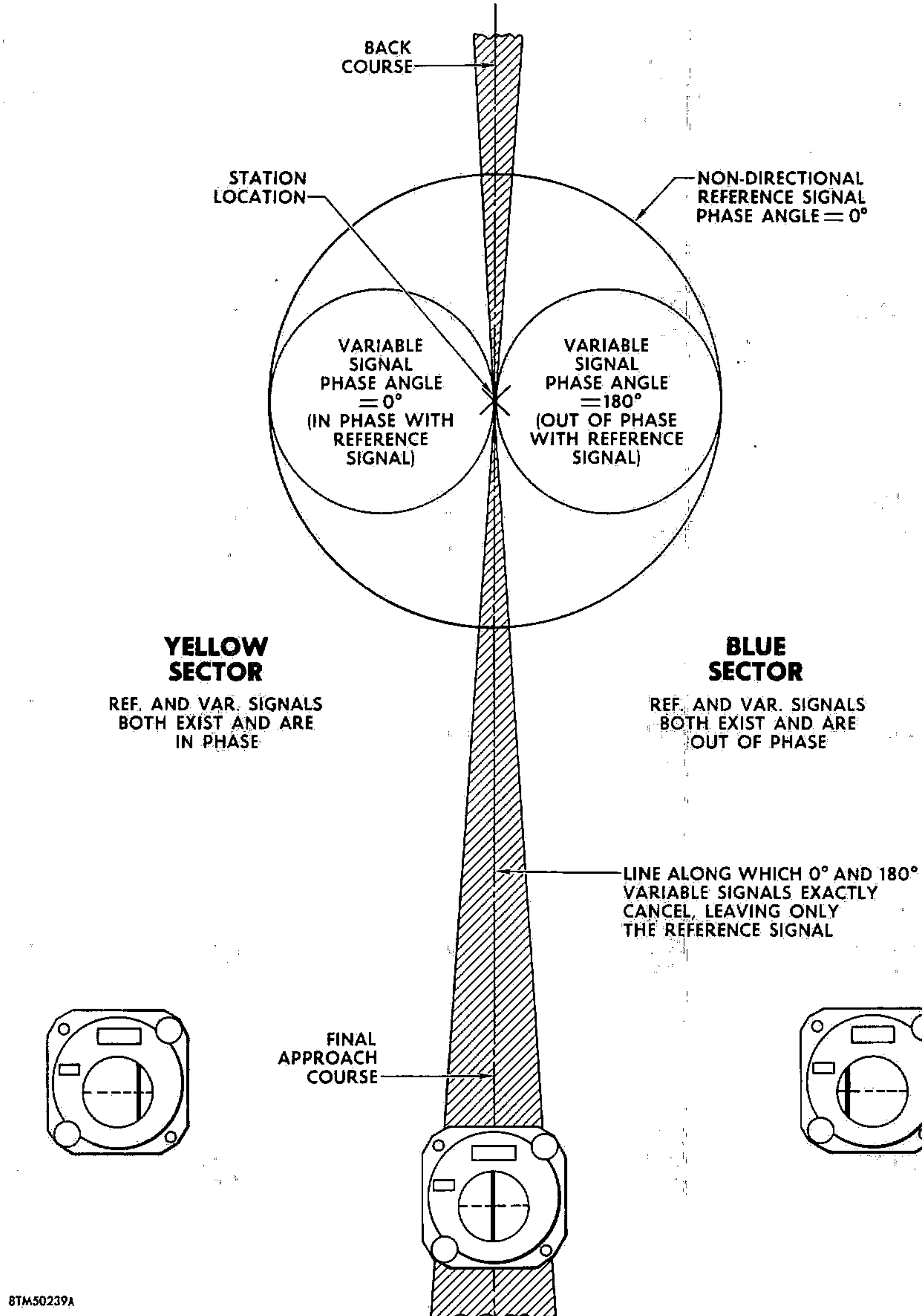

Figure 2-30. ILS Presentation with a Phase Comparison Localizer

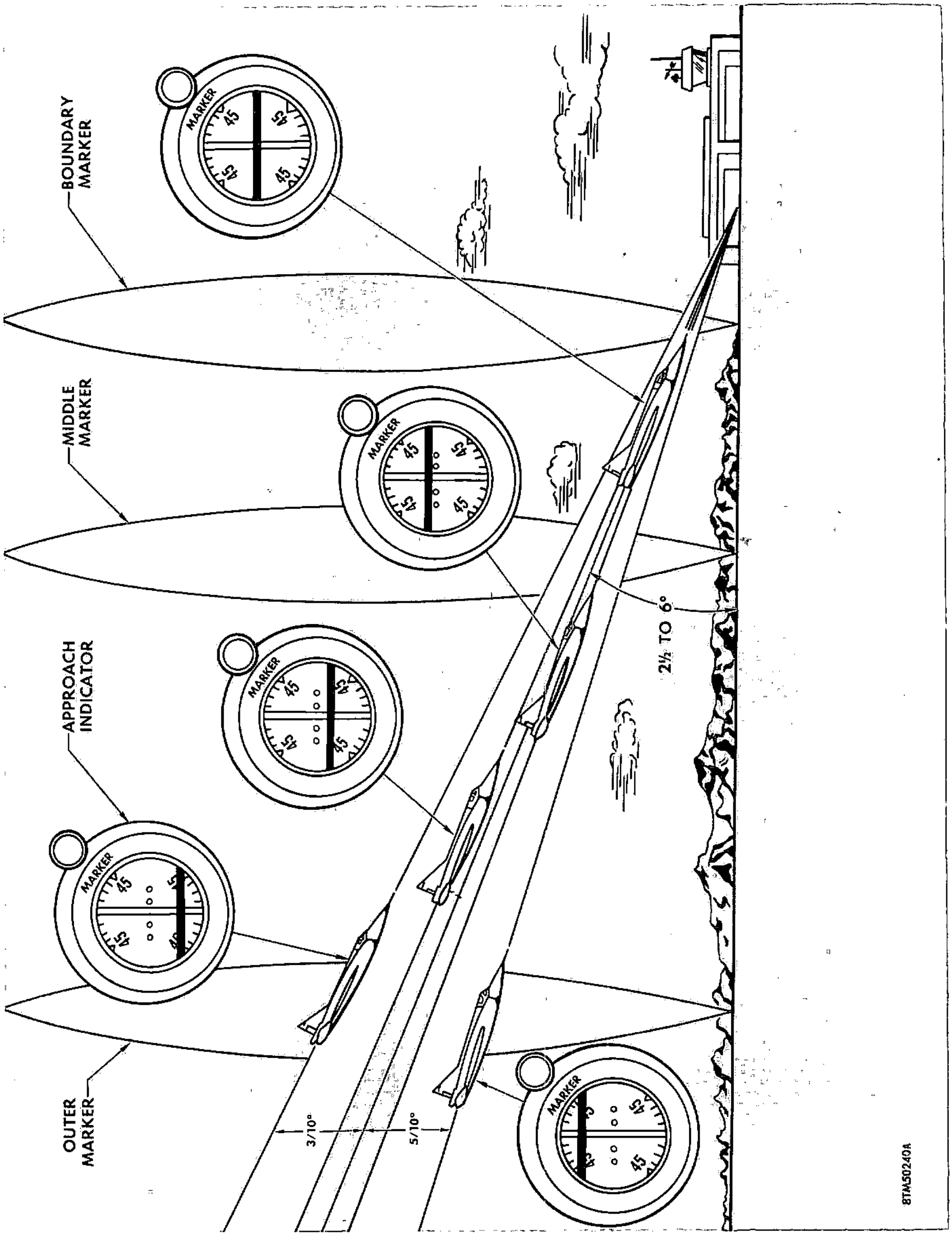

Figure 2-31. Glide Path Indications

one is nondirectional and the other two are directional signals with differing phase angles. Both types of localizers provide an imaginary vertical plane extending along the centerline of the runway in the direction of the final approach. With either type, the pilot must always turn toward the vertical pointer when it drifts from the centerline.

The relative heading pointer is used on an ILS approach in the same way it is used to fly omnirange. When the pilot sets the runway heading into his COURSE selector window, the pointer aids him in establishing and maintaining the correct wind drift angle. We've followed our F-102A pilot to within several miles of the runway, and he is lined up properly with the centerline. Now let's see how he keeps from overshooting or undershooting the field.

Glide Slope.

Another receiver in the F-102A picks up a signal from the glide slope equipment on the ground. This signal is similar to the tone localizer signal in that it also consists of one pattern modulated at 150-cycles and another at 90-cycles. The signal, shown in figure 2-31, is sent from the runway at a shallow angle that corresponds to the airplane landing glide slope. When the airplane is flying in the area of equal signal strength between the two patterns, the horizontal pointer of the deviation indicator is in the center. If the airplane drops too low, the pointer goes above the centerline; when it is too high, the pointer goes below the centerline. Therefore, the pilot must always correct toward this pointer too. The illustration shows how these positions appear on the indicator.

Marker Beacons.

Note in figure 2-31 that there are also three marker beacons; the outer, middle, and boundary markers. (Some ILS installations have only two.) Each marker beacon transmits a different signal. The marker receiver in the airplane picks up these signals as the airplane passes over the beacons and flashes a coded signal on the marker beacon indicator. This tells the pilot how close he is to the runway. When he sees the signal from the boundary marker flash on the indicator, it is time to "flare" out for the touchdown.

You have seen how the various components of the deviation indicator work, but no mention has been made regarding the maintenance of the indicator or its parts. This was not just an oversight. Aside from the routine cleaning of the instrument dials, all maintenance work on this instrument is accomplished by the radio specialist.

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Chapter III

ENGINE INSTRUMENTS

Contents — *Page*

The engine instruments are those instruments, installed in an airplane, that show how the power plant is operating and assist the pilot in proper operation of the engine; and conversely, they indicate if the engine is not operating correctly and may reveal at times which part of the engine is not operating correctly. A modern jet airplane requires some engine instruments that are not needed in airplanes equipped with reciprocating engines. On the other hand, some of the instruments that are used to show the operating conditions of piston engines are not needed in jets. You will find that the F-102A has fewer engine instruments than some airplanes equipped with piston engines, but all of the instruments are vitally important in the proper operation of the engine.

The F-102A engine instruments and their positions on the instrument panel are shown in figure 3-1. These instruments include the pressure ratio indicator, tachometer, exhaust temperature indicator, exhaust nozzle position indicator, fuel flow meter, and fuel quantity indicator. The alternate pressure ratio indicator which is used on some models of the F-102A is also shown at the bottom of the illustration. We will discuss each of these instruments and their systems in this chapter.

AFTERBURNER EXHAUST NOZZLE POSITION INDICATOR.

A jet airplane that is equipped with an afterburner always requires some means of adjusting the exhaust nozzle opening at the aft end of the power plant. An adjustable nozzle on the afterburner is necessary since the amount of pressure and heat produced by the power plant varies greatly between operations with afterburning and without afterburning. For normal operation without afterburner, the nozzle is closed (not completely shut, just restricted). In this position the opening is the best size to provide maximum exhaust velocity for the existing temperature and pressure conditions at the tailpipe. If the opening is too large, maximum thrust is not developed.

When the afterburner is ignited, the temperature at the nozzle increases greatly. This temperature increase causes additional expansion of the exhaust gases. If the nozzle opening is not enlarged, the pressure and temperature in the afterburner becomes excessive—you can understand why such a condition must be avoided. Therefore, the F-102A has an adjustable exhaust nozzle and an indicator to tell the pilot whether the nozzle is in either the OPEN or the CLOSED position. This indicator is located on the right side of the main instrument panel, adjacent to the pressure ratio indicator.

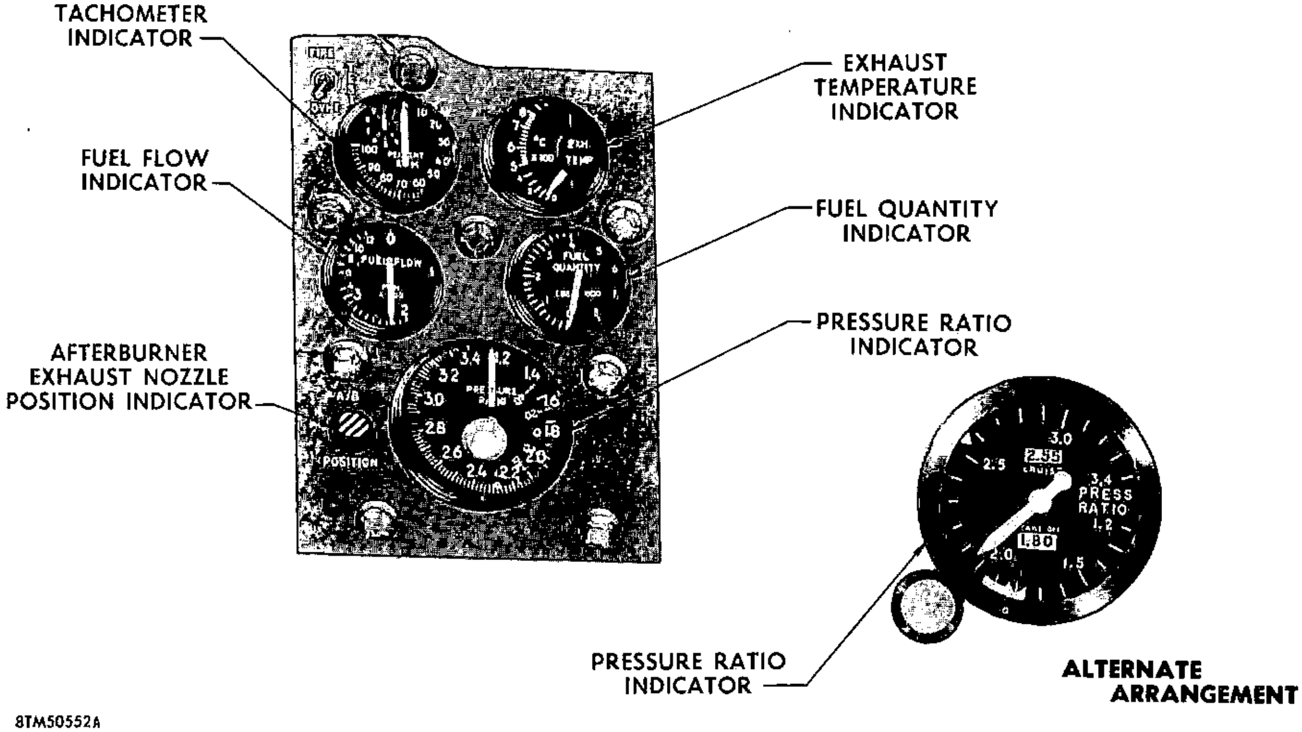


Figure 3-1. F-102A Engine Instrument Panel

The exhaust nozzle position indicator is a very simple instrument: its construction is shown in figure 3-2. It consists of two solenoids which rotate a spring-centered, three-position card within a small sealed case. A window in the front of the case shows one of the three positions of the card.

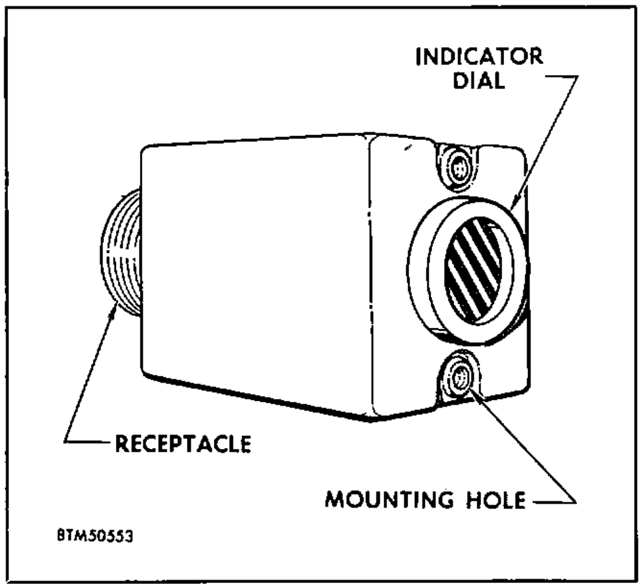


Figure 3-2. Afterburner Exhaust Nozzle Position Indicator

HOW IT WORKS.

There are three indications which the exhaust nozzle position indicator can give; the one which appears in the window at any particular time depends on whether one of the two solenoids is energized or neither solenoid is energized. In figure 3-3, you can see the three types of indications and the corresponding electrical schematic that causes each of them to appear. Note that there are two switches in the system, only one of which can be actuated at a particular time. In this diagram you see how the *open* solenoid is energized when one of the switches closes. The circuit is completed to ground from the airplane's 28-volt, d-c system, represented by EMF (electromotive force).

The middle diagram shows both switches open, so neither solenoid is energized. Diagonal lines appear on the dial because a spring holds the rotating card in the center position. This indication appears whenever both switches are momentarily open (as the cam shown in figure 3-4 moves between the OPEN and CLOSED switches) or there is a power failure (in which case the spring automatically returns the

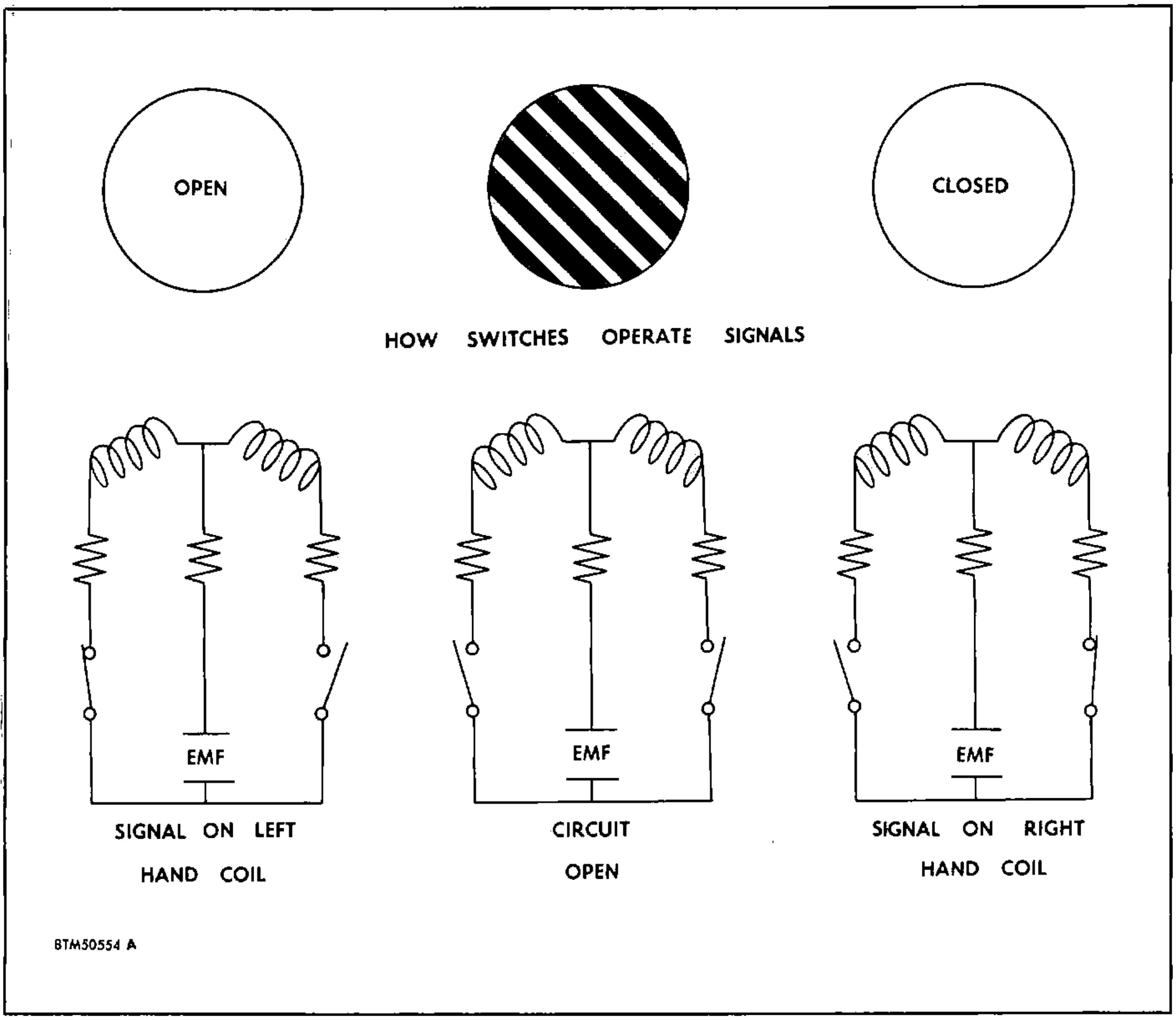


Figure 3-3. Afterburner Exhaust Nozzle Position Indication Circuit

card to the center position, although the nozzles could be in either the OPEN or CLOSED position). In the third diagram the circuit is completed through the other switch, energizing the *closed* solenoid. The CLOSED indication will appear as long as that switch is closed and the power suppy is not interrupted.

Now, let's discuss the two switches that are located on the power plant. As you can see in figure 3-4, the switch assembly is attached to the right side of the engine just below the compressor bleed valve. Two cables connect the switch assembly to the nozzle positioning mechanism. The upper cable is the nozzle position cable; the lower cable is a temperature compensating cable. Both are attached to spring plates in the switch assembly and serve to rotate a cam. The enlarged view of the switch assembly shows how this cam actuates the switches. Notice that the actuating arm of the CLOSED switch is depressed by the cam. In this position the CLOSED circuit of the indicator is energized as shown on the right of the circuit diagram of figure 3-3. The dotted lines show where the cam stops when the afterburner exhaust nozzle is open. This position results in an OPEN indication because the *open* solenoid is energized. It is obvious from this illustration that both switches are open momentarily as the cam rotates from one switch to the other. Thus, the diagonal lines are shown on the indicator very briefly each time the nozzle is opened or closed.

Remember that the switching mechanism does not open or close the afterburner—that's accomplished automatically when fuel pressure in the afterburner fuel control reaches certain limits. The sole purpose

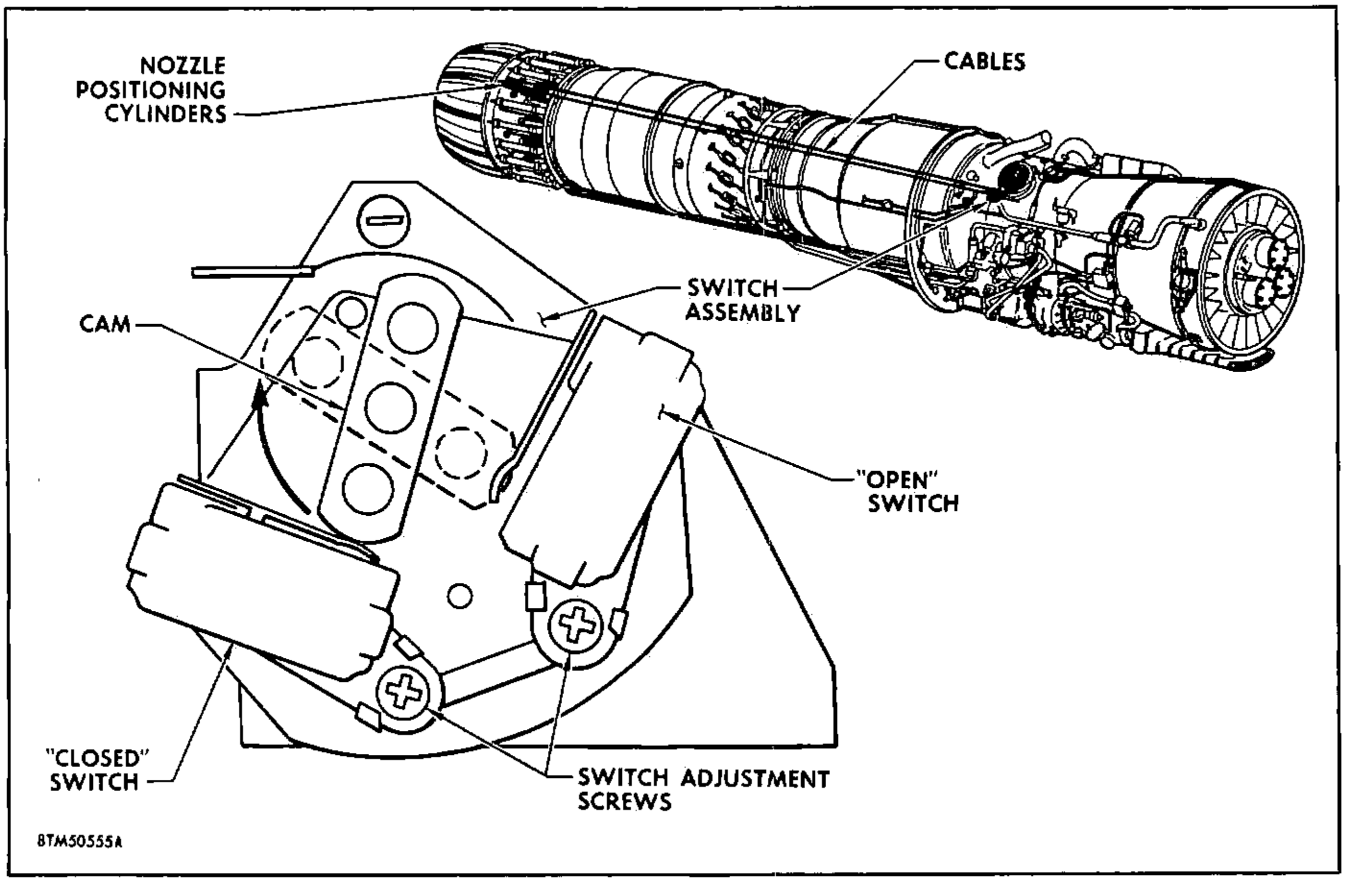


Figure 3-4. Afterburner Exhaust Nozzle Position Switches

of the switch assembly is to energize the correct indicator solenoid at the right time. If it does not accomplish that purpose, use the special rigging gage described in T.O. 1F-102A-2-9, to make the necessary adjustments. Before replacing the switches or indicator in the nozzle position indicating system, always check the system's circuit breaker. This circuit breaker is located on the left-hand aft circuit breaker panel in the cockpit of the airplane.

EXHAUST TEMPERATURE INDICATOR.

The exhaust temperature of a jet engine, like the cylinder head temperature of a reciprocating engine, is a good indication of the overall operating temperature of the engine. By way of further comparison, the jet engine exhaust temperature indicator is very similar to the instrument used to measure the cylinder head temperatures of piston engines. Both types are thermocouple-thermometers which measure the difference between electrical potentials of two metals in contact with each other. Figure 3-5 shows the exhaust temperature indicator and one of the four thermocouples used in the F-102A exhaust temperature indication system.

THERMOCOUPLES.

A thermocouple is a combination of two wires, each wire made of a different metal with each having a different electrical potential. If two such wires are connected together at one end and that junction point is heated above the ambient temperature of the opposite end of each wire, the joined wires become a source of electricity, the potential of which varies with the temperature. This physical phenomenon known as thermo-electric effect is the principle of every thermocouple.

The thermocouples used in the exhaust temperature indicating system are made of chromel-alumel material. They are mounted in probes which project into the tailpipe section of the engine just aft of the turbines, and are approximately 90° apart (see figure 3-6). Each thermocouple is connected in parallel, with the terminals connected to each of the two leads which carry the current to the indicator. These leads are also of chromel and alumel material. As you can see in figure 3-6, a calibrating resistor is included in the system. Note that it is located on the right side of the cockpit, above the master warning control box. You can use an ordinary Wheatstone bridge-type tester to determine the correct adjustment of the resistor.

THE INDICATOR.

The temperature of the thermocouples is shown on the dial of the indicator in degrees Centigrade—the scale is calibrated from 0° to 1000°C. As you can see in figure 3-5, there are no external controls or adjustments on the indicator. Internally, the indicator consists primarily of a moving coil, mounted on pivots, within a curved permanent magnet. Rotation of the coil is limited by two springs, one on each end of the coil. A pointer moves with the coil to show the temperature indication. The entire indicator is sealed and filled with helium.

Now take a look at the schematic illustration, figure 3-7. Note that the leads from the thermocouple connect to the coil through the springs, making a complete circuit. As you know, when current flows through a conductor, such as this coil, a magnetic field is set up. This magnetic field around the coil has both strength and direction, just as the field around a permanent magnet. Note also that the coil is situated directly between the ends of the curved permanent magnet. You learned in Chapter I how the magnetic flux around such a magnet is concentrated between the two ends. Thus, the coil tends to take a definite position between the ends of the permanent magnet.

You also remember that like poles repel and unlike poles attract, so the coil tends to turn until each of its poles is close to the opposite pole of the permanent magnet. If there were no restraining springs, the coil would always line up perfectly (with the poles in the normal relative positions) whenever there was a temperature difference between the "hot" end (thermocouple) and the "cold" end (indicator) of the system. Obviously, such an indicator would be of no value since it would always read the same. By the use of springs of the correct tension, the coil is only permitted to move an amount proportional to the strength of the magnetic field around it, which, as we mentioned earlier, varies with the temperature differential.

Temperature Compensation.

The coil springs we just discussed serve another purpose: they are temperature compensators. You will recall that the amount of rotation of the coil depends on the strength of its magnetic flux, which in turn is proportional to the difference in temperatre between the thermocouple and the indicator. But that isn't exactly what we want to know. If the exhaust temperature is 600° Centigrade, we want the indicator to say 600° Centigrade, regardless of the temperature in the cockpit. You can see then that the indicator must be set to read the cockpit temperature first so that the additional rotation of the coil will cause it to read actual exhaust temperature: The springs accomplish this for us because they are

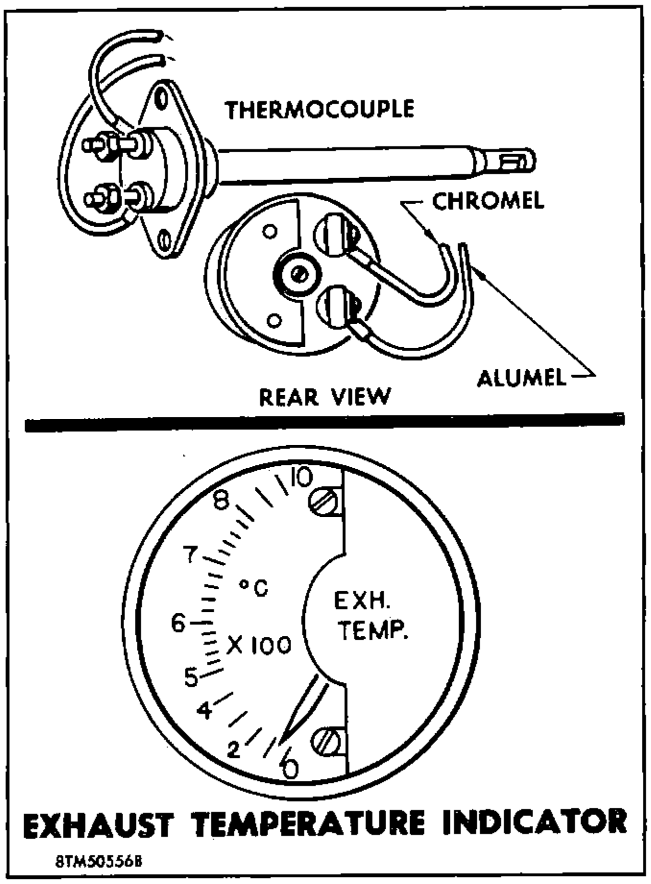


Figure 3-5. Exhaust Temperature Indicator and Thermocouple

made from laminations of different metals which react differently to temperature changes. Figure 3-8 shows how these bimetallic springs work.

Note that the strip of brass in the laminated metal expands more than the strip of iron when heat is applied, causing the laminated strips to bend. In the same manner the springs in the exhaust temperature indicator tighten up or straighten out with changes in cockpit temperature. Thus, the indicator pointer reflects the total of the temperature at the indicator plus the difference between the temperatures at the indicator and the thermocouple. In this way the coil, and therefore the pointer, is rotated to indicate the cockpit temperature. The additional rotation, caused by the difference in temperature between the indicator and thermocouple, brings the pointer to a position proportional to the total exhaust temperature. If you disconnect the thermocouple leads, the indicator will still show the approximate cockpit temperature.

Another temperature compensation problem results from the variations in electrical resistance of most metals with temperature changes. At a given temperature difference, the voltage generated in the circuit will produce a current inversely proportional

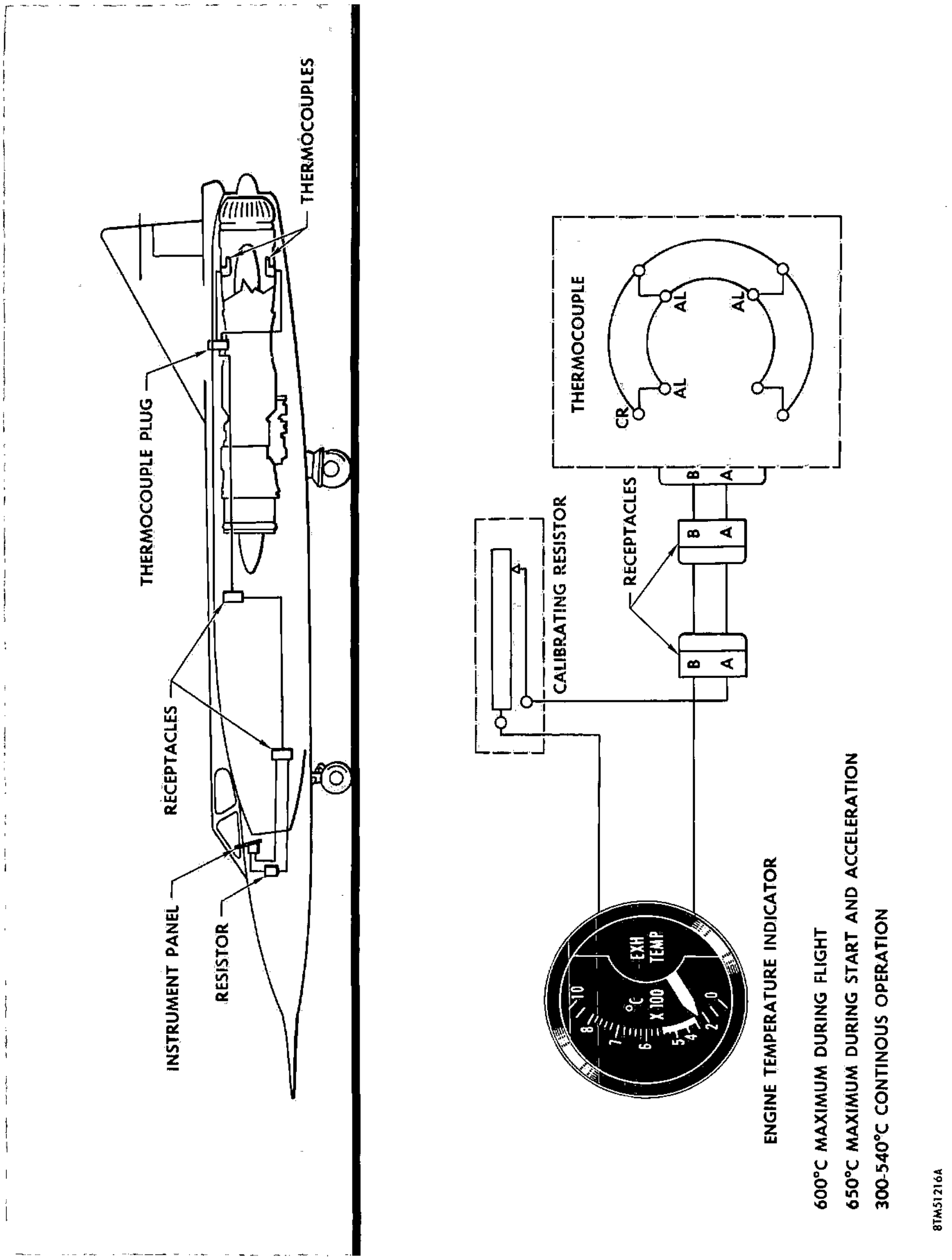


Figure 3-6. Wiring Diagram of Thermocouple

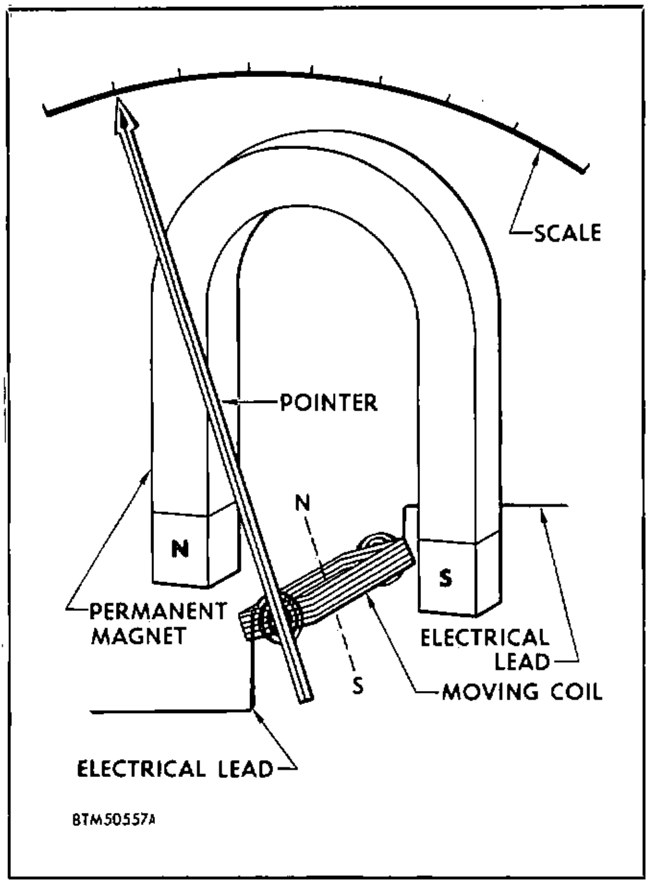


Figure 3-7. Exhaust Temperature Indicator Operational Schematic

to the resistance (Ohm's law). For any particular temperature difference between the "hot" and "cold" ends of the system, the current must always be the same. Therefore, a "neutralizer" is included in the indicator. This neutralizer is a resistor in which the resistance becomes less as the temperature increases. In that way it keeps the total resistance of the system constant for any temperature difference.

SYSTEM CALIBRATION PROVISIONS.

We have discussed the importance of keeping the resistance of the system completely constant so the indicator will receive the right amount of current. There are several things that might alter this resistance, thereby affecting the accuracy of the indication. For example, if you replace any of the components of the system, as would happen when you change engines, there could be a slight change in the resistance. Even changing a terminal on the leads could alter the resistance. A special testing unit (SE 0783) is required to test the accuracy of the complete system.

FUEL FLOW INDICATING SYSTEM.

Any large jet engine uses tremendous quantities of fuel. The rate at which the fuel is used varies greatly, depending on the power setting. Since space and weight restrictions limit the amount of fuel that can be carried, the jet pilot is always vitally concerned with how fast the supply is being consumed. Without this information he cannot estimate accurately how long he can stay away from the base by just knowing the amount of fuel remaining in his tanks. In addition, the rate of fuel flow is an indication of the efficiency of the engine. For these reasons the F-102A instrument panel includes a fuel flow indicator. Besides the indicator, the system includes a transmitter which is located on the engine. Both of these units are shown in figure 3-9.

THE FUEL FLOW TRANSMITTER.

The transmitter for the fuel flow indicating system is attached to the right side of the power plant (as shown in figure 3-10) and connects to the outlet fuel line from the fuel regulator. An a-c synchro transmitter is contained within the unit to send signals to the indicator in the cockpit. You learned in Chapter 1 how synchro transmitters operate, so we will just discuss the gage unit which drives the synchro. The operational schematic, figure 3-11, will help you to see how the flow transmitter works.

Fuel flowing through the transmitter enters the port on the lower right and leaves through the one on the lower left as shown by the arrows. Note that a hub with a vane projecting above it is mounted in the flow area. A light spring within the hub resists the force of the fuel flow, thus tending to keep the vane in the upstream position at all times. The actual position taken by the hub and vane assembly is, of course, dependent on the rate of flow (or the pressure it exerts) of the fuel which surrounds it. Therefore,

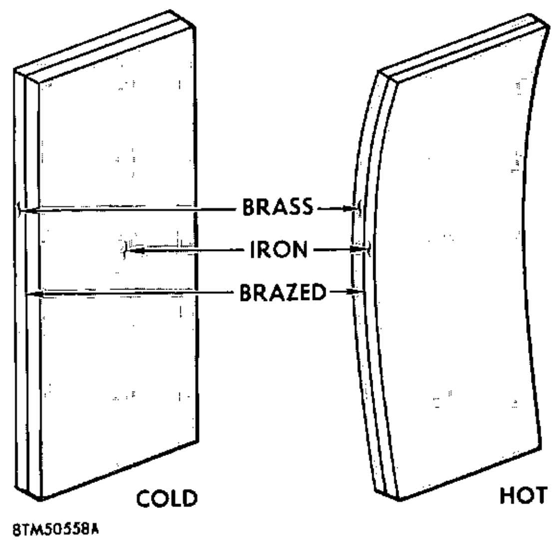
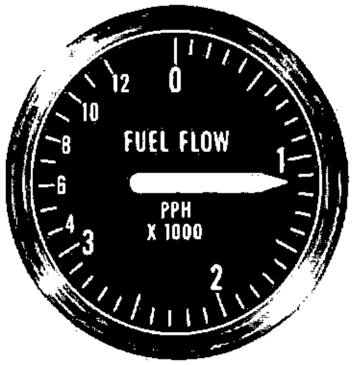
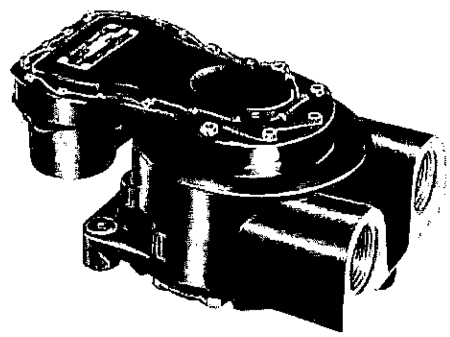


Figure 3-8. Effect of Temperature Changes on Bimetallic Strip



FUEL FLOW INDICATOR



FUEL FLOW TRANSMITTER

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Figure 3-9. Fuel Flow Transmitter and Indicator

we can call the vane and hub assembly the "*gage unit*" of the fuel flow transmitter.

Now, let's follow the train of action to see how the position of the vane and hub is carried to the synchro. Note that there is no direct mechanical connection between the gage unit, which does the measuring, and the transmitting unit, which signals the indicator in the cockpit. The shaft of the hub is not geared to the sector shaft. Instead, a permanent ring magnet on the hub shaft surrounds a permanent bar magnet which drives a pinion gear. This forms a kind of floating drag arrangement between the hub, the spring, and the magnet assembly. As you know, the relative positions of these magnets will tend to stay the same. When the gage unit rotates the ring magnet, the bar magnet turns with it so that the opposite poles of the two magnets are always lined up. You can see then how the rotation of the bar magnet and pinion moves the sector shaft and positions the synchro. This method of positioning the synchro permits the electrical section of the fuel flow transmitter to be isolated from the fuel-carrying section.

THE INDICATOR.

The fuel flow indicator is calibrated to show the rate of flow from 0 to 12,000 pounds per hour. As you can see in figure 3-9, the dial is graduated every 100 pounds up to 3000 pounds, and in 1000-pound increments from 3000 to 12,000 pounds.

Since the fuel flow transmitter positions a transmitting synchro, it is obvious that the indicator is a synchro instrument containing the receiving motor. Power to operate this synchro system comes from the airplane 26-volt, 400-cycle a-c source. You have already learned the fundamentals of synchros, so we won't go into further detail on this one.

If you should have any reason to doubt the accuracy of the indicator, disconnect the transmitter and plug the leads into a master synchro transmitter. If the indicator pointer smoothly follows the movement of the test transmitter, the problem lies in the airplane's fuel flow transmitter. There are no external provisions for adjusting either the indicator or the transmitter, so you must replace the faulty unit.

THE FUEL QUANTITY INDICATING SYSTEM.

The F-102A uses an electronic, capacitance-type fuel indicating system. This system measures the weight of the fuel instead of the volume. The quantity indicator in the cockpit can register the total weight of the fuel in all six tanks or the weight of the fuel in either of the No. 3 tanks. An electrical selector switch in the cockpit allows the pilot to obtain any one of the three different readings. The quantity indicating system is not designed to indicate fuel quantities for the jettisonable wing tanks which may be attached to the airplane wings.

The purpose of the fuel quantity indicating system, of course, is to keep the pilot informed of the existing fuel supply at all times. To accomplish this purpose the system uses several different components—12 tank probes, an amplifier bridge, a fuel-type compensator, a selector switch, two quantity relays, and an indicator. As you can see in figure 3-12, these components are located in several different places in the airplane. The amplifier bridge and the quantity relays (not shown) are in the nose wheel well, and the selector switch and indicator are in the cockpit. Two quantity probes are installed in each tank, and the fuel-type compensator is situated in the left No. 3 tank. A circuit breaker for the fuel quantity indicating system is located on the left-hand aft circuit breaker panel in the cockpit.

The components mentioned in the preceding paragraphs, the tank probes, the selector switch, the selector relays, the quantity indicator, and the bridge amplifier are shown schematically in figure 3-13. You can find a detailed explanation of their operation in the Fuel System Training Supplement.

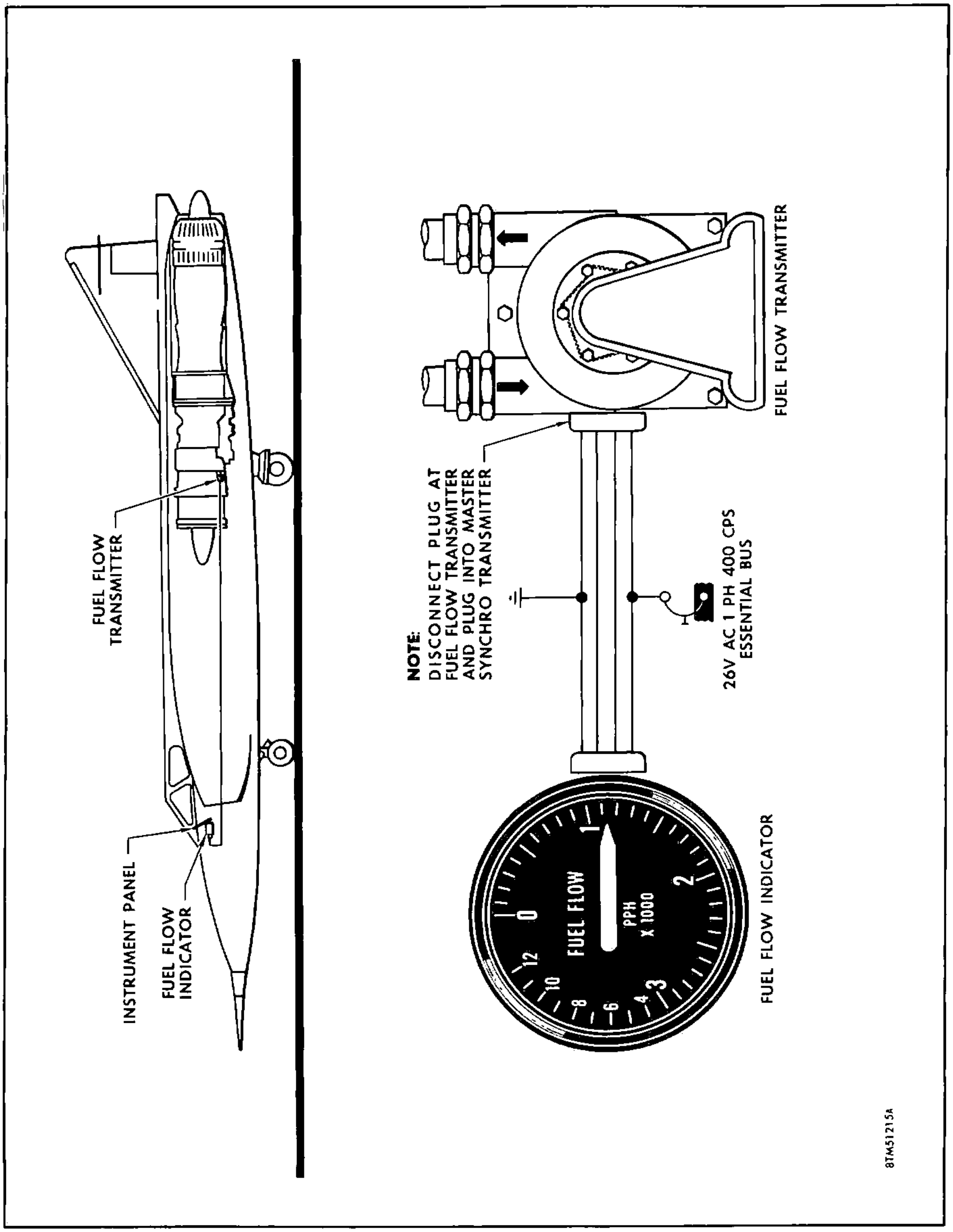

Figure 3-10. The Fuel Flow Transmitter and Indication Schematic

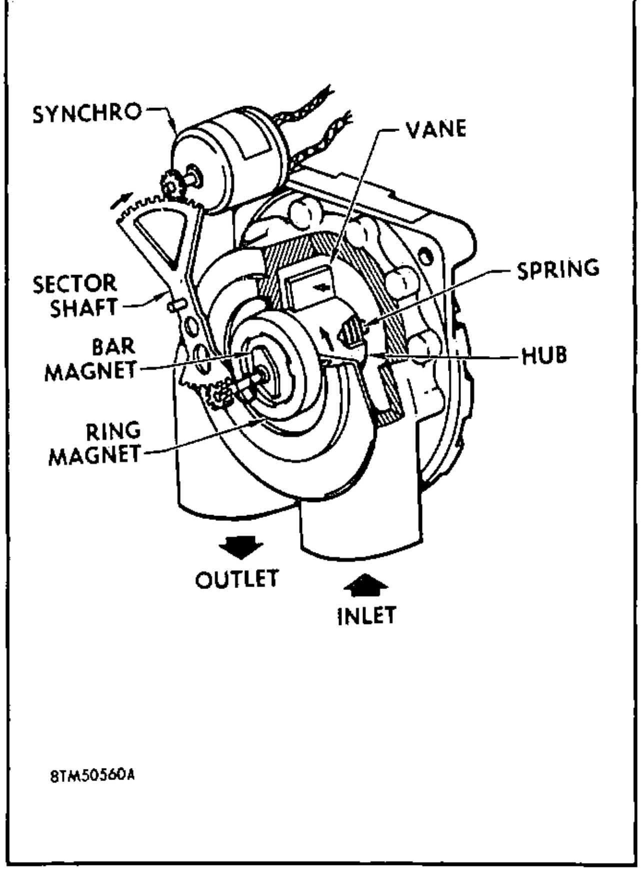

Figure 3-11. Operational Schematic of Fuel Flow Transmitter

SYSTEM OPERATION.

The electronic, capacitance-type, fuel quantity indicating system operates on the principle that the weight (density) of the fuel is directly proportional to the fuel's dielectric constant. If the indicating system can measure the dielectric properties of the fuel, it can translate this measurement into a weight measurement. To understand how this is accomplished, you should be familiar with the principles and construction of the electrical condenser.

Condensers store electrical energy, and for this reason they are commonly called *capacitors*. From this term the name *capacitance-type* indicating system is derived. The size of any capacitor determines the amount of electrical energy that it can store. This amount of stored energy depends upon three things: the area of the capacitor plates, the distance between the plates, and the dielectric material between the plates.

In a general sense, the tank probes in the F-102A indicating system are capacitors. They act as the sensing elements for the system. When the tanks are full of fuel, the tank probes (or capacitors) can store more energy than when the tanks are empty. The bridge amplifier in the indicating system measures the difference in the amount of stored energy, and then translates this electrical measurement into a dial reading on the indicator. Now we are ready to discuss the individual components. The F-102A Fuel System Training Supplement goes into considerable detail on these units, so we will only cover them briefly here.

Tank Probes.

There are two probes in each of the six fuel tanks of the F-102A. These probes are mounted to the wing structure, as shown in figure 3-14. It is necessary to have two probes in each tank, to provide an accurate quantity indication when the airplane is in attitudes other than level flight.

Each probe consists of an inner, an intermediate, and an outer tube. The fuel and air in the tank are admitted between the inner and intermediate tubes. These tubes are the plates of the capacitor. The dielectric is the fuel and/or air between the plates. When the tank is full, the space between the tubes is filled with fuel. Air replaces the fuel as the fuel level of the tank lowers. Thus, the amount of capacitance of each probe is greatest when the tank is full, and reduces as the fuel is used.

Fuel-Type Compensator.

As you know, the F-102A can use either JP-4 jet fuel or a blended mixture of oil and aviation gasoline. The dielectric constant of these two liquids is not the same, so the fuel quantity indicating system must be able to allow for this difference. A fuel-type compensator automatically determines which fuel is being used and transmits this information to the amplifier bridge. The type compensator is located in the left No. 3 tank, so it is able to detect the dielectric constant of the fuel even when the supply is low.

The Amplifier Bridge.

The "brain" of the fuel quantity indicating system is the amplifier bridge. This unit performs two basic functions. First, it compares the capacitance from the tank probes with a reference capacitance. This reference capacitance is controlled by the fuel-type compensator (see figure 4-8, Fuel System Training Supplement) so that it will be the correct reference for the type of fuel being used. Secondly, the amplifier bridge amplifies the signal—proportional to the difference between the reference capacitance and the tank probe capacitance—to provide the suitable signal to the indicator motor. Figure 3-15 shows how the amplifier bridge appears with the end cover removed. The three adjustable potentiometers which you see on the end of the unit are used in calibration—we will discuss them further in the section dealing with that problem.

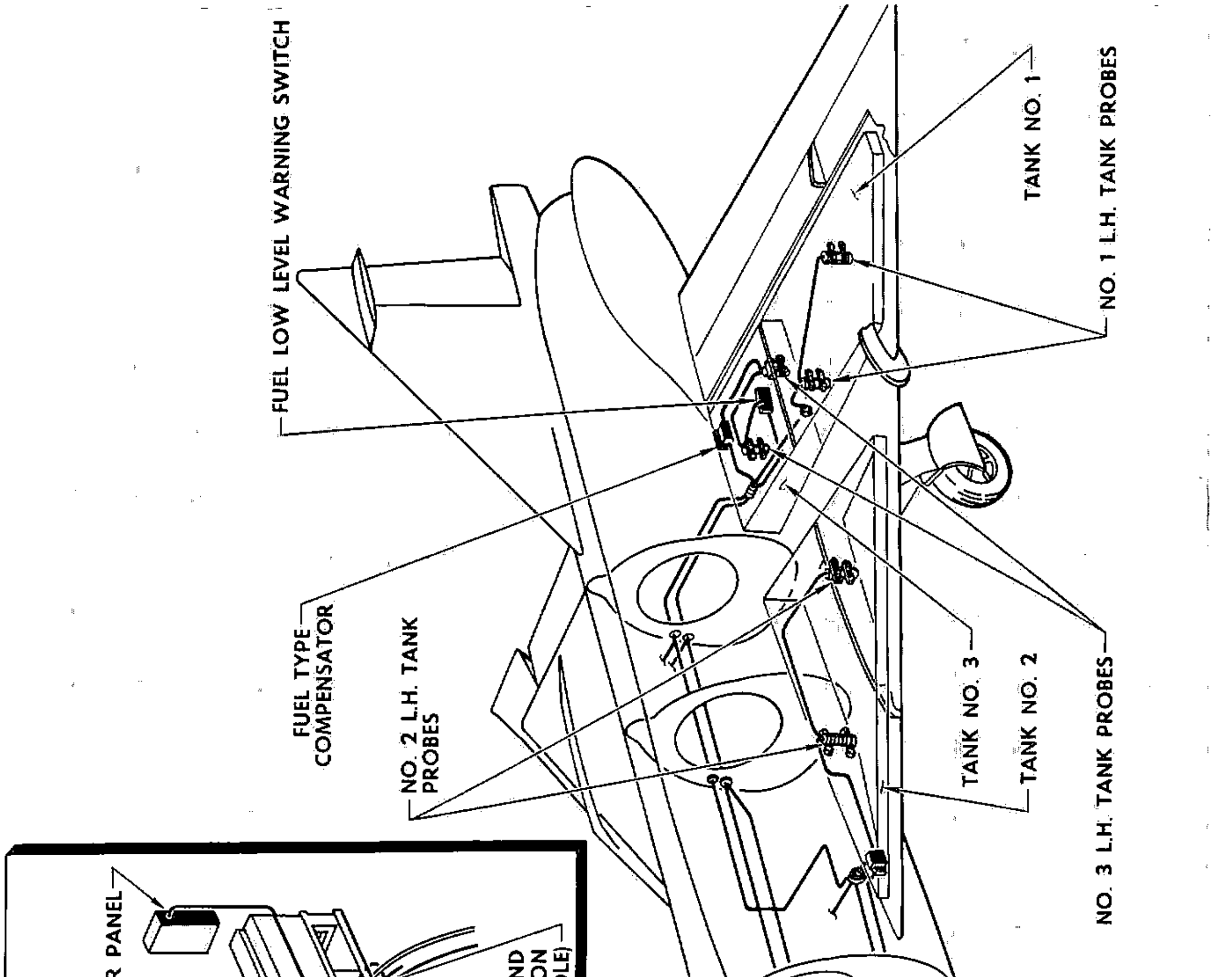

Figure 3-12. Fuel Quantity Indicating System Components

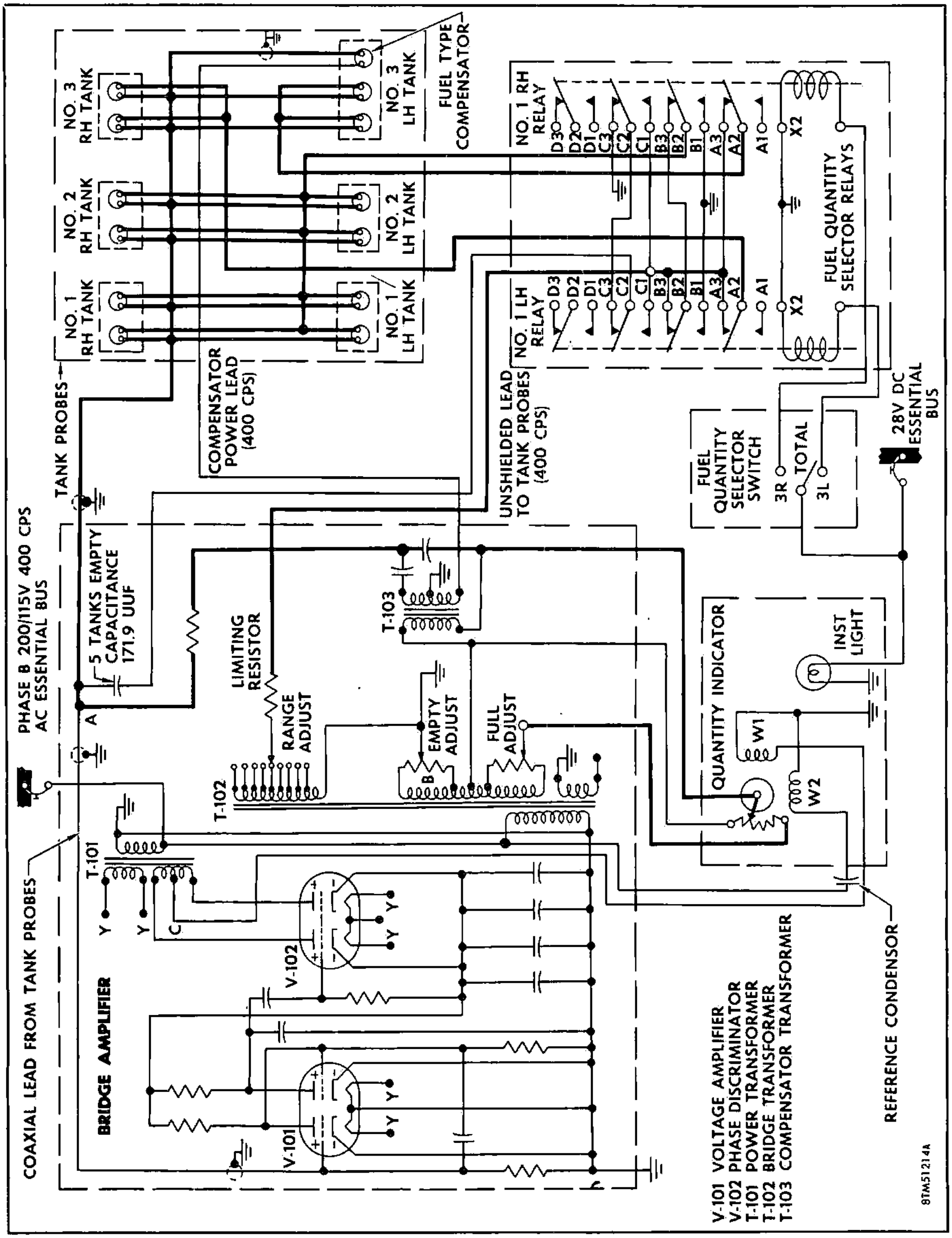

Figure 3-13. F-102A Fuel Quantity Indicating Circuit

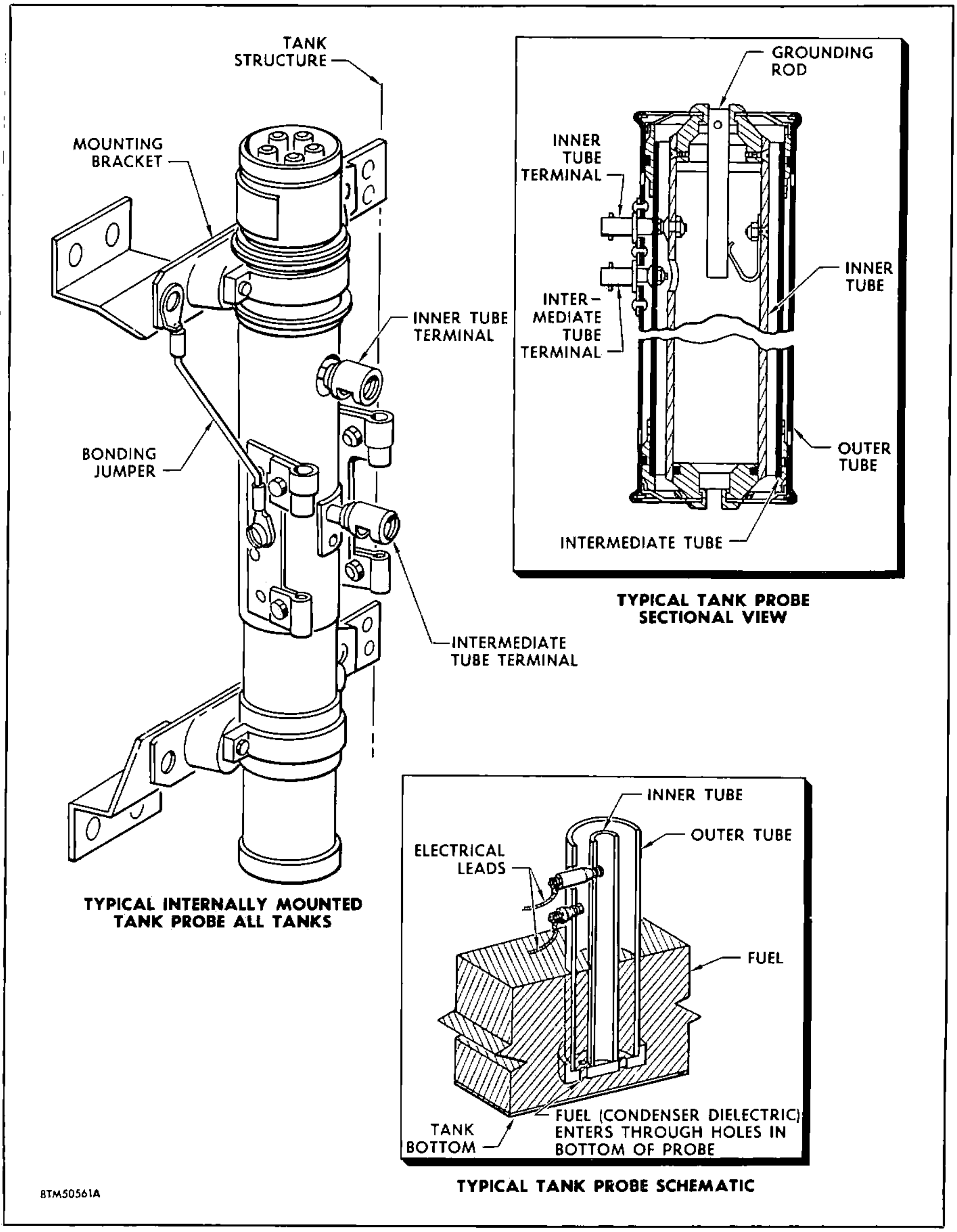

Figure 3-14. Fuel Tank Probe

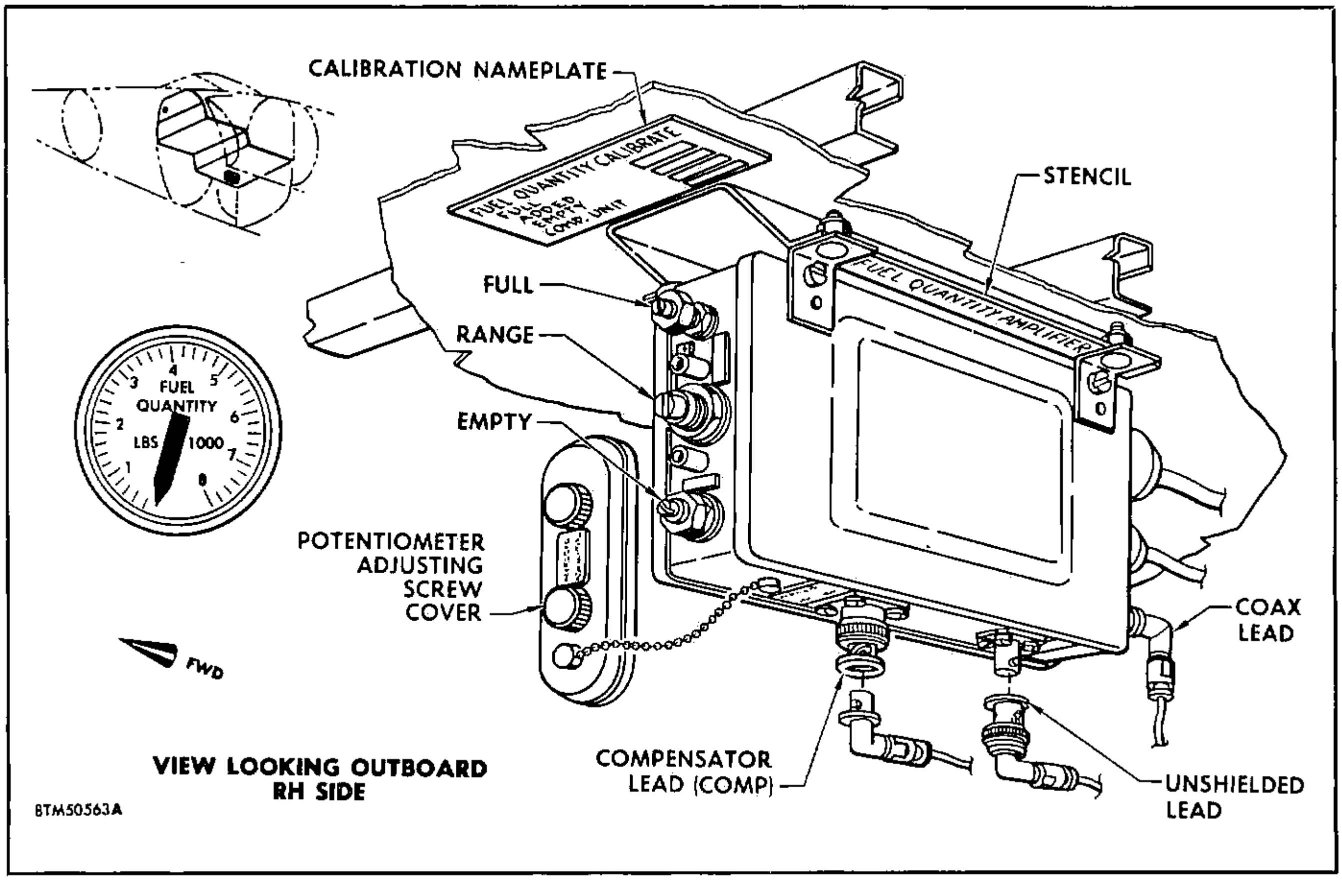

Figure 3-15. Amplifier Bridge Assembly and Fuel Quantity Indicator

The Indicator.

The fuel quantity indicator in the F-102A is located on the extreme right side of the main instrument panel. As you can see in figure 3-15, the indicator shows the available fuel supply in pounds, from 0 to 8000. The cutaway view in figure 3-16 gives you an idea of what makes the indicator "tick."

Internally, this indicator consists of a motor (10) driving a pointer (8) and a potentiometer wiper (1A) through a gear reduction train (11). The ratio of the gear train is about 3600 to 1; that is, the motor turns about 3600 revolutions to move the pointer and potentiometer wiper one revolution. Actually, the instrument is designed so that the pointer travel is limited to about 330° (slightly less than one complete revolution).

Two stator coils (5) and (7) are included in the two-phase a-c motor which drives the fuel quantity indicator. The rotor (6) will turn either clockwise or counterclockwise, depending on the signal received from the amplifier bridge assembly. Thus, the indicator will show either an increasing or decreasing fuel supply.

The potentiometer in the indicator is called a rebalancing potentiometer because it continually rebalances the bridge circuit. The bridge circuit becomes unbalanced when the capacitance of the tank probes differs from reference capacitance. You will recall that this condition is what starts the indicator motor turning in the first place, and it will not stop unless the bridge circuit is rebalanced. Rebalancing is accomplished by a signal picked off the potentiometer and transmitted back to the bridge circuit. In actual operation then, the indicator motor is continually starting and stopping as the fuel supply is used.

The fuel quantity indicator is a sealed unit with no external controls or adjustments. The pilot can, however, select individual tank readings from either the #3R (number 3 right tank) or #3L (number 3 left tank) positions of the FUEL QUANTITY switch. There is also a provision for reading total quantity. The pilot would normally select these individual or total tank quantity readings in the event of a WARNING light indication from the number 3 tanks, or if he thought the fuel quantity gage appeared stuck after a reasonably long period of flight.

Fuel Quantity Selector Switch.

As we mentioned earlier, the quantity indicator can show the contents of all tanks, or of either of the No. 3 tanks. The pilot can obtain any one of these

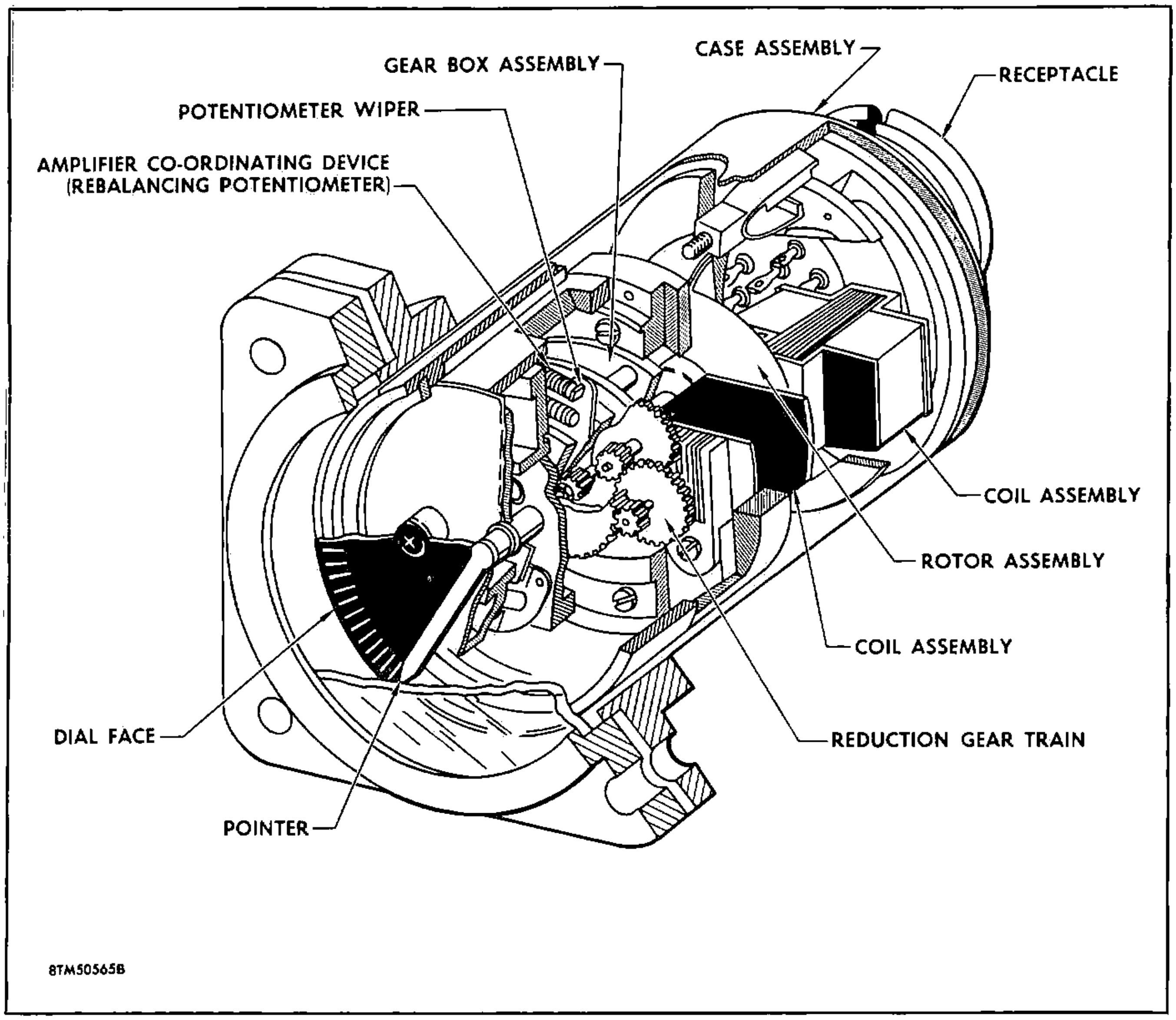

Figure 3-16. Fuel Quantity Indicator Cutaway View

three readings by means of the selector switch located on the fuel control panel on the left console. This switch has three positions—#3L, #3R, and TOTAL—as shown in figure 3-17.

The selector switch controls two fuel quantity selector relays situated in the nose wheel well. When the pilot moves the selector switch to one of the number 3 positions, the relays shut off the circuits of those tank probes which are not needed to obtain the correct readings. The indicator will then show the contents of that tank only. When the selector switch is in the TOTAL position, the relays allow current to flow through all of the circuits and the indicator shows the total contents of all tanks.

Fuel Quantity Indicator Test Switch.

Note that a test button is also located on the fuel control panel on the left console. If the pilot thinks that the pointer on the indicator has stuck in one position, he can check the pointer movement by momentarily depressing the test button. This will stop the flow of current between the indicator and the bridge amplifier, and the dial pointer will drop off the scale to a position toward empty or below. When the pilot releases the test button, the pointer should return to its previous position. If it does, the pointer is very likely moving correctly in response to electrical signals from the bridge amplifier.

CALIBRATION OF THE FUEL QUANTITY INDICATING SYSTEM.

Like all other mechanical and electrical devices, the fuel quantity indicating system is subject to malfunctioning. You will find that the most probable cause of any such trouble lies within the amplifier bridge. In most cases you will be able to recognize a defective

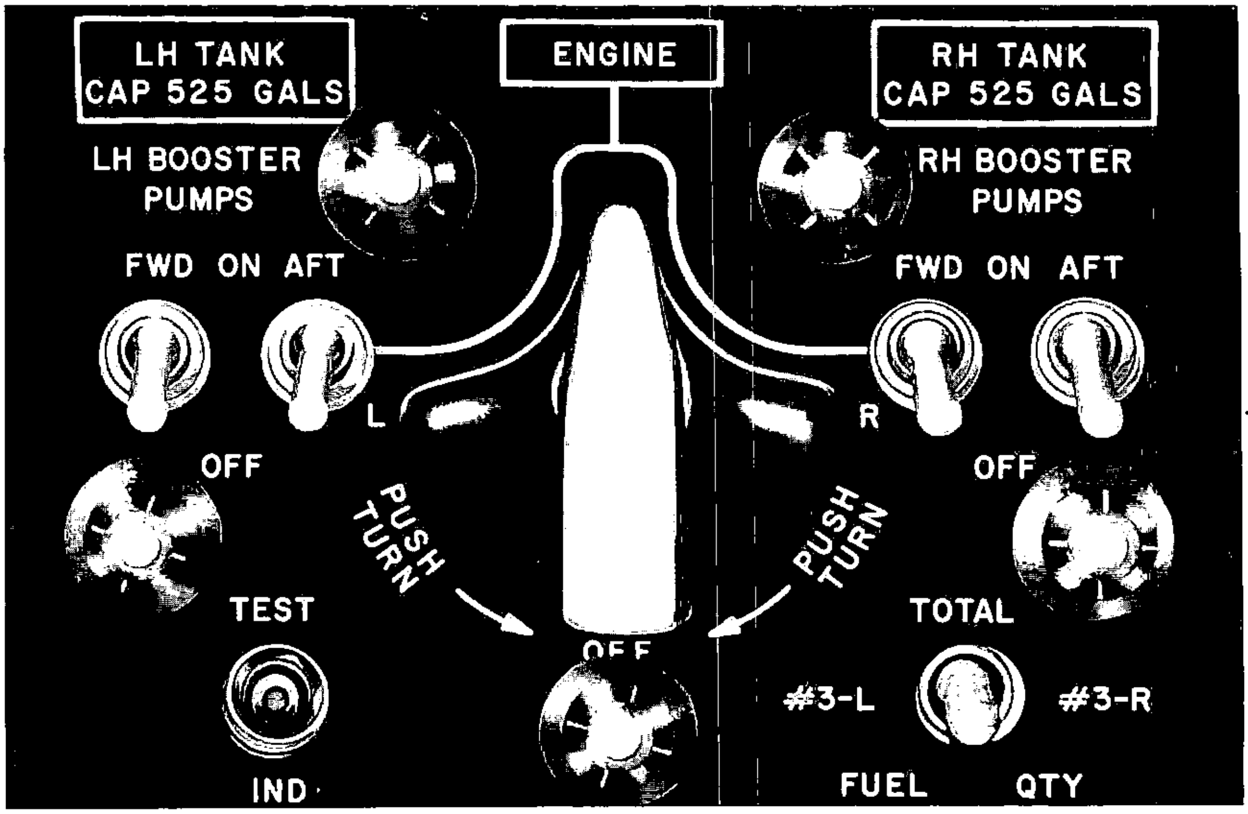

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Figure 3-17. Fuel Control Panel

amplifier by observing the readings of the quantity indicator. If the indicator reads excessively high on both right and left positions of the selector switch, or if the quantity indicator reads the same at all three switch positions, the amplifier is the most likely fault. In some instances a defective amplifier will prevent the quantity indicator from giving any reading, regardless of the quantity selector switch position.

Replacement of the amplifier necessitates calibrating the system. This means that you have to adjust the new amplifier to compensate for any slight differences in the airplane fuel systems, just as you might have to adjust a television set when you replace some of the components. Referring again to figure 3-15, note that there is a removable cover on one end of the amplifier. Under this cover are three screws, labeled FULL, EMPTY, and RANGE. The RANGE screw positions a switch; the EMPTY and FULL screws adjust potentiometers.

To calibrate the fuel system accurately, you will have to disconnect the unshielded coaxial and compensator electrical leads and then plug in a variable capacitance testing unit. This testing unit enables you to simulate the *full* and *empty* tank conditions without actually emptying and filling the fuel tanks.

With the test unit controls turned to the EMPTY position, the quantity indicator in the cockpit should read zero. If it does not, turn the adjusting screw until the quantity indicator gives the proper reading. The same procedure should be followed for calibrating the system for the *full* position. Set the RANGE adjustment screw in the third detent clockwise from the full counterclockwise position. In other words, back off the RANGE screw until it stops, then turn the screw clockwise to the third "click" or detent. This is the normal position for the RANGE screw in the F-102A installation.

The calibration procedures given here cover the basic steps required. Keep in mind, however, that this is only a very general description; for the detailed instructions and procedures you should refer to the Maintenance Technical Order for the F-102A fuel system, T.O. 1F-102A-2-5 and the Fuel System Training Supplement.

THE PRESSURE RATIO INDICATING SYSTEM.

The rpm that a reciprocating engine is turning up is a pretty good indication of the power it is putting out. Jet engines are considerably different in this respect —especially the twin-spool (two compressor) turbojet used in the F-102A. One compressor can operate near stall condition with no appreciable rpm change and without the pilot being aware of such a condition. To know exactly how the engine is performing, the pilot needs an accurate measurement of the thrust it is producing. The pressure ratio indicator, sometimes called a *thrustmeter,* provides this information. It was developed by modifying a machmeter, so you will recognize a resemblance between this mechanism and the airspeed and mach number indicator you learned about in Chapter II.

Two different pressure ratio indicating systems are used in the F-102A. Early models use an indicator which measures pressures *piped* to the instrument case. The other type consists of a remote indicator and a pressure transmitting device. Both types compare total tailpipe exhaust pressure and total engine inlet pressure to measure performance. Inlet pressure is taken from the pitot-static system; exhaust pressure is *bled* from a probe just aft of the turbine section of the engine. We will discuss the direct reading indicator first because it is the most simple.

DIRECT READING PRESSURE RATIO INDICATOR.

Figure 3-18 shows a face view of the direct reading pressure ratio indicator. The inner, fixed, dial is calibrated to show the ratio of the engine discharge pressure to its inlet pressure from 1.2 to 3.4. The outer, transparent, dial is lowered so you can see the markings. This transparent dial is rotated manually by the knob in the center of the dial to line up the temperature scale with the temperature index on the fixed dial. With the proper temperature setting, as determined by the pilot from his outside air temperature indicator, the most efficient power settings should keep the pointer in the area covered by the range mark. The pointer is rotated clockwise by the movement of two diaphragms within the airtight instrument case.

Now take a look at the operational schematic, figure 3-19. The diaphragm within the frame is the tailpipe pressure diaphragm. This diaphragm expands and raises the entire frame assembly when exhaust pressure enters it. The amount of expansion depends on the difference between the pressure inside the diaphragm and the pressure on its outer surface. Angular position of the frame is controlled by movement of the inlet pressure diaphragm shown in the illustration. This diaphragm is a sealed, evacuated unit (aneroid-type). Thus, it expands and contracts with changes of pressure on its outer surface only; since it is a sealed chamber, the inner pressure is constant. The pressure on its outer surface is the inlet air pressure.

Note in figure 3-19 that the rocking shaft is geared to the pointer. This shaft is turned by the vertical movement of the frame assembly. Just how much pointer movement results from a given displacement of the frame depends on the angular position of the frame. As altitude increases, the sealed inlet pressure diaphragm expands due to the reduced atmospheric pressure. This expansion rotates the frame so that the frame lever moves closer to the rocking shaft. Note that the closer the frame comes to the rocking shaft, the more the rocking shaft will be turned by a given amount of vertical displacement of the frame. It is just as though the actuating arm of the rocking shaft were being varied in length.

There is one disadvantage in using a direct-reading pressure ratio indicator—exhaust gases are piped into the cockpit of the airplane. A dangerous condition could result if anything happened to cause a leak in the system within the cockpit area. To eliminate that possibility, a remote-indicating system is installed in the later models of the F-102A.

THE REMOTE INDICATING PRESSURE RATIO SYSTEM.

If you understand the operation of the direct reading pressure ratio indicator, you won't have any trouble understanding this remote indicating system. It furnishes the same information to the pilot and in a similar manner, except that the measuring mechanism is mounted in the area between the two engine air intake ducts. It connects electrically to the indicator which occupies the same position on the instrument panel as the self-contained unit.

The Transmitter.

Figure 3-20 shows a side view and an end view of the transmitter assembly as it appears in the airplane. In the end view, note that there are two tube fittings, one marked LOW and the other marked HIGH. The low presssure connection receives the inlet air pressure while the high pressure connection receives the exhaust pressure. In the center is the electrical receptacle. It provides power to a synchro transmitter within the transmitter assembly and carries the output signal from the synchro to the indicator. These pressure and electrical connections are all on the base of the unit rather than on the transmitter itself, because the base is attached firmly to aircraft structure and the transmitter is shock-mounted to the base. The curved tubes from the base to the transmitter carry the pressures to the transmitter case without interfering with the shock absorbing mountings.

Internally, the transmitter contains the same sensing and measuring mechanism that is used in the direct reading indicator. Instead of rotating the pointer on the end of its shaft, the rocking arm in the transmitter is geared to a synchro transmitter. Thus, it puts out an electrical signal proportional to the deflection of the rocking shaft.

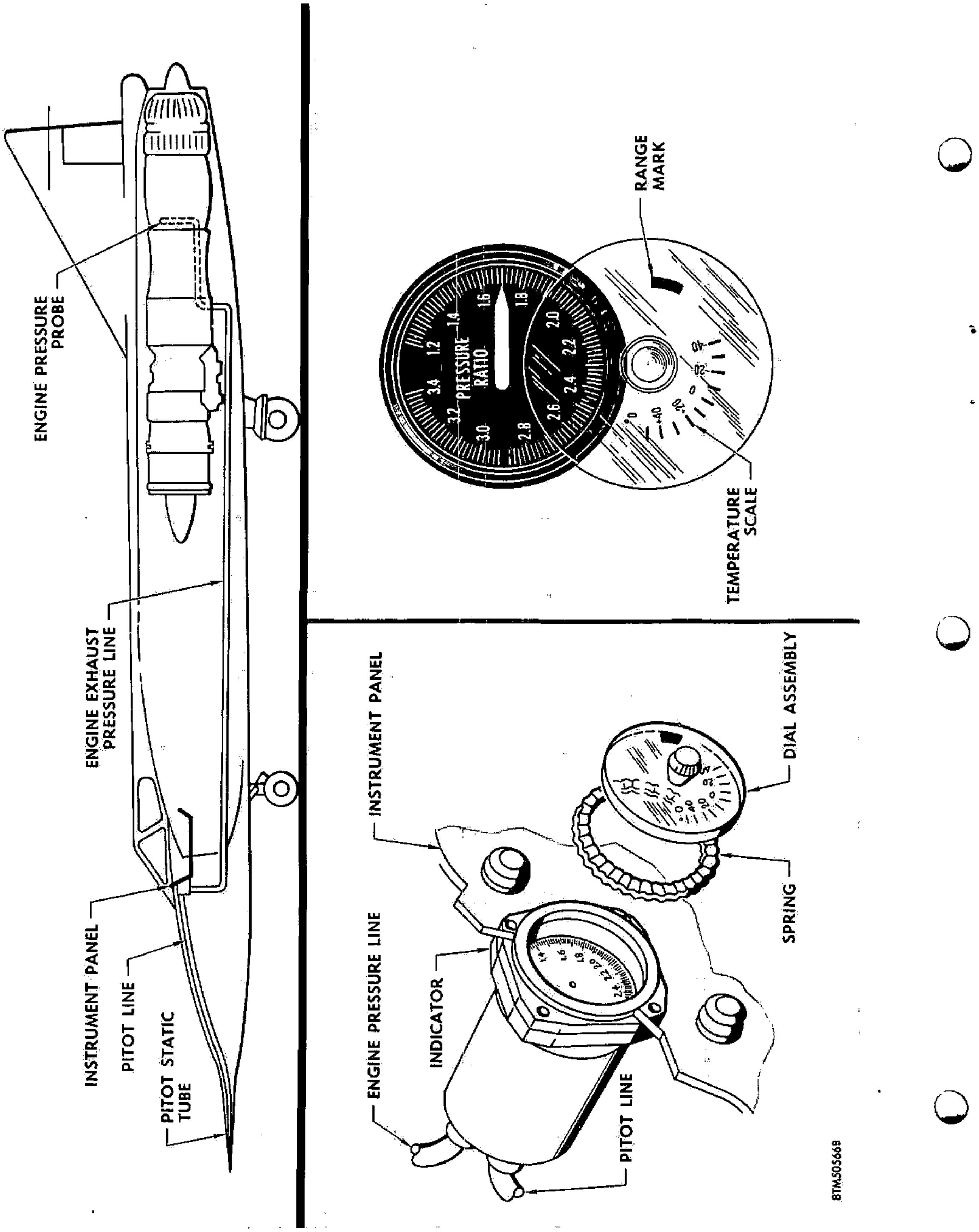

Figure 3-18. Direct Reading Pressure Ratio Indicator, Outer Dial Lowered

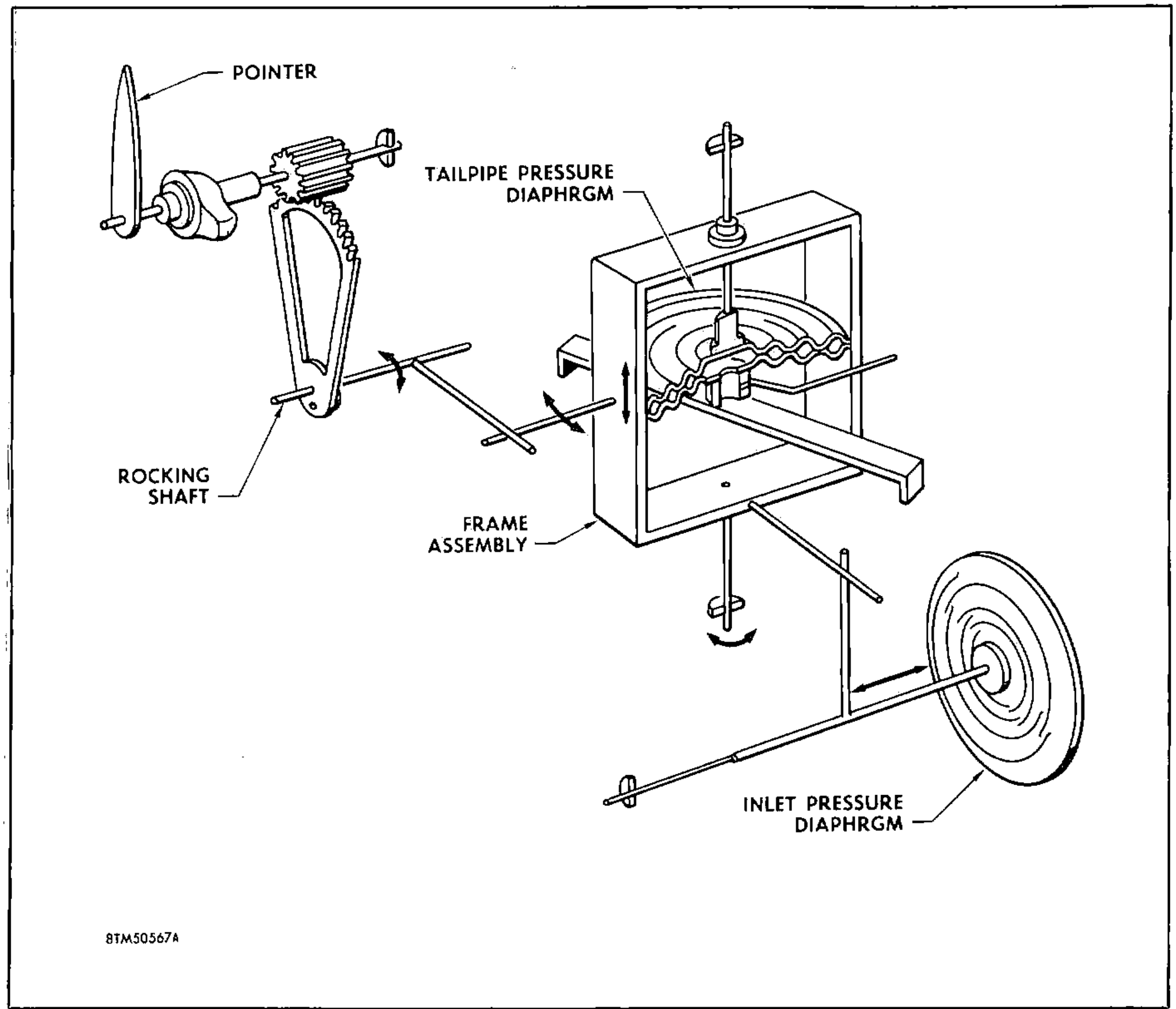

Figure 3-19. Pressure Ratio Indicator, Operational Schematic

The Remote Indicator.

There are several differences between the remote indicator and the direct reading instrument. In figure 3-21 note that the dial is calibrated for the same limits as the other instrument, but that it is positioned differently. There are two index markers and two windows in the face of the dial. The readings in the small windows and the position of the index markers can be changed by the push-pull knob on the front of the indicator. When the knob is pushed in and turned, it will turn the numbers in the lower window and reposition the lower index marker. When the knob is pulled out and turned, it will turn the numbers in the upper window and reposition the upper index marker. The upper window and marker indicate the cruise setting desired by the pilot; the lower window and marker indicate the takeoff reading which should be indicated when the takeoff is made. These dials and markers are connected mechanically to the knob and are set manually by it. In no way are they affected by the indicating mechanism that turns the pointer.

Inside the indicator case is a synchro receiver, a magnetic amplifier, and a servo motor. The position of the synchro in the transmitter unit provides a signal to the indicator through an electrical connection in the rear of the indicator case. This signal positions the synchro receiver, which in turn sends a signal via the amplifier to the servo motor. The servo motor then turns the pointer shaft to deflect the pointer and indicate the pressure ratio.

To set the remote indicator properly, the pilot must determine two factors. First, he contacts the control

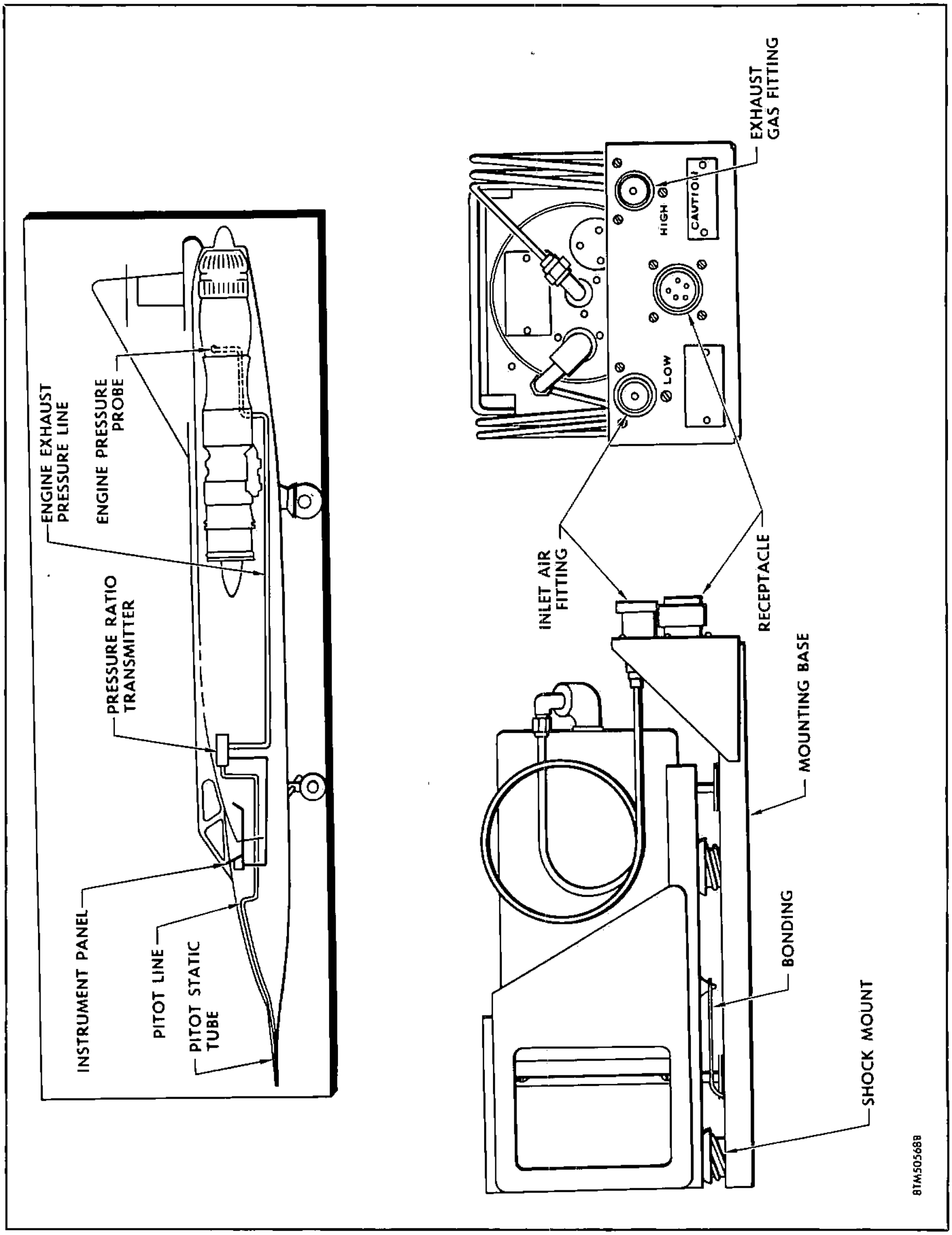

Figure 3-20. Pressure Ratio Transmitter

tower to get the existing temperature at the field and the estimated temperature at the altitude at which he intends to cruise. He then consults his performance charts to determine what the settings on the indicators should be for the two temperature conditions. When he turns the setting knob, the index marker and the numbers in the corresponding window will show the same reading.

You can see an example of these readings in figure 3-21. Note that the upper index marker is between 2.5 and 2.6, while the indication in its corresponding window is the same or 2.55. Note also that the lower window shows a reading of 1.80 and that its index marker is at the corresponding position on the dial.

The windows serve only to give the pilot an accurate and simple method of telling what his index setting is. Note that the lower index (for takeoff) has a raised portion on one end. This is the most desirable position for the pointer during takeoff. However, it is safe to take off even if the needle goes beyond that point, as long as it remains within the zone covered by the marker (between 2.5 and 2.6). When the aircraft reaches the selected cruise altitude, the pointer should match up with the upper, cruise index, marker. The pilot then knows that his engine is performing efficiently.

Since the pressure ratio indicator operates on the diaphragm and aneroid principle, it is subject to error if the tubing to the instrument (either the direct indicator or the remote transmitter) leaks or becomes restricted. In particular, you should check for leaks in the pitot and engine-transmitter line which will cause the indicator to read either too high or too low. You may also find that the instrument fails to register at all, in which case there is a malfunction in the circuits. You should check the circuit between the indicator and the transmitter, or in either one of these units. You will find more detailed maintenance information and procedures in T.O. 1F-102A-2-9, Instruments. In checking these systems, you should always follow the instructions in your maintenance technical orders very carefully.

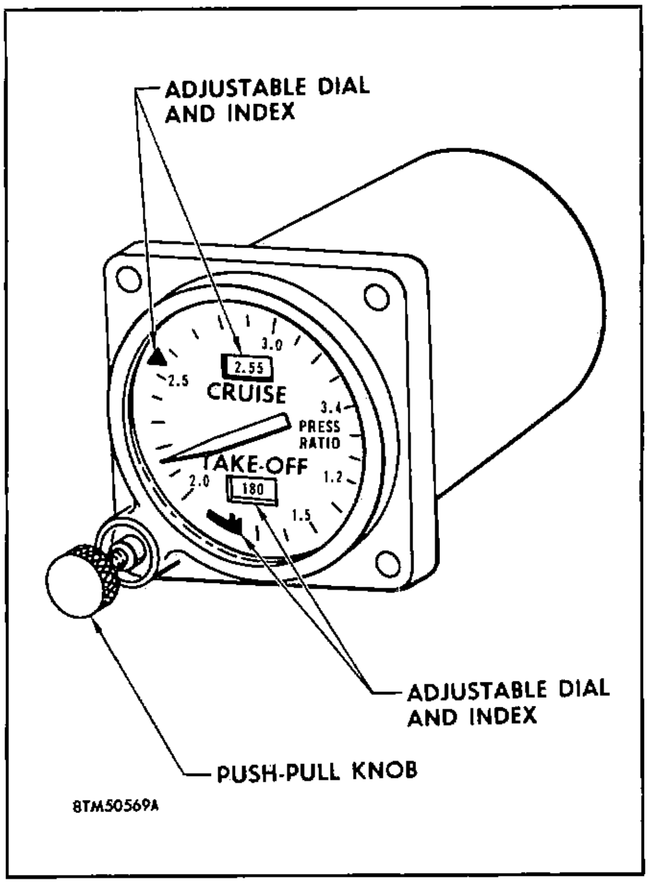

Figure 3-21. Remote Pressure Ratio Indicator

THE TACHOMETER.

The purpose of the tachometer is to determine the revolutions per minute at which the engine is operating, and to provide an indication of that engine speed to the pilot. There are several kinds of tachometers used in aircraft. In a single-engine aircraft which has the engine mounted directly ahead of the cockpit, a mechanical tachometer is frequently used. This type is driven directly from the engine by a flexible shaft, and operates much like an automobile speedometer. However, most modern aircraft use electrical tachometers, thus eliminating the direct drive from the engine. These tachometers consist of a generator attached to the engine, and the generator is connected electrically to an indicator in the cockpit. The generator and indicator are shown in figure 3-22. Let's consider the generator first, since that is the transmitting part of the system.

THE TACHOMETER GENERATOR.

On the F-102A, the tachometer generator is attached to the front of the accessory drive housing, on the right side of the engine. As you can see in figure 3-23, a pad-type mounting is used for its installation; that is, the generator is bolted to a flat, machined surface on the accessory housing. A square-end drive shaft projects from the generator into the accessory drive section where it is turned by reduction gearing at 4200 rpm when the engine is operating at maximum rated speed. This drive shaft runs through the center of the rotor (armature) shaft. A pin connects the two shafts at the end opposite the mounting pad. This arrangement permits the drive shaft to absorb any vibrations that might result from slight wear and misalignment at the connecting points. Thus, the rotor shaft—and consequently the rotor—spin at the same rate as the drive shaft. A bearing at each end of the rotor shaft supports it within the generator stator. Figure 3-23 shows a cutaway view of the entire generator.

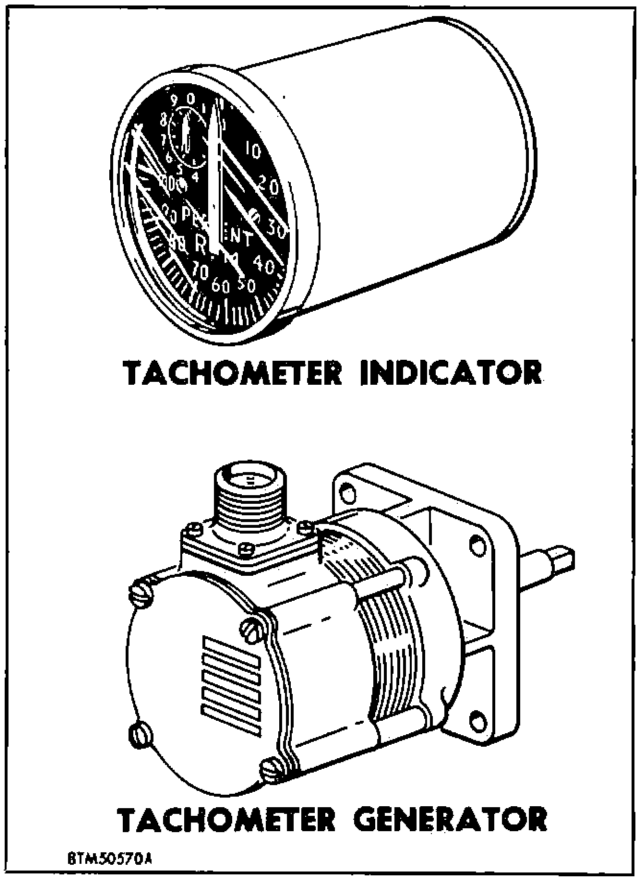

Figure 3-22. Tachmometer Generator and Indicator

The rotor of the tachometer generator is a two-pole permanent magnet made of very hard steel. Surrounding the rotor is the stator, made up of a laminated soft iron core around a three-phase winding. When the rotor turns, current is induced in the windings of the stator. Since the rotor has two poles (one North and one South), one complete revolution will result in one cycle of alternating current in each of the stator windings. Three wires come from the stator and carry this induced current to the generator receptacle, and from there to the indicator in the cockpit.

THE TACHOMETER INDICATOR.

There is considerable similarity between the generator and the basic mechanism of the indicator. That's understandable, since the indicator incorporates a synchronous motor having many of the same characteristics as a generator. The exploded view of the indicator in figure 3-24 shows the major components and how they are related to each other.

The rpm of the engine in the F-102A is indicated by the tachometer indicator in per cent of maximum rated speed. The dial of the indicator is calibrated from 0% to 100% of maximum rpm in 2% graduations, beginning at the top of the dial and extending around for 270°. The large pointer rotates around this dial. The small pointer rotates around its dial once for every 10% change in rpm. Using both scales, the speed of the engine can actually be read up to 110% of rated speed.

Power to operate the tachometer indicator comes directly from the tachometer generator. The stator is wound in the same manner as that in the generator, and the rotor is a permanent magnet, just as the rotor in the generator. As the induced current from the generator is fed to the stator of the indicator, a a rotating magnetic field is set up in the stator. You will recall from our discussion of magnetism in Chapter I how a permanent magnet tends to line up with the magnetic field around an electric conductor. Thus, the indicator rotor is literally dragged around at a speed proportional to the speed of the generator. Since the generator is geared to the engine, the speed of the indicator rotor is an accurate indication of the speed at which the engine is turning over.

The rotor of the tachometer indicator is different from the rotor of the generator in one respect—it is a four-pole magnet. In other words, it has two North Poles and two South Poles. Thus the speed of this rotor is one-half that of the rotor speed in the generator. For each complete cycle of current produced by the generator, the rotor of the indicator turns one-half turn.

Now, note that the shaft which turns with the indicator rotor turns another permanent magnet called the drag magnet. Around this drag magnet is a

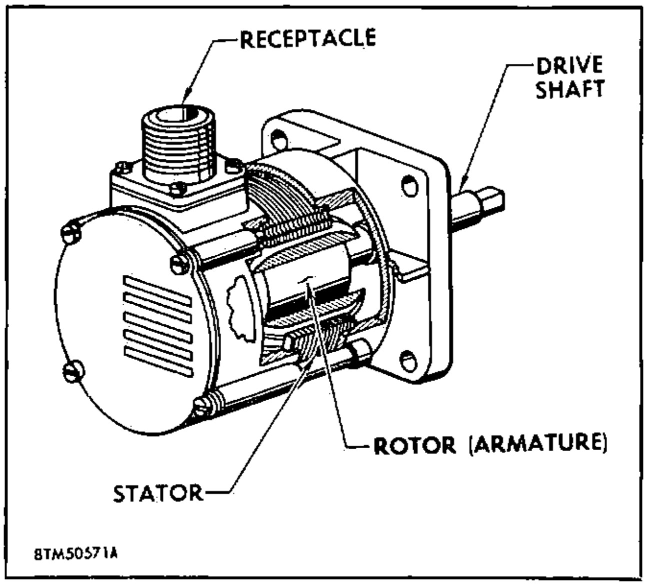

Figure 3-23. Cutaway of Tachmometer Generator

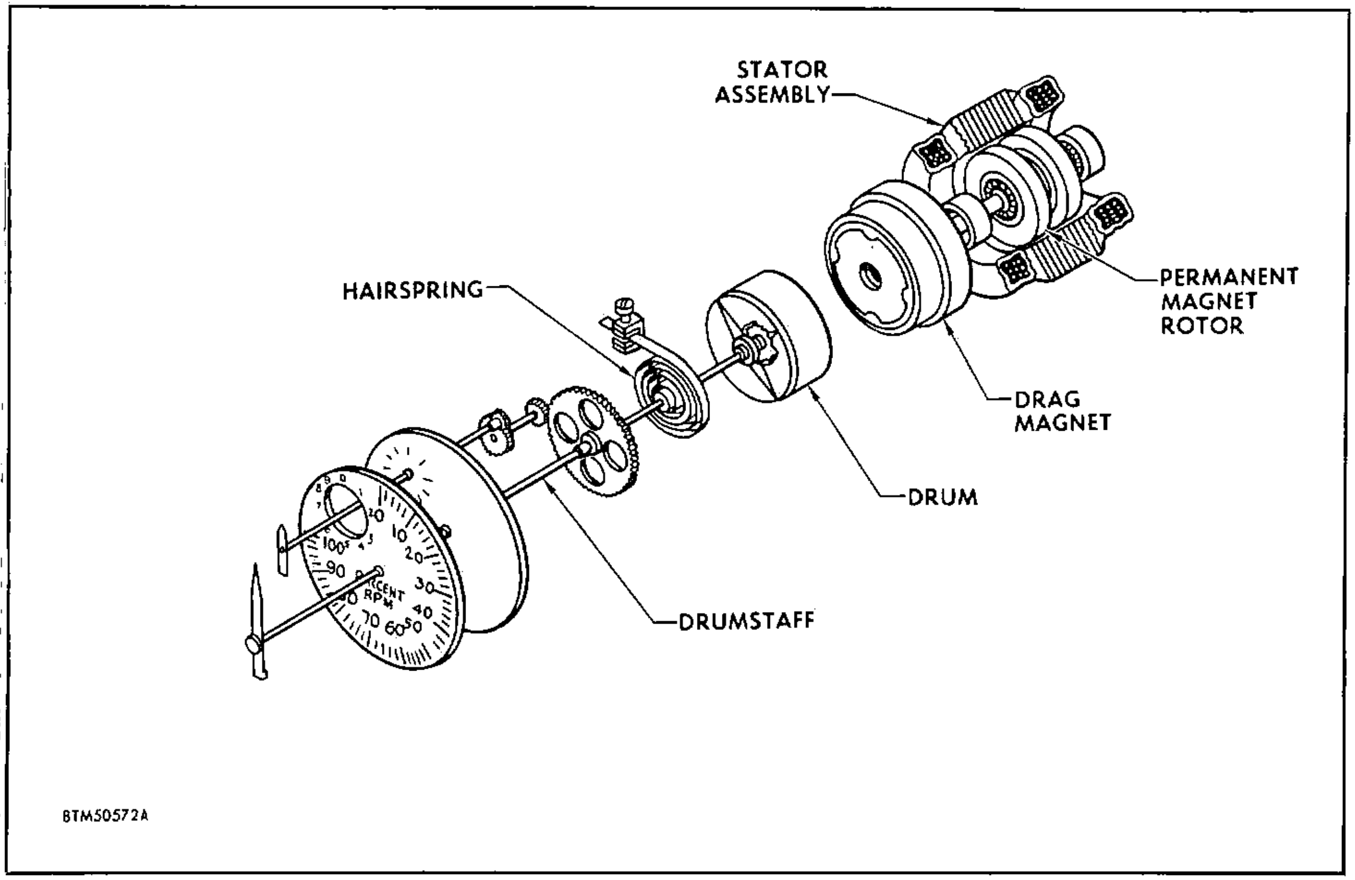

Figure 3-24. Exploded View of Tachmometer Indicator

drum which, because of its proximity to the magnet, is dragged around with the magnet. However, the rotation of the drum is limited by a spring, so the drum can only turn so far. Just how far it turns depends upon the speed of the drag magnet which turns it. As you can see, the drum and the pointer drive shaft are connected, so the pointer movement is directly proportional to the movement of the drum. Thus, the speed of the engine is transmitted to the generator by gears, then to the indicator by electricity, and to the indicator pointers by the magnetic force of the drag magnet.

You should not be alarmed if the tachometer indicator in the F-102A doesn't show 100% rpm when you make a full power engine runup. Remember that these engines turn up at very high speeds (approximately 10,000 rpm) and there is usually some variation in the speeds at which different engines produce their maximum thrust.

It is easier to remove the tachometer indicator than the generator. Consequently, if you doubt the accuracy of the indicator, you should first bench test the indicator on a tachometer tester or replace it with an indicator known to be serviceable. If the indicator operates satisfactorily, the only other possibilities of malfunctioning are the generator or the leads between the generator and indicator.

You can check the output of the generator with a field test unit that attaches to the generator output receptacle. Continuity checks will reveal the condition of the circuits. You can find the method for performing continuity checks in the Electrical Systems Training Supplement, Chapter III.

INSTRUMENT REPLACEMENT.

Sooner or later you will have to replace an engine instrument, so it might be well to mention here just how they are mounted in the airplane. Although the instruments discussed in Chapter II are bezel-mounted, most of the other instruments in the F-102A are clamp-mounted. Figure 3-25 shows a typical clamp mounting. As you can see, all that is required to remove or install one of these instruments is common sense and common tools, so we won't go into further detail on the matter.

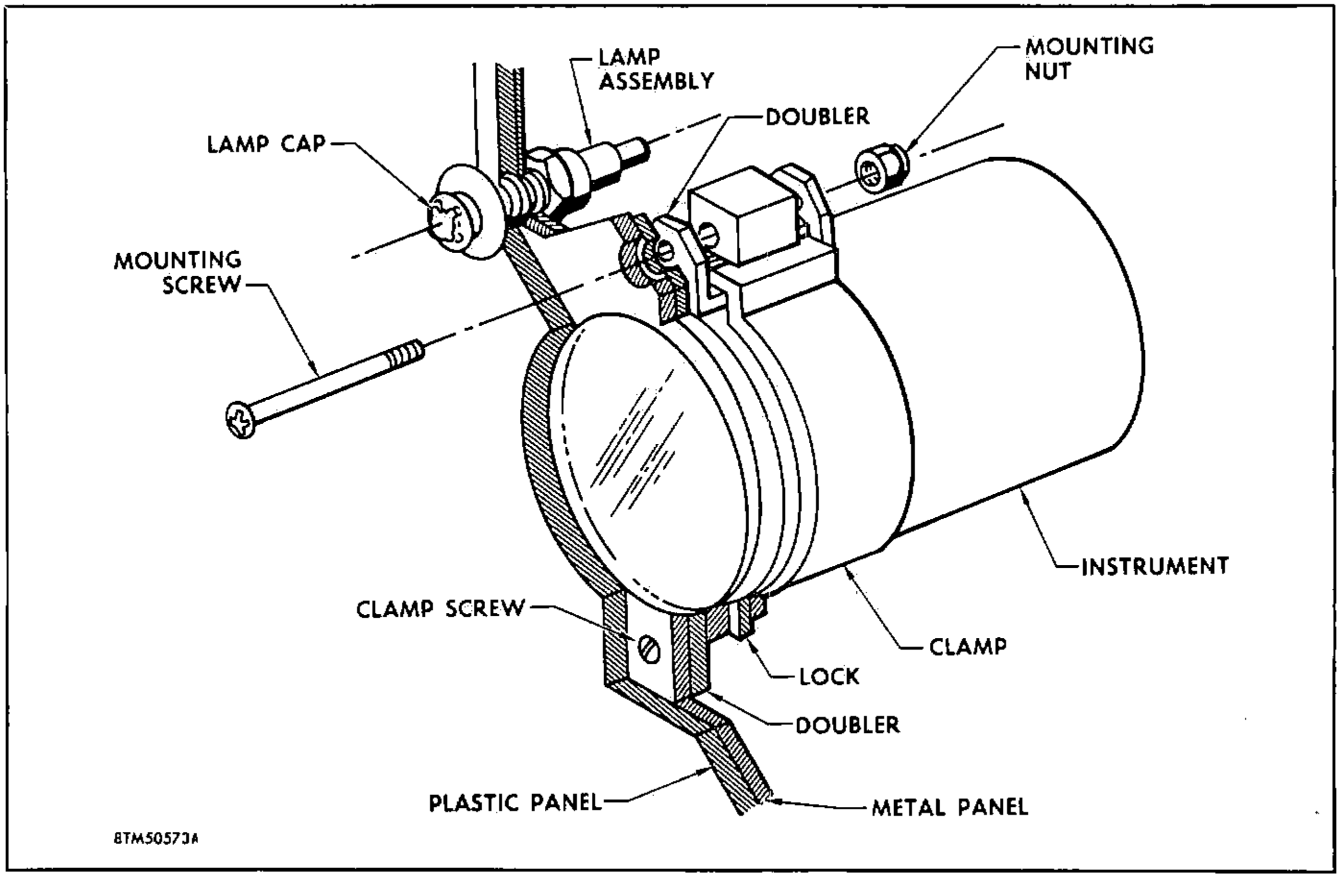

Figure 3-25. Clamp Mounting of Instruments

Chapter IV

POWER SUPPLY INSTRUMENTS

Contents *Page*

In the power supply group of instruments we include the instrumentation which indicates the amount of power available to operate the various systems and components within the airplane. We do not mean, however, the actual motive power (engine thrust) which propels the airplane through the air; instead we are talking about the power in the airplane systems such as electrical, hydraulic, and pneumatic.

The sources of power used in different types of aircraft vary considerably; in fact, some of the small private planes have no powered systems at all. The F-102A, on the other hand, requires three types of power systems: electrical, hydraulic, and pneumatic. The instruments used to indicate the power in these systems are the hydraulic system pressure gage, the hydraulic accumulator pressure gages, the pneumatic pressure indicator, and the a-c voltmeter.

There is no single location in the F-102A airplanes where you will find all of the power supply instruments grouped together. Instead, these instruments are placed at various locations throughout the airplane to serve their purpose best. For example, the a-c voltmeter is on the right-hand auxiliary instrument panel. The hydraulic pressure indicator for both the primary and secondary hydraulic systems is mounted on the main instrument panel on some models of the F-102A, and on the right console of other models. Neither the hydraulic accumulator pressure gages nor the pneumatic pressure gage is located in the cockpit area. They are installed only for your use in maintenance and servicing of the systems, and are positioned near their respective filler valves.

HYDRAULIC PRESSURE INDICATING SYSTEM.

The hydraulic pressure of both the primary and secondary hydraulic systems in the F-102A is shown by the hydraulic pressure indicating system. The indications given by this system do not show the pressures in the two accumulators; that information is provided by two other instruments which we discuss later in this chapter.

The hydraulic pressure indicating system is a remote indicating system consisting of two transmitters, an indicator, a selector switch, and the connecting electrical circuits. The hydraulic pressure indicator and one transmitter are shown in figure 4-1.

The transmitter shown is the type used in both the primary and secondary hydraulic systems.

THE HYDRAULIC PRESSURE TRANSMITTERS.

Both hydraulic pressure transmitters (for the primary and secondary systems) in the F-102A airplanes are located in the hydraulic accessory compartment, just forward of the right main wheel well. Both transmitters receive electrical power from the airplane 26-volt, 400-cycle, a-c supply. These transmitters can measure a pressure range from 0 to 4000 psi.

Now let's look at figure 4-2 which shows the components within a transmitter. Note that this transmitter incorporates a Bourdon tube and a synchro. Thus, the mechanism is quite similar to some which

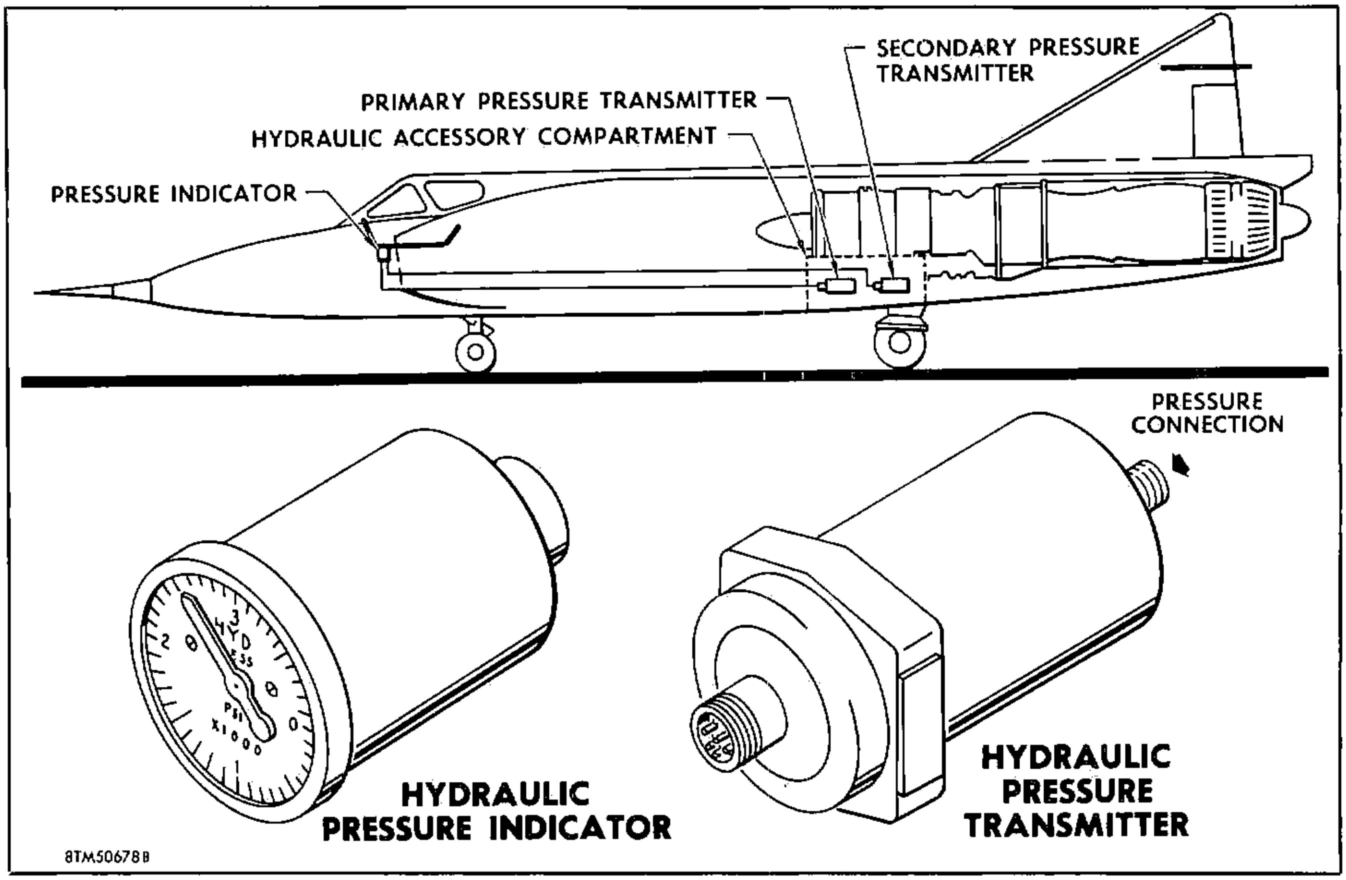

Figure 4-1. Hydraulic Pressure Indicator and Transmitter

we have already discussed. As you can see, the hydraulic pressure to be measured enters a fitting in one end of the transmitter and then flows through a capillary tube to the Bourdon tube. A connecting link on the "free" end of the Bourdon tube transmits movement of this tube to the rotating shaft. A sector gear attached to the rotating shaft meshes with the pinion gear mounted on the end of the synchro rotor shaft.

You can see then how the opening and closing of the Bourdon tube arc, caused by changes in the hydraulic pressure, position the synchro rotor. The amount of rotation of the synchro rotor is proportional to the hydraulic pressure in the Bourdon tube. When hydraulic pressure subsides the tube returns to its normally closed position, letting the rotor go back to a lower pressure position or to zero. A hairspring attached to the rotor takes up any slack in the linkage from the tube to the synchro rotor.

The snubber valve attached to the transmitter acts as a damper to reduce the effects of pressure surges and vibration on the Bourdon tube. You are probably familiar with the vibration that can occur in fluid lines, and undoubtedly you have experienced vibrating and chattering in water pipes when you turned a water faucet on. Obviously, without a snubber any vibration in the hydraulic system would cause the pointer in the indicator to oscillate if it were permitted to reach the Bourdon tube in the transmitter. In fact, hydraulic fluid could knock so seriously that you would also have to replace transmitters quite frequently. In addition to the damping action, the snubber valve includes a filter element to prevent foreign particles—if any exist in the fluid—from plugging the capillary tube.

The bellows shown in figure 4-2 is referred to as the "blowout bellows." If a leak should develop in the Bourdon tube, capillary tube, or in any of the connecting joints, the case of the transmitter would fill with hydraulic fluid under the full pressure existing in the system at that time. This pressure could rupture the case and cause fluid to be sprayed around the surrounding area. Needless to say, this could create a hazardous condition. The blowout bellows prevents this danger. If pressure builds up in the case, the blowout will "give" at a certain pressure, and permit the fluid to run out at a slow rate.

THE INDICATOR.

The indicating component of the hydraulic pressure indicating system shows the pressure of both the

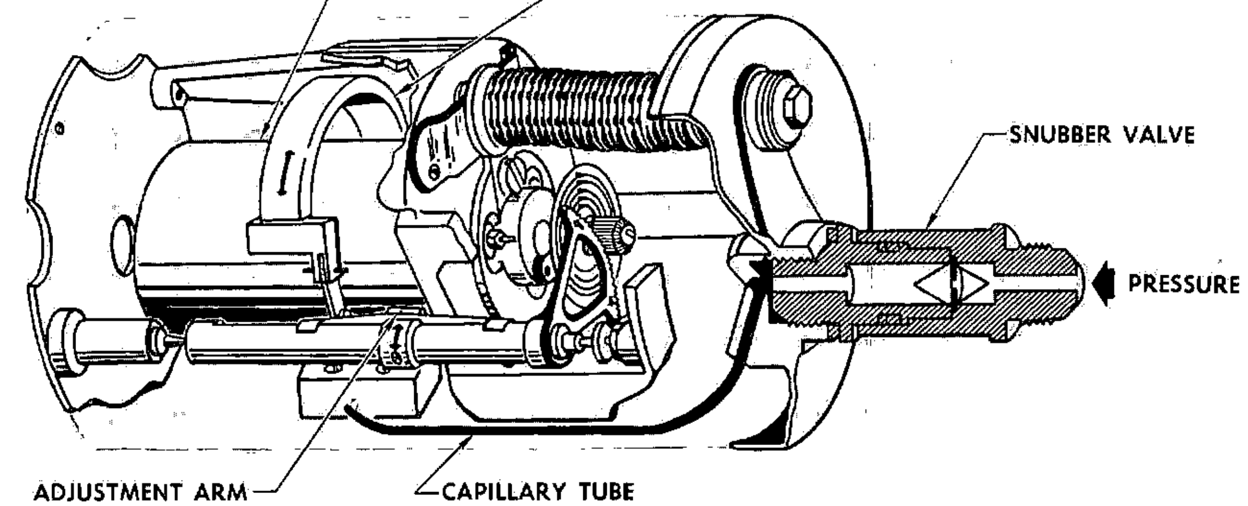

Figure 4-2. Hydraulic Pressure Transmitter—Internal Mechanism

primary and the secondary hydraulic system, but only one at a time. The system pressure which is indicated depends on the position of the hydraulic pressure selector switch adjacent to the indicator. Both hydraulic systems on the F-102A have normal operating pressures of 3000 psi, but the indicator will show pressures up to 4000 psi.

In figure 4-3 you can see the hydraulic pressure indicator with its case removed. Note that it is a very simple synchro instrument containing the receiving (or repeating) half of the synchro system; the transmitting synchros are in the pressure transmitters which we just discussed. Since the indicator contains only one synchro motor, it can indicate the pressure signals from only one of the two transmitting synchros at one time. The pointer is attached directly to the synchro rotor shaft, so it turns with the rotor.

THE SELECTOR SWITCH.

The hydraulic indicator selector switch is located adjacent to the hydraulic pressure indicator. It connects into the electrical circuit which carries the signals from the transmitters to the indicator. This selector switch is a two-position type. When this switch is in the UP or forward position, the primary hydraulic system transmitter is connected to the indicator; when the switch is in the DOWN or aft position, the secondary hydraulic system pressure is indicated. Thus, the pilot can read the pressure of

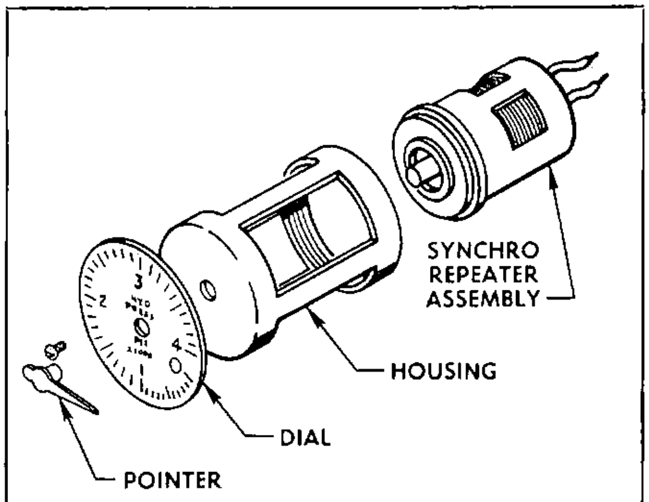

Figure 4-3. Hydraulic Pressure Indicator—Internal View

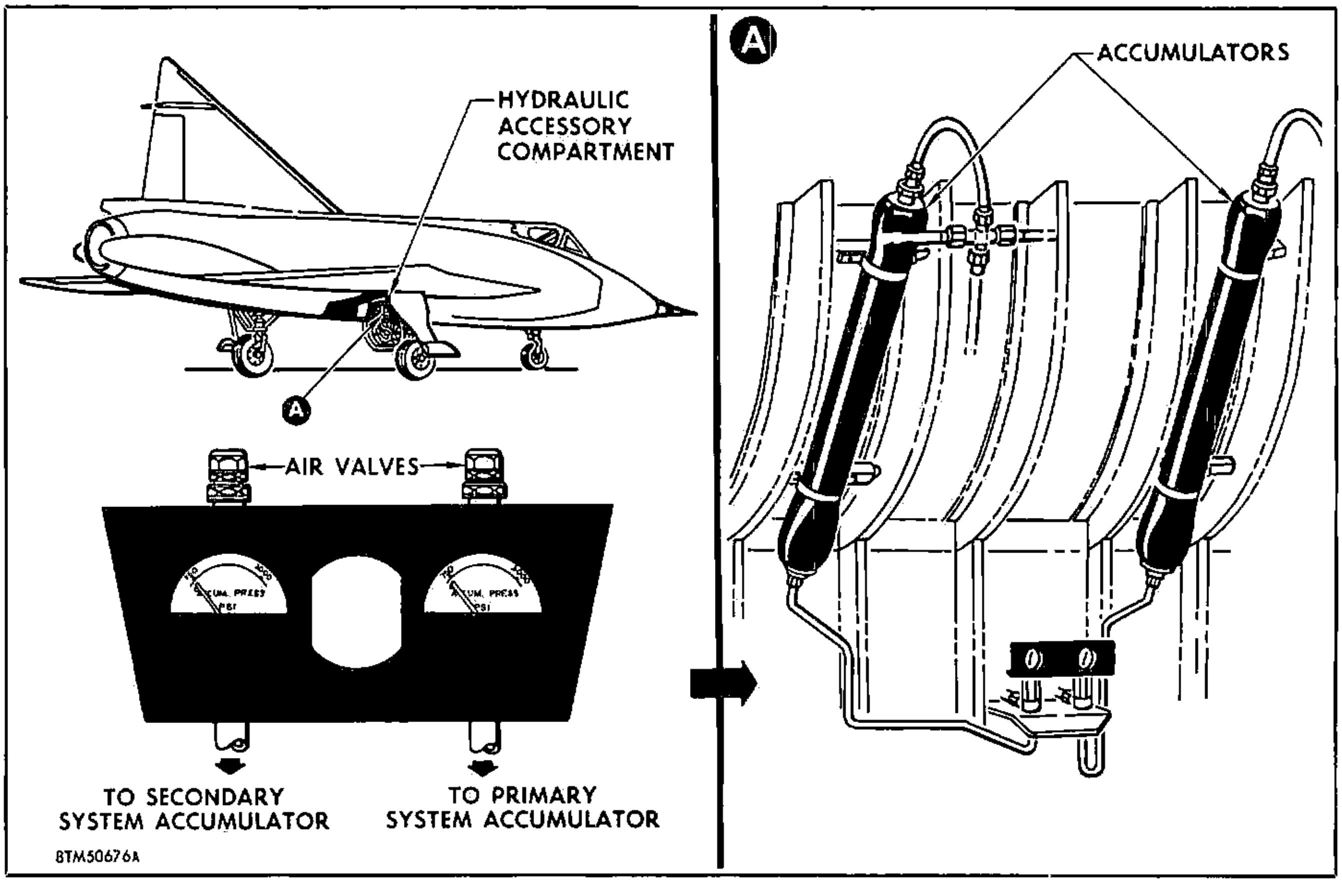

Figure 4-4. Hydraulic Accumulator Pressure Indicator

either hydraulic system on the same instrument. On some models of the F-102A the hydraulic pressure indicator and its system selector switch are located on the right-hand console, while other models have these units on the main instrument panel at the extreme right side.

SYSTEM MAINTENANCE.

The hydraulic pressure indicating system is usually very easy to troubleshoot. Since the indicator can receive signals from two separate transmitters, you can usually tell whether an erroneous reading is the fault of the indicator or one of the transmitters. If the indicator synchro is at fault, it will be in error regardless of the position of the selector switch. On the other hand, if a transmitter is at fault the indicator will read incorrectly only when switched to the malfunctioning transmitter. Of course, it is possible that the problem is in the electrical leads between the components of the indicating system. Your reading would not necessarily be erroneous, but you may receive no reading at all. A continuity check would determine if the wiring circuit is the cause of the trouble.

HYDRAULIC ACCUMULATOR PRESSURE INDICATORS.

Two hydraulic accumulator pressure indicators are installed in the F-102A—one for the primary and one for the secondary hydraulic system. Both instruments are identical, as shown in figure 4-4. You will find these indicators in the hydraulic accessory compartment, near the accumulators and reservoirs. They are mounted on a bracket and face outboard so that they are readily visible when the compartment door is open.

The hydraulic accumulator pressure indicators are simple Bourdon-tube gages which are installed for convenience in servicing and maintaining the hydraulic system. Each gage is calibrated from 750 psi to 3000 psi, but the pointer can go above or below these limits before striking stops.

INDICATOR OPERATION.

In figure 4-5, note that the tubing line to the indicator comes directly from the nitrogen end of the accumulator. This accumulator contains a floating piston which separates the hydraulic fluid from the pre-charge of nitrogen. As you can see, the indicator actually measures the pressure in the nitrogen

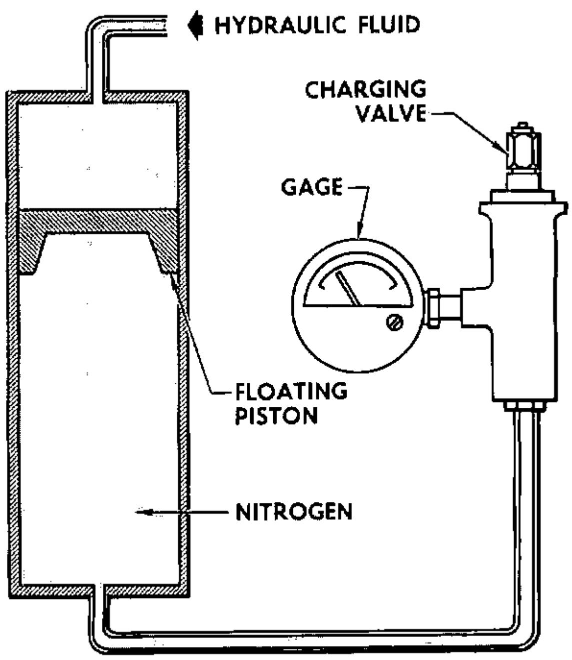

Figure 4-5. Hydraulic Accumulator Pressure Measurement

chamber. The purpose of the accumulators is to damp the pressure surges caused by the action of the hydraulic pump or relief valve, and to provide "backup" pressure to meet system demands until the pump furnishes the required flow. With the hydraulic pump not operating—when the engine is shut down—the accumulator pressure should be the normal pre-charge pressure of 750 psi. When the engine is operating, the hydraulic pressure should come up to about 3000 psi, moving the floating piston along the accumulator and compressing the nitrogen to the same pressure.

Thus, the hydraulic accumulator pressure indicators show the pre-charge pressures of the primary and secondary accumulators, when you are servicing the aircraft; and they show the actual pressure in the two hydraulic systems, when the engine is running, and the pressure pump is operating.

You can make a simple check of the accuracy of the accumulator pressure gages by comparing their readings with the indications of the hydraulic pressure indicator in the cockpit. Of course, these readings will only agree when the pump is in operation. An adjustment screw on the front of each accumulator pressure gage lets you correct the pointer position, but the gage should be checked against a master pressure gage before any adjustments are made.

PNEUMATIC PRESSURE INDICATOR.

The indicator which shows the pressure of the air in the high pressure pneumatic system is mounted on the forward bulkhead of the right main landing gear wheel well. This instrument is identical to the hydraulic accumlator pressure gages except for the dial graduations. To be more specific, it is a simple Bourdon-tube indicator. As you can see in figure 4-6, the dial is marked from 1000 to 3000 psi—3000 psi being the normal pre-charge pressure for this system. However, the indicator can show pressures up to 3500 psi.

PURPOSE OF THE INDICATOR.

This indicator is installed in the F-102A airplanes for convenience in servicing. In figure 4-6, note that the indicator connects to the tube between the filler connection and the priority air flask. Air pressure is stored in the flasks; they are recharged on the ground. The two permanent flasks are alwaye necessary equipment in the airplane; the alternate flasks can be added for they are required with certain types of armament.

When the air flasks are filled, pressure first builds up to about 1200 psi in the priority flask, then the priority air valve permits air to flow to the other three flasks. Conversely, when air pressure in the system drops to 1200 psi, the priority valve prevents any flow from the priority flask to the other flasks. Thus, approximately 1200 psi of air pressure is reserved to operate certain mechanisms which are connected to the priority flask. This air pressure is used to extend the landing gear and ram-air turbine, to deploy the drag chute, and to operate the main landing gear brake cylinders. For more detailed information, refer to the High Pressure Pneumatic System Training Supplement.

Like the hydraulic accumulator pressure gages, the pneumatic pressure gage has an adjustment screw just below the dial. You should use this adjustment only when the instrument is being compared with another indicator of known accuracy.

A-C VOLTMETER.

Power available from the alternating current system of the F-102A is measured by the a-c voltmeter. This instrument shows the voltage from the 26-volt, single-phase, step-down transformer, or shows the voltage of the 115-volt, 400-cycle, 3-phase, a-c supply. The voltage indicated at any particular time depends on the setting of a six-position selector switch. The voltmeter is situated on the right-hand auxiliary instrument panel; the selector switch is part of the

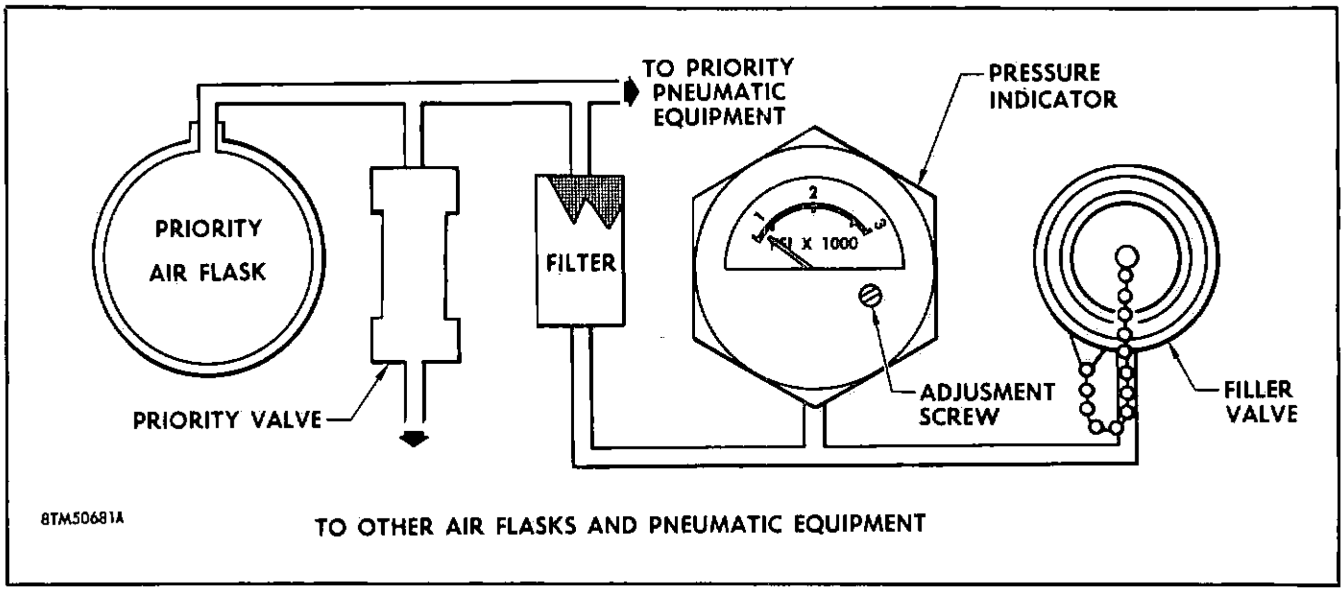

Figure 4-6. Pneumatic Pressure Indicator and Connecting System Schematic

power control panel which occupies the forward end of the rght-hand console, just below the voltmeter. In figure 4-7, you can see the voltmeter as it appears in the cockpit. You can find a complete description of the A-C System and components in the Electrical Manual, Chapter IV.

The voltmeter is a self-contained instrument which is mounted to the panel by means of locking nuts molded into the phenolic cover. Two electrical terminals, by which the voltmeter is connected to the a-c control panel, are located on the back plate. Just below the glass you can see the zero adjusting screw. This is the only adjustment that you can make.

OPERATION OF THE A-C VOLTMETER.

There are several types of voltmeters. The type which we are concerned with here is sometimes referred to as the iron-vane or movable vane type. Its mechanism consists of a stationary element and a movable element. The stationary element is comprised of resistors, a coil, a fixed vane, and supporting framework. A pointer, spring, and vane attached to a jewel-mounted shaft make up the movable element. These items are shown in figure 4-8.

Now let's see how the voltmeter comes up with an accurate measurement of the line voltage. Notice that the movable shaft and vane, and the fixed vane, are located inside the coil. As you know, when current flows through a coil a magnetic field is set up around the coil with lines of force most concentrated through the center. Since the iron vanes offer less reluctance (resistance) to the magnetic flux than the air around them, these vanes become magnetized. You learned why that is true in the discussion of electromagnetism in Chapter I.

Obviously, the magnetism of each vane is the same; that is, the north and south poles of each vane are in the same *relative* positions because they both reverse with each alternation of current. Thus, the vanes repel each other. One vane is fixed so the other one moves away, turning the shaft to which it is attached. The shaft rotates until the repelling force is equal to the restraining force of the spring, thus positioning the pointer accordingly. As you will recall from Chapter I, the strength of the magnetic force depends on the current in the coil. Since the current increases in proportion to an increase in voltage, the pointer position is an indication of the voltage across the instrument.

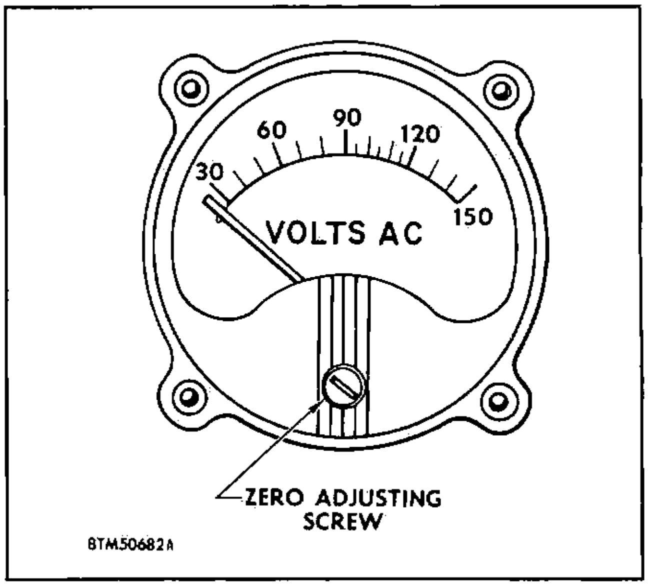

Figure 4-7. A-C Voltmeter

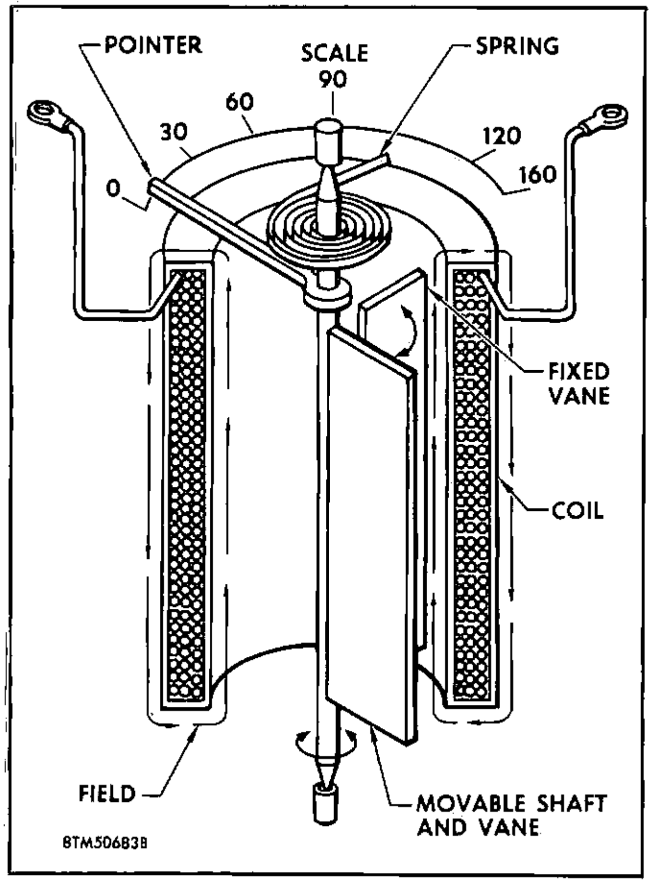

Figure 4-8. Voltmeter Movement—Operational Schematic

We have yet to consider the effects of temperature on the voltmeter. Naturally the electrical resistance of the coil, which is wound with many turns of copper wire, will increase with temperature increases. This resistance change would alter the reading of the voltmeter considerably if the total resistance of the instrument were low. To prevent these variations, the voltmeter contains two resistors which are connected in series with the coil. These resistors are wound of a high resistance alloy wire that is not affected by temperature changes. Therefore, the overall resistance of the instrument does not change much. You can even alter the length of the leads to the voltmeter without substantially affecting the total resistance.

Another feature of the voltmeter, which is of interest, is the damping provisions. Note that there is very little clearance between the outer edge of the movable vane and the inner wall of the cylindrical coil. Consequently, when the movable vane moves, it meets resistance from the air within the coil, since this air is being compressed. This resistance from the air reduces the swinging of the pointer caused by vibration or sudden changes in the line voltage. Of course, the air pressure quickly equalizes on both sides of the vane once the vane has stopped moving. The voltmeter is a very delicate instrument whose operation depends to a considerable extent upon proper balance and a minimum of friction. Should you doubt its accuracy, check the effectiveness of the zero adjusting screw. This screw adjusts the movement in a manner similar to the fast-slow adjustment on a watch. You should be able to move the pointer above and below the zero mark by rotating the screw a quarter of a turn in each direction from the zero position. To make a simple check for balance, remove the voltmeter from the panel. Hold the instrument in the normal mounted position and set the pointer exactly on zero. When you rotate the instrument a quarter of a turn in each direction, the pointer should not move far enough to show a visible gap between it and the zero line.

VOLTMETER SELECTOR SWITCH.

We mentioned earlier that the voltage indicated by the voltmeter depends on the setting of the selector switch on the electric power control panel. This switch is shown in figure 4-9. Notice that there are six positions or settings from which the F-102A pilot can choose. He can read the voltage of the 115-volt, 3-phase supply for phase A, B, C, AB, or AC, or the voltage of the 26-volt system. If the selector switch is in the 26-volt position and the generator is operating, the voltmeter should indicate 26-volts. In all other positions the reading should be 115-volts.

SUMMARY.

Although the location and purpose of some of the instruments in the power supply group are such that the pilots who fly the planes need not use them, it is still of utmost importance that these instruments be accurate. The safety of the pilot and aircraft depends on the proper functioning of the systems of which these instruments are a part, and any malfunctioning of the instruments could result in improper servicing of the systems.

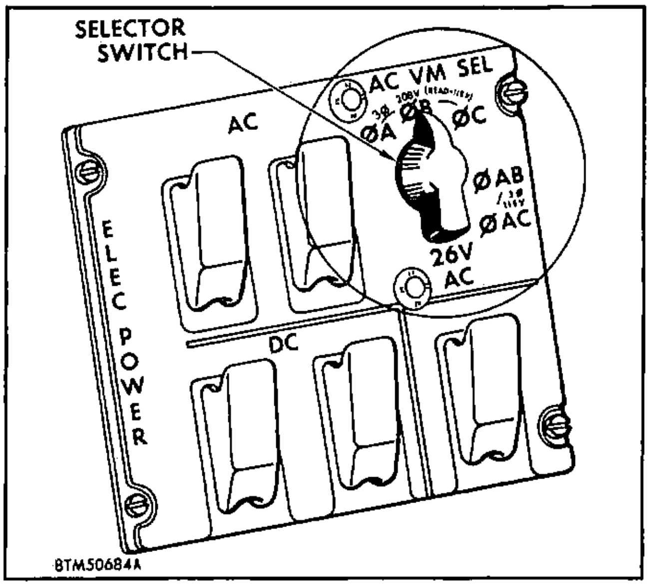

Figure 4-9. A-C Voltmeter Selector Switch

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Chapter V

MISCELLANEOUS INSTRUMENTS

Contents — *Page*

This chapter is devoted to the four instruments which, though not similar in function or operation, have one thing in common—they are not legitimate members of any of the instrument families that we discussed in the previous chapters. One of these instruments—the outside air temperature indicator—is used in conjunction with instruments in the flight and navigation group and in the engine group. However, only the early model airplanes of the F-102A series are equipped with this indicator; you will learn why that is true a little later in this chapter. This indicator is located on the right-hand console.

The other three instruments in this group are the cockpit pressure altimeter, the landing gear position indicating system, and the oxygen pressure and flow indicator. You will find all of these instruments on the left side of the cockpit. Since there is little similarity among the instruments covered in this chapter, we will discuss them in alphabetical order.

COCKPIT PRESSURE ALTIMETER.

The cockpit (cabin) pressure altimeter is a comparatively simple "cousin" to the sensitive pressure altimeter discussed in Chapter II. You should remember the basic principles on which altimeters operate and the definition of altitude from that discussion. In addition, the general description of aneroid instruments in Chapter I gives you the mechanical characteristics of this indicator. We won't repeat all of those things here; however, the important fact is that the cockpit pressure altimeter contains one aneroid diaphragm that turns a single pointer to show cockpit pressure up to an equivalent altitude of 50,000 feet. Figure 5-1 shows the cockpit (cabin) pressure altimeter as it appears in the F-102A. You will find this instrument on the left-hand auxiliary instrument panel under the CABIN AIR switch.

Understanding the actual mechanical operation of the cockpit pressure altimeter isn't too difficult, but you should know a little bit about why it is needed and what it tells the F-102A pilot. This instrument can also be helpful in certain tests you may make on the ground, and in giving you clues which will help in solving some problems encountered in the cockpit pressurization system trouble shooting. Remember, just because a pilot writes up a squawk on an instrument doesn't necessarily mean that the instrument itself is malfunctioning. In many instances an unusual indication is a sign of other faulty equipment that the pilot has no way of checking. Let's delve into this matter of cockpit pressurization a bit further to clarify what the indications of the cockpit pressure altimeter really mean.

COCKPIT PRESSURIZATION.

Since the molecules of air are evenly distributed, atmospheric pressure (under a given set of conditions and altitude) is equal at any point of contact. That means the air pressure on the human body is equal at all points, so we are not aware of pressure at any given level. We are, however, aware of *changes in pressure*—especially sudden changes like those felt when an airplane ascends or descends rapidly. For example, sudden drops in pressure, as experienced during a high speed climb-out, do not give the body time to adjust, and gases trapped in the body expand causing much discomfort or even death.

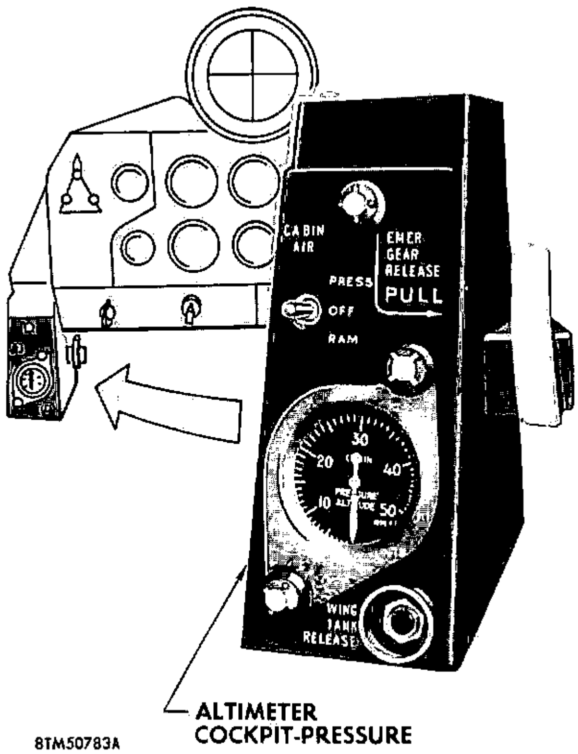

Figure 5-1. Cockpit Pressure Altimeter

Besides the dangers of rapid pressure changes, there are certain maximum and minimum pressure limits beyond which the human body cannot function properly. At 42,000 feet the average air pressure is only 2.47 psi (as compared with 14.7 psi at sea level), and that simply is not sufficient to maintain normal body functions.

Pressurization reduces the effect of the changing or reduced air pressure on the F-102A pilot by restricting the rate of pressure change within the cockpit, and by keeping the pressure at a safe level. Thus, a pressurized cockpit is one in which the air pressure is maintained at a level greater than the ambient pressure of the air around it. The most convenient way to express this pressure level is in terms of pressure altitude. The cockpit pressure altimeter case is vented to the cockpit rather than to oustide static air, so it shows the cockpit air pressure in terms of the corresponding pressure level in the theoretical "standard atmosphere."

You can see in figure 5-2 what the cockpit pressure altimeter should indicate at various altitudes. Note that the graph shows the airplane altitude in 1000 foot intervals and two altitude instruments; the upper instrument airplane altitude, and the lower instrument cockpit or cabin altitude. The gray band represents the approximate difference between the two altitudes. The example in figure 5-2 shows how an airplane altitude of 40,000 feet equals a cabin altitude of approximately 17,000 feet. By using this example you can determine the approximate cockpit altitude for any airplane altitude. On the right side of the graph, note that above 26,500 feet there exists a 5 psig pressure differential between the two altitudes.

Operation of the Cockpit Pressurization System.

We've established *why* the F-102A cockpit is pressurized—now let's briefly discuss *how* it is pressurized. Then you will be better able to interpret any trouble symptoms given by the cockpit pressure altimeter readings. Air can be brought into the cockpit of the F-102A either directly from the left-hand boundary-layer intake, or from the 16th stage of the J57 engine compressor. The cabin air switch on the left-hand auxiliary instrument panel (shown in figure 5-1) is used by the pilot to control the source of incoming air.

In the RAM position the boundary-layer ram air is used and the cockpit is unpressurized. In the PRESS position, the air intake is from the engine "bleed" air. This is the normal operating position. The RAM position is usually used only if the pilot feels that the other system of air intake is malfunctioning. When the switch is in OFF, all air flow to and from the cockpit is shut off, and air will gradually leak from the cockpit until the pressure is equal to the outside air pressure.

Air bled from the engine compressor is at high pressures and temperatures (sometimes as much as 800°F.). Part of this air is routed through the airplane's refrigeration unit where its temperature is reduced. The remainder of the bleed air bypasses the refrigeration unit. An outlet line from the refrigeration unit joins the bypass line. By the use of a manual temperature selector switch on the utility switch panel (on the instrument panel skirt), the pilot can control the temperature of the air which enters the cockpit. This is accomplished by a valve in the bypass line which regulates the amount of straight bleed air that is mixed with the refrigerated air.

The actual pressure of the air in the F-102A cockpit is automatically regulated by a pressure regulator, a safety valve, and a rate sensor. You can locate these components in figure 5-3. The pressure regulator, located in the cockpit floor between the rudder pedals, regulates the flow of air out of the cockpit by balancing cockpit pressure against static air pressure (ambient air). The safety valve performs the same function as the pressure regulator but it has higher

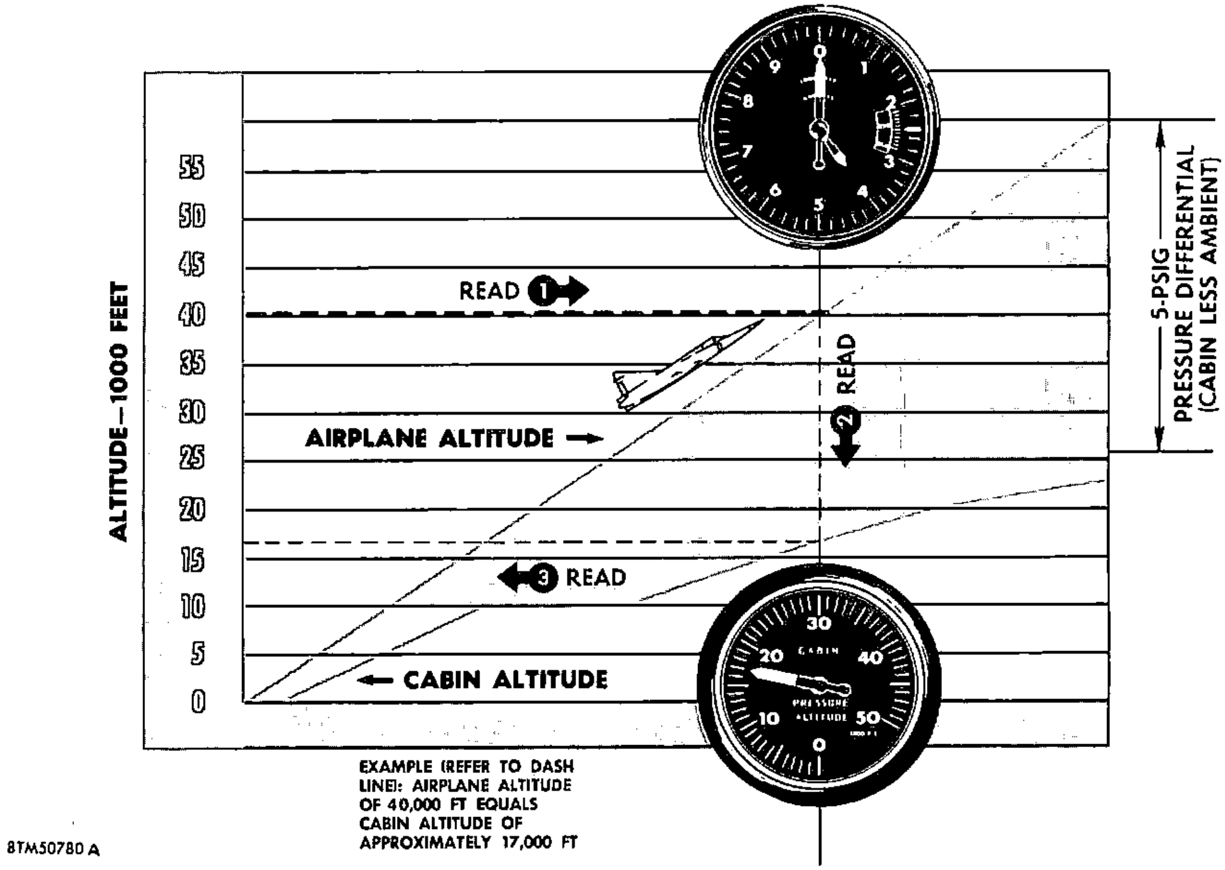

Figure 5-2. Cockpit Pressurization Schedule

relief settings. It is located on the bulkhead behind the pilot's head-rest. Thus, this valve does not function when the cabin air switch is in the PRESS position except when the pressure regulator fails. The rate sensor is connected by a pressure sensing line to a flow control valve in the engine compressor bleed line. The flow control valve is open whenever the switch is in the PRESS position. This rate sensor serves to limit the rate at which the pressure can build up in the cockpit to a maximum of 4 psi per minute. Remember, any rapid change in pressure endangers the health and efficiency of the pilot.

The pressure regulator or the safety valve regulates the cockpit pressure in three stages. Let's consider the pressure regulation first, since that is the normal regulating mechanism.

PRESSURE REGULATOR. Up to 10,000 feet of altitude the pressure regulator permits practically unrestricted flow of air from the cockpit; this is known as the *unpressurized operation* stage. Between 10,000 feet and 26,500 feet, the pressure regulator maintains the cockpit pressure at about 10.2 psi, which is equivalent to the normal atmospheric pressure at 10,000 feet. This is known as the *isobaric stage* (iso—equal, baric—pressure). Above 26,500 feet altitude, the air pressure in the cockpit is maintained at about 5 psi above the pressure outside of the cockpit (ambient air pressure). This is called the *differential operation* stage. As you can see, figure 5-2 shows graphically how the pressure altitude in the cockpit compares with the pressure altitude outside of the cockpit when the regulator is operating normally.

In the event of pressure regulator malfunctioning, the safety valve prevents excessive pressure build-up in the cockpit. If this did not occur, something would have to give—probably the cockpit canopy seal or the glass—and the pilot would be subjected to explosive decompression. That would be very bad. Once again, rapid change of pressure is tough on pilots.

SAFETY VALVE. There are also three stages of operation of the safety valve. The primary *differential operation stage* (corresponds to the unpressurized stage of the regulator) is in effect from 0 to 13,000 feet. In this stage the cockpit pressure can build up to about 1.5 psi over atmospheric pressure. The same term is used for the second stage of operation of the safety valve as is used for the regulator—*isobaric*. This stage is from 13,000 feet to 26,500 feet. From 26,500 feet on up, the *secondary differential* stage of operation is in effect. This is the same as the *third* (differential) *stage* of the pressure regulator, except that the upper limit is higher; it permits a pressure differential of about 5.25

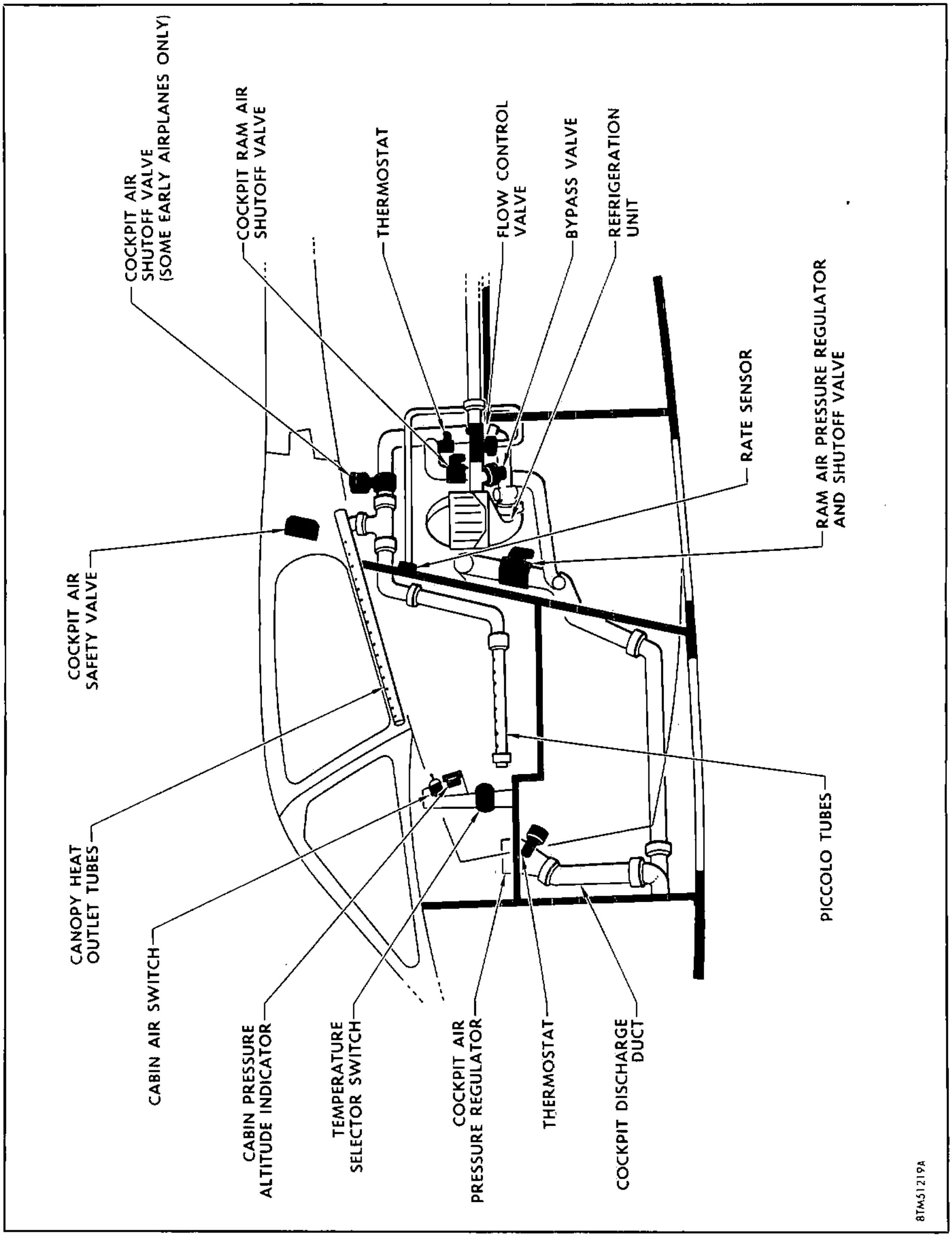

Figure 5-3. Cockpit Air Conditioning and Pressurization System

psi instead of 5 psi. Thus, you can see that the safety valve accomplishes the same basic job as the pressure regulator, except that it is inoperative unless the regulator goes out. The reason it is normally inoperative is, of course, due to its higher relief limits that are designed to relieve pressure in case of regulator failure.

OTHER PRESSURIZATION SYSTEM COMPONENTS. There are several other components which we should mention just briefly in our general discussion of the cockpit pressurization system. (For a more comprehensive discussion of this system, refer to the Low-Pressure Pneumatic System Supplement of this training series.) These components are the landing gear safety switch, the armament door switch, and the air control timer. The landing gear safety switch opens the safety valve whenever the weight of the airplane is on the main landing gear. If this pressure relief feature were not provided and the pressure regulator malfunctioned, the safety valve would permit cockpit pressure to remain at about 1.5 psi above ambient pressure at touch-down. If this pressure were not dumped (vented) when the airplane landed, the pilot would get quite a surprise when he unlatched the canopy.

The armament door switch and air control timer shut off all air intake to the cockpit during armament firing, and for several seconds thereafter. This feature prevents gases and corrosive dust (products of combustion of the armament propellant) from entering the cockpit. Since a certain amount of leakage around the canopy seal and through the structure is unavoidable, the cockpit pressure gradually reduces during the armament firing cycle. Thus, the cockpit pressure altimeter shows a continuous pressure altitude increase during the period of armament firing.

Having covered the highlights of the cockpit pressurization system, you can now view the cockpit pressure altimeter from the standpoint of utility. In other words, you can use its indications to pin-point specific problems.

EVALUATION OF ALTIMETER INDICATIONS.

Figure 5-2 shows what the cockpit pressure altimeter *should* indicate at various altitudes, but what if it doesn't? Well, there are lots of things that could cause false indications. Probably the least likely is an error in the indicator itself. Of course, since the cockpit altimeter has no external controls (and therefore cannot be adjusted for variations in sea level pressure), there could be some variation from the reading of the sensitive pressure altimeter even when the cockpit is totally unpressurized. However, this is nothing to worry about since the cockpit pressure altimeter is used only to indicate pressure conditions, not flight level or terrain clearance.

Without going into detail on trouble shooting of the pressurization system, let's consider what some peculiar indications might tell you. From our previous discussion on cockpit pressurization you know that the rate at which the cockpit pressure builds up is controlled by the rate sensor. If a pilot reports that the cockpit pressure altimeter fluctuates considerably —indicating pressure surges—the rate sensor may be to blame. Consistently *high* indications suggest a pressure leak somewhere in the system. On the other hand, the altimeter will indicate consistently *low* at some altitudes if the pressure regulator fails. This is because the safety valve permits higher cockpit pressures than the pressure regulator, and the greater the cockpit pressure the lower the altitude indication.

These are just a few examples which show you how knowledge of the cockpit pressure altimeter can help you interpret various trouble symptoms in the pressurization system. However, you should not assume that the instrument is *never* at fault—if you doubt its accuracy, it's a simple matter to remove it and check it out in the instrument shop.

LANDING GEAR POSITION INDICATING SYSTEM.

Position of the landing gear on the F-102A is shown on the upper left side of the main instrument panel. There are three separate dials; one for each main gear and one for the nose gear. Each of these dials can give three different indications. When one of the indications shows UP, the corresponding gear is up and locked. When the gear is down, a picture of the landing gear wheel shows. Diagonal lines appear on the dial if either the electrical power source is cut off or the landing gear is in transit between the *up-and-locked* and *down-and-locked* positions. As you can see in figure 5-4 these dials are arranged to correspond to the relative locations of the airplane landing gear.

THE INDICATORS.

The only difference between the landing gear position indicators and the afterburner exhaust nozzle position indicator (which was discussed in Chapter III) is the way in which the rotating cards are marked. As you will recall from that discussion, the indicator contains two solenoids and a spring-centered, three-position card. The indication given depends upon which position of the card shows in the window. This position indication depends upon whether the card is held centered by the spring or has been rotated by one of the solenoids. An operational schematic and the card markings for one of the landing gear position indicators are shown in figure 5-5. You can see by the directional arrows how the energizing of a solenoid rotates the card. When

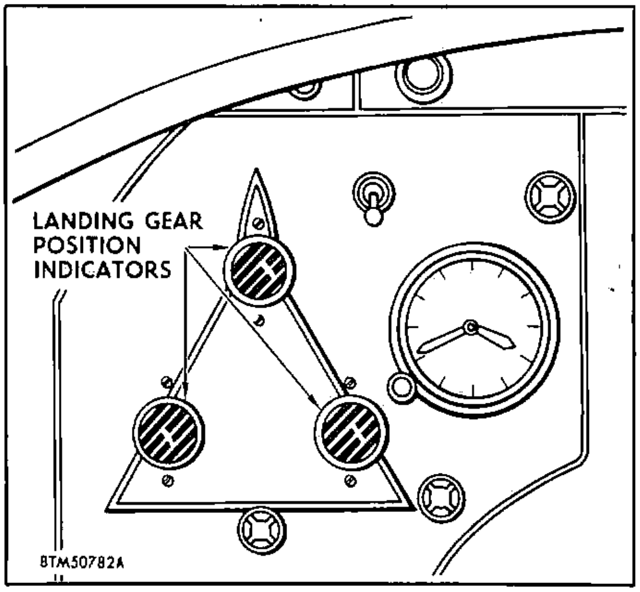

Figure 5-4. Landing Gear Position Indicator Arrangement

neither solenoid is energized, the spring centers the card. Remember that all three indicators are identical.

POSITION SWITCHES.

Several switches are involved in the landing gear position indicating system. They are actuated by contact with moving parts of the landing gear and wheel well door mechanisms to provide current from the 28-volt d-c supply to the indicators. Various combinations of switching action result in different circuits being energized and therefore different indications in the cockpit. The switching action used for the nose landing gear (NLG) is not the same as for the main landing gear (MLG), so let's discuss them separately.

NLG Switching Action.

Four switches affect the NLG position indication. These switches are the *NLG position up switch* located on the under side of the cockpit floor in the nose wheel well (NWW); the *NLG position down switch* which is also in the NWW on the canted (aft) bulkhead, the *NLG lock switch* mounted on the up and down latch mechanism of the gear drag strut, and the *NWW door cylinder switch* attached to the door actuating cylinder. Note the location of these switches in figure 5-6.

Now suppose that the nose landing gear is up-and-locked. To get an UP indication on the NLG position indicator, the UP circuit is energized as shown in the schematic (figure 5-7). Note that the power supply terminal of the indicator is energized and the UP solenoid is grounded. The power supply circuit is energized when the NLG lock switch is closed by the latching mechanism. Two switches are closed to provide the ground—the NWW door cylinder switch and the NLG position up switch. A lock within the door closing cylinder actuates the door cylinder switch, permitting current flow to the *NWW door position indicator relay;* this relay is also shown in the illustration. When the relay is closed, and when the NLG position up switch (contacts A and C) is held closed by pressure of the retracted gear, the circuit is completed to ground.

Only two switches are involved in the NLG DOWN indication. The power supply circuit is again energized by current flow through the NLG lock switch, just as in the up-and-locked position. To complete the DOWN circuit, the NLG position down switch is closed (contacts A and C). This switch furnishes the ground path for the DOWN indication circuit, and is actuated by the steer-damper unit linkage. Thus, the down solenoid is energized and the miniature wheel shows on the indicator dial.

MLG Switching Action.

Both MLG position indicators are energized by the same kind of switching action. The switches involved are the *MLG door closed switches* and the *MLG down-and-locked switches.* Each MLG has a forward and an aft MLG door closed switch, located on the forward and aft wheel well bulkheads, and a MLG down-and-locked switch on the downlatch mechanism on the side brace. Figure 5-8 shows the location of these switches on the main gear and in the main gear wheel well. Both right and left MLG position indicating switches are the same so only the left one is shown.

In figure 5-9, note that the power supply terminals of the left main landing gear position indicator is always connected to the airplane 28-volt d-c system, regardless of the gear position. To get an UP or DOWN indication on an indicator then, it is only necessary that the appropriate indicator solenoid have a path to ground. When neither solenoid is grounded, the rotating card of each indicator is centered by spring tension and the diagonal lines show.

The main landing gear on the F-102A is locked in the UP position by the wheel well doors. When these doors are completely closed, the forward and aft MLG door closed switches are actuated by the door actuator cylinder bell crank arms. This action grounds the up solenoids in the indicators, resulting in an UP indication. When the main landing gear is fully extended, the downlatch mechanism actuates the MLG down-and-locked switches. Thus, the down solenoid circuits are completed to ground and the DOWN indication appears on the indicators.

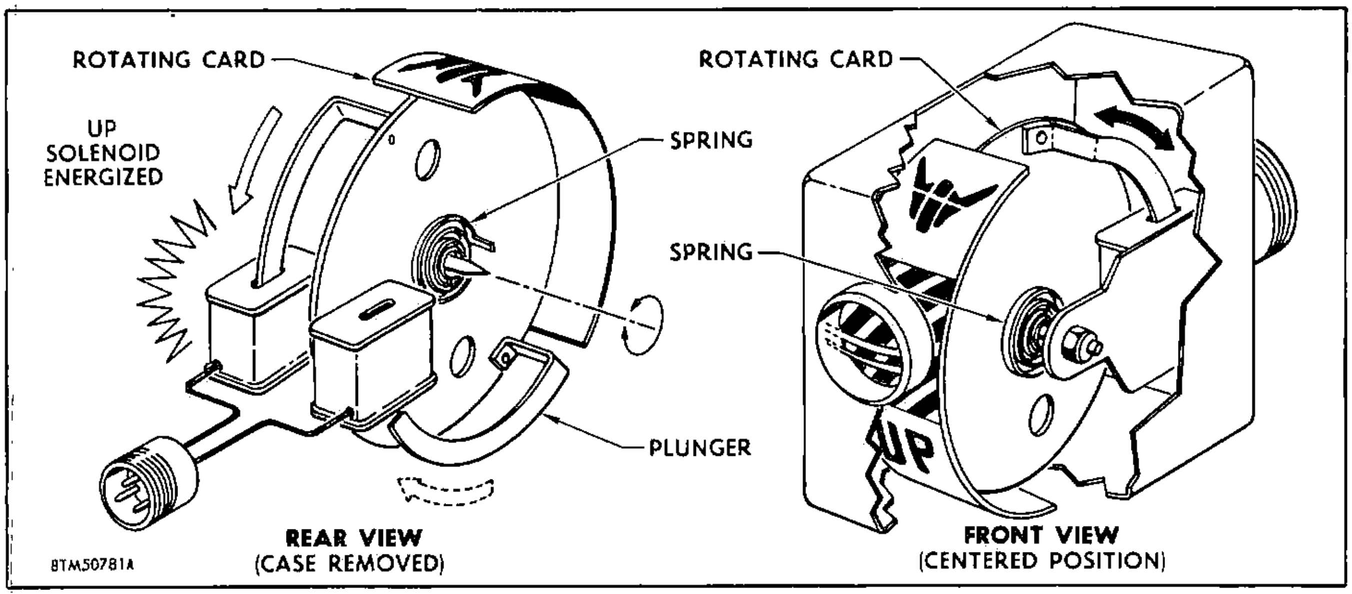

Figure 5-5. Landing Gear Position Indicator Operational Schematic

PROBLEMS OF THE SYSTEM.

You should have very little trouble with the landing gear position indicators, but any problems that occur anywhere in the system warrant your careful attention. All three indicators are identical, so switching the leads during a retraction test may help you to isolate a problem. For example, suppose that an F-102A pilot reports that his nose landing gear does not indicate *up-and-locked* when the control lever is in the *retract* position. Naturally, you have to determine whether the gear is not retracting fully or whether the indicating system is in error.

By switching leads on the nose landing gear indicator with one of the main landing gear indicators, you can tell if the trouble is within the indicator or some other part of the system. A faulty indicator will not indicate correctly when connected to any system. Similarly, if the nose landing gear isn't retracting properly, or if the switching action or leads are causing the error, any of the three position indicators will read incorrectly when connected to the nose landing gear indicating circuits.

The landing gear position indicators are sealed units, having no external adjustments, so they must be replaced if faulty. Any problem traced to causes other than the indicator should be checked out in accordance with instructions in your F-102A maintenance manual.

OUTSIDE AIR TEMPERATURE INDICATOR.

In our prevous discussions of instruments we have pointed out a need for air temperature information on the part of the F-102A pilot. For example, he must know the outside air temperature to compute true airspeed from his indicated airspeed. He also needs this information to set the marker (or markers) on his pressure ratio indicator correctly. Thus, the outside air temperature indicator is sometimes grouped with engine or with flight and navigation instruments.

Accurate ground temperature and estimated flight level temperatures are available from any control tower. Consequently many airplanes of the F-102A series are not equipped with an outside air temperature indicator. Nevertheless, you should know something about this indicator in case you work on the airplanes in which it is installed.

CONSTRUCTION.

The outside air temperature indicator is a resistance-type thermometer which shows the temperature of the air outside the airplane within a range of —50° to +50° centigrade. Internally, the indicator contains two coils mounted on a common shaft which turns within a core. The poles of a permanent magnet surround this core. Three springs carry current to and from the coils and rotate the movement so that the pointer is off the scale when the power supply is interrupted. Several resistors are included in the instrument circuit, and a plug at the rear of the case connects to a resistance bulb and the airplane 28-volt d-c power supply. Figure 5-10 shows the face of this instrument and the electrical schematic of its hook-up in the airplane. The components mentioned above are illustrated schematically when we discuss the operation of this instrument.

OPERATION.

Like the exhaust temperature indicator which we discussed in Chapter III, the outside air temperature indicator operates on the principles of magnetism. The

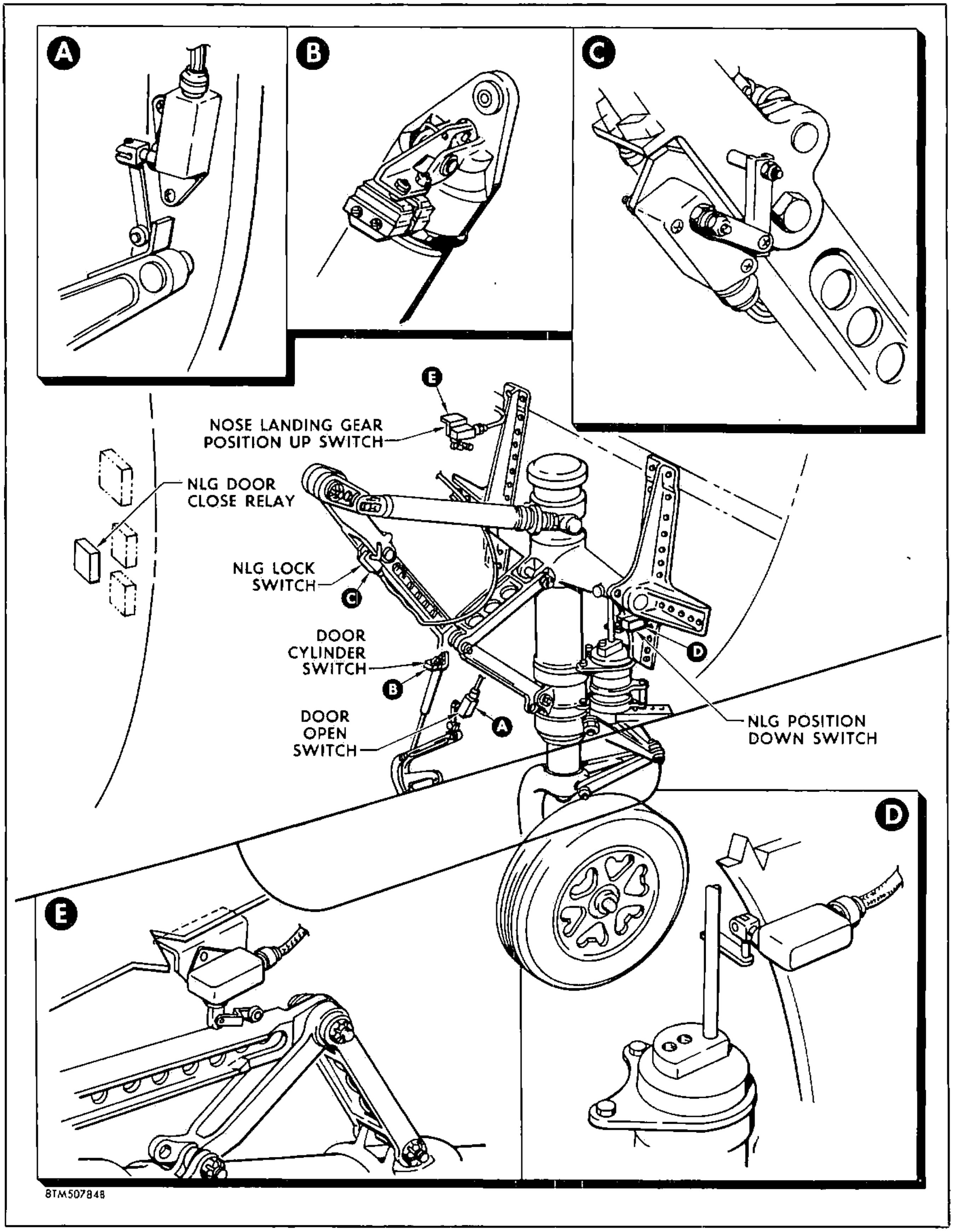

Figure 5-6. Nose Landing Gear Position Indicating Switches

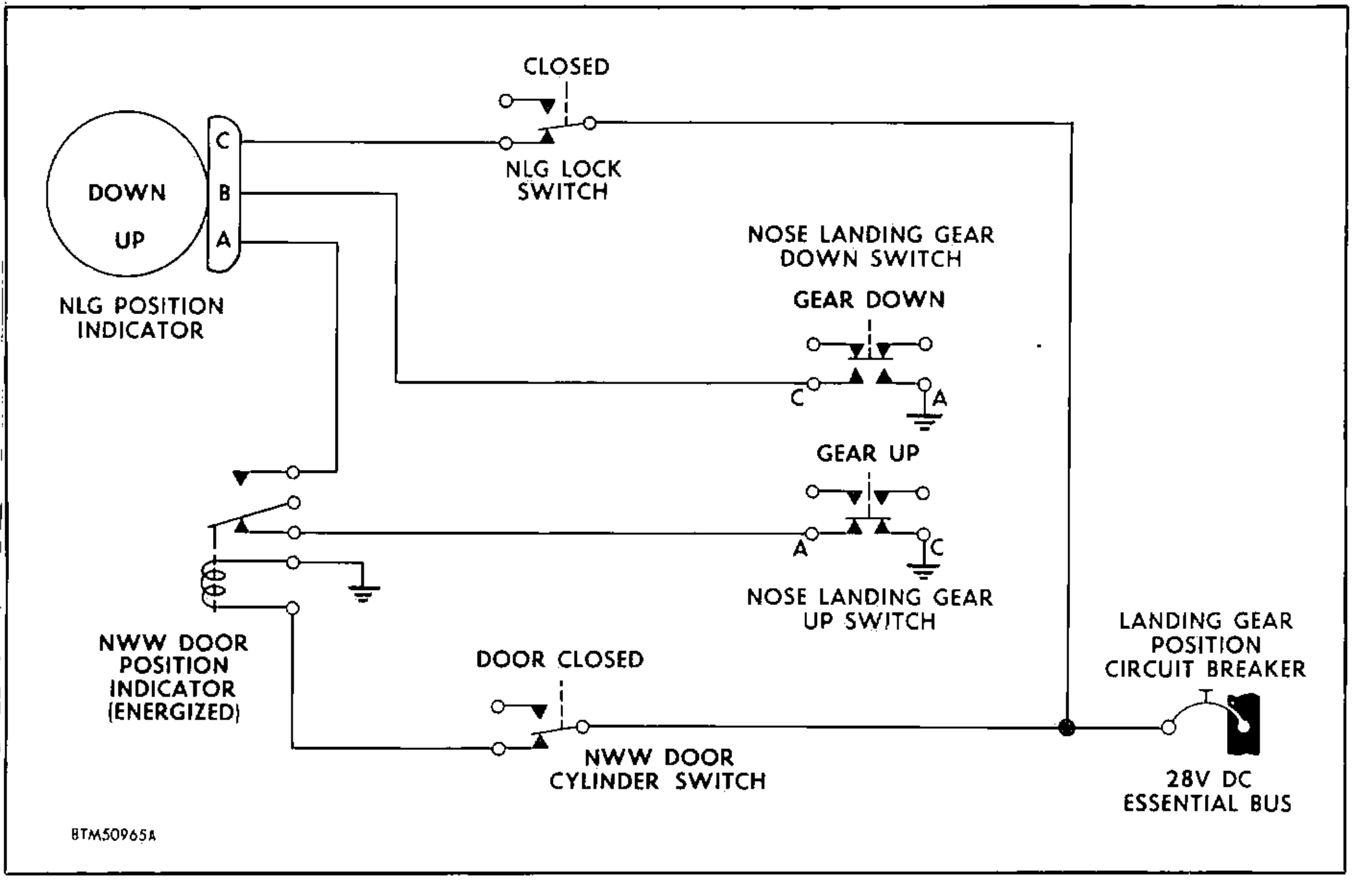

Figure 5-7. Nose Landing Gear Position Indicating Schematic

pointers of both instruments are positioned by the relative strength of electromagnetic and permanent magnetic fields. However, the similarity stops at this point. As you will recall, the exhaust temperature indicator operates on the thermo-electric principle—that is, the difference of electrical potential of two dissimilar metals. The outside air temperature indicator, on the other hand, requires power from the d-c system because the temperature range which it measures is not great enough to use the thermo-electric principle. Instead, this instrument simply measures the relative resistace of two parts of a bridge circuit. For that reason, it is sometimes referred to as a ratio-type indicator, or ratiometer.

The Ratiometer Mechanism.

To help you understand the operation of the ratio-type instrument, let's go back and briefly review some principles which are involved in its operation. First of all, you will remember that the magnetic field of force around any magnet takes the path of least reluctance. If the lines of force can flow through a core of metal, they will do so rather than flowing through the air. By the same token, if the two poles of a magnet are closer together in one place than in another, the flux will be most concentrated at the place where they are nearer together. Another point which you should remember is that when current flows through a coil of wire, a magnetic field is set up. When the amount of current increases, the strength of the magnetic field around the coil increases.

If this coil is suspended in the field of a permanent magnet, the force, or torque, which tries to rotate the coil becomes greater as the current increases; and the more loops of wire in the coil, the greater will be the torque produced. Thus, the torque depends primarily on three things—the number of loops of wire in the coil, the current flow through the coil, and the strength of the field of force (flux density) of the permanent magnetic field around the coil.

Now take a look at the operational schematic of the ratiometer-type of instrument. (See figure 5-11.) Note that the two coils of wire are mounted directly opposite each other on a common shaft. As you can see, a core is attached in such a way that the coils surround it and come between the core and the permanent magnet. Note also that the core is mounted eccentrically; that is, it is closer to the magnet in some places than in other places. In this instrument, the core is mounted high in the circular space between the poles of the permanent magnet. Thus, the flux density is greatest at the top of the core and decrease proportionally on both sides of the core.

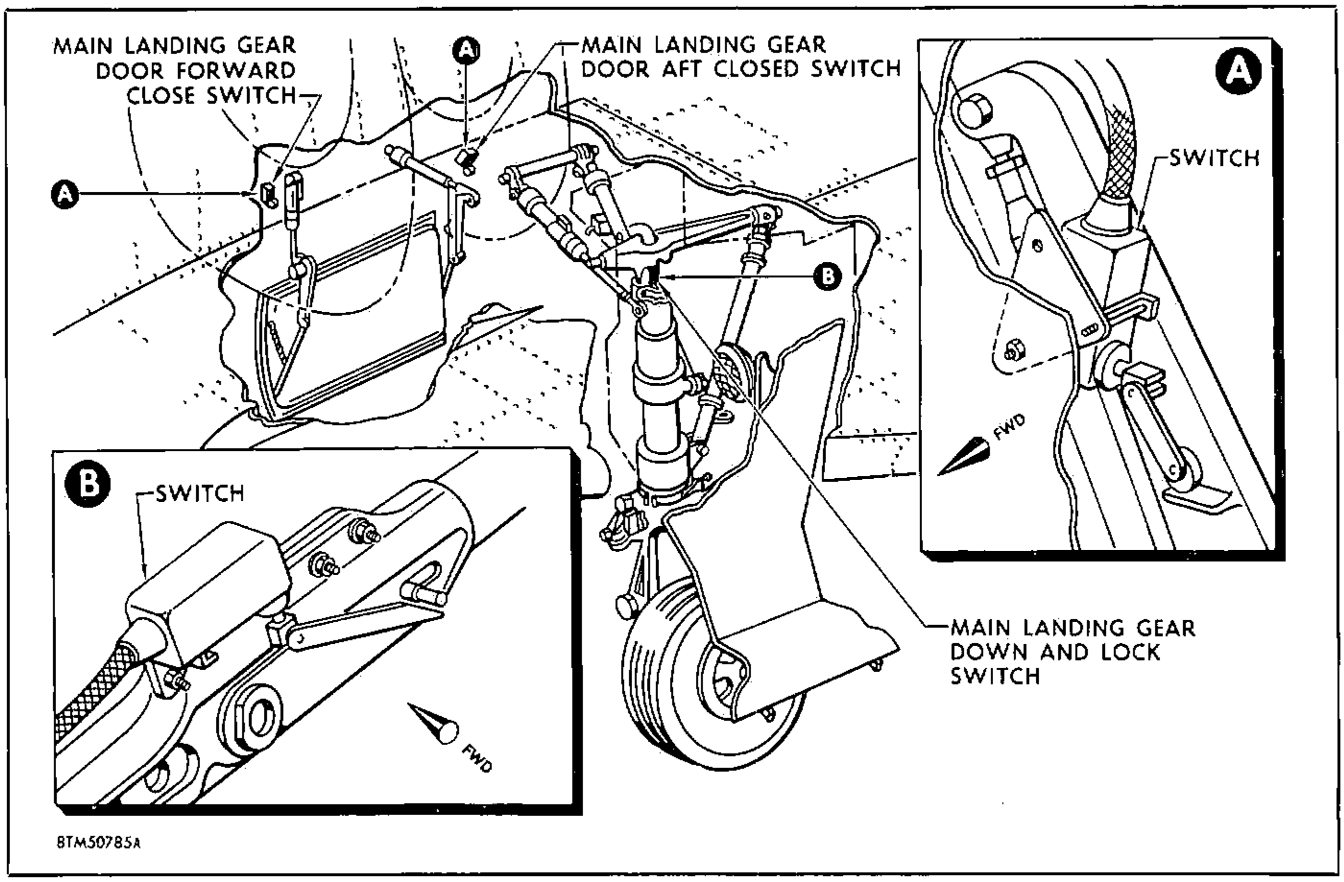

Figure 5-8. Left Main Landing Gear Position Indicating Switches

When the coils rotate, they move in this space between the magnet and the core. We have determined that the torque produced is proportional to the amount of current flowing through the coil and the flux density of the magnetic field in which the coil is positioned. Therefore, you can understand why the torque of these particular coils is determined by the resistance of the circuit, the voltage, and the magnetic field strength.

As you can see in figure 5-11, when one of the coils rotates, the other must also move, because they have a common shaft. Consequently the indication given by the attached pointer is the result of the combined forces acting on the two coils. These coils are wound in such a way that both of them attempt to turn, or apply torque, in the same direction as shown by the arrows in the cutaway view. Thus, as the lines of force flow from one end of the permanent magnet to the other, they cause both coils to push downward. The coil which has the greatest current flow will have the greatest rotational force and will rotate down, forcing the other coil upward.

In moving downward, the coil having the greatest current flow is moving from a position of high flux density to a position of low flux density, and the other coil is being forced upward to a position of greater flux density. Now, if the coil that is moving downward is losing torque because of the decreasing flux density, and the other coil is increasing its torque by entering an area of greater flux density, then the two coils will reach a certain point where the torques balance. At this balanced condition, the coils and pointer come to rest.

Getting back to our current flow, you will recall that the indicator is connected to the 28-volt d-c system. One of the instrument coils has a circuit containing certain fixed resistances. The other coil has the temperature sensing bulb connected in series with it. Since the temperature bulb varies in resistance with changes of temperature, the current flow through that coil will vary. At times it will permit a greater current flow than the other coil, and at other times less current flow. How much current it permits to flow through it at any one time determines how much torque is produced and therefore, determines the balance-point of the two coils. In so doing, of course, the current flow determines what position will be taken by the indicator pointer.

A discussion of the ratio-type measuring instrument wouldn't be complete without mentioning its particular advantages over other resistance indicators. As

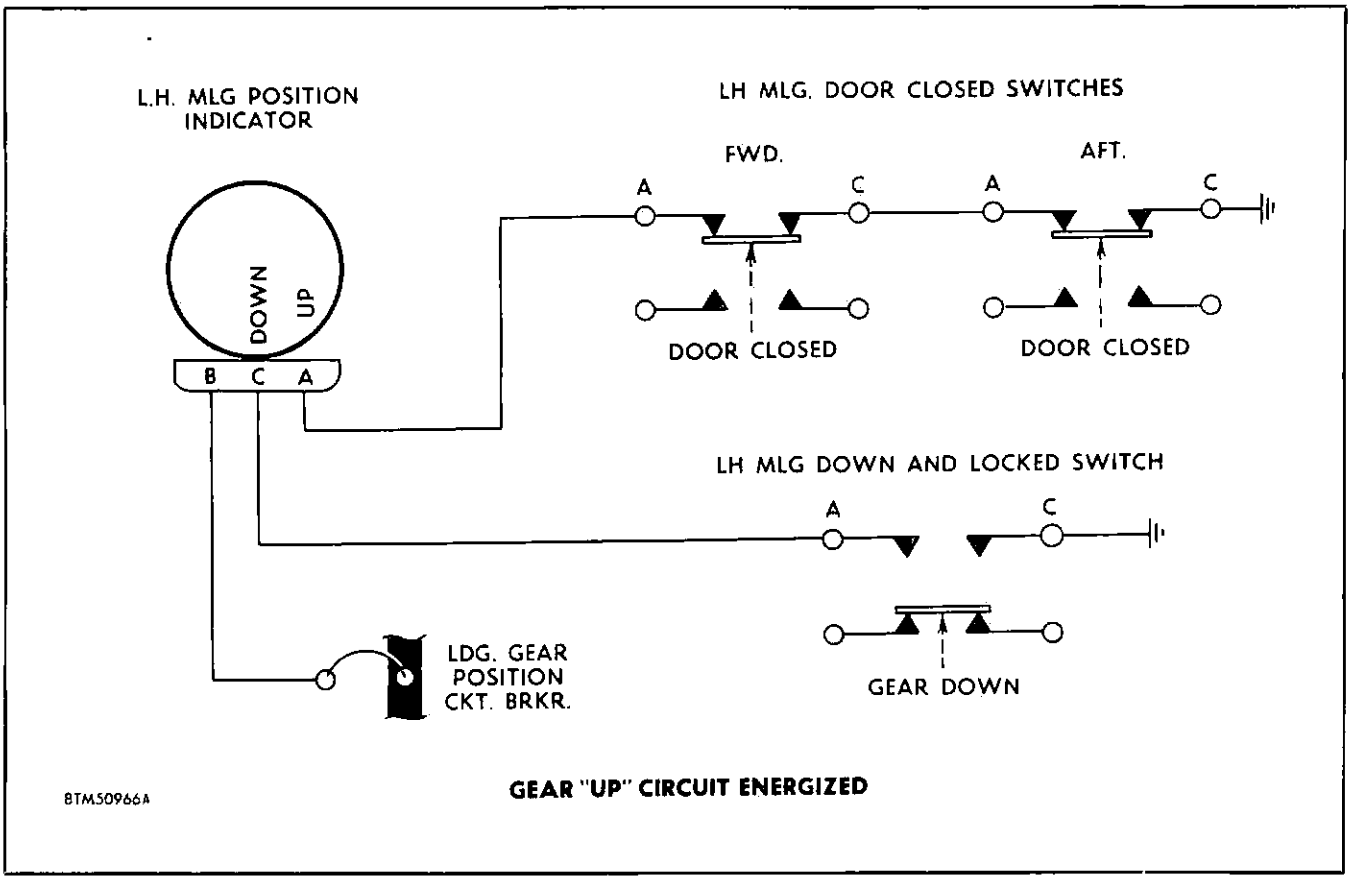

Figure 5-9. Left Main Landing Gear Position Indicating Schematic

you know, the 28-volt d-c system of an airplane is the generator-battery power system. Any direct current system of electrical power is subject to fluctuation. If an indicator is dependent upon the current input, there will be similar fluctuations in its indications. This is where the ratiometer instrument is superior to other types; it's practically unaffected by normal current fluctuations. As we pointed out, its readings are the result of compared torque values of two different coils. If the overall power input to one coil varies, the power to the other coil varies the same amount. Thus, the ratio of the two torques is the same regardless of current input.

The Bulb.

We can't disregard the resistance bulb, since it's an important part of the electrical circuit of one of the coils. As you can see in figure 5-12, this temperature sensing bulb is located in the engine air intake stub duct and looks much like the thermocouples used in the exhaust temperature circuit. However, this bulb contains a wire of a single material which is greatly affected by temperature changes—when the wire is hot, its resistance is much greater than when it is cold. The hotter it becomes, the more resistance it offers to the flow of current. This wire is encased in a hollow tube, sealed at one end and brazed to a fitting at the other end. A filling material insulates the wire from the tube. The only reason it is necessary to cover the wire, is that it could become corroded or the looped wire could become "shorted" in one way or another, thus destroying its normal resistance characteristics.

MAINTENANCE.

The outside air temperature indicator and temperature sensing bulb are both sealed units having no external controls or adjustments. If any problems arise which are caused by anything other than faulty wiring, you simply remove and replace the defective unit. The problem, then, is to determine where the trouble originates so that you do not have to replace parts which are in perfectly good condition.

The reasons for some malfunctions which you may encounter should be obvious from what you have already learned in this discussion. For instance, if the pointer remains on its off-scale position (mechanical zero, not the zero degree position) when d-c power is on, you can expect to find one of two causes: either the power supply circuit is open or there is a poor ground connection. Should the pointer remain on-scale with power off, the indicator mechanism is probably "hung up" due to friction or foreign matter in the movement.

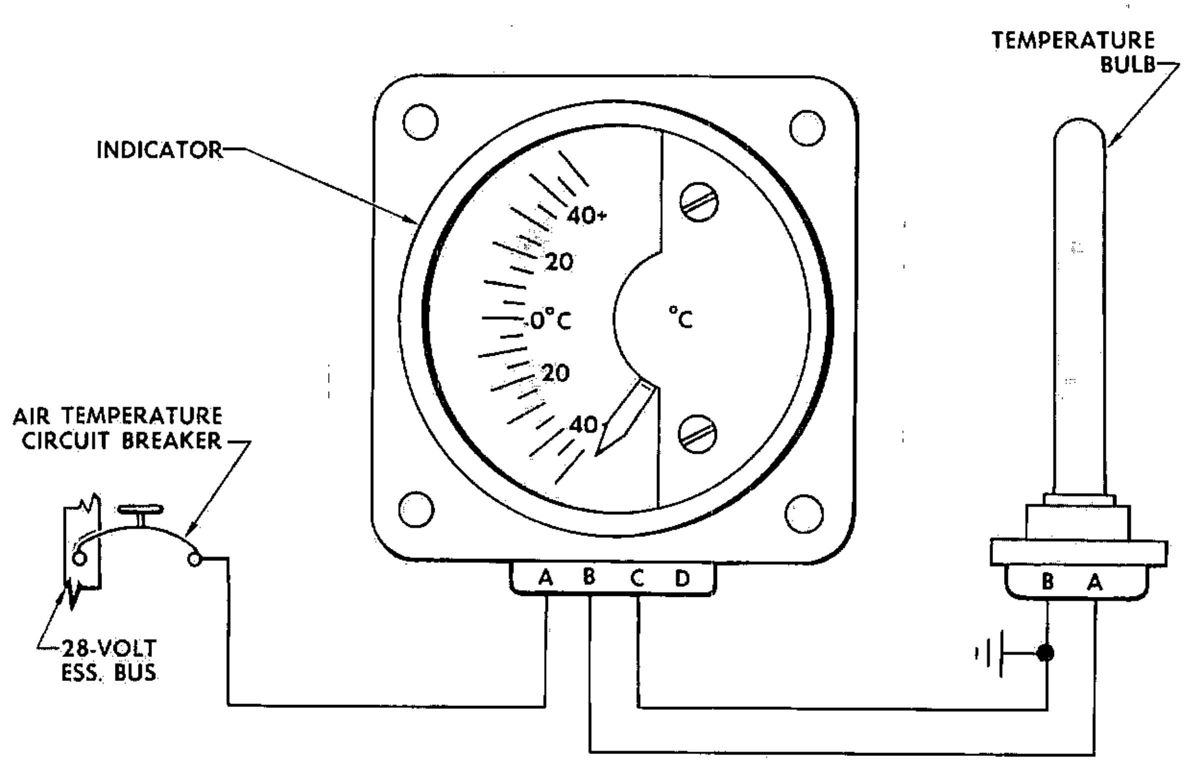

Figure 5-10. Outside Air Temperature Indicator

An excessively high reading indicates that the resistance bulb or its leads are cut off from the indicator, thus causing extremely high resistance in the coil connected to that circuit. On the other hand, if the pointer is forced off the low side of the scale, you probably will find that the leads or bulb are not providing enough resistance due to a short circuit. To test the outside air temperature indicator you should have either a thermometer tester or a decade resistance box. Complete instructions for testing all types of resistance thermometers are provided with this equipment.

OXYGEN PRESSURE AND FLOW INDICATOR.

The oxygen pressure and flow indicator in the F-102A is an integral part of the automatic positive pressure diluter demand oxygen regulator (oxygen pressure regulator, for short). This regulator constitutes the entire oxygen control panel on the left-hand console. In addition to the indicator, the regulator also includes a supply lever, emergency lever, and diluter lever, as shown in figure 5-13. The other major components of the F-102A oxygen system are the supply cylinder mounted in the forward end of the right-hand armament bay, the filler equipment in the nose wheel well, and the pilot's mask. We will confine this discussion to some background information on the need for an oxygen system and to the actual operation of the oxygen pressure and flow indicator. A complete explanation of the oxygen system is included in another training manual in this series.

PURPOSE OF OXYGEN EQUIPMENT.

Earlier in this chapter we discussed the reason for cockpit pressurization at high altitudes. You will recall that the cockpit pressure, during normal operation, is kept higher (a lower pressure altitude level) than ambient pressure. For example, the chart in figure 5-2 shows that when the airplane is flying at 40,000 feet, cockpit pressure is maintained at the 17,000 foot pressure level. However, that doesn't solve all the problems of pilot comfort and safety. Even at 17,000 feet the atmosphere is too rare for sustained breathing. Thus, military airplanes have carried oxygen equipment since long before cockpits were pressurized.

There is considerable variation in oxygen requirements of different persons and of any one person at different times. Generally it is considered bad practice to fly at altitudes above 10,000 feet without oxygen equipment. With a cockpit pressure altitude of 20,000 feet, the average individual will lose consciousness in about

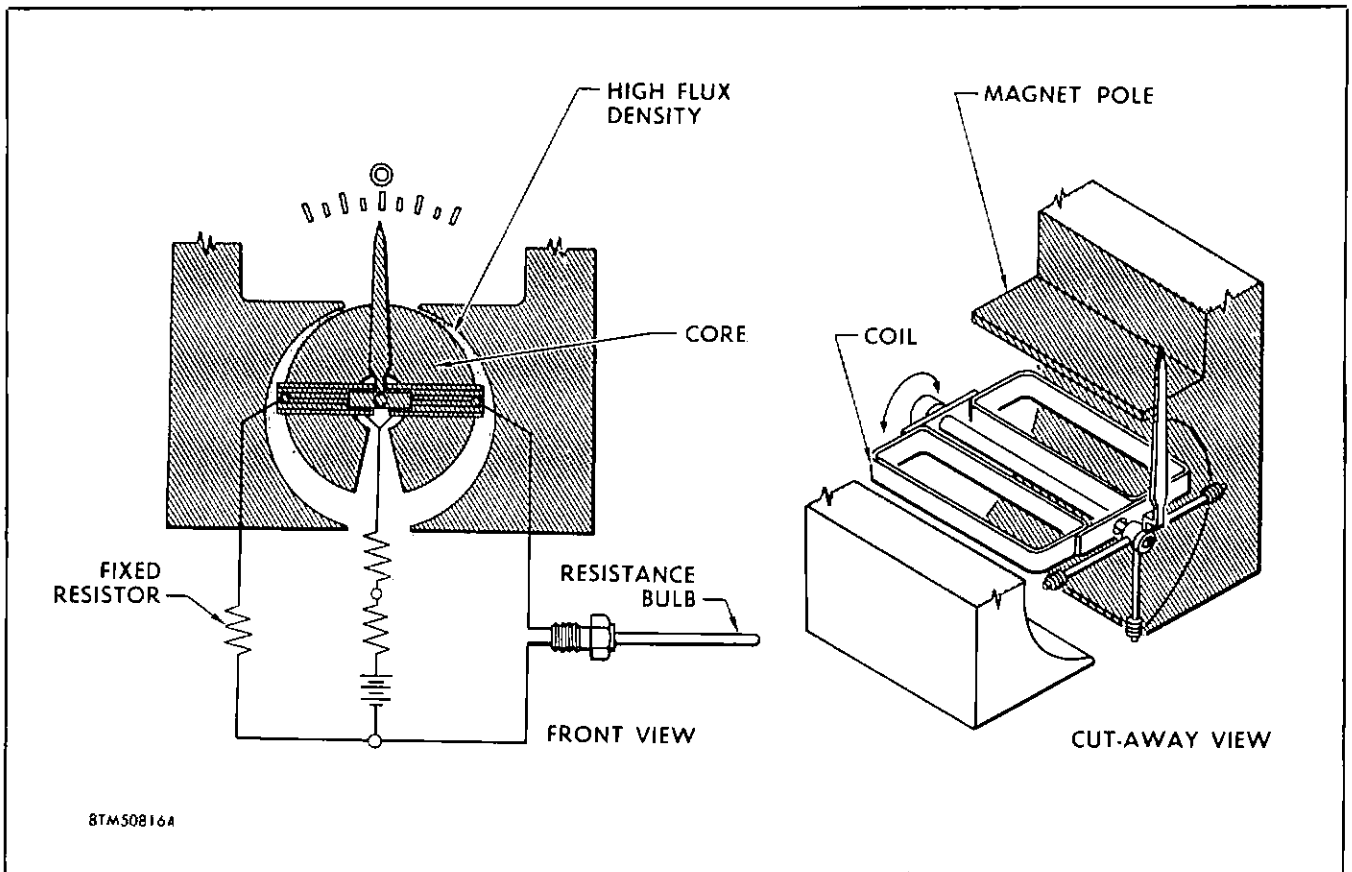

Figure 5-11. Operational Schematic Ratiometer Instrument

30 minutes unless he has oxygen equipment. Even when he is conscious, oxygen "starvation" will make him tired, confused, and forgetful. Insufficient oxygen also causes poor night vision, so oxygen breathing apparatus is required at night for even lower altitudes than during daylight hours.

We've established that oxygen equipment is necessary, but it isn't quite so clear why the pilot needs a pressure and flow indicator. It would seem as though he should know when he isn't getting sufficient oxygen. Frequently, this is not the case; oxygen starvation can occur entirely unnoticed. To avoid this danger, the pilot must know when his supply becomes inadequate. He must also know if something causes the flow of oxygen to his face mask to be interrupted. This is particularly important if the pressurization system fails while he is flying at high altitudes where, without a functioning oxygen system, he could lose consciousness in a matter of seconds. Thus, the oxygen pressure and flow indicator is a very important part of the oxygen system in the F-102A.

OPERATION OF THE OXYGEN PRESSURE AND FLOW INDICATOR.

The oxygen pressure and flow indicator illustration, (figure 5-14), shows the oxygen system pressure over a range of 0 to 2000 pounds. The word FULL appears at the 1800 pound position—this is the normal precharge pressure of the F-102A oxygen system. The slots in the lower half of the instrument dial are the flow "blinkers." Whenever there is flow to the face mask these slots alternately are black and white. Operation of these flow blinkers and the single pointer on the instrument are independent of any external adjustment.

Note that the measuring element of the pressure gage is a Bourdon-tube. You will recall that we discussed Bourdon-tube instruments quite thoroughly in Chapter 1, and to a lesser extent in some subsequent chapters. Since this one is no different than the others which you learned about, we won't go into further detail about it. Let's concentrate on the flow indicating part of the instrument.

The flow indication of the oxygen pressure and flow indicator results from the oscillation of a blinker plate. This plate mounts directly behind the instrument dial and pivots at its center. Alternate white and black segments on the lower part of the plate cause the blinking effect (through the slots in the dial) whenever the plate is rotated back and forth. To understand just what causes that oscillating movement

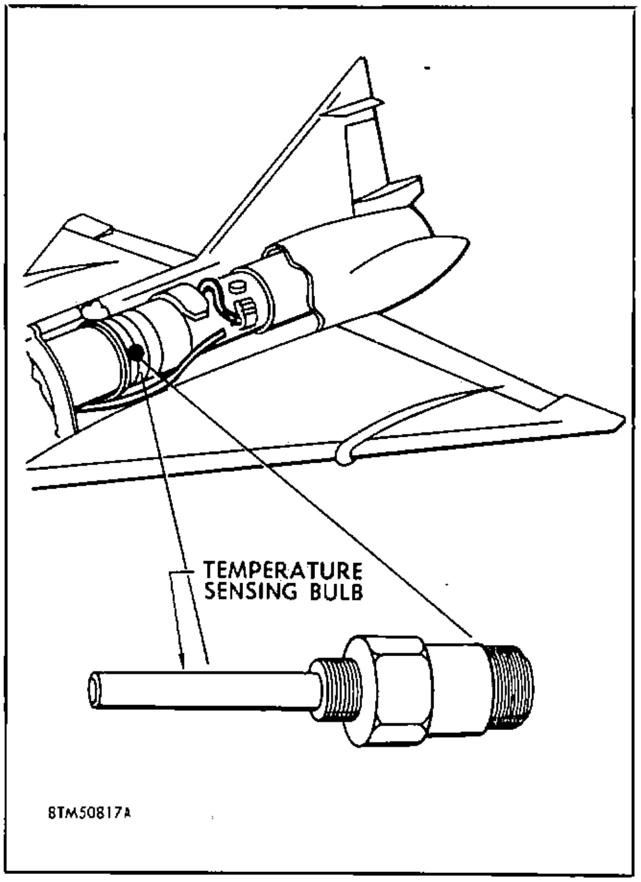

Figure 5-12. Temperature Sensing Bulb

we have to delve into the operation of the entire oxygen pressure regulator.

Referring again to figure 5-14, note that oxygen is admitted through an inlet valve and pressure reducer to the demand valve. This demand valve regulates the amount of oxygen made available to the pilot's mask. Another mechanism is responsible for opening and closing the demand valve at the proper time and at the proper amount. This mechanism, the aneroid assembly, is a system of sealed, spring-loaded bellows which expand and contract with changes of pressure on its outer surface. A semi-flexible diaphragm separates the aneroid assembly from the rest of the regulator case.

The same principles are involved in the operation of the oxygen regulator aneroid assembly that you learned about in the discussion of aneroid instruments in Chapter I. In other words, as the altitude of the airplane increases, the bellows assembly expands because air pressure on its outer surface decreases. In so doing, the demand valve is open sufficiently to bring in oxygen under pressure. When the oxygen pressure within the regulator reaches a certain level, the demand valve closes. However, each time the pilot inhales, the suction causes the diaphragm to move inward and the valve opens sufficiently to supply his demand. How far the valve opens, of course, depends on the pressure of the air in the cockpit.

Now we can return to the flow indicating mechanism. As you can see in the schematic (figure 5-13) when the pilot inhales, pressure is reduced in the regulator case permitting the bellows to open the demand valve. Pressurized oxygen released through the port of the demand valve then enters the aft chamber of the demand valve case and flows to both the pilot's mask and the flow indicator diaphragm. When the pressure reaches the flow indicator diaphragm, you can see that it forces the diaphragm forward. The lever attached to the diaphragm then rotates. Since the lever also connects to the blinker plate, the rotation of the lever causes a rotation of the plate. Thus, during inhalation, the blinker plate moves so that the white portion of the plate appears behind the openings in the dial. This tells the pilot that he is receiving oxygen.

When the pilot stops inhaling, pressure which had formerly gone to the mask, is now shut off and builds up in the regulator case. The trapped pressure then pushes against the diaphragm at the aft of the regulator, causing the demand valve to close. You can see

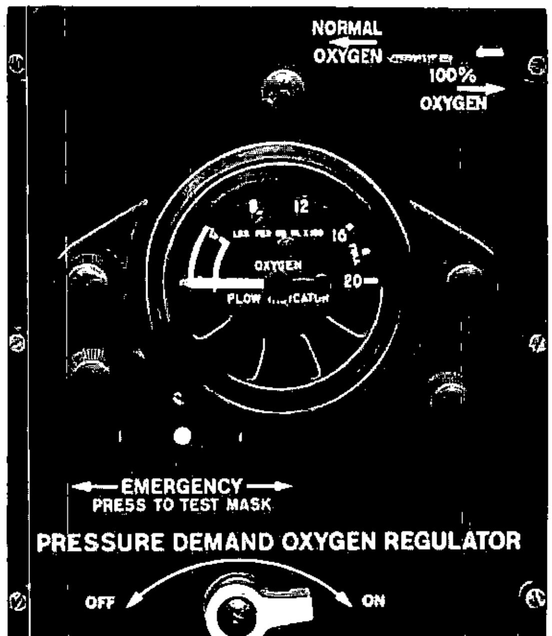

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Figure 5-13. Oxygen Pressure Regulator

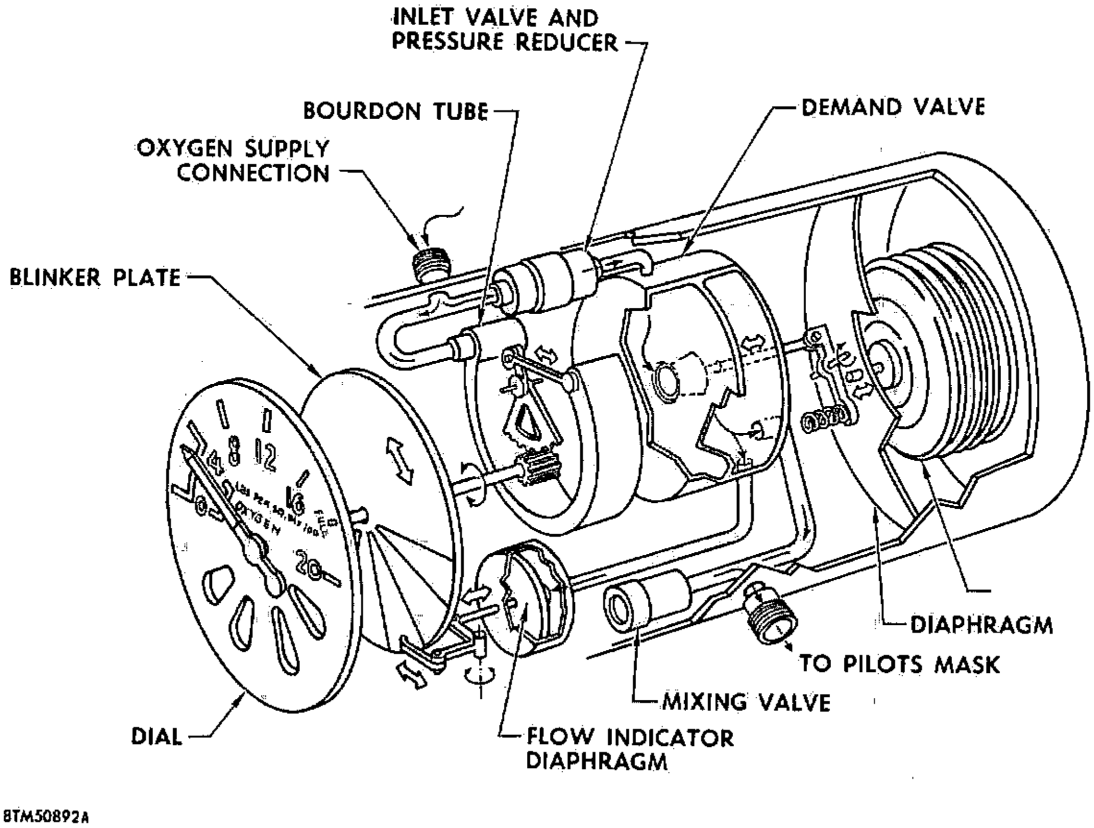

Figure 5-14. Oxygen Pressure and Flow Indicator Operational Schematic

that closing of the demand valve will cause the flow indicator diaphragm to move back to the aft portion of the indicator case. The movement of the diaphragm will move the lever and results in the corresponding movement of the plate. The black portion of the plate now appears behind the dial openings, informing the pilot that he is not receiving oxygen. In other words, the blinker plate flicks back and forth with each breath of the pilot.

MAINTENANCE OF OXYGEN PRESSURE AND FLOW INDICATOR.

We mentioned earlier in our discussion that the oxygen pressure and flow indicator is an integral part of the oxygen pressure regulator. As such, it cannot be removed from the airplane separately. If any problem develops with the instrument you must replace the entire regulator. However, the instrument is simple enough that it seldom gives any trouble—most problems which occur in the oxygen system can be traced either to other parts of the regulator or to leaks.

If you have to replace the oxygen regulator for any reason, there are certain precautions which you must observe. The most important of these is to keep oil and grease away from the oxygen system components. Whenever a petroleum-base material comes in contact with oxygen under pressure, there is danger of fire and explosion. Another thing to remember is that all open lines should be capped. Any foreign matter, including moisture, can foul up the regulator or lines, thus making it necessary for you to purge the entire system. To be on the safe side, it is always a good idea to consult your F-102A Maintenance Manual, T.O. 1F-102A-2-9, before making the replacement.

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Chapter VI

WARNING SYSTEMS

Contents *Page*

Like any other complex airplane, the F-102A has many warning systems. As the term implies, these systems are used to advise the pilot of dangerous or potentially dangerous conditions. They are, therefore, vital to the safety and efficiency of both the pilot and the airplane.

Current models of the F-102A series airplanes have 22 warning lights, all of which operate from the 28-volt d-c power source. Of these, 17 are part of the *master warning system.* The other five are for separate systems but have common dimming provisions. All but one of the warning lights (the landing gear light) contain a double socket with two lamps connected in parallel so that if one burns out the other will continue to give an indication. They are all listed in the table of contents of this chapter so there is no need to itemize them here.

Keep in mind that we are only concerned with the actual warning systems here. We will discuss the systems which they monitor only where necessary to understand the purpose and operation of the warning systems. For more complete coverage of the total systems, (such as fuel system, hydraulic system, electrical system and the like) refer to other training supplements in this series.

MASTER WARNING SYSTEM.

The master warning system is a combination of army warning circuits, which, for the sake of convenience the terminated at a common *warning indicator panel.* As you can see from figure 6-1, this panel consists of 16 individual slots or compartments. Each slot has an amber colored plastic grid which is lettered to indicate the system or equipment which it monitors. Warning lights are installed behind each slot to illuminate the appropriate lettering when an abnormal condition occurs. The warning indicator panel is mounted on the right-hand auxiliary instrument panel.

The other light in the master warning system is the *master warning light.* It is a rectangular amber plastic light located on the main instrument panel directly above the engine instruments, as shown in figure 6-1. The master warning light illuminates whenever any of the individual lights on the warning indicator panel light up. Because of its location and brilliance, the pilot's attention is immediately attracted. He then checks his warning indicator panel to see which individual system is malfunctioning. By momentarily placing the *warning test and reset switch* in the RESET position, the pilot extinguishes and rearms the master warning light.

If any other individual warning light on the warning indicator panel illuminates, the master warning light again illuminates and the pilot repeats the procedure. Placing the master warning test and reset switch in the RESET position *does not* extinguish the lights on the warning indication panel—they remain lighted until the malfunctions of the particular systems are corrected. For more detail of the complete system and a block diagram, refer to the Electrical Systems Supplement.

All circuits in the master warning system tie into the *master warning box,* mounted on the right-hand side of the airplane just above the cockpit floor and

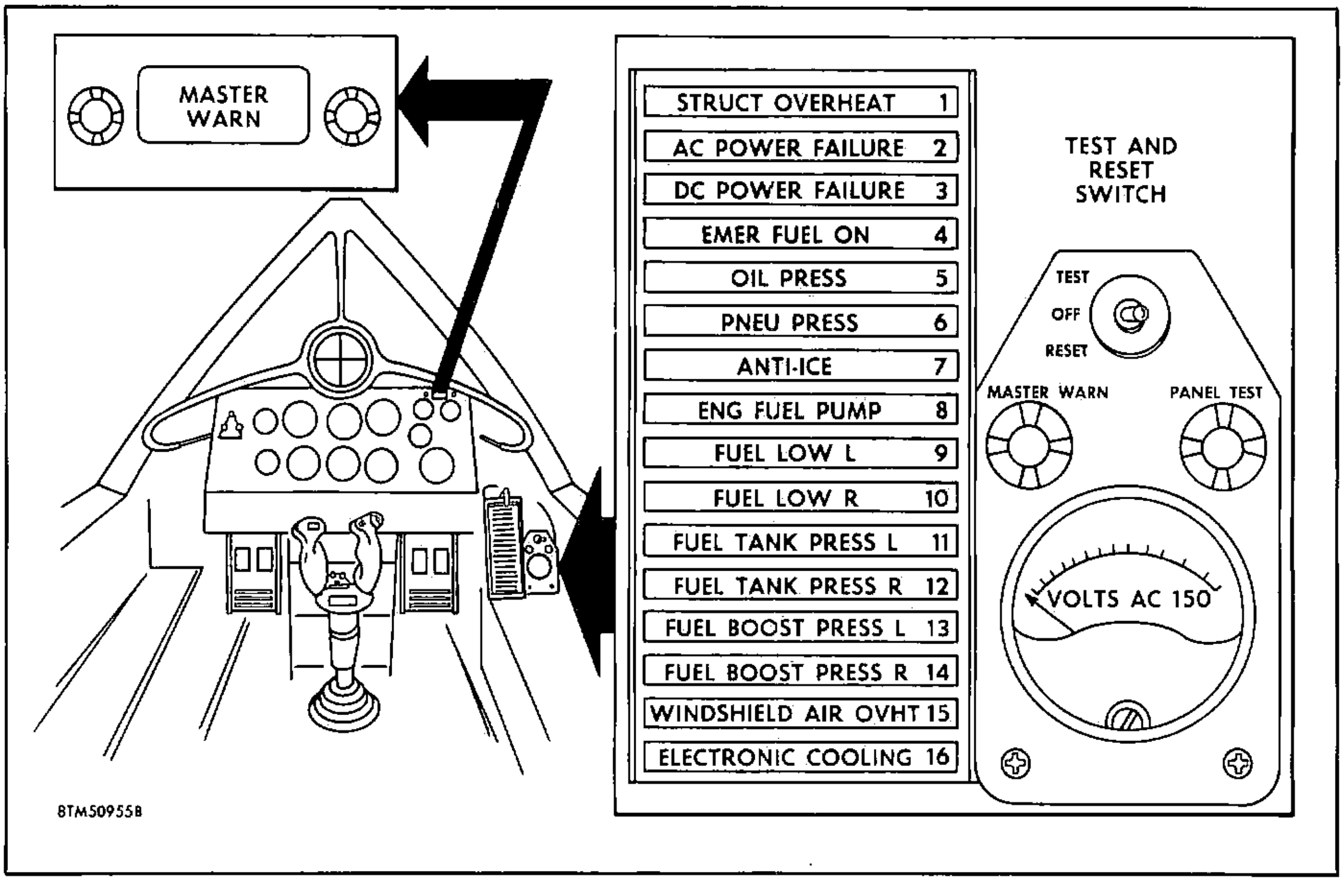

Figure 6-1. Master Warning Indicator Panel

forward of the instrument panel. Dimming relays and resistors in this box provide dimming control of the master warning system lights whenever the flight instrument lights are on and the thunderstorm lights are *off*. When the dimming relays are energized, current must flow through the resistors to get to the warning lights.

These relays are energized by 28-volt d-c power through the flight instrument rheostat in the ON position in series with the thunderstorm light switch in the OFF position. Thus, the brightness of the warning lights is controlled according to light conditions in the cockpit. Relays in the master warning box are also responsible for the master warning light coming on when any of the circuits to the warning indicator panel is completed. Now let's consider the various warning circuits which are part of the master warning system. We will discuss them in the order in which they appear on the panel.

STRUCTURE OVERHEAT WARNING SYSTEM.

The structure overheat warning light is the first light on the warning indicator panel and is used only on early model F-102A airplanes. This light is part of a system which notifies the F-102A pilot of excessive heat conditions in the fuselage structure at station 614.00—the area just ahead of the tail cone fairings.

The detecting device for the structural overheat system is a mercury-operated, temperature-sensing switch that is attached directly to the fuselage structure. Only that portion of the switch which touches the fuselage structure is sensitive to temperature changes, so it is limited to sensing just the structural temperatures. The switch is pre-set to actuate whenever the structural temperature rises above 245° F.

Both the master warning light and the structure overheat warning light come on when the temperature sensing switch is actuated. These lights indicate to the pilot that power should be reduced until the temperature decreases. If they come on during a ground runup, you should shut the engine down immediately and investigate the cause.

Figure 6-2 shows a simple electrical schematic of the structure overheat warning system. Note that current is taken from the 28-volt d-c essential bus through a push-pull type circuit breaker. This circuit breaker is located on the main wheel well circuit breaker panel and serves to protect the circuit from current overloads. Following the current from the essential bus

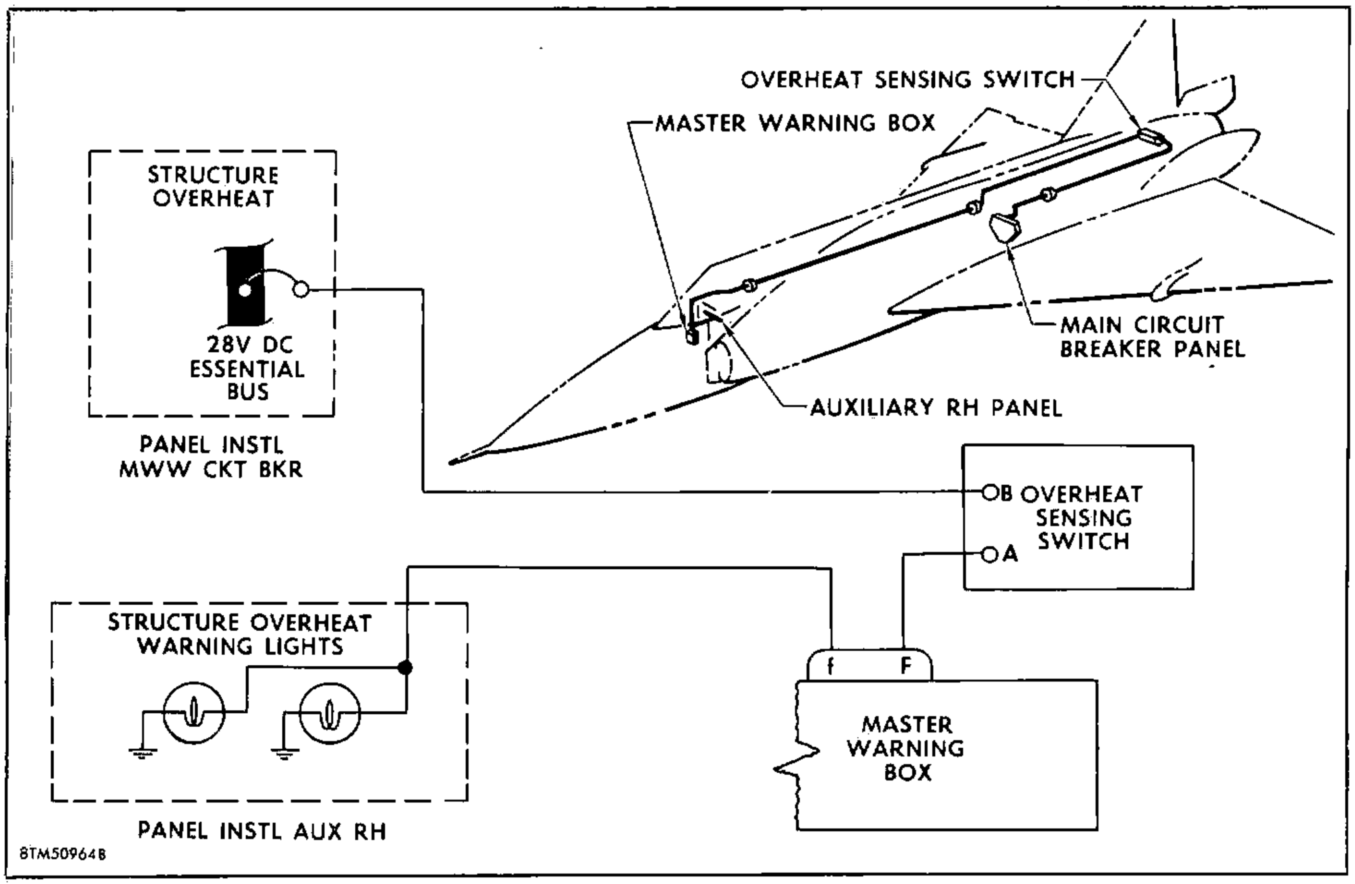

Figure 6-2. Structural Overheat Warning System Schematic

and the circuit breaker, you can see that it flows directly to the overheat sensing switch. Actuating the switch provides a current path to the master warning box, and from there to the master warning light and the individual system warning light.

A-C POWER FAILURE WARNING SYSTEM.

Second in the row of warning indicator lights is the A-C POWER FAIL light. Obviously, this is the indicator for the a-c power failure warning system. Its purpose is to notify the F-102A pilot if the main a-c power source fails, enabling him to take the necessary corrective action or revert to emergency operation. A-C power is normally furnished by a 30 KVA, 120/208-volt, 3-phase, 400-cycle generator. This generator is driven by the Sundstrand constant speed drive unit and supplies 115-volt 400-cycle regulated power to the a-c essential and non-essential buses. A 26-volt, 400-cycle bus is also energized, through a step-down transformer, from phase B of the essential bus.

Operation of the generator is controlled by the a-c generator switch and the a-c bus switch, both of which are on the electrical power control panel. When the airplane engine is first started, the a-c generator switch is momentarily placed in RESET, and then to ON. The a-c bus switch is placed on NORM, and, of course, the master switch is left on NORMAL. Now suppose that the A-C POWER FAIL light comes on during a flight. First the pilot will attempt to remedy the situation by placing the a-c generator switch in the RESET position, and then back to ON. This operates the exciter to "flash" the generator. If the warning light does not extinguish, the pilot will then move the a-c bus switch from NORM to EMER. This puts the emergency generator into operation. For more details and for a complete block diagram, refer to the Electrical Systems Supplement.

The a-c emergency generator is a 1 KVA, 120/208-volt, 3-phase, 400-cycle unit. It is located at the forward outboard side of the right main wheel well. A hydraulic motor powered by the secondary hydraulic system, is used to drive the generator. This motor is controlled by a solenoid operated hydraulic shut-off valve which receives power from the d-c essential bus. Whenever the a-c bus switch is in the emergency position and the generator switch is ON, d-c current energizes the valve solenoid. Operation of the emergency generator *does not* cause the A-C POWER FAIL light to extinguish. Let's study the circuit schematic, (see figure 6-3), to see what does control this light.

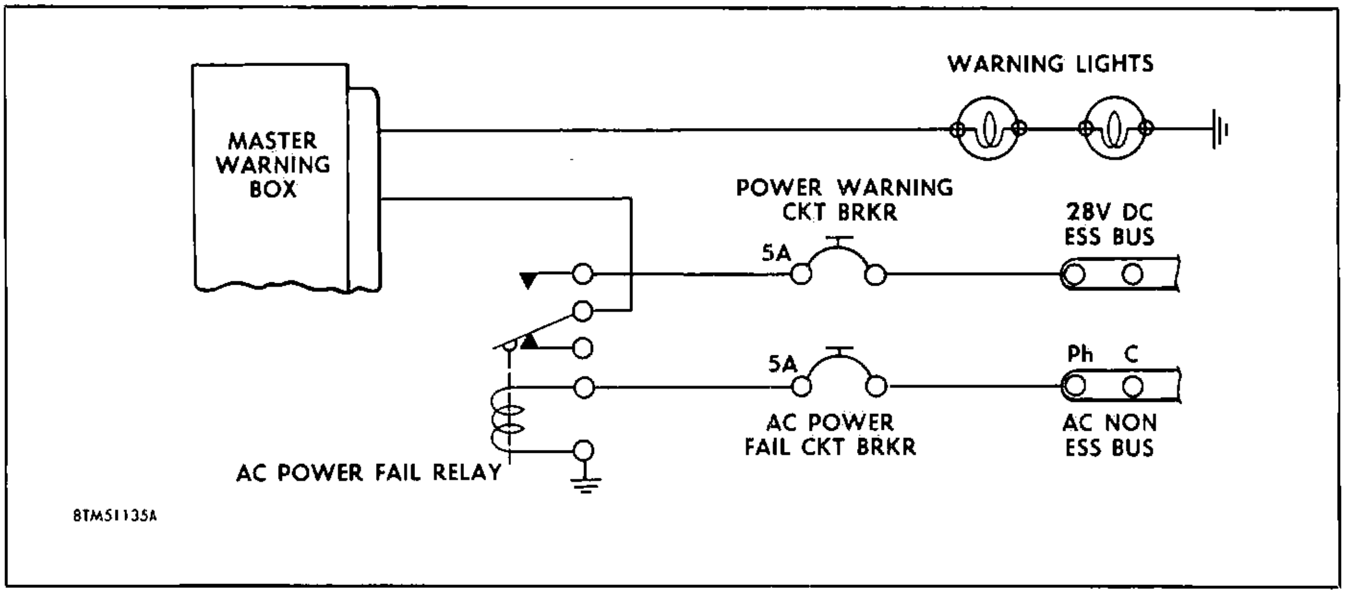

Figure 6-3. A-C Power Failure Warning System Schematic

During normal operation, both the essential and non-essential buses are energized. Under these conditions, current flows from the non-essential bus through the a-c power fail circuit breaker, and grounds through the a-c power fail relay. This relay is mounted in the cockpit, adjacent to the master warning box. Since the circuit is complete, the relay is energized and the relay switch is held open. The warning circuit is therefore not complete. See the Electrical Systems Supplement, for a complete diagram.

Now, if the main a-c generator fails there is no power to the non-essential bus—remember, the emergency generator is only connected to the essential bus. The a-c power fail relay is then deenergized, permitting the switch to move to the upper contact. Thus, the warning circuit is completed through the master warning box and the lamps of the warning light. The light will come on, warning the pilot that he must switch over to the emergency generator. It will continue to glow, reminding him of his undesirable predicament.

D-C POWER FAILURE WARNING SYSTEM.

There are two sources of d-c power in the F-102A airplanes. One source is the 24-volt battery located in the nose wheel well; the other is the d-c generator mounted on the forward end of the Sundstrand drive unit gear box in the engine accessory compartment. (See Electrical Systems Supplement.) The battery provides power to the d-c essential bus—through the battery relay—when the battery switch is in the ON position and the master switch is in the NORMAL position. Both of these switches are situated on the power control panel on the right-hand console. When the engine is operating, the d-c generator provides regulated power to both the essential and non-essential buses if the master switch is in NORMAL and the generator switch—also on the power control panel—is ON.

In the event of generator failure, all systems connected to the non-essential bus are without power. Systems which draw power from the essential bus have power for a very limited time only, since the battery will soon discharge. You can readily understand why the pilot must be warned immediately of this condition. When D-C POWER FAILURE shows on the warning indicator panel during a flight, he must find a place to land as quickly as possible.

The d-c power failure warning system, like all the other warning systems, relies on the d-c power for its operation. This may sound strange, but remember, all d-c power is not likely to be lost at the same time. In figure 6-4, note that the warning system connects to both the essential and the non-essential buses.

The non-essential bus receives its power from the generator only, but the essential bus connects to both battery and generator. Thus, there is battery power available from the essential bus after the non-essential bus goes "dead."

The warning system circuit is shown deenergized—that is, in the normal position. Power is available from the d-c non-essential bus, so the d-c warning relay located just above the master warning box is energized. This holds the relay switch in the open position and prevents current flow from the essential bus to the warning circuit. Should the generator fail, the

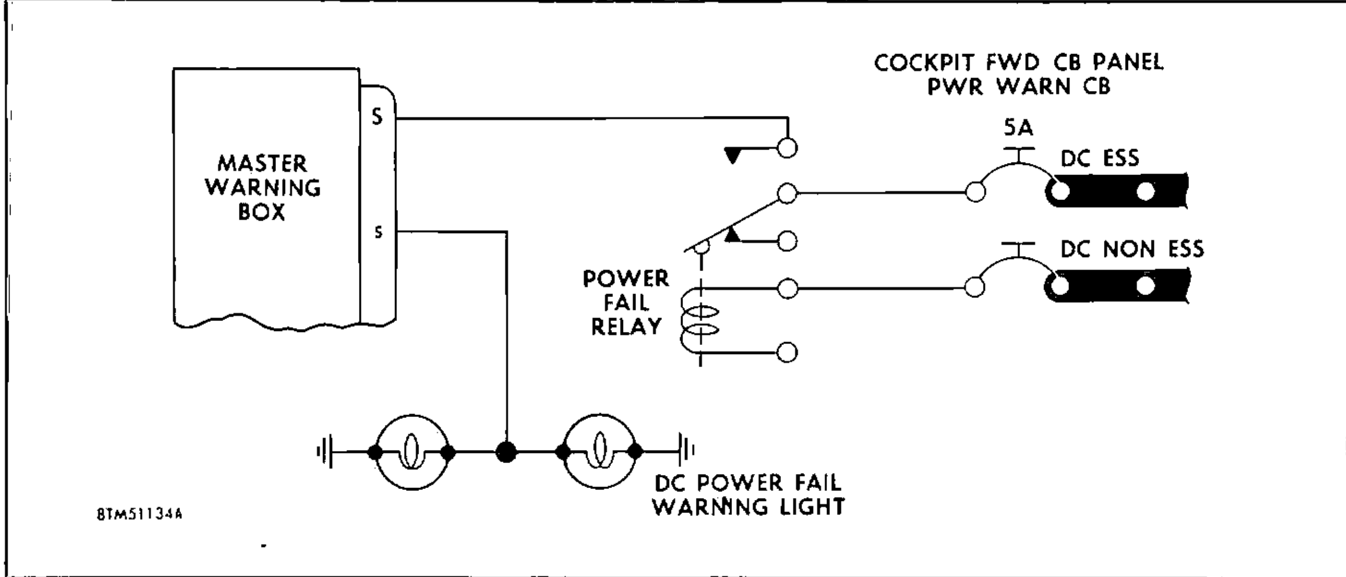

Figure 6-4. D-C Power Failure Warning System Schematic

non-essential bus is without power, so the d-c warning relay deenergized. The switch closes and the current flows from the essential bus through the master warning box and warning light to ground. The light will illuminate and continue to glow until generator power again reaches the non-essential bus, or until the battery power is exhausted.

EMERGENCY FUEL CONTROL WARNING SYSTEM.

The number four compartment on the warning indicator panel reads EMER FUEL ON. This compartment houses a warning light that glows constantly while the fuel control switch is in the EMERGENCY position. When this switch, located on the throttle quadrant, is in the NORMAL position the light should never be on. A brief review of the fuel control systems will clarify the need for this warning light. A complete description of the fuel warning circuits and schematics can be found in Chapter IV of the Fuel Systems Supplement.

During normal operations, fuel is metered to the J57 engine by an automatic fuel control. This device functions somewhat like a carburetor on an automobile, but must do more things. When you start a car on a cold day, the automatic choke enriches the fuel-air mixture to suit the needs of the engine. But, if you drive to the top of a high mountain, the mixture becomes too rich and the engine "floods" easily. To correct that problem you have to make manual adjustments to the carburetor. The automatic fuel control on a jet engine takes care of such problems automatically. It schedules fuel flow—for any given power lever setting—to avoid rich mixture "blowout" during acceleration, lean mixture "die-out" during deceleration, and to allow for varying conditions of altitude and of engine pressure and temperature.

In the event of automatic fuel control failure, the F-102A pilot can switch over to the emergency fuel control system. Under these conditions, fuel scheduling becomes a direct function of power lever movement, with fuel scheduling accomplished through an emergency throttling valve. Limited altitude compensation is achieved by this system, but there are no provisions for changing fuel flow during acceleration, deceleration, or for variations of temperature and burner pressure. Thus, the pilot must observe engine RPM, tailpipe temperatures, and pressures very closely during emergency operation. The warning light keeps him reminded of that fact.

Now refer to figure 6-5. Note that the fuel control switch is in the NORMAL position. In this position, the automatic fuel system is in use and no current flows through the warning circuits. Suppose now that the pilot switches over to the EMERGENCY position. The circuit from the 28-volt d-c essential bus is completed through the fuel control relay and fuel control switch to ground. The relay energizes, closing contacts A1 and A2 of the relay. This provides a current path through terminals K and J of the Pratt and Whitney relay box to the master warning box. The lamps are permanently grounded so the circuit is complete and the warning light glows. The light will remain on unless the fuel control switch is returned to the NORMAL position.

ENGINE OIL PRESSURE LOW WARNING SYSTEM.

The fifth item on the warning indicator panel is placarded OIL PRESS. This is the warning light for a system which notifies the pilot when his engine oil pressure is below safe operating limits. You have undoubtedly seen similar devices installed in some makes of automobiles.

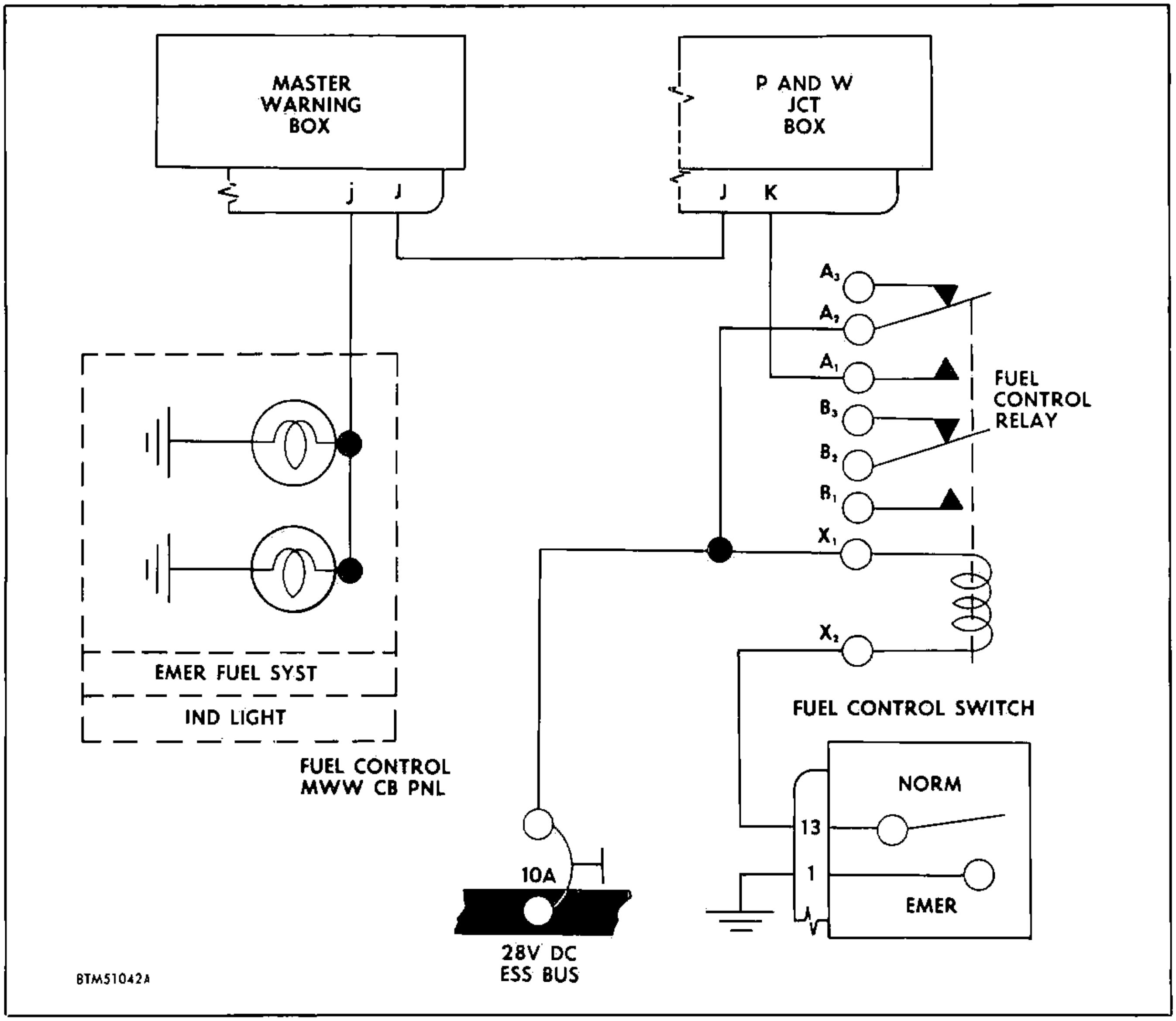

Figure 6-5. Emergency Fuel Control Warning System Schematic

The sensing element of the oil pressure low warning system is a simple pressure switch. It is mounted on the left side of the engine immediately aft of the oil tank. One fitting on the switch brings in oil pressure from the discharge side of the oil pump; another fitting is vented to atmosphere. Thus, the switch senses differential pressure between pump pressure and oil system breather pressure. For a complete diagram of this system, refer to Power Plant Installation.

Figure 6-6 shows a schematic diagram of the oil pressure low warning system. Note that the circuit is shown deenergized. It will remain that way as long as there is sufficient oil pressure into the switch—over 36 psi, to be exact. When the oil pressure falls below that pressure, the switch closes and current flows through to the master warning box. From there, current flows through the lamps to ground.

If the differential pressure at the oil pressure switch builds up to 40 psi, the switch contacts will open again. Thus, the warning system circuit will be de-energized and the light extinguished.

PNEUMATIC PRESSURE LOW WARNING SYSTEM.

The F-102A airplane uses air pressure from the high pressure pneumatic system for emergency landing gear extension, drag chute deployment, canopy ejection and cinch-down, armament displacement, and a variety of other purposes. As you learned in our discussion of the pneumatic pressure gage, Chapter IV, this system is pre-charged on the ground to 3000 psi. Once the system pressure is depleted, there's no way to replenish the supply in flight. For that reason, a warning system cautions the pilot if pressure in the high pressure pneumatic system becomes dangerously low. This warning system terminates at the warning

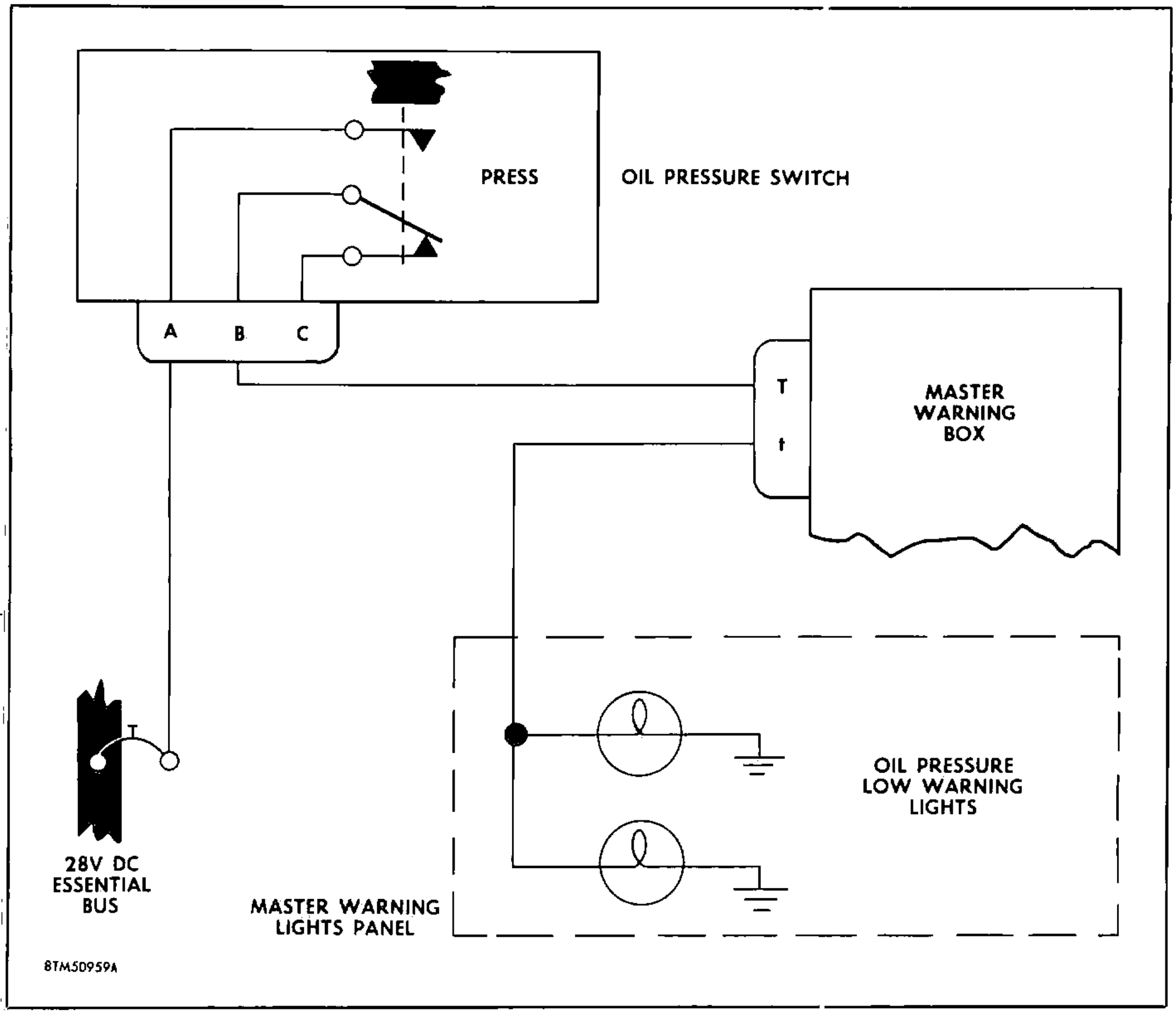

Figure 6-6. Engine Oil Pressure Low Warning System Schematic

indicator panel. Its light is placarded PNEU PRESS and is the sixth light on the panel.

In addition to the warning light, the pneumatic pressure low warning system consists of a pressure switch located in the pressure manifold line and connecting circuitry. Figure 6-7 shows the system in schematic form. For a schematic of the complete high pressure pneumatic system, refer to High Pressure Pneumatic System Supplement. The pneumatic pressure low warning light illuminates when the system pressure becomes insufficient for a combat cycle. A combat cycle consists of opening the armament bay doors, extending the armament racks, retracting the racks, and closing the doors. This cycle requires a pressure of about 1500 psi. Below that point, the pressure switch closes, permitting current flow from the power source through the master warning box and lamps to ground. The circuit is very simple and identical to others which we've already discussed, so there's no need for further comment on its operation.

The pressure switch in this warning system will open when the system pressure again builds up to approximately 1700 psi. Therefore, the warning light will remain on—while power is available at the 28-volt d-c essential bus—until the air flasks are refilled. If the pilot reports that the light goes on and off in flight, you should suspect malfunctioning of the pressure switch or the line leading to it.

THE ANTI-ICE WARNING SYSTEM.

Icing is a serious problem on jet airplanes, and is very difficult to contend with. To cope with the icing problem, the F-102A uses three methods of ice prevention and removal: hot air, electricity, and fluid

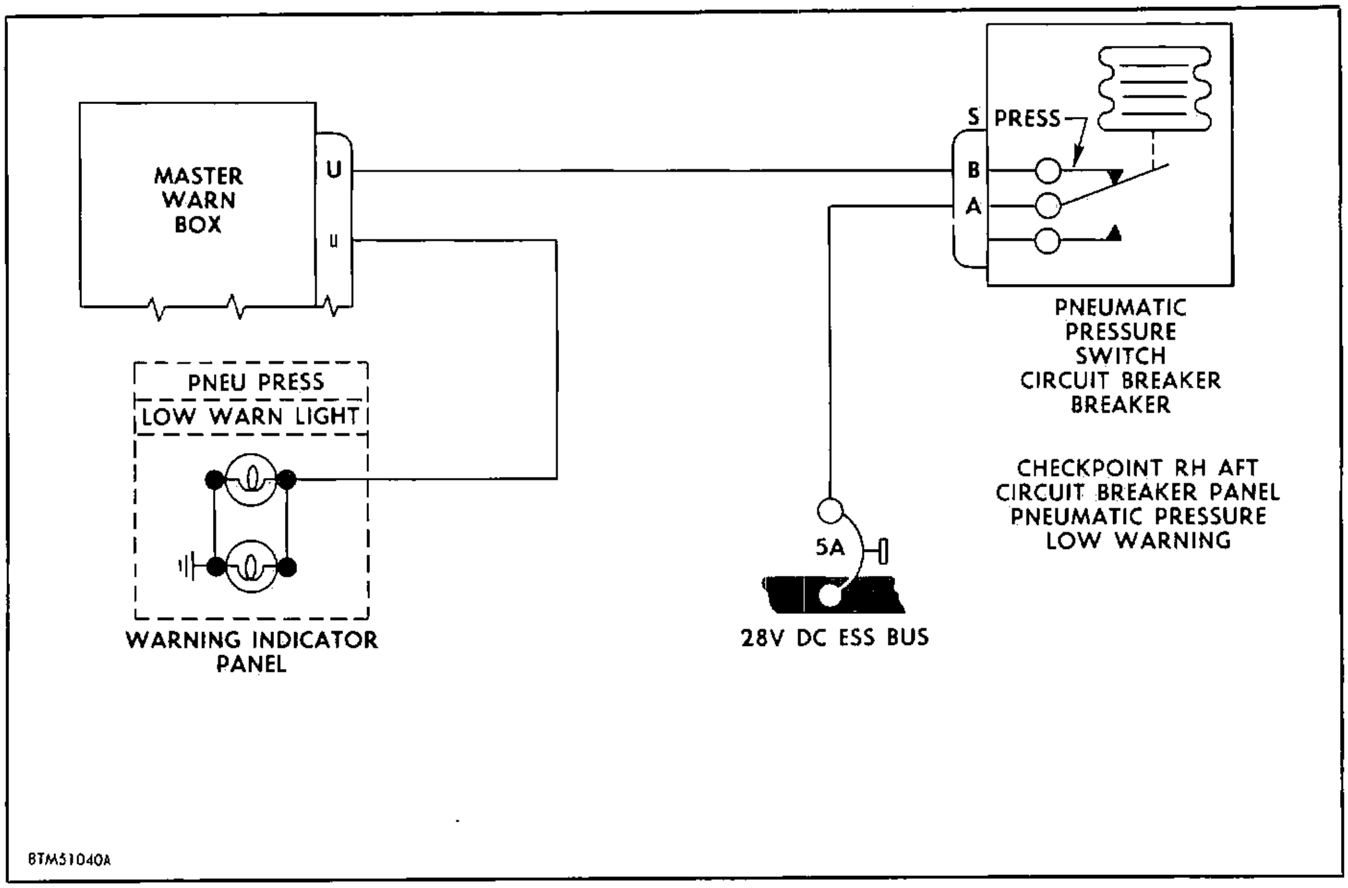

Figure 6-7. Pneumatic Pressure Low Warning System Schematic

(glycol). A number of different components and systems require anti-icing. In four of the systems which would be most seriously affected by ice, anti-icing is normally accomplished automatically. These systems are the power plant, the inlet duct, the "Q"-intake, and the radome. An ice detector probe in the engine intake duct triggers the automatic system. In the event of some malfunction of the system, the pilot must be warned immediately so he can switch over to manual control. Therefore, a warning system is incorporated in the automatic control system. This warning system *does not* indicate that ice is forming at any particular moment—it merely shows that the automatic control system is not operating the way it should. The seventh light on the warning indicator panel illuminates when this warning system is energized. It reads ANTI-ICE.

Since the warning system is an integral part of the automatic control system we will have to discuss them together; otherwise you won't know much about either system. Figure 6-8 shows the major components of the system. These items are adequately identified in the illustration. The detector detects the presence of ice on its probe when holes in the probe become constricted, limiting the intake air pressure to a diaphragm assembly. The interpreter receives a signal from the detector and energizes the control relay. The control relay actuates the anti-ice systems for as long as the interpreter keeps it energized. The ignition power relay affects the systems only during engine starting—we'll discuss what it does a little later.

Now let's "follow the electrons" through the schematic circuit (see figure 6-9). Keep in mind that all switches and relays in the diagram are shown in the normal—*no ice*—condition, and with no malfunctions existing.

All power for the ice detector and interpreter comes from the 28-volt d-c essential bus through the 10-amp anti-ice power circuit breaker. The current flows through contacts 2 and 3 of the ignition power relay and on to connections A1 and B1. From there it flows through contacts A and B on the ice detector switch—in the *no-ice* condition—and out B2 to A6. When there is no ice, therefore, both A1 and A6 are connected to power.

Now, notice that relay R1 is connected to A6 and is also grounded. Therefore, this relay is energized in the *no-ice* condition. When relay R1 is energized its switch is closed so there is another "hot" line across A1 and A6. As you can see, relays R3 and R4 are connected to B3 and contact C on the detector and

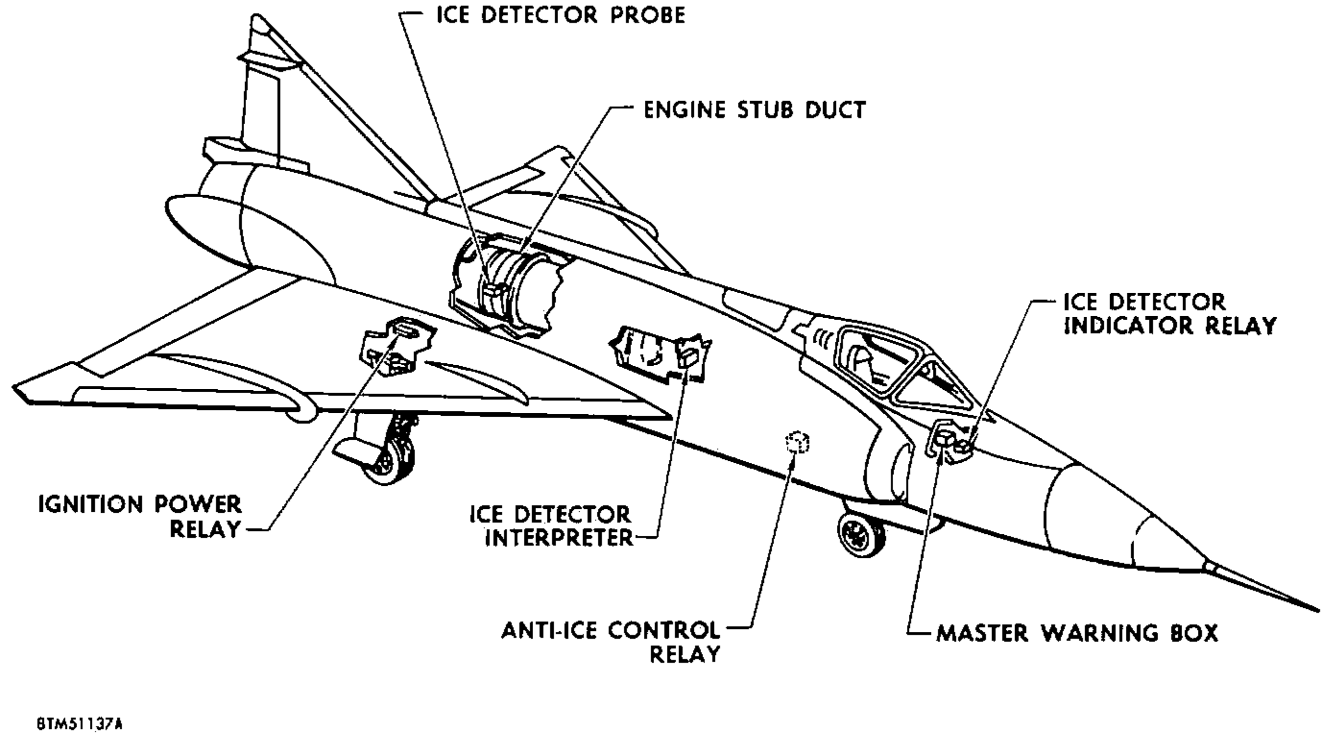

Figure 6-8. Anti-Ice Warning System Components

are not energized in the *no-ice* condition. Since R2 is energized only when R4 is energized it also is de-energized when there is no ice.

Let's see what happens when the detector probe ices up. First of all, the detector causes its switch to break contact with A and connects B to C. This opens one circuit between A1 and A6. But remember, A6 is still connected to A1 through the switch in relay R1. Therefore, B3 is now "hot," since it connects to A6 through C, B, and B2. Relays R3 and R4 are now energized. Current flows from point N, through the R3 relay switch to B4 and the probe heater. Since relay R4 is energized, current flows through its switch to energize R2 and the heater at R1. The R2 relay switch thus furnishes a ground—through connection A5—for the anti-ice control relay. When the control relay switches close, the four anti-ice systems are put into operation by current through the anti-ice control circuit breaker.

Now we'll go back to the probe heater which we left in the energized condition. Don't confuse this probe heater and the detector case heater. The case heater simply keeps the detector box warm enough for efficient operation. It doesn't affect the probe temperature. When the probe heater melts the ice from the probe, the detector switch moves back to the *no-ice* position. Current is cut off from B3, relays R2, R3, and R4 de-energize, and the probe heater and R1 relay heater are therefore de-energized. But, notice that there is a time delay mechanism at relay R2. This device keeps the R2 relay switch closed across contacts 2 and 3 for 60 seconds after the relay de-energizes. In so doing, anti-ice systems are kept operating long enough to do their jobs even if the heater de-ices the probe in just a few seconds.

When the anti-ice detection and control system is operating normally, the probe heater melts the ice at the probe and returns the switch to the *no-ice* position within 17 to 20 seconds. If it doesn't, the detector is malfunctioning. To prevent the probe from overheating and to de-energixe the relays durinig a malfunction is the job of the thermoswitch at relay R1. When the relay and probe heaters have been on about 17 to 20 seconds, the thermoswitch becomes hot enough to break the circuit at the R1 relay. The R1 relay switch opens, breaking the circuit from A1 to A6. Then there is no current to B2 or across the detector switch to B3, so relays R2, R3, and R4 de-energize. This also removes current from the heaters. Sixty seconds later the R2 relay switch opens, the anti-ice control relay de-energizes, and

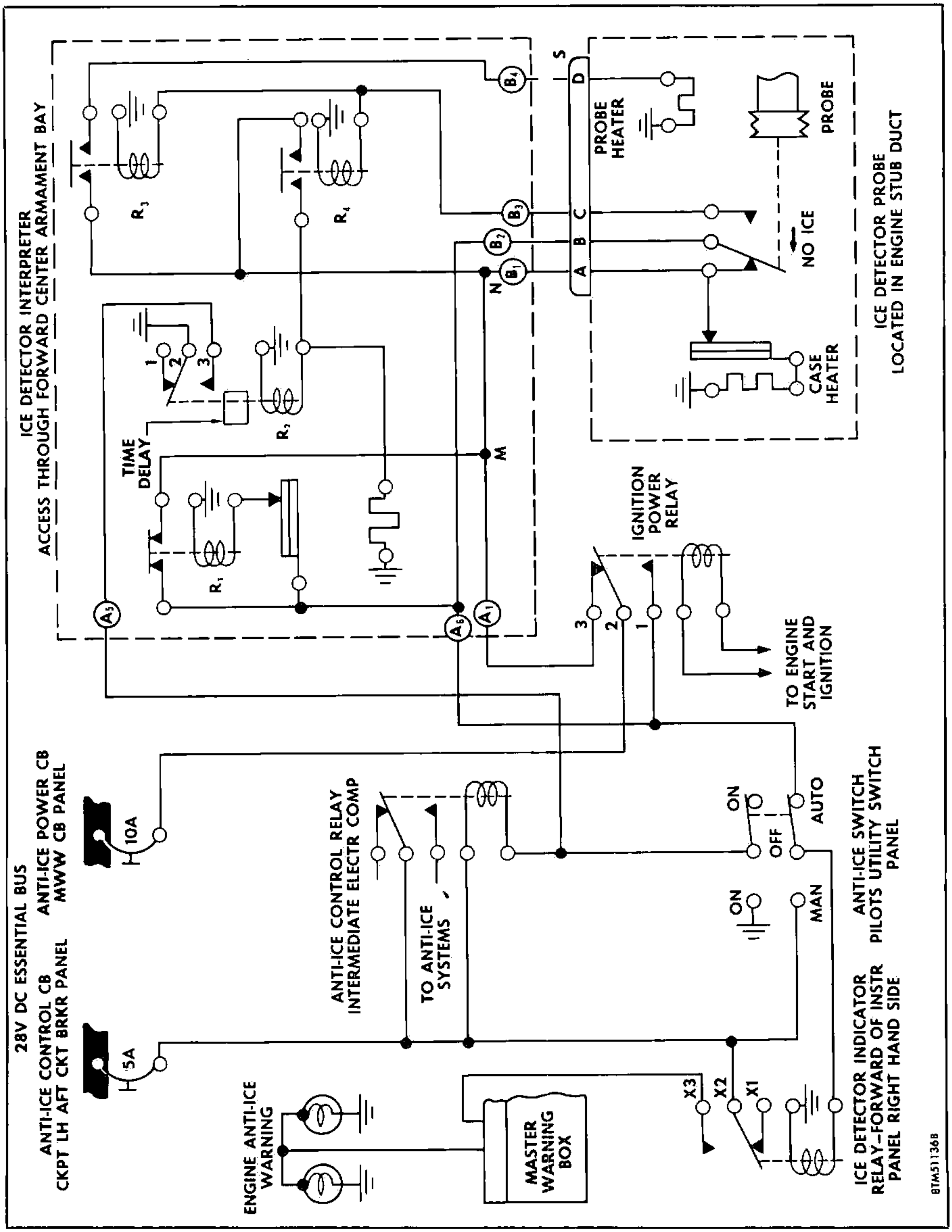

Figure 6-9. Anti-Ice Warning System Schematic

all power is shut off to the four anti-ice systems. Thus, the effectiveness of the automatic control system is destroyed.

Now we can concentrate on the warning system. All power to A6 has been lost during the malfunction because current can't flow from A1 through either the detector switch—it's in the ice position—or the R1 relay from point M. Since A6 is "dead," the ice detector indicator relay is deenergized. Contact X2 leaves X1 and moves to X3. This completes the circuit from the anti-ice control circuit breaker through the master warning box to the light, which is permanently grounded. The light illuminates, warning the pilot that the automatic anti-ice system has failed. He will then place the anti-ice switch in the manual (MAN ON) position.

You've learned why the warning light comes on when there is no power at A6. You can see, then, that the light will glow if the anti-ice switch is in the AUTO position with power turned on and the engine is not operating. When the engine is not operating, the pressure actuated detector switch is always in the *ice* position. R1 relay will not have been actuated, so power can not flow from M around to A6. The warning light also illuminates when power is on and the anti-ice switch is OFF. When the engine is being started with the anti-ice switch in AUTO, or when the anti-ice switch is in MAN ON, the warning light will always be off. During engine starting, the ignition power relay energizes, pulling the number 2 switch contact over to contact 1. Current then flows directly to A6 and is unaffected by the interpreter and detector. With the switch in MAN ON, current from the anti-ice control circuit breaker flows through the anti-ice switch to energize the ice detector indicator relay. In either of these cases, the ice detector relay switch contact X2 is held against X1 so current cannot reach the master warning box.

ENGINE FUEL PUMP FAILURE WARNING SYSTEM.

The purpose of the engine fuel pump warning system is to advise the F-102A pilot when the main engine stage of the fuel pump is malfunctioning. This system terminates at the eighth slot or compartment of the warning indicator panel. Its warning light reads ENG FUEL PUMP when the light is illuminated.

In addition to the indicator light, the engine fuel pump warning system includes a pressure switch. You will find this switch on the lower left side of the oil pump and accessory drive housing of the engine. A pressure sensing line connects the switch to the pressure side of the fuel pump engine stage.

Figure 6-10 shows a schematic diagram of the engine fuel pump warning system. The circuit is shown deenergized. If the pressure from the pump drops below 105 psi, the pressure switch closes the circuit between terminals A and B. This allows current to flow through the master warning relay box. Both lamps of the warning light are permanently grounded so that the light glows. A pressure of approximately 125 psi is required to open the pressure switch and extinguish the light. For a diagram of the complete fuel system, refer to the Fuel System Supplement.

Although not shown on this and some of our other schematics, you should remember that the master warning light is also tied into the master warning box. As stated earlier in Chapter V, the master warning light circuit is energized when any of the circuits to the warning indicator panel are energized. It will remain energized until the reset switch is depressed or until all warning indicator panel circuits are deenergized. Refer to the Electrical Systems Supplement for more details.

FUEL QUANTITY LOW-LEVEL WARNING SYSTEM.

The ninth and tenth items on the warning indicator panel are fuel low-level warning lights. These lights, placarded: FUEL, LOW L, and FUEL LOW R are part of a system which warns the pilot when his fuel supply drops to approximately 88 gallons in either or both number 3 tanks. This is enough fuel for a few minutes of normal cruising and one go-around with afterburner.

Each of the two circuits in the fuel low-level warning system is energized through a float switch in the corresponding tank. In figure 6-11 you can see these switches and the other components of the system. Note that the circuits for the left-hand and the right-hand warning lights are not identical; for that reason we will discuss them separately.

Current from the 28-volt d-c essential bus flows through a circuit breaker and directly to the RH fuel low switch. This switch, being the float type, is actuated by the changing fuel level in the No. 3 RH tank. It closes when the fuel supply is reduced to 88 gallons or less, thus providing a path for the current to terminal K of the relay box. The current leaves the relay box at terminal K and travels to the warning light for the RH tank. Since the bulbs in the warning light are grounded, the circuit is complete and the light comes on.

Now let's trace the circuit for the LH fuel low-level warning light. As you can see, the power source is the same as for the RH circuit. However, the current path to the LH fuel low switch is through the fuel low-level warning relay. Note that this switch is grounded instead of being connected to the master warning relay box. When the fuel supply in the No. 3

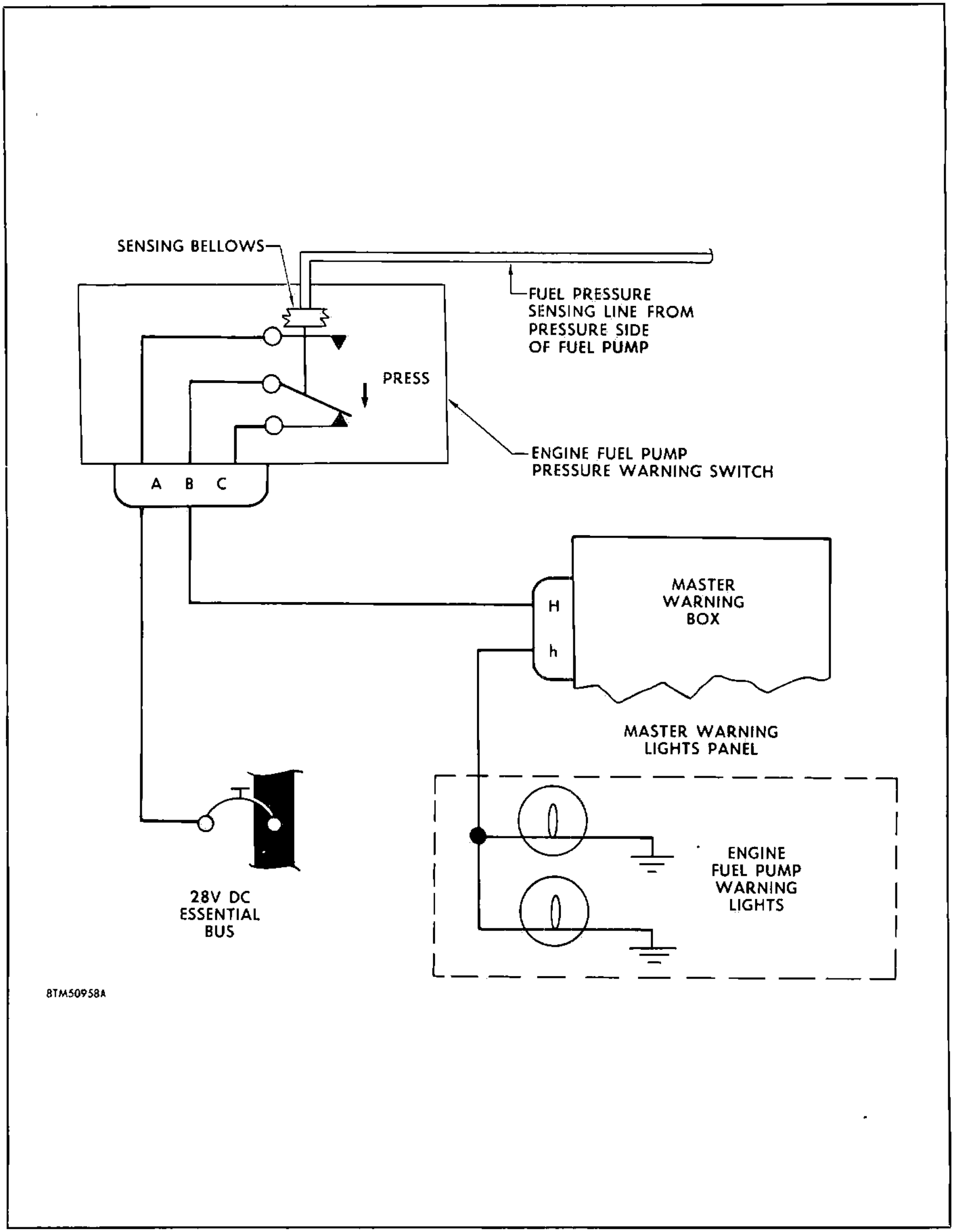

Figure 6-10. Engine Fuel Pump Warning System Schematic

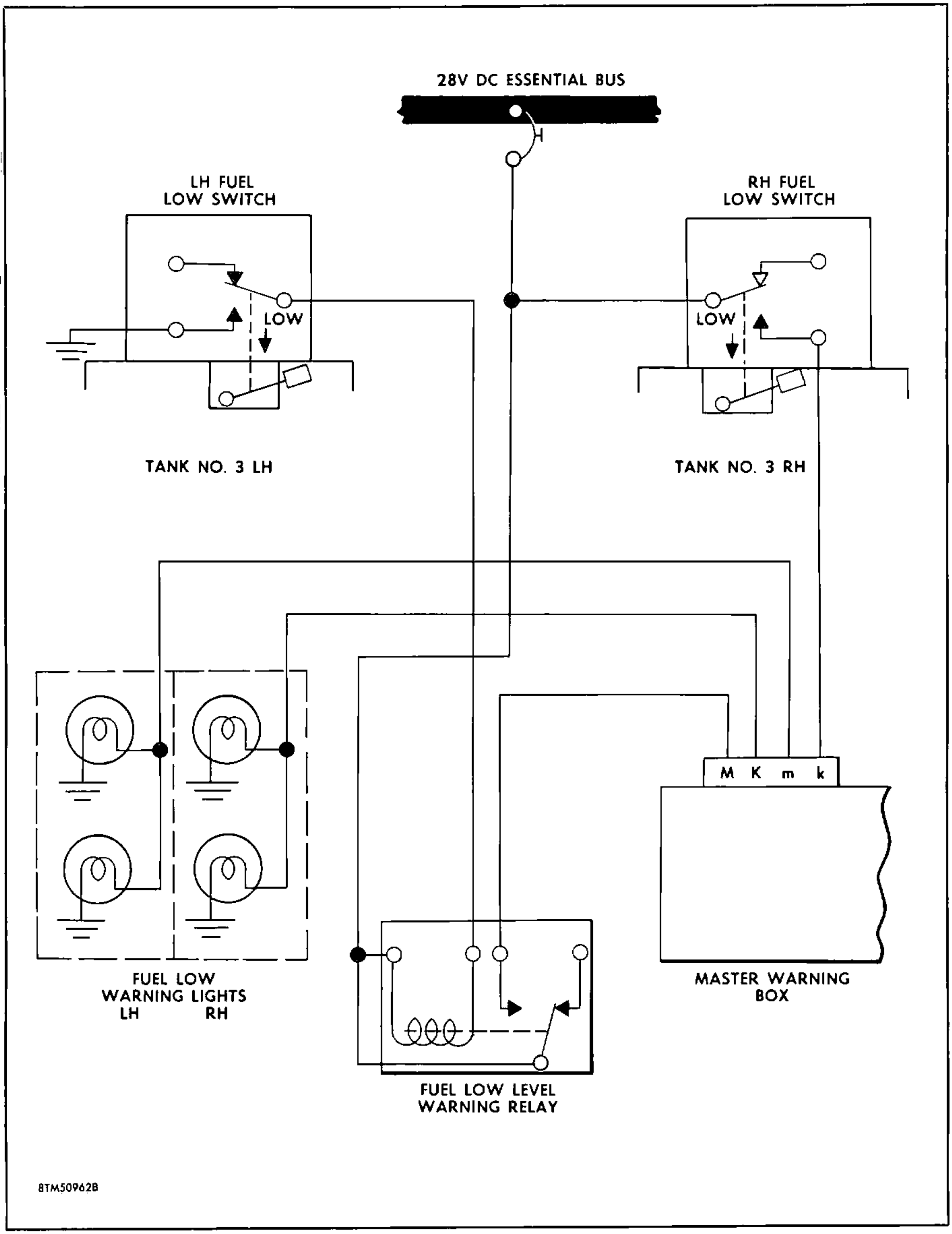

Figure 6-11. Fuel Quantity Low Level Warning System Schematic

LH tank drops to 88 gallons, the circuit is completed to ground and the low-level relay is energized. Current can then flow through the relay switch to terminal M on the master warning relay box. The current leaves the box at terminal M and flows to the LH warning light which is also grounded. Thus, the LH fuel low-level warning light comes on. Once the fuel low-level warning lights come on, they will remain on until the airplane is refueled. Refueling the tanks raises the float on the switches to the open position shown, thus extinguishing the warning lights.

FUEL TANK PRESSURE LOW WARNING SYSTEM.

The next two lights on the warning indicator panel (items 11 and 12) are placarded FUEL TANK PRES L and FUEL TANK PRES R. Their purpose is to warn the pilot when for any reason, the No. 1 fuel tank pressure drops below 0.5 psi above the ambient air pressure. This pressure—bled from the engine compressors—is needed to transfer fuel from the number 1 and 2 tanks to the number 3 tanks, from where it is pumped to the engine. As you would expect, the sensing devices for these warning subsystems are pressure switches. One of the switches is located outboard of each main wheel well area, forward of the number 1 tanks. Pressure sensing lines connect the switches to their respective tanks, and the switches are also vented to atmosphere. Thus, they operate on differential pressure, not absolute pressure. The pressure sensing switches and the rest of the fuel tank pressure warning circuitry are shown schematically in figure 6-12. For details of the complete fuel system electrical schematics, refer to the Fuel System Supplement, Chapter IV.

There is no difference between the LH and RH fuel tank pressure warning circuits. As you can see in the illustration, both pressure switches receive current directly from the 28-volt, d-c system to their B terminals. When tank pressure in either or both No. 1 tanks drops to less than 0.5 psi above ambient pressure, their respective switches close and provide a circuit through the A terminals to terminals D and E of the master warning box. Current from the left-hand pressure switch enters at terminal D and leaves at terminal d. Current from the right-hand switch enters at terminal E and leaves at terminal E. From these terminals, the current flows directly to the warning lights. Since these lights are connected to ground, they should always be lit when the pressure switches are closed. If the tank pressure rises to 1.5 psi above ambient pressure, the switches will open again and break the circuits.

Both of the fuel tank low-pressure warning circuits are shown energized in our schematic diagram. This does not mean that the two circuits are in any way tied together or that they will always be energized at the same time. They have the same power source, but are otherwise independent.

FUEL TANK BOOSTER PUMP PRESSURE LOW WARNING SYSTEM.

Just below the fuel tank pressure low warning lights on the warning indicator panel, you can see the LH and RH FUEL BOOST PRESS placards. These signs, when illuminated, warn the pilot that the fuel pressure from the booster pumps which feed fuel to the engine is abnormally low. When the pressure differential between ambient air and either pump outlet line falls below 10.5 psi, the corresponding warning light will glow. The light will extinguish if the differential pressure again builds up to 12 psi or more.

The booster pumps are located in the left and right number 3 fuel tanks. Pressure lines run from these pumps to pressure switches similar to those used in the fuel tank pressurization warning system. You will find these switches in the main wheel wells in the wings and just forward of the number 3 tanks. For details of the complete wiring schematic of this system, refer to the Fuel System Supplement.

Now refer to the schematic of figure 6-13, which shows the circuits of the fuel booster pump pressure low warning system. Note that this system is very similar to the tank pressure low warning system. The important difference, of course, is in the operating range of the switches. Also note that these circuits connect to different terminals of the master warning box. Other than that the circuits are identical so we will not go into further detail on this one.

WINDSHIELD AIR OVERHEAT WARNING SYSTEM.

Visibility through the windshield of any airplane is greatly reduced during rain conditions. At the high speeds attained by modern jets, the rain problem becomes particularly serious. Ordinary windshield wipers are not suitable under these conditions, so the F-102A uses hot air rain clearing. Engine bleed air is piped directly from the N2 compressor to an opening below the left windshield. By use of a switch on the utility switch panel the pilot can open a valve in the line, thus permitting the hot air to flow over the left windshield and form a barrier which prevents the rain from striking the windshield. At the point of discharge, this air is approximately 450° F.

This method of rain clearing presents some problems. There is always the possibility of damage to the windshield from too much heat. This windshield is of the sandwich type construction with a layer of plastic between two layers of glass. The glass can stand considerable heat, but the plastic cannot. To prevent damage to the windshield, the pilot must know when it is getting too hot. Then he can shut off the rain clearing air until the windshield cools. The next to the last light on the warning indicator

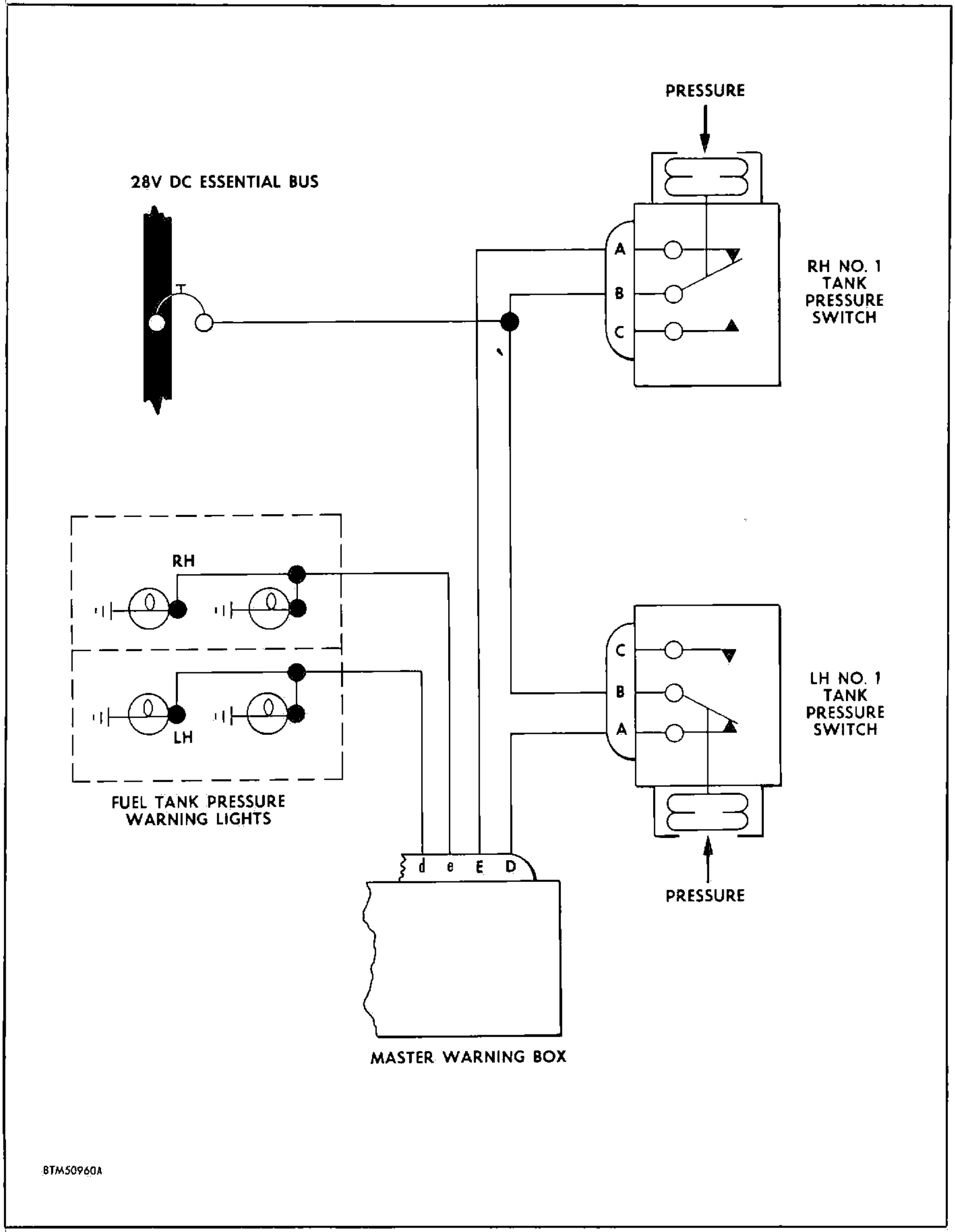

Figure 6-12. Fuel Tank Low Pressure Warning System Schematic

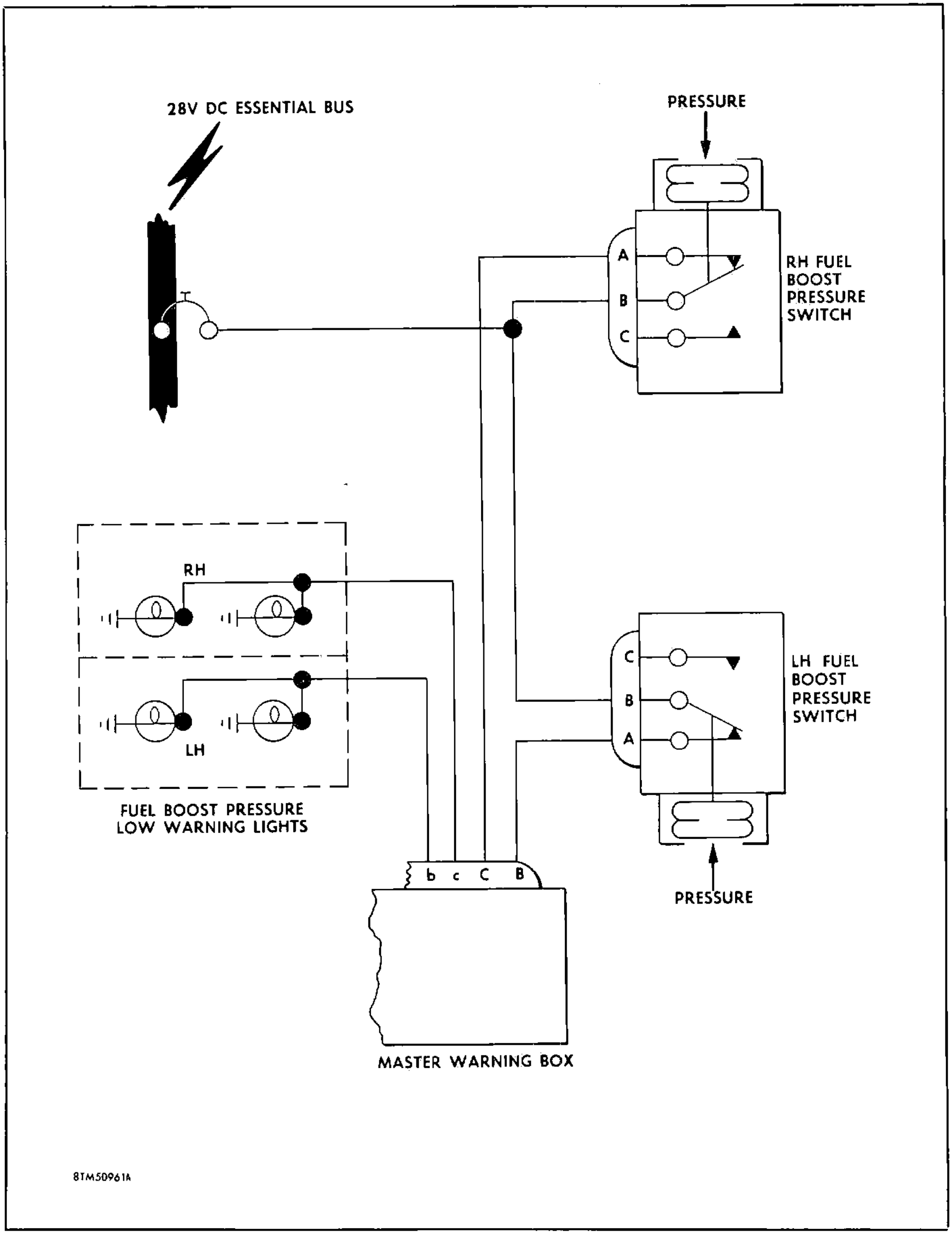

Figure 6-13. Fuel Tank Booster Low Pressure Warning System Schematic

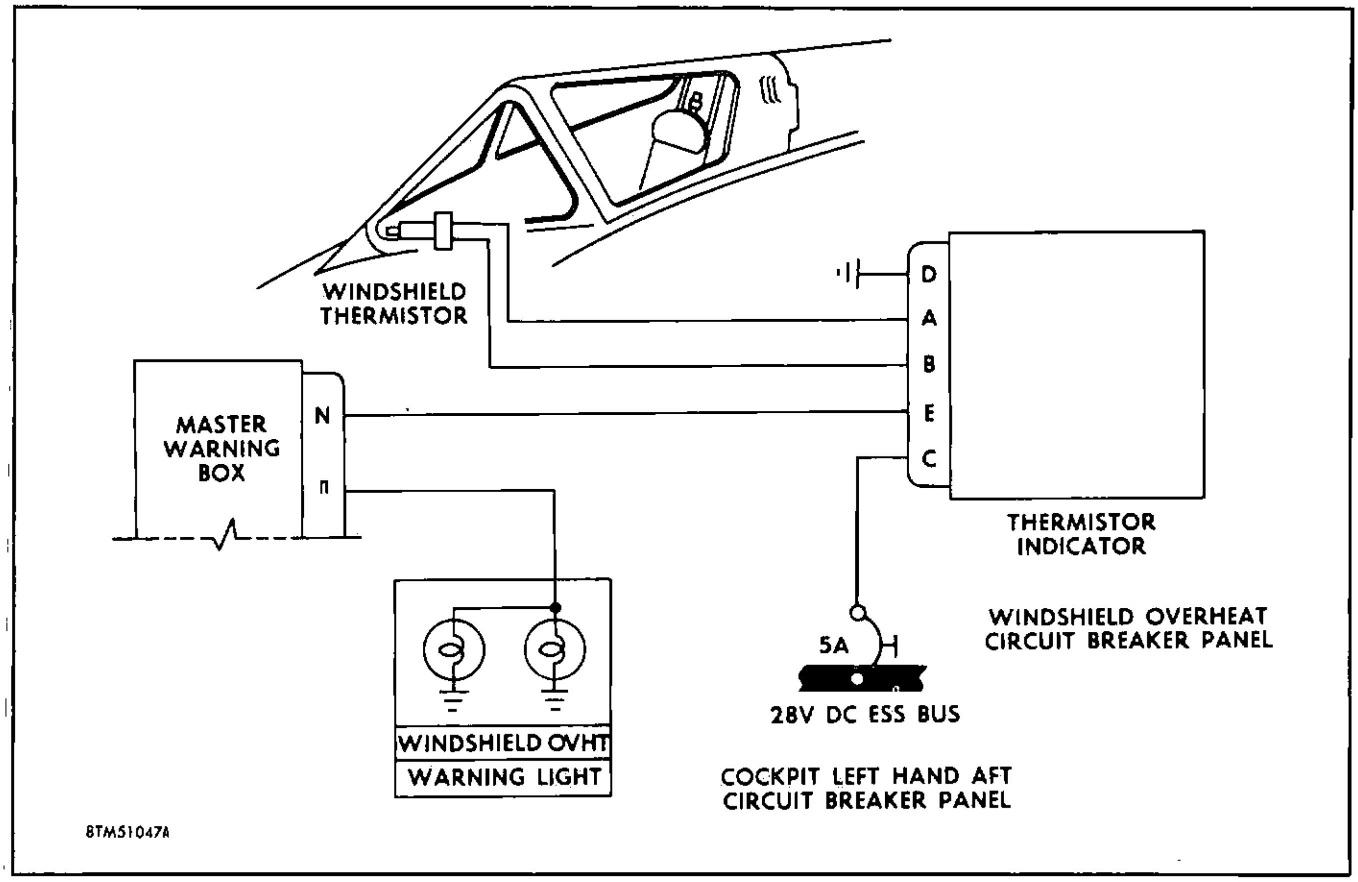

Figure 6-14. Windshield Air Overheat Warning System Schematic

panel, placarded WINDSHIELD OVHT, tells him when an overheat condition exists.

The windshield air overheat warning system consists of a thermistor, a bridge amplifier, and the usual warning light circuit. The thermistor is a resistance bulb which is sensitive only at its tip. It is mounted at the forward apex of the left windshield and can be removed from inside the cockpit. The bridge amplifier (sometimes called the thermistor indicator) is a small unit utilizing transistors instead of vacuum tubes. An adjusting screw is provided for temperature calibration. Aside from this adjustment, no changes or repairs are permitted on this unit. The amplifier, thermistor, and other components of the warning system are shown in figure 6-14.

Temperature sensing in the windshield air overheat warning system is accomplished by the thermistor. The tip of the thermistor is spring-loaded to maintain a good contact with the windshield glass. Whereas most temperature bulbs increase their resistance when temperature increases, this one does just the opposite. When the windshield temperature rises, the thermistor resistance drops. For that reason it is sometimes called a negative temperature bulb.

As you can see from figure 6-14, the only source of power to this warning system is the airplane's 28-volt d-c essential bus. Current flows through the circuit breaker to contact C of the amplifier. The amplifier reduces the voltage to about 5-volts in the thermistor circuit. Since the resistance of the bulb decreases when temperature increases, the current flow through the thermistor circuit increases. This acts to unbalance the bridge circuit in the amplifier, thus producing an electrical signal proportional to the temperature of the bulb. After amplification, the signal goes to a relay. When the current flow becomes great enough —due to low resistance of the thermistor bulb—the relay actuates, letting current flow through terminal E to the master warning box and the master warning light. The light will illuminate and remain on until the windshield temperature decreases below a predetermined level.

The windshield temperature at which the windshield air overheat warning light comes on is about 200° F. However, this figure may be subject to change in the future. You should always consult the latest applicable maintenance technical order for calibration data.

ELECTRONIC COOLING WARNING SYSTEM.

A considerable amount of heat is generated by the airplane's electronic equipment during its operation.

For that reason, cooling is provided for the forward, intermediate, and aft electronic compartments in the F-102A. When the airplane is flying, ram air furnishes a simple and reliable means of cooling. However, on the ground, methods of cooling are needed and it is here that the warning system is employed. The bottom light on the warning indicator panel informs the pilot of any malfunctioning in this cooling system. Several components are involved in the ground cooling of the electronic compartments. We will discuss them briefly from the standpoint of their relation to the warning system. A more comprehensive explanation of the entire cooling system is given in Chapter IV of the Low-Pressure Pneumatic System Supplement.

The most important of these components is the pneumatically operated jet pump. It consists of 6 nozzles which are connected through a solenoid shut-off valve to the engine bleed air supply line. These nozzles point forward and out of the left boundary-layer ram air intake. When the jet pump valve is open, hot bleed air leaves the nozzles in a high velocity "jet". This creates a suction in the distribution ducting resulting in a reverse flow of air through the ducting. Cooling air flows from outside the airplane, through the electronic compartment, and out the boundary-layer ram air ducts—just the opposite from the air flow during in-flight cooling, when the jet pump valve is closed.

The other components which we must consider in understanding the electronic equipment cooling warning system serve as control for the jet pump valve. Actually, we are more concerned with these controls than with the jet pump itself. They are the MLG (main landing gear) safety switch, the cabin air relay, the aft throttle position switch, and the structure overheat detection system (this system is not related to the structure overheat warning system around the power plant section). These items are called out in the schematic illustration of figure 6-15.

The MLG safety switch is on the left main landing gear and is actuated when the weight of the airplane compresses the strut on the ground.

The cabin air relay is in the intermediate electronic compartment and is energized when the MLG safety switch closes. Operation of the aft throttle position switch results from movement of the power control (throttle) lever—the switch is open whenever the power lever is forward of the idle position. The structure overheat detection system contains a sensing probe, a relay, and a detector. A switch in the detector closes and energizes the relay whenever the temperature around the jet pump gets too high.

Referring again to figure 6-15, note that all the switches and relays are shown in the position they are in when the airplane is on the ground with the power control lever at IDLE and the jet pump shutoff valve *open.* The structure overheat relay is shown deenergized, indicating that the structure around the jet pump is below the maximum allowable temperature. Now let's trace the current flow to the jet pump shutoff valve, keeping in mind that the valve is *open* when its solenoid is energized and *closed* when it is deenergized.

Note that power comes from the 28-volt d-c essential bus to contact 4 on the cabin air relay. From there, current flows through the switch to contact 3, and then to connection B on the throttle position switch. With the throttle at IDLE, current flows through the switch to connection C, then through point X to contact 10 on the cabin air relay. From contact 10 the current goes through the switch to contact 9 and on to contact 4 on the overheat relay. It continues through the switch to contact 5 and on to connection A on the shutoff valve solenoid. You can see that the current has passed four switches—through two switches on the cabin air relay, one in the overheat relay and one in the throttle position switch. All of these switches must be in the position shown for the current to reach the solenoid. The ground safety switch must be actuated, therefore the airplane must be on the ground. The power control lever (throttle) must be at IDLE and the overheat relay must not be energized.

To summarize the control system of the jet pump shutoff valve then, remember that the solenoid is *energized* when the airplane is on the ground, the power control lever is at IDLE, and the structure near the nozzles is not too hot. The solenoid should be deenergized when the airplane leaves the ground, when the power control lever is advanced beyond IDLE or when the structure overheats.

Now we are ready to discuss the electronic equipment cooling warning system. As mentioned before, this system includes a warning light on the warning indicator panel that tells the pilot of malfunctions which he cannot observe directly. The three basic conditions which can cause this light to glow are: shutoff valve *closed* when airplane is on the ground and the power lever is at IDLE; shutoff valve *open* when the airplane is on the ground and the power lever is advanced past IDLE; and shutoff valve open when the overheat relay is energized. Naturally, there are other defects that can occur in the system—such as short circuits, faulty switches, and the like, which can cause trouble. Whatever the cause, the electronic cooling warning light will illuminate in addition to the master warning light whenever current flows to the master warning relay box.

As you can see from figure 6-15, current flow to the master warning box is controlled by the position switch in the shutoff valve. This switch is always in either the *open* or the *closed* position. If it is in a

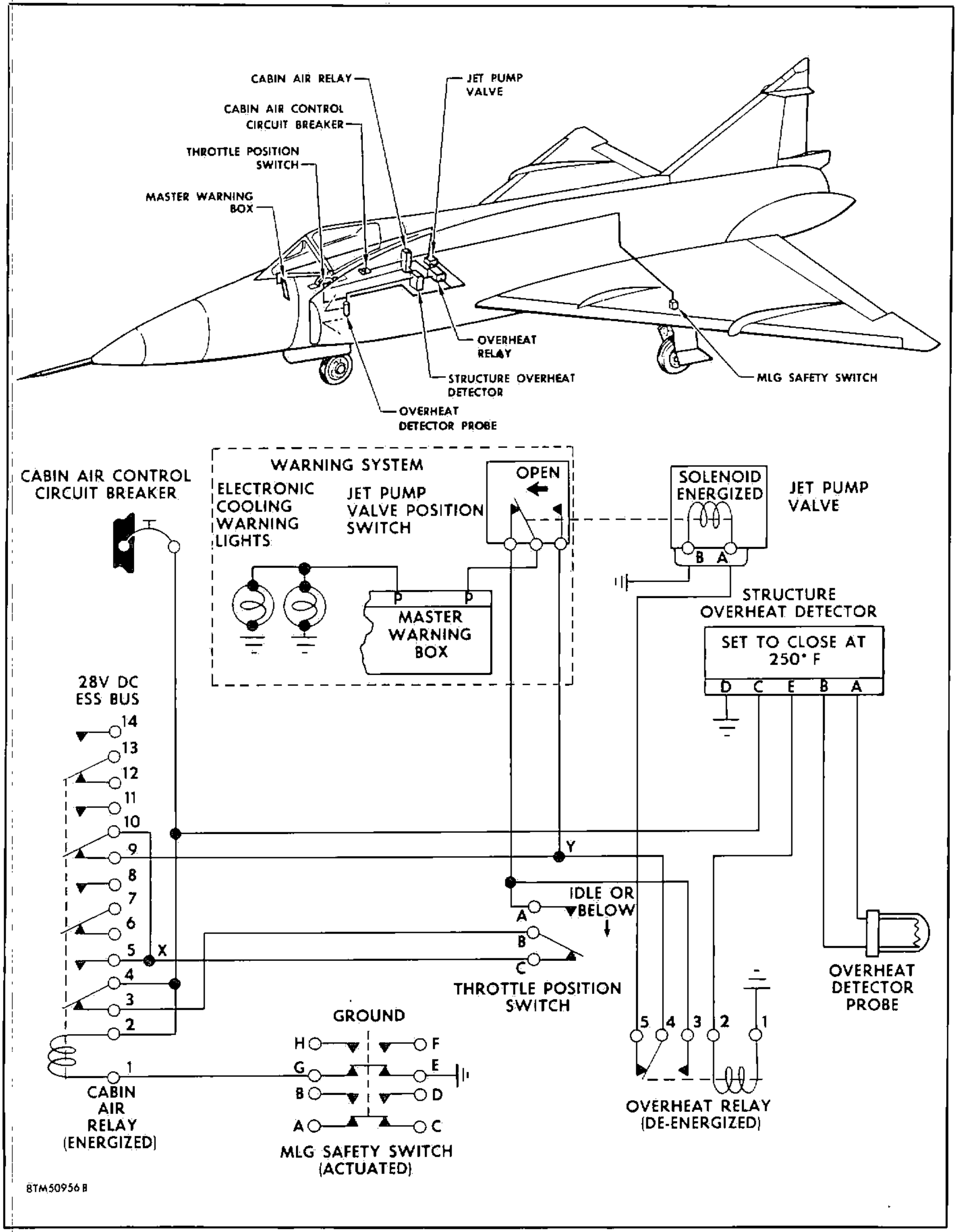

Figure 6-15. Electronic Cooling Warning System Schematic

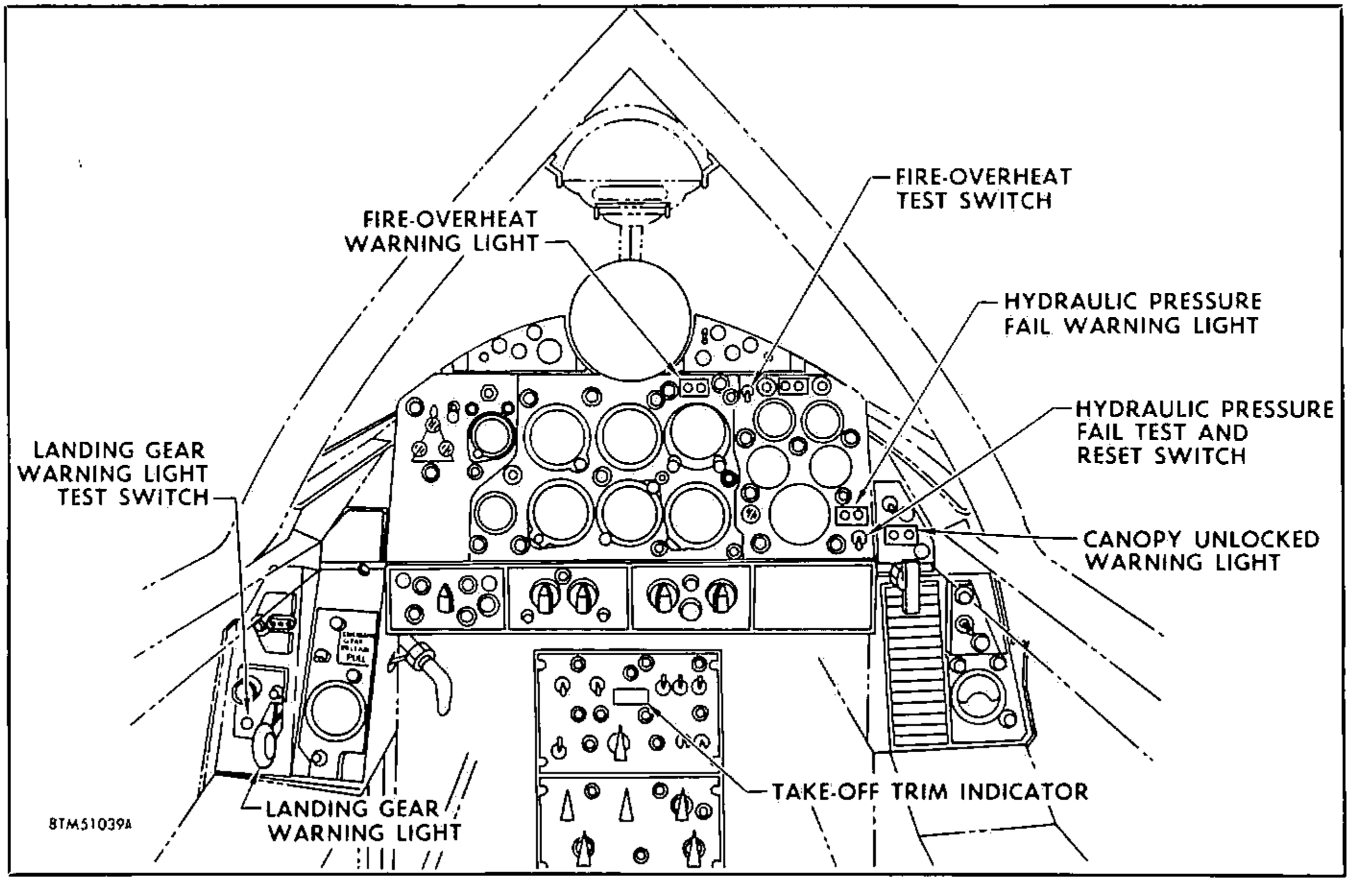

Figure 6-16. Miscellaneous Warning Light Locations

particular position at the wrong time, current is fed into the warning box and the lights come on. To understand how this takes place let's study what happens when we have the first of the basic conditions listed above. When the airplane is on the ground and the power control lever is at IDLE, the shutoff valve should be open. If it is closed, the position switch will also be closed and current will pass to the warning relay box from contacts 9 and 10 on the cabin air relay and point Y as shown on the schematic. The lights, being connected to ground, will then illuminate. If we have the second basic defect, that is—the valve is open, the airplane is on the ground, and the power lever is forward of the IDLE position—the position switch will be at the *open* position. Power then reaches the warning lights through contacts 4 and 3 of the cabin air relay and connection B and A on the throttle position switch. If we have the third defect—the valve *open* when the overheat relay is actuated—the current reaches the warning lights through the shutoff valve position switch—at the *open* position, contacts 4 and 3 of the cabin air relay, connections B and C of the throttle switch, point X, contacts 10 and 9 on the cabin air relay, and contacts 4 and 3 of the overheat relay.

MISCELLANEOUS WARNING SYSTEMS.

In addition to the warning systems discussed so far, there are five warning lights which are separate from the master warning system. They are the canopy unlock warning light (7), fire-overheat warning light (3), hydraulic pressure fail warning light (5), landing gear warning light (2), and the takeoff trim indicator light (8). Figure 6-16 shows the location of these warning lights in the cockpit. All but one of them are large rectangular-shaped lights just like the master warning light; that one exception is the landing gear warning light which is mounted in the plastic knob of the landing gear control lever. You can tell from the titles what systems are monitored by the various warning lights.

As stated in the introduction to this chapter, all of the individual warning lights have common dimming provisions. They are connected to a dimming relay panel in the master warning relay box, and are dimmed for the same reasons that the master warning system lights are dimmed. When the instrument panel lights are on, these warning lights will illuminate dimly if they come on. When the instrument panel lights are off, (as in daylight operations), or when the thunderstorm lights are on, the warning lights glow brightly. Like the master warning system, all these warning systems receive power from the airplane 28-volt d-c essential bus and all lights except the landing gear warning light, contain two lamps.

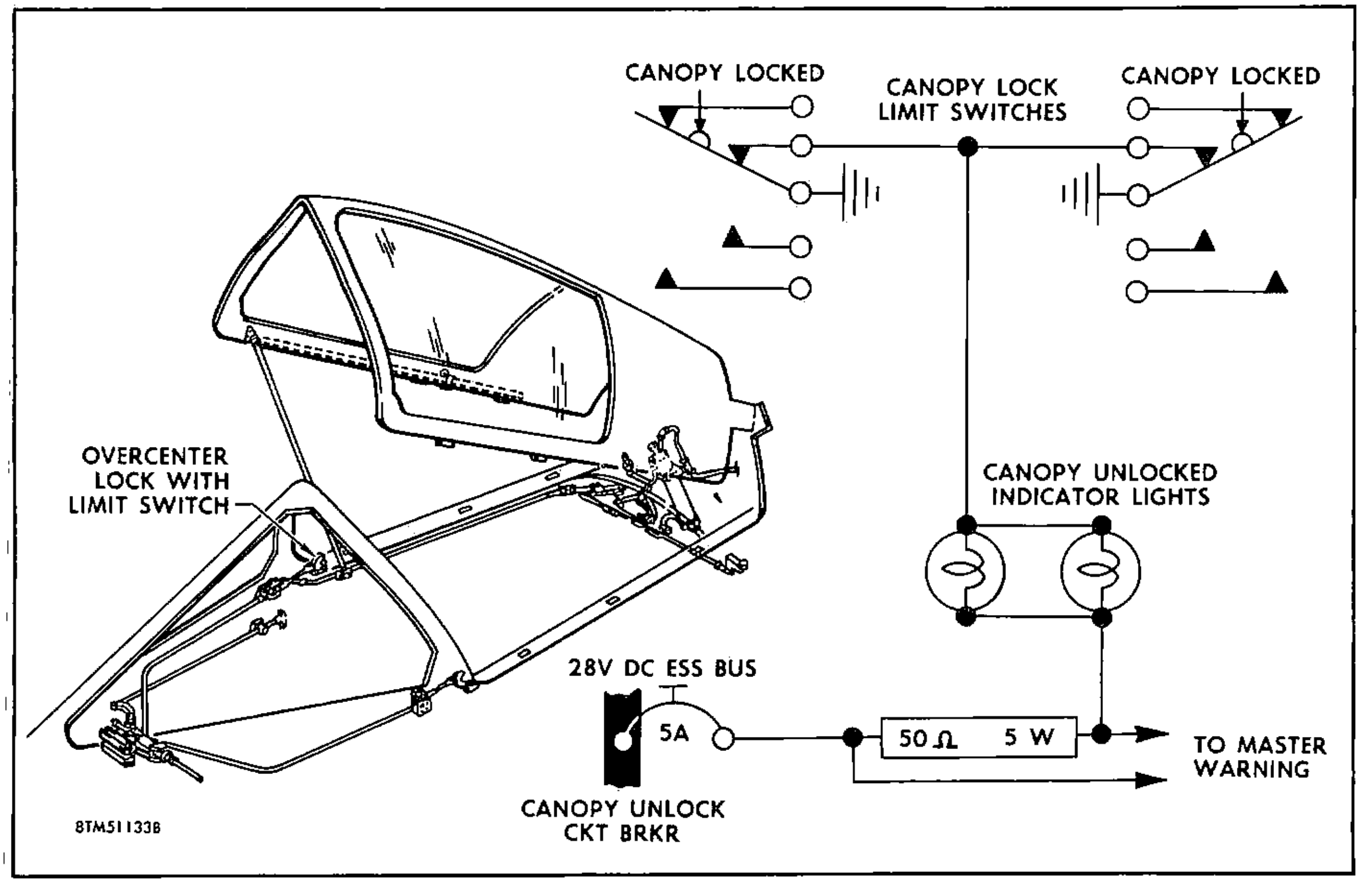

Figure 6-17. Canopy Unlock Warning System Schematic

CANOPY UNLOCK WARNING SYSTEM.

The F-102A cockpit canopy must be locked securely before takeoff to prevent its being torn off in flight, and to permit sealing of the canopy for cockpit pressurization. Canopy locking is accomplished by two latch hooks which engage rollers in the lower forward corners of the canopy. These hooks are actuated by a "T"-shaped push-pull handle on the right auxiliary instrument panel. In the event that either latching mechanism does not engage its roller, a switch on that latch will energize the warning circuit. The warning light for this circuit is located directly above the canopy push-pull control handle.

In figure 6-17 you see a schematic diagram of the canopy unlock warning system. Note that the canopy lock limit switch contacts are closed when the canopy is not locked. In this position current from the 28-volt d-c essential bus is provided a path to ground. The current flows through a circuit breaker and then through either the dimming resistor or the master warning box. If the flight instrument lights are on, the current must pass through the resistor. If none of the instrument lights are on or if the bright thunderstorm lights are on, then a relay switch in the master warning box is deenergized, permitting the current to by-pass the resistor: (see Electrical System Supplement, page 5-10). We discussed the reason for this arrangement at the beginning of this chapter. Regardless of the path taken by the current, it next flows through the warning light before grounding at the switches. Since both switches are grounded, the light will illuminate if either one of them is closed and d-c power is turned on.

The canopy lock limit switches are actuated by bolts protruding from the latching mechanism. If the warning light will not extinguish when the canopy is in the locked position, check that both of these bolts are adjusted properly to actuate their respective switches. If both switches are being actuated properly, you can check them out individually by disconnecting one at a time (either from the circuit or from ground) and actuating the other one. Be sure that the disconnected lead or switch is not permitted to ground out or the test will not be valid.

FIRE AND OVERHEAT WARNING SYSTEM.

Two different subsystems make up the fire and overheat warning system. One subsystem notifies the pilot of a fire condition in the engine compartment, while the other subsystem warns him when an overheat condition exists in this compartment. A single warning large light located on the main instrument panel to the left of the master warning light gives both warning indications. By routing the electrical signal from the overheat circuit through a flasher, the warning light will flash *on and off* whenever an engine overheat condition exists. Should a fire exist in the engine compartment, the warning light will come on and burn steadily. Don't confuse the engine overheat warning system with the structural overheat warning system that we discussed earlier in this chapter; they are not connected in any way.

Both the overheat and fire warning systems use detector loops. These detector loops—the temperature sensing elements of the system—are actually a type of coaxial cable with electrical connections on each end. The cables consist of an inner electrical conductor, which is separated from an outer sheath of corrosion resistant alloy by a thermistor-type heat sensitive compound. The electrical resistance of this heat sensitive compound varies inversely to the temperature. Under normal operating conditions the compound acts as a good insulator; when a *hot spot* develops anywhere along the detector cable, the resistance of the compound drops and allows current to flow from the inner conductor to the outer sheath. This completes the circuit to ground and the warning light comes on.

The components of the fire and overheat warning systems are shown in figure 6-18. In the upper portion of the illustration note that the structural overheat loop and the fire detector loop are situated in different sections of the engine compartment. The detector relays, overheat flasher, and detector control boxes are located in the upper electronics compartment.

Operation of the System.

To get a good idea of how the fire and overheat warning system operates, let's trace the current flow as shown in figure 6-18. Current for this warning system, like all other warning systems, is taken from the 28-volt d-c essential bus. It passes through the circuit breaker to the two detector control boxes and on to the junction point of the two relays. Note that the two lamps in the warning light must receive their current from either of the detector control boxes before the warning light will illuminate; however, the control boxes cannot send current to the warning light until the warning light circuit grounded at some point. This occurs when one or both of the detector loops are subjected to a "hot" condition that lowers the internal resistance of the separating compound. Whenever one of the cables grounds out, the respective detector control box circuit is completed and current flows to the light. If the overheat loop grounds, the current is first routed through the flasher unit and then to the warning light. This causes the warning light to flash on and off. Since the fire detector circuit does not have a flasher unit, a fire condition causes the warning light to burn steadily. The fire and overheat detector loops are easily damaged, and the usual result of loop damage is a warning indication when either a fire or an overheat condition exists. This is because the loop is likely to ground out in the damaged area. If this occurs, you will have to replace the loop segment that is faulty.

Fire and Overheat Test Switch.

A test switch on the main instrument panel is to energize the two warning circuits for test purposes. This switch, mounted just to the right of the fire and overheat warning light, has three positions. It is spring-loaded to the center, or off, position. By placing the switch in either the upper (fire) position or the lower (overheat) position, the pilot can check the functioning of these circuits without heating the detector loops. As you can see in the diagram, the switch merely provides a ground for the circuits.

HYDRAULIC PRESSURE FAIL WARNING SYSTEM.

A warning system is included in the hydraulic system to warn the pilot of critically low hydraulic pressure. If the hydraulic pressure in either the primary or secondary system drops to 800 psi or lower, a light on the instrument panel flashes a warning to the pilot. He then checks his pressure gage to determine which system is malfunctioning. If both systems go out, the light glows steadily. Therefore, this warning system monitors both hydraulic systems at the same time. The warning light extinguishes when the pressure again increases to about 1000 psi, or the pilot *resets* the warning light.

The warning system utilizes two hydraulic pressure switches, a flasher unit, a test and reset switch, and a warning light. The hydraulic warning system is more fully discussed in Chapter I of the Hydraulic System Supplement.

The pressure switches, which are located in the hydraulic accessory compartment, complete the circuit from the two hydraulic systems to the instrument panel warning lights. The two switches are hydraulically actuated and work in conjunction with each other to illuminate the warning light when low pressure causes them to close the electrical circuit to the light. Each hydraulic pressure switch has contacts that, when closed, complete a 28-volt, d-c electrical circuit to the hydraulic pressure warning light. The contacts in either switch close whenever the pressure in that system drops to 800 psi. No attempt should be made to disassemble or adjust these pressure switches since they are factory sealed and adjusted. However, if either switch malfunctions and gives false indication, you should replace the entire switch. When the portable hydraulic test stand is used during ground checkout operations, you can check the warning light system to see that it operates properly.

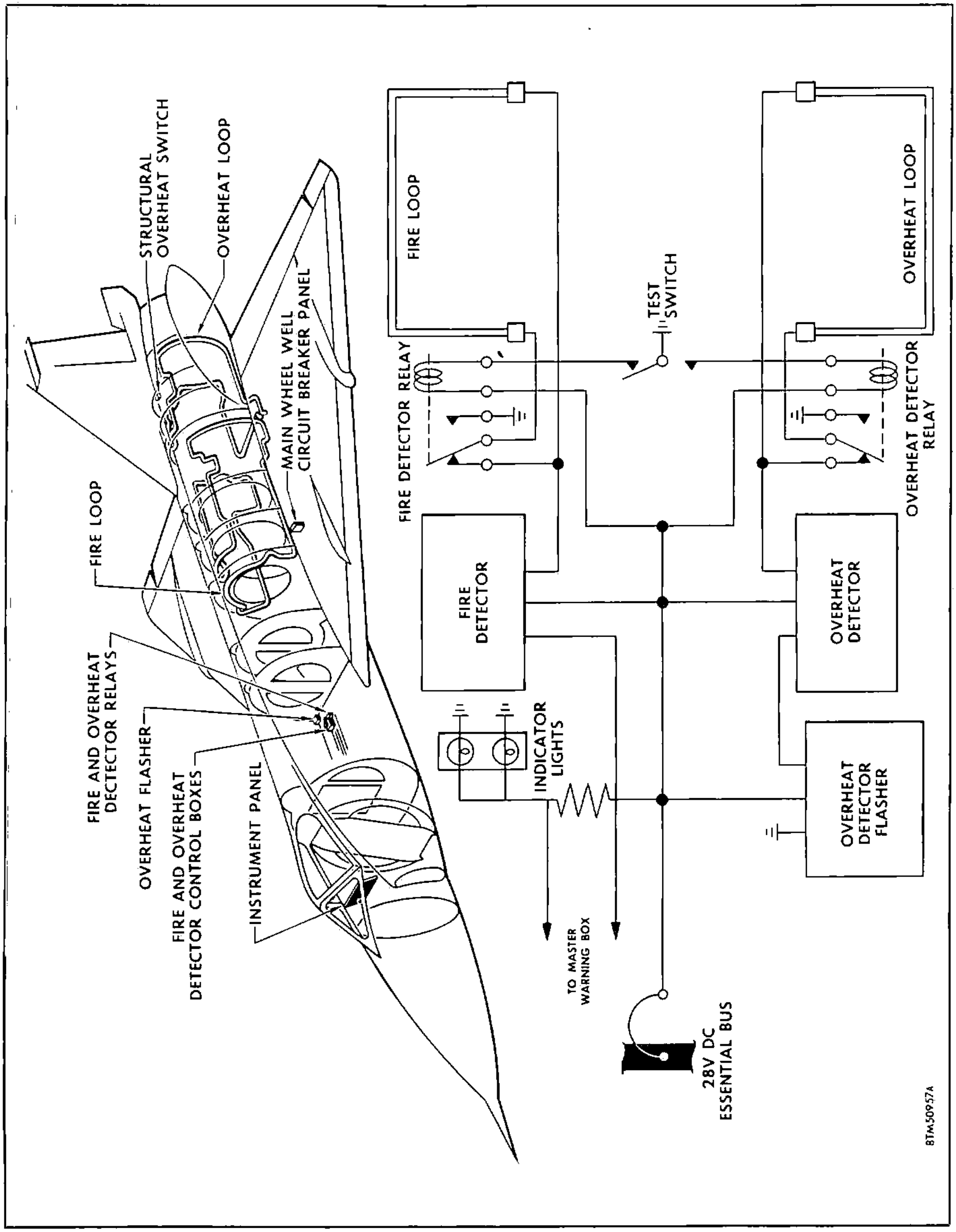

Figure 6-18. Fire and Overheat Warning System Schematic

Figure 6-19 shows the complete low pressure warning circuit for both primary and secondary hydraulic systems. Note the two hydraulic pressure switches, one for the primary system and one for the secondary system. Both switches complete the warning circuit to illuminate the warning light. Now let's see how this warning system functions.

Both pressure switches are shown actuated. This is the condition they would be in if the pressure in both hydraulic systems were 800 psi or less. Note the spring in each switch. These springs are compressed by system pressure which holds the electrical contacts *open* when system pressure is above 800 psi. However, the spring tension is sufficient to overcome hydraulic pressure of 800 psi or less and actuate the internal switches to the *closed* position.

First let's see how the warning system functions when the pressure in one system is low and the pressure in the other is satisfactory.

Assume that the primary system is below 800 psi and the secondary system is at 3000 psi. In this case the primary pressure switch would be as shown and the two switches D and A in the secondary pressure switch would be at terminals F and C respectively instead of the position in which they are shown. Power from the 28-volt d-c bus enters the primary pressure switch at terminal A, leaves at terminal B and goes to terminal B at the flasher unit. This same power goes to terminal A on the secondary switch; but since this switch is open at terminal C as we assumed above, the circuit is incomplete. From the flasher unit the circuit is completed through the resistor to the light causing it to flash on and off. At the resistor you will note that the circuit is connected to the Master Warning System—we will discuss this phase later.

With the flashing of the warning light, the pilot checks his hydraulic pressure gage to determine which system is low. Then, so he won't have the flashing light distract his attention, he presses the test and reset switch to RESET. This completes a circuit from the 28-volt d-c bus to the relay in the lower left corner of the schematic and energizes it, moving the contact to the other post. Since the relay is a holding-type relay, this breaks the circuit to the flasher unit—the lights go off, and the circuit through the pressure switch now energizes the relay even after the pilot releases the test and reset switch. The lights will not illuminate again until a low pressure indication is received from the secondary system. You will also notice that power entered the primary pressure switch at terminal D and went through terminal E to terminal D in the secondary pressure switch; but since this switch was open, power does not leave the switch.

Now let's assume that pressure in both systems is below 800 psi as shown in the schematic. The circuit through the primary pressure switch will pass through terminals D and E in the secondary switch. This power from terminal E will bypass the flasher unit and the circuit will be complete direct to the lights causing it to illuminate in a steady condition.

Referring back to figure 6-19, you should recognize the purpose of the resistor between the flasher unit and warning light. Remember that the resistor is bypassed under certain conditions where it is desirable to have bright illumination of the warning light.

The TEST position on the test-and-reset switch provides you a means of checking the system to determine that the lights and flasher unit are satisfactory. Placing the switch in TEST causes the lights to flash on and off.

LANDING GEAR WARNING SYSTEM.

In the preceding chapter you learned how the landing gear position indicating system tells the pilot the position of his landing gear. The landing gear warning system supplements this information. It warns him whenever the gear is not in the position which he has selected or whenever the gear is in an unsafe position for his condition of flight. The warning light for this system is in the plastic handle of the landing gear control lever. If the light illuminates, the pilot immediately sees it and checks his position indicators to determine which of the individual gears is not in the selected position.

Most of the landing gear warning system components are also used in the position indicating system and are therefore familiar to you. However, as you can see from figure 6-20, there is some additional equipment involved here. You may recall the airspeed and altitude switches from our discussion of the pitot-static system in Chapter I. These switches are located in the upper electronics compartment. The airspeed switch connects to the pitot tube and is vented to static air in the compartment. This switch contains a common airspeed diaphragm which permits a set of switch contacts to close whenever the airspeed is 250 knots or less. The altitude switch, also vented to static air in the compartment, contains an aneroid diaphragm. Below 10,000 feet, the altitude switch contacts are closed. Above that altitude, expansion of the diaphragm holds the contacts apart. All other switches in this warning system are mechanically actuated.

One more switch that we haven't mentioned before is the landing gear warning switch on the power control lever (throttle) quadrant. The contacts of this switch are closed whenever the power lever is aft of the TAKEOFF position.

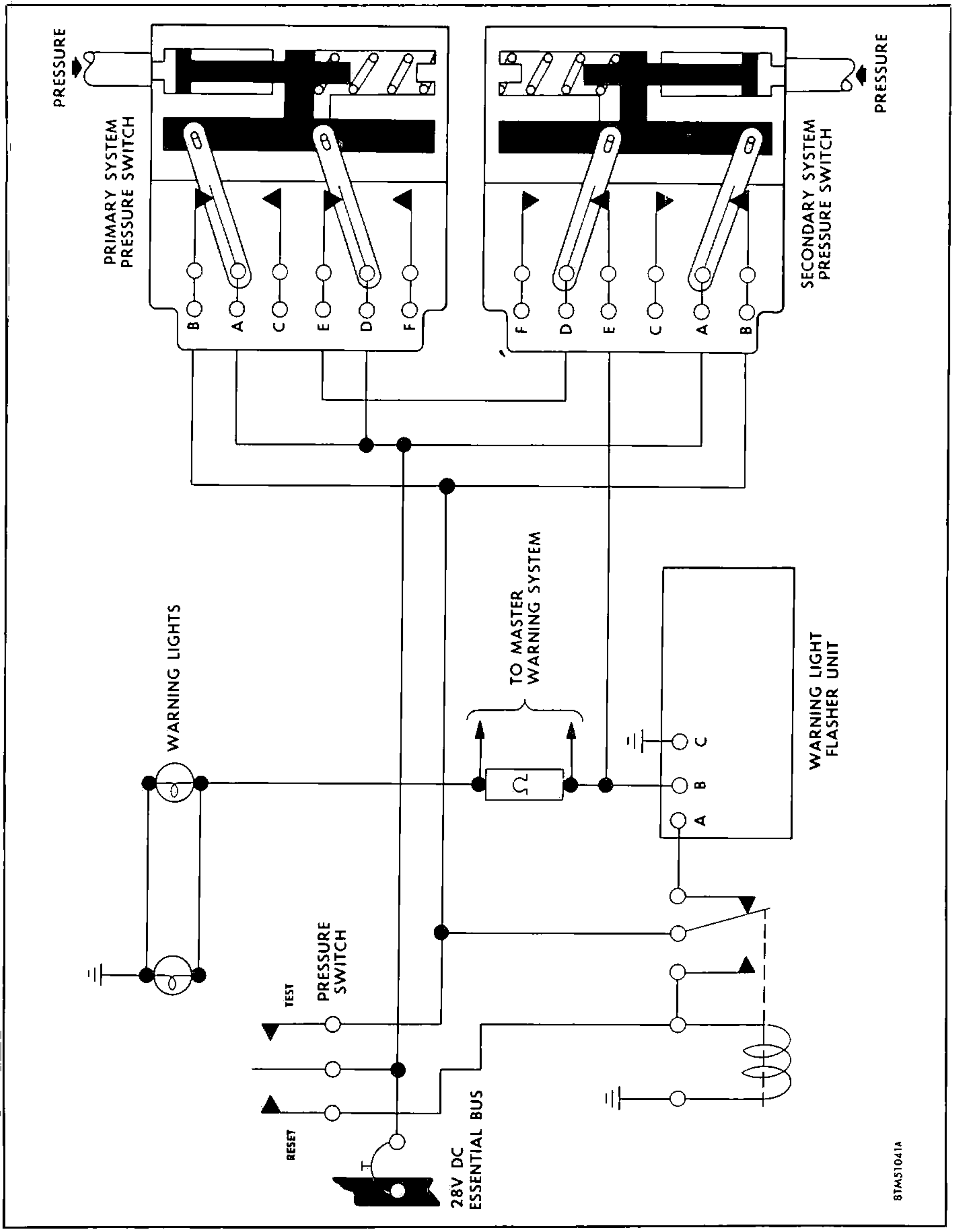

Figure 6-19. Hydraulic Low Pressure Warning System Schematic

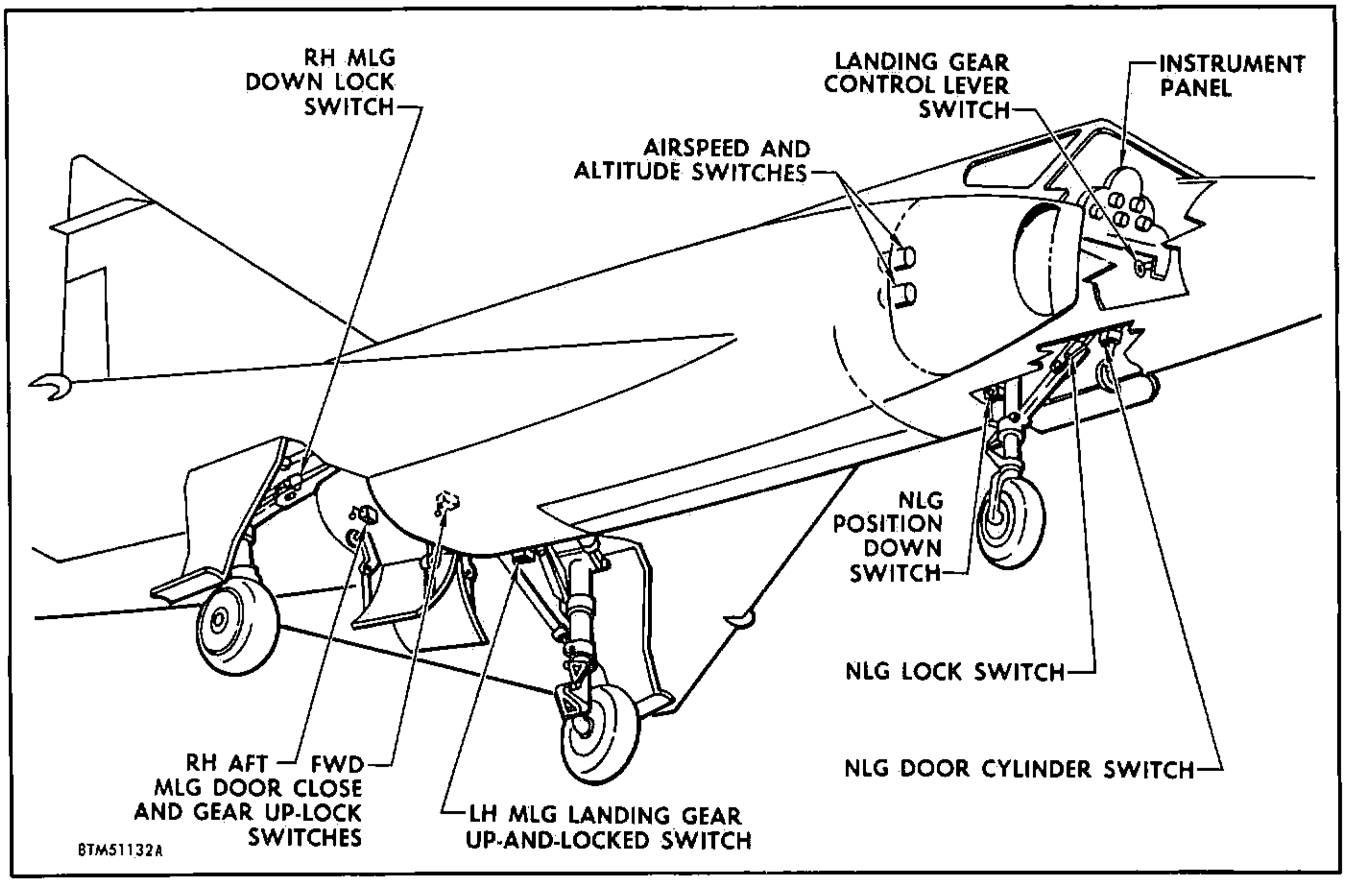

Figure 6-20. Landing Gear Warning System Component Locations

System Operation.

Now let's refer to figure 6-21. Note that all of the switches are shown in the positions they assume when *not actuated*. The arrows show what happens when each of the switches are actuated.

This discussion would be extremely lengthy if we covered every possible malfunction that would give a warning indication. Instead, we will trace the circuit as it functions in the normal *up-and-locked* and *down-and-locked* conditions, and then induce a couple of typical malfunctions. You will then be able to figure out other malfunctions in addition to those that we discuss here.

As we discuss the warning system operation, keep in mind two facts: *any one* (or more) of the landing gear position switches will cause the warning light to illuminate if not in the proper position relative to the control lever. The airspeed, altitude, and throttle quadrant landing gear switches will cause the light to illuminate only if *all three* of them are *closed* at the same time—if any one of these three switches is open, the light will not come on. Suppose an F-102A is waiting at the end of the runway, ready for takeoff. Assuming that the power control lever (throttle) is at IDLE and everything is normal, the switches will be positioned as noted below:

LANDING GEAR DOWN-AND-LOCKED, CONTROL LEVER DOWN

Switch	*Position*
NWW Door Cylinder	Closed
NLG Lock	Open
NLG Position Down	Open
MLG Forward Door Closed	Closed
MLG Aft Door Closed	Closed
MLG Up-And-Locked	Closed
MLG Down-and-Locked	Open
Throttle LG Warning	Closed
Airspeed	Closed
Altitude	Closed

Current leaves the 28-volt d-c essential bus through the landing gear position circuit breaker, and through points J, Z, and Y. Since the gear is down and locked, all of the MLG switches are closed, except for the *down-and-locked* switches. Current cannot reach points X and W, but it does flow through the closed switches to V, U, and M. From M, current cannot flow through the control lever switch, but could flow around through K, L, and O. Since the landing gear (LG)

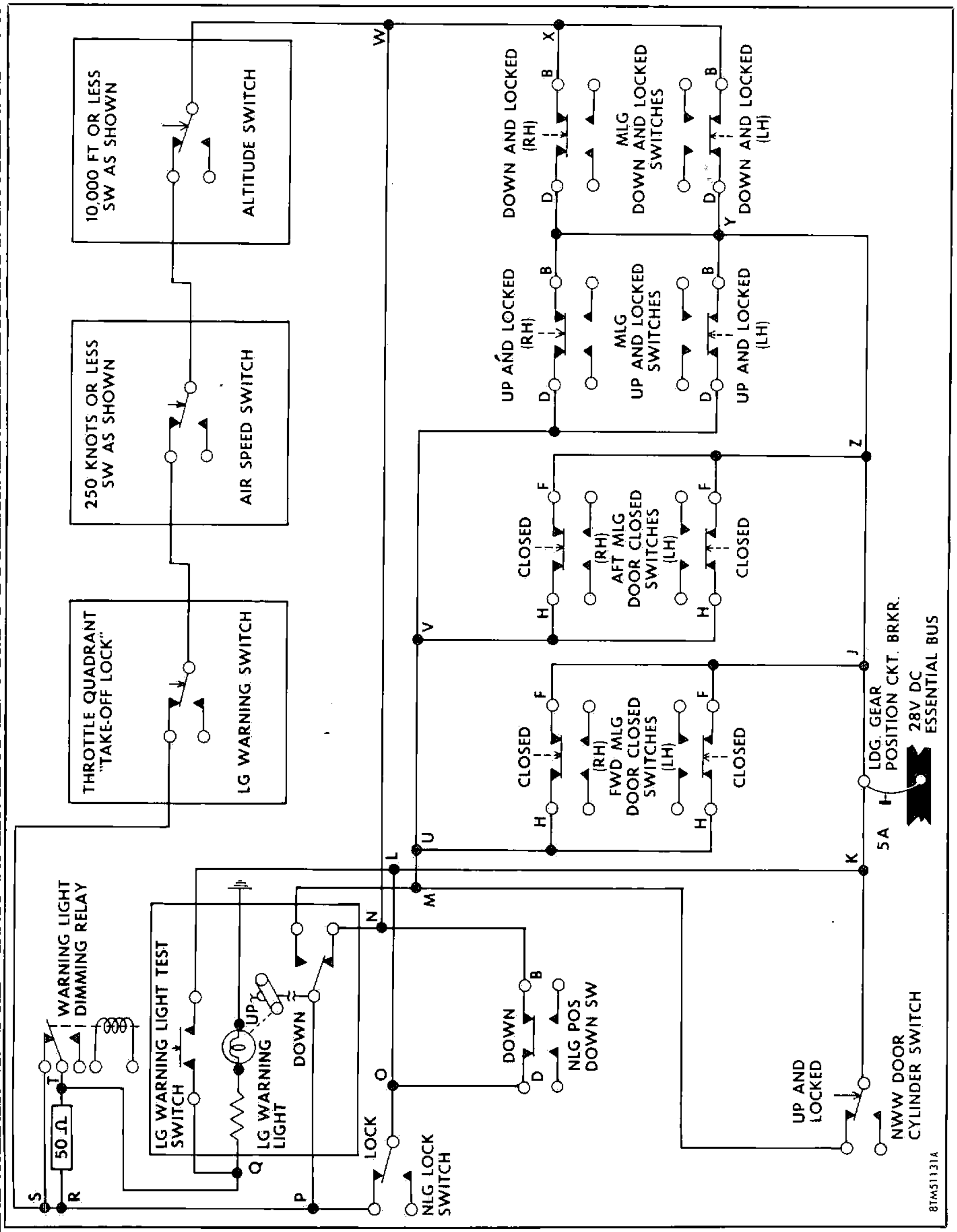

Figure 6-21. Landing Gear Warning System Schematic

warning light test switch, the NLG lock switch, and the NLG position down switches are all *open*, no current can reach the warning light. It would be possible for current to flow directly from the bus to points K, L, M, and O, but the flow would be stopped by the same open switches.

Now let's assume that our F-102A received takeoff clearance. The pilot applied takeoff power and is just airborne. He moves the landing gear (LG) control lever to the UP position and the gear is now up-and-locked with no malfunctions indicated. The positions of the various warning system switches will be as shown below:

LANDING GEAR UP-AND-LOCKED, CONTROL LEVER UP

Switch	*Position*
NWW Door Cylinder	Open
NLG Lock	Open
NLG Position Down	Closed
MLG Forward Door Closed	Open
MLG Aft Door Closed	Open
MLG Up-And-Locked	Open
MLG Down-And-Locked	Closed
Throttle LG Warning	Open
Airspeed	Closed
Altitude	Closed

Starting again at the 28-volt d-c bus, let's trace the current flow through the MLG part of the system. The forward and aft MLG *door closed* switches and the MLG *up-and-locked* switches are now open. However, the MLG *down-and-locked* switches are closed. Current can flow through both of these switches to points X and W. The altitude and airspeed switches are closed, but the landing gear (LG) warning switch on the throttle quadrant (connected in series with them) is open because the power lever (throttle) is in the *TAKEOFF* position. Thus, current cannot flow on from there. However, there is a circuit from W to point N and through the NLG position down switch to point O. But, since both the NLG lock switch and the lower contact of the LG control lever switch are open, current can go no further. Going in the other direction from the power source we find that current can flow through points K, L, and O but is stopped by the open NLG lock switch and the LG warning light test switch. Therefore, none of the circuits are complete through the warning light to the common ground.

The most obvious malfunction that could cause the landing gear warning light to come on is when the NLG fails to lock in either the *up* or *down* position. This provides a direct current path through points K and L, on through the closed contacts of the lock switch, and through either the dimming resistor or relay to the warning light. In the illustration, the dimming relay switch is shown closed, so the current will bypass the resistor.

Let's take a case where the pilot moves the LG control lever to DOWN, and the right main landing gear fails to lock down. Normally, all MLG switches should be in the closed position, except the *down-and-locked* switches. Current would have a path through these closed switches to point M and around through K, L, and O. At this point it would be stopped by the open NLG lock and the NLG position down switches. But there is one thing wrong. Both MLG down-and-locked switches *should be* open, but the right hand switch is still closed. This provides a path from point Y to point X. From there, the current flows to N through the lower contact of the LG control lever switch to P, and on through the dimming circuit to the warning light.

Perhaps we should briefly examine the one way in which the landing gear warning light illuminates without any malfunction existing. That can occur when the airplane "lets down" below 10,000 feet at a speed of 250 knots or less with the power control lever in any position aft of the TAKEOFF setting. If the landing gear is not down, current flows through the closed MLG down-and-locked switches to points X and W. From there it flows through the closed altitude, airspeed, and throttle quadrant switches to the dimming and warning light end of the circuit and to ground. The pilot notices the warning light and lowers the landing gear, thus avoiding an inadvertent "wheels up" landing.

Landing Gear Warning Light Test Switch.

The landing gear warning light test switch is located immediately outboard of the landing gear control lever. Its name explains its purpose and you can easily see how it operates in the warning system schematic (figure 6-21). Current from the 28-volt d-c essential bus is always available at one of the switch contacts through points K and L. When the button-type test switch is depressed, the circuit is completed through the warning light to ground. The warning light can be tested in this manner with the gear either up-and-locked or down-and-locked.

TAKEOFF TRIM INDICATOR LIGHT.

The F-102A does not employ trim tabs; instead trimming the airplane is accomplished by trim actuators which deflect the regular control surfaces. In flight, the pilot uses a switch on the control stick to energize the aileron and elevator trip actuators and a switch on the utility switch panel to energize the rudder trim actuator. The pilot can tell by the airplane's "feel" and by his instruments when the trim is satisfactory for the desired flight condition. On the ground, he has no way of knowing how much trim is set into his control surfaces. He cannot tell if his rudder and elevons are completely neutral when the stick and rudder pedals are neutralized; and, for example, if he were to begin his flight with nose down trim and left rudder trim, the airplane would have very undesirable takeoff characteristics. To eliminate that danger, this airplane has an automatic trim system. By pressing a button on the utility switch panel the rudder and aileron controls are automatically neutralized to the 0° position, while the elevator control is moved to the 5° up position. The takeoff trim indicator light comes on when the control surfaces arrive at these positions. In effect then, this light gives its warning by *not* illuminating. If it doesn't come on when the pilot presses the takeoff trim button, something is wrong with the trim system. Keep in mind that when speaking of ailerons and elevators we actually mean the aileron and elevator *function* of the elevons. As you know, the F-102A employs elevon surfaces for aileron and elevator control action.

Now refer to figure 6-22. Note that this system receives power from the 28-volt d-c essential bus through the nose wheel steering circuit breaker. In addition, the current is routed through the nose landing gear position up switch. This up switch is closed only when the nose landing gear is extended thus causing the takeoff trim system to be inoperative during normal flight.

When the takeoff trim push-button switch is depressed, current flows to point D at the aileron trim actuator. If the trim actuator shaft is retracted rather than at the neutral position, the contact arm from D will contact pin H. Current then flows through H and A to energize the "extend" winding of the actuator motor. The actuator shaft moves until a cam breaks the circuit between D and H, positioning contact arm D against terminal J. At this time, the actuator shaft is at the neutral position. From J, current flows through the "retract" contacts E and G, and on to the elevator trim actuator.

If, instead of being in the retracted position, the aileron trim actuator shaft is in the extend position, contact E will meet F, and the opposite winding of the motor is energized. In this case, contact D is at J so the "extend" winding is not energized. In either case—whether the trim actuator shaft is extended or retracted—the appropriate winding is energized until the shaft reaches the neutral position.

The process that we traced in the aileron trim actuator is repeated in the elevator trim actuator except for one difference—the automatic trim switches are set to open at 5° *up* elevator. After the elevons are properly positioned at 5° up elevator, the circuit is completed to the rudder trim actuator. The centering operation again takes place and the current flows on to the trim indicator light circuit. When the dimming relay contacts, 3 and 5 are together, the warning light glows brightly. When this dimming relay is energized, the current must pass through the resistor which dims the light. You will remember the reason for this dimming arrangement from our discussion of the Master Warning System earlier in this chapter.

Thus, the takeoff trim indicator light shows that the proper events have taken place in the automatic takeoff trim system. The pilot is assured that the airplane's control surfaces are pre-set to provide the most satisfactory trim conditions for takeoff.

MAINTENANCE OF THE WARNING SYSTEMS.

The warning systems in an airplane must be reliable. They are installed for the express purpose of monitoring other systems and equipment which are likely to be less reliable. The pilot depends on those warning systems to be accurate. If they don't warn him when they should, or if they give him false warnings, he would be better off without them. The problem, then is to keep the warning systems completely reliable.

The malfunctions that can occur in any of the warning systems are easy to classify. The warning lights may come on when they shouldn't, fail to come on when they should, or blink erratically when they should be either on or off. Let's discuss some of the factors that could be involved in these malfunctions.

Suppose that a warning light does not illuminate when it should. Two simple investigations will probably reveal the reason. One is to check the circuit breakers; the other is to check the lamps in the particular light. Circuit breaker panels in the F-102A are located along the sides of the cockpit, in the nose wheel well, main wheel well, and in the air conditioning compartment. If a circuit breaker will not remain in when power is on the system, you should make a continuity check for a short circuit. However, it is possible for a surge of power to burn out both lamps in a warning light unit without "popping" the circuit breaker. It's easy to check these lamps—just pull the light cap out from the panel the same way you would pull out an ordinary electrical plug. When you remove a light from the warning indicator panel the lamps come with it. The large light caps come out separately, exposing the light units. Don't worry about getting them mixed up; each one will only fit in its proper place. In addition, the light compartments on the warning indicator panel are all numbered.

If the lamp and circuit breaker check doesn't locate the cause of your trouble, there is obviously no current flowing through the system. Either the circuit is not getting power or it isn't grounded. The most probable

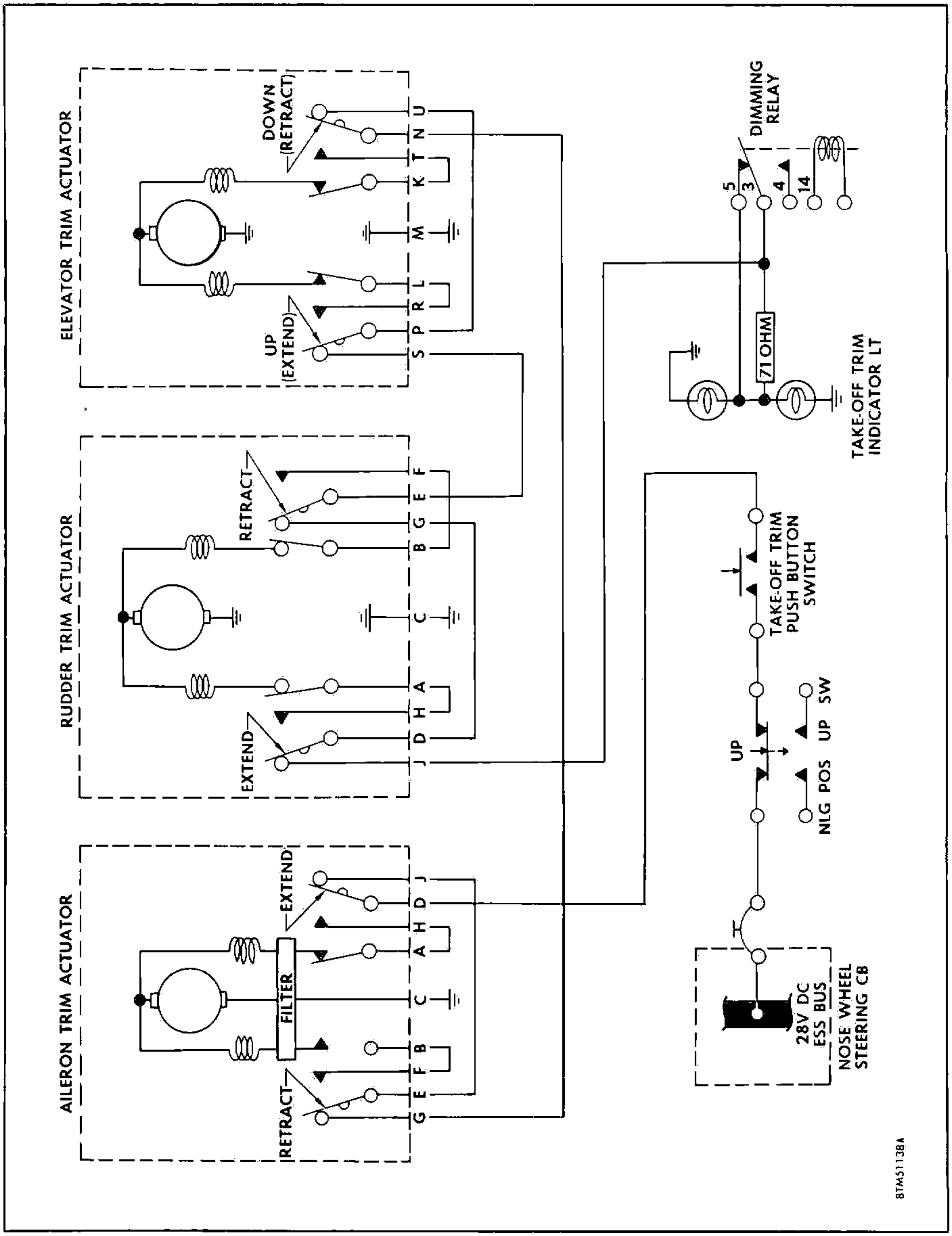

Figure 6-22. Takeoff Trim Indicator Light System Schematic

cause is a faulty switching unit in the system. Regardless of the kind of switch—whether a pressure switch, snap switch, or other type—it should be replaced rather than repaired.

In the event that a warning light illuminates when it shouldn't, it's obvious that something is causing the circuit to be energized at the wrong time. Most likely the switching device is faulty and will have to be replaced.

Erratic blinking of a warning light, (not to be confused with the steady blinking of the hydraulic pressure low warning light or fire-overheat warning light, which have flasher units), is caused by interruptions in the current flow. This usually calls for a careful continuity check to determine if there are any loose connections in the circuit. If the light blinks when it should be off, the system may be shorted.

A discussion of maintenance or trouble shooting of the warning systems would not be complete without considering the master warning box. Since this box is a common mating point for all the systems, it can cause trouble in all systems. You should suspect that it is the source of trouble if more than one light on the warning indicator panel malfunctions at the same time. The separate warning systems only tie into the warning box for brightness control. Therefore, a malfunction of the box will not cause complete failure of these systems. In any event, all malfunctions involving the master warning box should be checked out according to your F-102A electrical maintenance manual.

SUMMARY.

If you have read this training manual carefully and "digested" it thoroughly, you have acquired a good background in the fundamental principles of aircraft instruments. This background should help you to trouble shoot and maintain aircraft instruments and the systems with which they are involved. In addition, you are now particularly well grounded in the instruments used on the F-102A airplanes. Remember, though, that we have discussed the instrumentation which is used in these airplanes at this time. Some of these instruments may change in the future, and so will the F-102A airplanes. This fact should not be surprising to you if you have been in the service for very long. Your "jet age" Air Force is and must always be dynamic and changing. Today's most modern equipment may be totally obsolete tomorrow; however, the more you know about your present equipment the easier it will be to understand future developments.

Although this training manual will not always reflect the latest configuration of the F-102A, or of its accessories, remember that your technical orders will reflect them. They are revised periodically and can be relied upon for up-to-date information, but tech orders are necessarily very brief and short on reasons *why* an instrument operates the way it does. For that reason, you will occasionally want to refresh your memory on the purpose and the basic principles of operation of certain instruments. Herein lies the value of this training manual. Use it as a supplement to—not a substitute for—your technical orders.

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ALPHABETICAL INDEX

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F-102A

MAINTENANCE TRAINING SUPPLEMENT

(EXPERIMENTAL)

To Be Used in Conjunction With

T.O. 1F-102A-2-10

T.O. 1F-102A-2-13

ELECTRICAL SYSTEM

CONVAIR
F-102A

MAINTENANCE TRAINING
SUPPLEMENT

ELECTRICAL SYSTEM

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE

AIR FORCE—AFFS—SAN ANTONIO, TEXAS—5-57-2600

1 JULY 1956

Foreword

The F-102A Training Supplements have been prepared by Convair, A Division of General Dynamics Corporation, to provide you, the system mechanic or maintenance technician, with basic information concerning the F-102A airplane and its operating systems. There are ten of these supplements, each covering an airplane operating system or major component. The text is direct and informal and is supplemented with illustrations and diagrams to aid you in understanding how the systems operate. The following is a list of the Training Supplements on the F-102A airplane:

Title

Flight Control System
Hydraulic System
High-Pressure Pneumatic System
Low-Pressure Pneumatic System
Airplane General
Airframe Fuel System
Power Plant Installation
Airframe Armament System
Electrical System
Instruments

Each supplement describes an operating system or major component, how and why it operates, and maintenance problems that you may encounter. The entire major system is further simplified by describing each of its subsystems separately. Thus, when you understand how all of the subsystems operate, you will be familiar with the functions of the major system. This knowledge of the airplane systems will enable you to use other operational, service, and maintenance instructions.

Since the purpose of these publications is to familiarize and acquaint you with the airplane systems and components, specific values, measurements, and tolerances are not given. Any values that appear in these supplements are approximate only and are used to emphasize more clearly a certain operation or condition. You must refer to your 1F-102A-2-10 and -2-13 Technical Orders and other pertinent handbooks for specific values when adjusting or checking a system and its components. These Technical Orders are revised periodically to include the latest maintenance procedures and data on the equipment installed in the F-102A.

The F-102A Maintenance Technical Orders consist of the following handbooks:

T.O. 1F-102A-2-1	**General Airplane**
T.O. 1F-102A-2-2	**Ground Handling, Servicing, and Airframe Group Maintenance**
T.O. 1F-102A-2-3	**Hydraulic and Pneumatic Power Systems**
T.O. 1F-102A-2-4	**Power Plant**
T.O. 1F-102A-2-5	**Fuel Supply System**
T.O. 1F-102A-2-6	**Air Conditioning, Pressurization, and Anti-Icing Systems**
T.O. 1F-102A-2-7	**Flight Control Systems**
T.O. 1F-102A-2-8	**Landing Gear**
T.O. 1F-102A-2-9	**Instruments**
T.O. 1F-102A-2-10	**Electrical Systems**
T.O. 1F-102A-2-11	**Radio-Communication and Navigation Systems**
T.O. 1F-102A-2-12	**Armament and Armament Electronics**
T.O. 1F-102A-2-13	**Wiring Data (F-102A)**
T.O. 1F-102A-2-13A	**Wiring Data (TF-102A)**

The Air Force is vitally interested in seeing that its equipment functions properly and is maintained as efficiently as possible. The Air Force, like a manufacturer of an automobile or commercial appliance, provides printed instructions for use with its equipment. The printed instructions you will use most frequently in maintaining the F-102A are the dash-2 Technical Orders listed above. They are available for your reference at the Maintenance Office on your base. You must refer to them frequently so that you will always have the latest maintenance information. The Training Supplements will help you to use these Technical Orders by answering many of the "hows" and "whys" that may puzzle you from time to time.

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Chapter III—F-102A D-C Electrical Power System

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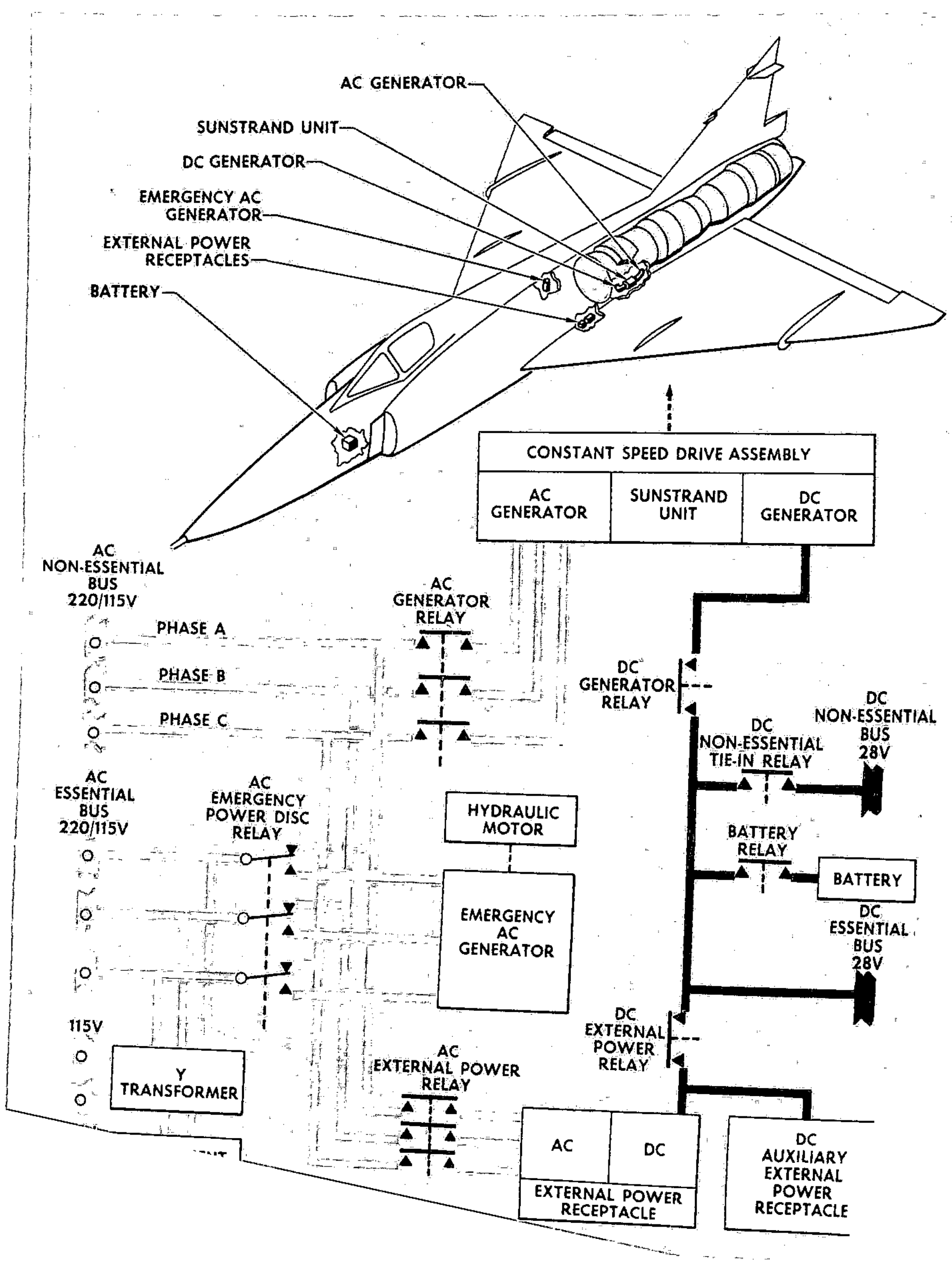
Introduction

This supplement consists of five chapters which cover the F-102A Electrical System. Chapters I and II discuss the sources of electrical power and how electrical power is distributed and controlled. Chapter III describes the D-C Power System and its components; Chapter IV describes the A-C Power System and its components. Chapter V presents the aircraft lighting systems and a circuit analysis of the airplane operating systems. A summary concludes this supplement.

F-102A MAINTENANCE TRAINING SUPPLEMENT



Foreword

The F-102A Training Supplements have been prepared by Convair, A Division of General Dynamics Corporation, to provide you, the system mechanic or maintenance technician, with basic information concerning the F-102A airplane and its operating systems. There are ten of these supplements, each covering an airplane operating system or major component. The text is direct and informal and is supplemented with illustrations and diagrams to aid you in understanding how the systems operate. The following is a list of the Training Supplements on the F-102A airplane:

Title

Flight Control System
Hydraulic System
High-Pressure Pneumatic System
Low-Pressure Pneumatic System
Airplane General
Airframe Fuel System
Power Plant Installation
Airframe Armament System
Electrical System
Instruments

Each supplement describes an operating system or major component, how and why it operates, and maintenance problems that you may encounter. The entire major system is further simplified by describing each of its subsystems separately. Thus, when you understand how all of the subsystems operate, you will be familiar with the functions of the major system. This knowledge of the airplane systems will enable you to use other operational, service, and maintenance instructions.

Since the purpose of these publications is to familiarize and acquaint you with the airplane systems and components, specific values, measurements, and tolerances are not given. Any values that appear in these supplements are approximate only and are used to emphasize more clearly a certain operation or condition. You must refer to your 1F-102A-2-10 and -2-13 Technical Orders and other pertinent handbooks for specific values when adjusting or checking a system and its components. These Technical Orders are revised periodically to include the latest maintenance procedures and data on the equipment installed in the F-102A.

The F-102A Maintenance Technical Orders consist of the following handbooks:

T.O. 1F-102A-2-1	General Airplane
T.O. 1F-102A-2-2	Ground Handling, Servicing, and Airframe Group Maintenance
T.O. 1F-102A-2-3	Hydraulic and Pneumatic Power Systems
T.O. 1F-102A-2-4	Power Plant
T.O. 1F-102A-2-5	Fuel Supply System
T.O. 1F-102A-2-6	Air Conditioning, Pressurization, and Anti-Icing Systems
T.O. 1F-102A-2-7	Flight Control Systems
T.O. 1F-102A-2-8	Landing Gear
T.O. 1F-102A-2-9	Instruments
T.O. 1F-102A-2-10	Electrical Systems
T.O. 1F-102A-2-11	Radio-Communication and Navigation Systems
T.O. 1F-102A-2-12	Armament and Armament Electronics
T.O. 1F-102A-2-13	Wiring Data (F-102A)
T.O. 1F-102A-2-13A	Wiring Data (TF-102A)

The Air Force is vitally interested in seeing that its equipment functions properly and is maintained as efficiently as possible. The Air Force, like a manufacturer of an automobile or commercial appliance, provides printed instructions for use with its equipment. The printed instructions you will use most frequently in maintaining the F-102A are the dash-2 Technical Orders listed above. They are available for your reference at the Maintenance Office on your base. You must refer to them frequently so that you will always have the latest maintenance information. The Training Supplements will help you to use these Technical Orders by answering many of the "hows" and "whys" that may puzzle you from time to time.

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Introduction

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Chapter I

SOURCES OF ELECTRICAL POWER

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In this day of complex weapon systems and high speed, high altitude aircraft, our concern is not so much with what an electrical component *is,* but what it *does.* This was also the primary concern of the electrical engineer who developed the components and the circuits that control them. It should be yours. You are responsible for the proper function and maintenance of all electrical equipment and systems installed in the F-102A interceptor. On your electrical knowledge depends the successful completion of a mission and the safety of the pilot and the airplane itself.

ELECTRICAL KNOWLEDGE.

Without knowledge there is little progress. It took knowledge to construct the atom bomb. It took knowledge to build the F-102A. It took knowledge to conceive a magnetic amplifier which automatically governs alternator output. By the same token, it takes knowledge to govern the individual's output.

More often than not most of us are too prone to confuse *knowledge* with *familiarity.* We develop this false sense of knowledge because of everyday association, because we can with pride identify and even classify this gadget or that. But when asked, "What does it do?" or "Why does it do it?" we find ourselves at a loss for a satisfactory answer. In like manner, in the maintenance of electrical systems and their components we often know what should be done but the HOW TO and WHY TO is not always clear.

EXACT KNOWLEDGE IS ESSENTIAL.

Let's face it! New developments in aircraft make new demands on men. The purpose of the F-102A interceptor is to act as this country's first line of defense against all air attacks. Radar warning nets will detect, and the F-102A will intercept and destroy any hostile aircraft attacking this country. What does this mean to you? It depends on you! The F-102A will bear the brunt of any attack and must be in a constant state of readiness if it is to perform its mission with maximum efficiency. If you are to maintain and support this all important task, exact knowledge of the electrical power systems installed in the F-102A is essential. This manual will give you everything that you need to know of the F-102A electrical systems and their components, and why they do what they do.

ELECTRICAL SYSTEMS OF THE F-102A.

We all know that cold facts are about as interesting as a corpse; we know also that when we find out the living nature of the particular corpse our disinterest is usually displaced by interest. The living nature of an object is the facts that make that object live. The electrical systems in the F-102A are what bring this present object of our attention to life.

The F-102A is an arsenal on wings. Underneath its sleek skin are the electrical nerves and muscles that control its operation. The F-102A has an electrical and

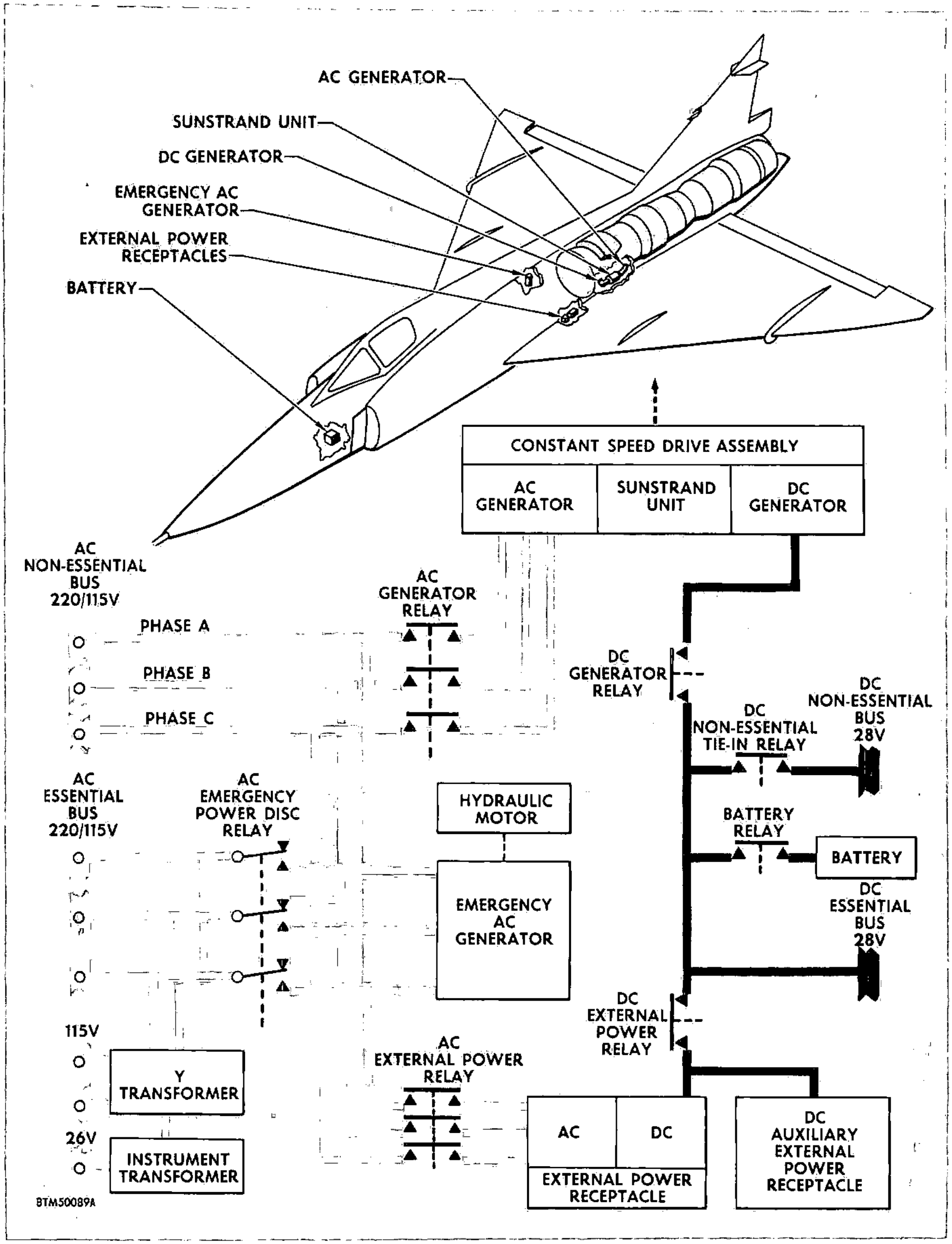

Figure 1-1. F-102A D-C and A-C Electrical Power Systems

electronic system comparable to an intricate IBM machine. It has electrical wiring sufficient to wire 15 modern houses, and this does not include its electronic network. The cubic-foot displacement of its electrical, as well as its electronic equipment, would fill a good sized truck. Let's learn some additional facts.

Further along in our text the d-c and a-c power systems will be discussed at length, so for the sake of familiarization and simplicity: the F-102A interceptor has a 28-volt d-c power system and 115/200-volt, three-phase, 400-cycle a-c power system coupled with a 26-volt, a-c, single-phase, regulated system. Both d-c and a-c system generators are driven by a Sunstrand constant-speed drive unit in the engine accessory compartment. The front end of this unit provides output for the d-c generator which varies with the engine speed. A constant-speed output powers the a-c generator on the other end. The drive unit is driven from a gear box mounted on the engine. The constant-speed drive assembly consists of the drive unit, gear box, and the a-c and d-c generators.

D-C POWER SYSTEM.

The d-c power system is a conventional 28-volt d-c single wire ground return system. If you will recall, single wire ground return systems save considerable material and weight, only one wire being used to connect an electrical device to the power source. The circuit is completed to the battery or generator through the metallic structure of the airplane.

D-C Power Sources.

There are three possible power sources for the d-c system. During in-flight operations all d-c power is supplied to the essential and non-essential buses by a 30-volt, 200-ampere generator. To avoid confusion, let's pause for a minute to clear up a point which might throw some of you off base. The d-c electrical system in the airplane is a 28-volt system, and the power is supplied by a 30-volt generator. This simply means that the maximum output capacity of the generator is 30 volts and has been adjusted to the systems requirements of 28 volts at the time of installation. Any further adjustments of the voltage regulator will depend upon the operating area of the airplane. We will cover this in a more suitable place.

The second source of d-c power is a 24-volt, 24-ampere-hour battery. It furnishes power for emergency operations while in flight and can be used for limited ground operation.

The third source is external ground power. For the convenience of ground handling, a main d-c external power receptacle is located in the left main wheel well on the F-102A. Since operational tests may be made on the landing gear at times, another d-c external power receptacle is installed in the aft electronics compartment.

D-C Power Control.

The generator-battery d-c power system is controlled by a master switch on the right-hand console next to the pilot. When external d-c power is being used, this switch is normally OFF—the external power being controlled from the d-c ground support equipment cart.

A-C POWER SYSTEM.

In this system, the a-c generator provides 115/200-volt, three-phase power to essential and non-essential buses. Two step-down transformers regulate the 26-volt and the 115-volt supply during both emergency and normal operation.

You will remember that all airplanes equipped with radio and radar equipment, electronic gages and regulators, and electrically controlled flight and remote control instrument systems must either convert 28-volt d-c electricity to higher voltages or manufacture a-c electrical energy to the correct voltage and frequency. For this reason airplanes boasting high performance, sonic speeds, and the latest equipment are equipped with two or more single or three-phase inverters, with dynamotors, or with alternators.

While we are on this subject it might be well to refresh your memory on the functions of these components. You learned in electrical school that inverters and dynamotors are not considered sources and generators of electricity. This is because they are driven by a primary source of power and merely convert electrical energy from a battery and generator system to a higher voltage—the dynamotor to d-c, the inverter to a-c. An alternator, as the name implies, is simply a large a-c generator. It is mechanically driven direct from the engine through some extended linkage or flexible drive and is a prime source of a-c electricity. In the F-102A the alternator is driven by a Sunstrand constant-speed drive unit. In this manual we will refer to the alternator as an a-c generator.

A-C Power Source.

In the a-c system of the F-102A, the a-c generator for normal in-flight operation is a 30-kva, 120/208-volt, 400-cycle, air-cooled generator. It provides 115/200-volt, three-phase power. A portion of this power is stepped down by a transformer to power the 26-volt, single-phase system. This 26-volt system is required by certain instruments, such as the primary and secondary hydraulic system pressure transmitters, the fuel flow indicator, and others. Another transformer in the system steps down the voltage to 115 volts, for phases "B" and "C" of the three-phase power. You will be given a more detailed description of this wye-to-delta transformer and its function within the system when we discuss the a-c system in Chapter IV.

As in the d-c system, the a-c system in the F-102A has three possible sources of power. We have discussed briefly the first of three—normal in-flight operation. The second source is a 1-kva, 120/208-volt, three-phase, 400-cycle, air-cooled a-c generator for emergency use. This generator is driven by a hydraulic motor powered by the secondary hydraulic system. The third source of power is received from the a-c side of the external ground power cart. The a-c external power receptacle is located alongside the d-c receptacle in the left-hand main wheel well.

A-C Power Control.

During both normal and emergency operation the a-c power systems are controlled by the a-c generator and a-c bus switches. These switches are on the electrical control switch panel above the d-c power control switches and the master switch. This panel is situated on the right-hand console in the cockpit. The a-c bus switch selects the power source to be connected to the distribution buses. During normal in-flight conditions, this switch is in the NORM position and a-c power is supplied to all the a-c buses. When the a-c bus switch is in its EMERGENCY position, the hydraulically driven 1-kva a-c generator supplies power only to the essential a-c buses.

MASTER SWITCH.

In both the d-c and a-c systems power is normally distributed through the master switch to the essential and non-essential buses. But when it is necessary to select emergency power, the non-essential buses of both systems are deenergized. The master switch is provided to deenergize all buses regardless of the position of any other switch. In Chapter III, you will learn that the master switch must be in the NORMAL position to energize any of the distribution buses in either system. *When the master switch is OFF, all circuits in the airplane are deenergized, including all essential buses.*

This short resume of the cold facts regarding the electrical systems in the F-102A should give you a good idea as to what we will be talking about throughout this manual. The electrical power systems installed in the airplane are not too complicated and are easily understood when your knowledge of *why a component does what it does* is applied. At the beginning of this discussion we stated that familiarity is often confused with workable knowledge. This is a confusion to be found in all of us; and so that this *false sense of knowledge* may not interfere with our efficiency to do the job demanded of us, let's turn back to the building-block stages of electrical power systems.

THREE BRANCHES OF ELECTRICITY.

A cold plunge into the study of an electrical power system leaves too much unanswered. It's like setting off a blast without knowing how it was triggered or the end result. So, before taking the plunge into the building-block stage of electrical power systems in general, it might be a good idea to refamiliarize ourselves with the classes of electricity. It might be a good thing, also, to remember that it is impossible to neglect any of the following branches of electricity without harming your efficiency. Bearing this in mind, let's divide electricity into three branches: electrostatics, electrodynamics, and magnetism. When these have been discussed we will review their prime mover—electromotive force.

ELECTROSTATICS.

Electrostatics is simply static electricity. Originally, static electricity was considered electricity at rest, but with the acceptance of the electron theory—even though static electricity may be temporarily stationary—it is far from inactive. Static electricity builds up mainly through friction and continues to accumulate because it has no particular place to go. When it builds to abnormal proportions and is attracted by a like accumulation, it becomes a serious hazard to any object that has the power to attract it. You may recall, from your study of basic electricity, that this power of attraction is characteristic of electrified objects and that all *matter* is electrical in nature.

Those Free Electrons.

The F-102A airplane that you maintain, the air it flies through, your body, the body of the pilot, and the airplane landing strip all constitute *matter*. In other words, *matter* is a body or substance having weight and occupying space. It is also anything that offers resistance to change in its condition of rest or motion. And when we have resistance to change we have friction and a resulting build-up of *free electrons*—electricity's most important product.

Those Two Opposite Kinds of Electricity.

Sometimes an object accumulates a large number of these free eletrons and cannot get rid of them very easily either because it is not a conductor or, because even though it is a conductor there is no other object nearby to conduct this overabundant store of electrons. Such an object has a static negative charge of electricity. You may remember from studying ionization that an atom structure that has lost one of its electrons is called a *positive ion*. In like manner, when an object loses an abnormal number of electrons it has a static positive charge of electricity. You may remember also that when we speak of "positive charges" or "negative charges" we are talking about two opposite kinds of electricity, and the first hidebound law of electrostatics is: *like charges repel each other; unlike charges attract each other*. So, when objects with a positive or negative charge of electricity come close to another object to which they can release electrons, or from which they can gather electrons, a spark is produced. As these electrons are attracted to one another, they jump from one object to the other.

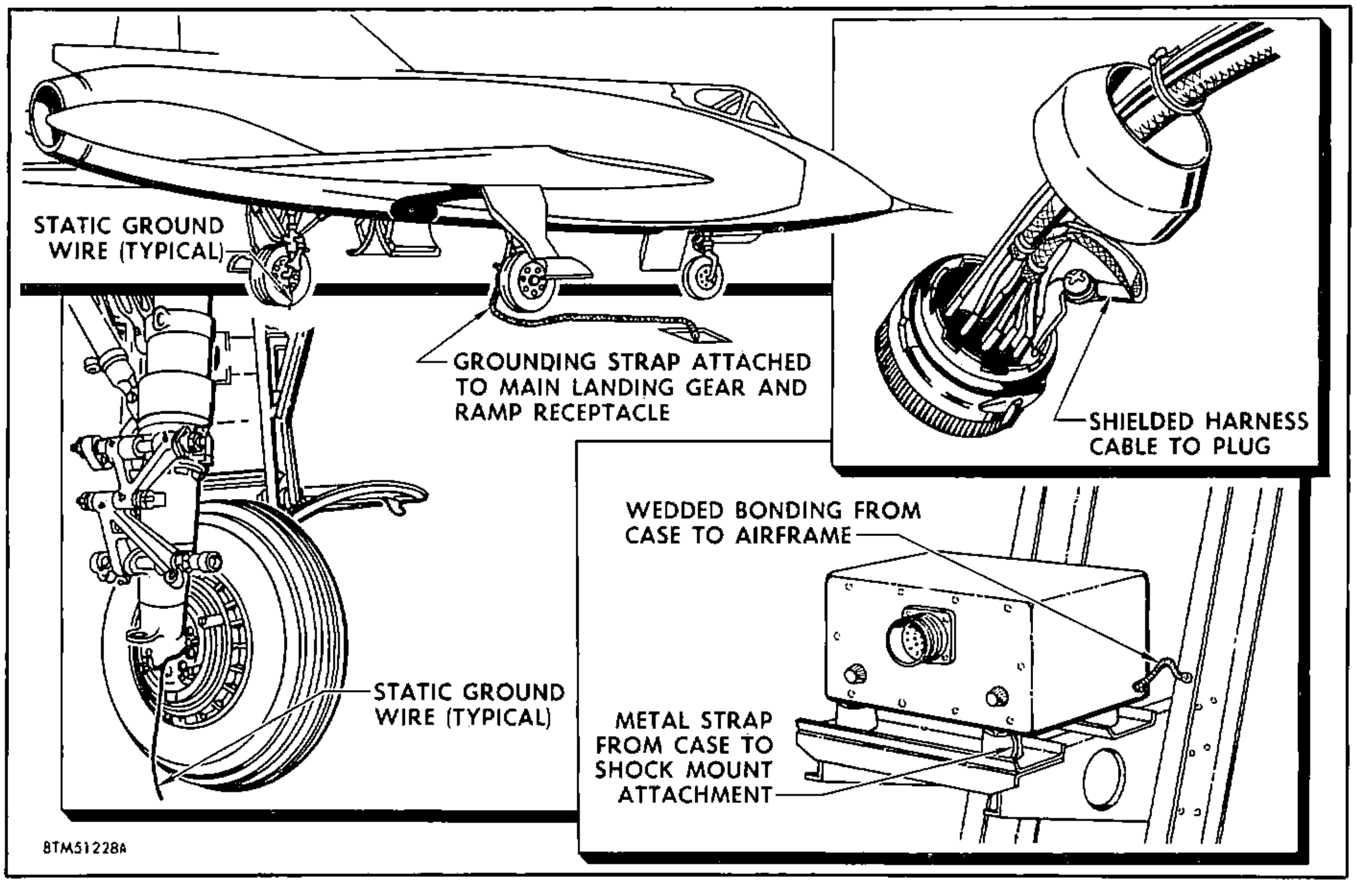

Figure 1-2. F-102A Bonding and Grounding Devices

For example, static electricity is generated when a person slides from or into the upholstered seat of an automobile, especially on a hot, dry day. Let's say your wife has driven to the Air Force base where you are stationed, you reach over as you slide behind the wheel to kiss her, a spark passes between you—this, friend, is static electricity. If, however, either of you were touching the metal of the car, there would be no spark—the charge has flowed back into the frame of the automobile. Electrically speaking the positive and negative charges have intermingled in equal quantities and produced a neutral charge.

Static Electricity and the Airplane.

Static electricity, as you know, has been in the past and still is a serious problem with aircraft. Free electrons may accumulate on an airplane while it is in flight and a heavy static charge may remain on the airplane after it has landed. Most airplanes are provided with static dischargers which dump most of the static build-up back into the airstream. They are likewise equipped with ground wires—like a gasoline truck or a truck carrying high explosives—which drag the ground on touchdown, discharging the excess static electricity.

Bonding.

Another anti-static device is the method of bonding. In all probability you are familiar with bonding, but here again let's examine it in its relationship to static electricity. Did you know that bonding was provided as a means of equalizing the potential of the airplane with that of the earth on touchdown, and keeping them the same until scramble? You see, *bonding* is the process of connecting *all* of the component metallic parts of the airplane until they collectively form a gathered electrical mass. If any two parts do not touch each other metallically, a piece of bonding braid must be attached across them (as shown in the detail of figure 1-2). Any electrical mass has an electrical potential. The earth is at zero potential electrically, so when a bonded airplane lands the static ground wire equalizes the electrical potential of all the various components in the airplane making any part of it safe for such processes as refueling and rearming.

The F-102A.

The F-102A is heavily bonded and is equipped with three static ground wires in the form of metal probes, much the same as those attached to an automobile for parking. The static ground wires on the F-102A are attached to the landing gear and make contact with the zero potential of the earth's surface just before the plane touches the landing strip. In the illustration figure 1-2, you can see the grounding probes. Without such necessary precautions, a static discharge of electricity during refueling might result in a fatal explosion. So much for electrostatics.

ELECTRODYNAMICS.

Electrodynamics, or dynamic electricity, is the subject with which we are primarily concerned when studying the electrical circuits of the F-102A. But, dynamic electricity and magnetism are so closely related that we can hardly understand one without an understanding of the other. When discussing electrostatics, we said that *matter* is a body or a substance which offers resistance to change. Well, when matter undergoes a change the result is *energy*, and *energy* as you know is the capacity for doing work.

Energy and Work.

Dynamic electricity cannot be fully understood unless we have knowledge of how the electrical nature of things is put to work. You will recall in electrical fundamentals, or in physics in high school, that the work required to overcome resistance is usually wasted as heat; but that the work done, in giving a substance speed or in changing its position or condition, is stored up in the system as energy and may be recovered.

Four Kinds of Energy.

For the sake of review, energy may be classified into at least four kinds, according to the composition of matter:

MATTER	ENERGY
Substance or body	Mechanical
Molecule	Heat (Thermal)
Atom	Chemical
Electron	Electrical

Mechanically speaking, energy may be either *potential* or *kinetic*. Potential energy is energy that a substance has due to position or condition. Kinetic energy is the energy a substance has due to its motion. The wound spring in a clock, a raised hammer, and a rock resting on a hillside all possess potential energy. It is only when the clock spring unwinds and drives the gears, the hammer falls and drives the nail, and the rock crashes down the hillside that the potential energy is transformed into kinetic energy.

Briefly, let's say a dam is constructed in the state of Washington, Tennessee, or Arizona. It is filled by streams and springs. Once full and at rest, it is potential energy—a reservoir of energy. When released, the flow is kinetic energy. This also illustrates how one form of energy may be converted into another; the dam was filled through kinetic energy but once stored it became potential energy.

ENERGY'S LAW. An important law to remember, when thinking in terms of energy, is that *energy can neither be created* (it's already there) *nor destroyed*—the total amount of energy in the Universe, regardless of its form, always remains constant. We simply convert one kind to another, or in some cases back to its original form. A good example of this would be in the steps taken to energize a dynamo, if driven by a steam engine.

THAT BIN OF COAL. First, we must have a bin of coal—a bin of stored *chemical energy*. The moment we throw some of this coal into the steam engine's firebox and ignite it, we have *thermal or heat energy*, which drives the engine and gives us *mechanical energy*. Mechanical energy drives the dynamo, converting it into *electrical energy*. If the direction of current derived from the dynamo flows through a resistance coil, the electrical energy is reconverted into *heat energy*.

The time it took for each conversion to happen is called *power*. Energy is a commodity which can be bought and sold; power is simply the speed of the transaction. In the above process of buying one form of energy to sell to another, a certain amount of heat escaped during the transaction into the surrounding atmosphere, but *none of the energy was actually destroyed*.

Electrical Energy.

Unlike other forms of energy, electrical energy is actually a negligible natural resource. Not only does it have to be obtained largely by conversion from the other three forms, but also, it is of little use as such and must be reconverted before use.

Now, why should this double conversion be employed more and more, year after year, instead of the direct application of electrical energy as found in nature? The answer to this is comparatively simple. For example, compare the qualities of electrical energy with a sound currency like our own monetary system; both are highly *convertible, conveyable,* and *controllable*.

THOSE FREE ELECTRONS. As you know, a molecule is a combination of two or more atoms. But perhaps what is not so clear is that the elements in the make-up of the structures of those atoms are joined together by the interaction of their free electrons as they move about the nucleus—a solar system in miniature. These electrons are the smallest particles of negatively charged electricity known to science. The fields of force made by them is the basic idea behind magnetism, or for that matter, electricity itself.

These free electrons move about the nucleus at a high rate of speed, somewhere in the neighborhood of 40,000 miles per second under ordinary temperatures. Some of them are only loosely held by the nucleus,

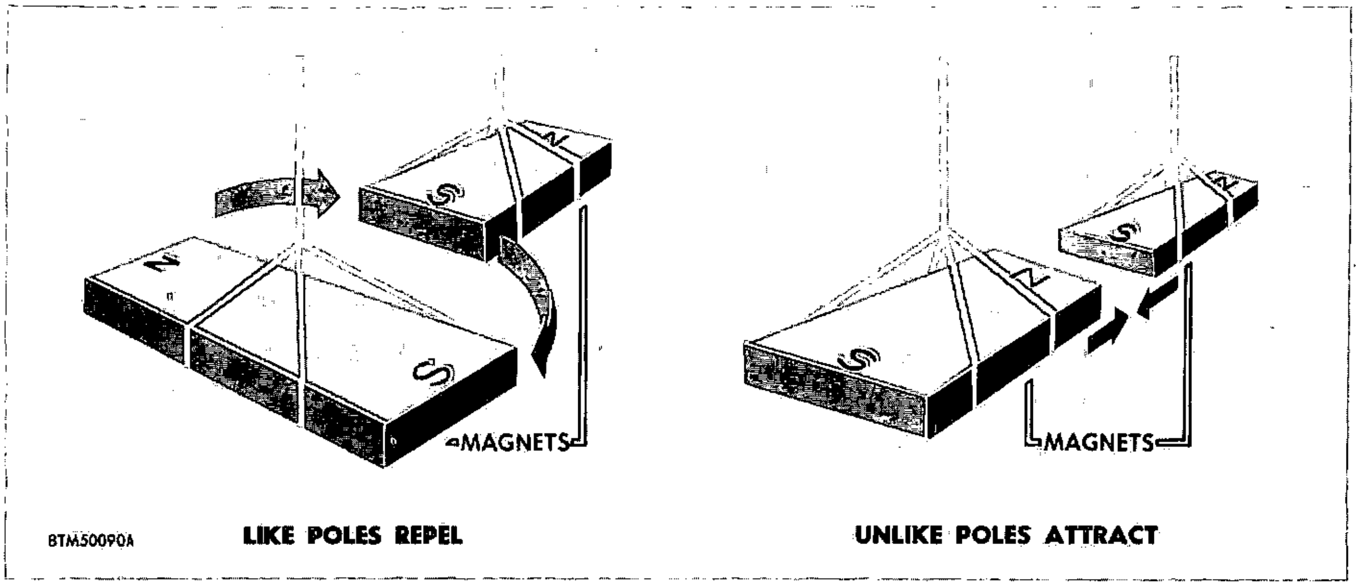

Figure 1-3. Like Poles Repel—Unlike Poles Attract

and under sustained energy and force break away to join another atom structure when electrical pressure is applied. The resulting flow of electrons is dynamic electricity, or *electrical current.*

ELECTRON MOVING FORCE. If work, as the result of sustained energy or the force used to overcome an obstacle, produces movement or flow, it follows that work results when electrons are added to an inert atom —as in electrostatics—to give it a negative charge, or subtracted to give it a positive charge. Charging a substance stores electrical energy; discharging a substance releases it.

Also, work is performed whenever electrons flow from a conductor to a positively charged substance, or from a negatively charged substance to a conductor. So, since charging a substance gives it work capacity electrically, in like manner discharging a substance releases its work capacity electrically. All this is developed through an external force being applied—the battery, or the generator. It is this same external force or pressure which causes electrons to flow in a definite direction through conductors and electrical components. This electron-moving force is familiarly known as electromotive force (emf), voltage, or difference of potential—all three have the identical meaning.

MAGNETISM.

You are aware that energy, for the operation of most electrically operated equipment in an airplane, depends on electrical energy supplied by a generator. A *generator* is nothing more than a machine used to convert mechanical energy into electrical energy by electromagnetic induction. It is common knowledge to those of us who have studied even the most basic electricity that practically all electrical apparatus, from electrical doorbells to generators, depend on the action of magnetic fields.

Molecular Theory of Magnetism.

When we discussed electrostatics you were reminded that like charges repel each other while unlike charges attract. Magnetically speaking, if two bar magnets are suspended at their midpoints by a string, and are free to move as shown on figure 1-3, their like poles repel each other and their unlike poles attract each other. Just as the earth itself is a large magnet, having a north pole and a south pole, every molecule of matter is also a tiny magnet, having a north and a south pole. This phenomenon is due to the action of the atom structures within the molecules which set up tiny individual fields of force. But the magnetic properties in some molecules are stronger than in others. Let's take a look at the arrangement of molecules in a piece of unmagnetized iron and those in magnetized iron. If all the molecules in the unmagnetized bar, as shown in figure 1-4, were aligned in such a manner that all the north poles faced in the same direction as they do in the magnetized bar, the iron would be a magnet. This is the condition found naturally in the loadstone in which the original phenomenon of magnetism was first observed.

Magnetic Lines of Force.

A convenient way to regard lines of force is to think of them as taut rubber bands which in trying to shorten themselves establish a stress field of mutual repulsion, against whatever holds them apart and their normal state. In like manner, when two magnets are brought together their fields of stress interact causing repulsion or attraction. What is a stress field? If you place a sheet of glass or just a simple piece of paper over a magnet and sprinkle iron filings on it, the pattern of a magnetic field can be readily seen. When you tap the glass gently the filings will arrange themselves along the lines of force. For this reason, at any particular point in the space around the magnet there is a state of stress which exerts force on any pole brought into the vicinity of the magnet.

Lines of force have a definite direction. They leave their *north pole* and re-enter their *south pole* and continue on their journey through the magnet from south pole to north pole. Notice in figure 1-5 the pattern taken by these lines of force under this law of polarity. In the upper left corner of the illustration are shown the magnetic fields of force about a bar magnet and the continuous paths taken by these fields of force as they leave the north pole and re-enter the south pole. Below this bar magnet are the magnetic fields about a U-shaped or horseshoe magnet. In figure 1-5, you can see how these magnetic fields of force have been harnessed for work in a simple two-pole and four-pole magneto. The permanent magnet in these simple magnetos acts as a keeper. The *keeper* concentrates the magnetic fields into a more orderly stream and thereby strengthens the field. This is a *closed magnetic circuit.*

Terminology.

Various technical terminologies are used by different authorities to distinguish the lines of force inside and outside the magnet. These have been devised to explain the fine details of magnetism. For our purpose, however, let's stick to the more simple term, magnetic lines of force. Later, when we cover the action of the generator and when magnetic flux is mentioned, remember that magnetic lines of force and magnetic flux are interchangeable—*both have the same meaning.* And furthermore, let's remember that the entire space around a magnet is the magnetic field sometimes referred to as a *field of stress.*

Laws of Magnetism.

Because lines of force, magnetic fields, and electromagnetism will be reviewed before we discuss generators, certain laws governing magnetism will be quoted here. These laws might be good to keep in mind while we move right into the production of electromotive force. You will find them applicable to almost any study or understanding of what an electrical component does and why it does it.

First of all, magnetic lines of force are continuous and *always form closed loops.* Magnetic lines of force have a tension (remember our rubber bands), along the direction of the lines, which tends to shorten them. As another example, when two unlike poles are brought near each other, the lines of force existing between them cause them to move toward one another. Magnetic lines of force never cross one another—they are conducted by all materials. And last but not least, magnetic

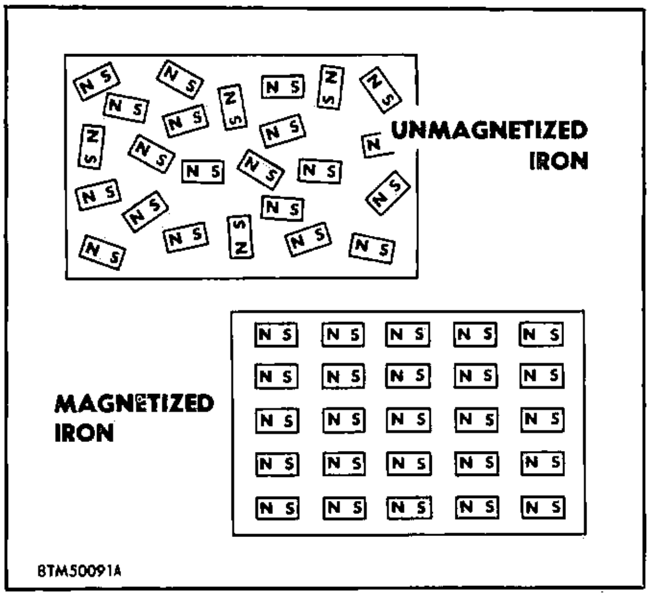

Figure 1-4. Arrangement of Molecules in an Iron Bar

lines of force that have the same direction, like those that materialize from the same poles, tend to push one another apart. This particular fact accounts for the repulsion between like poles of our two bar magnets when these poles are brought close to each other.

ELECTROMOTIVE FORCE.

All normal atom structures are electrically neutral—the number of positive charges equals the number of negative charges. Since all matter is composed of molecules and their accompanying atoms which in turn are composed of positive and negative electrical charges—it should be clear that all matter is electrical in nature. Now, how is the electrical nature of things put to work? It was brought to your attention (when we were discussing energy and work) that an electron-moving force, or flow of electrons, is what is more familiarly known as electromotive force (emf), voltage, or difference in potential—all three having the same meaning.

METHOD OF PRODUCTION.

The two most common methods of producing electromotive force are by chemical reaction and by mechanical means. The first method involves transforming chemical energy into electrical energy, as in the storage battery or the dry cell. The other method involves rotating conductors of electricity in a magnetic field in such a manner that an electrical voltage is generated. This arrangement, of course, is the generator.

We have said, atoms having equal amounts of positive and negative charges are electrically neutral—they are balanced and in a condition of momentary rest. Like the stored water in that dam, they are simply minute bundles of stored potential energy waiting for release and conversion into electrical energy.

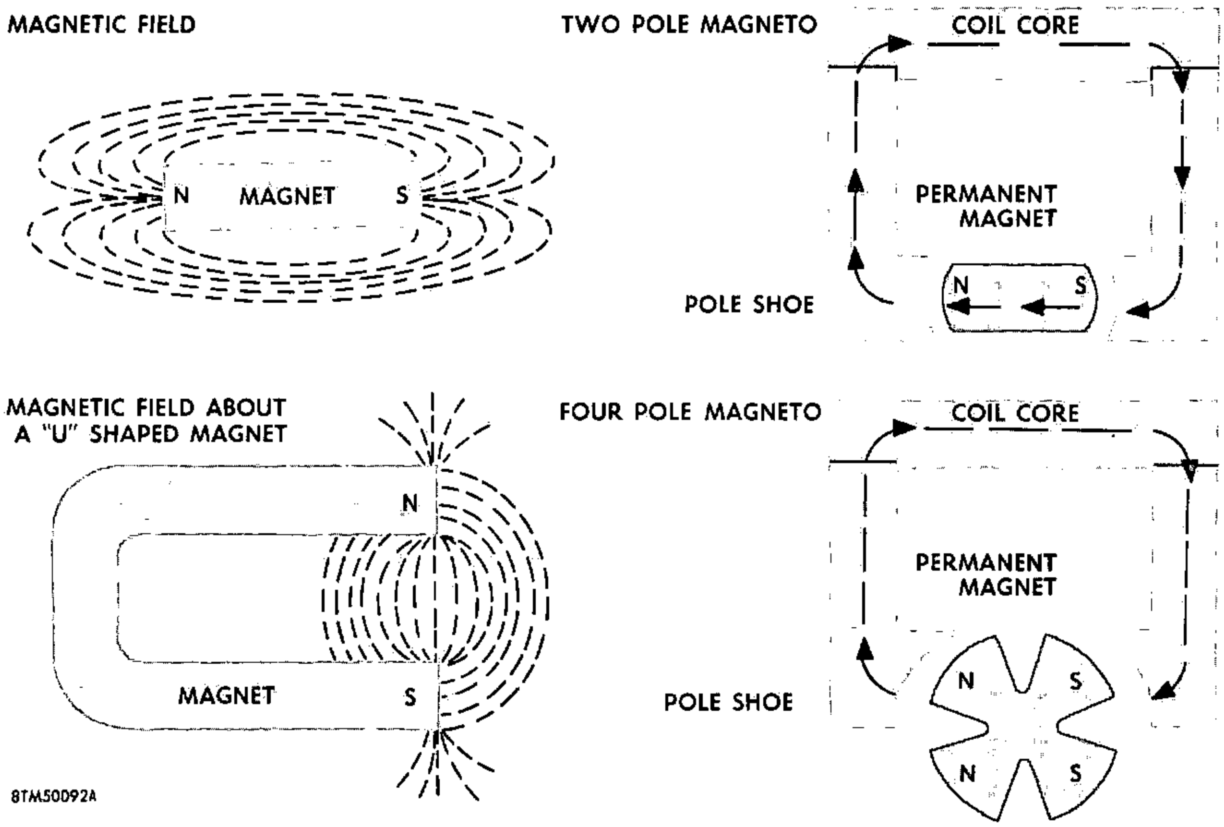

Figure 1-5. Magnetic Fields and Lines of Force

Electromotive force then is the force that releases the atom's free electrons and forces them through conductors and the various devices that form electrical systems. This electron-moving force, this electrical pressure, is measured in *volts.* As it moves through the electrical system, from electrical device to electrical device, it varies in pressure or in electrical potential. The difference in pressure between two points in a stream of water may be called the difference in level, the drop in level, or merely the drop. Similarly, the difference in voltage between two points in an electrical circuit is said to be the *difference in potential,* or the *drop,* such as voltage drop.

CURRENT AND THE ELECTRON FLOW.

The velocity of an object is the speed at which an object moves. If you remark of a river, "It has a swift current," you mean only that the velocity of the water is high. Technically, current measures the time it takes to displace some substance, not the speed at which it moves. Water, for example, is measured in the gallons per minute that flow past a certain spot. In electricity, it is the number of electrons that flow past a particular point in an electrical circuit. The measurement for this is the *ampere;* the measurement for the quantity of electricity is the *coulomb.* If we were to say that current is flowing past a certain point in an electrical circuit at the rate of one coulomb per second, there would be one ampere of current. One ampere of current is equivalent to 6.28 billion billion electrons flowing past a given point per second. (This figure is merely mentioned to make our point—you'll never have to remember it.)

ENTER THE ELECTRONIC ERA.

In other supplements of this series, when we study the flight control and the armament control systems in the F-102A, electronics will and must come into our discussion. Today, anyone studying electricity is sooner or later going to run into electricity's big brother, electronics. So, before we move into a closer study of power sources, let's pause for a minute and go back to the discovery of the electron.

SOME BASIC CONCEPTS OF THE NATURE OF THINGS.

When we were discussing electrodynamics we mentioned matter and defined it. But perhaps at this point we should learn a little more about it. Undoubtedly you remember that *matter* is any substance having weight and occupying *space.* But what is it composed of? You probably learned this in high school, and are familiar with the term, but do you have the knowledge of it?

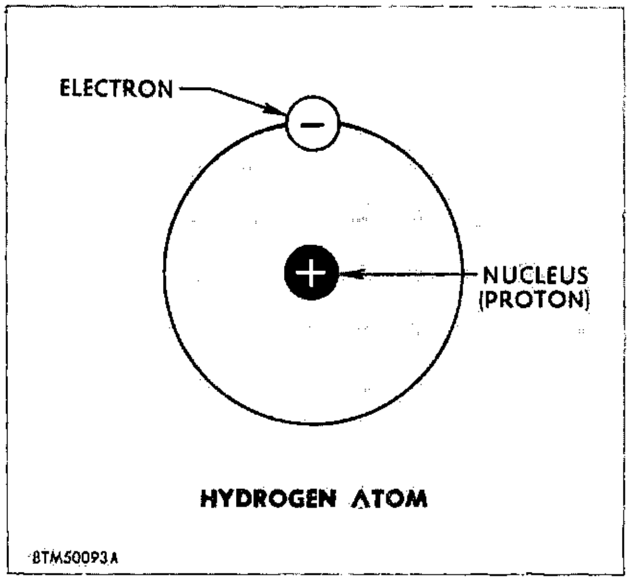

Figure 1-6. The Hydrogen Atom

The Composition of Matter.

Matter consists of 98 or more known elements and appears in three forms: one, a pure elemental substance such as hydrogen; two, it may be compounded to form the chemical unions of two or more elements as in carbon dioxide; and three, it can be a mixture of several elements or compounds not bound by chemical action—an example of this is copper and ground glass (silica). But regardless of the molecular form matter takes, it can always be broken down into atoms and the electron.

Molecules.

A molecule is a grouped combination of atoms. It is the smallest particle into which any substance can be divided and still retain its characteristics. For example, two atoms of hydrogen (H_1) form the molecule H_2 of hydrogen; two atoms of hydrogen and one of oxygen form the molecule H_2O, water. In other words, the molecule of a compound always contains two or more atoms while an element like oxygen consists of but one.

Atoms and Electrons.

Atoms are the smallest divisions of matter that can ordinarily be obtained chemically. As we mentioned earlier, the structure of the atom can be compared to our solar system in miniature. As the planets revolve around the sun, the atom's most important product, the *electron*, revolves around the atom's nucleus (made up of protons and neutrons). And remember, the electron is always the same regardless of its source and is the smallest known particle of negative electricity. Contrary to belief, the electron is *not* a part of its so-called nucleus. It can be compared more to a temporary visitor than to anything else.

Atomic Structures.

The simplest of atomic structures is the hydrogen atom, shown in figure 1-6. This structure consists of one negatively "charged" electron revolving around a positively "charged" proton. It is the number of protons in an atom's nucleus that determines the charge, and it is this number that gives the atom its atomic number. In this case, the atomic number is 1 (H_1). The quantity of electricity, or charge, in both the proton and electron is equal—this atom is therefore neutral.

To go a step further, let's take a look at the structure of the helium atom as illustrated in figure 1-7. Here we have a nucleus which, in its natural state, contains four protons. The atomic number of the helium atom, however, is 2, (H_2), not four. Why? Because, the charge on two of the protons has been neutralized by two electrons held captive by protons in the form of neutrons or *bound electrons*. The resulting positive charge on the protons is just enough to hold the negatively charged *free electrons* on their orbit as they revolve around the nucleus. As in the simpler structure of the hydrogen atom, we again have a resulting quantity of electricity or electrical charge which adds up to zero. The atom is inert, at rest.

What does this really mean to us? Simply that electrification, as we know it, consists not in making or producing electricity, but merely separating the equal charges of opposite kinds of electricity we find present in a neutral substance or body. It's that easy.

THAT MAN THOMPSON.

Until a man named J. J. Thompson discovered the electron in 1897, it was assumed that current flow was

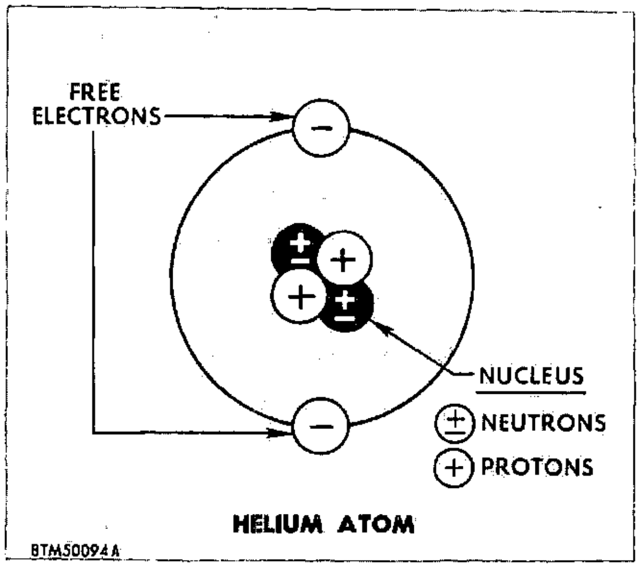

Figure 1-7. The Helium Atom

from (+) positive to (−) negative. Rules of behavior, whether we are talking about electricity or not, become the conventional way of doing or thinking about something. So it was, that this plus to minus thinking became known as the conventional direction of current flow. But, since Thompson discovered the electron, scientists are now agreed that electrons in motion are current, and that current actually flows from a negative to a positive potential. As you undoubtedly learned in electrical school, either the conventional direction of current flow, or the electron theory of current flow, may be used in tracing electrical circuits.

You must remember, however, that the Air Force recognizes only the *Electron Flow Theory.* The reason for this is that electron flow is the only theory applicable to electronics; therefore, for standardization purposes, the Air Force had to adopt it.

CONDUCTORS AND INSULATORS.

The number of free electrons varies with the substance; and the greater the number of free electrons contained in a substance, the less resistance there is to electrical current. Substances characterized by their relatively large content of free electrons are called conductors; those characterized by very few or the lack of free electrons are known as insulators and are used to insulate conductors and keep the electron flow from moving in undesired paths. Common among non-conducting substances are mica, glass, rubber, and bakelite. The materials used to insulate electrical conductors in airplanes are specially treated cambrics and nylons. It might be a good thing to remember, however, that *there is no perfect conductor and no perfect insulator.*

PRODUCTION OF ELECTROMOTIVE FORCE BY CHEMICAL ACTION.

All that we have been talking about thus far is fairly familiar to you. But, because *exact knowledge* is of such great importance to you when dealing with ultramodern aircraft, it is felt there is a need for even such basic concepts as the construction and process of as simple a gadget as the lead-acid storage battery. After all, it is the life-blood of the average automobile — it acts as the d-c blood bank for the modern jet airplane.

Basically, the free electrons we have been talking about are weak—they wander. When this occurs, if you will remember, we have ionization. An atom which has lost one of its electrons is called a positive *ion;* an atom which has gained a wandering electron is called a negative *ion.* This addition and subtraction of the number of free electrons in an atom is the same process that takes place in an aircraft storage battery.

THE STORAGE BATTERY.

The aircraft storage battery is a reservoir in which energy is stored until needed. It is the method of producing electrical energy through chemical reaction. So let's review the electrical auxiliary power source of most airplanes, the storage battery.

The main source of the d-c electrical power systems in airplanes is the direct current generator. The battery, we can say, goes along for the ride and is used as an auxiliary or emergency source of d-c power during in-flight operations. A number of similar articles collected together is termed a battery. Storage batteries consist of a collection of individual cells connected together, thus the word "battery" as used in electricity. Figure 1-8 shows a cutaway of a typical storage battery. Each cell contains positive plates coated with lead peroxide, and negative plates of a spongy pure lead. Undoubtedly you have heard these cells referred to as *lead-acid cells.* This designation is the result of acid being poured into the cells in the form of a liquid called *electrolyte*—a mixture of sulphuric acid and water. It is the union of the positive and negative plates through the electrolyte which produces chemical energy and its end result, electromotive force. Lead-acid storage batteries for airplanes are fundamentally the same as the one in your automobile.

When a battery is connected to an electrical circuit through which it can force current flow, the chemicals within the battery will react to produce voltage. Just how and why is this brought about?

The Primary Cell.

Let's talk about cells for a minute. You will see in figure 1-9 two basic cells, one wet and one dry. When two unlike metals, or it could be a metal and carbon, are suspended in a solution that produces a greater reaction on one than on the other, a difference of potential will exist. And, as we know, if a conductor is connected between them, electrons will flow. The arrows in the illustration show the flow of electrons, and the difference in potential is shown in the meter connected across them. This arrangement is called a primary cell; the two metals are electrodes, and the solution is the electrolyte. This set-up is merely the common dry cell battery.

The difference in potential is the result of material from one or both electrodes mingling as additives to the electrolyte. During this process the ions are formed in the vicinity of the electrodes. The resultant negatively or positively charged ions give the electrodes their respective electrical charges. The amount of difference in potential between electrodes depends on the metals used. The type of the electrolyte and the size of the cell have little or no effect on the electromotive force produced.

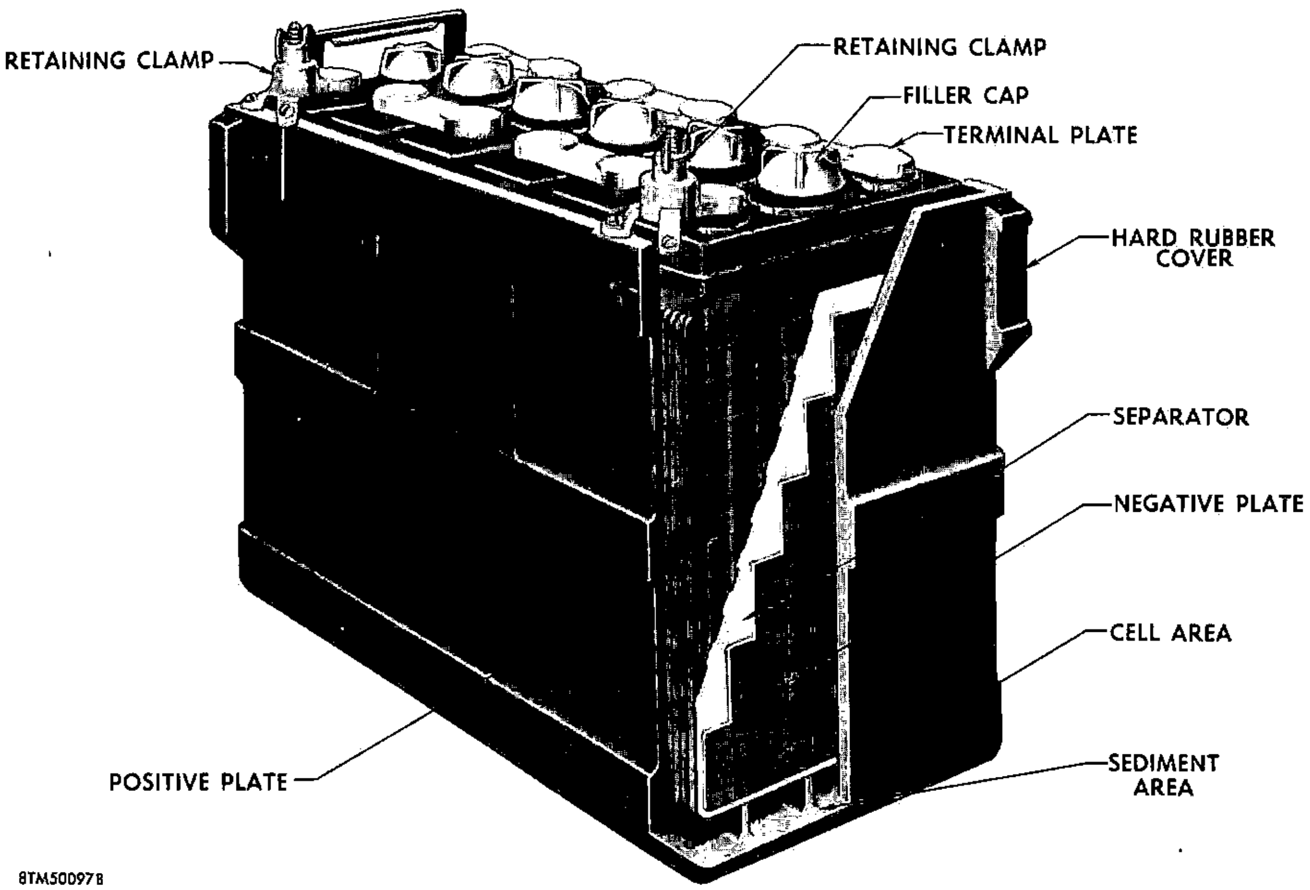

Figure 1-8. Storage Battery Construction

Internal Resistance.

In any conductor there is a certain amount of resistance to electron flow. Similarly there is some resistance in the primary cell we have been discussing. This is due mainly to resistance found in the electrodes and eletrolyte. So, the amount of resistance depends on the kinds of materials used in the battery and the size and the spacing of the electrodes. In large cells however, with their close spacing of electrodes, we find less resistance than those with smaller cells made of the same material.

Secondary Cell.

It is the primary cell which delivers electrical energy direct from the reaction of the chemicals. But, if we are to have sustained voltage, there must be a *secondary* or storage cell which stores this energy before it becomes an electromotive force delivering the electron flow demanded by the circuit. This method, called charging, is what happens in a storage battery. In other words, when a source of higher voltage is connected to a storage battery the proper terminals of the chemical action is reversed from that of a primary cell. The secondary cell is being charged or the electrical energy is being converted back to chemical energy which is stored in the cell. In discharging, the stored chemical energy is re-converted into electrical energy when the battery is connected to our electrical circuit through which it can force the flow of current. This is what is meant by the much used statement, ". . . when a voltage is impressed across a substance."

A simple secondary cell contains two lead electrodes immersed in a dilute solution of sulphuric acid. When a current is forced through the cell, the surface of the

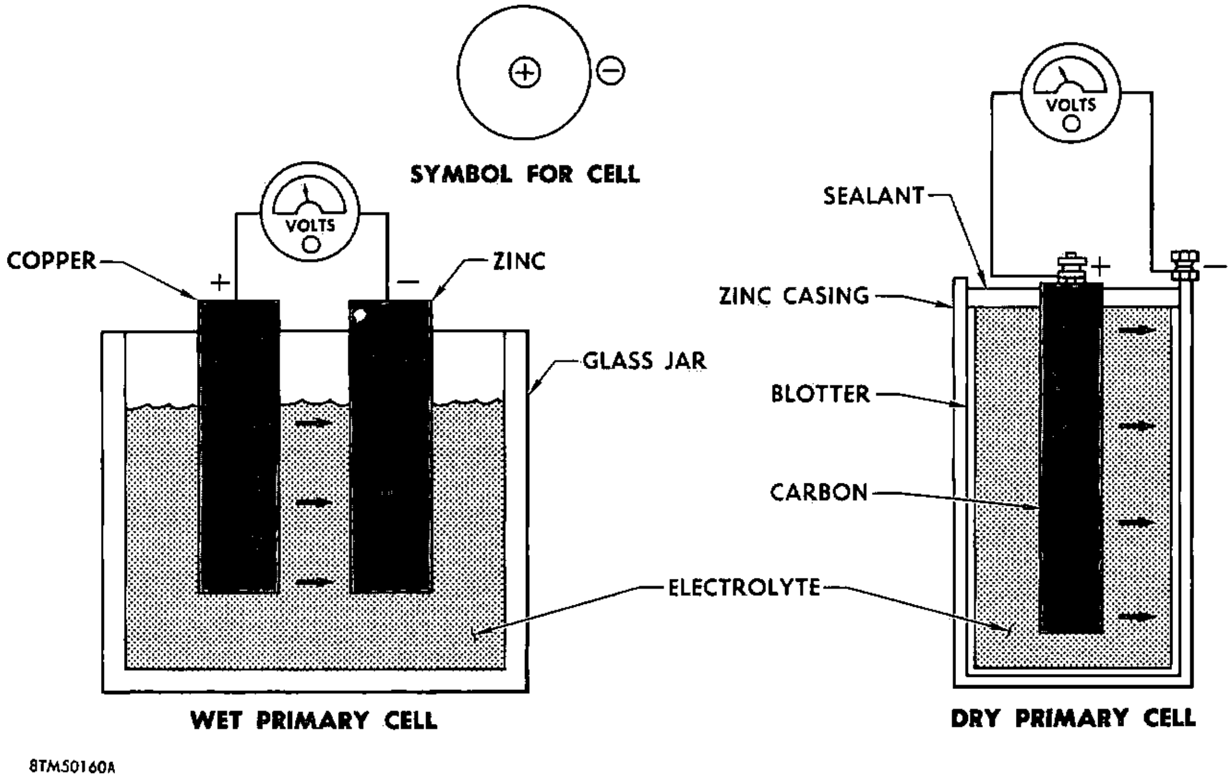

Figure 1-9. The Primary Cell

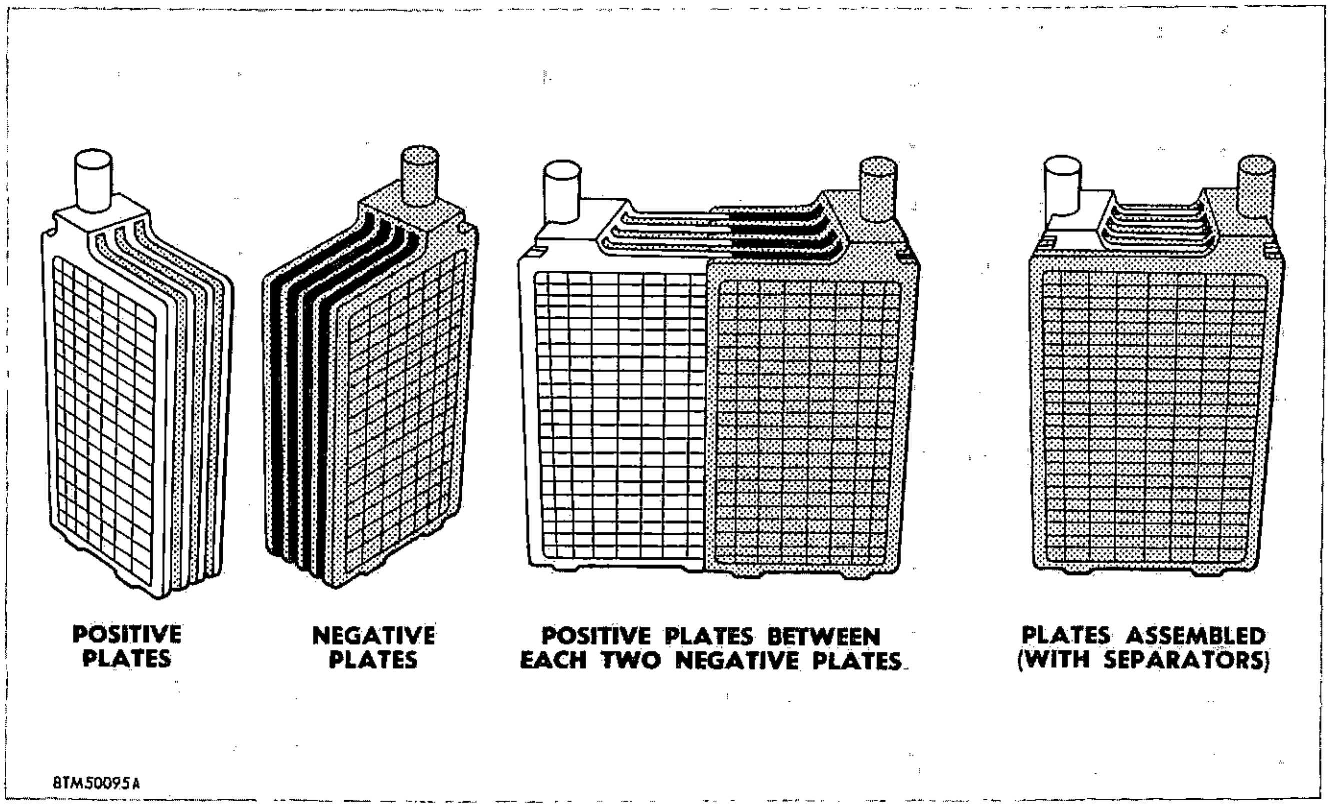

Figure 1-10. Cell Construction Features of a Storage Battery

electrode will be changed to lead peroxide and the surface of the other electrode will be changed to a spongy lead. The surface of the electrode is connected to the positive terminal of the charging voltage.

Discharge of the Lead-Acid Cell.

During discharge, lead sulphate is formed on both the positive and negative plates of the storage battery. The acid content of the electrolyte is decreased as discharge continues, while the water content increases. By the same token, the amount of lead sulphate on the plates increases until the sulphate coatings become so thick that the weakened solution of electrolyte cannot effectively reach the lead and lead peroxide. It is at this time that chemical reaction is held back and the output in the cells is reduced. In actual practice the cell is not permitted to discharge until this condition exists. The reason is, because the thick coatings create a high internal resistance which, when introduced into any circuit, reduces the current to a voltage value too low for practical use.

Charging the Lead-Acid Cell.

As we mentioned before, when the battery is connected to a source of higher voltage the chemical action is in reverse. It is when a cell is being charged that lead sulphate is removed from both the positive and negative plates and sulphuric acid is formed: the electrolyte is now decreased, and as this takes place its density naturally increases.

Open-Circuit Voltage.

The open-circuit emf or the voltage on a typical lead-acid cell is approximately 2.2 volts when there is no load on it drawing current. The voltage is the same on all lead-cells regardless of plate size, and it remains at this value (level) until the cell is practically "dead" regardless of its rate of discharge. It is only when the cell nears total discharge that its sustained voltage level drops off rapidly.

Closed-Circuit Voltage.

This refers to the voltage of an acid cell under load or when connected into an electrical circuit which is drawing on its stored energy. As the cells discharge, the closed-circuit terminal voltage gradually decreases. This is due to the gradual increase in internal resistance of the cell due to the sulpher coating on the plates—to give it a more technical term, the *sulphation of the plates.* At the end of a normal discharge, the internal resistance of a lead-acid cell is more than twice as high as it is when fully (or newly) charged.

Open-Circuit Versus Closed-Circuit Voltages.

The difference between the open-circuit and closed-circuit terminal voltages is due to the voltage drop inside the cell. Mathematically speaking, this drop is equal to the current the load draws multiplied by the internal resistance in the cell. If you recall from your tech school study of Ohm's Law, there is a definite relationship between voltage, current, and resistance of any circuit or any part of a circuit. If the voltage increases, the current increases, and as the resistance increases, the current flow decreases. If you will remember, this relationship is mathematically written $I=E/R$ (I for current, E for voltage or volts, and R for resistance). In our present situation, however, it would be written as: $E=IR$. Therefore, the voltage which a lead-cell will supply under closed-circuit conditions is equal to the open-circuit voltage of the cell, minus the IR drop in the cell.

Characteristics of Construction.

If we are to have a high discharge current as well as a high terminal voltage under load, then the battery must have a low internal resistance. One way to reduce internal resistance is to increase the total plate area. To achieve low internal (IR) resistance, several sets of plates are used in each cell. The positive plates of a cell are all tied together in parallel by one connecting bar. The negative plates are connected together in the same manner. Plates connected in parallel decrease the IR factor without affecting the level of the individual cell voltage—the voltage in this type assembly is relatively the same as a single pair of plates of the same materials. The illustration in figure 1-11 is typical of storage battery construction.

The Assembly.

As you see, each plate of a lead-acid cell consists of a framework called the grid. To this is attached the negative, spongy lead plates and the positive peroxide coated, lead plates. The plates are assembled so that the positive plates fit in tandem: negative, positive, negative, positive, etc., or in series. In this manner the end plates of each cell are negative. Between each plate are fitted porous separators which act as insulators. These prevent the plates from touching, which would cause the cells to arc-over or short out. These separators have vertical rims on their sides facing the positive plate. This construction permits the eletrolyte to circulate freely around the plates, and allows an unobstructed path for the sediment to settle to the bottom of the cell.

Each cell is contained in a hard rubber case sealed by a special compound. On top of each cell are terminal posts and a threaded hole. The hole provides a means of testing the strength of the electrolyte solution and for adding water. A non-spill vent plug seals this opening. This vent plug permits gases to escape from the cell and at the same time discourages any leakage of the electrolyte. Let's take a look at one in action.

In the illustration, figure 1-12, you can see that in the normal flight position the lead weight inside the vent plug allows the dangerous hydrogen gases to escape,

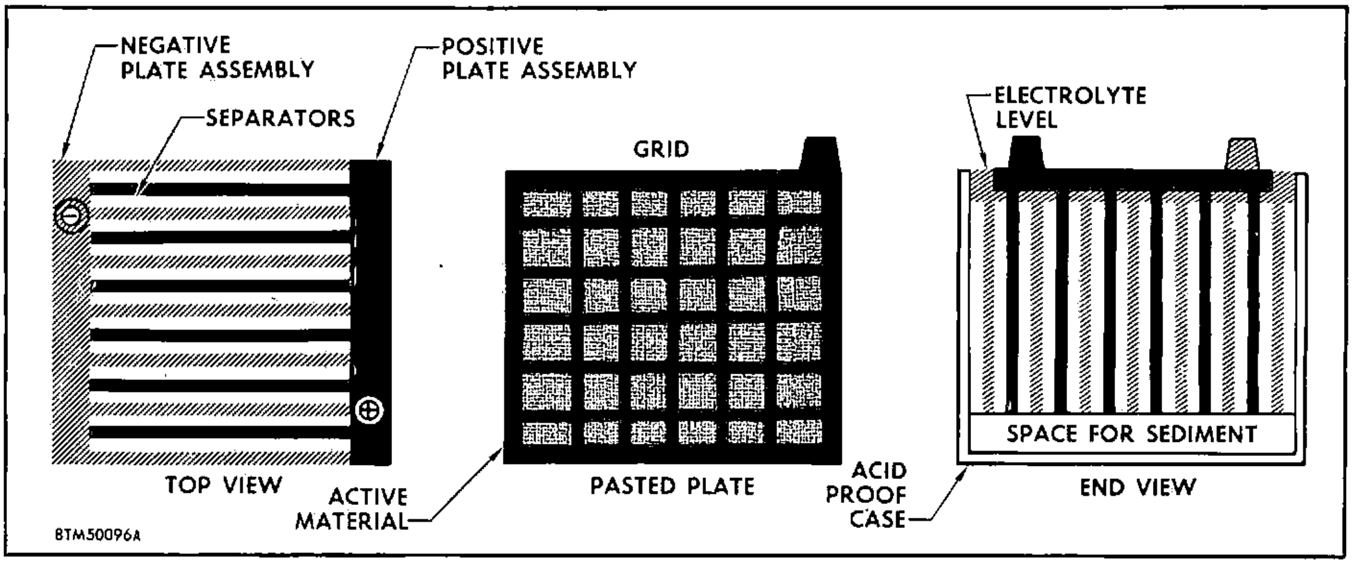

Figure 1-11. Secondary Cell Construction

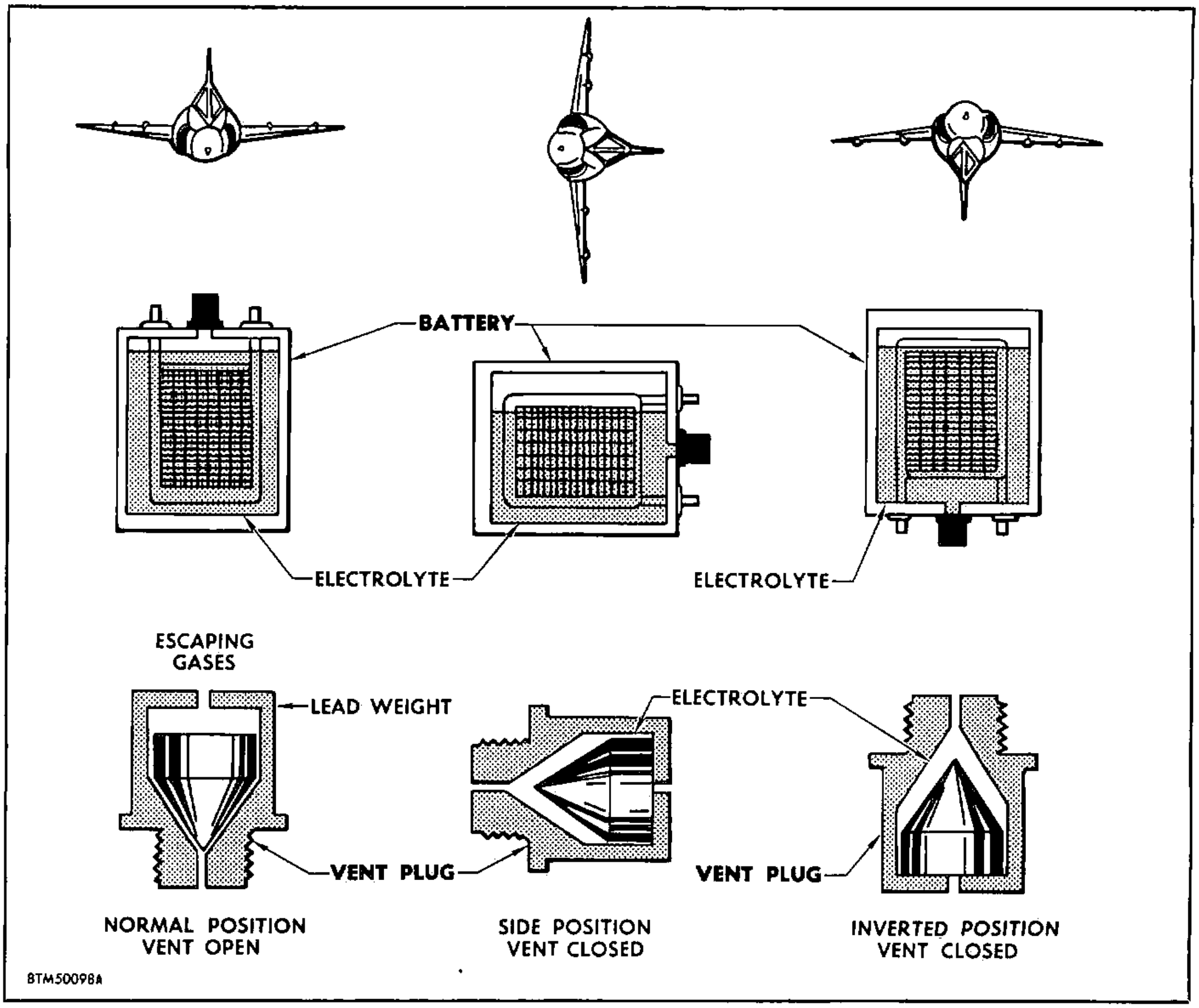

Figure 1-12. Battery Non-Spill Vent Plug

and at the same time prevents the electrolyte from spilling. When the plane is on its side or inverted, the lead weight seats securely, thus sealing the non-spill vent opening. A ventilating system carries the hydrogen gases escaping through the vent plugs away from the battery surface and exhausts them overboard.

Battery Ventilating.

On most airplanes the battery ventilating system is a simple one. Outside air usually enters through a small air scoop in the belly of the plane and is distributed through tubing to the battery container. This ventilating air routes the escaping battery gases into a jar containing pads soaked in baking soda (sodium bicarbonate). The baking soda neutralizes the battery gases so that they do not corrode the airplane skin as they exhaust overboard from the jar.

In the F-102A the air intake tube assembly is slightly different from this conventional method. We will discuss the F-102A battery ventilation when we reach the d-c system in Chapter III of this supplement.

RATING OF STORAGE CELLS.

A storage cell is rated in ampere-hours capacity. The battery used in the F-102A for example, has a 24-ampere-hour capacity. This simply indicates the number of amperes that a cell is able to furnish continuously for a given period of time. In theory, a 100-ampere-hour battery could furnish 100 amperes for one hour, 50 amperes for two hours, or 20 amperes for five hours.

The Ampere-Hour.

In use, the ampere-hour output of a particular battery depends largely on its rate of discharge. As we know, heavy discharge currents heat the battery and decrease its efficiency which naturally decreases the ampere-hour output. A five-hour period has been established for most airplane batteries as the discharge time from full charge under continuous usage. This five-hour period, however, is only a basis for rating and does not necessarily mean the length of time the battery is expected to furnish current. Under actual in-flight conditions the battery may be completely discharged in a matter of minutes. Under emergency conditions, the battery in the F-102A will dissipate its d-c charge in 6 to 12 minutes.

The ampere-hour capacity of a cell is dependent on its total effective plate area. If you connect cells in parallel, it increases the ampere-hour capacity. On the other hand, if you were to connect them in series it would increase the total voltage, but not the ampere-hour capacity. A good example of this is seen in figure 1-13, using the primary cells of a dry cell for the sake of clarity. As you already know from your basic training in electricity, electrical circuits are divided into three classifications: series, parallel, and series-parallel. In airplanes where more than one battery is used, batteries are connected in parallel but the voltage is only equal to that of one battery as you see in the middle view of our example above. Further along in our manual we will review more extensively the electrical circuits classified above, but for now let's get back to the storage battery and the all important factors governing its life.

FACTORS AFFECTING BATTERY LIFE.

Earlier in this chapter we were talking about a 50-cent word—"sulphation." Well, sulphation is one of the chief causes of a short battery-life. If you remember, lead sulphate is deposited on both the negative and positive plates during discharge, and the coating on the plates—the thickness of the sulphate build-up—is governed by the load thrown on the battery during discharge. It is most important that this sulphate deposit be changed back into an active material to mingle with the electrolyte. This is accomplished by *the proper charging* of the battery *at the proper time.*

During the in-flight operations of the airplane, direct current from the airplane generator charges the battery. This method of charging is the "constant-voltage" method. The charging voltage is held constant by the voltage regulator, which is set at a point slightly above the normal voltage reading of the battery, 26.5v. The reason for this is the necessity of continuous charging at a low rate. This continuous method of charging at a low rate—low rate means low differential voltages—minimizes the possibility of overcharging the battery. For this reason, frequent checks on the generator output must be made if the battery is to maintain a proper low rate of charge.

What happens when a battery overcharges? Plenty! First of all, it will cause the electrolyte to boil. When this happens, an excess amount of electrolyte overflows into the battery drain sump, regardless of the non-spill vents, putting its usefulness to an end and causing the battery to overheat. The end result? The cell plates buckle. But this is not all: as the electrolyte level becomes low, the upper parts of the separators dry out and an arc-over, or short circuit, may occur. The result? An explosion!

HYDROGEN GAS.

Remember we spoke earlier of the hydrogen gas which escapes from the battery vents? It is highly explosive. With the battery drain sump out of commission, the explosive qualities of hydrogen will fail to neutralize. This causes a highly dangerous situation to exist, which endangers the mission, the pilot, and the plane itself. So the utmost care of the battery is essential, and if the maintenance is efficiently administered, the battery should perform for its normal life span. The life of a battery—unlike a man's—is predetermined by its developer, and if carefully maintained, may reach 36 to 48 months.

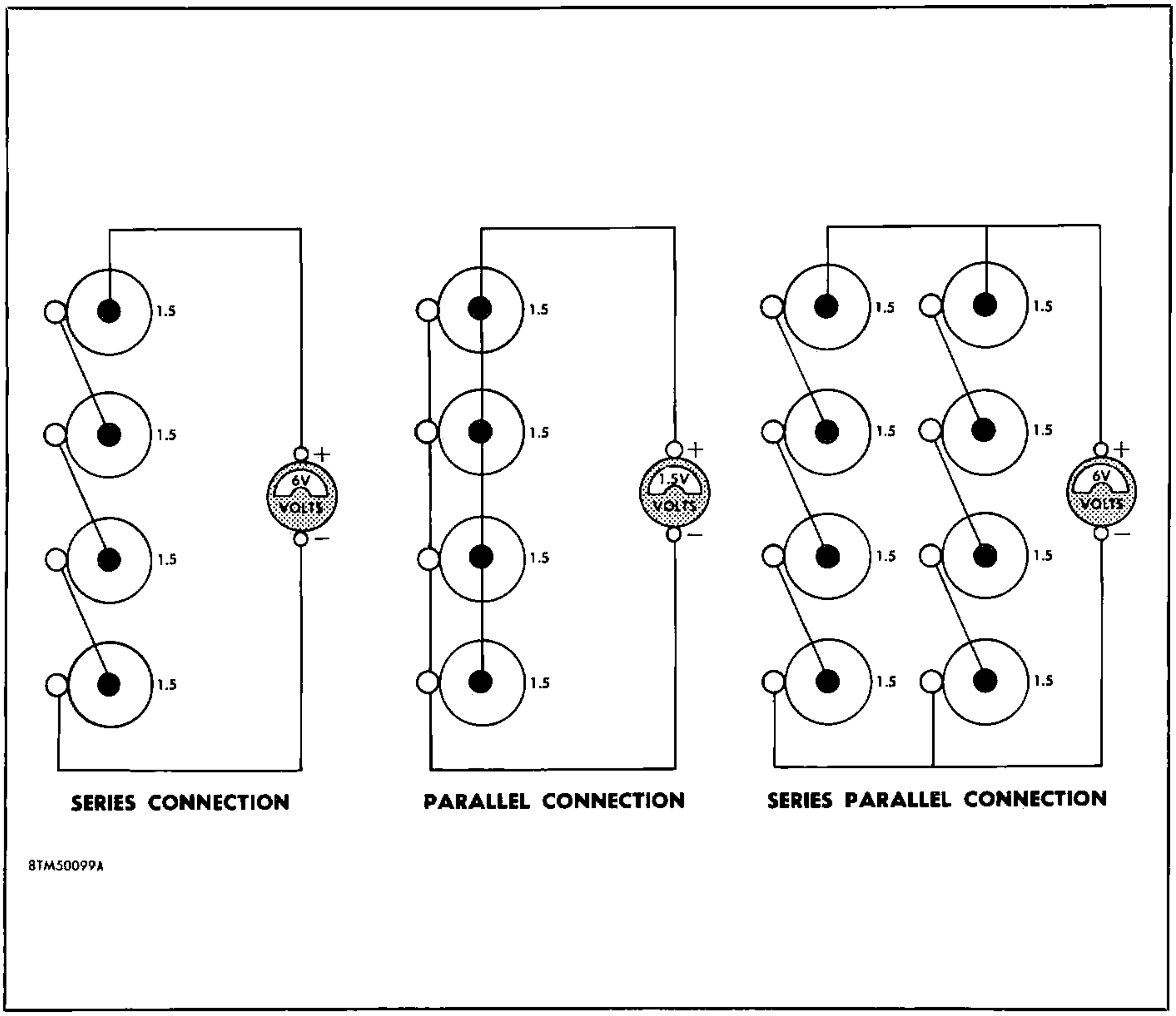

Figure 1-13. Series and Parallel Connections

When we discuss the d-c system in the F-102A and the battery used, we will go into methods of testing, charging, and the maintenance and inspection of the storage batteries. It has seemed sufficient here, to simply review its theory and action.

PRODUCTION OF ELECTROMOTIVE FORCE BY ELECTROMAGNETIC INDUCTION.

Sounds impressive, doesn't it? All the heading means is that electromotive force (emf) is produced by a machine known as a generator. The generator, as you know, converts mechanical energy into electrical energy and emf by using the principle of induction. A voltage is induced in a conductor if there is relative motion between the conductor and a magnetic field. We mentioned this method previously in our brief discussion of magnetism, but we are now at a point when the meat of magnetic theory must be cut and served in palatable slices if we are to understand the full scope of the principles of d-c, or for that matter a-c generators. It should also be kept in mind that these same magnetic principles may be applied to the solenoid relays which you will learn about in Chapter II. Some of these slices will be thick, others thin, according to their importance in our study leading up to generator principles.

ELECTROMAGNETISM.

It is over a century ago, now, that a Danish physicist by the name of Oersted discovered that a current-carrying conductor is surrounded by a magnetic field. This discovery was one of the factors which led to our man Thompson's discovery of the electron a number of years later. It started a train of thought at least, and this is the reason, in our discussion of electromagnetism,

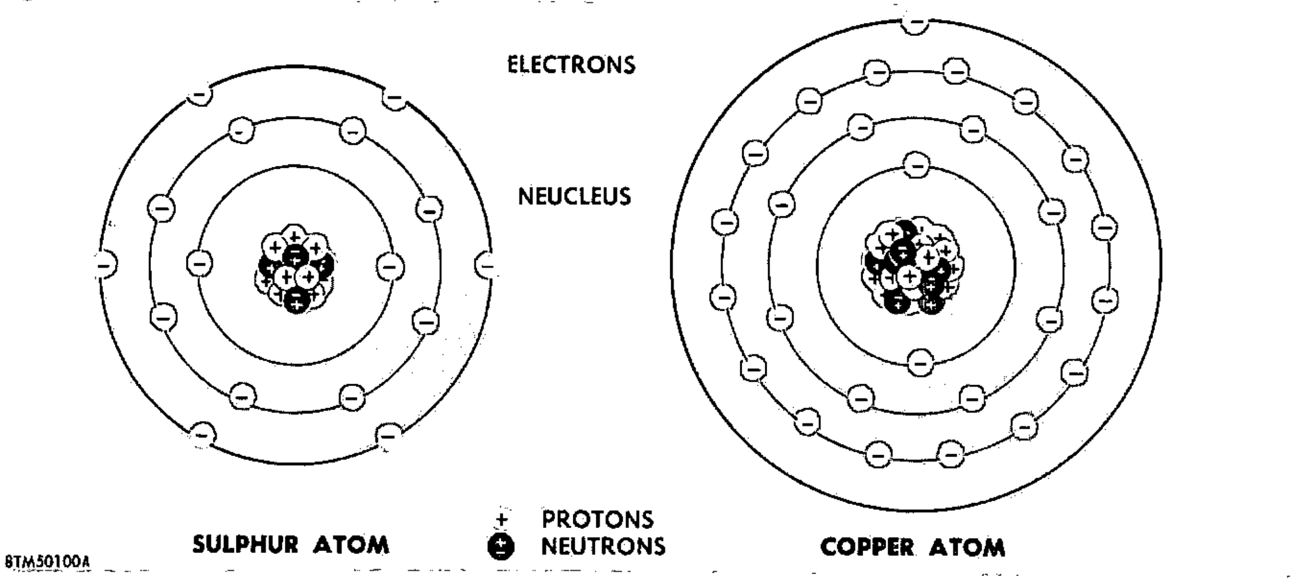

Figure 1-14. Structures of Sulphur and Copper

that the electron and atomic structure comes back into our story, if for no other reason than to emphasize the relationship of the fields of force set up in atomic structures, and those created by electromotive force as it is forced through a conductor. The only difference in the two is movement by direction. Let's not split hairs. The fine lines drawn by scientists between fact and fancy do not enter the scope of this supplement, but analogies do, and when we find an analogy that will punch home two lines of thought and give us a short cut to understanding, let's use it.

Magnetic Fields.

We know what a conductor is, and we should know by now that when electrical pressure is applied to an atom structure, the result is electrical current through the magnetic pull of one structure (the conductor) being greater than the magnetic pull on the other (the atom structure). You know what is meant by an atomic structure—you saw this in the simple hydrogen and helium structures. Figure 1-14, shows the more complex atomic structure of copper. As you know, copper is the common electrical conductor used in airplanes. It can be readily seen by its structure why it is preferred over all other conductors.

If you will remember, a conductor is characterized by the content of free electrons. This means the higher the atomic number the more readily the electron flow will move. The atomic structures of the two examples—figure 1-14—are arranged in layers, as are all others from the simplest of atomic structure, the hydrogen atom, to the more complex of atomic structure, copper. It was our friend Thompson who suggested that the electrons were arranged in this manner. And if you remember, it was he who discovered the electron. This brings us back to the Danish physicist, Oersted, who as we have already said, found that whenever a conductor of electricity has a current flowing through it, magnetic fields exist about the conductor, as shown in figure 1-15.

What does this remind you of? Electrons as they move about their nucleus? Right. The difference being, however, that electrons have no definite direction of movement; at least not until an external pressure is exerted to produce an electromotive force sufficient to force them through a conductor in a definite stream, and in a definite direction.

Left-Hand Thumb Rule.

The lines of force that form about a current-carrying conductor, as shown above, travel in either a clockwise or counterclockwise direction. This depends on the direction of electron flow. If you remember, the direction of the lines of force is determined by the *left-hand rule.* This states, that if you grasp a conductor in your left hand in such a manner that your thumb points in the direction of the electron flow, the fingers of your left hand will indicate the direction of the lines of force. Figure 1-16 will simplify your understanding of this rule.

Magnetic Field About a Loop.

If you bend a straight conductor into a *single turn loop* as shown in figure 1-17, the lines of force enter the loop on one side and leave at the other. Now, if you wind several turns of wire or a series of loops into a coil, the magnetic fields all have the same direction at every point.

The Solenoid.

A solenoid consists of a coil of wire wound around a hollow cylinder, and is used *to produce a magnetic field.*

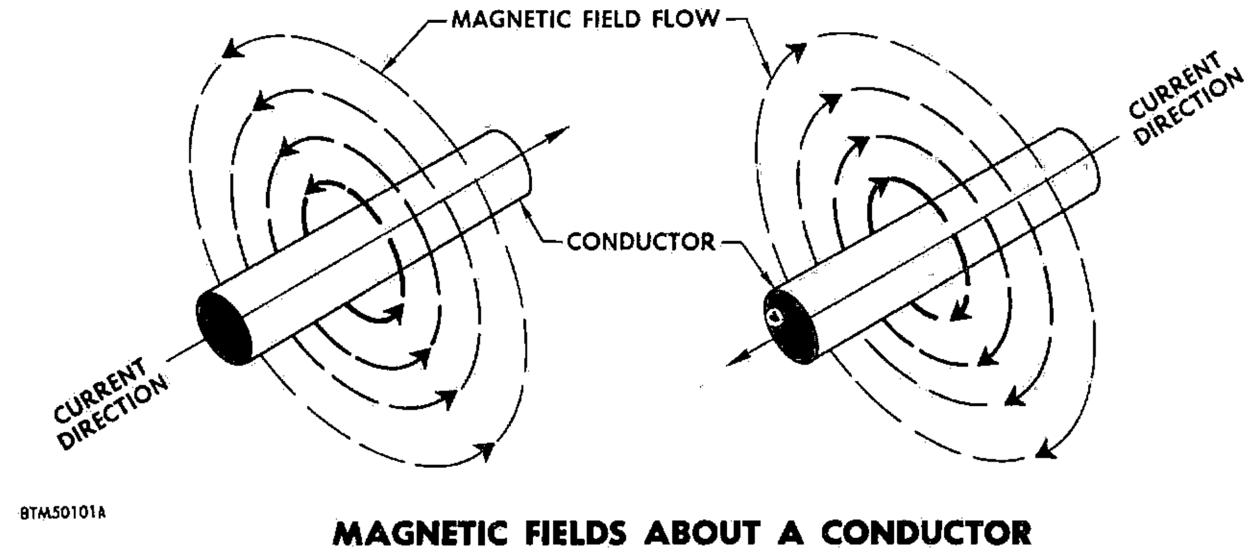

Figure 1-15. Magnetic Fields About a Conductor

Now, if you were to place a movable core of soft iron inside the cylinder, the magnetic field of the coil will tend to center the core into the coil when current is turned on—inserting a soft iron core into a solenoid greatly increases the number of magnetic lines of force. This increase in magnetic lines, however, is not from an increase in the intensity of the field, which depends only on current and turns per unit length, but from the additional lines of force produced by the magnetization of the iron core. Solenoid coils with movable cores are used extensively in the F-102A for the remote control of various units such as solenoid-operated valves and relay switches.

Let's take a simple example of a solenoid and explain its theory, for it is an important device in airplane circuits today. The magnetic field associated with a helical coil, or solenoid, is much the same as the field surrounding a bar magnet. The comparative drawing of figure 1-18, shows you what is meant by this.

If an equal current is forced through the coil, let's say it consists of eight loops, and at the same time through a single-turn loop of the same diameter, you will find the magnetic fields are almost identical. The magnetic field strength of the 8-turn coil, however, will be approximately 8 times that of the single-loop. The reason should be clear enough: in the 8-turn coil its

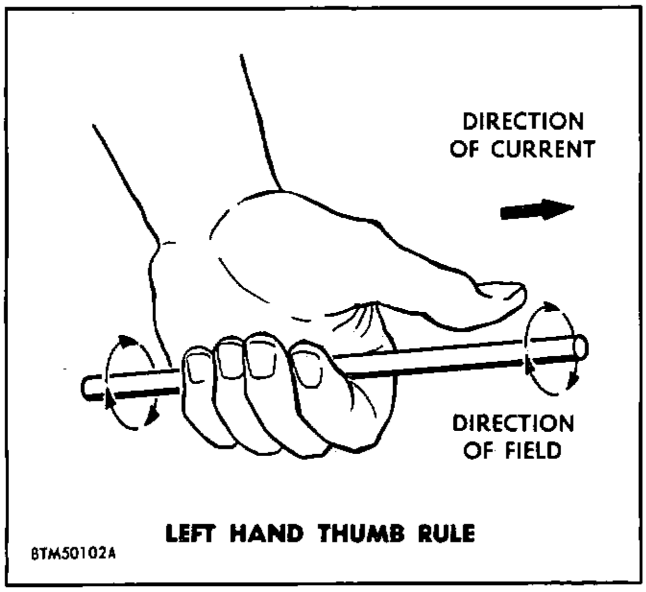

Figure 1-16. Left-Hand Thumb Rule for Conductors

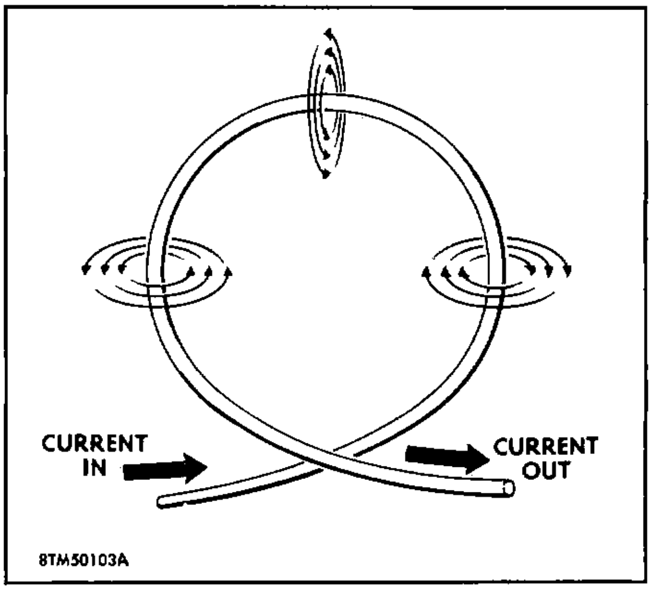

Figure 1-17. Effect of a Loop on Magnetic Fields

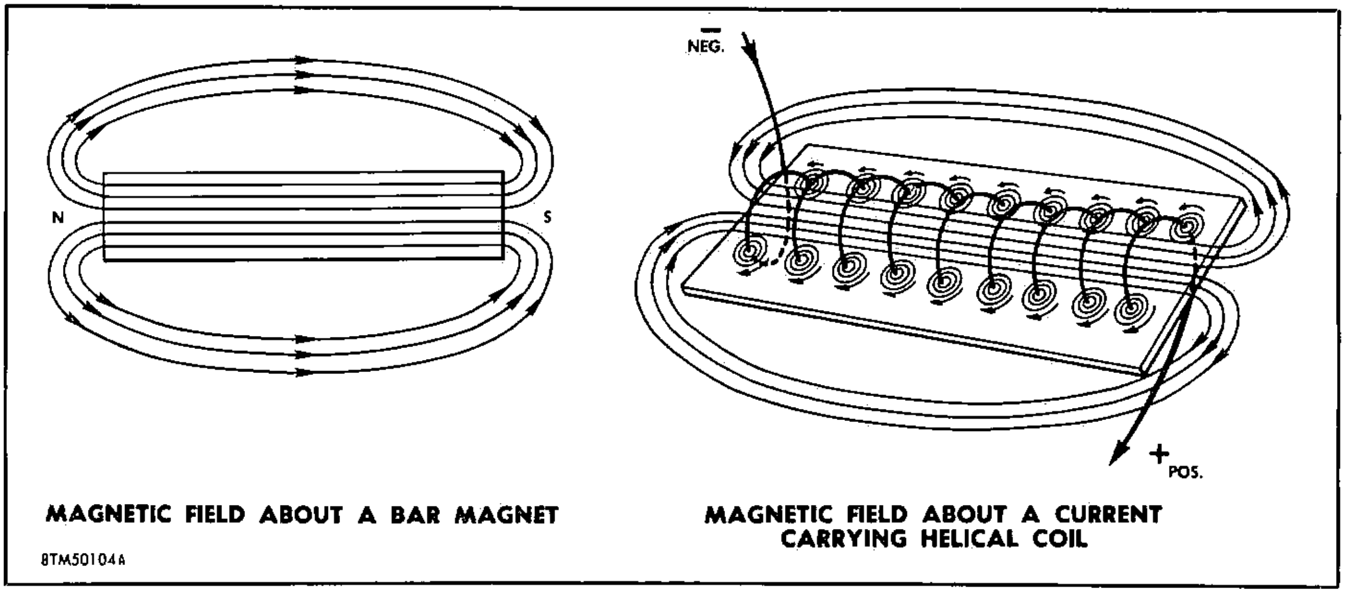

Figure 1-18. Magnetic Fields

magnetic fields are paralleled with each other at every point, and their effects—by adding one to the other—are greater, or cumulative. Let's spread our 8-turn coil into a helical coil, or solenoid. The magnetic field between the turns will now be very weak because the fields of adjacent turns are opposite in direction, they tend to cancel each other as can be seen in the illustration, figure 1-19. But, *inside* the coil they are strong because their magnetic fields are cumulative.

This cumulation is a strong field of fairly uniform intensity, because it represents nearly straight lines of force inside the coil. What has happened? An electromagnet has been induced through the action of several fields of force. It is better than an ordinary magnet because it can be magnetized and demagnetized at will. Thus it can be linked to a switch arm, as in a solenoid-type relay, and remotely open and close a circuit in response to either an automatic or manual controlled power source.

Electromagnets.

The strength of an electromagnet is based on three important factors: the number of coil turns of wire, the kind of metal in its core, and the amount of current flowing through the coil. In the F-102A, electromagnets are used in the landing-light relays, voltage regulator, flight control system, generator relay switches, disconnect relays, warning relay systems, power shutoff, generator field coils, and in many other places in its overall system. We will discuss these units in detail in appropriate sections of this supplement.

Determining the Polarity of an Electromagnet.

Here again we have a left-hand rule. As can be seen in figure 1-20, grasp the coil with your left hand so that the fingers point in the direction of current flow in the coil, your thumb is extended at right angles to the fingers, and points the direction of the north pole. If the current is reversed the poles will reverse.

Magnetic Flux.

The total number of lines of force around any *magnetic current,* or in any region of magnetic activity, is called magnetic flux. A simple example of this can be seen in figure 1-21. If you remember, one of the basic laws of magnetism is that lines of force are continuous and always form closed loops. Now, because magnetic flux lines of force form closed loops, the paths followed by flux loops are called *magnetic circuits.*

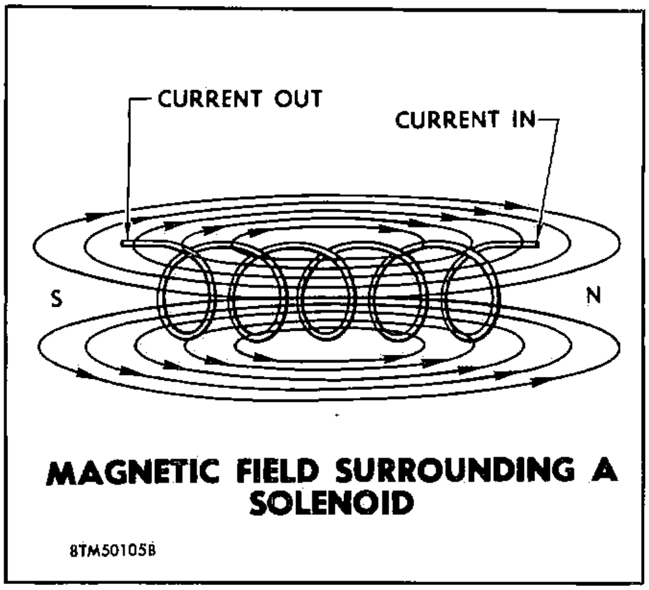

Figure 1-19. Magnetic Field Surrounding a Solenoid

Magnetic Circuits.

In the magnetic circuit, flux is similar to current in an electrical circuit. As you know, the force which produces a flow of electrons in an electrical circuit is electromotive force (emf). In a magnetic circuit, the force producing flux is called magnetomotive force (mmf).

Resistance, Conductance Versus Reluctance, Permeability.

In an electrical circuit, it is *resistance* that opposes current flow; in a magnetic circuit, *reluctance* holds up the magnetic flux. *Conductance* in an electrical circuit is the ease with which the electrical current flows, but in a magnetic circuit it is *permeability* that is the ease with which a substance conducts magnetic lines of force.

MMF.

The amount of magnetomotive force (mmf) is expressed in ampere-turns. One ampere of current flowing, say through one turn of a coil, is one ampere-turn. A generator pole having 20 turns of wire around the pole, with say 10 amperes flowing through it, will exert 200 ampere-turns of mmf.

Hysteresis.

The molecular theory of magnetism explains at one point, that if you hold a piece of steel parallel with the earth's magnetic field and strike it several times with a magnetized steel hammer you will magnetize it. What happens is, the blows struck shift the tiny molecular magnets in the steel into uniform lines. By the same token, you can demagnetize a permanent magnet by heat. Heating the magnet accelerates the motion of the molecules throwing them into their former state of confusion.

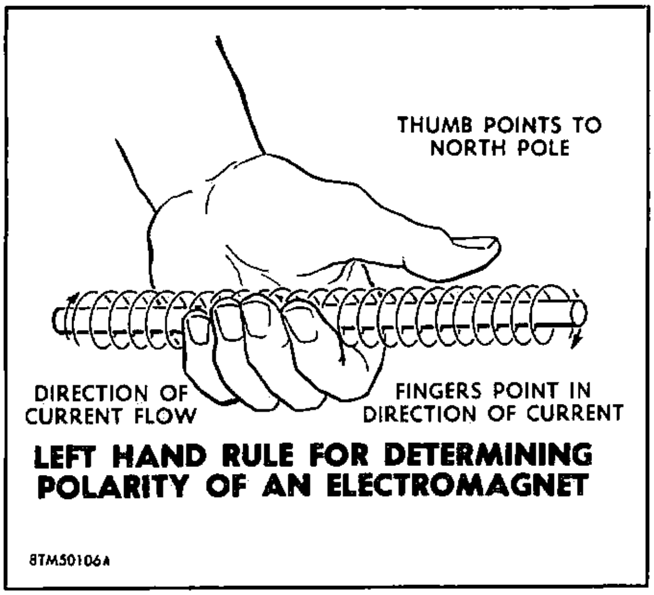

Figure 1-20. Left-Hand Rule for Determining Polarity

Soft iron conducts magnetic lines of force more readily than steel, because soft iron has more permeability than steel. In other words, steel will hold magnetism longer than soft iron, but it does not conduct as easily. Speaking of iron, let's return to our example of a magnetized and non-magnetized piece of iron mentioned earlier in this chapter. The former's molecular structure is aligned uniformly, the latter's haphazardly. In the magnetization of a piece of iron, considerable energy is expended in the form of heat due to the acceleration of molecular movement.

If this is the case, think what happens in the life of molecules when they are aligned first in one way and

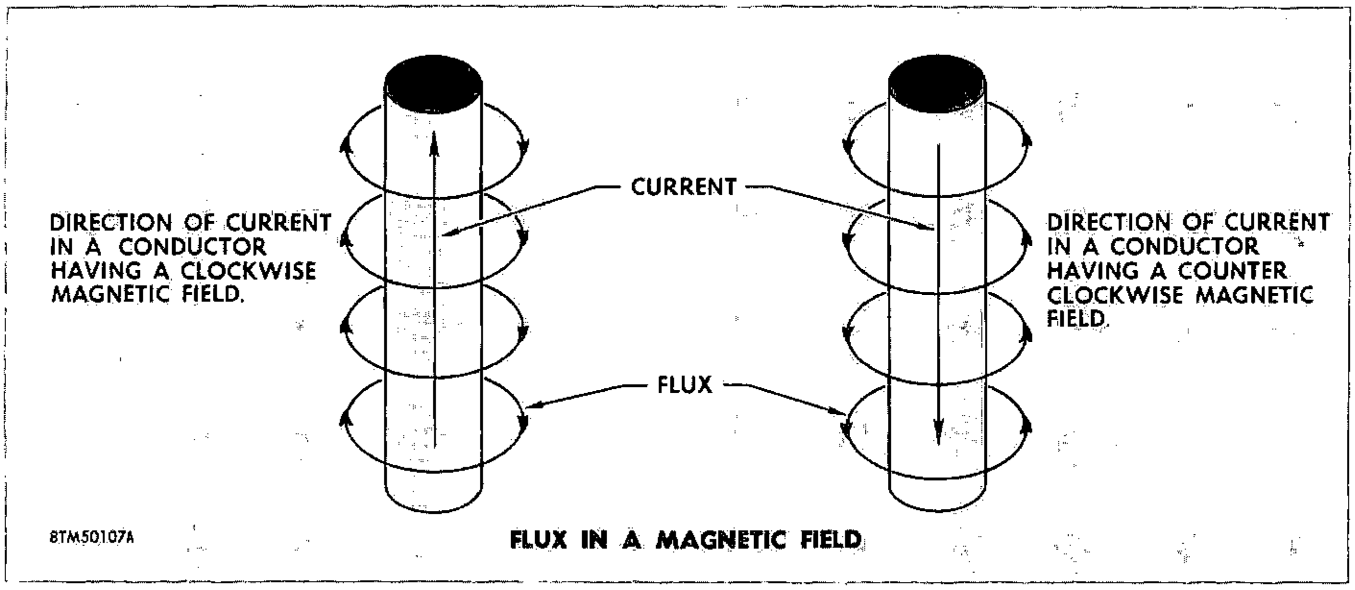

Figure 1-21. Flux in a Magnetic Field

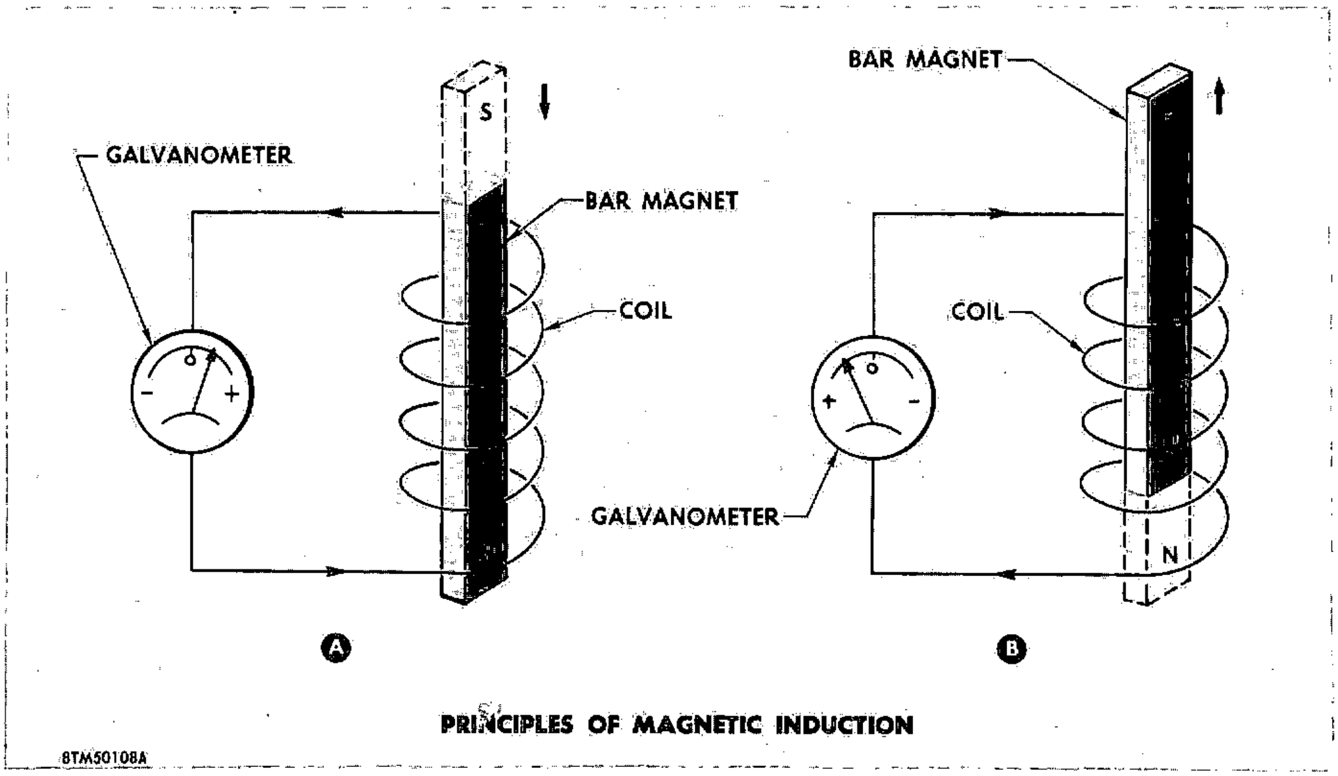

Figure 1-22. Principles of Magnetic Induction

then in another many times per second. Complete hysteria. An example of this is when an alternating current is sent through ferromagnetic material such as iron. The waste of energy through heat is great. It is this wasted energy which is known as hysteresis loss. In electromagnets, this hysteresis loss in the magnetic core prevents its polarity reversal at the same time that the magnetic current reverses its flow.

It stands to reason then that iron is not the material to use in an alternating current electrical component when a high pattern of efficiency is demanded. In alternating current transformers, or where alternating current electromagnets are used, steel showing a low hysteresis loss, through tests, is customarily used.

PRINCIPLES OF INDUCTION AND CAPACITANCE.

The major discussion in this portion of the chapter is the production of electromagnetic force by electromagnetic induction. We have been some time getting to the actual induction, but unless electromagnetism is understood there would be little sense in discussing induction at all. Volumes have been written on all phases of electricity covered to this point. It is hoped that with your basic training in electrical school, and the practical application of that training in the field, your knowledge is more than the false sense of knowledge we spoke of earlier — familiarity.

Electromagnetic Induction.

The production of voltage and current in a conductor, when it cuts the magnetic field of a magnet, is called *electromagnetic induction*. Actually, the conductor itself does not have to be in motion to cause this electrical phenomenon—it is enough if the magnetic field is a moving field and the conductor is stationary. Figure 1-22 will help to emphasize this point.

As you can see, the coil is connected to a galvanometer, which as you probably know, is a sensitive instrument used to detect small voltage changes. In section A of the illustration, when the bar magnet is plunged into the coil, the meter deflects to indicate voltage. If the magnet stops at any point within the coil, the voltage indicator returns to zero—a voltage no longer exists. If the magnet is withdrawn, as in section B, the meters indicate that a voltage of opposite polarity has been produced. How is this explained?

The value of this induced voltage *at any instant* depends upon the number of lines of force and the speed at which they are being cut at that particular instant. This is sometimes explained through *flux linkage*, but for our purpose it is simpler to use the expression, *lines of force*—for they mean the same thing.

If the conductor is moved, cutting these lines of force at a greater velocity, say twice as fast, the induced voltage will increase to twice its original value. In a like manner, if the velocity is decreased, there will be a corresponding decrease in the induced voltage. Also, if the strength of the magnetic field is increased or decreased in value, the result is a corresponding increase or decrease in the value of the induced voltage. In what other way may we increase or decrease the value of the induced voltage? That shouldn't be too difficult to determine, simply increase or decrease the length or size of the conductor.

Now, what are the "whys" and "wherefores" of what we have been discussing? If you will remember, we stated that energy can neither be created nor destroyed. Therefore, the electrical energy must be produced at the expense of some other energy. And if you will recall, we said that energy is a commodity which can be bought and sold. In this instance, mechanical energy (the motion of the magnet) is changed to electrical energy *by the coil.* The motion of the magnet after all was necessary for a flow of current. This brings up a rather important point: the method of determining the direction of the induced voltage and current in an electrical circuit.

Lentz's Law.

A fellow by the name of Lentz, in 1834, summed up induction somewhat like this: The voltage, due to electromagnetic induction, must be of such a direction that the current flow and its resultant magnetic field will oppose the action which produced them. Let's apply this to our illustration of the principles of magnetic induction.

It can be seen that when the magnet is induced, the field that is produced directs the flow of current in the direction of our left-hand rule for coils. A magnetic field has been produced and current flows from north to south. As the magnet is withdrawn the polarity is reversed, opposing the initial action which produced the voltage and current in the first place. Let's add another rule.

Generator Left-Hand Rule.

Let us say a conductor is made a part of a closed circuit. This would result in a voltage being induced in the conductor. Now, to determine the direction of this resulting voltage and current, we would use the left-hand rule for generators. This is simply carrying Lentz's law a step further and determining the relationship between the direction of the magnetic field and the direction of the induced voltage. Figure 1-23—a man's hand and the fundamental principle of a generator—will give you a good idea of what is meant.

If you place the first and second fingers and the thumb of your left hand at right angles to each other in such a position that the index finger points in the direction of the magnetic flux and the thumb points in the direction of motion, then the middle finger points in the direction of the induced voltage. If the conductor is part of a closed electrical circuit, the current will flow.

To further understand this rule, position the fingers of your left hand as noted above and shown in the left-hand view of figure 1-23, then place the back of your left hand on the illustration next to the right view. Your index finger will point toward you, and by looking at the directional arrows, the path of the current flow is clearly indicated.

Self Induction.

When a coil is connected in series with a battery and a switch, and when the switch is closed, the current in the circuit produces a magnetic field. The current in the circuit does not build to a maximum instantaneously, as it would if the circuit contained resistance only. It builds gradually, as is shown in the current curve, figure 1-24, acting as though something in the circuit opposed its build-up. This is actually the case — the opposition is caused by what we have been talking about, electromagnetic induction in the coil. The coil has generated a voltage called a *voltage of self-induction.*

Now, the instant the switch is closed the current starts to increase as magnetic lines of force form about each turn of the coil. These lines expand outwardly, cutting and linking up with each other. Technically, this process is called *flux linkage.* Their varying degrees of strength as the current increases, induce into the conductors in the coil a voltage which is opposite in polarity to that of the original current—caused by the impressed voltage when the switch was closed. This self-induced voltage on the coil is called a *counter voltage,* and ceases as soon as the current reaches a maximum value and the magnetic field becomes steady.

Energy in a Magnetic Field.

What we have just discussed is rather important to your understanding of the electrical nature of things. So, let's carry this a step farther.

At some time after closing the switch, the amount of current in the circuit stops—the lines of force (the flux linkages) no longer waver, and the induced counter voltage returns to zero. Now, if at this time you opened the switch, there would be an arc-over between the switch contacts as shown in figure 1-25.

This arc-over, or spark, represents electrical energy. It indicates also that a high voltage is present. The electrical potential for the spark is supplied by the energy stored—remember our dam—in the magnetic field in the form of expanding flux, or magnetic lines of force. During the time of the current build-up, energy was stored in the magnetic field by the *work* performed *by the current* to overcome the opposing counter current of self induction. Let's make a comparison. Say you have a screen door held shut by a spring. When you open the door, the spring is stretched taut. With the door open, the spring has energy stored which represents the ability to do work. The energy was stored there in the first place by the work done in opening the door. When the door is released, the stored energy in the spring pulls the door shut. Well, in the

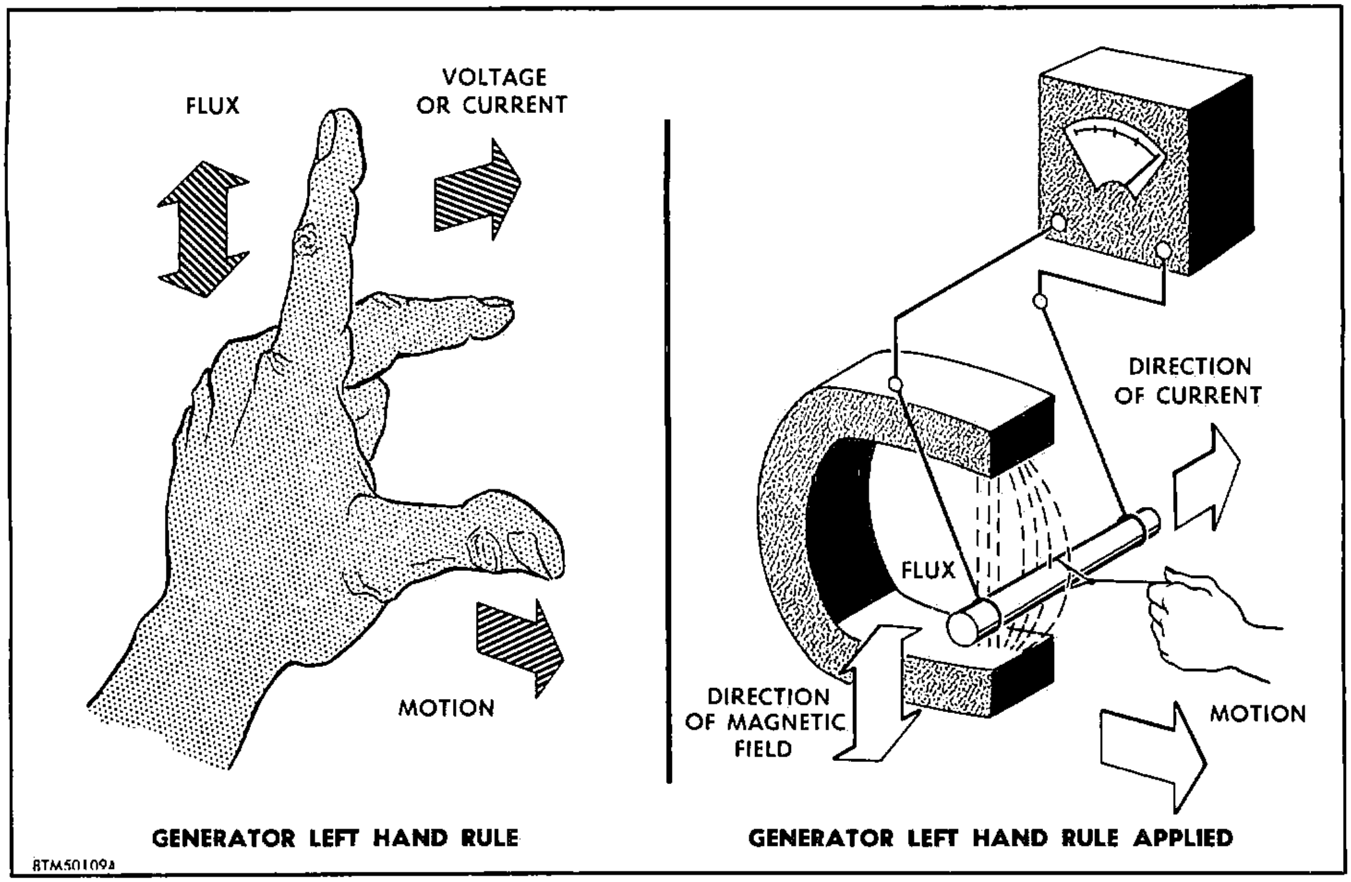

Figure 1-23. Generator Left-Hand Rule

same way, the work done by the current to overcome opposition to the self-induced counter voltage stores energy in the magnetic field of our simple circuit. When the switch is open, the magnetic field collapses (the door slams shut) and the energy stored is returned to the circuit (the door at rest against its jamb).

Let's return to our friend Mr. Lentz for a moment. While the field is collapsing, there are variations in the flux linkages, or the magnetic lines of force, which induce a voltage *in the coil.* According to Lentz's law, this voltage opposes any decrease in current. The high self-inducted voltage must go some place just as any

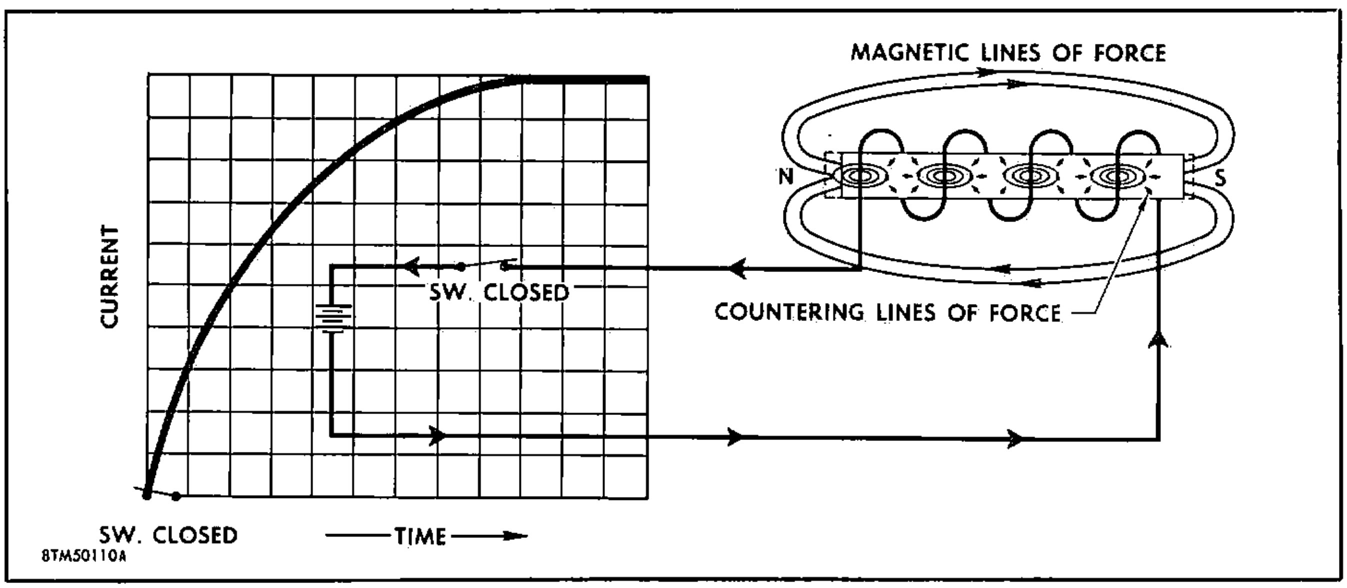

Figure 1-24. Self-Induction and Counter Voltage

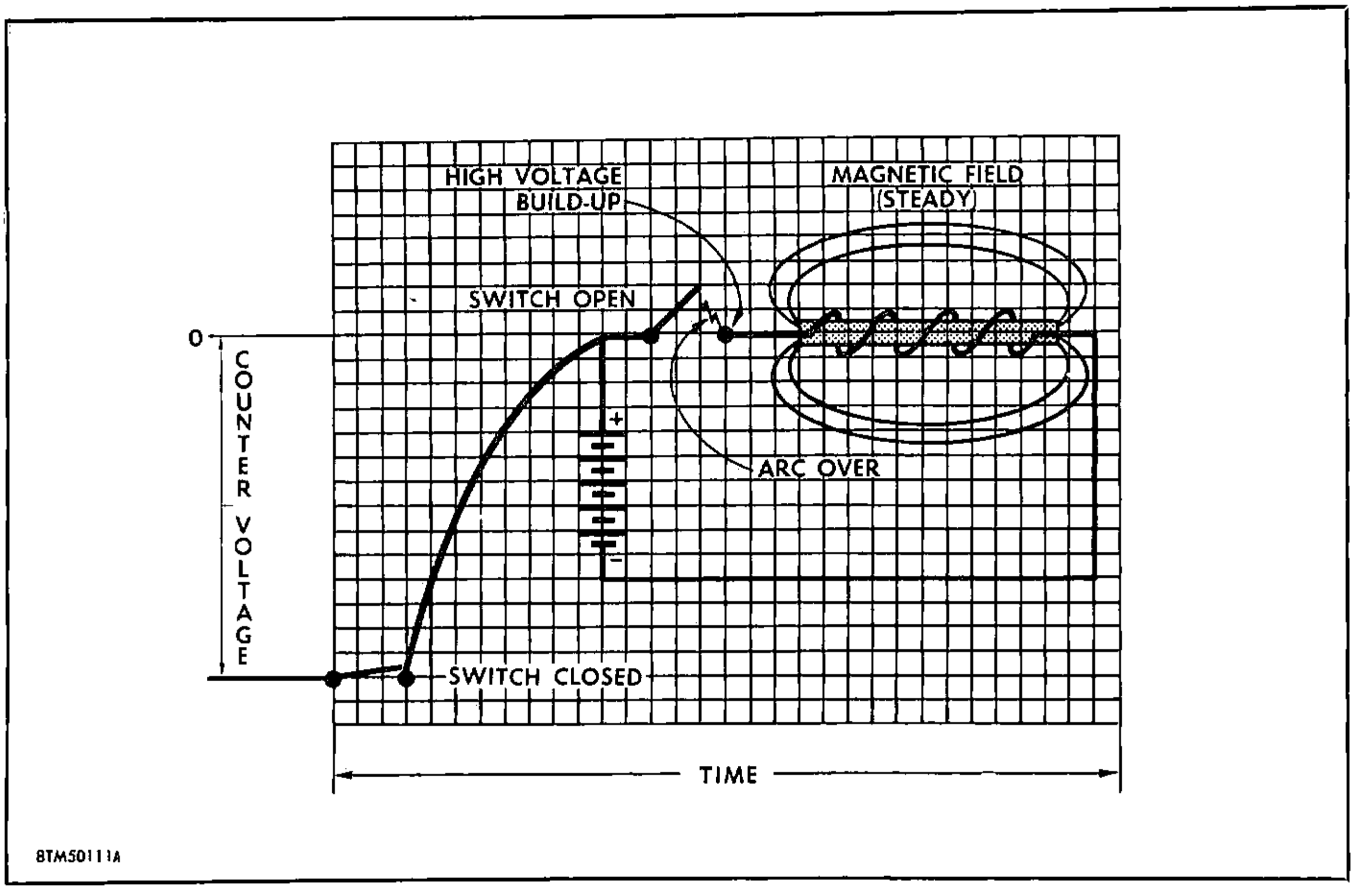

Figure 1-25. Energy in a Magnetic Field

gathering of free electrons—remember our discussion of electrostatics—so the stored energy jumps across the switch contacts. The strength of the stored energy is determined by the amount of inductance and current in the circuit. The current, however, is a greater factor than the amount of inductance. If the inductance is doubled, the strength of the arc (stored energy) is double. *But,* if the current is doubled the strength of the arc increases four times.

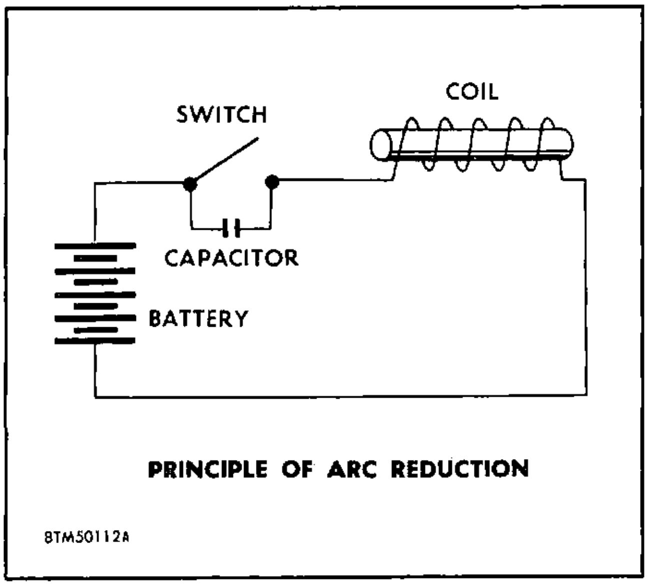

Figure 1-26. Principles of Arc Reduction

The principle of arc reduction is shown in the illustration, figure 1-26. In this illustration, a capacitor is connected across the switch contacts eliminating the arc, and the electrical energy—which formerly did not have a place to go—is stored in the capacitor for future use. But before we discuss this capacitor let's talk about another form of inductance, or mutual induction.

Mutual Inductance.

In our electrical circuit, a change of current is always accompanied by a change in the magnetic field surrounding a circuit. If the current is increasing, the magnetic field is said to be expanding, that is, its intensity at any particular point is increasing. On the other hand, if the current is decreasing, the field is said to be collapsing or decreasing in its intensity. When a conductor is placed in a magnetic field of a circuit in which the collapsing or expanding lines of force cut the conductor, a voltage will be induced in it. (See figures 1-27 and 1-28.)

You learned, when we discussed self-inductance, that when a current flows through a coil, the lines of force produced a linkage with the other turns of the coil.

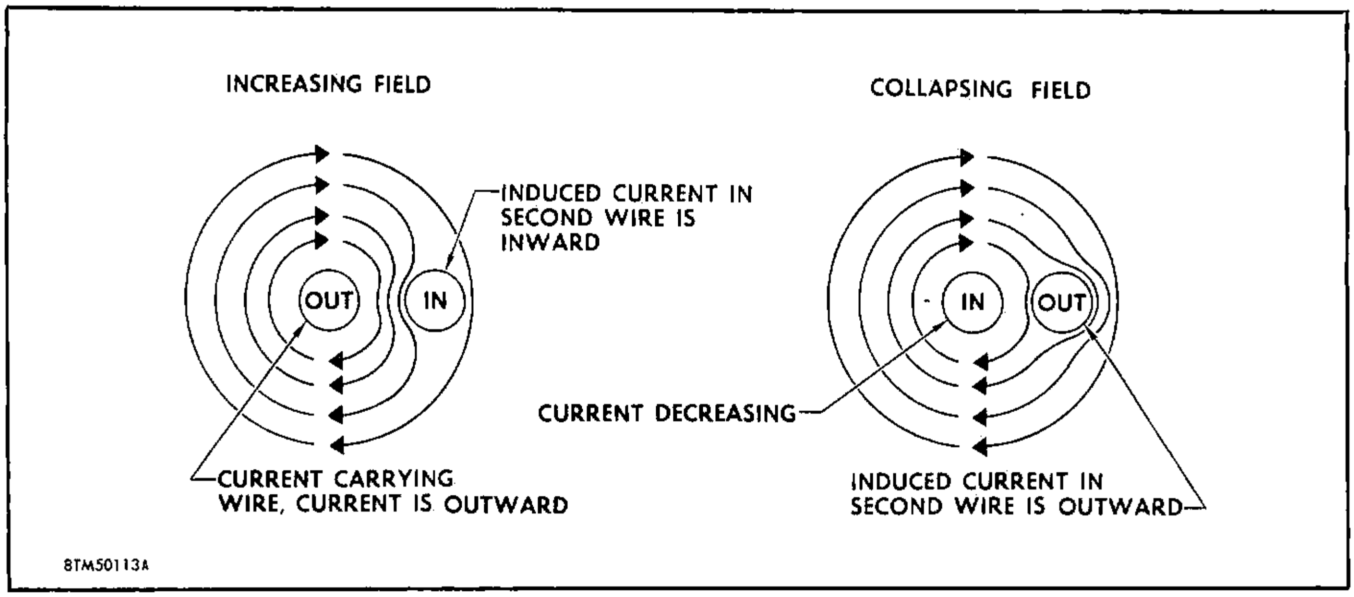

Figure 1-27. Mutual Induction

And also, that whenever a change in the current occurs, the corresponding change in these linkages induces a voltage (voltage of self-induction) in the coil. Now, when a second coil (not connected to any source of voltage) is placed next to or near the current-carrying coil—the two coils being oriented so that the lines of force of one links with the turns of the second coil—any change in linkages will induce a voltage in the second coil. This is what is meant by a voltage of mutual induction.

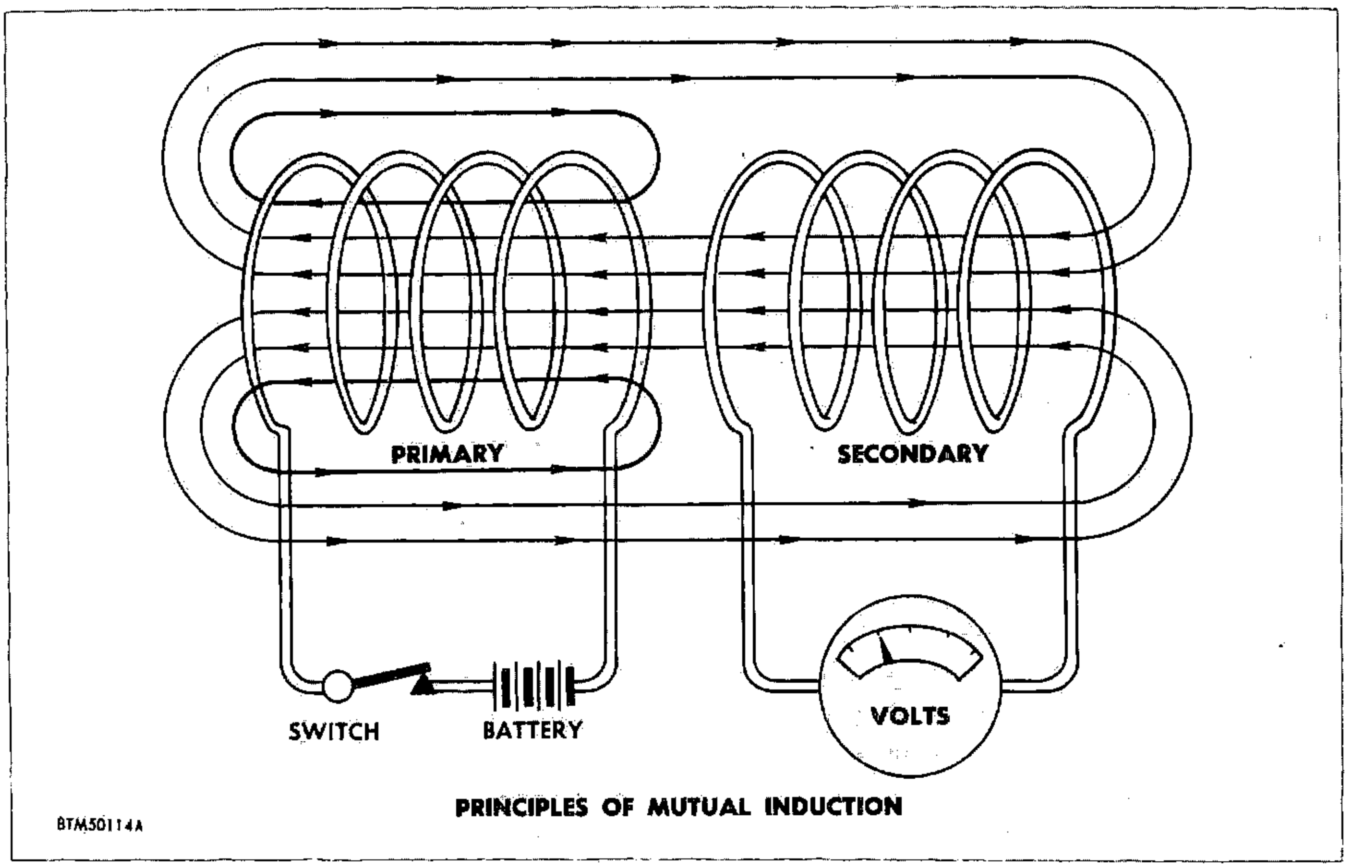

Figure 1-28. Principles of Mutual Induction in Coils

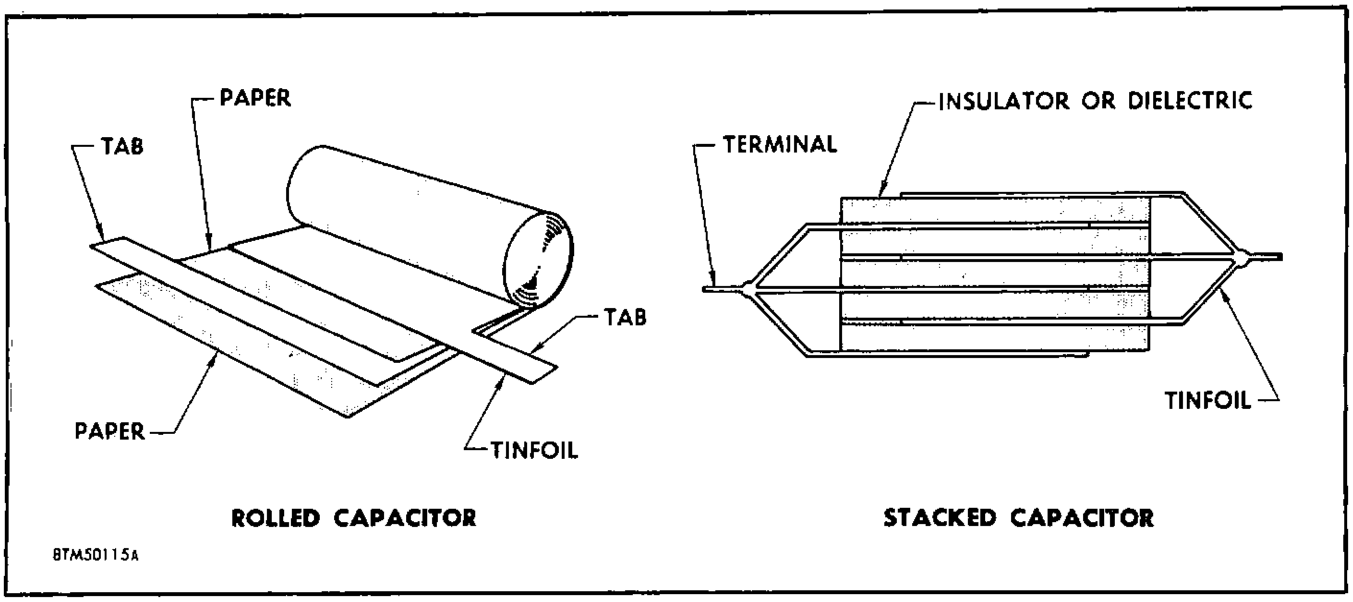

Figure 1-29. Rolled Capacitors

Let's repeat, when two coils are so arranged that a change of the magnetic lines of force in the current-carrying coil causes air induced voltage in the second coil, *both* coils have a property known as mutual induction.

You will recall that reluctance in a magnetic circuit is the corresponding term for the resistance in an electrical circuit. Well, when the reluctance of the magnetic circuit in coil number one is constant, the amount of mutual inductance between the two coils depends on the number of magnetic lines of force linked to the second coil, and the unit rate of change in the current reading of coil number one. Let's summarize: the coil carrying the original current is called the *primary* coil.

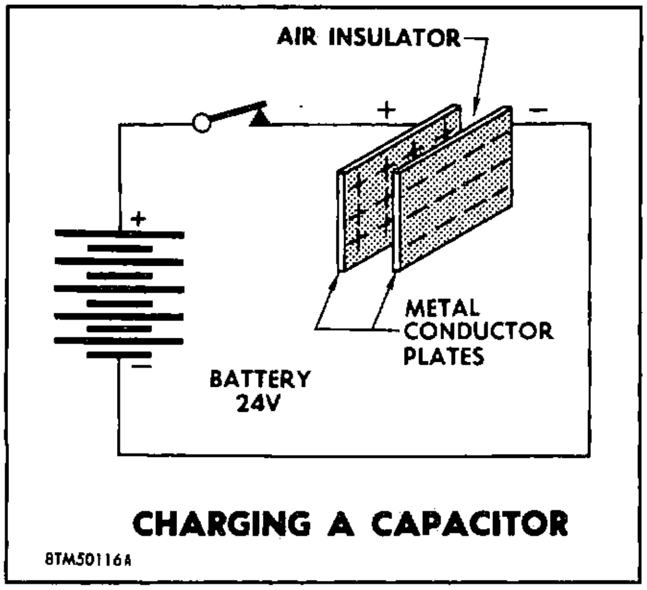

Figure 1-30. Charging a Capacitor

Its companion coil is called the *secondary.* When a current flows in the primary, a strong magnetic field appears about the primary coil. Now—and remember our curve in explaining self-inductance—as the current increases, or decreases, this strong field *cuts each turn of the secondary coil.* And when the two coils have the same number of turns, the voltage induced in the secondary will be the same as that applied to the primary. If, however, there are more turns in the secondary than in the primary, the voltage in the secondary will be higher. On the other hand, if there are fewer turns in the secondary, then there will be less voltage in the secondary. In other words, it is the number of turns in the secondary coil as compared to those of the primary coil which determines the amount of voltage induced in the secondary coil by the primary coil.

Capacitors.

In our diagram of a simple electrical circuit, we provided it with a gadget known as a capacitor—across the switch contacts — to store the high voltage build-up thereby eliminating the arc-over. A simple definition of a capacitor is a device used for the temporary storage of electrical energy. Let's pull one apart. As you can see from figure 1-29, the essential parts of a capacitor are the plates (usually tinfoil), an insulator between the plates (paper or a dielectric), and the connecting leads or terminals. The two capacitors shown above are common types. In the stacked capacitor, the alternate plates are connected together and provided with two sets of plates to which the terminals are connected. In a capacitor with high storage capacity, the plates are made of metal foil, and the dielectric of wax paper or mica.

The capacity of a capacitor, which as you know, simply means the amount of electrical energy it can store, depends on the area of the plates and the material used as the dielectric or insulator. It might be a good idea here to illustrate the action of a capacitor as a storer of electricity. The process of storing electricity in a capacitor is called *charging the capacitor.* When the wires from our 24-volt battery are connected to the terminal posts of the capacitor—the switch being closed —the current flows to the capacitor plates until they are charged to an equal amount of the battery voltage. If we disconnect the wires from the battery and bring them together, the current will flow in the opposite direction until the capacitor is discharged.

An example of the practical application of this is a capacitor connected across a d-c circuit to stabilize the voltage and to eliminate radio interference. In this case, capacitors absorb the small ripples in the d-c voltage which frequently cause the static that is heard. What happens is, when the d-c voltage rises, the capacitor becomes an electrical reservoir which overcomes the fluctuating voltages, and the d-c ripples smooth out to a steady d-c voltage.

GENERATORS.

The F-102A uses one d-c generator as its regular source of d-c current. This is an air-cooled generator which delivers 200 amperes at 28 volts to the buses at a minimum speed of 4000 rpm. The a-c power system uses two generators—one for its normal source of a-c power, the other for its emergency power. The first one is a 30-kva, 120/208-volt, three-phase, 400-cycle, a-c generator, and like the d-c generator, it is air-cooled. The other is a 1-kva, 120/208-volt, three-phase, 400-cycle, a-c generator used only for emergency operations.

The entire discussion in this manual thus far has been more-or-less a survey, a knitting together, of the basic concepts of electrical power. It is hoped that this review has brought into focus all that you have previously learned, or have known, about these elusive characteristics of what we know as electricity. From here on out we will be talking about the concrete components that circulate the electricity needed in the airplane. We have covered the principles of the storage battery, in what we feel was the appropriate place for it, and call it the blood bank for the F-102A. And it is just that. But, generators are the real heart of the matter. They make the "juice," that all important "electromotive force," which brings an airplane such as the F-102A to life. So let's discuss generators.

Generator Principles.

We have been discussing the principles of electromagnetic induction because the generator is a machine which employs these principles to convert mechanical energy into electrical energy. A generator which produces alternating-current energy is called an *a-c generator;* one which produces direct-current energy is called a *d-c generator.* Both a-c and d-c, and this is important, operate by the induction of an a-c voltage! Why? In our discussion of inductance we spoke of the varying and wavering amounts of magnetic flux which intertwine, like threads, as it passes through the coils. As you know, alternating current, unlike direct current, changes its flow—first in one direction and then in another direction—at regular intervals. This varying of the magnetic flux in the coils results in an a-c voltage. What does this prove? Simply that all electromagnetic generators are based on the principles of an alternating current. This alternating current, which is developed in the armature of a simple electromagnetic generator, may be delivered to an external electrical circuit in one continuous flow by means of a *commutator* mounted on the armature shaft. It then becomes a direct-current generator. The illustration of the two elementary generators in figure 1-31 will illustrate what we mean by this statement.

The principle difference in the two generators shown is the method in which the generated energy is taken from the generator. In the a-c generator, the energy is taken from the *slip rings;* in the d-c generator, it is taken from the *commutators.* Both methods will be explained later. But for the time being let's keep in mind that internally both types of generators produce electromotive force—electrical voltage—by the same method, and both produce a-c electricity.

BASIC A-C GENERATOR.

The basic thinking behind an electromagnetic generator was introduced earlier, where you learned that whenever lines of magnetic force are cut by a conductor, such as a wire, passing through them, a voltage is produced in the conductor; and the strength of the voltage is dependent upon the speed of the conductor and the strength of the magnetic lines of force; and, if the ends of the wire are brought together to form a closed loop, a current is induced in the conductor.

The simplest form of a generator is the one shown on figure 1-31. It consists of a conductor formed into a loop, *ABDC,* and is located in the magnetic field between the poles (north and south) of an electromagnet. The two ends of the loop are connected to the slip rings *x* and *y* and they in turn make contact with two brushes which are connected to the resistor *R,* or load.

The sequence diagram in figure 1-32, will take the loop through one clockwise revolution. The output voltage of this revolution will be indicated in degrees of rotation on the waveform chart. Remember that the resulting electricity formed will be a-c.

This is our simple type generator, as shown in figure 1-31, making a complete cycle of 360°. Its single turn coil, or loop, is suspended in a magnetic field; its polarity, north to south. The two ends of the loop, as

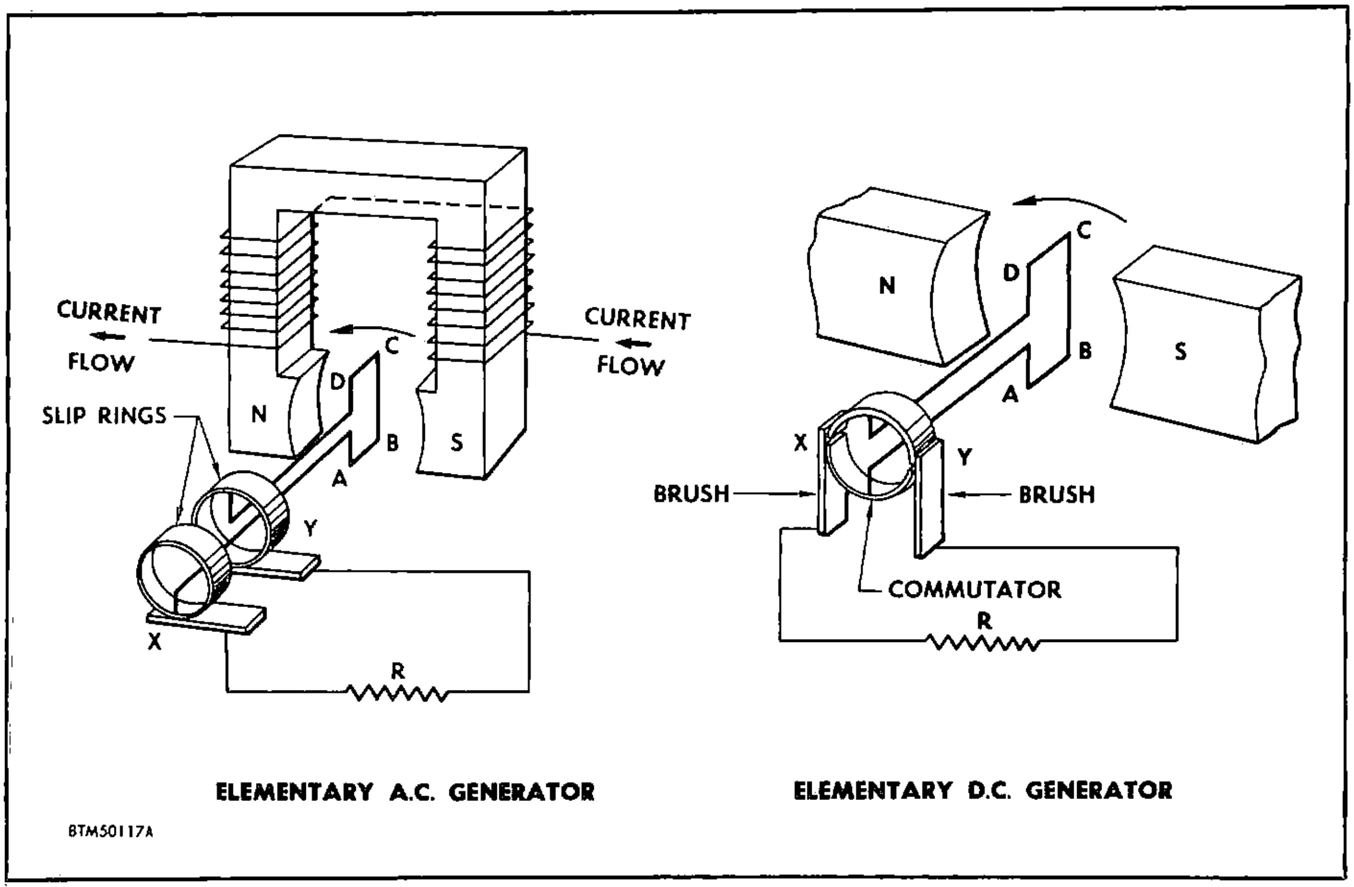

Figure 1-31. Elementary A-C and D-C Generators

we mentioned before, are connected to the slip rings *x* and *y*. These rings are attached to the shaft, which rotates the coil, and are electrically insulated from one another. A stationary brush—made of either carbon or metal—rides on each slip ring, making a constant electrical contact with it as the coil rotates. It is through the slip rings and their brushes that the voltage generated in the coil is transferred to the load, shown as a resistor.

The Generation of Voltage.

The desired voltage, to produce the current delivered to the load, is generated by rotating the coil in a magnetic field at a uniform rate of speed. Let's assume that our coil (see figure 1-31), is to be rotated in a clockwise direction. In position *A,* we find that the voltage induced in the coil is zero; the coil side *DC* and *AB* are not cutting the lines of force, but are traveling parallel to their direction. As the coil begins to rotate farther, it starts to cut the lines of force, and as the cutting angle increases the lines of force are cut at a greater rate of speed. What happens? The voltage induced into the coil keeps increasing in magnitude. And by the time the coil reaches the 45° position, view (B), the sides of the coil cut the lines of force at such an angle that the voltage induced in the coil has a magnitude shown at point *b* on the waveform chart.

Now, when the coil rotates to the 90° position as shown in view (C), the sides of the coil are traveling perpendicular to the lines of force. At this position the lines of force are being cut at the greatest rate of speed. On the waveform chart you can see that the magnitude of the voltage being induced in the coil at this point is maximum. But, the coil does not stop here, it continues on until it reaches the 135° position, view (D) —the same position it held when it began its cycle at view (B). As it moves on, you can see how the cutting angle starts to decrease, and at point *d* on the chart the voltage has also begun to decrease. The magnitude of the voltage at both views (B) and (D) are the same. From here on, the rate at which the sides cut the magnetic field diminishes until the coil reaches the position in view (E) at which time the voltage becomes zero. The coil has traveled one-half its cycle.

Let's recap a little, during the first half cycle, as the coil side *AB* moved down through the magnetic field, *DC* moved up through the field. A voltage was induced in *D* toward *C,* and in *A* toward *B.* These two voltages added together in series make brush *x* positive and brush *y* negative. The current then must flow out of the coil side *AB* into the load in the direction of the arrow and back into the coil side *CD.* In the external circuit of a generator, the current always flows from a negative potential to a positive potential.

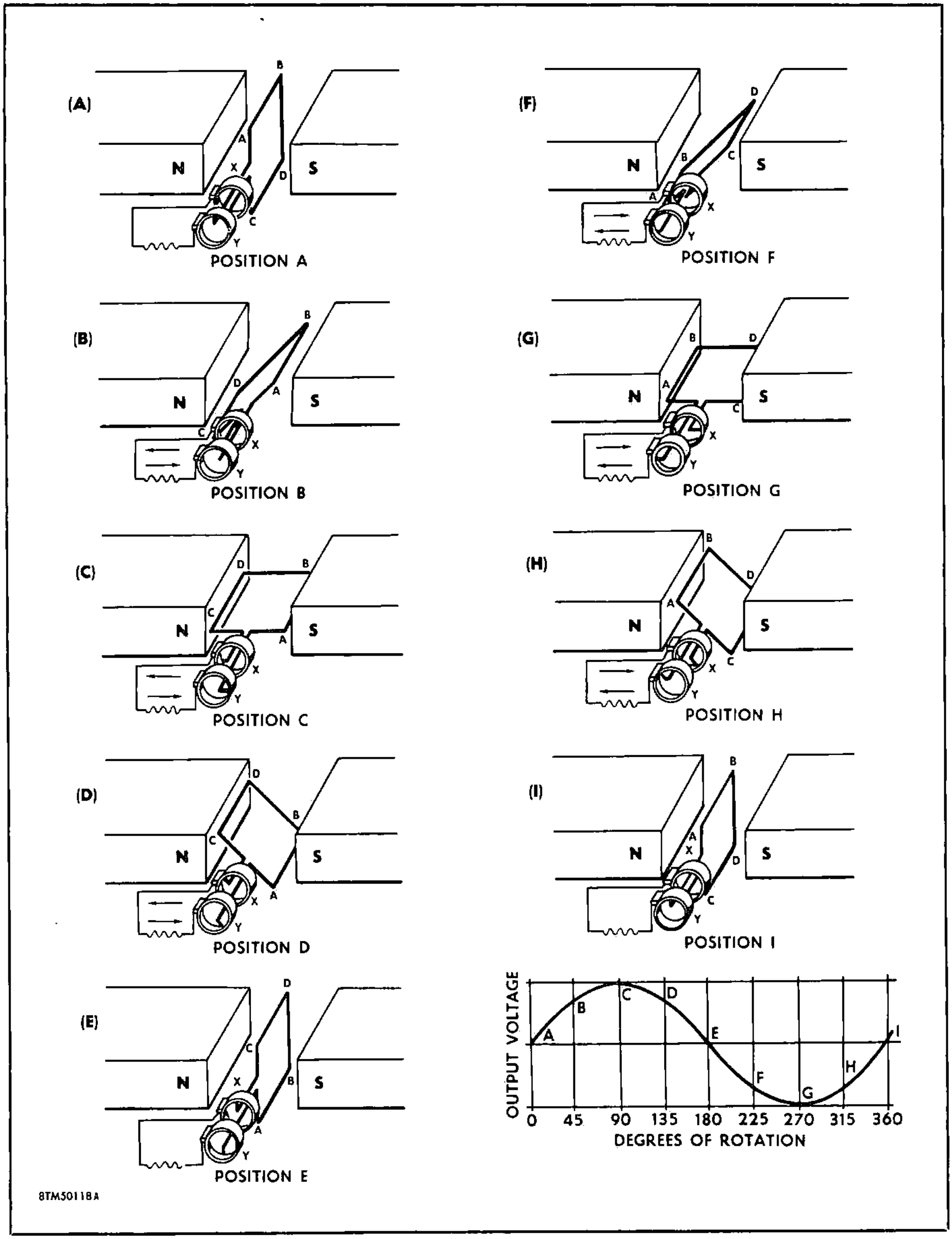

Figure 1-32. Cycle of A-C Generator

In the rotating coil of a generator, however, our electrons flow from the positive to the negative terminals. This is just the opposite from their flow in the external circuit of the generator. The reason for this is the confused state of our molecules at this point, and this is caused by a mechanical force being exerted upon them rather than by the electrical potential that forces them through the coil.

During the final half of the coil's cycle, the coil sides *AB* and *CD* travel through positions *e* to *i*, which as you can see, cut the lines of force at the same rate and same angles as they approach the corresponding positions of the first half of the cycle. The magnitude of the induced voltage varies in the same manner in both cases; however, there is one important difference during the last half of the cycle—the coil side *AB* is now moving up through the opposite (negative) side of the magnetic field, while *CD* is moving downward through the former path of *AB*. The polarity has been reversed. What happens then to the voltage and current? It's rather simple if we give it some thought and remember some of our fundamentals. The magnitude of the voltage and current varies in the same manner that they did in their first half cycle—from zero to maximum and back to zero—their direction is all that has changed, they simply reversed themselves. This is reflected in the waveform chart from the 180° position to the 360° position.

ELECTRICAL DEGREES VS MECHANICAL DEGREES. What we have been talking about is one complete cycle of alternating current. It is divided into 360 equal units which are called "electrical degrees." The coil actually rotated through 360 mechanical degrees, or two alternations of 180° each, thus generating a varying voltage through 360 electrical degrees. It should be clear, then, that one electrical degree in a basic alternating-current generator represents the same time interval as one mechanical degree.

A PRACTICAL A-C GENERATOR.

The elementary a-c generator just discussed can be expanded into a practical a-c generator by increasing the number of poles. Now we must bring the term armature into this discussion. You undoubtedly are familiar with it, but before going on let's define it. A simple armature is nothing more than the coil we have been talking about in our basic a-c generator. The conductors mentioned above are merely addition loops. In a practical a-c generator, the armamture consists of a flexible steel shaft on which are located a soft iron core and the armature windings, or loops.

A practical a-c generator which has a single set of coils, all of which are connected in series so that the electromotive forces are added together, is called a *single-phase generator,* and its output is called a *single-phase current.* If, however, a generator armature is divided into a number of sets of coils (so arranged that the voltage output of each reaches a maximum point or a minimum point at different positions, as the loop or armature rotates), we have what is called a *polyphase* generator. And so we arrive at a point in our discussion which is of the utmost importance to us, the *three-phase generator.* It is this type of a polyphase generator with these sets of coils, which is used in the a-c system of the F-102A.

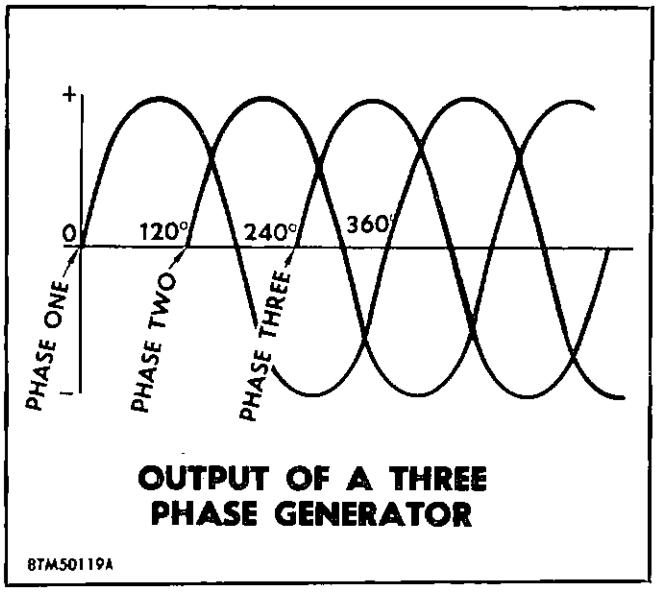

Figure 1-33. Output of a Three-Phase Generator

THE THREE-PHASE GENERATOR.

Because of the heavy current taken from the armature windings in a three-phase generator, and because from four to six brushes are usually needed, the armature in this type generator is usually stationary. It is the magnetic field that rotates, producing three phases of voltage with but one set of brushes. The wave forms of this voltage differ from each other by one-third of a cycle, or 120°. The waveform chart in figure 1-33 illustrates this difference in output.

As previously mentioned, the principle difference between an a-c generator and a d-c generator is the method used in connecting their external circuits; the a-c generator is connected to the slip rings, the d-c generator to commutators. An a-c generator and a d-c generator then are identical in the way they generate voltage in a rotating loop. If the current is taken from the loop by slip rings, you have alternating current and the generator is called an a-c generator. If the current is collected by a commutator, you have direct current and the generator is a d-c generator.

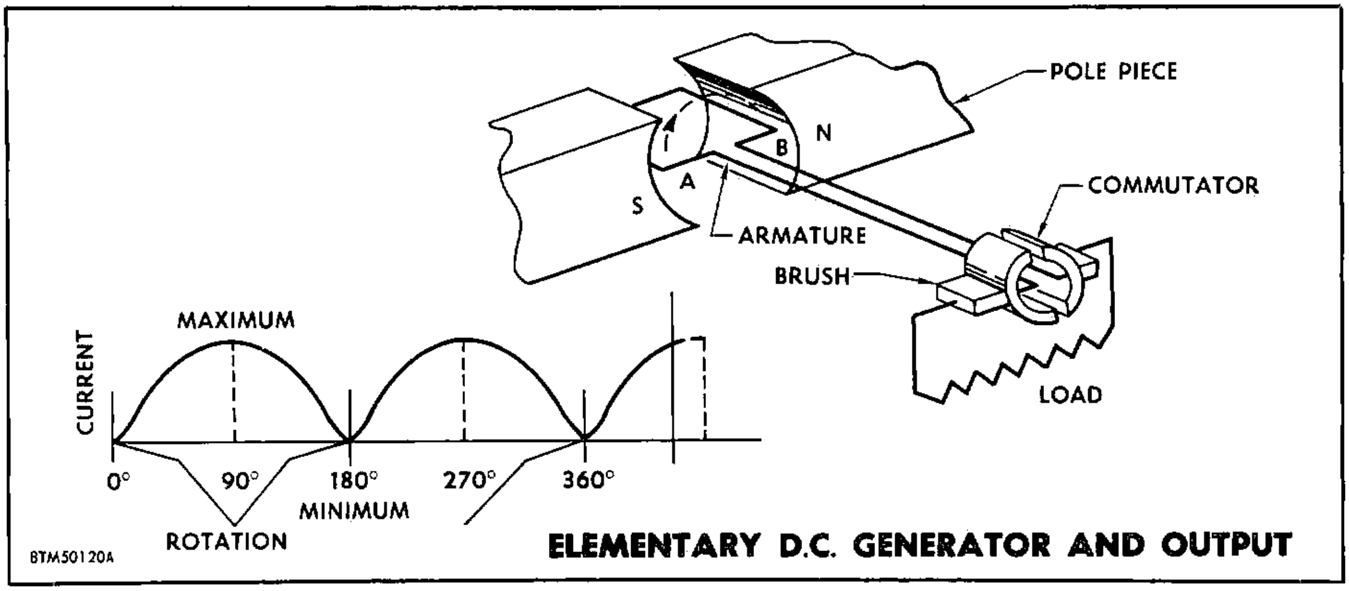

Figure 1-34. Elementary D-C Generator and Output

D-C GENERATOR.

As you were previously told, to convert our simple alternating-current generator into a direct-current generator, it is necessary only to replace the slip rings with a commutator. A commutator is a ring cut in two halves. These halves are insulated from each other and are connected to one end of the generator coil. The commutator serves to reverse the connections of the loop to the external circuit. This is achieved by means of brushes used at the instant that the polarity of the induced voltage in the loop coil reverses. This can be seen in figure 1-34 which shows an elementary d-c generator and its output.

Now, how exactly is a direct current produced? If the brushes we have mentioned are set so that each half of the commutator moves out of contact with one brush and into contact with the other, right at that point when the loop is passing through the positions we used to illustrate one cycle of an a-c rotation, and when the induced voltage is at minimum, the generator will produce a d-c voltage. In this particular generator, however, the current is not smooth; it pulsates, as can be seen in the waveform chart of figure 1-35.

The minimum point of contact with our imaginary line is sharp and reverses positively. It is not sweeping from positive to negative as is the curve in an alternating current output. How is this compensated for? If we had, instead of just two commutator segments and a single loop, many commutator segments and a number of coils —with the end of each coil connected to a different segment and provided the coils are so placed that several of them are always in the maximum position—the generator would generate a much more steady current (see figure 1-35). This is the d-c generator we are most interested in; so let's call it a practical d-c generator.

Principles of Practical D-C Generators.

There has been a lot said so far concerning electromagnets. In fact, the main heading of this section of the manual is: The Production of Electromotive Force by Electromagnetic Induction. The simple generators that we have discussed operate with an armature consisting of a loop of wire rotating between the poles of permanent magnets. And as you know, a permanent magnet can be made by striking a piece of hardened steel with a natural magnet. This permanent magnet can be used to magnetize another piece of steel, and the process can go on indefinitely. It is the electromagnet in an airplane generator which supplies the magnetic field.

This electromagnet is made up of a laminated iron that is capable of strong magnetism. Remember our discussion concerning hysteresis? The hysteresis loss due to heat, when our molecules are in a state of hysteria, is due to being aligned first one way and then in the other. It is the lamination of the electromagnet which cuts down on this hysteresis loss; and on these laminations, called *pole pieces,* the field windings are placed. These pole pieces are then mounted to the field frame which forms part of the field's magnetic circuit. The current for the electromagnet is supplied by the generator itself.

Four-Pole Field Structure.

Figure 1-36, shows a four-pole field structure along with its resulting magnetic field. The electromagnets consist of the coils, which we mentioned above, wound on a soft iron core. By varying the applied voltage, it is possible to vary the strength of the field of the electromagnet. This process controls the output of the generator.

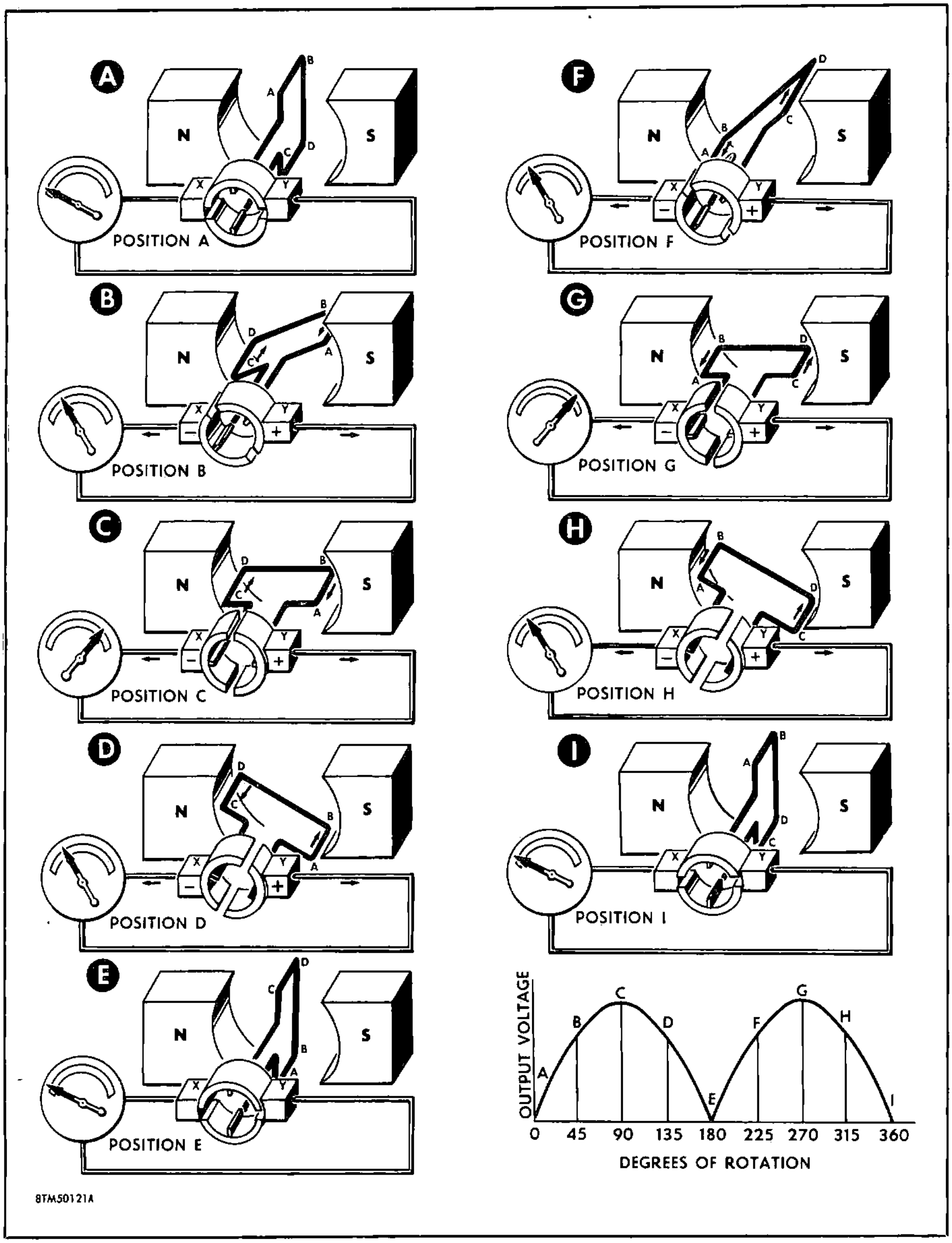

Figure 1-35. Cycle of D-C Generator

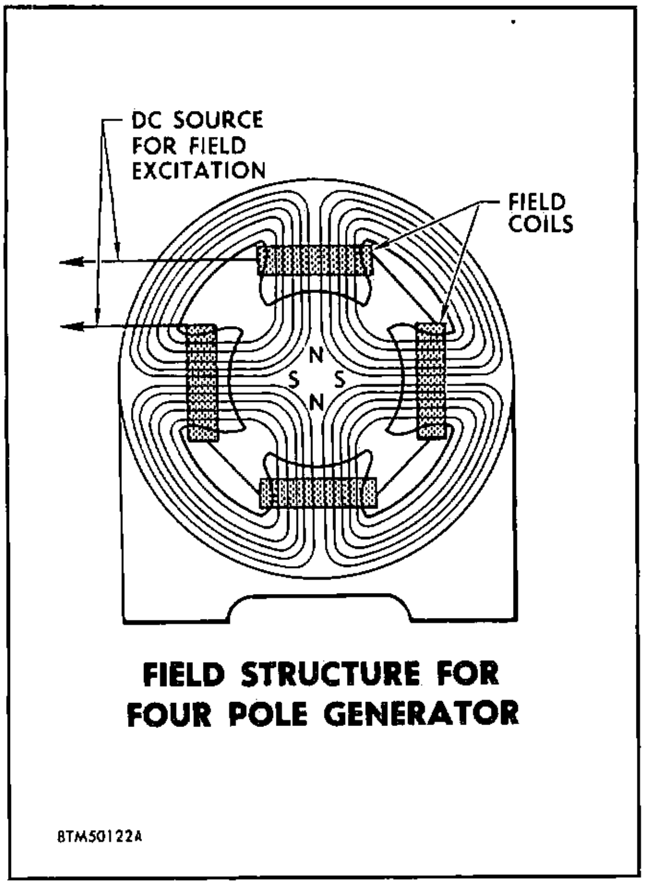

Figure 1-36. Field Structure of a Four-Pole Generator

BRUSHES. Usually, the number of brushes in a generator is equal to the number of poles. Each brush is located approximately halfway between each pair of poles. The positive brushes are connected together to form the positive output, and the negative brushes to form the negative output. This arrangement in a four-pole generator provides four parallel paths through which the armature current flows, and also lowers the internal resistance of the machine. The voltage generated is the voltage of any of the parallel paths. The current capacity is equal to the sum of the current within the four paths.

THE SELF-EXCITED GENERATOR. A generator which supplies its own field current is called a *self-excited* generator. The reason for this term is that when the power plant in an airplane is started, the generator armature turns over cutting a small amount of magnetic flux (those lines of force) which has been retained in the pole pieces. This residual magnetism causes a small induced voltage to be developed in the armature. The current flow from this voltage, naturally, strengthens the magnetic field, thereby producing a higher induced voltage in the rotating armature. The higher voltage results in a still larger current in the field coils. Unlike our simple battery circuit flowing through the coil and held back by its counter voltage, the aircraft generator builds up to its final voltage and current value almost instantaneously. Self-excited generators are classed according to the type of field connections they employ. We shall get to generator classification in a moment.

ARMATURE WINDINGS. The more armature windings an airplane generator contains, the greater the generated voltage and the more constant is its output. Constancy of output is what we want. This is achieved by the appropriate spacing of these windings. And when carefully engineered, an almost steady direct current is developed. There are, however, slight pulsations still in evidence, but these cause no serious effects for most power purposes. These are called *commutator ripples* and cause considerable disturbance in the radio equipment. These ripples are eliminated by a condenser or capacitor, connected across the generator. The condenser builds up the low values of the output voltage and lowers the higher voltages, thereby equalizing the rippling and smoothing the voltage to a point where radio interference is minimized.

GENERATOR RATINGS.

An airplane generator, or for that matter any generator, is designed to operate at a specified voltage. The rating is usually given as the number of amperes the generator can safely supply at its rated voltage. For example: the F-102A direct-current generator operates at 30 volts and delivers to the buses, at the minimum rate of speed of 4000 rpm, 200 amperes of current. To further describe it; it is a four-pole, shunt-wound generator which has interpoles, and compensating and equalizing armature windings. The regulation of the generator output is described in Chapter II, of this supplement. Let's take a look at some types of generators.

TYPE CLASSIFICATION OF D-C GENERATORS.

There are three types of d-c generators: series, shunt, and compound. These classifications are given them according to the relation of the field windings they employ to their external circuit.

Series-Wound.

In the *series-wound* generator, the field coils are connected in series with the armature. The current that flows in the load (the external circuit) also flows through the field coils. This type generator has such poor regulation that it is never employed in an airplane. But it might be a good idea if we learned why it has poor regulation and why it is not used as such, for it may help us understand the other types better.

Briefly then, the field coils in this type generator are composed of a few turns of large wire, which make them very low in resistance. The magnetic field strength is obtained from the current flow rather than from the number of turns in the coils. It is for this reason, that series-wound generators have very poor voltage regulation under a changing load.

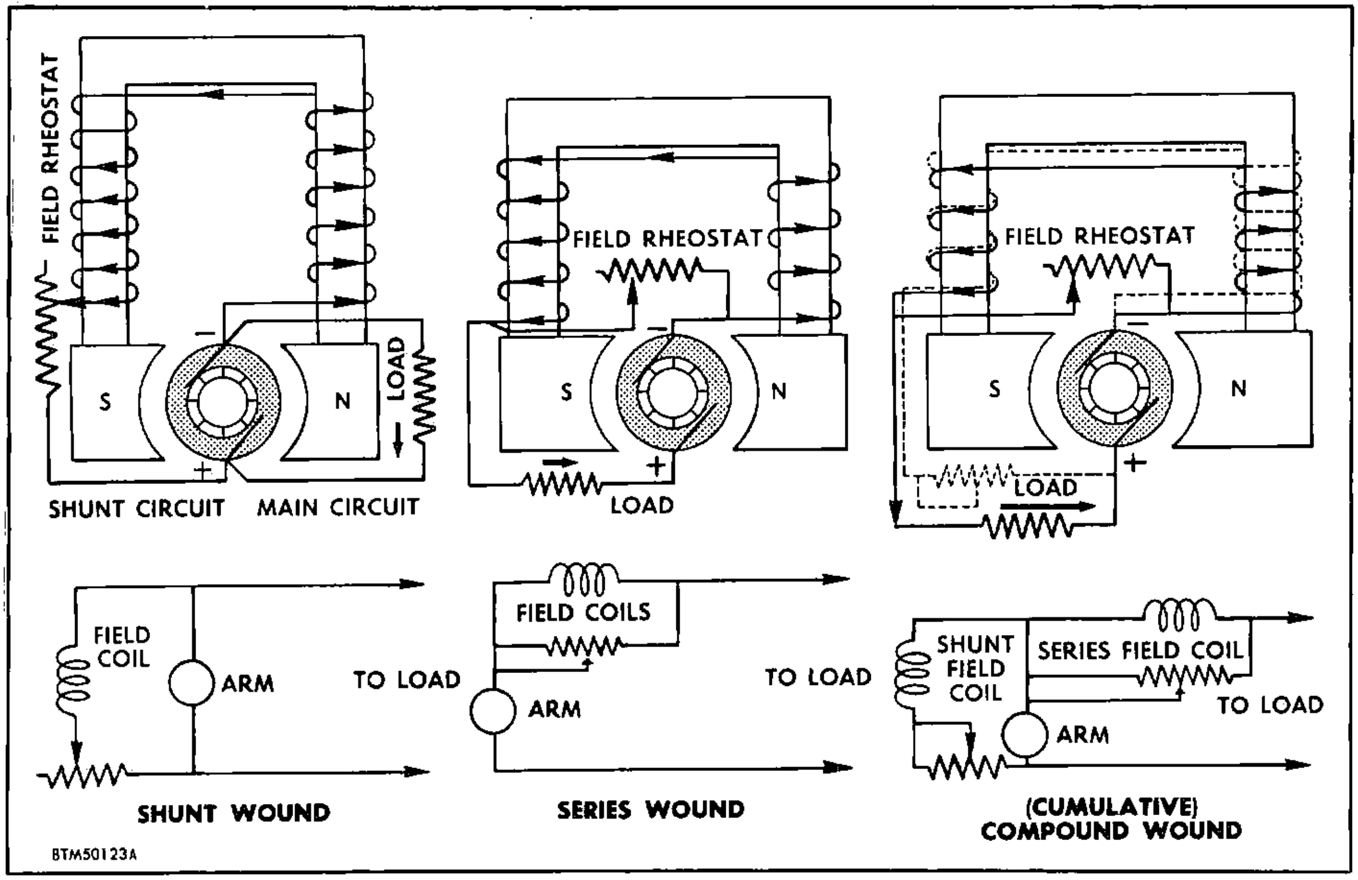

Figure 1-37. Types of Generators

Let's explore this a bit further. The greater the rate of displacement of current flowing through the field coils to the load, the greater is the emf and the *terminal voltage*, or *output voltage*. In other words, when the load increases the voltage increases; and when the load is decreased the voltage is decreased. Thus with such variation in the strength of the magnetic field and resulting lack of a constant output, the series-wound generator cannot be used in airplanes—the electronic communication and navigation systems require a constant voltage and this alone is reason enough for not using this type generator.

Shunt-Wound.

The word "shunt" means to by-pass. A generator whose field windings are connected in parallel with the armature terminals is called a *shunt-wound* generator. Its field coils contain high resistance windings, consisting of many turns of small wires, and produce a maximum voltage output only under no-load conditions; consequently, the voltage drops with the load. There are two reasons for this undesirable voltage drop under load.

First, all of the increase in the load current flows through the armature. And if the armature resistance (R) is constant, the increase in current makes the IR drop greater. And as you recall, IR drop = the current × resistance. In this case the internal resistance of the armature decreases the output voltage.

The second reason is a result of the first reason. This lower voltage allows the field current to decrease, thereby decreasing the magnetic field. Thus, the output voltage drops a little more, and this again is a disadvantage because of the variation in output voltage.

Compound-Wound.

The illustration figure 1-37, will give you a clearer picture of what we have been talking about in the series and shunt-wound generators, as well as the one we are about to discuss.

A compound-wound generator has *both a shunt and a series field*. The shunt field coils are connected across the external load as in the shunt-wound generator illustrated in figure 1-37. The series winding is connected in series with the external circuit. In other words, both series and shunt fields are employed simultaneously. These series windings contain from a fraction of a turn to only a few turns of heavy wire. These are wound on the pole pieces so that the magnetic flux they produce is added to the flux produced by the shunt windings. Therefore, since the series field is in series with the external load circuit, the same amount of current flows in the load circuit as flows in the series field winding. When the load is increased, more current flows through the series field winding, thus causing an increase in the strength of the field in which the armature rotates. An action of this sort tends to increase the generator output; and by the same token, any losses due to voltage drops in the generator increase as the load increases; however, the terminal voltage remains the same or constant. This is exactly what we are looking for in an airplane generator. The shunt-wound generator alone is not suitable for rapidly fluctuating loads if a constant voltage is desired. But, in the compound-wound generator, and this is its principle advantage, we have a generator whose terminal voltage varies less with a load than do any of the others.

SUMMARY.

In this chapter you learned why your exact electrical knowledge is essential to maintain the F-102A. You were given a brief resume of what you will encounter in its electrical systems. And then, as a review, the three branches of electricity were covered. Next came energy, work, and how the electrical nature of things are put to work. This introduced the nature of the molecule, the atom and its structure, and electricity's most important product: the *free electron*—the smallest particle of negatively charged electricity known to man. Then, we reviewed how these electrons work when electromotive force is produced by chemical reaction—the storage battery, or the blood bank for the F-102A d-c electrical power system.

We then returned to a broader study of magnetism—the atom again—because without an understanding of electromagnetism, our final section of the chapter would have been worthless. So at this point, in our study of the F-102A electrical power systems, we should know that electrical energy is highly *convertible, conveyable,* and *controllable*. We know what makes it, how it is made, and where it comes from—we have it in the palm of our hand. Let's put it to work; let's take this electromotive force, difference in potential, voltage—the electrical current—and find out how it is distributed through the electrical systems of the airplane.

How it is conveyed and how it is controlled and converted once again, will be the primary concern of Chapter II. Again, keep in mind: *on your electrical knowledge hangs the successful completion of a mission — the safety of the pilot and the airplane itself.*

Chapter II

F-102A D-C POWER DISTRIBUTION AND CONTROL

Contents *Page*

The best power and distribution system for any airplane is one tailored to meet the particular demands of the particular airplane. This is true of the F-102A, and as we said earlier, new developments make new demands on men. How reliable an airplane's power system is depends on the inter-connected network which distributes the electrical current demanded by this or that circuit. This is particularly true of systems such as those the F-102A carries—where there are two or more routes of power to the important loads; and when an emergency arises, the airplane can still function efficiently even though this efficiency is limited to minutes. But, when we think of the speed at which the F-102A travels and its cruising radius, even 12 minutes can be a lifetime. And if you will remember, this is the maximum life of the storage battery in the F-102A's d-c power system under ideal emergency conditions.

In the preceding chapter, you learned that whenever a conductor moves in a magnetic field in such a way as to cut lines of force, voltage is induced in the conductor. When the output from the conductor is taken from slip rings, you will recall (from our discussion of generator principles in Chapter I) that the output is an alternating current. In an alternating current, the voltage rises and falls, and reverses polarity every half cycle. The magnitude of the voltage is also constantly changing.

On the other hand, when a voltage is induced into a conductor and the resulting output is taken from a commutator, the current is constant in magnitude and direction, and we have direct current. Direct current changes its magnitude only when the circuit is open or closed. So that the wave forms taken by these two currents remain firmly fixed in your mind, let's compare the wave forms shown in the illustration, figure 2-1.

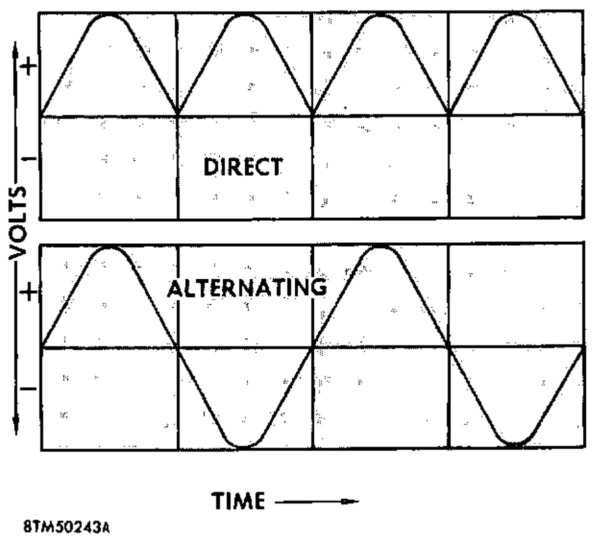

Figure 2-1. Wave Forms of Direct and Alternating Current

The differences in these wave patterns are due to the difference between the output of a direct-current generator and alternating-current generator and the change of polarity in their brushes. In a d-c generator the polarity of the brushes remains unchanged. In an a-c generator on the first half of its cycle, or the positive alternation, the polarity of the brushes is positive; in the second half of its cycle, or negative alternation, the polarity changes to negative.

POWER DISTRIBUTION IN THE F-102A.

The power distribution systems in the F-102A are fundamentally the same as those used in other modern high speed, high altitude aircraft. In Chapter I, the power systems installed in the F-102A—their sources and controls—were given you in capsule form. The reason for such a thumb-nail sketch of these systems was to familiarize you with what we would be talking about in this manual. From now on we will give you knowledge of them and how they function. You already know that electrical energy is convertible, conveyable, and controllable; you know what makes it, how it is made, and where it comes from. In this chapter you will learn how it is distributed through the electrical systems of the F-102A. And keep in mind that the more deeply rooted you are in all things of which distribution systems are comprised, the better able you are to perform your job.

In Chapter I, we said that the F-102A uses a conventional 28-volt, single-wire, ground return direct-current system. What is meant by this? Simply that when an airplane is properly bonded, the metallic structure of the modern airplane provides an excellent return path for most of the circuits used. The voltage drop of the path *for this system* is negligible. The main advantages in such a system should be clear enough—simplicity and the weight saving factors. And as you probably know, a large part of the weight in any aircraft electrical system is in its wiring. The principle behind this system is shown in figure 2-2 which depicts a simplified version of a common single-wire d-c system.

The F-102A also uses a 115/200-volt, 3-phase, 400 cycle a-c system, and a 26-volt, single-phase, 400 cycle a-c system. It receives its initial voltages from a 30-kva, 120/208-volt, 400 cycle generator. Just what does this mean? If you recall, in an a-c system a number of different voltages are required. We obtain these voltages with transformers, as needed within the power systems.

In the case of the F-102A, two transformers are used; for example, the voltage requirements of certain instruments such as the fuel flow transmitter and the hydraulic pressure indicator are much lower (26V) than those (115V) required by the attitude gyro or the instrument panel lights. The type of generator used when a single-phase load is required, is usually a 208-volt, 3-phase, 3-wire system, having a voltage from line to ground of 120 volts. This is essentially the generator used in the F-102A's a-c system. For its emergency a-c supply, a 1-kva, 120/208-volt hydraulically driven generator is used. The d-c system, in which we are particularly interested at this point, is supplied by a 30-volt, 24 ampere-hour, lead acid-cell battery for emergency power. We discussed the fundamentals of this battery and the factors governing its life in Chapter I.

During normal inflight operation, both the a-c and the d-c generators furnish regulated power to the essential and non-essential buses of the power distribution systems. This power is distributed by means of feeder lines to the essential and non-essential buses. These buses in turn supply the electrical power through connecting wires to the operating electrical components throughout the airplane.

Stick your head into any compartment of the F-102A and you will find thick bundles of wire, all numbered and all identified by code. These bundles are distributing the power needed to bring the F-102A to life with enough wire to electrify fifteen average modern homes. Today, so many units of an aircraft are operated by electricity that the electrical systems are as important as the nervous system in the human body; and the F-102A is what might be called a unit airplane.

This chapter will deal with the distribution of d-c power in the F-102A airplane the circuits used and their protection and the d-c generator system and its control.

D-C POWER CIRCUITS.

In Chapter I, you learned that if a greater voltage is needed, a number of cells in a battery are connected in series—the negative terminals of each cell to the positive terminal of the succeeding cell. Now what about the current? Is it higher, or does it stay the same? If the voltage of a battery is equal to the total voltages of the individual cells, why isn't the current proportionately greater? This shouldn't be difficult to figure out when you know that the unit of emf is the volt and that

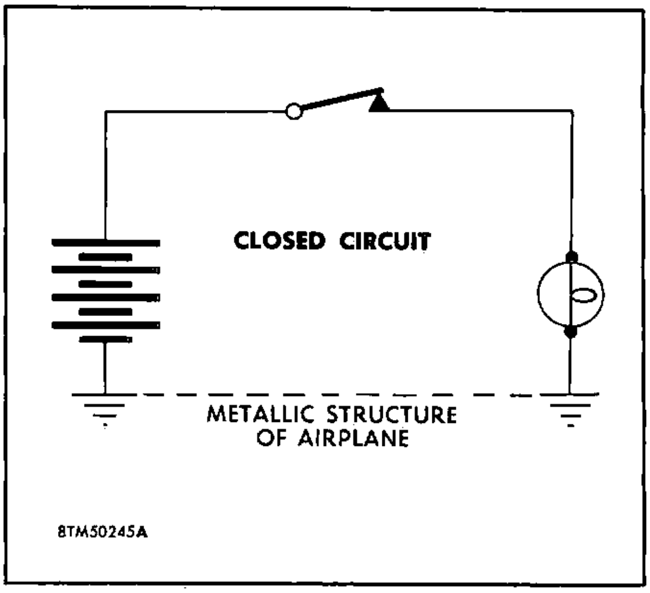

Figure 2-2. Simple D-C Single-Wire Ground Return System

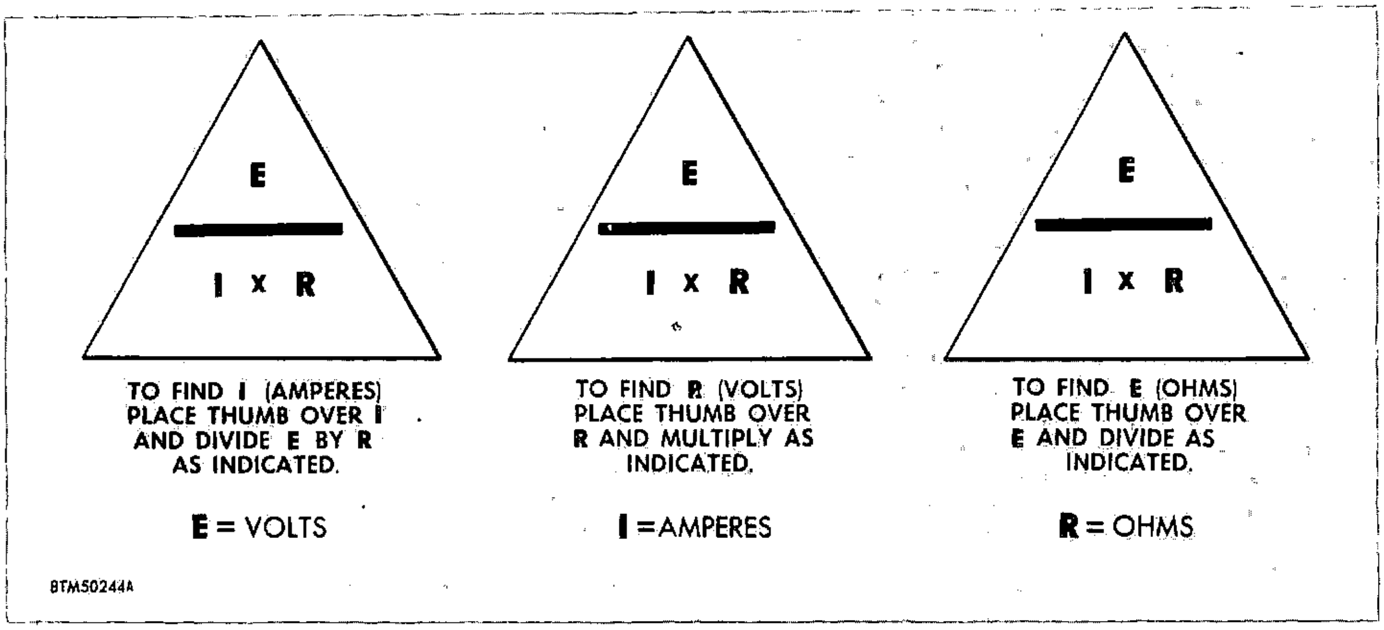

Figure 2-3. Ohm's Law Chart

a volt is the amount of electrical force or pressure required to maintain a current of one ampere (6.28 billion billion electrons) through a resistance of one ohm. The current a battery can supply, when connected in series, is equal only to the current of the original cell.

You also learned that when a greater current is desired, the cells are connected in parallel. However, in this arrangement the voltage is only equal to the voltage of one cell, and the total current is the sum of the individual currents. Why is this? Because, unlike the series connection, in the parallel connection the current of one cell does not flow through the other cells. All the positive (+) terminals are connected together and all negative (−) terminals are connected together to form but one positive terminal and one negative terminal. Each cell must have the same voltage, otherwise a cell with a higher voltage will force current through the lower voltage cell and thus carry the greater part of the load. Combined, these two connections are what is called a series-parallel connection which provides *both* a greater voltage output and a greater current output.

You found, when we discussed and classified generators in Chapter I, that series-wound generators lack constant terminal output voltages. You also learned that in the shunt-wound or parallel connected generator there are two causes for the instability of voltage output: the IR of the armature decreases the output voltage, and this lower voltage decreases the field current and in turn the magnetic field. The result—there occurs another drop of output voltage. If, however, we combine the series and shunt-wound characteristics we have a generator whose terminal voltage is fairly stable, and as you know, this is a series-parallel wound generator.

We have reviewed briefly the basic principles behind series, parallel, and series-parallel connections within a battery and a generator. The magnitude of the electromotive force, voltage or difference in potential, produced by these two processes are what we are most interested in, and how the resulting flow of free electrons are pulled from a point of low potential to a point of high potential through an *external circuit.* Let's divide the external d-c electrical circuits into three general classifications and discuss them—series, parallel, and series-parallel.

SERIES CIRCUITS.

The simplest form of an electrical circuit is the series circuit. In this circuit the current flow is the same at all points in the circuit. When you were first introduced to electrical fundamentals in electrical school, you learned there was a definite relationship between the voltage, current, and resistance of any electrical circuit or part of any electrical circuit. This relationship is expressed through the Ohm's law. The chart shown in figure 2-3 will help you to refamiliarize yourself with its principles.

In the series circuit shown in figure 2-4, three resistances, represented schematically by the symbol-∿∿, are connected in series across a 24-volt d-c battery. As can be seen, series circuits are those having only one closed path for the flow of electricity. Let's rework this simple diagram and apply it to our single-wire, d-c, ground return system as illustrated in figure 2-2. This reworked diagram gives us a series ground return system as shown in figure 2-5. In both diagrams there is but one path for the current to follow, and even though the current is forced through each resistance, the *current* values are the same throughout the circuit.

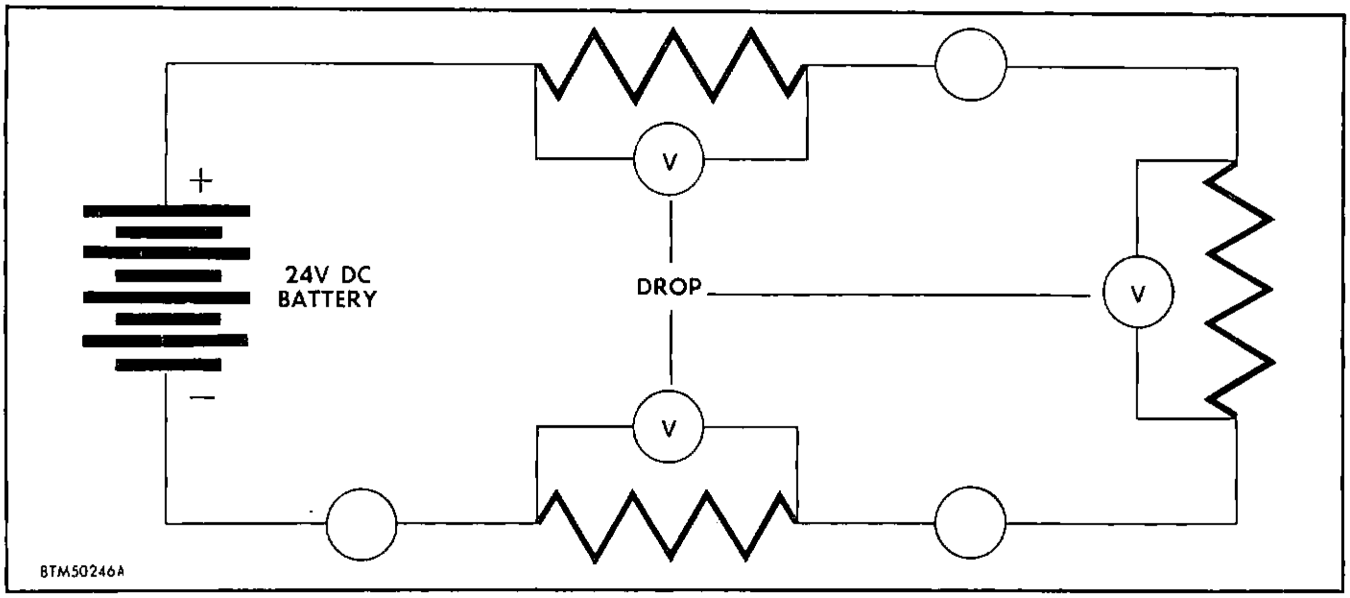

Figure 2-4. Series Circuit

The total voltage drop in the circuit is equal to the sum of the voltages across each of the resistances. Let's assume that R_1 has a resistance of 2 ohms, R_2 has 3 ohms, and R_3 has 1 ohm. What would the voltage drop be across each resistor? We know that the voltage is 24-volts d-c, and we know that there is a resistance of 2 ohms in R_1. According to our Ohm's law chart, the method of solving this problem be: $E = I \times R$. But we do not know I nor do we know the current. Well, let's figure the current first: $I = \frac{E}{R}$, in this case, so, $I = \frac{24}{6}$. Six being the sum of *all* resistance within the circuit, divided into emf which we know is 24 volts.

So our current value is four amperes or 1/6 of the voltage. To get back to our original problem, to find the voltage drop across each resistor, we would use:

$$E = I \times R$$
$$E = 4 \times 2 = 8 \text{ volts}$$
$$E = 4 \times 3 = 12 \text{ volts}$$
$$E = 4 \times 1 = 4 \text{ volts}$$

Adding these together we have the sum of the original power source, 24 volts.

What does all this mean? Simply that the current (I) in a series circuit is always the same in all parts of a circuit;

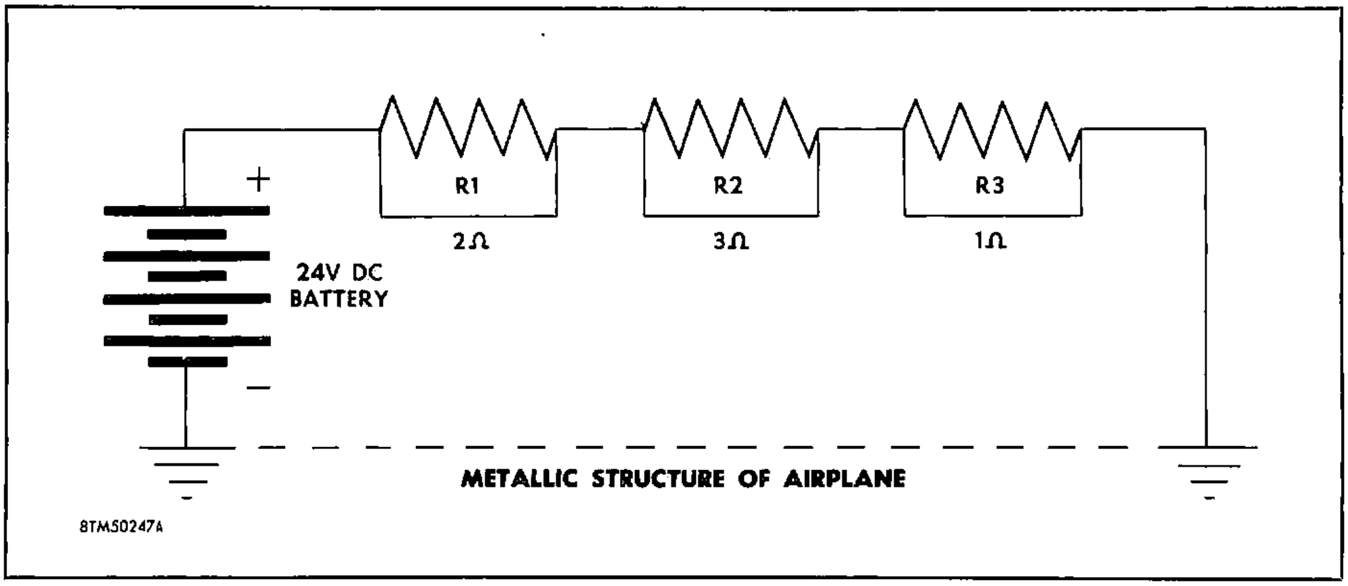

Figure 2-5. Series Ground Return System

the total voltage (E) in a series circuit is equal to the sum of the voltages across the different parts of the circuit, and that the total resistance (R) of a series circuit is equal to the sum of the resistances connected in series.

Now suppose we have, within our series circuit, a lamp or a motor and one or the other malfunctions. What happens to the current flow? It is, of course, interrupted completely and the other devices in the system fail to operate.

Resistors.

Before we become involved in a discussion of parallel-circuits, let's get squared away on resistors and their function within an electrical circuit.

As you know, any opposition to the flow of current by conductors or the electrical devices within a circuit, is called *resistance*. In Chapter I, we learned that: substances characterized by their content of free electrons are called *conductors;* those characterized by the lack of free electrons are known as *insulators*. There are other factors, however, besides the electron content of a conductor which decide the amount of resistance a conductor has— namely its length, cross-sectional area or circumference, and temperature. Regardless of these natural resistant qualities of conductors, it is usually necessary to reduce the amount of voltage applied to devices in a circuit and thereby reduce the rate of electron flow. This unit which reduces the amount of voltage is called a resistor.

As you can see in figure 2-6, a resistor is usually cylindrical in shape. Its construction varies with the resistance demands. Some are sections of carbon rod, some are combinations of carbon and other substances, and others are wire wound. One common form of a resistor is a length of high-resistance wire wound about a ceramic core which is usually an insulating, heat-resisting material.

Resistors are marked by a color coding with the ohm value and percentage of tolerance, or power rating, printed on its coating. We shall discuss power and power rating after we deal with the characteristics of the parallel and series-parallel circuits. Resistors may be of the *fixed type* in which their value cannot be raised, or of the *variable type*. Variable resistors are called *potentiometers*, or *"pots,"* and *rheostats*—both are adjustable to any amount of resistance within their capacity.

Before going into more complex circuit analysis, we would like to interpose two laws for governing electron flow and voltage which we hope will improve your knowledge of electrical circuits. You are familiar with the basic concepts of the Ohm's law and both of these fundamental concepts are an outgrowth of that law.

Law of Continuity of Current.

In our diagram of an undivided circuit or simple series circuit there are a number of ammeters placed at intervals within the circuit. At each point in the circuit these readings would be identical. This is the proof of the pudding as far as the law of current continuity is concerned—to wit, an electrical current in an undivided circuit is the same at all sections of the circuit. Now, how does this law affect a divided circuit? Let's look at the illustration figure 2-7, which shows a divided or parallel circuit. Here again we have ammeters distributed throughout the circuit. What does this prove? Simply the second part of the continuity law of current. The sum of R_1, R_2, and R_3,—the branches of a divided circuit—is equal to the current of the undivided part of the circuit, or

$$R_1 + R_2 + R_3 = L_1$$

Kirchoff's Laws.

A fellow by the name of Kirchoff, sometime between 1824 and 1887, formulated from the Ohm's Law and the law of current continuity, two laws which can be used to solve any problem which may arise in an electrical network.

We saw when discussing a simple series circuit how circuit values are arrived at by applying the Ohm's law. This is not only very difficult in a complex network of conductors, generators, and other electrical devices such as we have in the F-102A, but often impossible. Granted, these computations and engineering headaches may not be your concern at the moment. But they might be!—after all this is a manual which is supposed to tell you *how* a thing works, *why* it works, and *what* it does when it works. So it is important for your knowledge of the strange behavior of electricity to understand what Kirchoff was talking about—it is very simple.

KIRCHOFF SAID, ONE. The usual *conventional* signs to take current toward a junction point of a load element in a circuit or away from a junction point of a load element are: plus (+) when current is moving toward, and minus (−) when current is moving away. Let's actuate the master switch in the cockpit of the F-102A.

After the switch has been closed and the current has become constant in all branches of the network, Kirchoff says as much electricity must flow away from any junction in a second as flows toward it. In other words, in *any* circuit the total amount of current leaving a given point must be exactly equal to the total amount approaching that point.

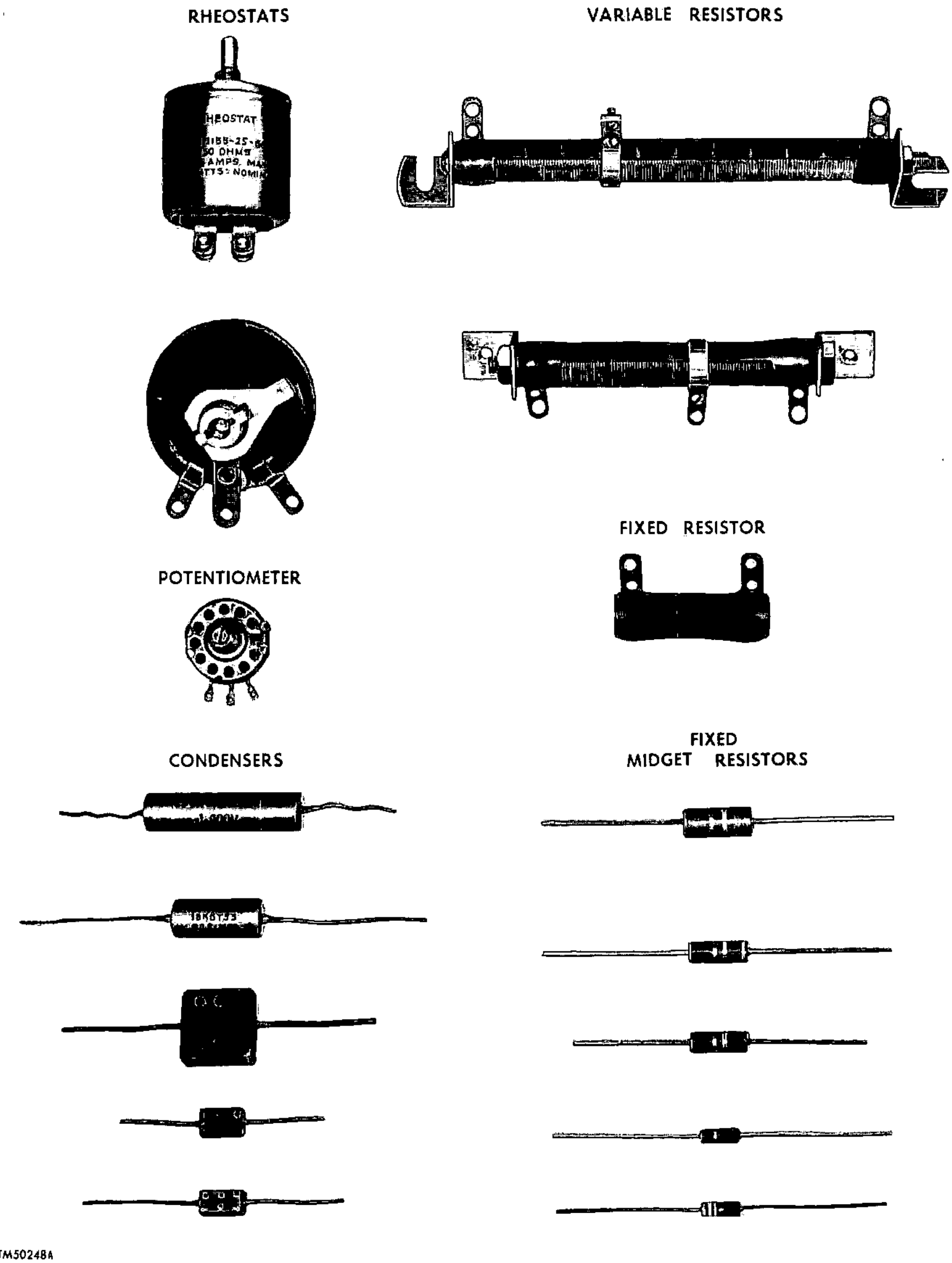

Figure 2-6. Typical Resistors

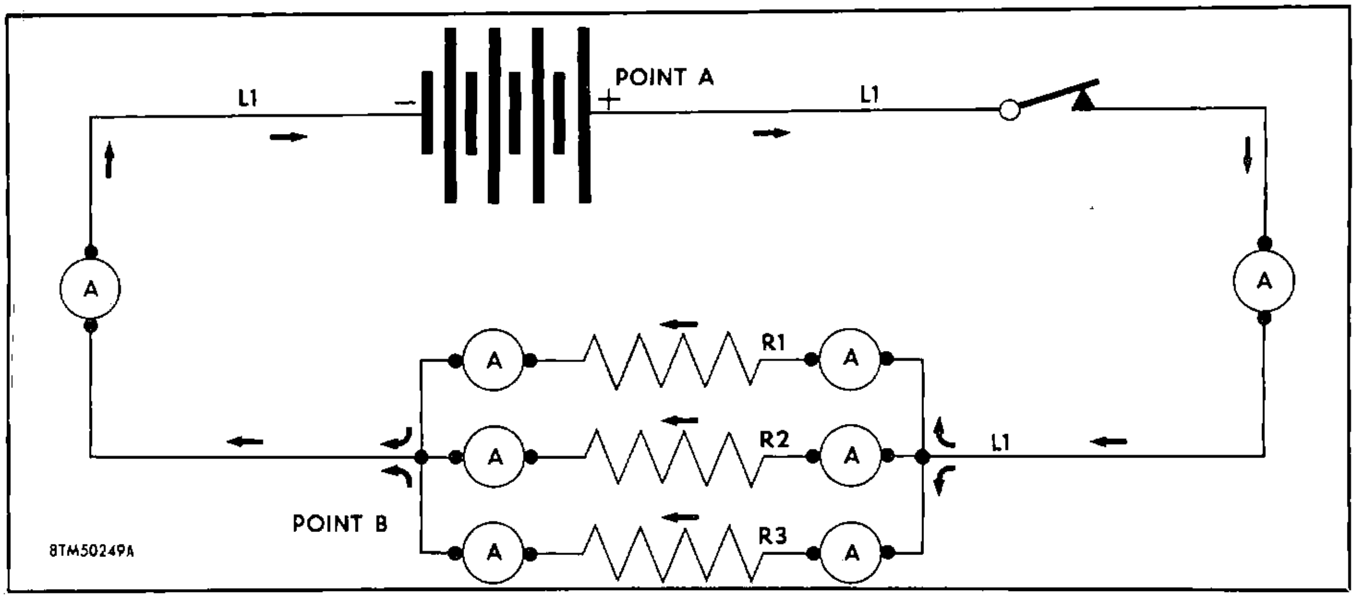

Figure 2-7. A Divided or Parallel Circuit

Taking a look at our diagram of a divided circuit (figure 2-7) we can see R_1, R_2, and R_3 are connected so that *part* of the current supplied by the battery goes through each of them. Kirchoff says, the current at point B must equal the sum of the currents through R_1, R_2, and R_3. And this law applies regardless of the number of resistors.

KIRCHOFF SAID, TWO. When we discussed the simple series circuit we found that the voltage drop across each resistor is equal to the original voltage of the circuit. For example, we had a 24-volt battery connected across three resistors—R_1, 2 ohms; and R_2, 1 ohm. You found a drop of 8 volts across R_1; 12 volts across R_2; and 4 volts across R_3. The sum of these drops equalled 24 volts, and $24-24=O$. Well, this is how Kirchoff formulated the second part of his law: that the total voltage drop around *any complete path* is equal to zero. To express it another way: the voltage between two points in a circuit is the same, no matter what path is taken to go from one point to the other. Now, let's take up the parallel circuits.

Parallel Circuit.

Most of the electrical devices in airplane electrical circuits are connected in parallel. In a parallel circuit two or more electrical devices provide independent paths through which our busy electrons flow. And, as we have learned from Kirchoff, the voltage across each device in parallel is equal, just as the total current in the circuit is equal to the sum of the currents flowing through all devices.

If you remember, Ohm's law says that if a voltage is increased the current *increases* proportionately, and if the resistance is increased the current *decreases* proportionately. Therefore, if the total amount of current in a parallel circuit is greater than the current in any individual part in the circuit, the total resistance of the circuit—as a whole—must be less than the smallest resistance in it. To clarify this: the more electrical devices or resistors connected in parallel, the greater the total current and hence, the smaller the resistance of the complete circuit.

Electrical devices are connected in parallel to decrease the total resistance and to allow them to operate independently. If one device in a parallel circuit burns out, the others can still operate. For example consider your Christmas tree lights. When a bulb burns out in one of the less expensive string of lights, they all go out — this is a series connection. But, if a bulb burns out when they are connected in parallel, the others in the string continue to glow.

In the illustration figure 2-6, we have the parallel circuit we've been discussing. To find the total resistance of a parallel circuit the following method is used. Only two paths are solved mathematically at a time.

$$R_T = \frac{\text{Resistance of 1st unit} \times \text{resistance of 2nd unit}}{\text{Resistance of 1st unit} + \text{resistance of 2nd unit}}$$

For example: in the above diagram three load units (resistors) are connected in parallel. If we want to find the total resistance for the first two paths, R_1 and R_2, we substitute their values in the method used for finding the total resistance:

$$R\text{ (1 and 2)} = \frac{R_1 \times R_2}{R_1 + R_2} = \frac{12 \times 4}{12+4} = \frac{48}{16} = 3\text{ ohms}$$

Remember we said that, ". . . *if the total amount of current in a parallel circuit is greater than the current in any individual part in the circuit, the total resistance of the circuit as a whole—must be less than the smallest*

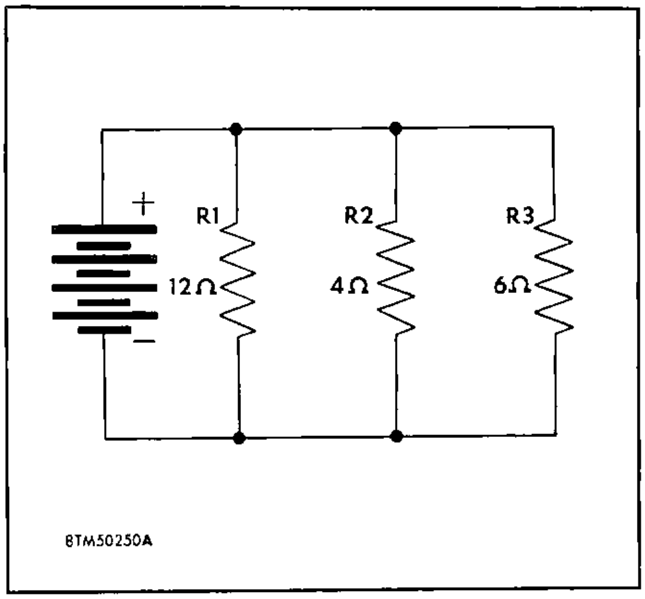

Figure 2-8. A Simple Parallel Circuit

resistance in it." Well, in this instance 4 ohms was the smallest resistance in the circuit. So what do we do now? A 12—and a 4-ohm resistor is not necessary in this case, we simply substitute a 3-ohm resistor and redraw the circuit as illustrated in figure 2-9.

Now what is the total resistance?

$$R_T = \frac{R\,(1+2) \times R_3}{R\,(1+2) + R_3} = \frac{3 \times 6}{3+6} = \frac{18}{9} = 2 \text{ ohms}$$

As you see the total resistance of the circuit is now only 2 ohms, less than the resistance of any of its individual paths. If we redraw this diagram we would have what is called an equivalent circuit to the one above. Now, what is an equivalent circuit? Well, it is nothing more than a circuit that could replace a particular circuit under certain conditions—a circuit of equal values. To carry this a bit further, if one circuit can replace another circuit—without altering in any way the electrical operation of the electrical system external to that circuit—they are said to be "of general equivalence."

When the load units in parallel have equal resistance, we reduce our method to:

$$R_T = \frac{\text{resistance of one unit}}{\text{the number of units}}$$

For example, in a parallel circuit with six, 5-ohm load units the ohm value would be:

$$R_T = 5/6 \text{ ohm or } .08333 \text{ ohm}$$

Equivalent Circuits.

For emphasis, let's trace the circuit shown in figure 2-10 back to our equivalent circuit just discussed. In this diagram there are three lamps connected in parallel across a 24-volt source of power. The voltages were measured with a voltmeter and the current measured with an ammeter—they were found to be as indicated.

Let's find the voltage drop across each lamp, the total current, the resistance of each lamp, and the total resistance. In the first place, a voltage in a parallel circuit is the same across each unit. Why? Because Kirchoff said, ". . . the voltage drop across each unit in a parallel circuit is equal to the original emf." If you checked the voltage with a voltmeter—in parallel with the circuit—you would find that our friend Kirchoff was correct. By the same token, the total current (let's call it I_t) in a parallel circuit is equal to the sum of the currents in each path; so referring back to our diagram we have:

$$I_t = I_1 + I_2 + I_3 = 2 + 6 + 4 = 12 \text{ amps}$$

MR. OHM AGAIN. To find the resistance of each lamp in the circuit, we return to our friend Ohm and divide the voltage across each lamp by the current which flows through it. If you refer to the Ohm's Law chart, this will read:

$$R = \frac{E}{I}$$

Therefore: R sub 1, would be equal to E sub 1 over (divided by) I sub 1, and with the numerical values substituted for the alphabetical values it would look something like this:

$$\frac{24}{2} = 12 \text{ ohms}$$

In the same manner:

$$R_2 = \frac{E_2}{I_2} = \frac{24}{6} = 4 \text{ ohms}$$

$$R_3 = \frac{E_3}{I_3} = \frac{24}{4} = 6 \text{ ohms}$$

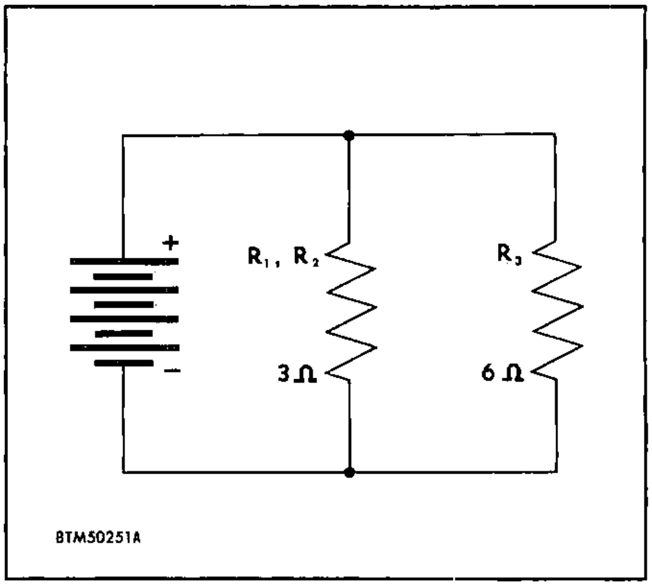

Figure 2-9. An Equivalent Parallel Circuit

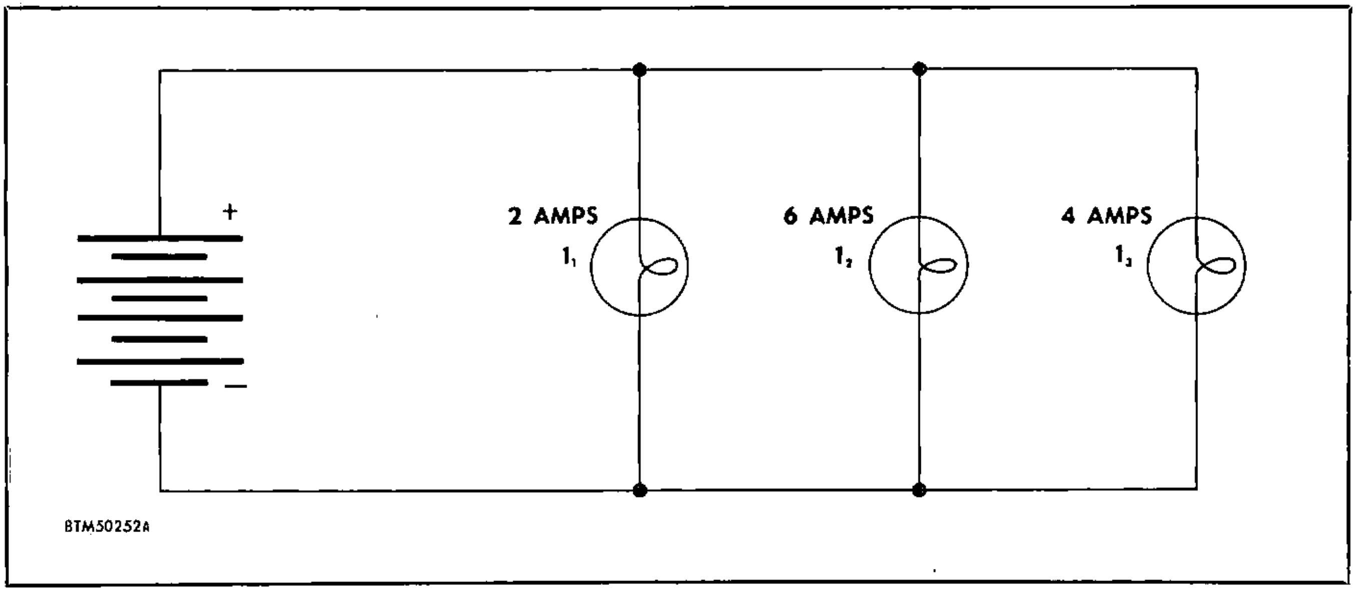

Figure 2-10. Lamps Connected in Parallel

There are two methods for finding the total resistance (R_T) in a parallel circuit when we wish to find equivalent circuits. First, let's do it the Ohm's way:

$$R_T = \frac{E}{I_t} = \frac{24}{12} = 2 \text{ ohms}$$

In other words, the total resistance is equal to the total voltage divided by the total current. The Kirchoff way:

$$R\ (1 \text{ and } 2) = \frac{R_1 \times R_2}{R_1 + R_2} = \frac{12 \times 4}{12 + 4} = \frac{48}{16} = 3 \text{ ohms}$$

At this point in our discussion we assume that the circuit we have been talking about is equivalent to that shown in figure 2-11. By combining R (1 and 2) and R_3, according to Kirchoff, the total resistance would now be:

$$R_T = \frac{R\ (1 \text{ and } 2) \times R_3}{R\ (1 \text{ and } 2) + R_3} = \frac{3 \times 6}{3 + 6} = 2 \text{ ohms}$$

And when we redraw the above circuit it would look like that shown in figure 2-12.

Just what does all this mean? It simply means that the above circuit is equivalent to our circuit of three parallel lamps. This is because: current through a series circuit is the same as the current through each separate part; resistance of a series circuit is equal to the sum of the resistances through each separate part; and that voltage across a series circuit is the sum of voltages of the separate parts.

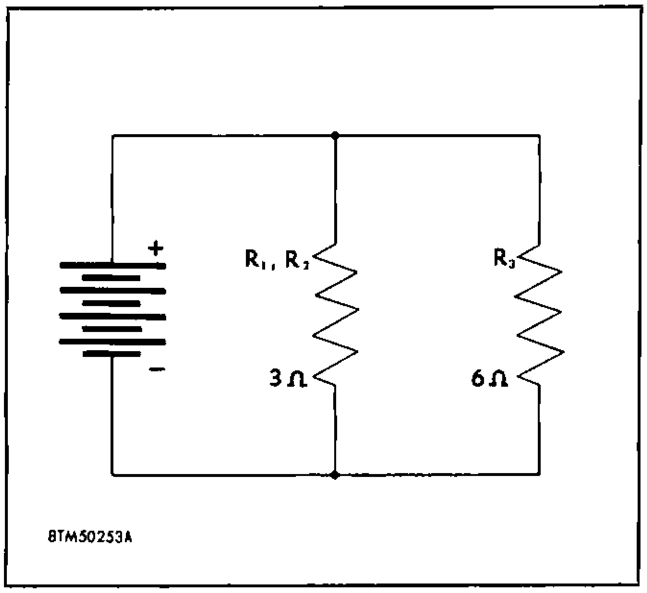

Figure 2-11. First Equivalent Circuit

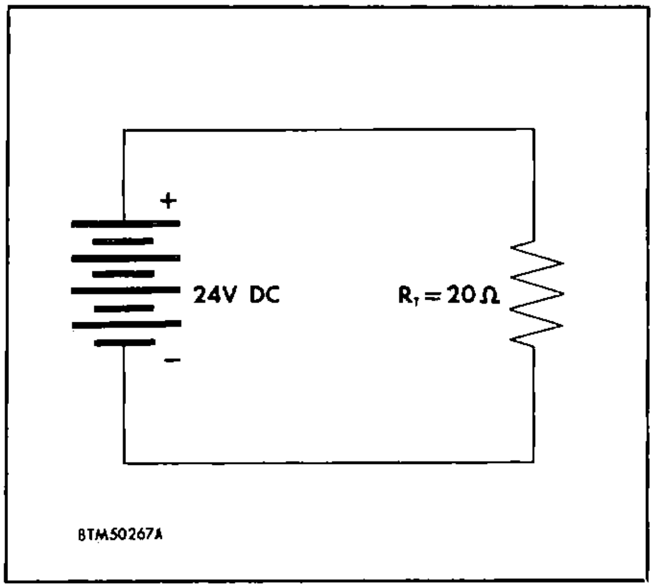

Figure 2-12. Final Equivalent Circuit

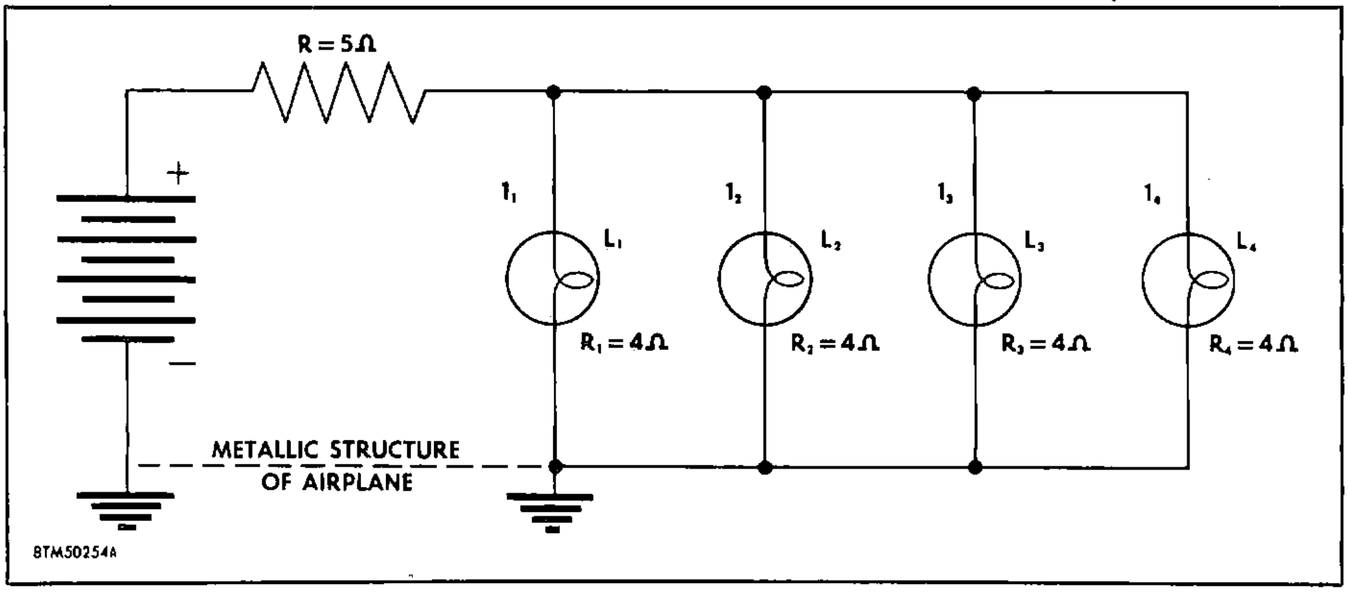

Figure 2-13. Typical Series-Parallel Circuit

In a parallel circuit, voltage across a parallel circuit is the same as voltage across each parallel branch of the circuit. Current through a parallel circuit is the sum of the currents through all the branches. And, the resistance of a parallel circuit is less than the resistance of that branch having the smallest resistance.

SERIES-PARALLEL CIRCUITS.

We should know by now that in a series circuit all the electrical devices of a circuit are connected in tandem—one after the other. In other words, the positive (+) terminal of one device is connected to the negative (−) terminal of the other, and so on down the line: +, −, +, −. In our discussion of parallel circuits we found the circuit divided into two or more branches—each branch carrying a separate fraction of the current. Another way of looking at it would be: all the devices in a parallel circuit are connected so that one-half of the terminals are fastened to a common point or to a common conductor, and the other half are tied into another common point or another common conductor. In this circuit if one or even two devices become ineffective, the rest of the circuit will still carry the electron flow.

In a series-parallel circuit some devices are in series and others in parallel. Figure 2-13 is a typical series-parallel connection. Note that there is a resistor in series with the four lamps which are connected in parallel. By measurement we found the resistances in each of the four parallel branches (R_1, R_2, R_3, R_4) to be 4 ohms—the original power source being a 24-volt d-c battery. Let's find the amount of current flowing through the various parts of the circuit.

To solve a problem such as this one in a series-parallel circuit, you must first convert it to a series circuit by substituting equivalent resistance for the parallel resistance. When you have done this, solve the series circuit.

In the circuit of figure 2-13, the resistance in each of the lamps is 4 ohms. If you will recall our saying that when load units in parallel have equal resistance the method for finding the total resistance is reduced to the following equation:

$$R_T = \frac{\text{resistance of one unit}}{\text{the number of units}}$$

Well, in this case we have four units of 4 ohms apiece, therefore, the equivalent resistance of the four lamps in parallel is:

$$R_T = 4/4, \text{ or } 1 \text{ ohm}$$

By substituting a 1-ohm resistor for the four lamps we have an equivalent circuit as shown in figure 2-14.

You now have two resistors in series across a 24-volt d-c battery and the total resistance in the circuit is 5 + 1, or 6 ohms. To find the total current in the circuit, our original problem, we revert to the Ohm's law:

$$I = \frac{E}{R}$$

and by substitution we arrive at the amperage value in the circuit:

$$I = \frac{24}{6} = 4 \text{ amperes}$$

You know then, since the total current in the circuit must flow through the 1-ohm resistor, that the current through it is 4 amperes. Therefore, the current through all four lamps in our original circuit is 4 amperes. Since they all have an equal amount of resistance, the current through each lamp is 1 ampere.

POWER.

Energy is a commodity which can be bought and sold and power is the speed of the transaction. In any transaction—especially when energy, force, and speed are involved—a certain amount of frictional heat is expended. This is doubly true of electrical energy. The heating effect of electrical current is well known—we see it every day in heaters, toasters, your automobile cigarette lighter, and other comforting electrical appliances—it is electrical power at work.

The Watt.

The unit of electrical power is the *watt*. Electrical power is the rate at which electrical energy in a circuit is expended. Another way of expressing the same thing —power is the rate of doing work and is equal to the voltage multiplied by the current in a circuit; or, power in watts = emf in volts $\times$ current (I) in amperes, or $P = E \times I$.

When the emf forces current through a conductor, the resistance encountered causes the conductor to become heated. This heating effect, as previously stated, is electrical energy. The power rating of a resistor, for example, indicates the maximum current that can flow through a resistor without danger of burning it out because of overheating. Let's take the simple series circuit as illustrated in figure 2-15, and find the power rating of a particular resistor. In this simple series circuit the applied voltage (E) is 100 volts and the resistance of the resistor (R) is 2000 ohms. By using our Ohm's Law we find that the current is 0.05 ampere. Now when this current flows through R, power is expended. To find out just how much is expended we use our power formula, $P = E \times I$:

$$P = 100 \times 0.05$$

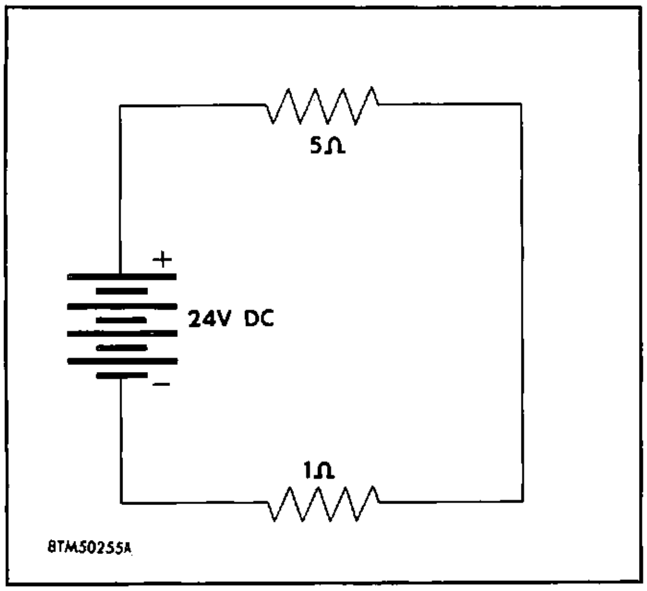

Figure 2-14. A Six-Ohm Equivalent Circuit

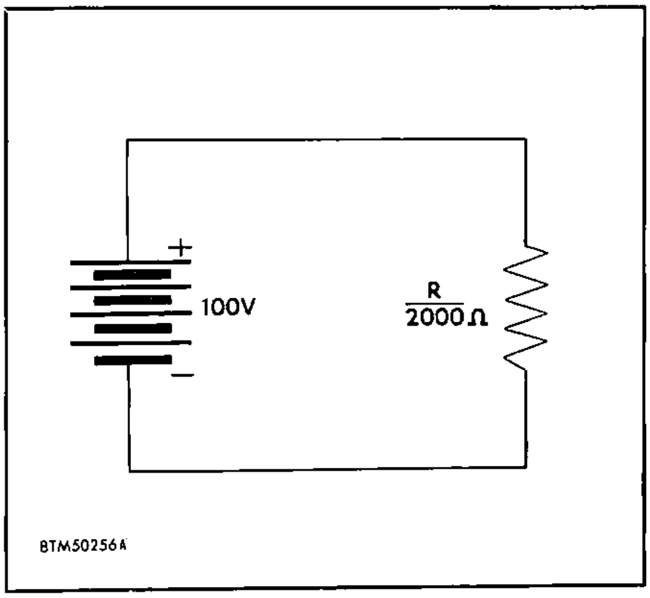

Figure 2-15. A Resistor Power Rating

$P = 5$ watts, or the power expended. It stands to reason then, that if the current, voltage, and resistance of a circuit is considered, then power must also be taken into account.

Horsepower.

Electrically speaking, power is the rate of doing work and as you know is expressed in watts. A watt is the power consumed in a circuit when 1 ampere flows under an emf (pressure) of one volt. But power is measured in various units: an airplane engine for example is rated in horsepower and in turn thrust horsepower—one of the few things a reciprocating and a turbojet engine have in common. One horsepower is the rate of doing work in raising a 550 pound weight one foot in one second. And one horsepower is equal to 746 watts. The electrical power rating of the overall electrical power systems in the F-102A, if expressed in horsepower, would be somewhere in the neighborhood of 64 horsepower.

Rating of Electrical Devices.

Electrical devices are rated—first according to the voltage that should be applied to them, and secondly, according to the power they require. The simplest example of this is an electric lamp. One lamp might be rated as a 115-volt, 40-watt lamp; another as a 115-volt, 20 watt lamp. This means that both lamps require a 115-volt circuit, but that twice as much power is required to operate the first lamp as the second. As you already know, you can find the wattage of an electrical unit—the power it requires—by multiplying the current flowing through it by the voltage applied to it. As an example, in a piston engine's starter motor using 70 amperes and with an emf of 24 volts, there would be 1680 watts of electrical power. If you converted this wattage to horse-

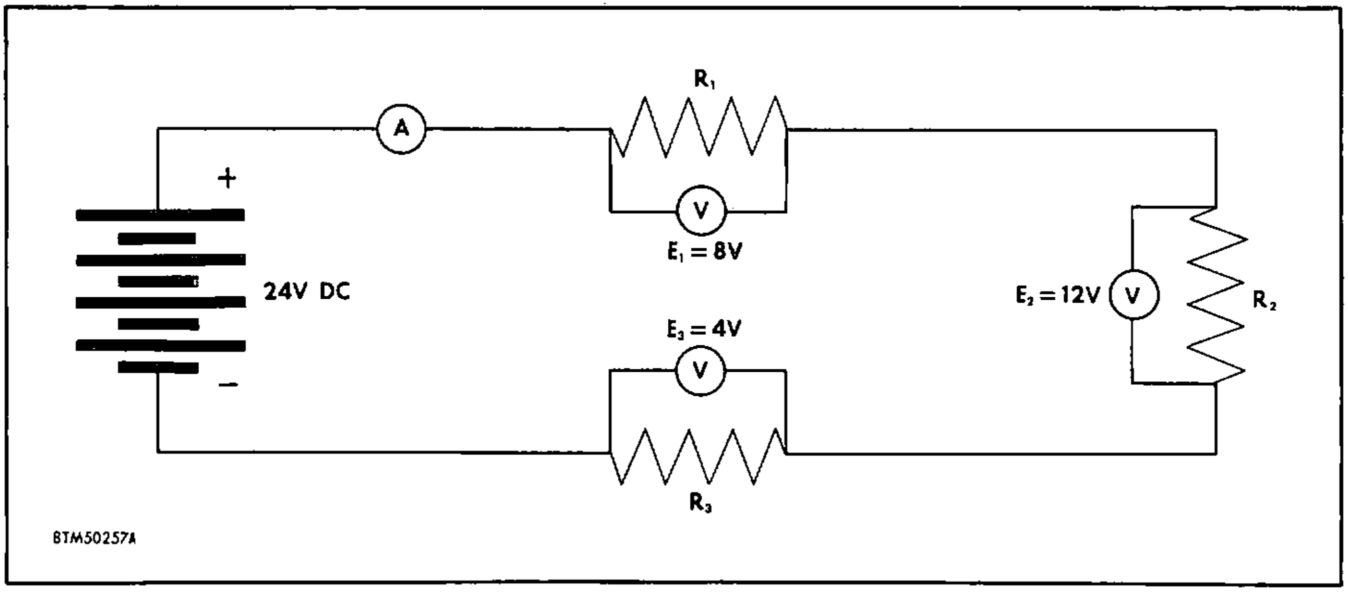

Figure 2-16. Power Rating in a Series Circuit

power by dividing by 746 watts (the electrical equivalent of 1 h/p), you would find this starter motor developed approximately 2 horsepower.

Power in Series and Parallel Circuits.

To find the power consumed in a series-parallel circuit, you first determine the power used in each unit and add their power values together. In either a series or parallel circuit, however, the power used in each unit is equal to the voltage across that unit multiplied by the current through the unit:

$$P = E \times I = \text{watts}$$

Let's draw our original series circuit that we used at the beginning of this discussion on circuits: substituting in our power formula to find the power used in each load unit; (see figure 2-16):

$$P_1 = E_1 \times I_1 = 8 \times 4 = 32 \text{ watts}$$

$$P_2 = E_2 \times I_2 = 12 \times 4 = 48 \text{ watts}$$

$$P_3 = E_3 \times I_3 = 4 \times 4 = 16 \text{ watts}$$

Adding to find the total power:

$$T_t = P_1 + P_2 + P_3 = 32 + 48 + 16 = 96 \text{ watts}$$

In a parallel circuit, to find the total power, the individual paths are solved first (see figure 2-17):

$$\text{Path 1: } P_1 = E \times I_1 = 24 \times 2 = 48 \text{ watts}$$

$$\text{Path 2: } P_2 = E \times I_2 = 24 \times 6 = 144 \text{ watts}$$

$$\text{Path 3: } P_3 = E \times I_3 = 24 \times 4 = 96 \text{ watts}$$

The total power then is:

$$P_T = P_1 + P_2 + P_3 = 48 + 144 + 96 = 288 \text{ watts}$$

To summarize our brief and basic discussion of circuits —before we move along into circuit protective devices— let's figure out the box score in the fundamental principles of computing circuit value (see figure 2-18).

CIRCUIT PROTECTIVE AND CONTROLLING DEVICES.

There cannot be power distribution without adequate circuit protection; there cannot be power distribution without circuits to channel this power to the proper buses, where it is distributed to other circuits. The first part of this chapter was given over to a discussion of basic circuits, this section will be devoted to circuit protective and controlling devices.

The first four chapters of this manual are devoted to a discussion of systems of the F-102A. The discussion covers the sources of power to the essential and nonessential buses. Every switch, relay, fuse, circuit breaker (trip-free or non-trip), and other circuit protective, controlling or voltage regulating devices are of the utmost importance to your knowledge of the power systems installed in the F-102A. The devices about to be explained are applicable to both the a-c and d-c systems.

CIRCUIT PROTECTIVE DEVICES.

As you have learned in the first part of this chapter, if the resistance is extremely small, the current will be extremely great. And whenever you have a high rate of current there is a continuous danger of a short circuit, that is, if there is no control or protection. Let's cite an example. If the wires from a battery to a motor were to touch each other, a short circuit would result. What would happen? There would be a great deal of trouble,

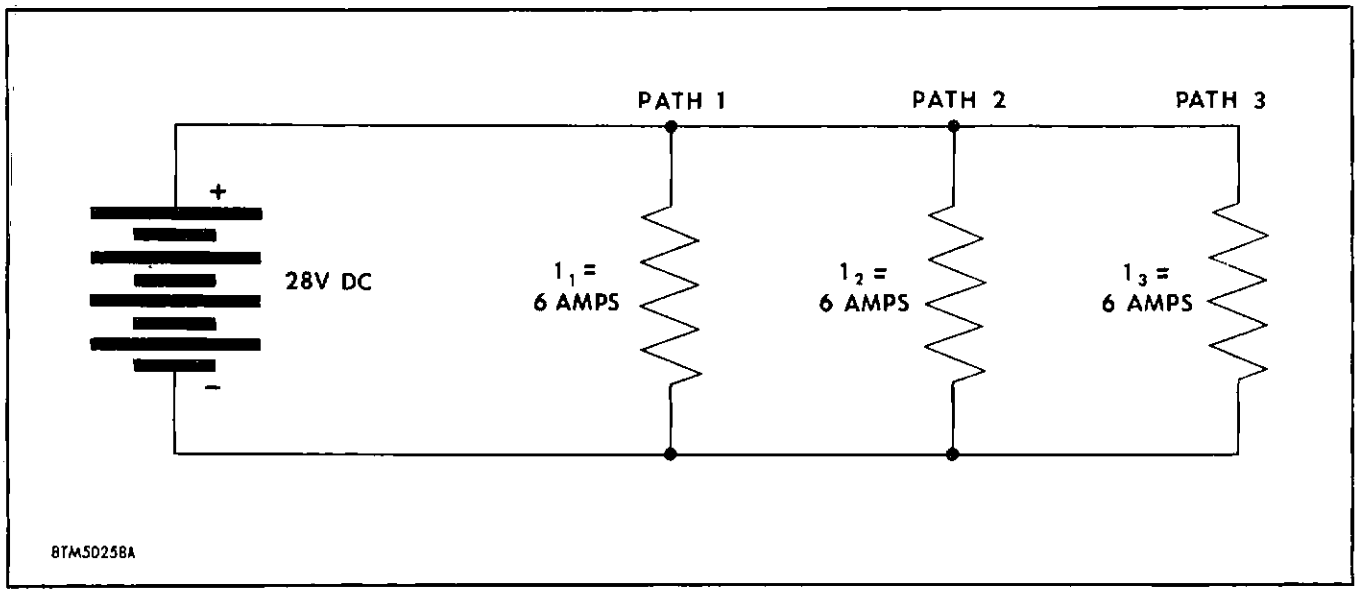

Figure 2-17. Power Rating in a Parallel Circuit

you can bet on that: the motor could certainly fail because practically all the current would be going through the "short circuit." The sudden load thrown on the battery would either cause it to run down, or the rapid generation of heat by excess current flowing through the conductors would burn through the wires before the battery was completely dead, causing a dangerous fire hazard.

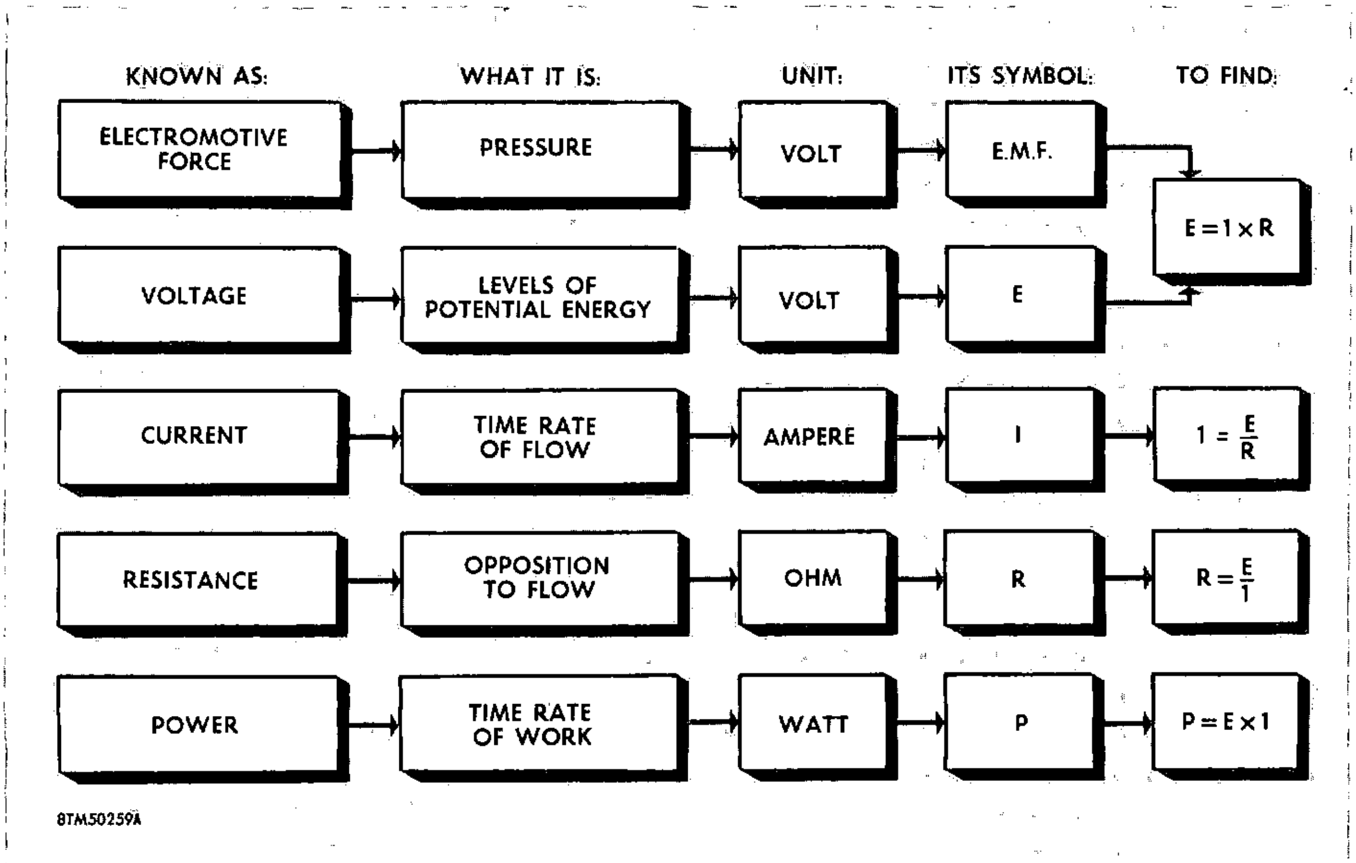

Figure 2-18. The Box Score

To protect airplane electrical systems from damage and failure due to excessive currents, protective devices such as fuses, circuit breakers, and various circuit protectors are incorporated into these systems.

Fuses and Current Limiters.

The fuse is perhaps the most common device used for everyday circuit protection. To "blow a fuse" has become a figure of speech. It is especially applicable early in the morning when the lights, the electric alarm clock, heater, radio, coffee percolator, and electric shaver are all buzzing and "perking" along in fine shape. Suddenly, they all stop functioning at the same time—you have blown a fuse literally and figuratively.

Well, anyway, a *fuse* is a strip of metal with an exceedingly low melting point. It is connected in series in the circuit which it protects. Its rating is based on the number of amperes it will carry. This rating represents the maximum possible current a particular circuit will carry. The capacity of the fuse will of course be greater than the requirement of the unit or units it protects—a 15-amp fuse could not be used to protect a motor which requires say, 20 amps. Since a fuse is a protective device it is most important to use one that fits the need.

In most fuses the metal strip is an alloy of tin and bizmuth. When this strip melts due to excess current the circuit is open, although a short circuit may have caused the overload. In figure 2-19 you will see a typical fuse used in the F-102A. A fuse whose metal strip is made of copper (usually a calibrated copper link) is called a *current limiter*. These, while essentially a fuse, will not melt on usual overloads, and can be said to isolate a particular circuit. Some current limiters will carry double its nominal rating and not melt until the current overload is four or five times its rated current. For example, to permit circuits controlled by heavy duty relay switches (which we shall discuss in a moment) to carry heavy current, and still protect them against short circuits, a current limiter is sometimes used. The two fuses most used in the Air Force are the "plug-in" and the "clip" types. The F-102A uses very few fuses, actually only six, and these are used in the fuel booster pump circuit.

Circuit Breakers.

A *circuit breaker* is an electrical device which breaks the circuit when the current reaches a predetermined value. Circuit breakers are used today almost exclusively in places of fuses, because they not only give circuit protection but also eliminate, in some cases, the need for a switch. In the F-102A they are used almost exclusively as circuit protection.

The feature which distinguishes a circuit breaker from a fuse is the reset value—a circuit breaker can be reset, a fuse must be replaced. As you know, there are several types of circuit breakers used in the Air Force. Those used in the F-102A are in almost every case the trip-free, push-pull type.

These trip-free circuit breakers operate on a thermal (heat) overload principle. The contacts are closed, but as an amperage builds up in excess of the load capacity (5 to 10 amps usually in the F-102A) the bimetallic strip bends away from a catch on the contact lever and permits the contacts to open. However, once a trip-free, push-pull, manual-reset circuit breaker is tripped it cannot be overridden immediately. The cooling-off period for the tripping element is approximately one minute. When closing the circuit, it might be a good thing for you to remember that to insure relatching of the tripping element, you must pull the button out as far as possible before pushing it to the closed position. Figure 2-20 shows the construction of a trip-free circuit breaker used throughout most of the circuits in the F-102A.

CIRCUIT CONTROLLING DEVICES.

When we become more deeply involved in the a-c and d-c circuits of the F-102A, your knowledge of switches and relays will be of prime importance to your understanding of the circuits. The chart on figure 2-21—showing the symbol designations used for switches and relays in the circuit diagrams of the F-102A—should give you a good idea of the number and the types of switches and relays that you should have knowledge of if you are to perform your task efficiently.

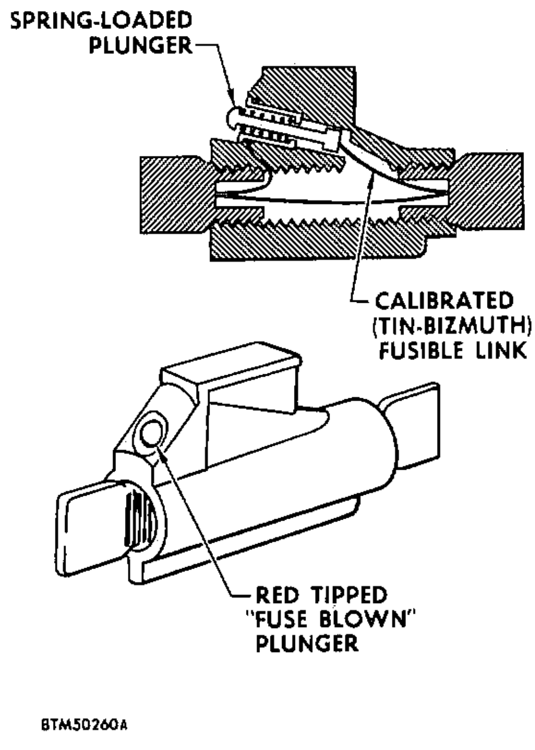

Figure 2-19. Typical Fuses

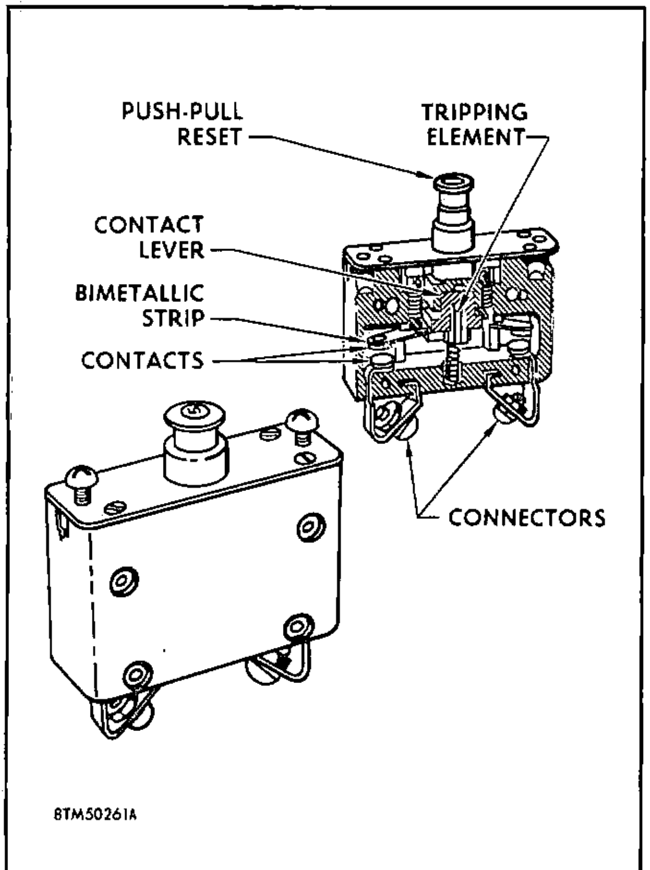

Figure 2-20. Typical Circuit Breaker

The Switch.

Aircraft circuits are equipped with switches to provide a quick and an easy way of starting, of stopping, and of changing the direction of the current flow. They are designed for specific usage within the circuits, with insulating qualities to carry the voltage as well as the current.

Switch Designation.

Spst, spdt, dpst, and dpdt—these are simply abbreviations for switch designations used to save space and to avoid distracting your mind by the needless spelling out of repetitious words or phrases. It would be desirable if there were more of them. What do these particular ones stand for? Let's find out.

In the first place switches are designated by the number of *poles, throws,* and varied *positions.* The *pole* of a switch is the movable blade which makes the contact and closes the circuit, or which breaks contact opening the circuit. The number of poles is equal to the number of terminals by which the current can enter or leave the particular switch.

The *throw* of a switch indicates the number of circuits each pole can complete through the switch. The number of positions in a switch simply means the number of places the switch's actuating device may be set, to open a definite circuit, close a circuit, or channel a new current flow.

Because of the crude method of operation and because of exposed contacts, knife switches are seldom if ever used in aircraft electrical systems. But, for the purpose of explaining the more commonly used toggle switch, they are most beneficial. In figure 2-21, we will compare—for the sake of clarity—the switch abbreviations found in the switchgear symbol chart with their knife switch counterparts. In this illustration, a switch through which only *one* circuit can be completed is designated as a *spst,* or single-pole, single-throw switch.

If two circuits can be completed through a single-pole, it is designated as a *spdt,* or single-pole, double-throw. A *dpst,* or double-pole, single-throw designation is a switch with two poles. In this type switch, two individual circuits can be completed through each pole. And finally, there are the *dpdt's,* or those switches having two poles through which two circuits can be completed, and they are further described as double-pole, double-throw switches.

Toggle Switches.

A toggle switch, which comes to rest at either of two positions—opening the circuit in one and completing it in another—is described as a *two-position* switch. On the other hand, if we have a toggle switch which is spring-loaded to the OFF position—one that is manually held in the ON position to complete the circuit—it is a *single-position* switch. A *three-position* toggle switch is one that will come to rest on any of the three positions, ON, OFF, ON.

A *normally open* switch is one that stays open, except when it is held in the *closed* position. A *normally closed* switch is one that stays closed, except when held in *open* position. Both of the above types are spring-loaded to a particular position and when released trip to their original position.

All toggle switches are self-enclosed and are used in the airplane more than any other type switch.

Pushbutton Switches.

These switches are more-or-less self defined, having one stationary contact and one movable contact, as shown in figure 2-23. The movable contact is attached to the pushbutton by an insulator. This switch is also spring-loaded and is the momentary contact type when snap action is required.

Limit Switches.

Figure 2-24 shows a normally closed limit switch, which is more commonly known as a microswitch. When the operating plunger is pushed in, the three-bladed spring

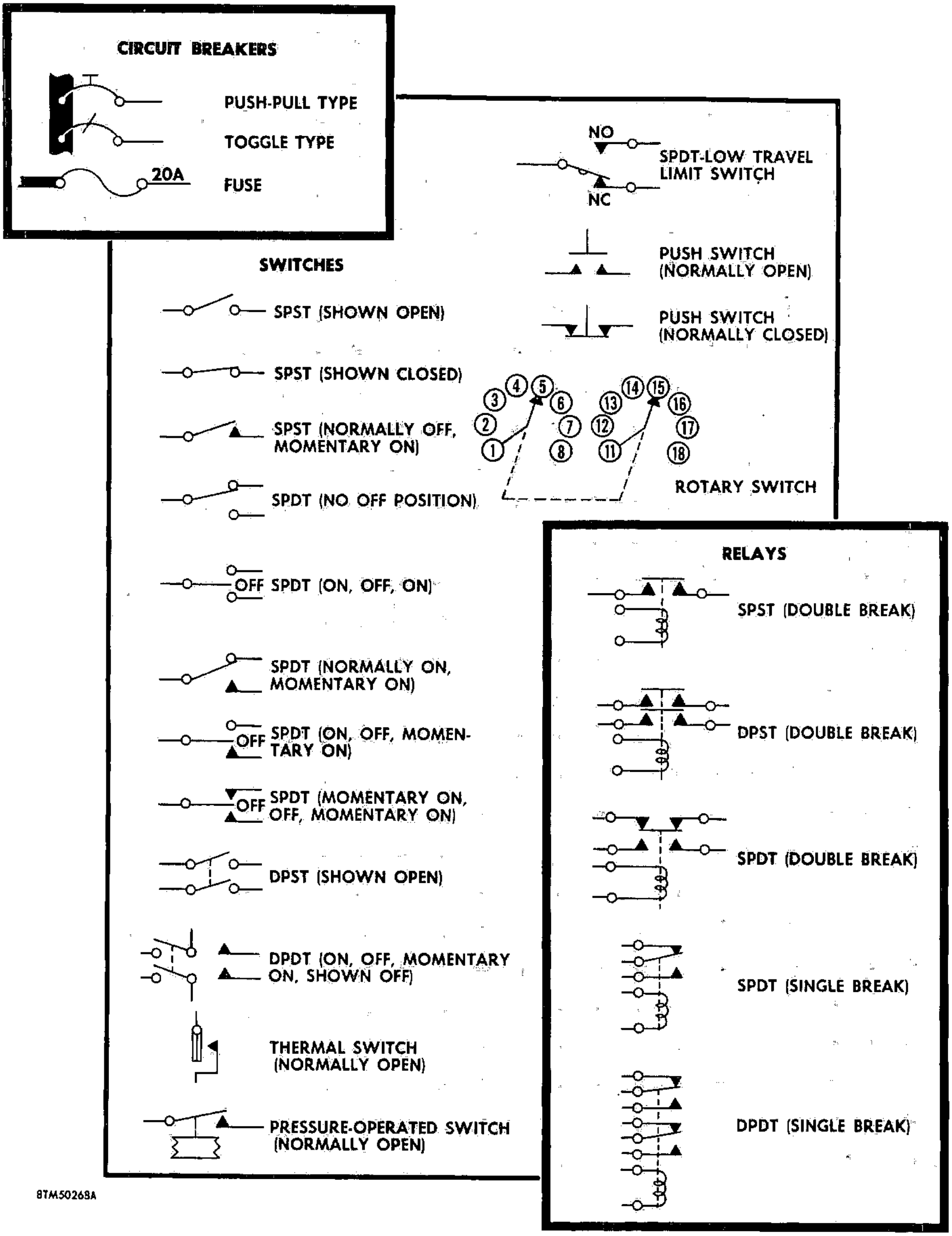

Figure 2-21. Electrical Symbols for Circuit Controlling Devices

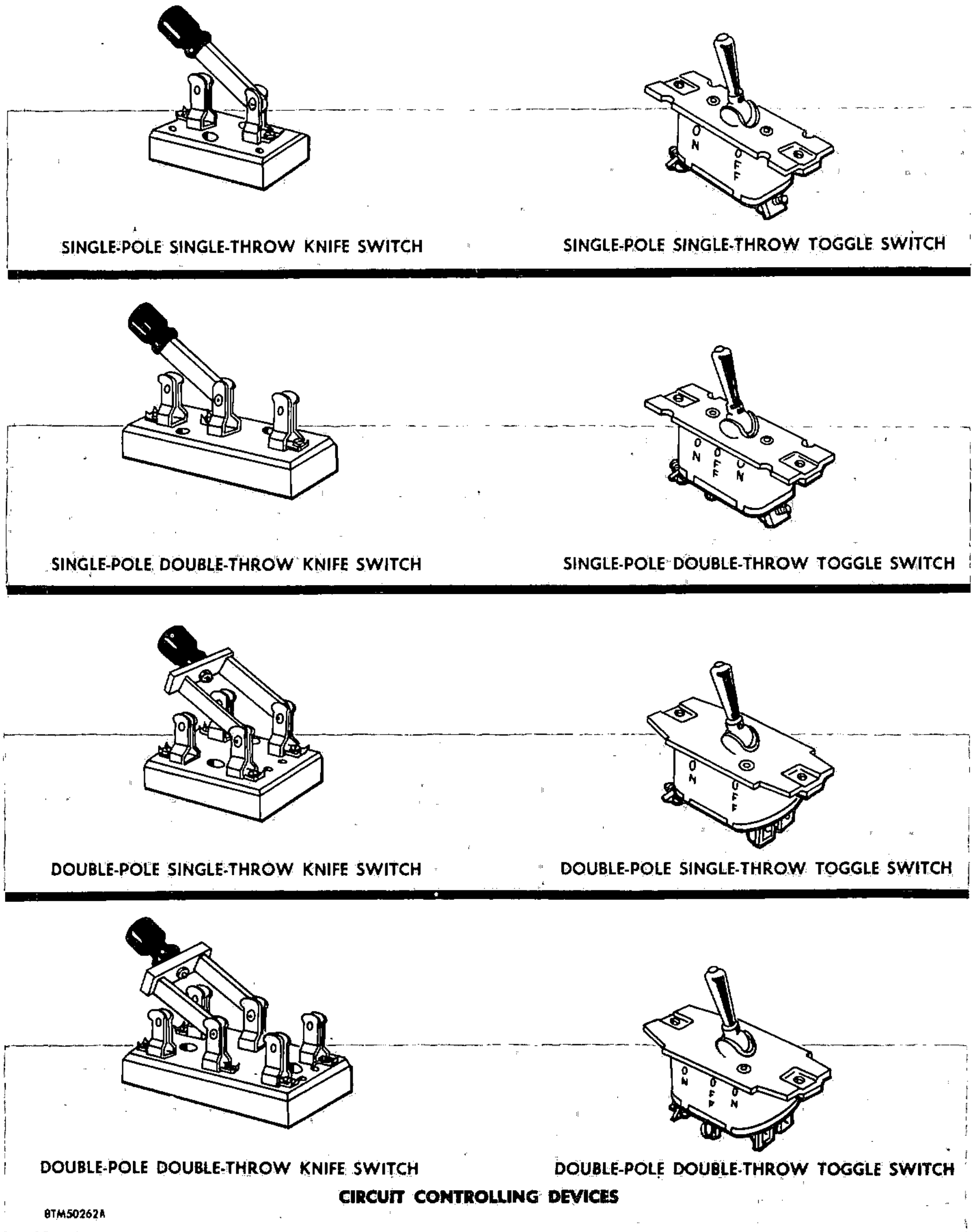

Figure 2-22. Toggle Switches

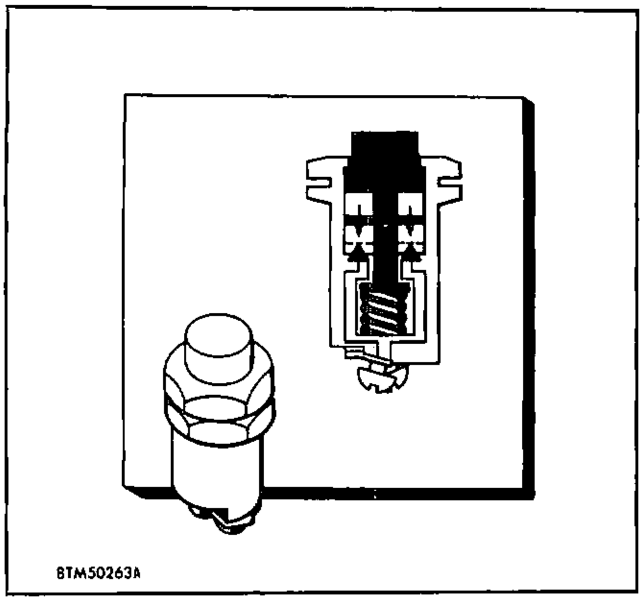

Figure 2-23. Pushbutton Switch

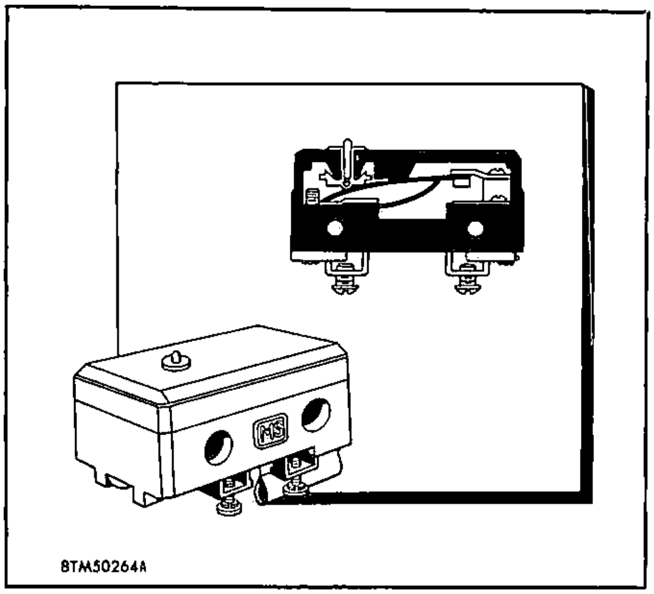

Figure 2-24. Limit Switch

is pushed down and the contact point attached to the spring is separated from the fixed contact, thus opening the circuit. Microswitches open or close a circuit with a minimum of movement. A sixteenth of an inch or less is all that is required generally to move the tripping device. When used as limit switches, they are in most cases of the pushbutton variety.

Rotary Switches.

The rotary selection switch (selector or wafer switch), as the name implies takes the place of several switches, as you can see in the illustration, figure 2-25. By turning the selection knob from one position to another, one circuit is opened and the other is closed. A rotary switch may contain a single wafer or several wafers depending on the circuit requirements. An example of this is the armament selector switch which controls the armament system of the F-102A; this selector switch has 12 wafers mounted on a common shaft. In this way, a single movement of the shaft opens and closes a number of circuits. The number of circuits involved is only limited by the number of contacts on the wafers.

Relay Switches.

In our symbol chart (figure 2-21) you can see that relay switches are abbreviated in the same manner as toggle switches: spst, dpst, spdt, and dpdt. For example, spst (double break) means that we have a single-pole, single-throw relay switch with two sets of contacts—a double break—to open or close a circuit.

THE RELAY SWITCH IN CIRCUITS. Relay switches are used for remote control of heavy-current circuits. They can be adjusted for the time-delay and sequencing factors in these circuits and others. Usually they are placed directly between the souce of power and the units controlled so that the cables carrying high voltages will be as short as possible. This cuts down the possibility of fire as well as line loss.

In Chapter I of this supplement, you learned the basic theory behind the helical coil or solenoid—that the magnetic field associated with this type coil is much the same as the field surrounding a bar magnet. The relay switch we are about to discuss is based on the fundamental principles of those electromagnetic fields intertwining the windings of a solenoid coil.

In the F-102A, relays are used not only in the a-c and d-c power distribution systems, but in almost every other associated system in the airplane. For example, the d-c

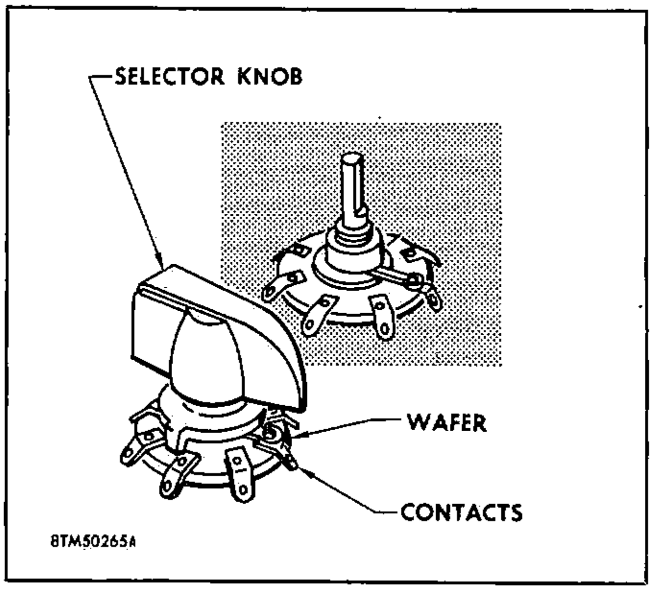

Figure 2-25. Armament Selector Switch

sources of power are connected to the d-c essential and non-essential buses through relays. The power is then distributed through circuit breakers—from the buses—to the various switches and relays that control the other d-c powered electrical devices. The following are some of these relays which we will be talking about in the d-c system: the *overvoltage relay* which has an inverse time voltage characteristic consisting of a magnetic piston enclosed within a solenoid (its principles of operation will be discussed later in this chapter); the *d-c warning relay,* which connects the d-c essential bus to the master warning system through the power warning circuit breaker, which in turn lights the master warning and the power failure warning lights when the d-c generator ceases to function; then we have the *battery relay,* the *d-c disconnect relay,* and others which we shall get to at the appropriate time.

The relays that control the a-c power system of the F-102A are used in much the same manner. Besides an overvoltage control, there is an *a-c power failure relay* and an *a-c power disconnect relay* whose function it is to connect the a-c generator to, and disconnect it from, the a-c buses. There are others, but for our purpose here, it is sufficient to acquaint you with some of them simply for future reference. Let's discuss for a moment, in a general way, some fundamental information concerning relays.

RELAY SWITCH CONSTRUCTION. A relay switch consists of a coil or solenoid, an iron core, and fixed or movable contacts. Small high resistance wires connect the solenoid coil terminals with the source of power usually through a control switch. When this switch is closed manually or automatically, an electromagnetic field is set up around the coil. If you are not clear at this point, on just what an electromagnetic field about a coil is, it is suggested that you refer to "Electromagnetism" of Chapter I and restudy that portion of the text dealing with electromagnetic fields.

As you know, there are many types of relays with variety in construction—some large, some small, and some tiny. Some are of two coil construction, others single. Then there are the a-c power frequency relays and differential protection relays used for protecting the a-c generator and its leads against line-to-ground and phase-to-phase faults. And as we have said before, all are usually used where heavy currents are involved.

Let's discuss the basic fundamentals of a dpdt (single break) relay for a moment. The relay B in figure 2-26, will give you a good idea of just how it functions. As you can see, the iron core is fixed. The instant that the control switch closes, the core becomes magnetized by the magnetic field set up around the coil. The magnetic pull of the core—on the piece of soft iron which as you learned in Chapter I, is highly susceptible to magnetization and has a high degree of permeability—overcomes the force of the spring, drawing the spring taut and closing the contacts of the relay, thus completing the circuit. Technically, the movable contactor is called the armature. Perhaps you will recall our analogy of a spring pulling a door shut. Well, when the control switch is opened, the field about the coil collapses, and the mechanical energy stored in the extended spring separates the contacts and opens the circuit.

Basically, in a spst (double break) relay, part of the core is movable. Relay A in the illustration will show you this arrangement. The contacts on the "T" head of the core, while attached to the core, are insulated from it. When the switch controlling this particular relay is closed, the magnetic field around the coil causes the movable part of the core to be drawn into the coil. You will remember that in Chapter I we said, ". . . the magnetic field of *the coil* will tend to center the core into the coil when the current is turned on." This strong magnetic field of uniform intensity compresses the return spring thus closing the contacts and completing the circuit. Again, as in the case of our dpdt (single break), when the control switch is opened the magnetic field about the coil collapses and the compressed spring returns the movable core to its original position, breaking the closed circuit.

There are some relays designed for continuous operation; others for intermittent operation. A conventional starter relay, for example, is constructed to function intermittently and would overheat if used continuously. The battery relay switch in the F-102A, however, can be operated continuously; the coil assembly is wound so that it has a high resistance, and therefore will not overheat readily.

In any relay or electrical switch of any kind there will be a certain amount of arcing. It may be small or infinitesimal, or it may be of more magnitude and result in burning the switch contacts. For this reason, the return springs used in most relays have high tension qualities, so they open or close circuits quickly. One exception to this is the spring used in most battery relay switches which have just enough tension to open the battery circuit after those units requiring heavier currents have been disconnected.

F-102A D-C GENERATOR POWER SYSTEM.

In the F-102A d-c generator power system, the generator is mounted on the forward end of the constant-speed drive unit gear box. This generator delivers 200 amperes at a regulated 28 volts to the d-c buses even at its minimum speed of 4000 rpm. Its speed range, at present, is from 4000 to 8000 revolutions per minute. Its continuous operating speed is rated at 6000 rpm, and its required maximum speed, for regulation, is 7000 rpm. This generator is a highly efficient d-c electrical power plant that is capable of operating in altitudes in the neighborhood of 65,000 feet.

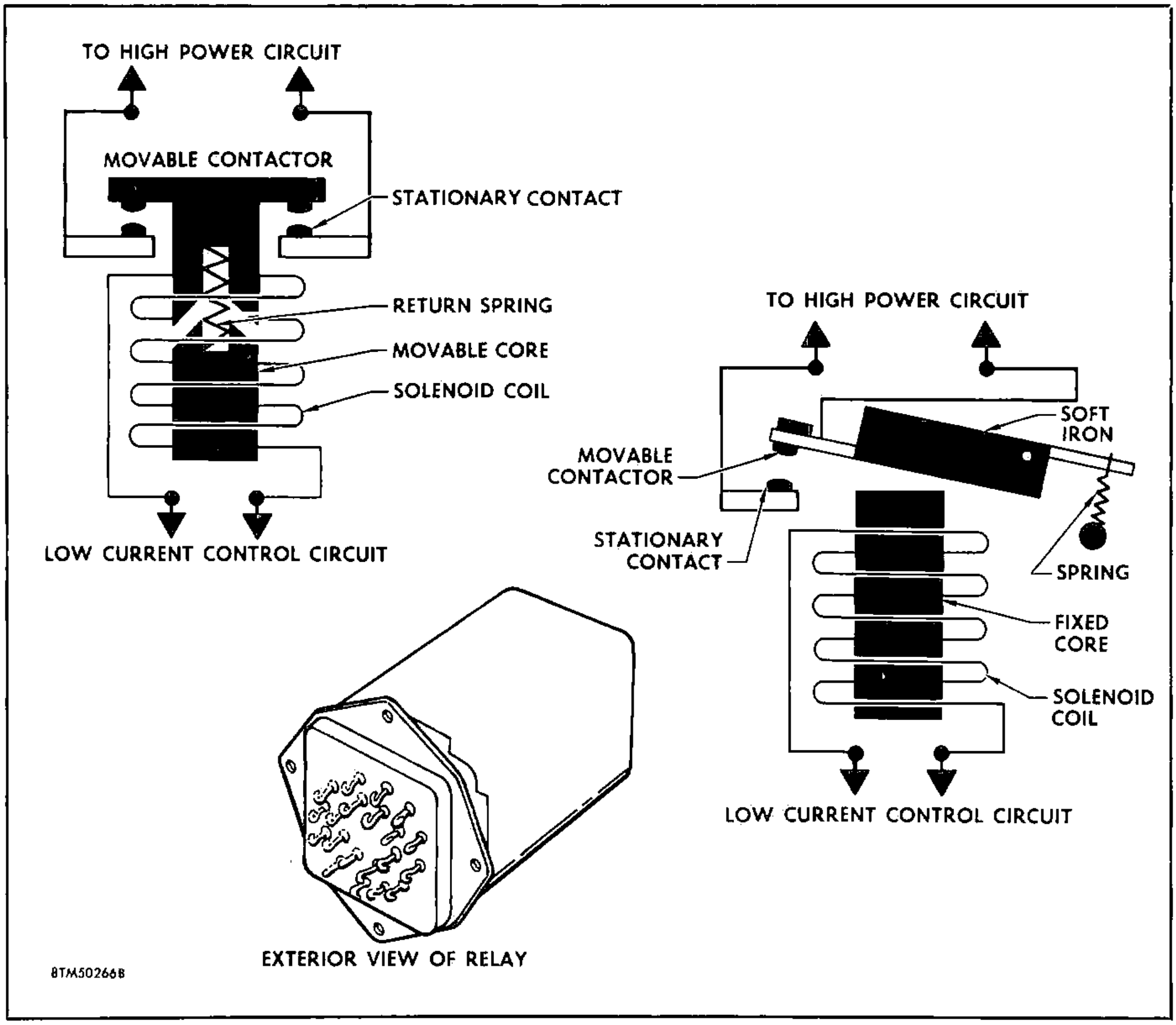

Figure 2-26. Relay Switches

As you probably know, it has only been in the last few years that a satisfactory generator has been developed to operate specifically from the high speed pads of a jet engine. Heretofore, generators were primarily designed for the 6000 rpm piston engine speed.

In Chapter I of this supplement you were given the basic principles of generators—both d-c and a-c. What we have been talking about in the preceding sections of this chapter are more-or-less the basic factors governing the distribution of d-c power throughout the F-102A, or for that matter almost any interceptor, fighter, or bomber in the skies today. We have discussed d-c circuits, their initial problems, and the devices for protecting and controlling those circuits—their voltages and current—from overloading or other faults.

To understand the d-c power system on the F-102A, you must have a thorough knowledge of not only what has been discussed thus far, but particularly those sections we are about to discuss. And like our review of the three basic kinds of electricity in the beginning of this manual, we cannot have exact knowledge of one section without exact knowledge of the other. Chapter I, covered briefly, the practical d-c generator and its type classifications—series-wound, shunt-wound, and compound.

THE D-C GENERATOR.

You are now entering a phase in your knowledge of the d-c power system in the F-102A, where the practical application of the theory behind the production of electromotive force (emf) will be of great value to you. You will find that all generators used in the Air Force, while they differ somewhat in design, have the same general construction and operate similarly. The reason for this is of course the difference in manufacturers, but regardless of design—whether they have two or three major assemblies—they must come up to the rigid specifications demanded by the Air Force.

Figure 2-28 shows the components of the F-102A high-output 30-volt generator; a bit more complicated than our generators depicted in Chapter I, but the same principles of operation apply. As you can see, there are but two major assemblies—the housing assembly and the armature assembly. The subassemblies are designated as the support and brush holder assembly, the drive shaft assembly, the bearing cap and condenser assembly, and the brush assemblies.

In electrical school you learned that most typical d-c generators are broken down into three major parts—the armature assembly, the field frame assembly, and the commutator end frame assembly, (see figure 2-29). In the generator we are interested in at the moment, we may speak of the field frame and end frame assemblies as being incorporated into one unit. The armature assembly is, and always has been, a single integrated assembly. Let's talk about this assembly first.

Armature Assembly.

This assembly consists of a hollow steel shaft on which are located the core, the armature windings and a commutator.

The core is a soft iron cylinder of laminated construction. By laminated, we mean a layered buildup of iron strips which, as you have learned, cuts down on the hysteresis loss. The core serves as the magnetic circuit, and at the same time the mechanical support for the armature windings which are placed in insulated slots running the length of the core.

The armature windings, as you learned in Chapter I, are composed of a number of insulated copper wires. These are called inductors in which the voltage is induced through the rotation of the armature in the magnetic field. In Chapter I, you were told that the more windings an armature has, the greater the generated voltage and the more constant the output. The ends of each winding are brought out to the commutator. Here, the coil leads, as they are called, are pressed into the commutator riser and hard-soldered, or, as in the case of the F-102A, silver brazed to the commutator risers. Because the rotation is exceedingly high in aircraft generators, binding bands are used to further safeguard the coil leads from coming loose. It is, of course, the centrifugal force acting against these conductors which could and has caused a disconnect. In the F-102A generator we have a wedge-type armature construction—the conductors being secured in their slots by means of wedges in addition to the wire banding reinforcement at both ends of the armature.

The Commutator.

The commutator in an aircraft generator—unlike our fundamental machine in Chapter I — is made up of a number of hard-drawn copper segments. Hard-drawn simply means that the copper has been freed from iron particles by drawing a magnet through the copper mass, thereby giving it great hardness and by the same token, strength. The segments are insulated from each other, from the shaft, and from the commutator frame. It is most important that commutator insulation be unaffected by moisture, changes in temperature, and high altitudes. For this reason, mica is generally used as the insulator. Mica, while being hard and brittle, also has enough elasticity to fill the spaces between the commutator segments as they expand or contract through temperature variations.

Housing Assembly.

The housing assembly (or as it is designated in some generators, the field frame assembly), is the cylinder in which the generator field is located. At one time field frames were made of cast iron, but because of the weight factor and inferior permeability they are now constructed of rolled, sheet steel. Located within the field frame are the polls on which the field coils are wound. These poles and coils constitute the electromagnet which we explained in Chapter I. And if you will recall, the field coils are wound so that the poles—north and south—are to the right and left of each other. In this way the lines of force in travelling from their south to north poles must pass through the frame, which becomes part of the magnetic circuit. This is the main reason why frames must have a high degree of permeability and not be of the cast iron construction we spoke about earlier.

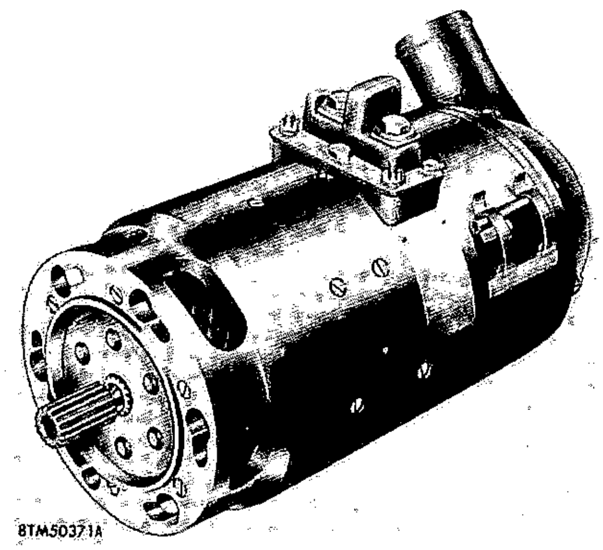

Figure 2-27. F-102A Generator

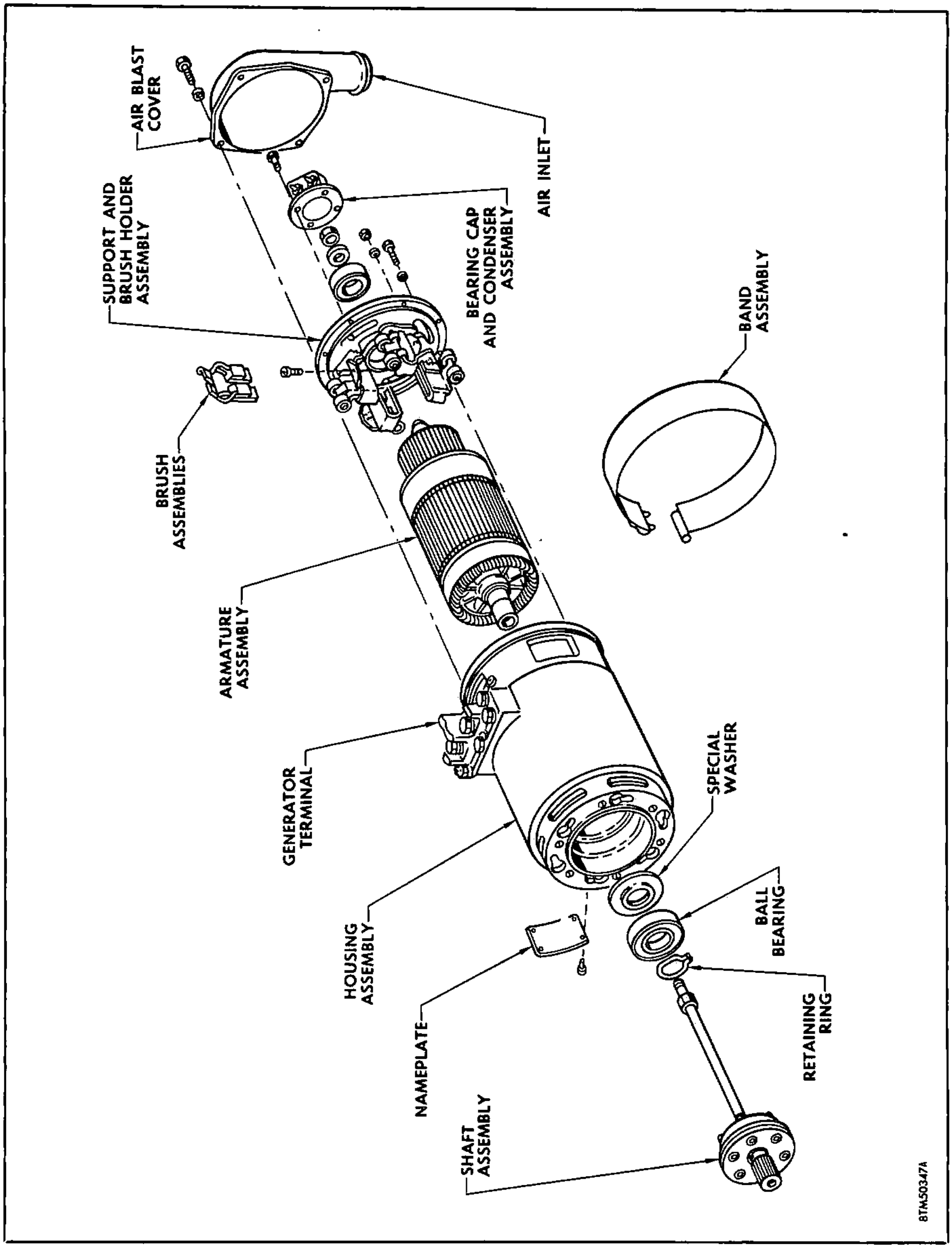

Figure 2-28. Components of F-102A Generator, Model G35-5

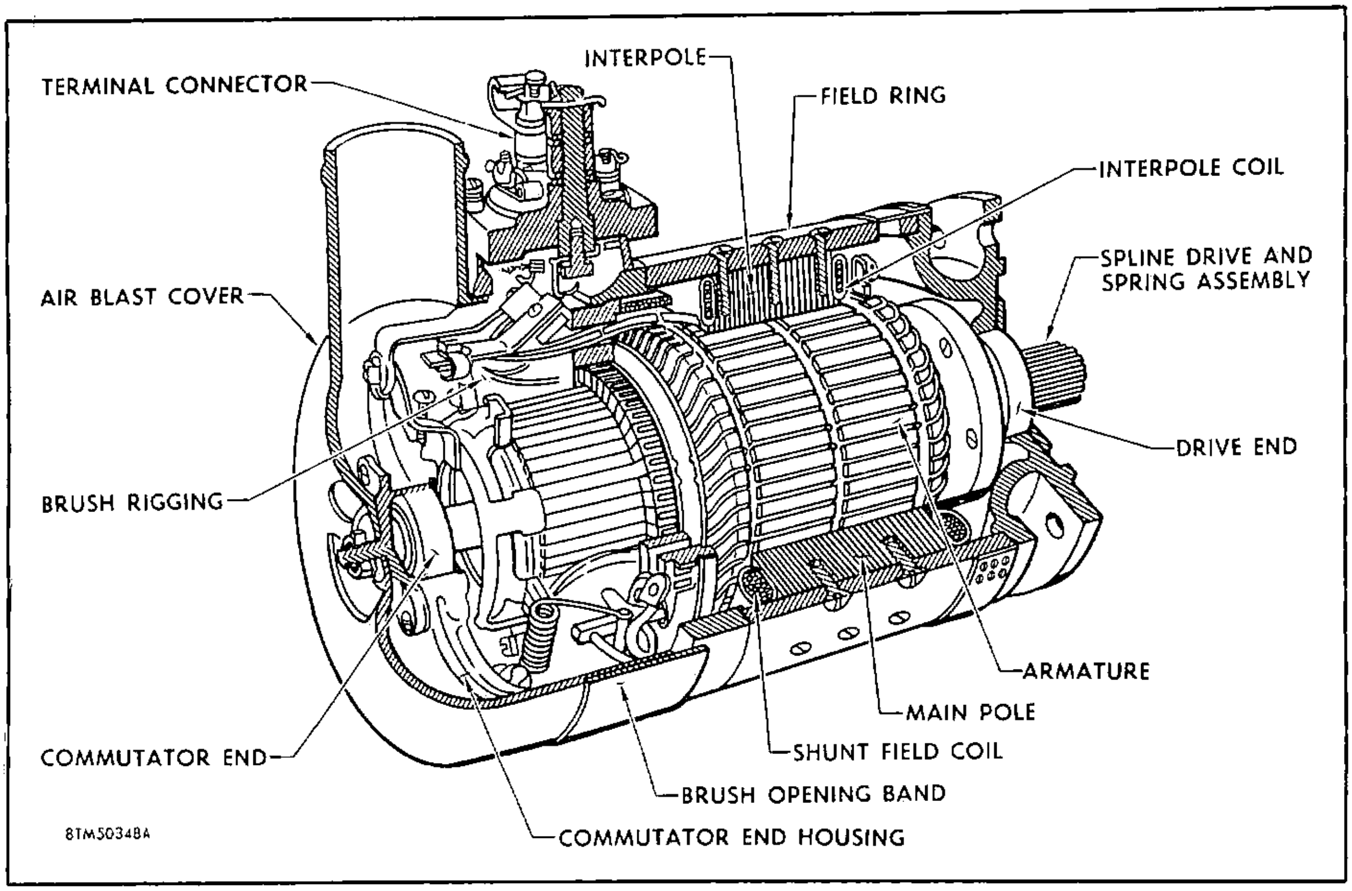

Figure 2-29. Cutaway of Typical Aircraft Generator

The commutator end frame assembly (in the illustration of the F-102A generator, this is part of the housing assembly, see figure 2-28), forms a support for the brush holder assembly and armature. Its end flange serves as an attach pad for the air inlet housing. The air inlet cover may be attached in any one of eight positions depending on the location of the cooling air duct in relation to the generator installation in the airplane.

The drive end of the F-102A generator housing assembly contains the armature bearings on which the armature shaft rotates. It is the flange, or mounting head, at this drive end which secures the generator to the mounting pad on the forward end of the constant-speed drive unit.

COMMUTATION.

As you have learned—and it might be well to emphasize it again—the emf induced in any one coil of a direct-current generator is alternating. Therefore, for the current in the load circuit to flow in one direction only (d-c), rectification is necessary. And, as you probably know, to rectify means, to make an alternating current vary solely between zero and maximum in its wave form.

The rotation of the commutator, with its many segments, provides this rectifying action. In Chapter I (see "D-C Generator") we showed the complete cycle of a basic d-c generator and its output. At that time our discussion was brief, mainly because we felt that this was a more appropriate section in which to expand our discussion. The operation of a commutator can be understood by reviewing the complete revolution of a d-c generator. Refer to "D-C Generator," Chapter I of this supplement.

If you remember in the rotation positions *a* to *e*, the current flows in the rotating coil from *A* toward *B*, and *D* toward *C*, undergoing, during the process, changes in its magnitude from zero to 180 degrees. At position *e* the coil completes a half-revolution, and at this point (*e*) the induced voltage is zero.

During the second half-revolution the current flow reverses itself. And if you recall, at the moment the current in the coil sides, AB and CD, changes its direction, the commutator segments reverse their connection with the brushes. In this manner we have rectification—the electron flow through the load is the same in the second half-revolution (from 180 degrees to 360 degrees) as it was during the first half-revolution. In other words, in a d-c generator, even though the current flow in the coil is alternating in direction, the current flow in the external load is constant and in one direction due to the action of the rotating commutator.

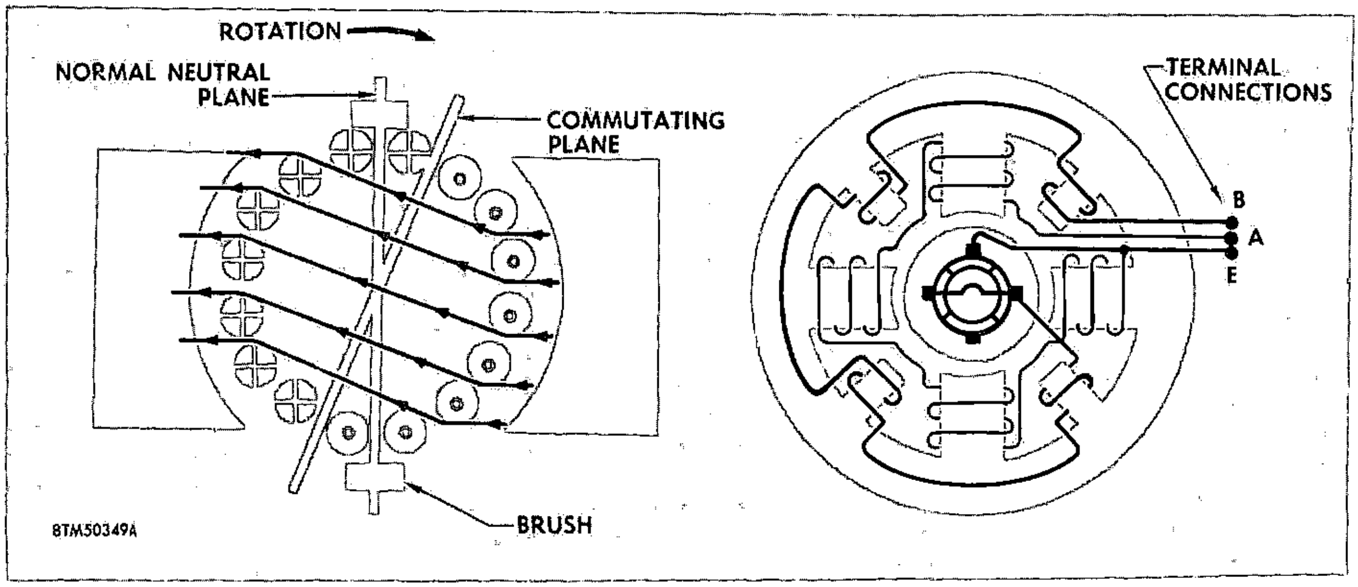

Figure 2-30. Generator Without Interpoles, Generator With Interpoles

FIELD DISTORTION.

In a d-c generator it is the current flowing through the armature which sets up the electromagnetic fields in the windings. These new fields tend to distort—to bend—the magnetic flux between the poles of the generator from their usual straight line configuration. An example of what we mean can be seen in figure 2-30. Since armature current naturally increases with load, this distortion becomes more pronounced as the load is increased. Suppose we analyze this statement.

Armature windings in generators are spaced so that at certain positions during the rotation of the armature the brushes contact two adjacent segments on the commutator, thus shorting the armature windings to these segments. Now, if the magnetic field is not distorted as shown in figure 2-30, no voltage will be induced into these shorted windings and therefore no harmful effects occur. If, however, the field is distorted and a voltage is induced, considerable sparking takes place between the brushes and the commutator. This excessive sparking pits the commutator segments, and at the same time the wear on the brushes becomes unusually great. This, of course, reduces the normal output of the generator. The higher an airplane flies the more pronounced this becomes, and we have what is known as "commutation." Fortunately, this fault has been reduced to a minimum through engineering "know-how."

Commutation has been minimized by three methods. First of all, engineers in correcting this condition found that if brushes are so positioned that the plane of the shorted coil (shown in figure 2-30) is perpendicular to the distorted field, the commutation fault is diminished considerably. There were, and still are, a few generators where the brushes can be manually shifted ahead of the normal neutral plane to the neutral plane which is caused by field distortion. These adjustable-brush generators, however, are not practical for aircraft use. They have been mentioned simply as a comparison to those more applicable to modern airplane usage. The generators installed in airplanes today are the non-adjustable brush-type. It is the manufacturer who sets the brushes for a minimum of sparking and a maximum desired voltage output.

The second method of checking this excess sparking was by the addition of interpoles. An interpole is a pole placed between the main poles of a generator. It has the same polarity as the next main pole in the direction of the armature rotation. The magnetic flux produced by an interpole causes the current in the armature to change its direction as the armature winding passes under it. This cancels out the electromagnetic fields about the armature windings. How does this minimize field distortion? It is very basic. We simply go back to our very important friend the atom. All normal atom structures are electrically neutral, the number of positive charges equals the number of negative charges—they are electrically balanced. In our present case, the magnetic strength of the interpoles varies with the load on the generator. So, if the field distortion also varies with the load, the magnetic fields set up by the interpoles must counteract the effects of the distorted fields about the armature windings. Another way of putting it would be: the interpole acts as an electrical balance to keep the neutral plane in the same axis for all loads thrown on the generator. The generator used in the F-102A is a four-pole machine with interpoles. In generators of this class, field distortion, under normal flight conditions, is kept at a minimum for maximum efficiency. The life of the brushes is improved immeasurably by this method, and the output is kept at a constant level.

The generator brush was the third offender of commutation trouble, especially at high altitudes. The brushes used in the F-102A generator are what are known as "halide treated" cord brushes. Halide is a binary, or a two-element chemical compound, usually consisting of chloride, bromide, iodide, or flouride. When impregnated into the brush, halide acts as a scouring agent on the commutator at high altitudes. This cuts commutation even more and prolongs the life of generator brushes.

While we are on this subject of brushes it might be a good idea to give you a little of the background behind the hurry and scurry of a few years ago, when it was found that above certain altitudes generator brushes became the real bugaboo of the aviation industry. It's a sad story, but it has a happy ending, thanks to engineering knowledge which resulted in the brush now used in the F-102A, whose operation at high altitudes makes unusual demands on brush efficiency.

With the advent of World War II, serious generator brush trouble was encountered at altitudes of 25,000 feet and above. The situation became so critical that high altitude flying and precision bombing had a very questionable future. Brushes that would last several hundred hours under normal flight conditions would, in some cases, disintegrate in a matter of minutes. Fires, power failures and crashes resulted—lives were lost. Completely new generators were installed for each mission. To simply replace brushes was too risky. A number of things were tried. Brushes were tipped with paraffin, soaked in compounds, impregnated, hardened, tested and retested until finally high altitude brush life was increased 50 times beyond that of the very best of previous brushes. It was due to this research that we now have the "halide treated" brush. With halide treated brushes set for sparkless commutation, and with the additional features of interpoles and compensating windings in the negative leg of the circuit, we should have, under most in-flight conditions, a constant voltage output at the generator terminals. If a generator is not loaded beyond its rating, and if the simple maintenance procedures outlined in Chapter III of this supplement are followed, it should give satisfactory service for hundreds of hours.

Unfortunately, generators in the field do not operate under ideal conditions. There are climatic, high altitude, and dust-ladened atmospheric conditions, as well as tropical operations to be faced. Even with these abnormal operational characteristics, if proper preventive maintenance is used, few power failures will result, and a constant output at the generator terminals will make everybody happy.

GENERATOR TERMINALS.

On most aircraft generators, the electrical connections are made to the generator terminals marked, B, A, D, and E. This is also true of the generator installed in the F-102A.

Internally, the positive armature lead in the generator connects to the B+ terminal. The internal wiring schematic, figure 2-31, will help you to follow this explanation. As you can see, the negative armature lead is connected through the interpole and compensating windings to the E− terminal. The positive end of the shunt field winding is tied into terminal A. The opposite end of this winding connects to the negative generator brush. In this manner, terminal A receives current from the negative generator brush through the shunt field winding. This current then passes through the voltage regulator and back to the armature through the positive brush. Terminal E is connected to ground.

You will remember, that in most airplanes, terminal D is used when two or more generators are operating in parallel. In the F-102A, where only one d-c generator is used, it is still designated as the equalizer terminal, but is connected to the low sides of the field relay trip coil through the d-c generator control panel, which we will get to in a moment. But first let's discuss the regulation of generator voltage generally.

REGULATION OF GENERATOR VOLTAGES.

At this point in our discussion it should be quite clear that the efficient operation of the electrical equipment in an airplane depends on a constant voltage supply from the generator. Among the many factors determining the voltage output of a generator, only one—the strength of the field current—can be conveniently controlled. The illustration in figure 2-32, can best describe what we mean by this.

If you set the rheostat to increase the resistance in the field circuit, less current flows through the field windings, and the strength of the magnetic field in which the armature rotates decreases. How does this affect the output? It too, decreases. But, if you decrease the resistance in the field circuit by regulating the rheostat, more current flows through the field windings which, of course, simply means that the magnetic field becomes stronger and the generator therefore produces a greater voltage.

It might be a good thing to remember that a voltage regulator must automatically control generator voltage over a wide range of generator speeds as well as loads. This is especially true in a d-c electrical power system. But, whether we are talking about a direct-current or alternating-current power system, the voltage regulator cannot do the job alone. This brings us to the additional relays and protective devices that make up the d-c control panel used on the F-102A.

F-102A D-C GENERATOR VOLTAGE CONTROL.

Without voltage controls and reverse current protection, a d-c generator system is worthless. Circuit protective

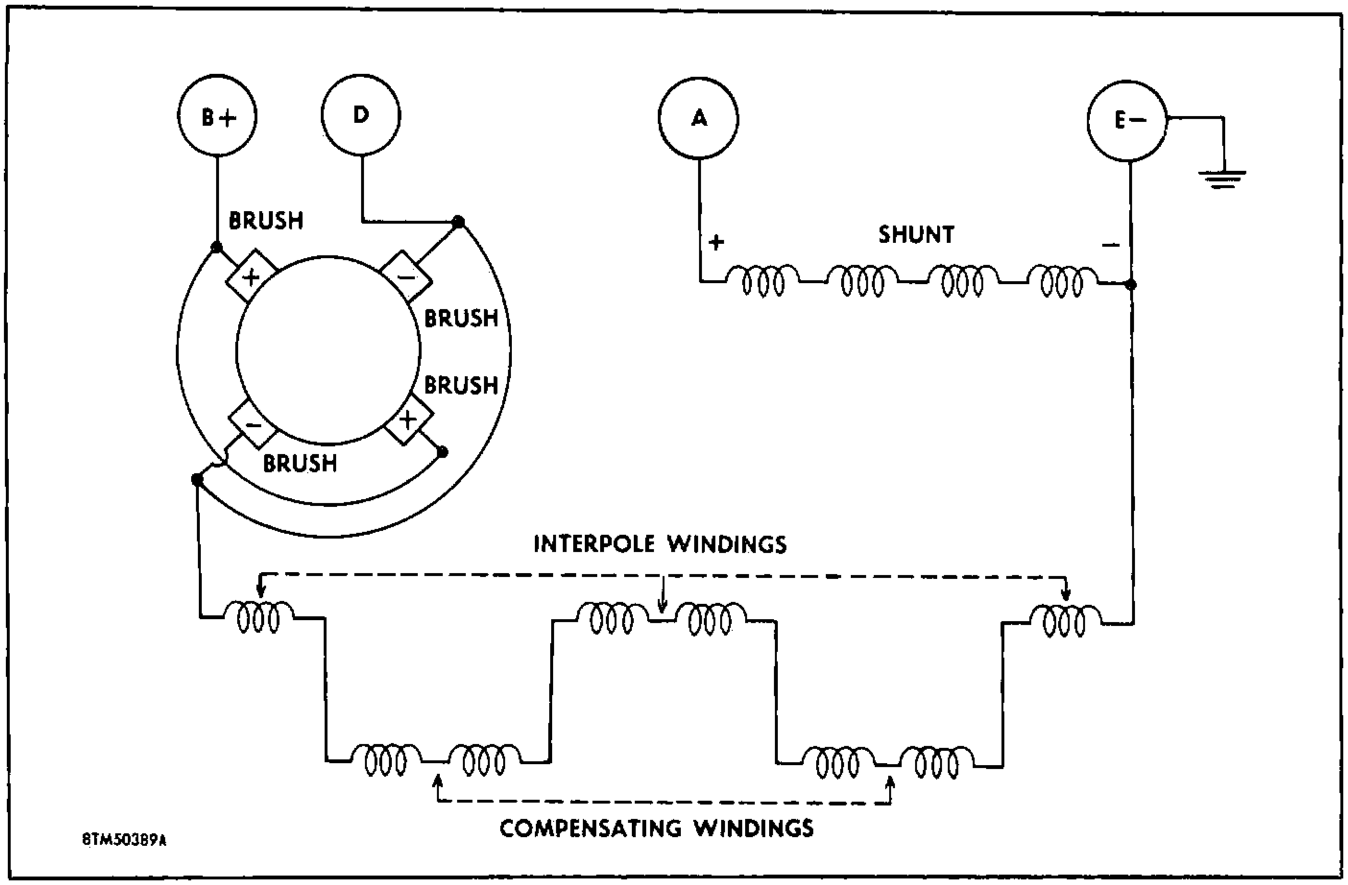

Figure 2-31. F-102A Generator Wiring Schematic

and controlling devices were discussed at length in another portion of this chapter. The fuses, circuit breakers, and relays mentioned are as important to an electrical power system as the brakes and steering knuckle on your automobile. By the same token, airplanes as we know them won't fly without wings—they won't fly, for very long at least, without the proper voltage controls, just as it was found that they wouldn't operate at high altitudes with the conventional pre-war carbon generator brushes.

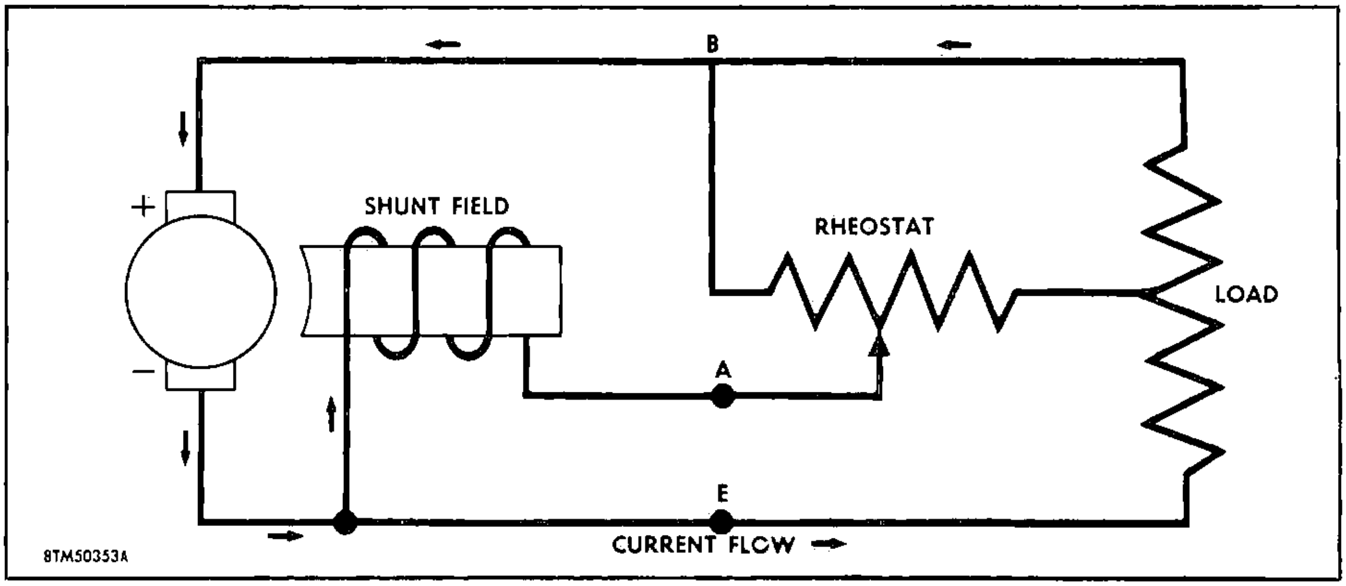

Figure 2-32. Generator Voltage Regulation by Field Rheostat

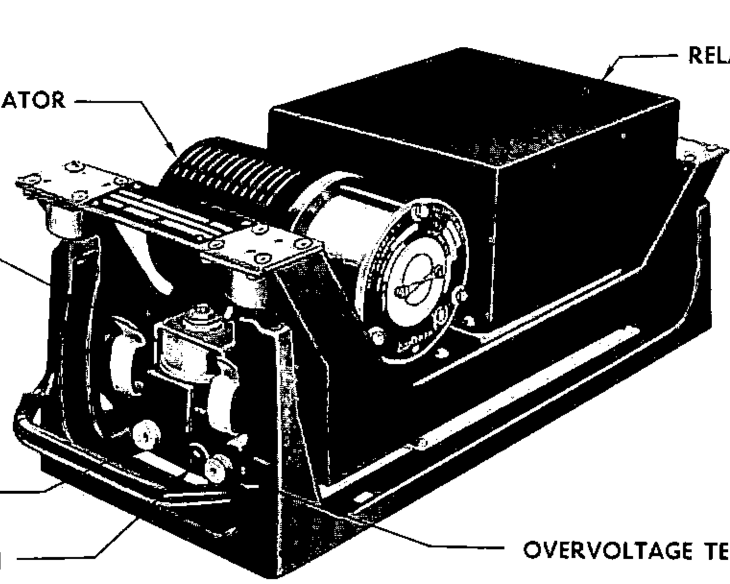

Figure 2-33. F-102A D-C Control Panel

THE F-102A D-C CONTROL PANEL.

The d-c control panel in the F-102A is in the nose wheel well. While it is easily accessible for testing and adjustment on the ground, it cannot be adjusted from the cockpit while the airplane is in flight. The simplified thinking behind this is, why bother the pilot with an extra gadget when there is no particular reason for having one—the panel functions automatically. There is no d-c voltmeter on this airplane either. Here again, there is no need. More than likely you have noticed that in some of the more modern automobiles the ammeter has been left off the dashboard. Instead, a small red light glows when the generator in the automobile is not producing. Once the emf has built to the proper magnitude, the light flicks off and won't come on again until either the generator speed is below minimum rpm, or malfunction exists. In the F-102A, however, the main line d-c power system functions properly, or not at all. There is no half-way, or look-out-something-is-to-happen, warning, until the battery (the airplane's blood bank) starts feeding the essential bus. Not unlike the dashboard light in an automobile, a d-c power failure in the F-102A will be indicated automatically through the d-c power failure relay, which illuminates the d-c power failure warning light in the cockpit of the plane.

The basic purpose of the control panel is to provide complete protection for the 28-volt d-c electrical system. As you can see in figure 2-33, there are two main divisions in the panel: the carbon pile regulator and, under cover, the various relays and other components which will be discussed further along in the text.

As a flight-line maintenance man, the scope of your duties will rarely encompass overhaul or adjustment of the field relay, overvoltage relay, differential relay and other components found in this covered section of the panel. Nor will your duties necessitate overhaul of carbon pile voltage regulator. You will however, be called on to adjust the output voltage of this regulator through the rheostat control, and to make periodic checks of the regulator output by connecting a voltmeter across the voltage test jacks marked, "VJ+" and "VJ–." The discussion of this control panel will therefore deal with the reason for its existence in the system and the functions it performs in the system.

CARBON PILE VOLTAGE REGULATOR.

The carbon pile voltage regulator on the F-102A is adjusted to 27.5 volts. Its regulation limits are from 26 to 30 volts, plus or minus 0.7 volts. What is the theory behind this adjustment? All other things being constant, the output voltage of our generator depends directly on the speed of the armature as it cuts those lines of force. The armature speed is sustained by the variable speed drive mechanism in the constant-speed unit, which in turn is affected by the varied speeds of the airplane power plant. As we discussed previously, the amount of field flux is determined by the field current. So, by controlling the field current, it is possible to control the output voltage. A variable resistance, therefore, in series with the field winding in the generator can and is used as the controlling element. In other words, all that is needed for effective automatic voltage regulation is a suitable resistance device that utilizes the main line-voltage variations to control this resistance. We have this in the carbon pile regulator—the silent sentinel in the aircraft.

In the cutaway of a carbon pile voltage regulator, figure 2-34, you can see that the carbon pile itself consists of a stack of carbon discs. It is the resistance of these discs and the amount of air between them that determines the efficiency of the regulators. If you were to

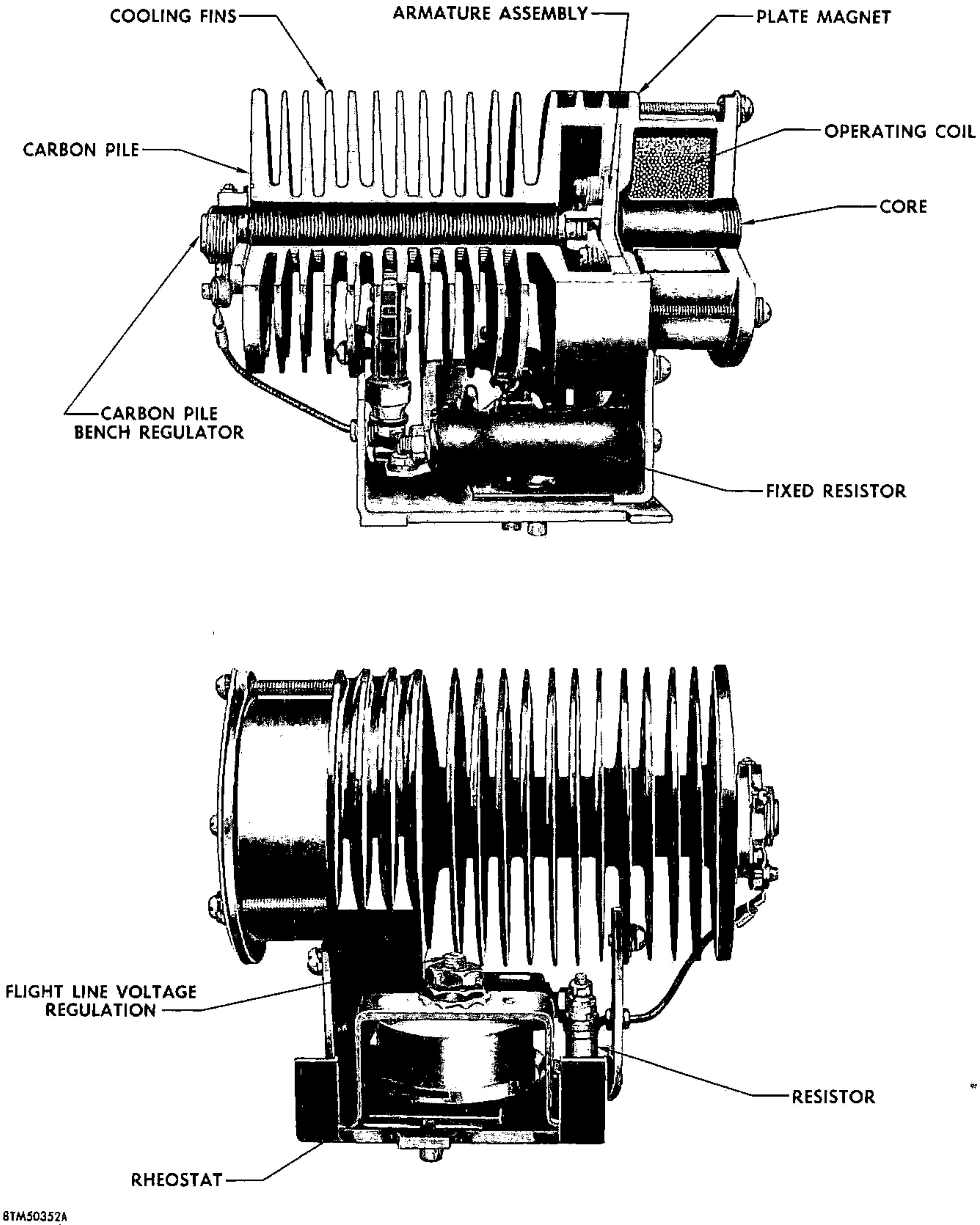

Figure 2-34. F-102A Carbon Pile Voltage Regulator

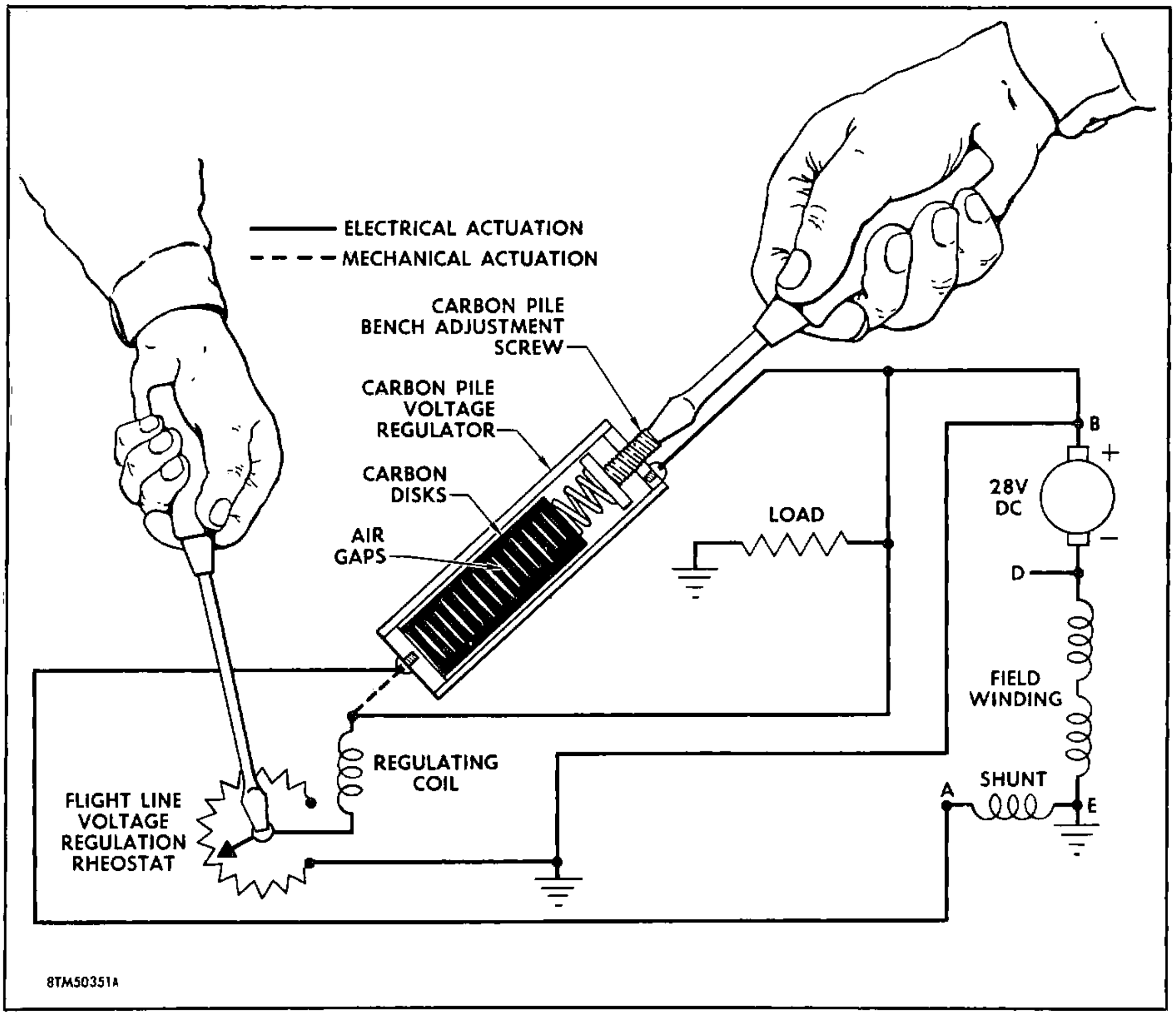

Figure 2-35. Simple Carbon Pile Voltage Regulator

insert a screwdriver into the screw's slot and turn it very slowly clockwise, the stack would be compressed and the air gap between the discs would be reduced, thereby reducing the resistant qualities of the regulator. What happens now to the voltage? Is it increased or decreased? Increased, of course. By turning the adjustment screw counterclockwise, the area between the discs is increased and the resistance is therefore increased. And the voltage! That's right! It is decreased. *This adjustment procedure, however, would be used only in shop overhaul. The actual flight line voltage regulation is made at the rheostat, which we shall cover later.*

The pressure on the carbon pile depends upon two opposing forces—spring tension and the action of an electromagnet. The spring tends to compress the carbon pile, and the electromagnet exerts a pull which tends to decrease the pressure. Actually, the voltage regulator is controlled by the electromagnet. The main winding of this *solenoid* is connected across the generator output voltage. The current flowing through this magnet coil depends on the output voltage of the generator, and decreases and increases with that voltage. The core of the magnet is so arranged that an increase of voltage releases the pressure on the carbon discs, while any decrease in voltage allows the spring and core to exert more pressure. If the generator output tends to rise above the voltage setting (set by the rheostat), the current flowing through the magnetic shunt windings will also increase. This increase in current produces an increase in attraction on the regulator's armature which reduces the pressure on the carbon pile. And the carbon pile, being in the field circuit, increases the field resistance. This increase of course reduces the current in the

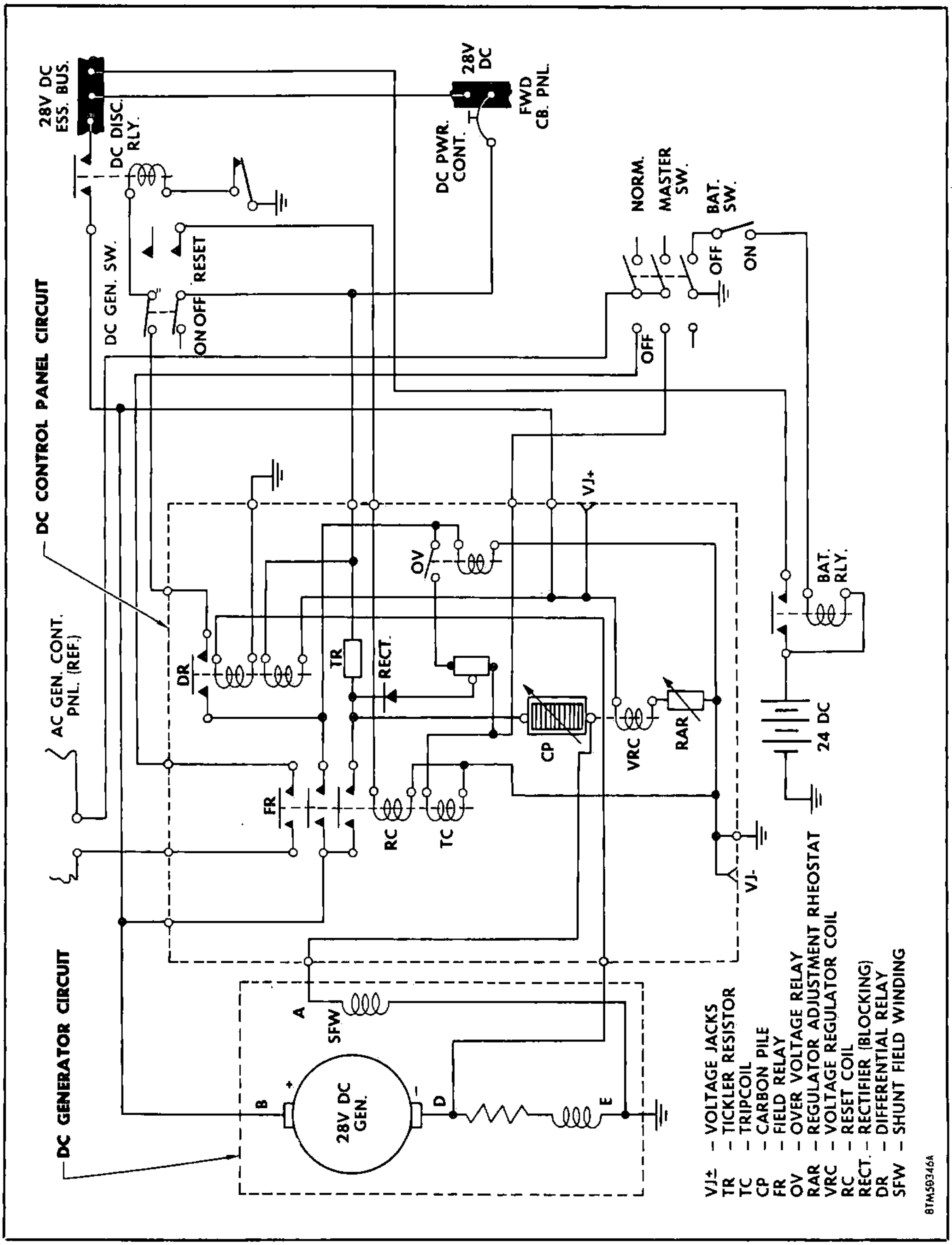

Figure 2-36. D-C Control Panel Wiring Schematic

field and in turn, the output voltage. Now, if the voltage output should fall below the voltage set by the rheostat, the entire action of the regulator, as we have explained it, is reversed.

In summary, there are four advantages to using carbon pile regulators in aircraft: Perhaps the major benefit found in this regulator is its sensitivity and the resulting resistance changes which take place almost instantly in response to any fluctuation in generator output. Increased power capacity gives it a greater range than would the vibrating relay voltage regulator or one of the carbon-plate types. Radio interference is also minimized due to the absence of movable contacts. And, a carbon pile regulator is not affected by the normal stress and strain of flight. It is possible under heavy G loads for the carbon discs to stick together in a temporary binding condition, but this condition was overcome by placing the voltage regulator near the center of gravity of the aircraft.

CONTROL PANEL RELAYS.

The covered portion of the control panel in the F-102A contains a differential relay, overvoltage relay, field relay, and what is called a "tickler" resistor. A blocking rectifier is also used in the F-102A for additional reverse current protection. These relays are the remote control system of the generator circuit.

Basically, whenever the generator output voltage is higher than the bus voltage, it is a generator relay switch that automatically connects the generator to the essential and non-essential bus bars. Bus bars, as you know, are the distribution points where the current is then channeled throughout the airplane. If for any reason the generator voltage drops below the required bus voltage, the switch disconnects it from the bus bars and the battery takes over. It is in this way that reverse current flowing from the battery to the generator is prevented from damaging the generator. Whenever the generator control switch is opened, the generator relay switch opens and disconnects the generator from the buses. This is what is meant by remote control of the generator circuit. The schematic, figure 2-36, is typical of the generator remote control system found in the F-102A. All symbols shown on this schematic are explained in Chapter III when we analyze the d-c circuits.

DIFFERENTIAL RELAY.

The reverse current differential relay, as it is generally referred to, controls the main line switch connected between the generator and the essential bus. Usually a differential relay switch connects the generator to the essential bus when the generator output voltage exceeds the bus voltage by 0.35 to 0.65 volt. The closing of this relay is brought about through the combined efforts of the differential voltage coil, reverse current coil, and the voltage relay. The differential voltage coil senses *at all times* the difference in voltage between the generator and the bus side of the generator to the main line switch, or contactor. The reverse current coil disconnects the main line contactor of the system when current flows from the bus through the generator to ground. The voltage relay controls both. The illustration, figure 2-37, is typical of the internal connections in the differential relay as used in the F-102A control panel.

The relay marked A, is the differential relay. Relay B is the voltage relay. Both relays consist of permanent magnets which pivot between the pole pieces of temporary magnets on which the relay coils are wound. Voltages of one polarity set up fields about the temporary magnets. Their polarities are such that the permanent magnet moves in a direction which causes the relay contacts to close, and the voltages of the opposite polarity to establish additional fields causing the contacts to open. The differential relay has two coils wound on the same core. The coil-operated contactor, as shown here, and called the main contactor, consists of movable contacts operated by a coil that has a movable iron core.

When the pilot positions the d-c generator switch to RESET momentarily (in case the d-c control panel may have tripped) and then to ON, he connects the generator output to the voltage relay coil as shown in figure 2-37. When this voltage builds to 22 volts and the correct polarity is reached, the current flows through the voltage relay coil closing the contacts of the voltage relay. These contacts are magnetically latched in either the open or closed position. It is this action which completes the circuit from the generator to the battery through the differential coil. Whenever the generator voltage exceeds the bus voltage by ½ volt, this ½ volt is impressed on the differential coil. This causes sufficient current to flow in the proper direction to close the main contactor, and then to open it before any reverse current surges can occur during the generator buildup on an energized bus.

Let's pursue this a bit further. The reverse current coil is connected in parallel with the series field of the generator and in this way senses the generator load current. When the generator delivers current to the bus, the magnetomotive force (mmf) impressed by this winding on its core, tends to close the contacts even tighter. As the generator drops off in speed, the circuit starts to draw a reverse current from the bus (battery voltage) and the current in this winding reverses. When this reverse current reaches sufficient magnitude it weakens the magnetic field about the temporary magnet of the differential relay. This weakened field permits the spring to open the contacts, which breaks the circuit to the coil of the main contactor relay, and disconnects the generator from the bus. This generator-battery circuit is also broken when the pilot presses the generator switch to the OFF position. This opens the magnetically latched contacts of the voltage relay and deenergizes the differential relay coil.

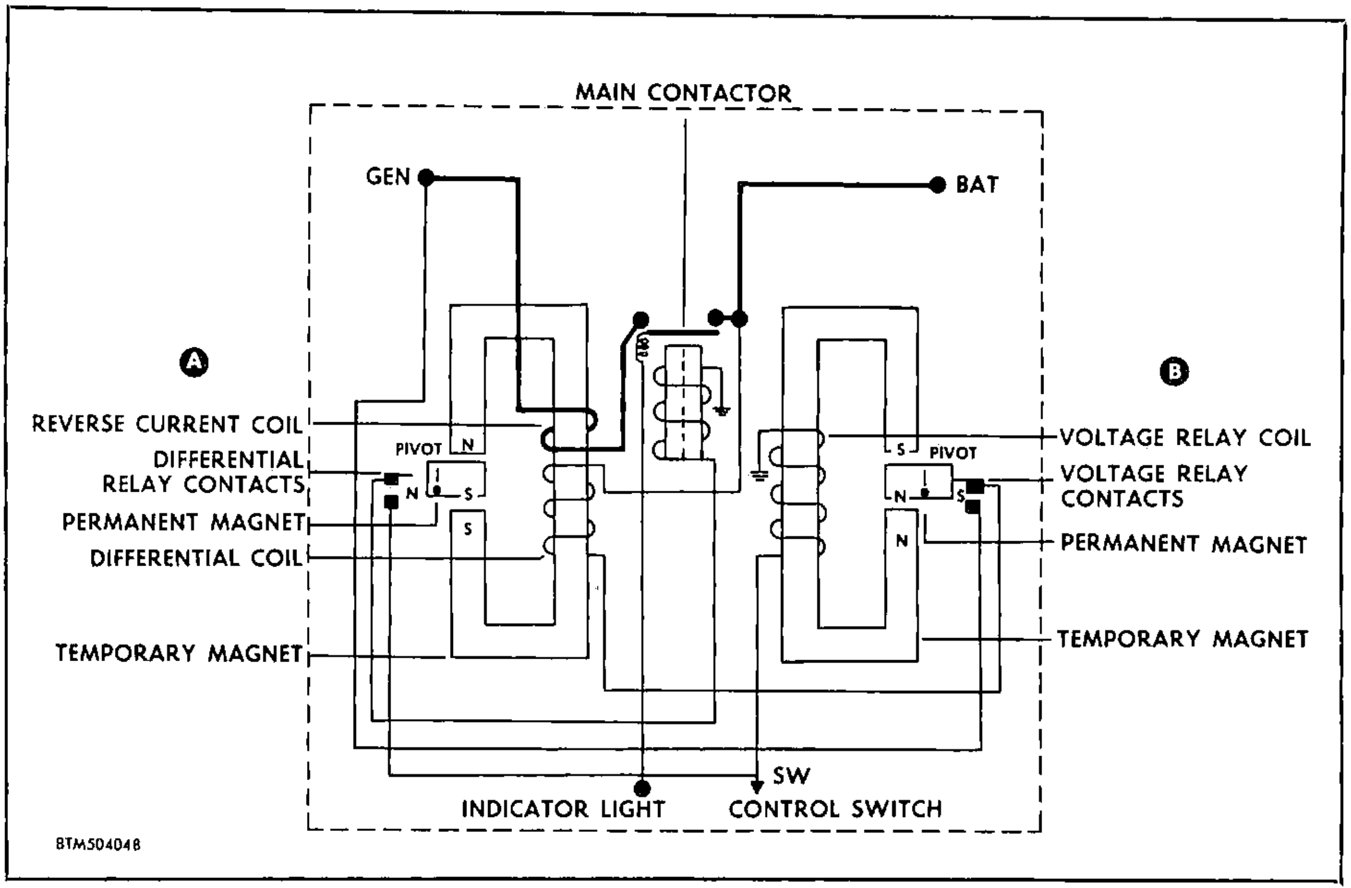

Figure 2-37. Typical Differential Control Relay

OVERVOLTAGE RELAY.

If for any reason an excessive overvoltage occurs in the circuit, it is the overvoltage relay which protects the system. In the F-102A, if there is a sustained overvoltage from 1 to 3 seconds of over 32 volts, the relay coils are energized closing the contacts. It is not, however, quite as simple an action as it sounds. This relay has an inverse-time characteristic. (See "Relay Switch in Circuits," in this chapter.) It is also spring-loaded. Inverse-time, in this case, simply means that nuisance, or reversal trips, are prevented due to a transient, or short-lived voltages, during switching operations.

This relay consists of a magnetic piston enclosed in a solenoid, as shown in figure 2-38. You can also find the overvoltage relay illustrated in the schematic, figure 2-36. The piston is hollow, closed at one end, and is normally located in the end of the cylinder away from the relay's armature. When we speak of an armature in relation to a relay, we are talking about an armature as defined in a magnetic circuit—a soft iron core used to connect the poles of adjacent magnets. To activate the movement of the magnetic piston of this relay, air must flow between the piston and the inside wall of its tube or housing. In this way a time lag is produced. When an overvoltage is applied to the solenoid coil, the piston moves toward the armature descreasing the air gap. When the air gap has decreased to a factory-calibrated point, the armature closes and operates the relay contacts. The higher the applied voltage, the faster the relay operates. The spring returns the piston to its normal position.

Further overvoltage-overload protection is given the circuit by what is called a biasing coil. This coil reduces the minimum trip point when the generator is overloaded due to an overvoltage. Without this the generator might be damaged due to overload. And, if this occurred it would prevent the generator voltage from rising to the normal minimum trip level of the overvoltage relay. This brings us to the field control relay.

FIELD CONTROL RELAY.

We said, that the overvoltage relay would close when there was a sustained voltage of over 32 volts for 1 to 3 seconds. When this happens a circuit is completed to the trip coil of the field control relay. The closing of the field control relay trip circuit opens the generator shunt field circuit and completes it through a resistor. This of course, causes the generator voltage to drop. The field relay interlock provides a trip-free reset action. This field relay reset circuit is energized from the essential bus when the spring-loaded reset switch is actuated from the cockpit.

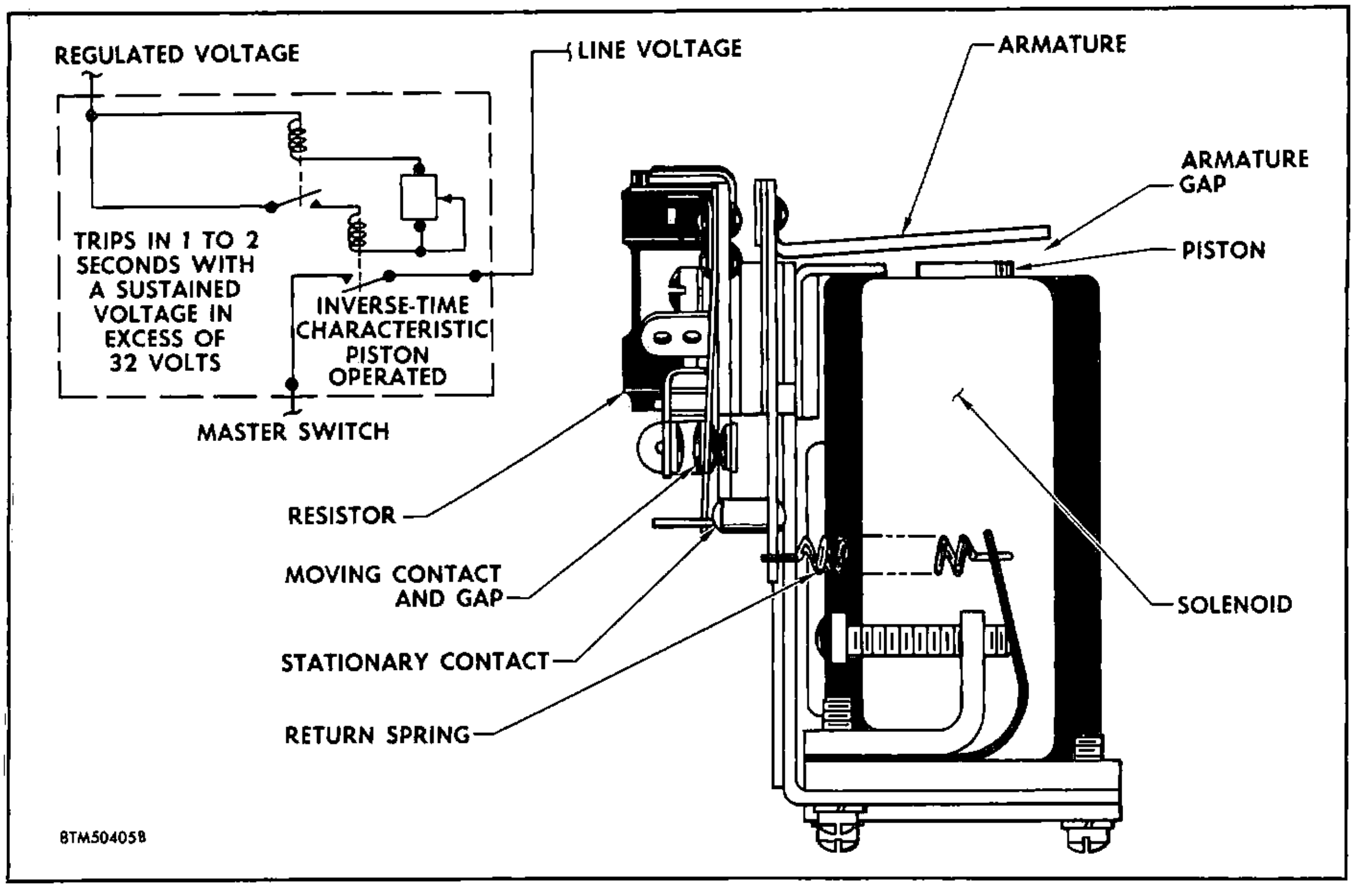

Figure 2-38. Overvoltage Relay

F-102A ESSENTIAL AND NON-ESSENTIAL BUS SYSTEMS.

In the sections leading up to this one, we discussed the protective and controlling devices in the d-c electrical power system between the source of power, the generator, and the essential and non-essential buses. In this section we will discuss the distribution point of this power, namely the buses, to the other circuits installed in the F-102A.

The word "bus" is a contraction of the word "omnibus," which as you probably know means a "carry-all." We see these vehicles of distribution every day, in cities, towns, and on the open road—you probably arrived at your present place of duty in one. The difference between this type of bus, used for transportation of humanity, and the one that we are presently interested in, is that one delivers people while the other delivers electrons, to their particular points of destination.

The first airplanes with electrical starters had batteries placed close to the engine to reduce the heavy wiring. Starting was the only heavy load. The lights and instruments required little current. For this reason it was possible to connect the battery and generator directly to a small distribution bus located in the cockpit. Gradually, the increased demand for power brought heavier loads to the central distribution points, until today buses are used throughout aircraft electrical power systems for distribution to the branch circuits. The bus system used in the F-102A is the essential and non-essential bus system.

ESSENTIAL AND NON-ESSENTIAL BUS SYSTEM.

The d-c electrical power in the F-102A is distributed from buses designated as *essential* and *non-essential* The essential bus supplies those circuits that we consider essential to the *safety* and *continued flight* of the airplane. Figure 2-39 shows those circuits supplied by the essential bus, and those fed by the non-essential bus.

As you can see, the power to the d-c non-essential bus is routed through a non-essential bus tie-relay. If for any reason there is generator failure during normal in-flight operations, this relay is deenergized, thus disconnecting the non-essential bus from the essential bus. During ground operations, when external power is used, a d-c external power interlock relay routes power to the solenoid of the non-essential bus tie-relay. In this manner the non-essential bus is energized. We will discuss this operation more fully in the next chapter.

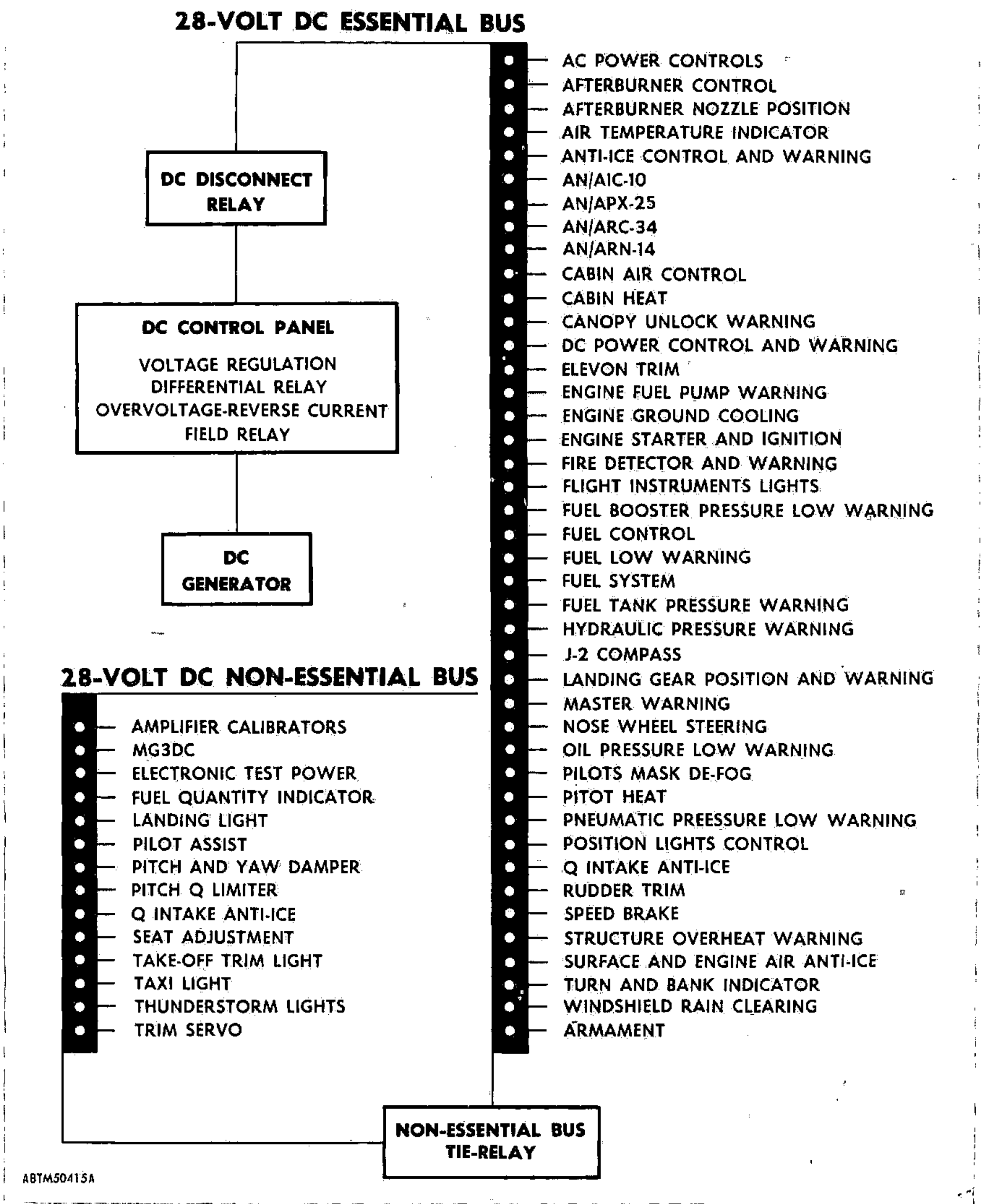

Figure 2-39. F-102A Essential and Non-Essential Buses

CIRCUIT BREAKER PANEL LOCATIONS AND CONTENTS.

There are nine individual circuit breaker panels scattered throughout the F-102A. Each of these panels contains a segment of either the essential or non-essential buses. These buses are located in the rear of the various circuit breaker and switch panels. These panels contain both the d-c and a-c buses and the circuit breakers. But for our purpose in this chapter, when we are discussing the d-c system, only the d-c buses and circuit breakers and their location will be listed. Figure 2-40 shows the location of these circuit breaker panels. The largest of these panels is the main wheel well circuit breaker panel (E). Then we have the nose wheel well circuit breaker panel (C), the nose wheel switch panel (B), and the panel in the upper electronics compartment (G). In the left-hand side of the cockpit is the forward auxiliary circuit breaker panel (D), located slightly above the cockpit floor at the forward end of the left-hand console and below the cabin altimeter. The left-hand aft circuit breaker panel (A), is located immediately below the cowling angle and above the "G" suit control valve at the rear of this console. Opposite this panel, and on the right-hand side of the cockpit in relatively the same position, is the right-hand aft circuit breaker panel (F). Just below this panel, on the aft end of the right-hand console, will be found the right-hand console communication and radar circuit breaker panel (F). At the forward end of this console is the electrical power switch panel. The utility switch panel is located on the skirt below the pilot's instrument panel.

The illustrations and tables concluding this chapter show you where each circuit breaker panel is, what it looks like, and what it contains. The load capacity of the individual breaker is given, its bus, and the circuit protected. While there are bus connections in these panels expressly for the a-c electrical system, those located in the following charts are for the d-c electrical system only. The a-c will be covered in Chapter IV.

In the first column, "Circuit Breaker Decal," you will find, in some cases, two different decal designations. The first one quoted is on the earlier airplanes; the second is on the later models.

MAIN WHEEL WELL CIRCUIT BREAKER PANEL.

CIRCUIT BREAKER DECAL	AMPERAGE AND BUS	CIRCUIT
FIRE DET FIRE DETECTOR	5-amp, essential 5-amp, essential	Fire detector
STR OVHT STRUCTURE OVER HEAT	5-amp, essential 5-amp, essential	Structural overheat warning
OIL PRESS OIL PRESSURE LOW	5-amp, essential 5-amp, essential	Oil pressure low warning
TANK PRESS TANK PRESSURE LOW	5-amp, essential 5-amp, essential	Fuel tank pressure warning
BSTR PRESS BOOSTER PRESSURE LOW	5-amp, essential 5-amp, essential	Fuel booster pressure low warning
FUEL LOW FUEL LOW	5-amp, essential 5-amp, essential	Fuel low warning
FUEL PUMP FUEL PUMP FAILURE	5-amp, essential 5-amp, essential	Engine fuel pump warning
RH BSTR PUMP BOOSTER PUMP RH	5-amp, essential 5-amp, essential	Fuel system
LH BSTR PUMP BOOSTER PUMP LH	5-amp, essential 5-amp, essential	Fuel system
ANTI-ICE ANTI-ICE POWER	10-amp, essential 10-amp, essential	Surface and engine air anti-ice
IGN PWR IGNITION POWER	10-amp, essential 10-amp, essential	Engine starter and ignition
FUEL CONT FUEL CONTROL	10-amp, essential 10-amp, essential	Fuel control system

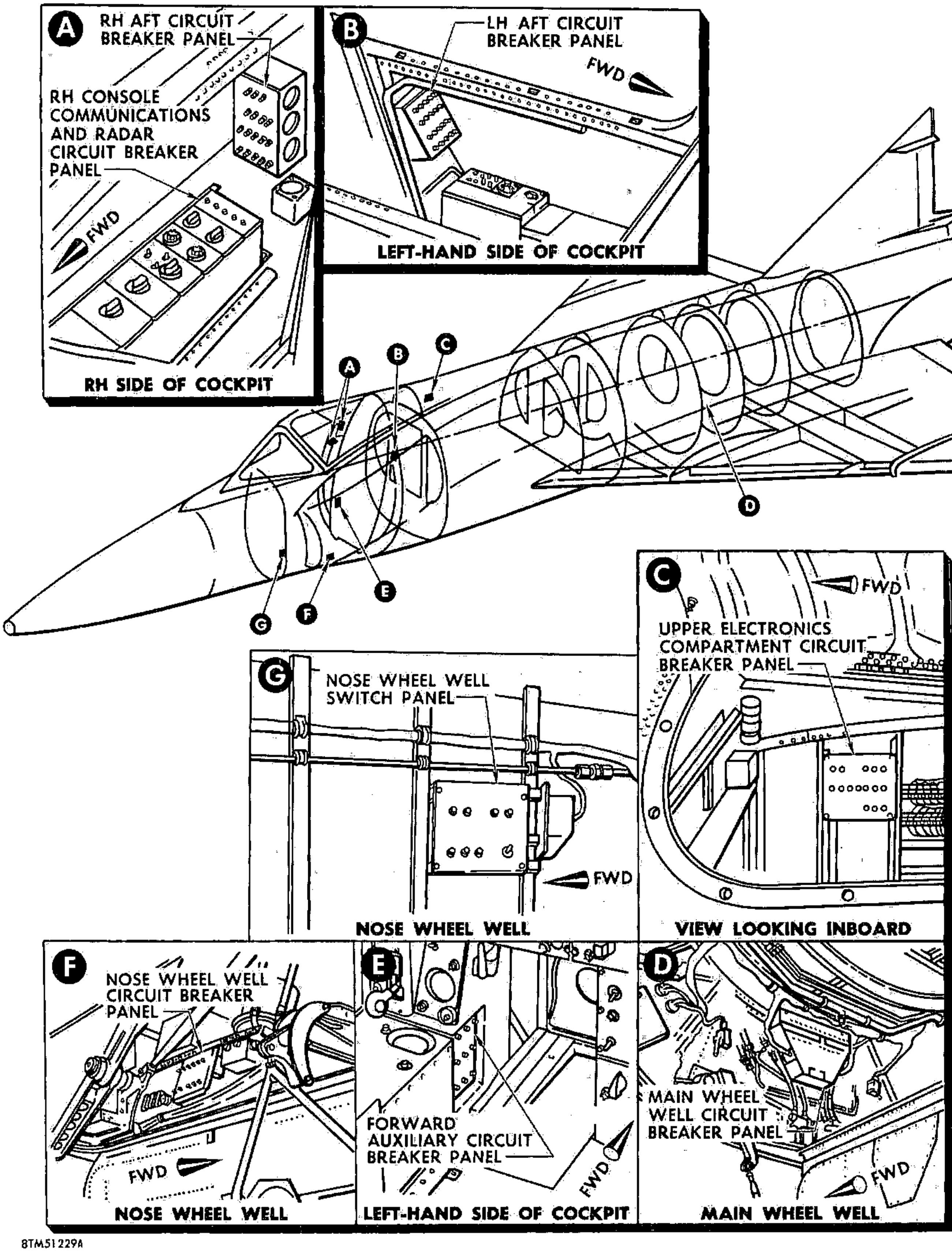

Figure 2-40. Circuit Breaker Panel Locations

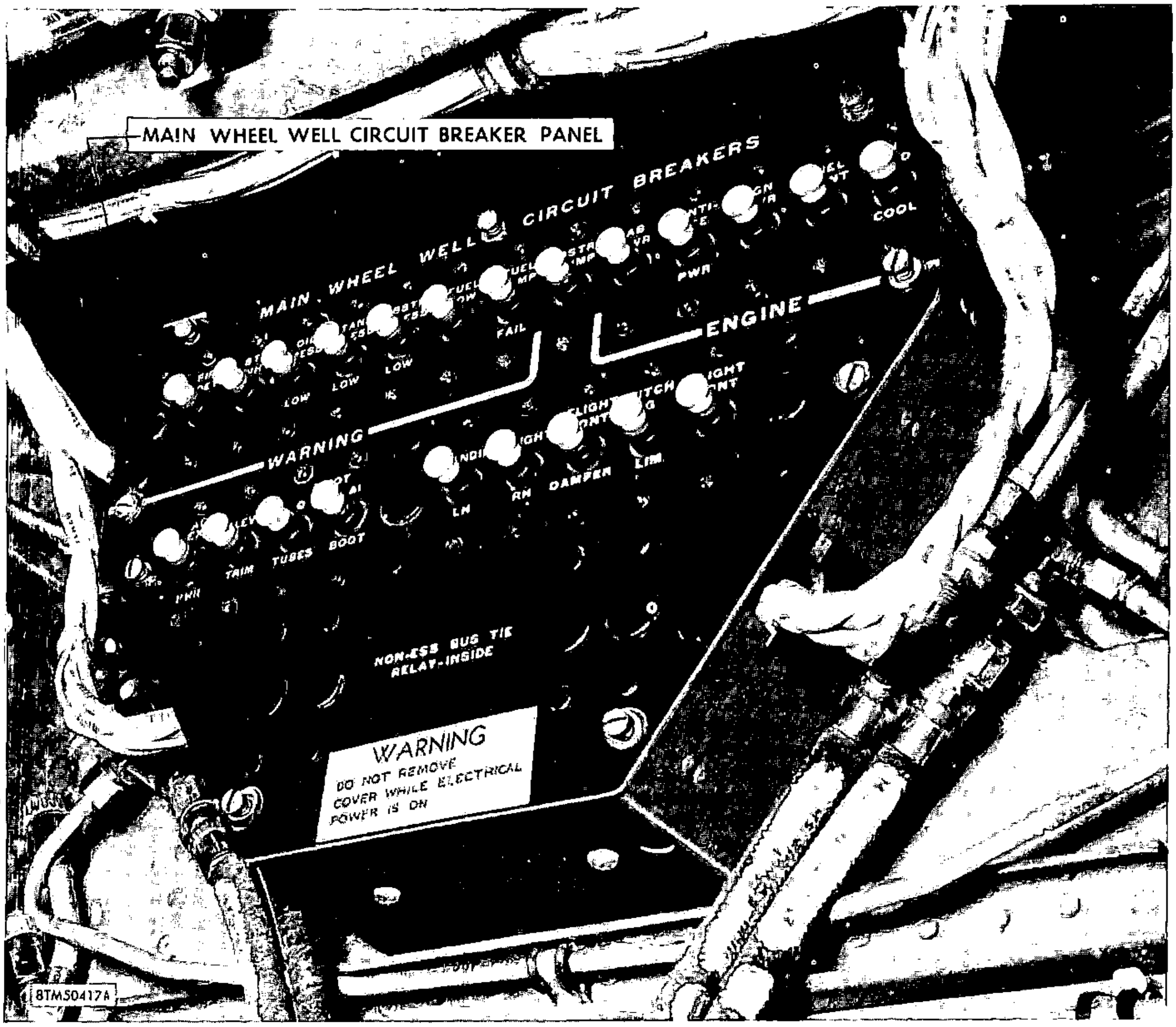

Figure 2-41. Main Wheel Well Circuit Breaker Panel

MAIN WHEEL WELL CIRCUIT BREAKER PANEL (Cont).

CIRCUIT BREAKER DECAL	AMPERAGE AND BUS	CIRCUIT
ENG GRD COOL ENGINE GROUND COOLING	5-amp, essential 5-amp, essential	Engine ground cooling
AC EXT PWR EXTERNAL POWER AC	5-amp, essential 5-amp, essential	A-C power
ELEVON TRIM ELEVON TRIM	5-amp, essential 5-amp, essential	Elevon trim
Q TUBES (PITOT HEAT) PITOT Q TUBES	10-amp, essential 10-amp, essential	"Q" intake anti-ice
AI BOOT (Airspeed Pitot Heat) PITOT ANTI-ICE BOOT	5-amp, non-essential 5-amp, non-essential	"Q" intake anti-ice

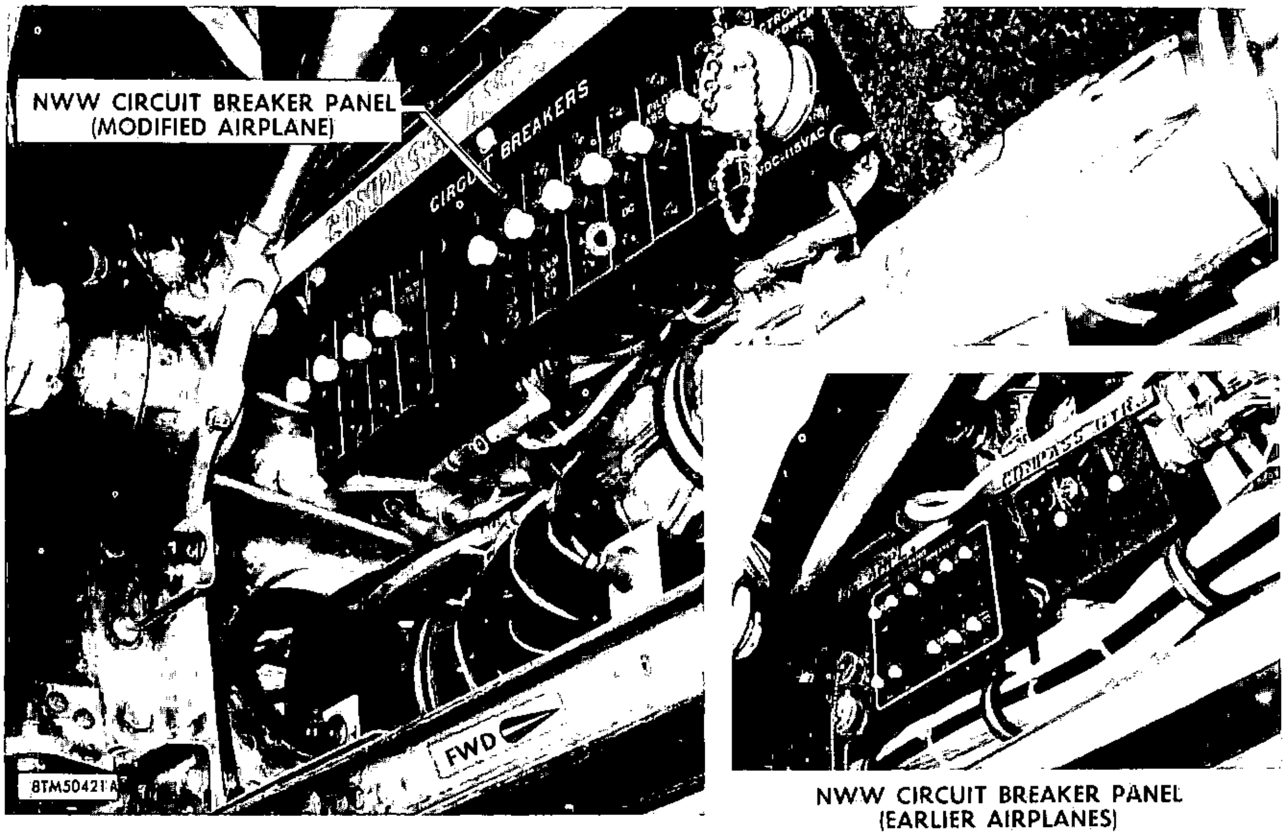

Figure 2-42. Nose Wheel Well Circuit Breaker Panel

MAIN WHEEL WELL CIRCUIT BREAKER PANEL (Cont).

CIRCUIT BREAKER DECAL	AMPERAGE AND BUS	CIRCUIT
LH LDG LT LH LANDING LIGHT	10-amp, non-essential 10-amp, non-essential	Landing lights
RH LDG LT RH LANDING LIGHT	10-amp, non-essential 10-amp, non-essential	Landing lights
FLIGHT CONT LT FLIGHT CONTROL DAMPER DC	5-amp, non-essential 5-amp, non-essential	Pitch and yaw damper
PITCH G (Limiter) PITCH G LIMITER	5-amp, non-essential 5-amp, non-essential	Pitch and yaw damper
ROLL RATE CONT ROLL RATE LIMIT CONTROL	10-amp, essential 10-amp, essential	Pitch and yaw damper
AB PWR AFTERBURNER POWER	10-amp, essential 10-amp, essential	Afterburner control
EXT FUEL EJC EXT FUEL EJECT	10-amp, essential 10-amp, essential	External droppable fuel tank trigger circuit
EXT FUEL SHUTOFF EXTERNAL FUEL SHUTOFF	5-amp, non-essential 5-amp, essential	External fuel circuit

NOSE WHEEL WELL CIRCUIT BREAKER PANEL.

CIRCUIT BREAKER DECAL	AMPERAGE AND BUS	CIRCUIT
ATTITUDE GYRO CONT, 28 VDC	5-amp, essential	Attitude gyro
NOSE WHEEL STEERING	5-amp, essential	Nose wheel steering
PWR SHUTOFF	5-amp, essential	D-C power

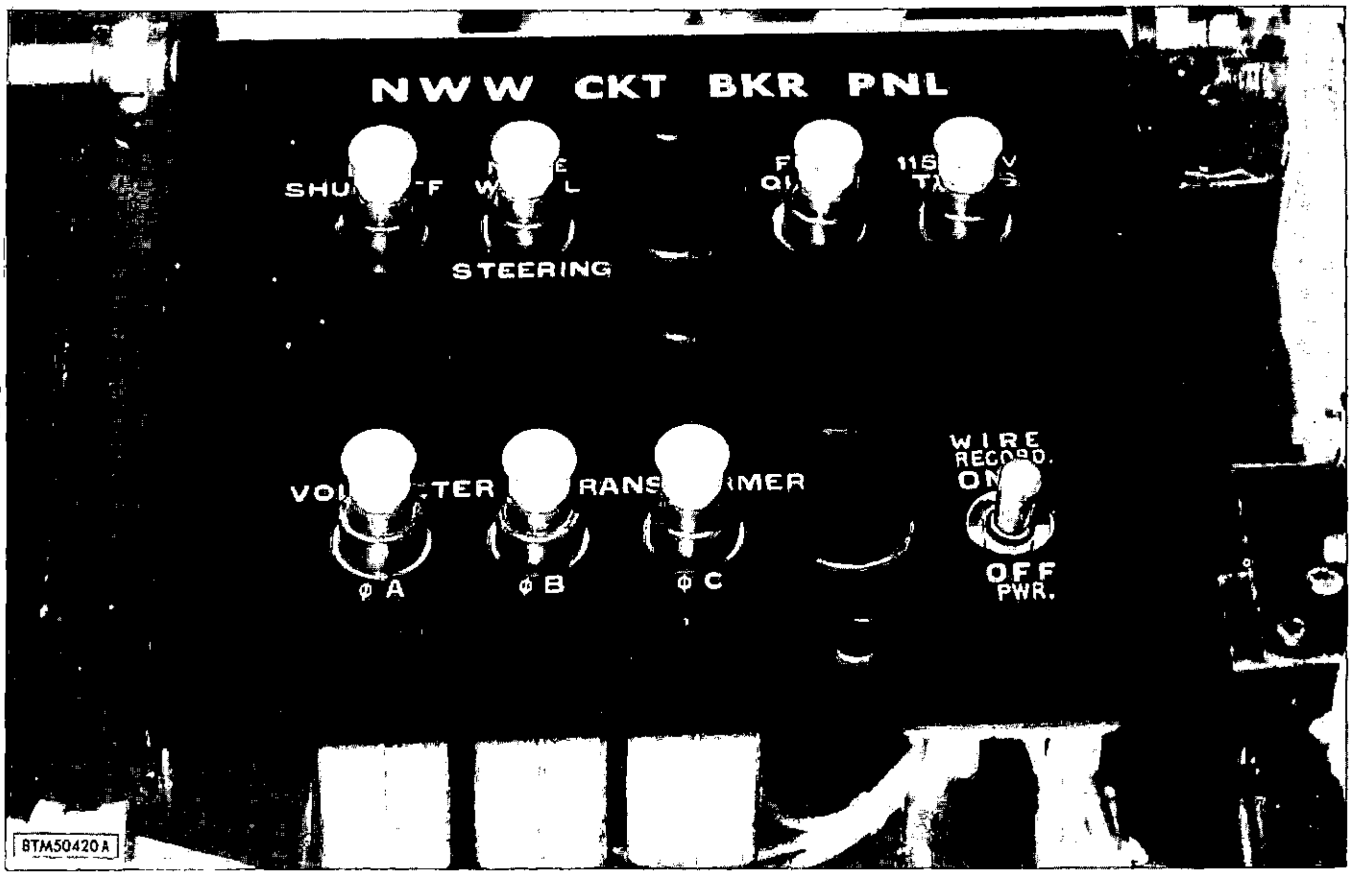

Figure 2-43. Nose Wheel Well Switch Panel

NOSE WHEEL WELL SWITCH PANEL.

CIRCUIT BREAKER DECAL	AMPERAGE AND BUS	CIRCUIT
TRIM SERVO DC	5-amp, non-essential	Pilot assist
PILOT ASSIST	5-amp, non-essential	Pilot assist
TAXI LT	10-amp, non-essential	Taxi light
J-2 COMP	5-amp, essential	J-2 compass
ARMAMENT POWER	5-amp, essential	Armament control
ARMAMENT RESET	5-amp, essential	Armament control
ELECTRONIC TEST POWER DC	10-amp, essential	Electronics test power
SIDE SLIP PROBE HEAT	5-amp, non-essential	Pitch and yaw damper system

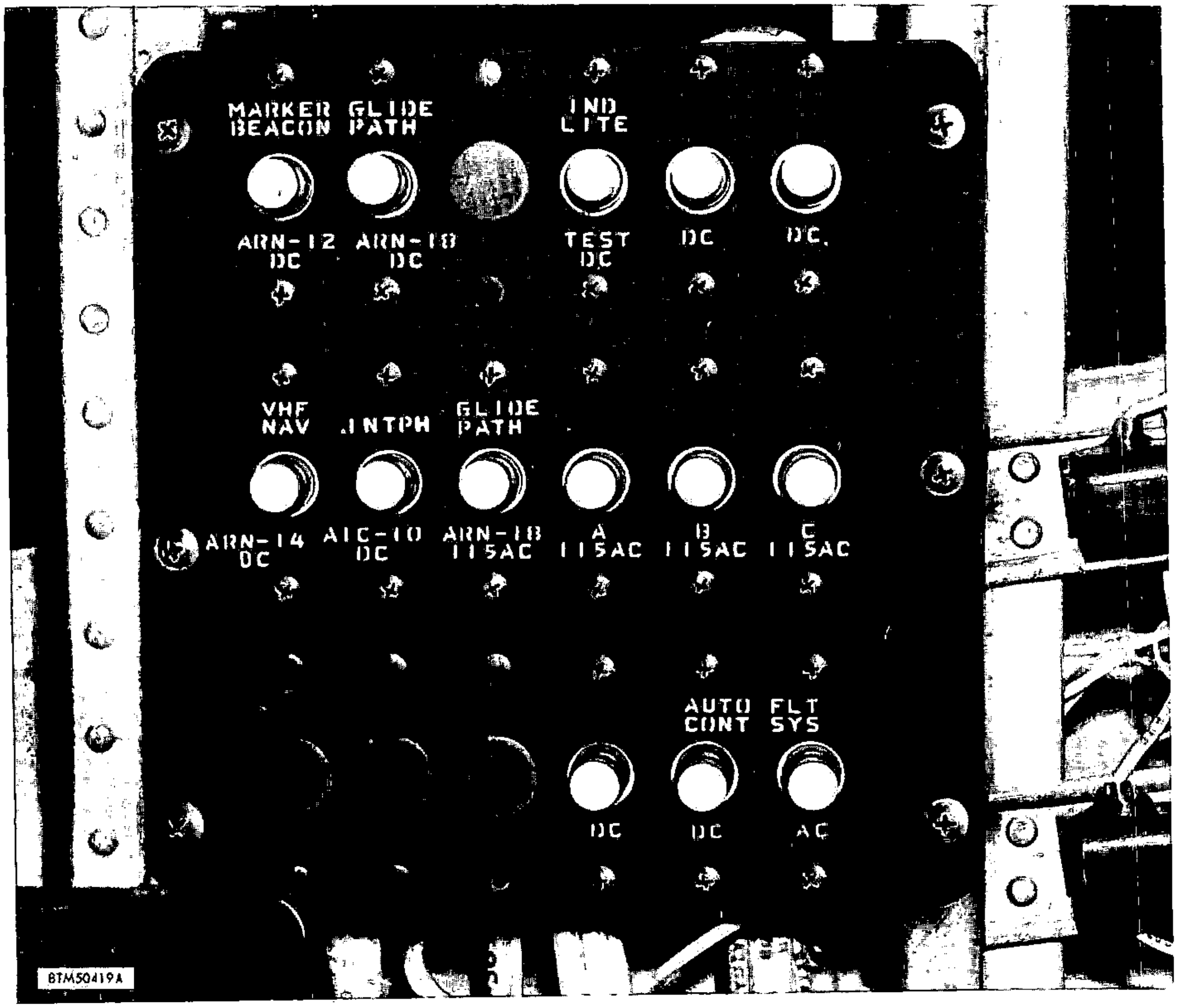

Figure 2-44. Upper Electronic Compartment Circuit Breaker Panel

UPPER ELECTRONICS COMPARTMENT CIRCUIT BREAKER PANEL.

CIRCUIT BREAKER DECAL	AMPERAGE AND BUS	CIRCUIT
MARKER BEACON ARN-12 DC	5-amp, non-essential	Marker beacon
GUIDE PATH ARN-18 DC	5-amp, non-essential	Glide slope, ARN-18
IND LITE TEST DC	5-amp, non-essential	VHF Navigation, ARN-14
MG-3 DC	50-amp, non-essential	Fire control system
MG-3 DC	50-amp, non-essential	Fire control system

UPPER ELECTRONICS COMPARTMENT CIRCUIT BREAKER PANEL (Cont).

CIRCUIT BREAKER DECAL	AMPERAGE AND BUS	CIRCUIT
VHF NAV ANR-14 DC	10-amp, essential	VHF Navigation, ARN-14
INTPH AIC-10 DC	5-amp, essential	Interphone, AIC-10
DATA LINK DC	10-amp, non-essential	Fire-control system
AUTO FLT CON SYS DC	5-amp, non-essential	Fire-control system

FORWARD AUXILIARY CIRCUIT BREAKER PANEL (COCKPIT).

CIRCUIT BREAKER DECAL	AMPERAGE AND BUS	CIRCUIT
PWR CONT DC	5-amp, essential	D-C power
PWR WARN	5-amp, essential	D-C power
FUEL SELECTOR LH	10-amp, essential	Fuel system
FUEL SELECTOR RH	10-amp, essential	Fuel system
LDG GEAR CONT	5-amp, essential	Landing gear control
A/B CONT	5-amp, essential	Afterburner control
START	5-amp, essential	Engine starter and ignition system

LEFT-HAND AFT CIRCUIT BREAKER PANEL (COCKPIT).

CIRCUIT BREAKER DECAL	AMPERAGE AND BUS	CIRCUIT
CABIN HEAT	10-amp, essential	Cabin heat control
CABIN AIR	5-amp, essential	Cabin air control
RAIN CLEAR	5-amp, essential	Windshield rain clear
ANTI-ICE	5-amp, essential	Surface and engine air anti-ice
WINDSHIELD OVHT	5-amp, essential	Windshield overheat control
MASK DEFOG	5-amp, essential	Pilot's mask de-fog
A/B NOZ POS IND	5-amp, essential	Afterburner nozzle position
LG POS IND	5-amp, essential	Landing gear position and warning
FUEL QTY	5-amp, essential	Fuel quantity
RUDDER TRIM	5-amp, essential	Rudder trim
SPEED BRAKE	5-amp, essential	Speed brakes
DE-FOG CANOPY	10-amp, essential	Anti-ice and de-fog
SEAT ADJ	5-amp, non-essential	Seat adjustment
EXT FUEL TANKS	5-amp, essential	External fuel circuit

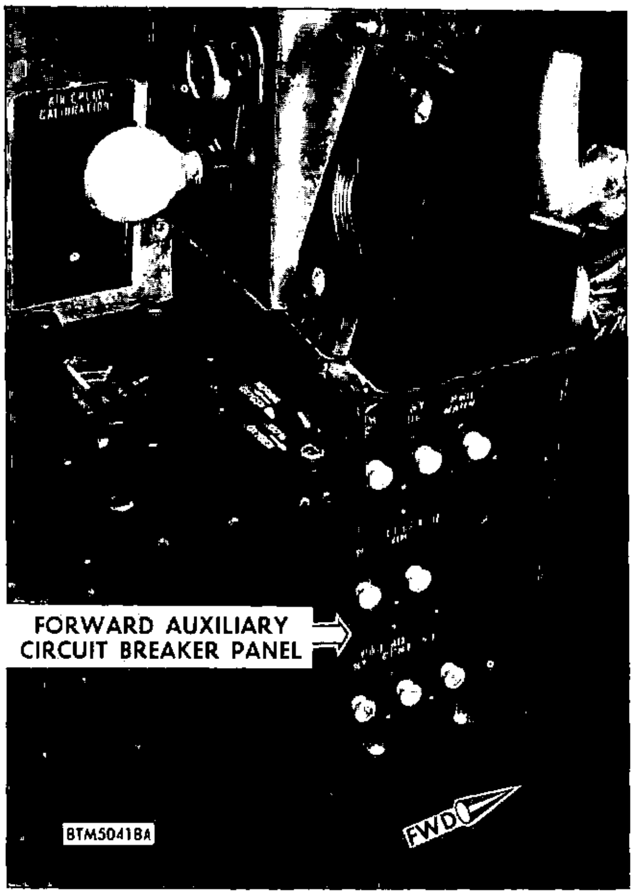

Figure 2-45. Forward Auxiliary Circuit Breaker Panel

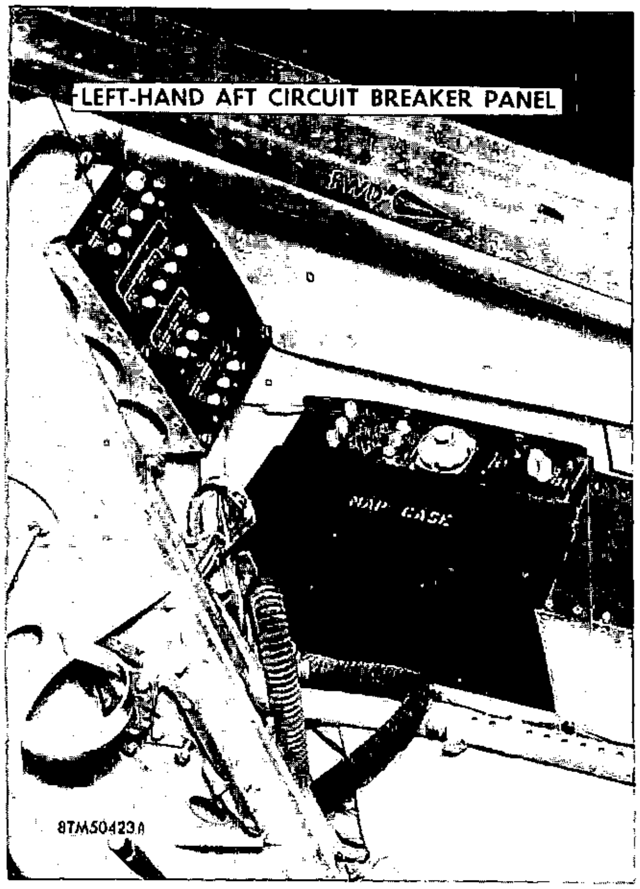

Figure 2-46. Left-Hand Aft Circuit Breaker Panel

RIGHT-HAND AFT CIRCUIT BREAKER PANEL (COCKPIT).

CIRCUIT BREAKER DECAL	AMPERAGE AND BUS	CIRCUIT
MASTER	5-amp, essential	Master warning
DC FAIL	5-amp, essential	D-C power
CANOPY UNLOCK	5-amp, essential	Canopy unlock warning
PNEU PRESS LOW	5-amp, essential	Pneumatic pressure low warning
HYD PRESS	5-amp, essential	Hydraulic pressure warning
LIGHTING FLT INST	5-amp, essential	Flight instrument lights
AIR TEMP	5-amp, essential	Air temperature indicator
STORM	5-amp, non-essential	Thunderstorm lights
POSITION CONT	5-amp, essential	Position lights
TURN & BANK	5-amp, essential	Turn and bank indicator

RIGHT-HAND CONSOLE COMMUNICATION AND RADAR CIRCUIT BREAKER PANEL (COCKPIT).

CIRCUIT BREAKER DECAL	AMPERAGE AND BUS	CIRCUIT
UHF COMM DC	25-amp, essential	UHF communication ARC-34
GROUND TO AIR DC	5-amp, essential	IFF radar
AIR TO AIR DC	5-amp, non-essential	IFF radar

UTILITY SWITCH PANEL.

CIRCUIT BREAKER DECAL	AMPERAGE AND BUS	CIRCUIT
PITOT HEAT	10-amp, essential	Pitot heat

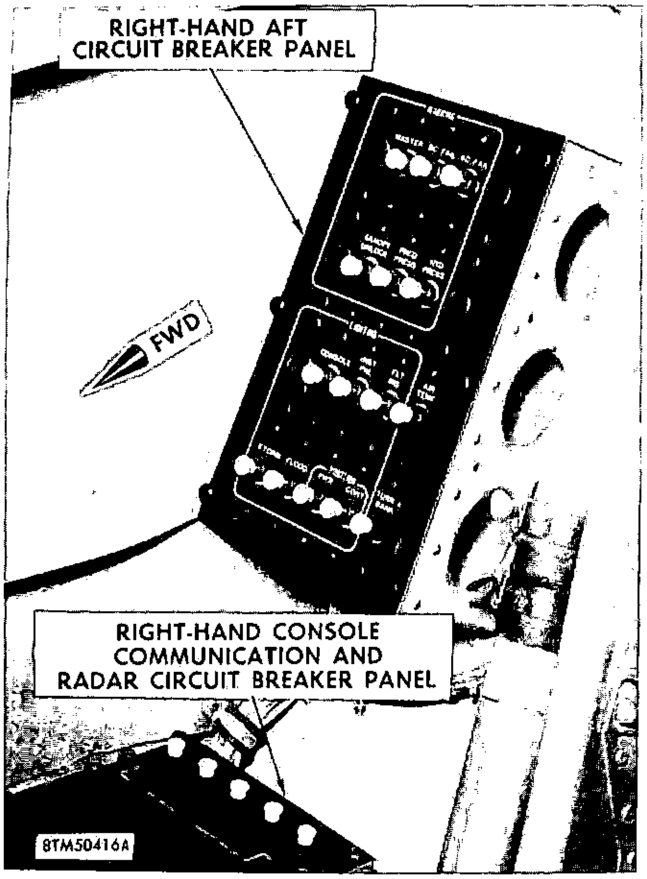

Figure 2-47. Right-Hand Console and Right-Hand Aft Circuit Breaker Panels

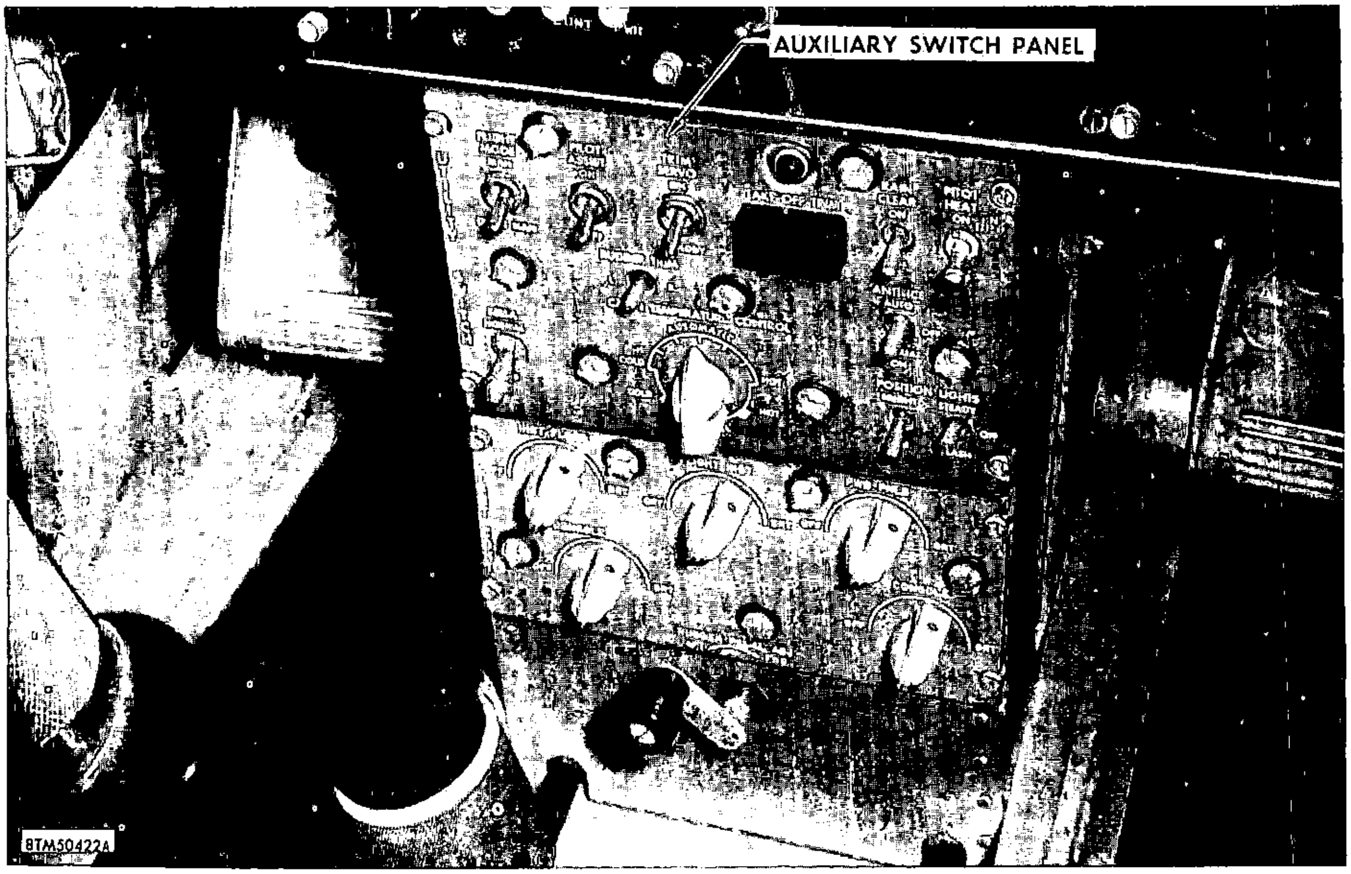

Figure 2-48. Utility Switch Panel

Chapter III

THE F-102A D-C ELECTRICAL POWER SYSTEM

Contents *Page*

In the last chapter we were interested mainly in the individual components and controlling devices used throughout the direct-current, electrical distribution system of the F-102A; why they are in the system, how they work, and where they function within the system. In this chapter we will wire them together, energize them, and analyze the d-c electrical system as a whole. But first, let's turn back a few years and find out how and why direct-current electrical systems have become what they are today.

D-C ELECTRICAL POWER IN EARLY AIRPLANES.

It was cold that morning. An icy wind had swept in from the Atlantic and across the sand dunes. Five men and a boy shivered in their heavy overcoats. They were witnessing the first successful flight of an engine-driven, heavier-than-air, man-carrying flying machine. The man on the flying machine, Orville Wright, was cold too—but happy. It was December 17, 1903.

Ever since that icy-cold morning near Kitty Hawk, North Carolina, to the present day, all aircraft has demanded direct current. It was the ignition system that first made the demand—it still does. But other systems have come to the fore that have overshadowed its initial importance to powered flight. There will be others. But for the time being it is still with us, whether we are talking about engines that operate on the Otto-cycle principle, or one that is based on the principle of jet propulsion. The Otto-cycle, in case you have forgotten, is the working stroke of an engine.

The real beginning, however, of diverse electrical power systems in the airplane came with the Liberty engine. On the earlier Libertys the magneto used was found inadequate. The manufacturers were forced to do some quick thinking, and they came up with a battery-operated ignition system. This battery was a four-cell, lead-acid, 8-volt storage battery; an engine-driven generator kept it charged.

Later a 12-volt battery installation was selected, because at the time some automobiles used a 12-volt system and a certain amount of electrical equipment was readily available. This system proved satisfactory until larger airplanes, with heavier electrical loads, gave critical priority to further development in electrical systems.

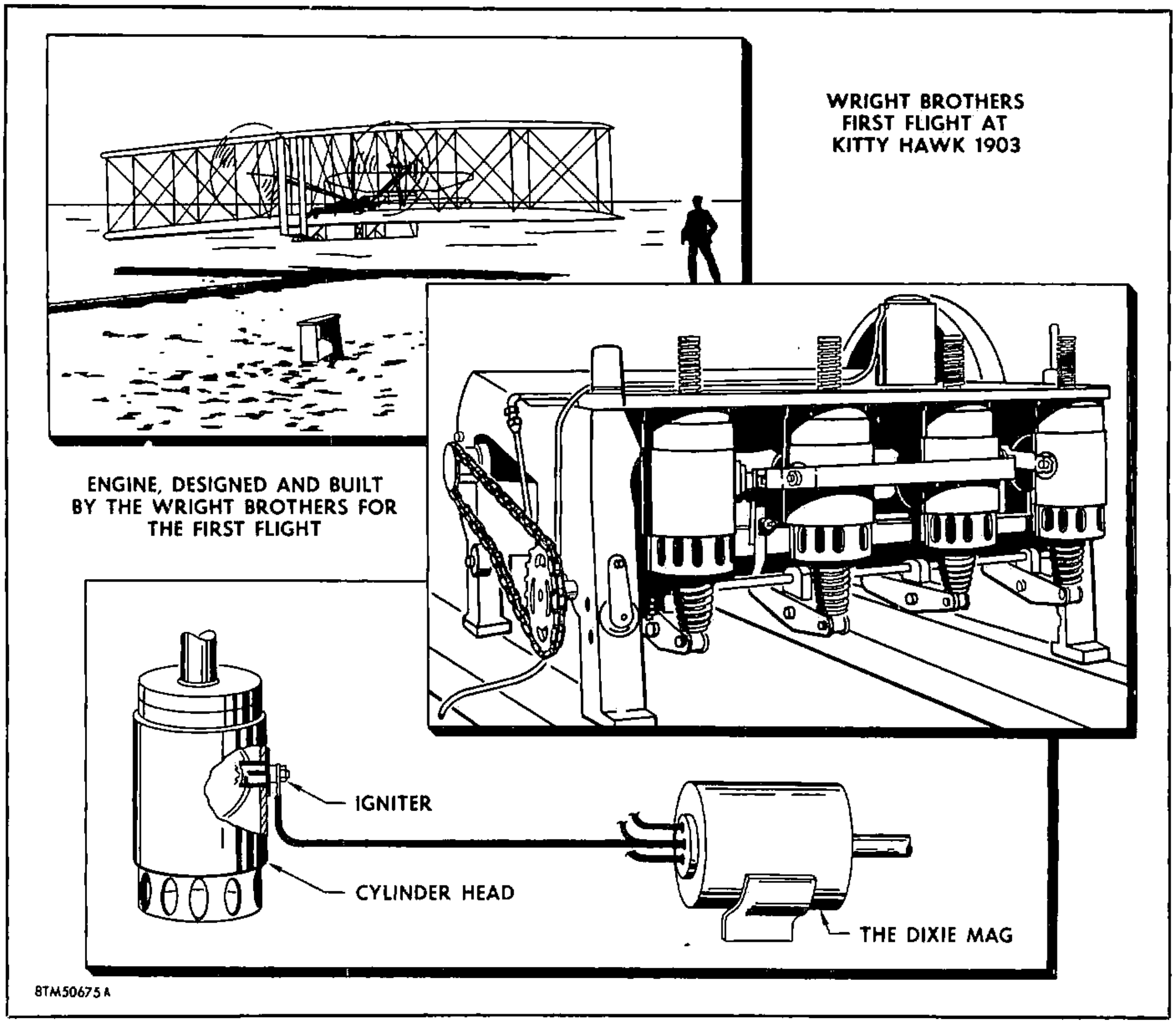

Figure 3-1. First Airborne D-C Electricity

The 24-volt system was first used in Europe. In 1938, the United States Air Force made this voltage standard. This seemed adequate until new designs in aircraft made further demands on engineering knowledge. Then with the development of high-output generators, the whole concept of power supplies in airplanes changed—almost over-night. Let's take a look at some of the things we have been discussing.

BASIC POWER SUPPLIES.

As you know, the basic power supply system is that portion of the electrical system lying between the source of power and the bus, or distribution point. We mentioned in Chapter II that the first airplanes that were equipped with electric starters installed the battery as close to the engine as possible, to reduce the amount of heavy wiring and to simplify maintenance. A master switch was not used.

Figure 3-2 shows an example of this circuit. As you can see, when the starter switch was closed the solenoid (located in series between the battery and the starter), became energized, completing the circuit to the starter. When the engine turned over, the starter switch was opened and the generator switch closed. This permitted the flow of current from the generator to the battery through the distribution bus. This system worked satisfactorily until the ever-increasing demand for more power threw heavier and heavier loads on the central distribution point.

When a master switch and ground power were incorporated into the system, power supply circuits became slightly more complicated, as shown in figure 3-3.

In this system an independent generator feed and ground power connector were used. The master switches

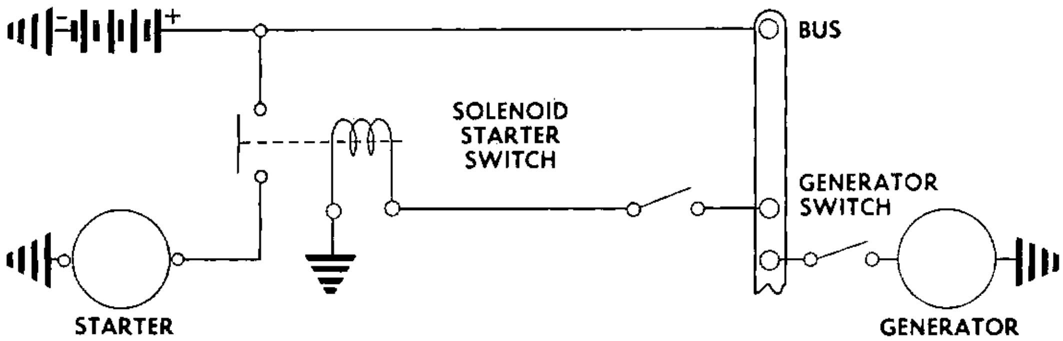

Figure 3-2. Simple Power Supply Circuit

were placed on the battery side of the main distribution panel. They were arranged so that the pilot could open the circuit in case of a crash, thereby eliminating a fear factor, the amount of "live-wire." On the ground, the mechanic could switch from battery to an outside source of power for starting or other ground operations. The master switches were also installed to provide the pilot with a means for battery disconnect in case the battery failed or in case of a short circuit.

This system worked very well, but it was far from perfect. For example, if both generators in a twin-engine airplane were used to charge the battery at the same time, one of the generators would occasionally reverse the field windings of the other generator, putting it out of commission; that is, until someone thought to "flash" its field. In those days this simply meant that a jumper wire was held firmly to the field terminal, generally "A" of the generator, while the other end of the wire was "sparked" off the positive terminal of a storage battery.

This, as you know, is an out-moded method of generator field flashing and is frowned upon in good-practice circles. Air Force regulations prohibit such haphazard but effective "flashing." We will discuss "field flashing," as it applies to the F-102A, later on in this chapter. This problem of field reversal was partially met by introducing switches, which enabled the pilot to exchange batteries with either one or the other of the two generators.

Also, he could operate his radio from only one battery, while the other furnished current for his lights and

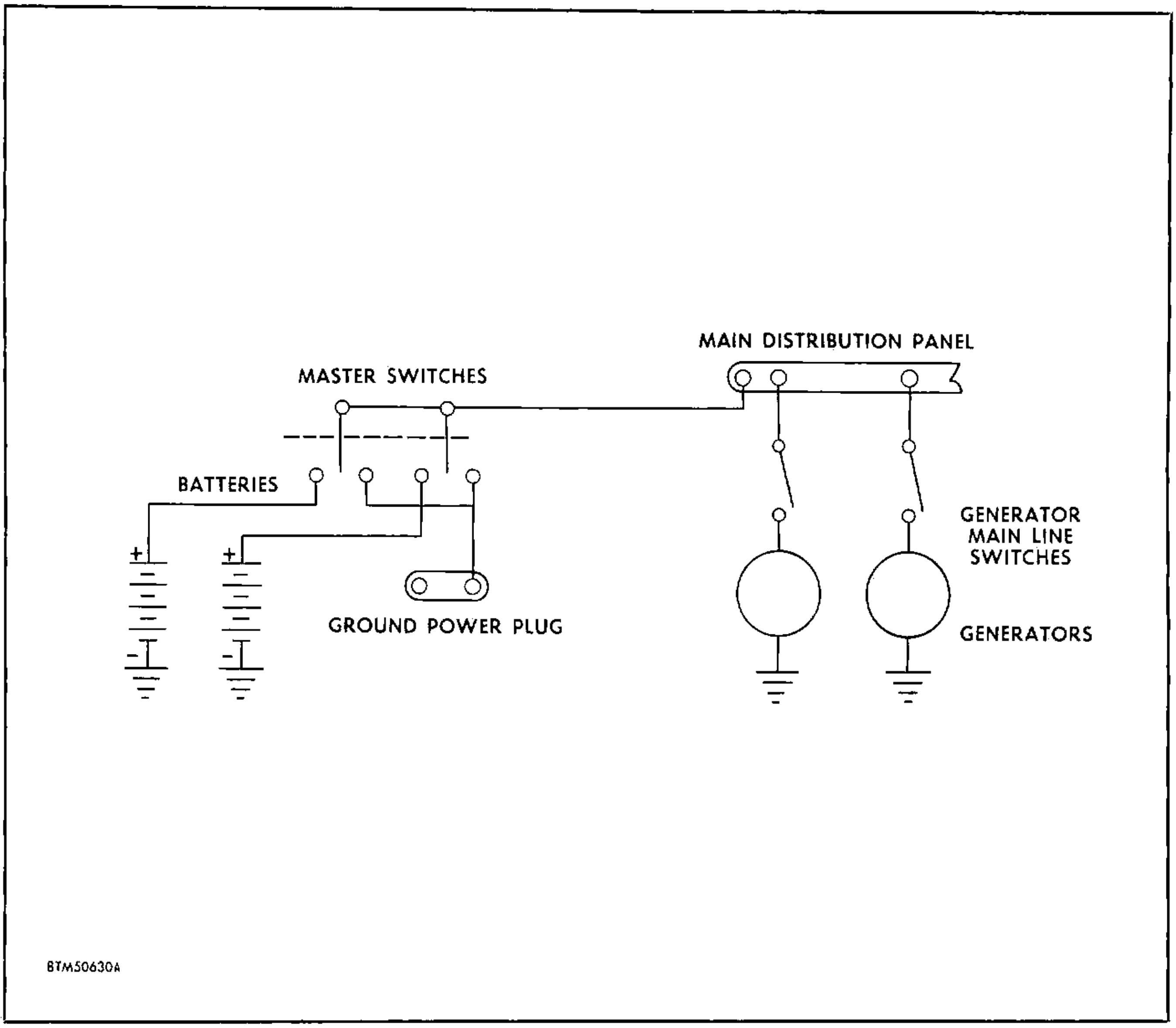

Figure 3-3. Power Supply Circuit With Master Switch

instruments. This procedure eliminated the annoying fluctuations in voltage which caused lamps to flicker and instruments to give inaccurate readings when the radio sending equipment was in operation. Figure 3-4 is typical of this system.

With the advent of high-output generators and multi-engined aircraft, it was necessary to design a system with control units to balance, or equalize, the load between generators. When two or more generators are operated at the same time to supply power for a load, they are operated in parallel. This simply means that each generator supplies a proportional part of the ampere-load. If an operation of this magnitude is to function successfully, each generator must share the load on an equal footing. This was achieved through a twin-engine equalizing circuit, whose purpose is to automatically help the two voltage regulators in lowering the voltage in the high generator, and in raising the voltage of the low generator. (See figure 3-5.)

MAINTENANCE AND RELIABILITY OF ELECTRICAL EQUIPMENT.

A reliable person is generally thought of as a trustworthy person. Sometimes, however, during the stress and strain of everyday living, there comes a time, or times, when even this person may succumb to a pattern of behavior not in keeping with his more dependable characteristics. In the same manner, electrical equipment under shock, certain "G" and atmospheric conditions, or inefficient and slipshod maintenance, will set a pattern of behavior not intended by its manufacturer.

Before installation, the reliability of electrical equipment is tested and retested under all conceivable conditions.

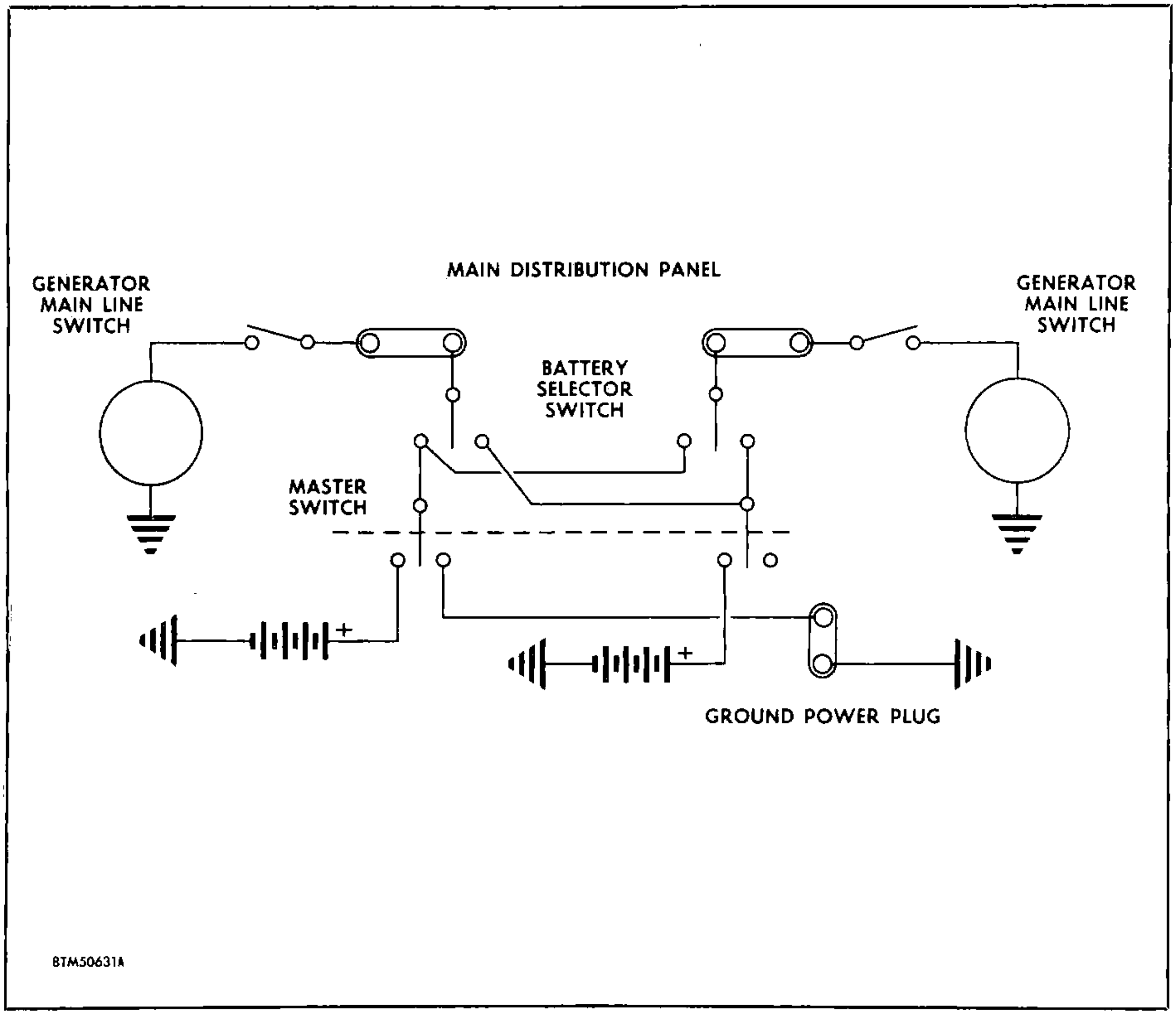

Figure 3-4. Twin Power Supply Circuit

And because facts, not fiction, are desired, simulated performance areas are set up in laboratories to test this equipment under rigid specification requirements. Devices are tested for altitude performance, as high as 75,000 feet; for arctic, desert, and tropical operations, and just plain rugged dependability. Laboratory testing today is useful and it certainly has its place, but at best it is only comparative. Unfortunately, there is the wear factor and certain other conditions which cannot be foreseen. One of these is the reliability of the line maintenance man.

Electrical components, as they assume more and more of the vital functions within the electrical system of a modern airplane, demand efficient maintenance. There are few electrical parts that do not require periodic inspection and maintenance. Once a device has proven its reliability through actual flight testing (and there is no substitute for this), it is the flight-line maintenance man who must be relied on to keep it in proper working order. But even with all this, there is questionable reliability even in dependable electrical equipment that is not properly maintained. On the other hand, with unreliable equipment to maintain, you might as well pick up a magazine and forget the whole thing—you're not a magician.

Speaking of printed matter, did you know that manufacturers of airplanes keep a digest on unfavorable field reports? Well, they do. They call these "Safety Reliability Digests." And these are the facts—not the fiction—of flight performance. For example, during a takeoff run, the pilot of a twin-engine jet fighter received a false fire warning light. Because these fire warning lights were so close together, the pilot was unable to distinguish which engine was afire. He elected

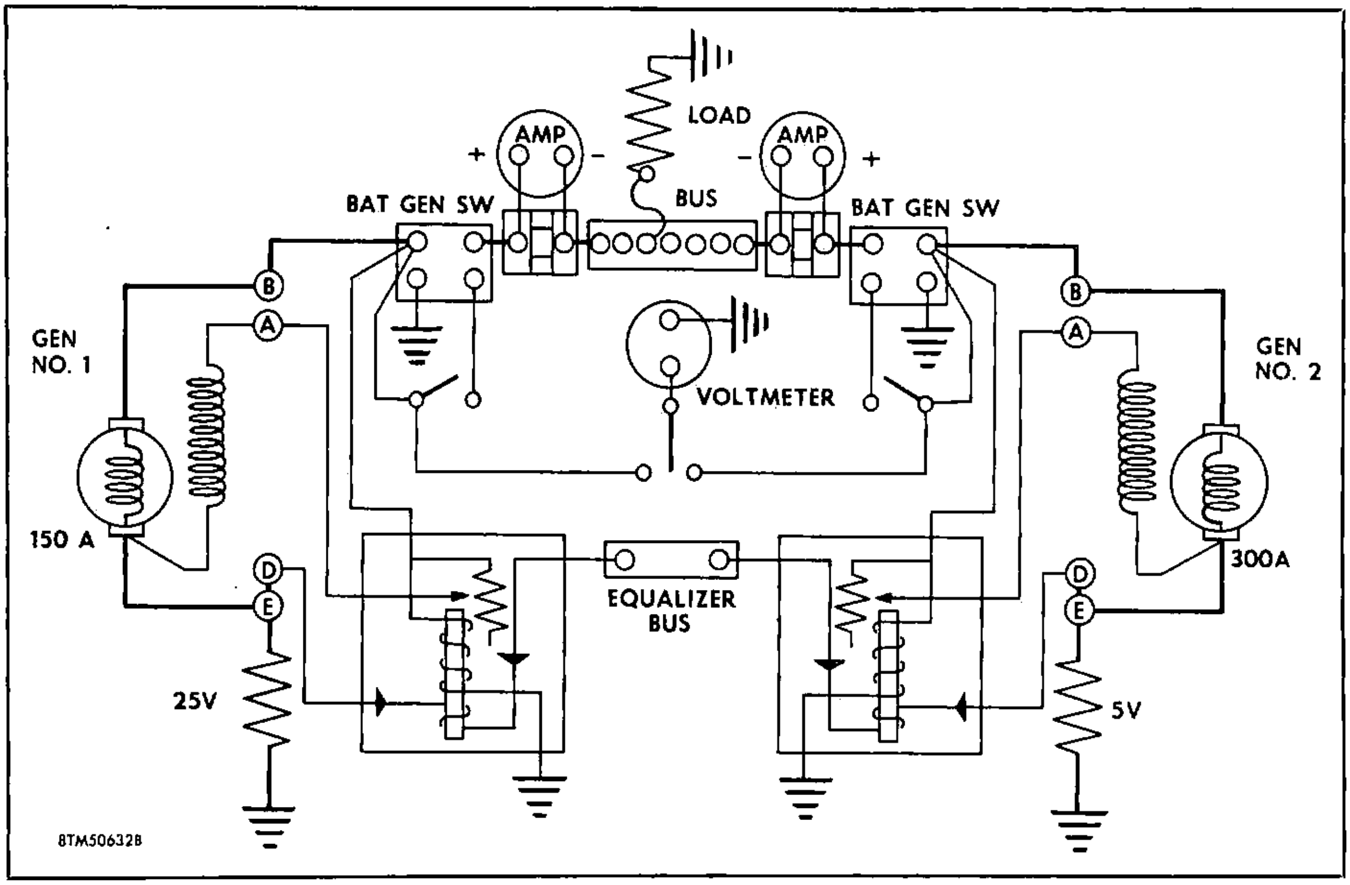

Figure 3-5. Equalizer Power Supply Circuit

to abort the mission. Two dangerous conditions were established—one, the warning lights were so close together that a quick, accurate identification was impossible; and two, this *false fire warning was traced to a corroded connector,* part of the fire warning system.

A lot of thought is given to the design of panels and instrument placement in the cockpit of an airplane, and in this particular airplane it was thought that the placement was ideal. This case history proved otherwise. This particular condition had never come up before, nor has it since. But the core of the trouble, the corroded connector, is a different matter. This was maintenance oversight, and there is no place for inefficient or ineffective maintenance in aircraft. Men's lives are at stake. The best maintenance, therefore, is the preventive and continuous type.

ACCESS TO ELECTRICAL EQUIPMENT.

Time is a precious commodity, especially in airplanes designed for a continuous alert status. For this reason electrical design engineers make special provision for easy access to electrical equipment. You may get a stiff neck or an aching back working on them, but they are accessible. For example, connectors, or plugs, are so placed that they can be inspected at regular intervals and any corrosion removed. Quick disconnects and snap-removal features are incorporated into equipment. The man-hour consumption to make maintenance easier for you is a staggering figure. A very good example of this is in the F-102A. If you will think about it a minute, after you have stuck your head into the various compartments and wheel wells, you will realize how many design-hours are expended to make your job easier for you; not to mention the barked elbows, skinned knuckles and the frayed tempers.

A typical example of the planning that makes maintenance easier is the example of the cockpit electrical components. In figure 3-6, you can see that the instrument panel can be dropped forward for access to the wiring in the front of the cockpit, and the switch panels on the consoles can easily be removed for inspecting and maintaining the switch and control circuits concealed beneath the consoles.

ENVIRONMENT AND ITS EFFECTS ON ELECTRICAL EQUIPMENT.

Our friend Mr. Webster defines environment as the sum total of all the external conditions and influences affecting the life and development of a human being or a complex thing.

Whether we are at sea level or in the higher altitudes within one of the great blankets of atmosphere

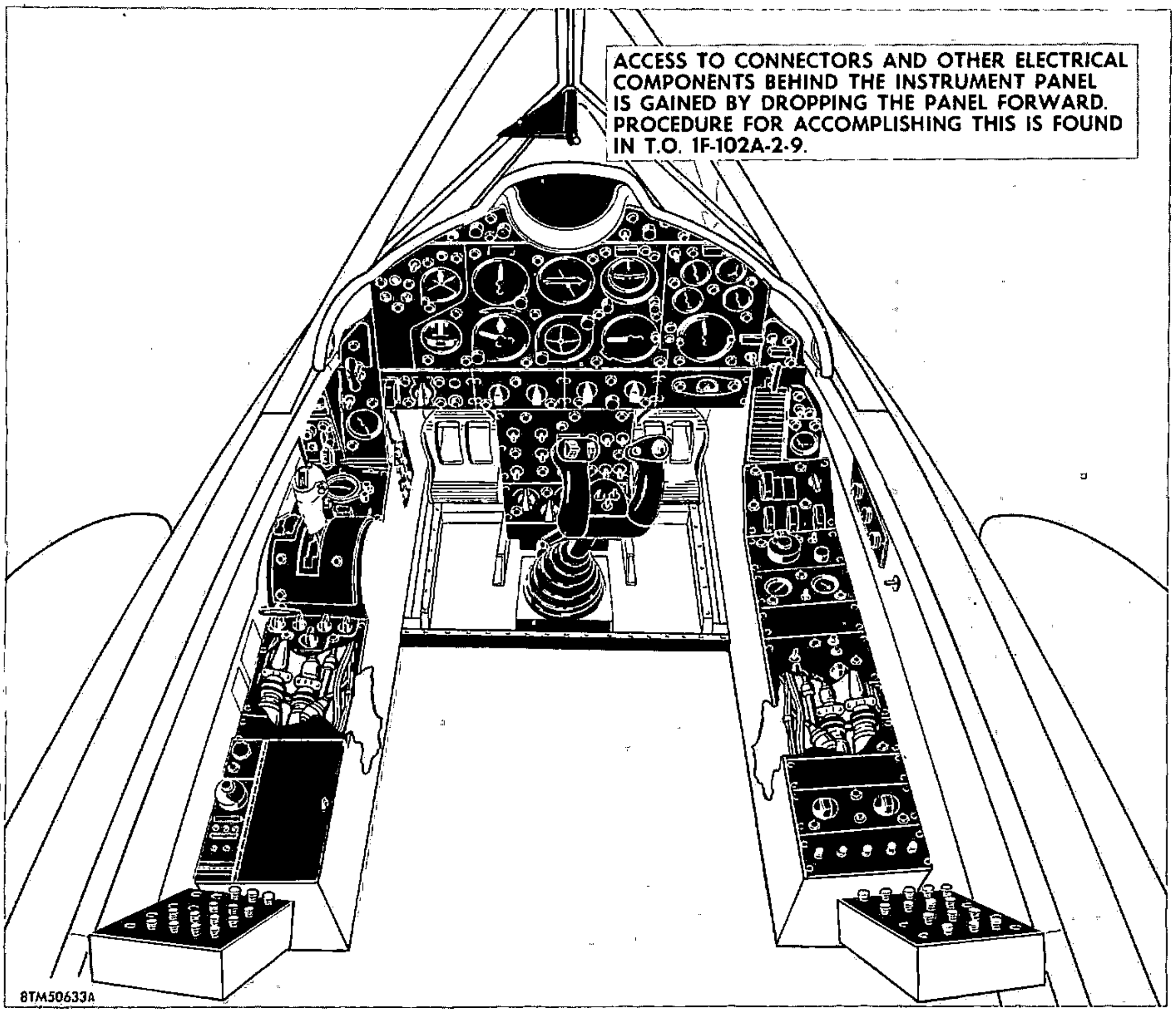

Figure 3-6. Electrical Equipment Location and Access

surrounding the earth, atmospheric conditions (environments) vary from day to day. You breathe the atmosphere around you. It may be hot, warm, brisk, cold, windy, or frigid. And how you breathe it, where you breathe it, and when you breathe it, affects your physical balance. It might make you feel good, bad, or indifferent—it might even kill you. In the same manner, electrical components react to atmospheric changes (environment). They too have their good moments and their bad.

Figure 3-7 shows the environments in which the F-102A —an all-weather interceptor—must be ready to operate at a moment's notice. You must know something about them, especially their effects on electrical systems.

Natural environment presents problems of worldwide extremes in temperature; it presents additional problems in humidity, fungus, rain, snow, solar radiation, lightning, wind, sand, and dust. It, also encompasses pressures at the earth's surface and those found in the upper blankets of atmosphere.

Induced environments are the most difficult to guard against, because they vary with atmospheric conditions and the performance of the aircraft within those conditions. Induced environment is characterized by high temperatures, temperature shock, vibration, acceleration, explosive vapors, and nuclear radiation. Temperature shock, for example, is induced by rapid acceleration to very high speeds, rapid changes in altitude, or a combination of both.

ALTITUDE AND THE ELECTRICAL SYSTEM.

We are entering a scientific era, an engineering era, when the upper atmosphere and its properties are as

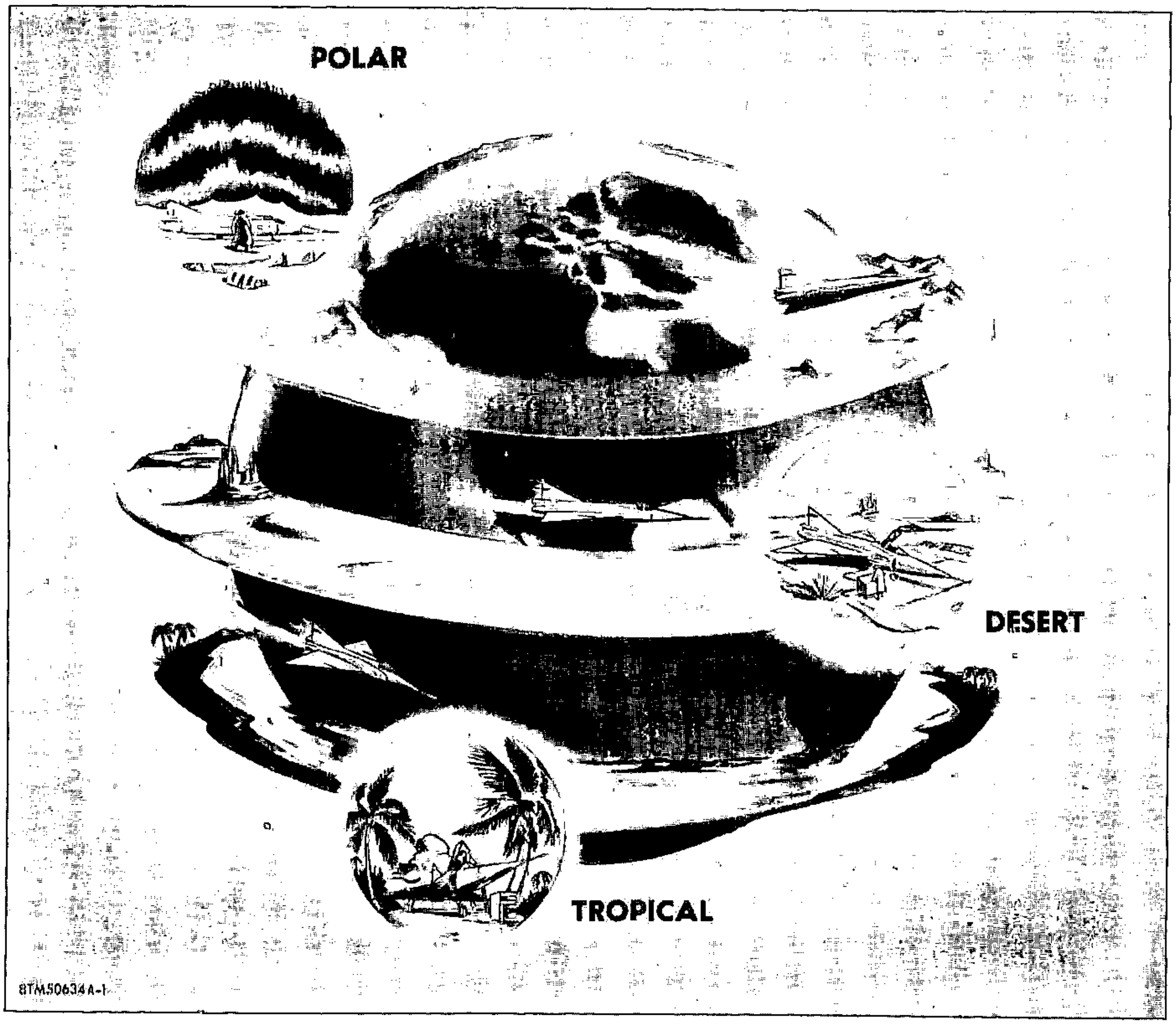

Figure 3-7. Environment and Its Effect on Electrical Equipment

important to your knowledge of aircraft maintenance as your knowledge of the atom and its temporary visitors, the electrons, or why an airplane flies. The expression "high, wide, and handsome" can certainly be applied to Air Force operations today. In the future they will be higher, wider, and more handsome—and you'll be there.

Just as an electric blanket has different and varying temperatures, the great blankets of atmosphere which lie above us are varying and different in their temperatures, as can be seen on the temperature chart in figure 3-8. You will also see that there are four atmospheric blankets, one above the other—in layers: the ionosphere, the troposphere, the stratosphere, and the mesosphere.

Each of these conditions presents a different problem to flight; but for our purposes our greatest interest is in the troposphere.

"Why?" you might ask. "How about all this talk about man-made satellites, and rocket-powered aircraft flying at 90,000 feet. How about interplanetary space travel?"

How about it? Well, it doesn't do any harm to dream and plan, does it? And it is true, Major Arthur Murray did fly a rocket-powered airplane in the neighborhood of 90,000 feet, as far back as the summer of 1954.

But the fact remains that nearly all present-day military and commercial flying is still within the troposphere; our first blanket of atmosphere, which extends from sea level to fluctuating altitudes of between 35,332 feet and 40,000 feet.

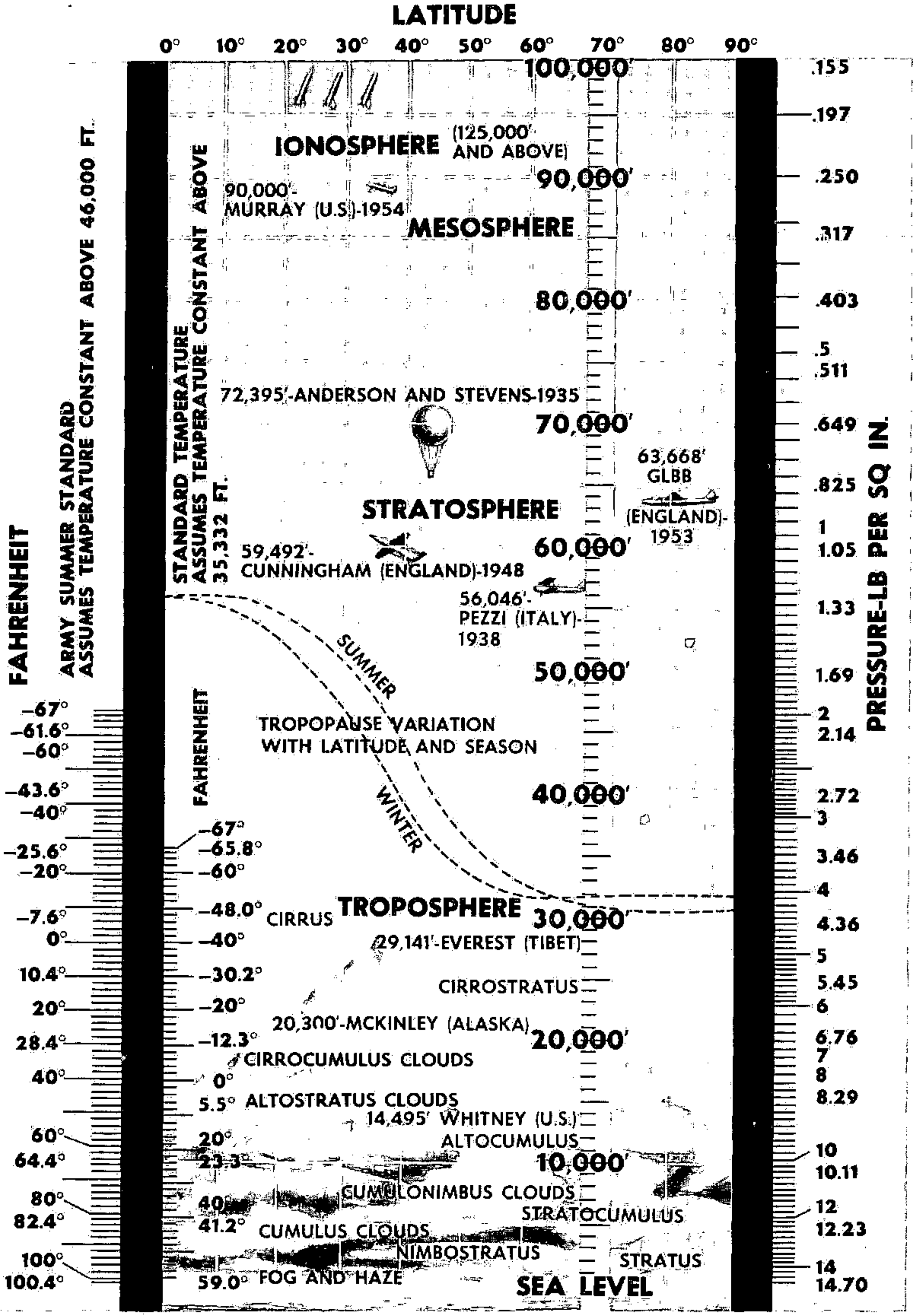

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Figure 3-8. The Upper Atmosphere

These figures, however, are more or less subject to change. The tropo*pause,* which is the upper portion of the troposphere, is where you find it. The upper extremity of the tropopause has been recorded at 23,000 feet. In other words, it varies with the time of year, the time of day, and the weather conditions existing on the earth's surface—and that part of the world where it is measured.

For example, at Point Barrow its extent might be 30,000 feet, while in Southern California it could reach beyond the 40,000-foot level we spoke of—the average mean extremity. For our discussion let's think of it as extending from sea level to the 40,000-foot level.

In the troposphere there is a gradual and nearly constant reduction in temperature, from 59°F at sea level to −67°F at about 47,000 feet, which could well be the extent of the troposphere at certain seasons of the year. Because the warm air at the earth's surface lies below this cold air, a great deal of movement takes place in the form of wind. Most of our water vapor and clouds make the troposphere their concentration center. Above the troposphere lies the stratosphere, which stretches from about 40,000 feet to about 70,000 to 100,000 feet where the temperature remains practically constant. We shall get to this blanket of atmosphere later.

Those Four Kinds of Altitude.

While we are on the subject of altitude, it might be well to define some terms you will undoubtedly hear while serving in the Air Force. Altitude is generally thought of as a measurement—and so it is; but it should also be readily understood as the relation of an airplane or rocket to the earth's surface. Scientifically, in aerodynamics, the term *altitude* is important only as a means of describing the characteristics of the air through which an airplane must fly.

As you have learned, the atmosphere around and above us is not the same from day to day. For this reason, certain standards have been established which are based on average atmospheric conditions in the United States. As closely as possible, they approximate true atmospheric conditions as they may be found at sea level, in the troposphere, and on into the stratosphere.

Each altitude is assigned a complete set of characteristics; these characteristics are pressure, temperature, and density. These altitudes are known as NACA (National Advisory Committee for Aeronautics) standard atmospheres.

These standards have also led to some new definitions of altitude: First of all there is *tapeline* altitude. This is the actual height in feet above sea level. Next, we have *density* altitude, which is the height in feet above the earth at which a given air density is found. *Pressure* altitude and *temperature* altitude is the height in feet above the earth at which a given atmosphere pressure or atmosphere temperature is found.

For example, a pilot could be flying at a tapeline altitude of 40,000 feet, but his density altitude might be only 37,000 feet, his pressure altitude 36,943 feet; and his temperature altitude could be but 32,000 feet. They are all interrelated and are defined in mathematical equations which are out of the scope of this manual. These altitudes have been basically defined so that you will know what someone is talking about in regard to altitudes. It isn't necessary that you know how they are arrived at, simply what is meant by them.

Airborne Electrical Equipment.

The operating temperature of the electrical equipment will increase approximatey 1% over that occurring at sea level for each 330 feet of tapeline altitude in excess of 3000 feet. Altitude shortens the life of electrical equipment.

In an earlier portion of this chapter we discussed the testing and retesting of equipment for its electrical reliability under various environmental conditions, and for its rugged operational characteristics. In Chapter II we discussed the effects of high altitude on generator brushes. You learned that, for awhile at least, the situation was so critical that the future of high-altitude flying and precision bombing was extremely doubtful. The reasons for this excess wear—and in some cases complete disintegration—were a lack of moisture, lack of oxygen and general rarity of the atmosphere at altitudes of 25,000 feet and above.

Air, as you know, is a gas or a combination of gases. Ordinarily gases are poor conductors because they contain few *ions*. We discussed *ions* in Chapter I, when we reviewed the storage battery. We said then that an atom which has lost one of its electrons is called a *positive ion,* and one that has gained an electron is called a *negative ion;* and when these ions build up we have ionization.

Another way of putting it, an ion is an atom or group of atoms that have either an excessive positive or negative charge. This, as you know, is what takes place in the electrolyte solution in a battery.

Ionization of gas is produced by either of two methods: by collision impact, or by the absorption of radiation. The first method is simply when electrons are forcibly detached from ions and molecules by the impact or bombardment of other ions or molecules. The second method is best described by the phenomenon which takes place some 25 miles above the surface of the earth (132,000 feet) in the ionosphere. This is a region of electrically charged air—ionized air—which is charged by radiation from the ultraviolet rays from the sun. But it will be a few years before we will be flying in this blanket of atmosphere. So, for the time being, let's get back to earth.

The insulating effect of air varies as we take off and climb to higher altitudes. At high altitudes it is found that insulation breaks down and flash-overs, or arcing, is more likely to occur because the air is more readily ionized. This is due to poor conductance qualities found at these altitudes, allowing corona discharges to take place.

These discharges are nothing more than a buildup of two electrical potentials until a crackling spark passes between them—a miniature flash of lightning. The corona may be in the form of a bluish glow and at sufficient altitude appear on the surface of a conductor, or (without static discharge) at lower altitudes a similar phenomenon may appear in the form of static electricity and roll off the wing of a plane in what pilots call "Saint Elmo's Fire."

It is the ozone and acids, generated by coronas, which rapidly corrode metal and attack insulation at high altitudes. The ruggedness of electrical systems is further challenged by the tremendous increase in moisture content as the airplane descends to lower altitudes, plus the rapid change of atmospheric pressures and temperatures. Thanks to our electrical "know-how" these severe atmospheric changes have been fairly well met; by using hermetically sealed panels, sealed electrical contacts, potted components, pressurized compartments, cooling systems; and as in the case of our generator brushes, a chemical impregnating treatment with halide.

COLD WEATHER AND THE ELECTRICAL SYSTEM.

The group of photos in figure 3-9 has not been reproduced from actual conditions and inserted in this manual to sell you on a pleasant vacation spot, nor is it placed here to recruit you for duty in the far north; rather it is to show you just how critical the maintenance must be under these conditions.

Cold temperatures make insulation brittle—it tends to weaken it. Batteries freeze and lose their charge unless properly maintained. But the most serious effect cold-weather operation has on aircraft is that it reduces your own efficiency. This is recognized, and it certainly is not held against you—after all you are human. But it is a critical problem, and as long as you recognize the fact that it does lower your efficiency, concentrate that much harder on the particular task ahead of you and the problem will not be as serious as it is cracked up to be. In other words, you will be a brass monkey out there in that bitter cold but don't think like one.

For the maintenance of the aircraft storage battery and other components in the aircraft during cold weather and polar operations, you are referred to T.O. 8D2-1-31 and T.O. No. 00-60B-1.

THE DESERT AND THE ELECTRICAL SYSTEM.

The desert is designated as a dust-laden atmosphere, and is harmful to electrical equipment. As a rule, desert operations do not shorten the life of electrical equipment if it is protected from sand and dust storms. Severe damage can be caused, however, by dust or sand which is carried in large amounts in the atmosphere; the presence of these particles between moving surfaces is almost sure to cause failure.

In some locations atmospheric dust peculiar to that locality causes corrosion or the breakdown of insulation. Coral dust is a conducting material and may cause arcing of contact points. Volcanic dust, in addition to its abrasive qualities, also causes corrosion, particularly when combined with moisture.

THE TROPICS AND THE ELECTRICAL SYSTEM.

In the tropics it is the fungi and moisture—the high humidity—which attack electrical systems. Perhaps the best way to present these damaging effects is in chart form. We spoke of moisture when presenting high-altitude operations and its effects on a system; the following chart emphasizes what sustained moisture coupled with fungi can do to a system without adequate maintenance and protection.

THE EFFECTS OF MOISTURE AND FUNGI ON MATERIALS

The Material	*The Effects*
Metals.	Moisture causes corrosion. Loss of sensitivity of delicate mechanisms, unbalancing of electrical circuits and arcing. Fungus settles on organic dust and etches, or eats metal, destroying vital operating surfaces.
Wax impregnation.	Non-fungus-inhibiting waxes support fungi, causing destruction of insulating and protective qualities, and permitting the entrance of moisture, which destroys parts and unbalances electrical circuits.
Glass: Instrument lenses, component inspection, windows.	Fungus settles on organic dust. Almost immediately it etches the surfaces, destroying optical properties and visibility.
Fiber: Washers, cable supports.	Moisture causes swelling, resulting in binding of support parts. Fiber is attacked and destroyed by fungi.
Laminated plastic: Electrical system terminal strips, mounting panels.	Insulating properties are lost, leakage paths cause flashover. The bond between laminations is destroyed.
Cotton, linen, paper, cellulose derivatives: Insulation, covering, webbing, belting, laminations, and dielectrics.	Insulating and dielectric properties are lost or impaired, causing arcing and flashovers. These materials are rotted by moisture, and completely destroyed by fungi.

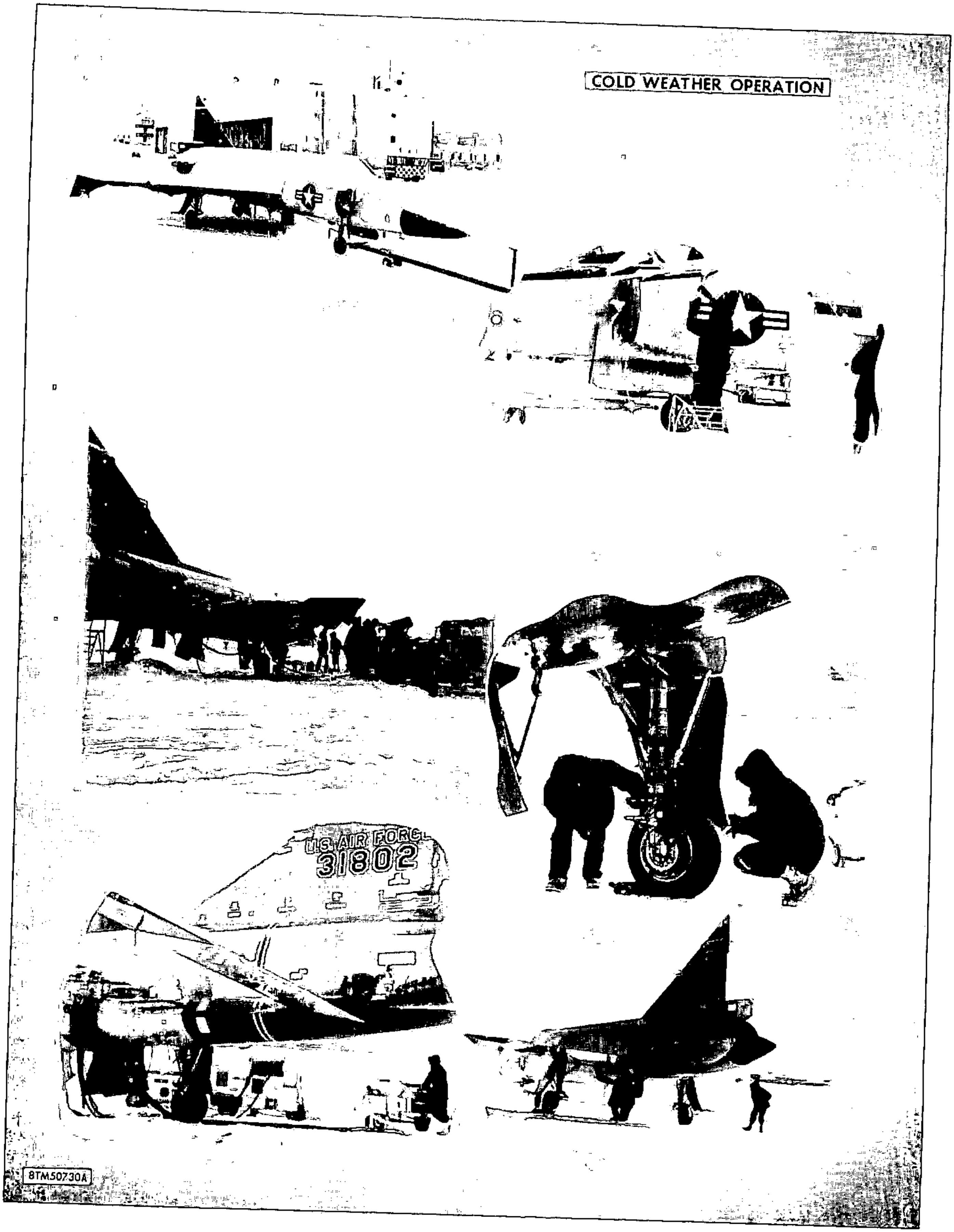


Figure 3-9. Equipment Maintenance—a Critical Problem

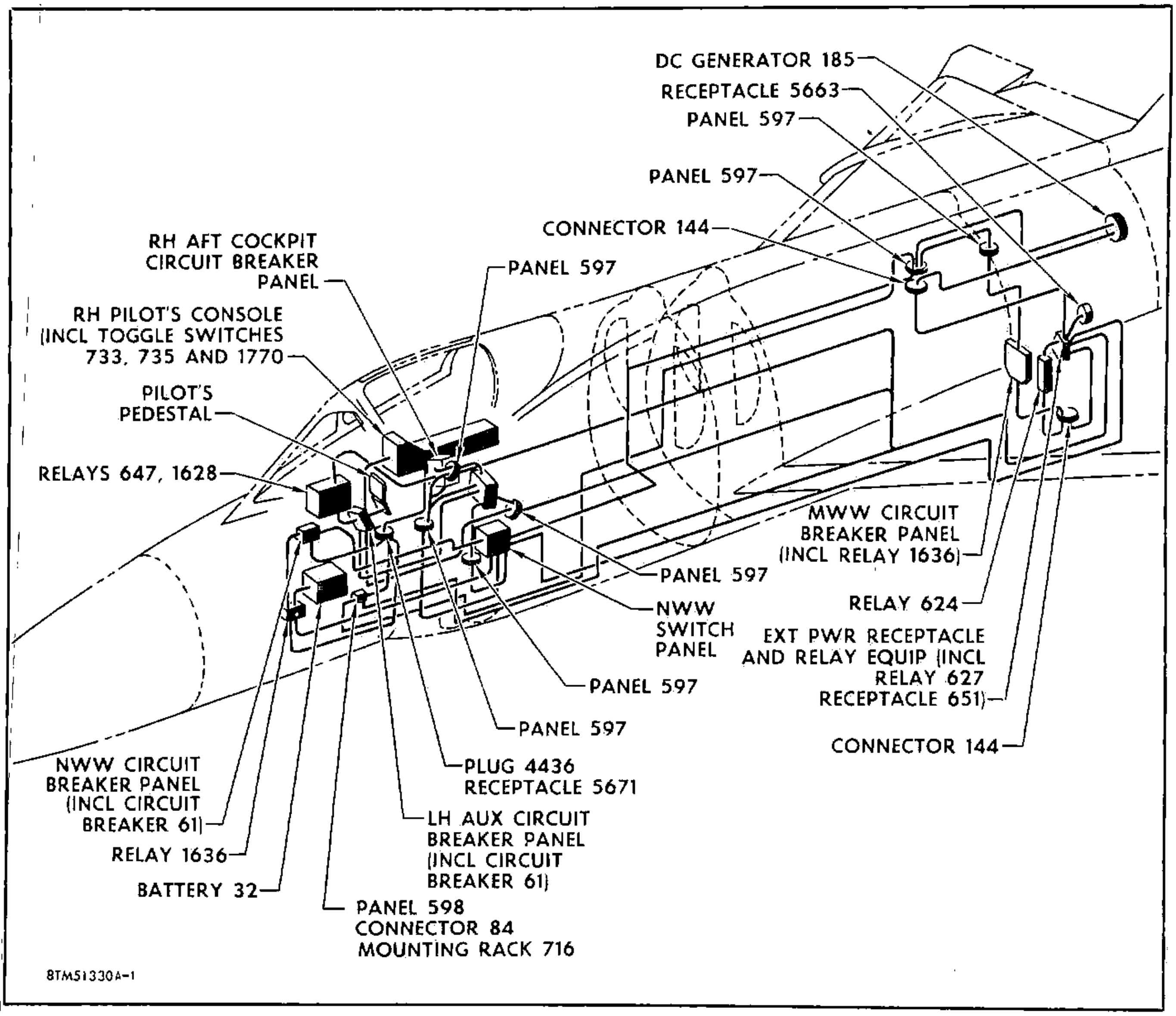


Figure 3-10. Location of Electrical Units and Associated Wiring (Sheet 1 of 2)

The knowledge of high-altitude and climatic effects on electrical equipment is most important to you as a maintenance mechanic and you will hear, from time to time, a great deal more about this subject. The discussion just presented has been nothing more than a thumbnail approach to familiarize you with this aspect of your electrical knowledge to help you to understand the problems under different climatic conditions.

WIRING.

The wiring in an airplane comprises a large percentage of the total weight of the electrical system. As previously stated, the F-102A has electrical wiring sufficient to wire 15 average modern homes, and this does not include the electronic network of the aircraft.

Figure 3-10 shows the location of the major components of both the d-c and a-c electrical systems. Although we are discussing only the d-c system in this chapter, both systems are shown to give you some idea of the vast amount of wiring required to connect all the components of the electrical systems. (In Chapter IV, you will learn about the a-c system and its components.)

Keep in mind that the lines shown here represent all the wires and cables used to connect the components, and do not actually show the exact routing of wiring you will find in the F-102A. Also remember that these illustrations show only the power system, and do not take into account all the wiring used to connect power to the separate systems such as armament, communications, and flight controls.

The primary function of any wiring system, which should be quite clear at this point in our discussion, is to provide a conducting path from the source of power to the units involved and from the units back to the

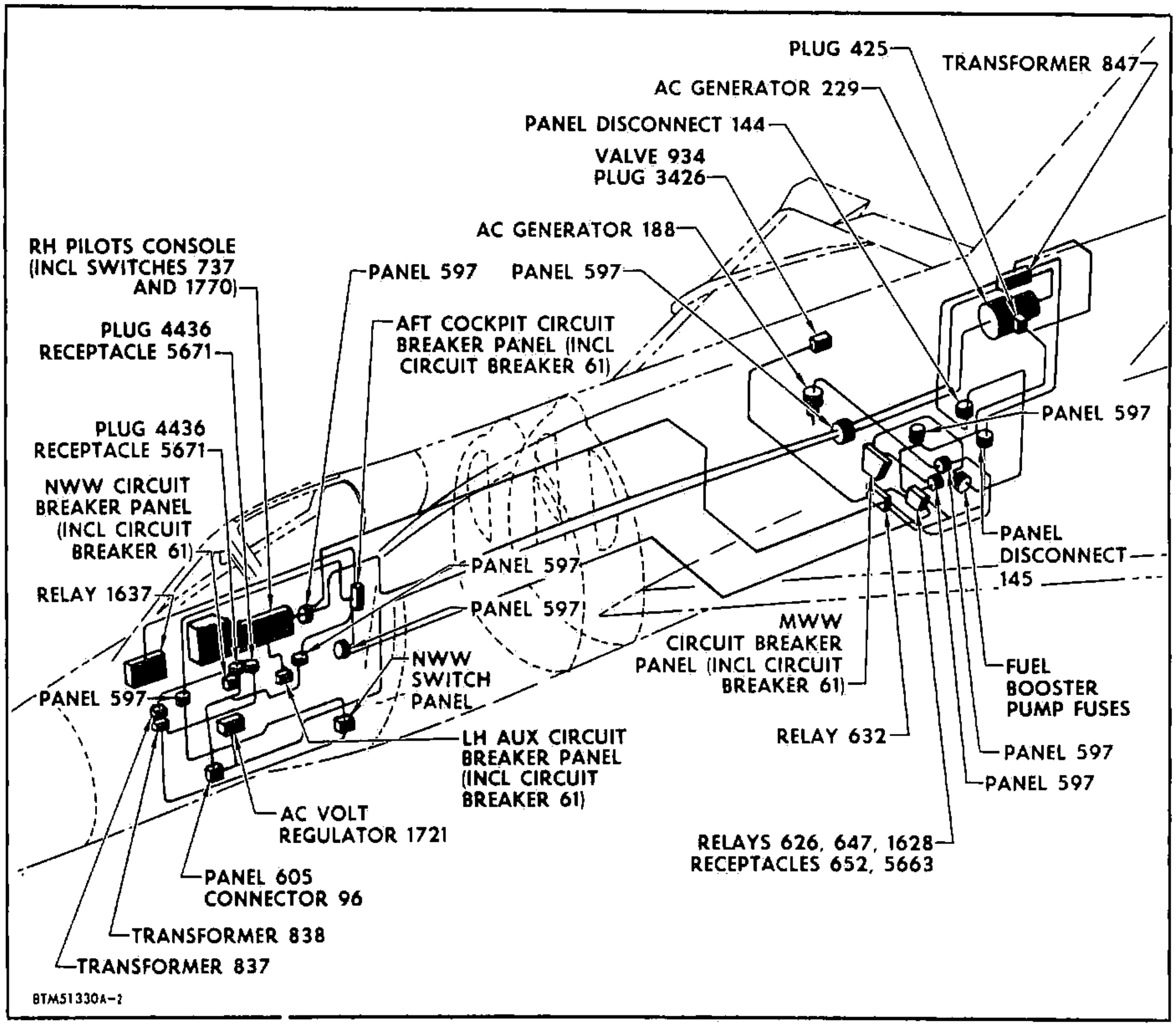


Figure 3-10. Location of Electrical Units and Associated Wiring (Sheet 2 of 2)

source. In addition, it should carry the current without excessive heating or voltage drop. For this reason, there are rigid specifications governing the wiring used in aircraft.

For example, the wire used must be strong and flexible; its insulation should be resistant to abrasion, normal wear and tear, and moisture, and it should be able to withstand high temperatures without losing its insulating qualities. All wiring must meet these standards, and many others, before it is installed in the airplane. In other words, aircraft electrical wiring installation practice is based on quality standards to insure safety, proper operation of equipment, and service life.

F-102A WIRE IDENTIFICATION.

Each wire used in an aircraft electrical system carries an identifying number. A code letter prefix indicates the general circuit in the system of which the wire is a part. Figure 3-11 shows a typical example of this method of coding as used in the F-102A. Its accompanying table gives the various letters used for system designation.

WIRE IDENTIFICATION TABLE

Letter	*System Designated*
A	Armament
C	Controls, Surface
D	Hydraulic and Pneumatic
E	Engine Instruments
F	Flight Instruments
G	Landing Gear
H	Heating, Ventilation, and De-Icing

WIRE IDENTIFICATION TABLE (Cont).

Letter	*System Designated*
J, K	Starting and Ignition
L	Lighting
M	Miscellaneous
P	D-C Power
Q	Fuel and Oil
W	Warning
X, V	A-C Power and Emergency Power

As we have said, the letter prefix indicates the system circuit; in this case, the letter P stands for d-c power. The number 122 calls out the actual number of the wire, and the next letter indicates the wire segment. There may be any number of segments in a circuit, but usually you will find five is an average number. In this particular case, "D," the fourth letter of the alphabet, means that this wire is the fourth segment from the power source. The last number, 16, indicates the size of the wire. If, in some cases, the letter "N" is found tacked on to this alphabetical-numerical coding, it simply designates "ground."

If an exception to the above method of coding is found in a bundle, it will be a part of the electronic network in the airplane. For example, one of the most delicate and critical wires found in the F-102A is one coded 3F799A, and black in color. Extreme caution should be used in handling this wire. The reason for this is the gauge of the seven twisted wires encased in this conductor, which, when measured individually with a micrometer, are only 0.004 inch. Actually this is a small coaxial cable. It is shielded and has a plastic cover over the shielding. The plastic covering will melt before it burns.

The importance of the wiring in an airplane and its maintenance cannot be overemphasized. In Chapter I, we stated that there is no perfect conductor and no perfect insulator, and this is a good rule to follow. But great strides have been made toward perfecting insulating materials used to cover the wiring and cables used in aircraft.

An example of this is the fibreglass insulation used where temperatures become exceptionally high—the area surrounding the engine between the engine shroud and the fuselage frame. And as you have already learned, high-altitude flight has made research into this field especially critical. This is also true of connectors, or plugs, as you will find when we discuss the comparatively new type "potted" plug used throughout the F-102A.

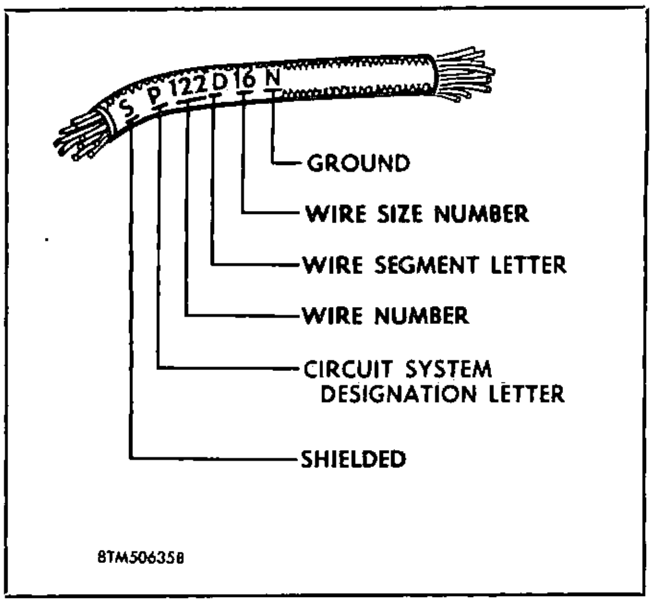


Figure 3-11. F-102A Wire Identification

As you have learned, coded identification is imprinted on most cabling in the airplane. There are some exceptions to this, such as coaxial and other shielded cables, which will not take printed identification—high-temperature wires are one example. In such cases a fiberglass pressure-sensitive tape is used with the specified coding. When vinyl, plastic, or another protective covering is used, the code numbers will be imprinted on protective covering. Harnesses for electronic systems, such as radar and armament systems, have their own identification system. Identification tapes used in these special cases are specially treated so that they will not dry out in service or produce chemical reaction with the particular cable insulation. Tapes which absorb moisture are never used.

Aircraft power cables are constructed of a large number of small, tinned copper wires. The reason for this is the need for cable flexibility. The wires are braided or twisted together and then covered with specially treated cambrics, nylons or any other suitable insulating material. Although smaller wires give more resistance than a single large one the same size as the cable, the cable-type is preferred for its flexibility.

AIRCRAFT CABLE SPECIFICATIONS.

Let's take a look at some aircraft cable specifications. In the following table you will find the Army-Navy, or as they are commony know, the "AN Specs" for aircraft cables. There are more than those shown here, but in all probability those listed will be all you are required to know, except in isolated cases or as new type cables are made available for specific purposes.

WIRE SPECIFICATION TABLE

Wire Size A.W.G.	Resistance of Cable (ohms per 1000 feet at 20°C.)	Rating (amperes)	Fuse Capacity (amperes)
AN-20	10.25	4.0	5
AN-18	6.44	7.0	15
AN-16	4.76	11.0	20
AN-14	2.99	16.0	30
AN-12	1.88	23.0	35
AN-10	1.10	33.0	50
AN-8	0.70	45.0	70
AN-6	0.436	65.0	100
AN-4	0.274	90.0	125
AN-2	0.179	130.0	175
AN-0	0.114	185.0	250
AN-00	0.090	220.0	300

You will notice that the size of the cable is designated in the American Wire Gage number (A.W.G.) and that only *even numbers* are used in this particular table. The reason for this is that in the Air Force only even numbered cables are used. The larger the number, the smaller the cable. In the above table AN-20 is the smallest cable and AN-00 is the largest. The ampere rating, or current rating, of the cable is the amount of continuous-duty current a cable will handle without appreciable heating. The fuse capacity of any cable is the maximum current the cable will carry without causing the strands to fuse together. This brings up a rather important point—selecting the correct cable.

CABLE SELECTION.

When a cable is chosen for a particular installation, the cable size selected is one having a current rating equal to, or higher than, the maximum current the cable must carry. For example, a certain size cable will carry a load compatible with its heating effect tolerance, but because of its *length* there is the danger of an excessive voltage drop; to overcome this a larger cable with a higher current rating would be used. While we are on this point, in a 28-volt d-c system the allowable voltage drop for a continuous-duty current is 1 volt. If it is intermittent-duty current, a 2-volt drop is permitted. Actually intermittent-duty current is nothing more than current which is used in a circuit that has intermittent operation within the system. We can define intermittent operation as full loading of a circuit for not more than three minutes out of 20 minutes; and to break it down even more, that within those three minutes there will not be any one period of continuous current flow that exceeds 45 seconds.

CABLE ROUTING.

The F-102A, being a modern tactical airplane, eliminates conduits almost entirely. The reason is to make your job easier, facilitating cable installation and practical maintenance. It also saves weight, and in actual combat operations makes the electrical system less vulnerable to enemy gunfire.

A word of caution—when it becomes your job to replace a cable, make sure you install it the way it was originally routed. There should be no shortcuts, short wires, an excess in slack, or an excess in tension in the replaced cables or individual wires. Perhaps the best way to think about a wire you are replacing is that it is a fine piece of linen thread, which is strong but not too strong. Don't treat it as you would the string on a guitar or banjo. We don't want a tune out of it—just performance.

The routing of cables—the bundles of wire—is carefully considered with certain objectives in mind. First of all, the conducting reliability is considered; then the voltage drop, and the job the particular cable is to do. Routed through here, will it reliably carry the current to its junction point?

The safety factor is taken into consideration—the safety of the airplane. For example, it would be extremely dangerous to channel wires where chafing and abrasion might occur, or where damage from battery acid fumes, fluids, exposure, or excessive heat would impair its reliability.

Figure 3-12 shows a typical example of cable routing in the nose wheel well of the F-102A. Sometimes it is necessary that cables be placed in critical areas, but when they are, they are carefully insulated and installed. That makes your job doubly important, to see that the same precautionary measures are taken in replacement.

It cannot be emphasized too emphatically that when you see cables routed in what you might think a roundabout way, this has been done for a purpose. Simply *put them back the way they were installed.* Below that bundle there may be a fuel, an oil, or highly combustible hydraulic fluid or oxygen line, which could cause serious consequences (even loss of life) if an electrical power line were placed too close to this plumbing. There is a prescribed distance for electrical wiring in these critical zones—adhere to them; and if in doubt, consult your Air Force directives on this subject.

Along the lines of the above discussion, there is another important consideration when routing cable—to prevent the malfunctioning of equipment by induction. Finally, every effort is made to provide accessibility for maintenance, for inspection and checkout, and for easy replacement of individual wires.

TERMINALS.

Terminals are devices installed at the ends of a conductor to make a firmer, easier, more dependable and safer connection between the source of power and the unit to be operated. There are two general categories

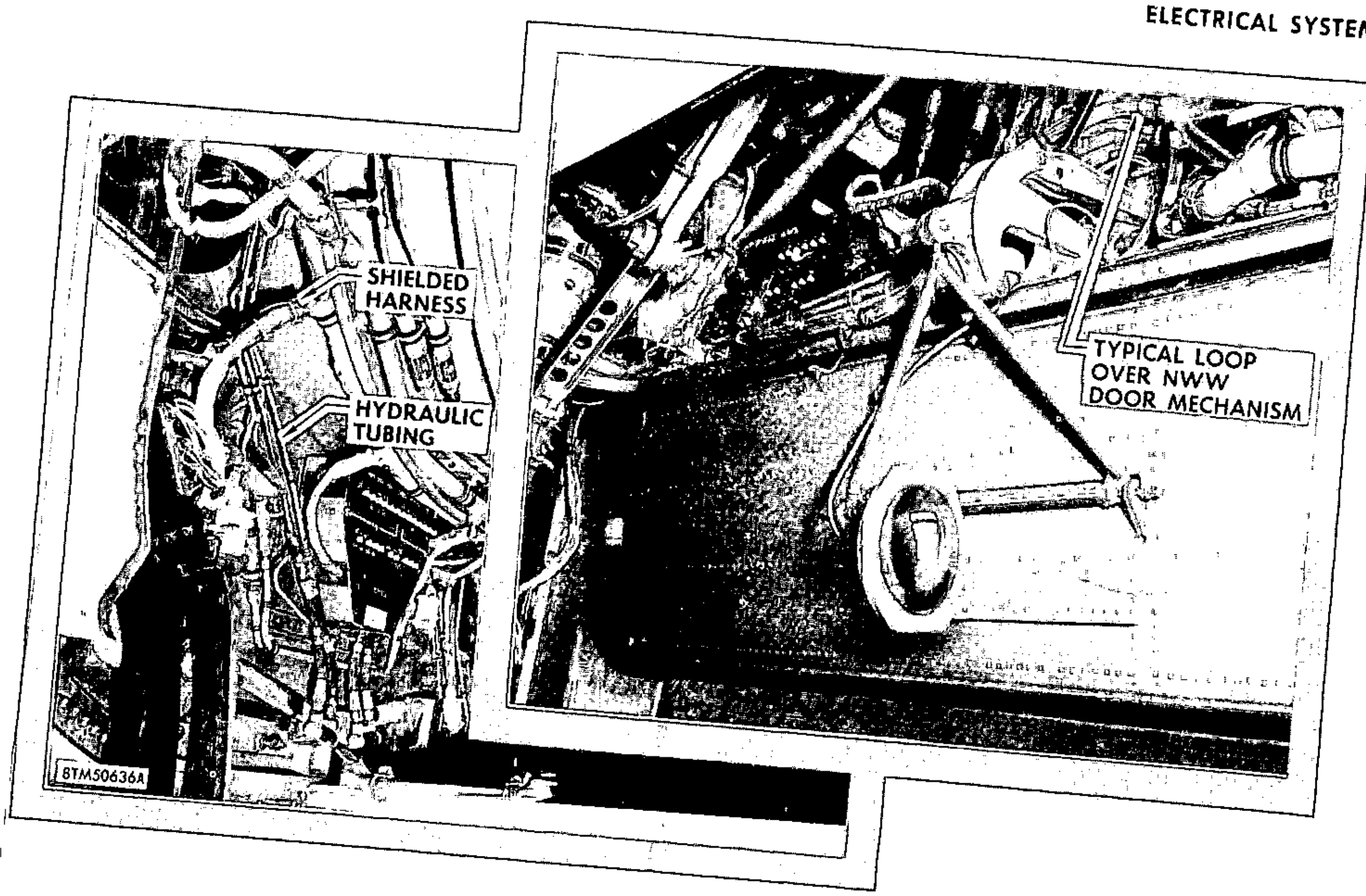


Figure 3-12. Typical Cable Routing

into which these terminals fall—the *soldered* and the *solderless*. The solderless terminal is the one most generally used and recommended for aircraft use.

Here again, Air Force directives should be consulted before bending or straightening any terminal. As a rule, terminals are never bent unless a very definite necessity exists and rarely if ever if a wire is larger than an AN-12. On wires smaller than this (AN-12 to AN-20), terminals are bent only under extreme conditions; to provide proper clearance, or to otherwise improve installation—but first consideration is always given to the proper routing of the wires.

CONNECTORS USED IN THE F-102A.

In electrical school you learned that connectors are devices attached to the ends of cables and sets of wires to make them easier to install or to remove. Ordinarily connectors consist of a coupling ring, the plug itself, the vinyl sleeving, a space adapter, a grommet, the tapered sleeve, and the plug adapter.

Figure 3-13 is a disassembled view of a conventional electrical plug. The "X's" show these portions of the plug eliminated through "potting" in the F-102A electrical system. There are however, a few conventional plugs still used. These will be found near the engine compartment where high temperatures make the use of the potted plug impractical.

THE POTTED PLUG.

The potted plug is comparatively new and is used chiefly on high-altitude aircraft. The F-102A interceptor is the first combat-ready airplane to use the potting processed plug. The major advantage of this plug is the circuit reliability it gives at high altitude. If you will recall we said that one of the real bugaboos to high-altitude flying was the breakdown of insulation. In potting, the possibility of shorting out between connections in plugs is practically eliminated.

This shorting out, if you remember, was due to rapid changes in atmospheric conditions—the lack of moisture one minute, quantity the next, and the overall, seemingly endless process of evaporation and condensation. Here again is a word of caution, *there is no perfect conductor and no perfect insulator;* but in the long run, the potted plug does the job.

As can be seen in figure 3-14, the wire is soldered into the solder pot. The potting compound acts not only as an insulator between wires, but as a buffer against sudden stress and the possibility of wire breakage at the solder pot joint. The dotted line is the outline of the mold and the extent of the compound.

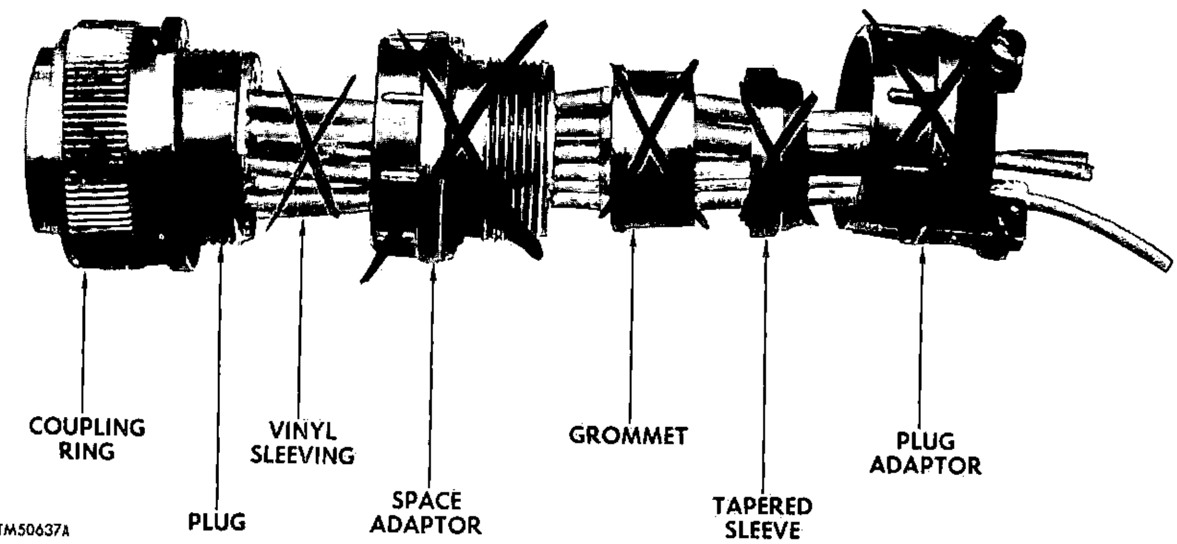


Figure 3-13. Exploded View of Electrical Connector

Because this is a comparatively new process, and because there will be flight-line "spot potting," it is felt that you should know something of the process as a whole, and have a complete explanation of flight-line maintenance of potted electrical components. That's right—components, plural. Not only connectors are potted, but also any of the exposed relays, such as the d-c warning relay, d-c power disconnect relay, and others. Figure 3-15 shows a number of potted electrical components used in the F-102A. You will find that the wires leading into the master warning control box and those leading into the armament disconnect plug are also potted. In addition to these components, all limit switches in the interlock network in the armament circuit are potted. The potting process of any electrical component is identical regardless of nomenclature.

How Electrical Components Are Potted.
The compound used in potting is called "Thickol," a type of synthetic rubber. In color and consistency it is similar to taffy candy before it is pulled—tan in color and pourable. When you were a kid your mother probably made taffy, let it cool slightly, greased your hands with butter, and then by "pulling," the taffy gradually stiffened and when hardened, it was usually eaten. It was pretty good stuff. Needless to say, potting compound is not eaten, nor is it cooled.

In the field, "Thickol" will come to you in two separate containers of proper proportions for mixing. It must not be stored where the temperature is above 75°F. At present the potting compound is a mixture of EC-1120 sealer and EC-1031 accelerator. The accelerator accompanying the sealer must be used with this sealer *and this sealer only*.

This accelerator is highly combustible before it is mixed, and it acts as a positive catalyst to the potting compound. (A catalyst, as you probably learned in chemistry class in high school, is an accelerating agent which produces a quickening reaction on a substance. Water, for example, might be called a catalyst when mixed with cement.)

Once mixed, the potting compound is *immediately* poured into plastic injection gun containers, which are placed in portable freezers whose temperatures are kept at −15°F or cooler. Each container is labeled with its contents, date, time of mixing to the nearest five minutes, and the batch number. It is transported to the working area in these refrigerating units.

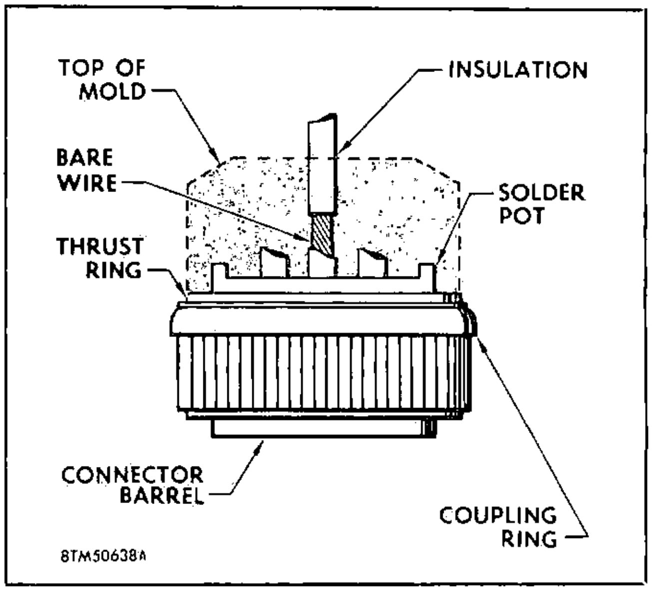
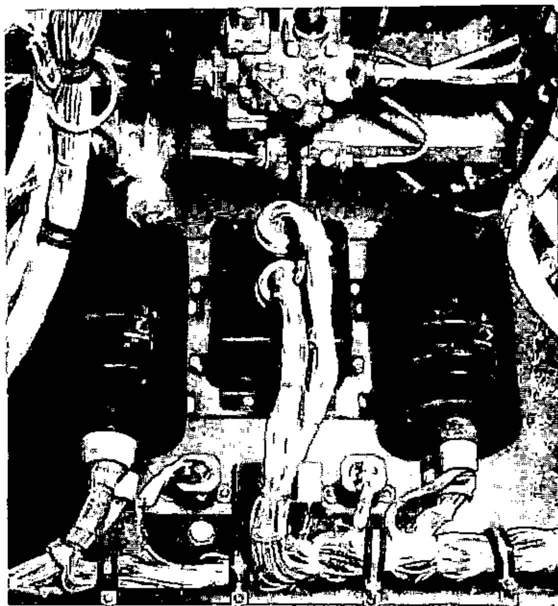
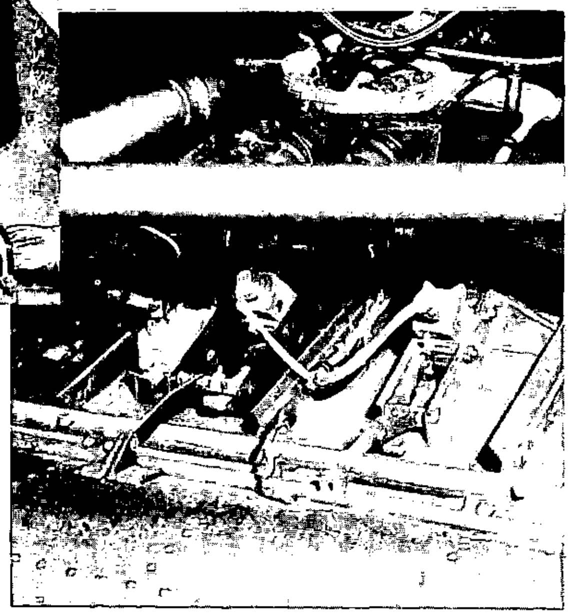


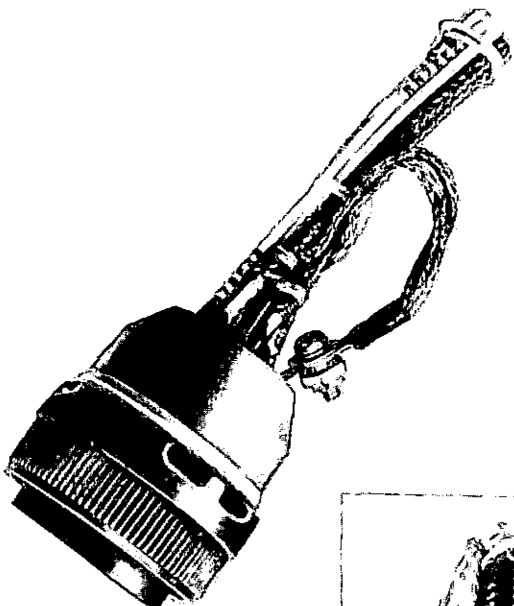
Figure 3-14. Connector Prepared for Potting



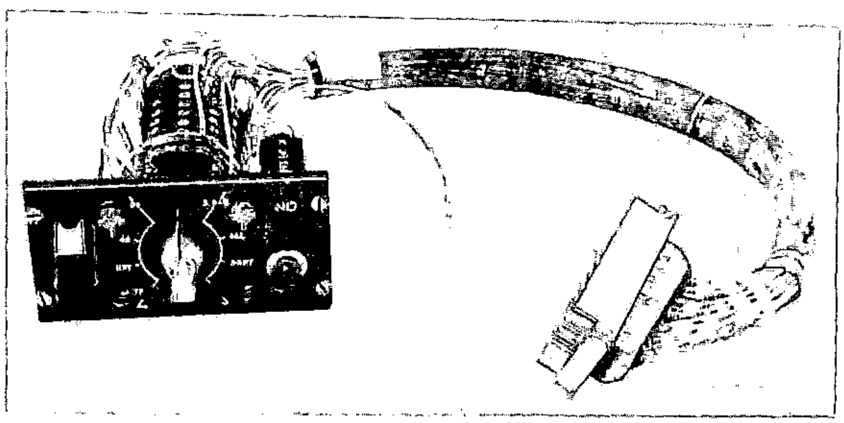
POTTED RELAYS AND CONNECTORS



LIMIT SWITCHES



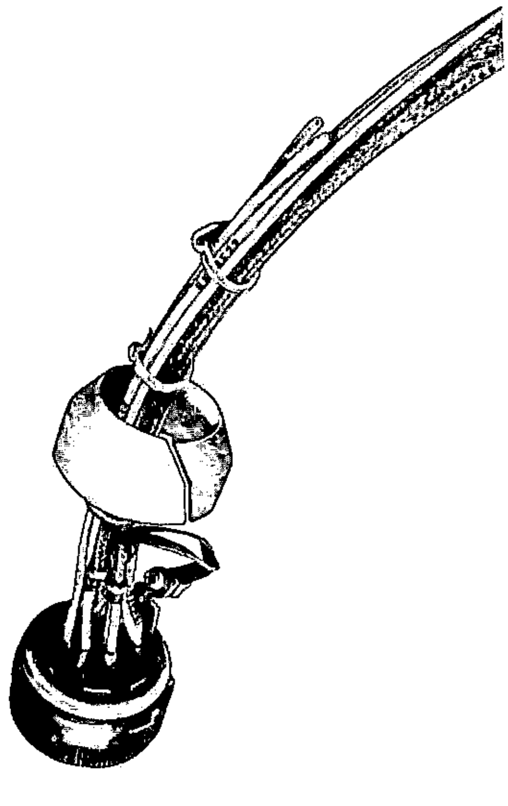
THE POTTED PLUG



POTTED SELECTOR SWITCH

8TM50639A

Figure 3-15. Potted Components in the F-102A



BTM50640B

Figure 3-16. Plug Prepared for Potting With Mold

From 8 to 20 minutes before a plug or relay is to be potted, the container is taken from the freezer. In this period the compound will attain the working viscosity of warm taffy, or heavy molasses. It is not necessary to heat the compound. The injection gun container is then seated in the injection gun, and the air pressure is adjusted until there is a smooth, bubble-free flow of compound. The compound is now ready to inject into the mold. (See figure 3-16.)

When filling the mold, the potter is careful that the compound is solidly packed around all solder pots, between the wires, and the bushing barrel from the bottom to 1/4 inch above the mold. By solid we mean that during the process (as shown in figure 3-17) if any air bubbles are detected, a small putty knife or doctor's tongue depressor is used to tamp them out. If this does not do the job, a clean piece of wire is inserted into or as close as possible to the bubble and moved up and down until the bubble disappears.

One air bubble could cause serious trouble by acting as a reservoir for moisture. The potted portion of an electrical component must be airtight, otherwise we might as well forget potting as a new and revolutionary advancement in circuit protection—one just as important at high altitudes as any relay or other protective device discussed so far in this manual.

The Butyrate Mold.

The butyrate mold is made by dipping steel mold patterns into a hot solution of butyrate, as shown in figure 3-18. The pattern is rotated until an even coating forms on its surface. It is then dipped into tap water which instantly sets and hardens it. The mold is removed from the pattern by cutting around the base of the mold and then diagonally from the top to the base. It is then peeled from the pattern like an orange skin from an orange. The butyrate type mold is clear and flexible, and air bubbles can be seen during the filling process.

Another type mold is made of a cloudy plastic, and air bubbles are not as clearly detected. You will find that this type mold is generally left on the plug as it is more difficult to remove than the butyrate mold. It can, however, be removed with a knife and used for flight-line spot potting—which we will get to in a moment.

After the butyrate mold has been removed from the pattern, the mold is placed around the wiring (the harness) and slipped down until its base is seated on the thrust ring of the connector as shown in figure 3-18. A piece of plastic tape holds the mold in place. The mold is sealed along the diagonal cut with a soldering iron. (See figure 3-16.) The connector and harness are secured rigidly and must be held in this position during the entire potting process which has already been discussed. The potted plug, relay, limit switch, or other electrical component is then allowed to stand for a curing period. For example, the curing time at 75°F, or approximately room temperature, is 24 hours; at 85°F, it is 12 hours. The use of a heat lamp for emergency curing will be discussed under Flight-Line Maintenance of Potted Components.

FLIGHT-LINE MAINTENANCE OF POTTED COMPONENTS.

From time to time you will be called on to initiate wiring changes, the replacement of broken wires, or even the substitution of an electrical component. These, of course, will be repair duties that fall within the scope of organizational (flight-line) maintenance.

The replacement of a *potted* plug, *potted* relay, or *potted* limit switch, and minor wiring modifications and broken wires will require on-the-spot "potting." For that reason we are including in this manual a discussion of potting for flight-line maintenance. The method to be used must be followed to the letter if effective results are to be obtained. The storage and mixing of potting compound will not be covered at this time. The primary reason for this deletion is that the mixing and the care of compound is critical and should not be attempted unless rigidly controlled.

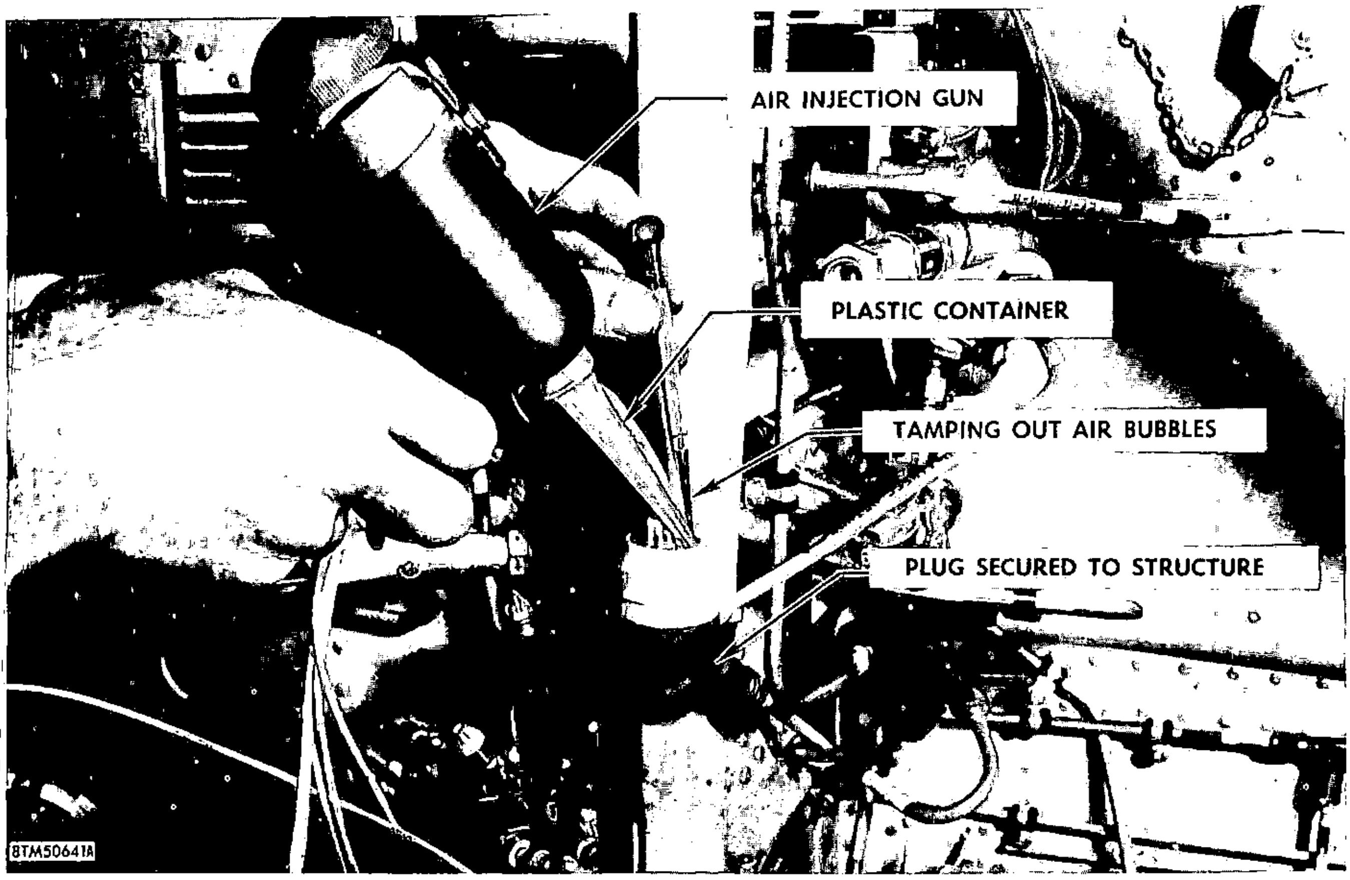


Figure 3-17. Potting an Electrical Connector

The Easy Out.

As long as we are on the subject of plugs, or connectors, *let it be well understood* that the removal or partial removal of the socket contacts of a female connector, or the pin contacts of a male receptacle or connector, is frowned upon as poor maintenance procedure. This is especially applicable to high-speed, high-altitude aircraft such as the F-102A.

In the conventional connector there are times when it is necessary to disassemble a plug for cleaning or repair. In a potted plug this should not be necessary if the proper inspection against corrosion of receptacles and connectors is maintained.

The reason for this restriction of pin and socket removal should be clear enough. The pin or socket can become so loose in the insert hole that poor electrical contact results when they are mated. Because of this loose mating, the insert holes become a moisture collecting agency which, as you know, causes breakdown of insulation which in turn arms the greatest enemy to an electrical system, *corrosion*. So unless there is a critical emergency, and you have received the go-ahead signal from your crew chief, do not remove or even partially remove a pin or socket from a potted plug. *Never attempt pushing contacts up and through the potting compound.* Let's keep that airplane in a combat readiness status, and not endanger the success of a mission or the life of the aircraft pilot. Let's not look for what might seem the "easy out" (and we are not talking about tools) when maintaining airplanes.

Flight-Line Potting Equipment.

The only difference between *potting* on the line and potting on the bench level or manufacturing level is an even stricter adherence to the potting rules and some plain old common sense.

Let us assume that our compound is mixed, and is standing by in a portable freezer, ready for use. The freezer is on a cart and under the wing out of the sun. Next to it on the cart is a potting kit consisting of: various sized potting molds, a small sharp knife, an injection gun, clean Teflon injection tubes, a roll of clean cheesecloth, a small oil-free, dust-free paint brush, a narrow bladed putty knife or some tongue depressors such as doctors use, and a bottle of CVAC SOL 1050-56, a solvent for cleaning the surface of the component to be serviced. An isopropyl alcohol may be used, but *only as a substitute.*

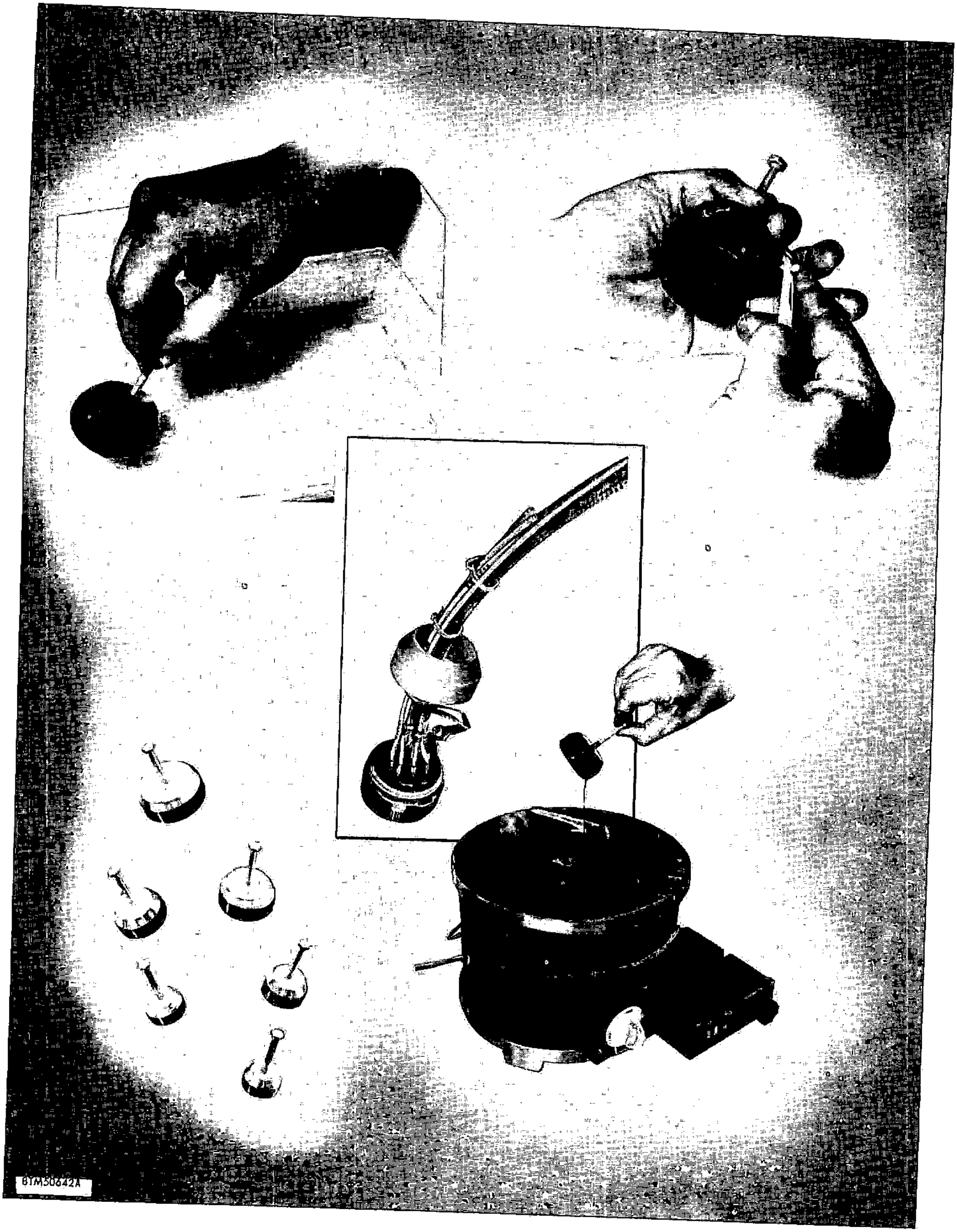


Figure 3-18. Making Butyrate Molds

Flight-Line on-the-Spot Potting.

As you have learned, there are a number of reasons why it may be necessary for you to perform flight-line on-the-spot potting. (See figure 3-19.) If it happens to be new wire and there are a number of pigtails already in the plug, there is no problem. You simply select the proper pigtail and make a permanent splice. A pigtail is a six or eight-inch spare piece of wire used for wiring additions, and is connected to an unused contact in the connector. In the F-102A the tips of these pigtails are also potted. For the purpose of this discussion, let's assume pigtails are not a part of this particular plug assembly, or that a break has been traced close to the solder pot within the potted portion of the plug. What do we do now? *There is only one safe method of repair for a potted plug—complete removal of the potting and repotting.*

REMOVAL OF POTTING. If the plug in question happens to be the type of which we spoke earlier—as having the cloudy, grey mold, as part of the assembly—take the small sharp bladed knife from the potting kit and cut the mold from the top to the mold base. Remove the mold sleeving. The base of the mold is left on the connector. If you are short of molds, or the correct size mold has been misplaced, the mold sleeving just removed may be used for repotting. The potting is removed with the knife. Be very careful while digging out the potting compound from between wires—a cut wire may mean another broken circuit. Work slowly and cautiously. When the potting is removed, scrape the surface clean and examine the other wires for secured connections, or any possible damage which may have occurred during the removal of the potting. Remove the broken wire with a small soldering iron.

CLEANING FOR REPOTTING. A plug must be surgically clean for repotting. This is the reason for the solvent, the clean paint brush, and cheesecloth in your potting kit. Pour a small portion of the CVAC SOL 1050-56 into a clean pan. Take the dust-free, oil-free paint brush, dip it into the solvent and clean the face of the rubber insert, between the wires, the wires themselves and the empty solder pot. When you are sure the surfaces are clean, moisten one end of a piece of cheesecloth with solvent and clean the entire surface of the connector. When dry, secure the connector in an upright position. The connecting face of the plug should be held on a flat surface or tied firmly. The less movement during repotting, the better the job. A good job of repotting is the only acceptable job.

REPOTTING. We are now ready to repot. Place the correct size mold around the harness and slide it down until it fits snugly against the mold base. Wrap the mold with plastic tape. If this particular potted plug is one which did not come with the mold on it, the repotting mold is seated against the thrust ring of the connector and secured by means of plastic tape. If the mold is butyrate it is sealed lightly along the cut edges with a soldering iron. If it is a plastic mold it is wrapped with tape.

To pot a connector we must have some compound. If you remember, we left the refrigerated compound in the freezer under the wing. Take the compound from the freezer and let it stand for 8 to 20 minutes, or until it is pliable under finger pressure. During this time interval attach an air hose to the injection gun and insert a clean Teflon injection tube. When the compound is sufficiently defrosted, insert the compound container in the injection gun and proceed to pot as explained earlier in this chapter.

CURING THE POTTED PLUG. The curing period for a potted plug varies with air temperature. In cold climates it may be necessary to use a heat lamp. This method of curing may also be used in milder climates when an emergency arises, but is not recommended. For the best results, the curing or hardening process of a potted plug should be carried out under temperatures not lower than 70°F or more than 85°F. The following chart gives the prescribed curing time for potted electrical components and approximate curing temperatures:

CURING TIME FOR POTTED COMPONENTS

Curing Temperature (Max.)	*Curing Time (Hours)*
75°F.	24
85°F.	12
95°F.	6

CARE AND MAINTENANCE OF CONNECTOR PLUGS.

When we discussed the maintenance and reliability of electrical equipment, we referred to a case history where a corroded connector caused a false fire warning. We said this was a maintenance oversight, and that is what it was. The fact that this particular plug was located in the pilot's console was no excuse. A plug is as accessible there as it is in a wheel well or any other compartment in the airplane. There are no excuses for maintenance oversights. No excuses, period.

Connector Plug Inspection.

Plugs are inspected periodically for corrosion, any indication of overheating, and for secure connections. To check a plug for corrosion, break the safety wire and unscrew the coupling ring with a strap wrench or other approved tool. Disengage the plug as shown in figure 3-20, by pulling it straight out by the metal body of the plug. Under no circumstance should the plug be disconnected by pulling on the attached wiring.

When the mated connectors are parted, look for corrosion. It may be found in either the female contact or

Figure 3-19. Flight-Line on-the-Spot Potting

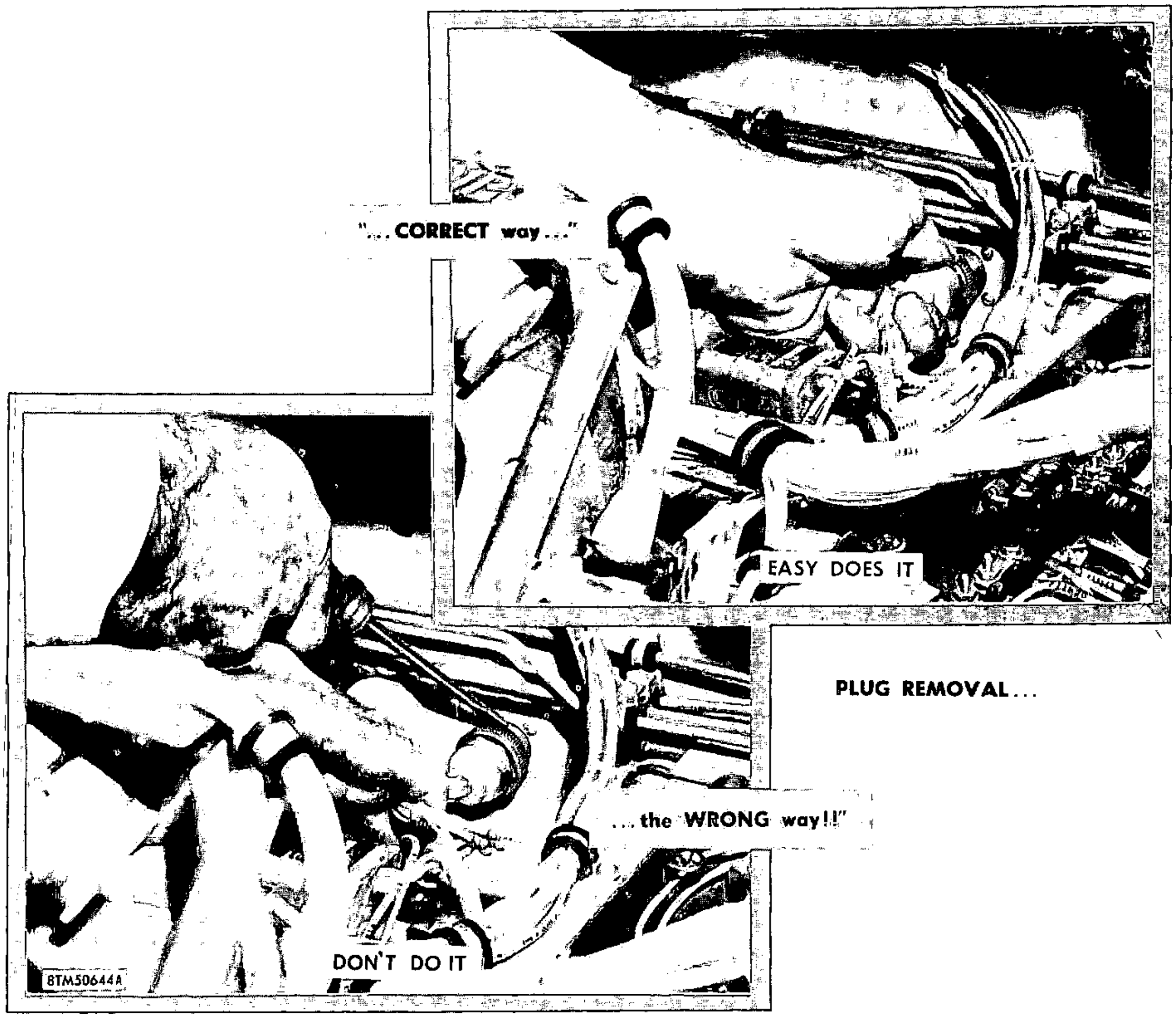


Figure 3-20. Disengaging a Connector

on the surface of the male contact. Corrosion is a chemical or electro-chemical destruction of metal. It can be detected in a plug by its tell-tale grey, white, powdery deposit. It can be found anywhere. It acts on metal the way termites undermine the structure of a home. One day you have a home — the next day it falls out from under you.

Corrosively speaking, one day you might have fine performance in the airplane you are maintaining; the next day poor performance (intermittent operation) or the aircraft may fail to return to the base. You will wonder why—you may not sleep well that night or for many nights. A lot of people will not sleep, a lot of people will wonder why the airplane did not come back.

Don't let this happen to the aircraft you maintain. There is no reason for it to happen, if when you detect corrosion the contacts of the plug are cleaned with a brush or clean rag which has been dipped in a non-corrosive solvent, or alcohol.

After you have inspected the contacts for corrosion, check the coupling ring for battered threads. If the threads are not in good condition, the coupling ring must be replaced. In a potted plug, this would entail a complete de-potting operation in order to remove the coupling ring. If such a case occurs, consult your crew chief and let him decide the best method of repair. If it is a plug in or about the engine area (these plugs are not potted), then a complete disassembly of the cable end of the plug is required. The moral to the above discussion is: When attaching or detaching connectors, carefully avoid damaging the coupling ring. Stripped threads or a battered or bent coupling ring will, as you know by now, require the complete replacement of the ring in question.

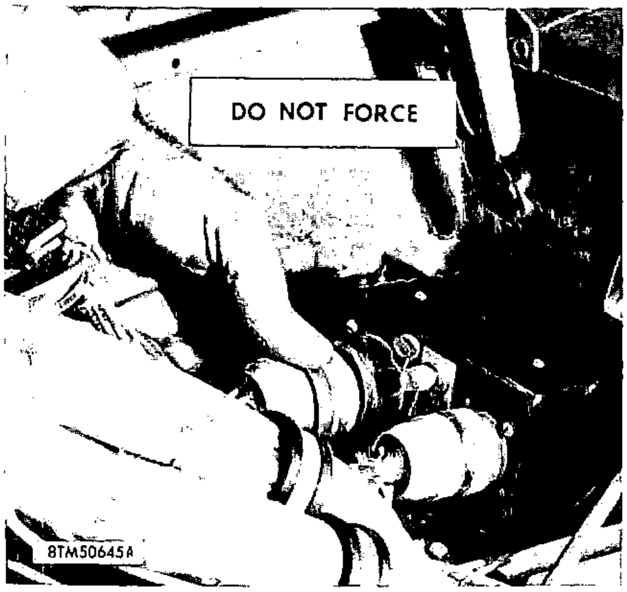


Figure 3-21. Engaging a Connector

Attaching the Connector.

Connector plugs are easily installed. It is not necessary to force the plug into the receptacle. Before attaching the plug to the receptacle, however, lubricate the threads with an anti-seize compound. Be sure you apply the compound to only the first two or three threads. The plug is then positioned in such a way that the key of the one part is lined up with the keyway or groove of the other part; the plug is then pressed into the receptacle with a light hand pressure—not forced! Be careful also, when engaging the threads of the two parts that they mate smoothly; then tighten the coupling ring *finger tight*. Use a strap wrench, or other approved tool (such as water-pump pliers covered with tape) *to tighten the coupling 1/8-inch beyond finger tight.* Use new safety wire to secure the connector. Do not attempt to salvage the old safety wire. It's about as useless as the string of a hurriedly opened bundle of laundry.

TROUBLE SHOOTING THE F-102A.

In discussing trouble shooting, let's get straight to the point. There is nothing simple about trouble shooting. But if you understand the circuits involved, your job is simplified immeasurably. You should know by now that any gadget having moving parts is subject to malfunction at one time or another no matter how well it is engineered. You should also know by now that due to the peculiarities of electricity, under certain atmospheric conditions and the stress and strain of flight, electrical components are unpredictable.

Unfortunately, things do wear. If this were not true there would be little need for you as a mechanic, of inspection, of maintenance as such, or of this training supplement. In this section, trouble shooting or circuit testing will be discussed generally. Testing equipment, such as the voltmeter, ohmmeter, and continuity tester, which is used in trouble shooting, will also be covered. For the trouble shooting charts applicable to the F-102A you should refer to T.O. 1F-102A-2-10. This is also true for the operational checks and testing of the F-102A electrical systems. The inspection requirements will be found in T.O. 1F-102A-6.

But as long as the above subjects have been brought to mind, the inspection and the operational checks and testing are as important as any function you might perform on the line. They are vital! This is maintenance, this is what we mean when we speak of "... your efficiency in maintaining . . ." a certain aircraft. So, in the inspection, maintenance, and operation of aircraft electrical equipment, you must and will be called on often to measure voltage, current, resistance, and to make continuity checks.

A number of instruments have been developed for this purpose. You have heard of them—you have probably used them. Let's talk about them. Let's find out what makes them "tick" and how and why they are used. In order that you understand these instruments thoroughly, perhaps we should touch on a measuring device known as the galvanometer. It is basic. The principles of the D'Arsonval galvanometer have been applied to the design and construction of both the voltmeter and the ammeter.

THE GALVANOMETER.

When we talked about the effect of a loop on a magnetic field in Chapter I, we were talking indirectly about the basic principles of the D'Arsonval galvanometer. To review: if you bend a straight conductor into a loop and current flows through the conductor, a magnetic field is set up within the loop. Now, if you were to place a compass needle inside the loop, as shown in figure 3-22, the induced magnetic field would cause the needle to deflect. This is the galvanometer in its simplest form.

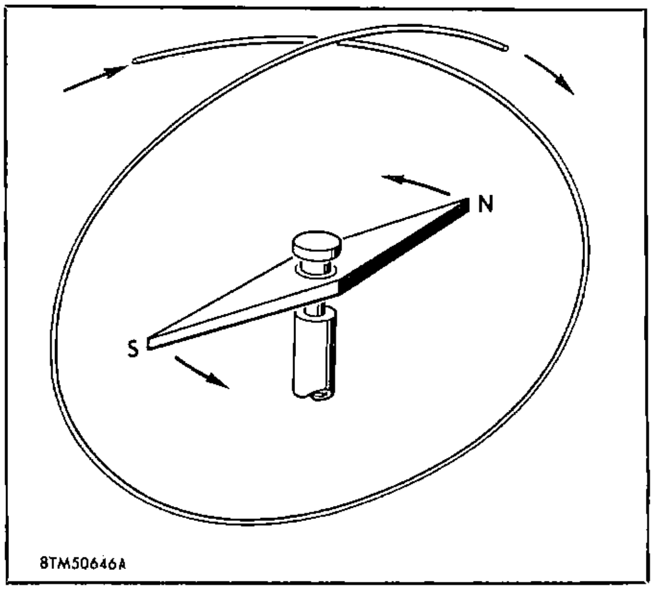


Figure 3-22. Simple Theory Behind the Galvanometer

The galvanometer movement is used in voltmeters, ammeters, thermocouple thermometers, electrical tachometers, and other similar instruments. Figure 3-23 shows the internal wiring of a galvanometer.

The galvanometer used for airplane electrical systems is the moving coil D'Arsonval type. This particular galvanometer consists of a coil of many turns of very fine wire, having a resistance of hundreds and even thousands of ohms. It is wound usually on an aluminum frame which is suspended between the poles of a horseshoe magnet. The core is securely fastened. The coil frame is on a shaft and is free to move on jeweled bearings attached to the core.

When current is induced, the coil acts like a small motor and starts to turn. Its rotation is opposed by a spiral spring whose ends also serve as current leads, so that the current flowing into the coil by the minus (−) lead is counterbalanced by the amount of current in the plus (+) lead. A calibrated scale indicates the movement that takes place.

You have probably seen old-fashioned scales used for weighing sugar, coffee, and the like—when a scoop of sugar must be balanced by its equal weight on the other side of the scale. Chemists and druggists still use a sensitive version of these old scales for weighing very minute quantities of chemicals and drugs.

In the same manner a galvanometer is so sensitive that its needle will be deflected by extremely small currents. The current flowing through the armature of the coil establishes a magnetic field. In this way one side of the core becomes a north pole and the other a south pole. From your knowledge of magnetism you should be able to figure out what happens when a current flows into and through a coil which is suspended between the poles of a permanent magnet. Our law of attraction and repulsion is applied. The south pole of the coil seeks the north pole of the permanent magnet, and at the same time, the north pole of the coil seeks the south pole of the permanent magnet.

Let's apply Ohm's law to this. We know that the extent of the rotation in the coil depends on the amount of induced current. Since the amount of current flow depends on the applied voltage, and since the resistance in the coil of a galvanometer is constant, the amount of deflection of the coil, according to Ohm's law, will indicate the amount of applied voltage. Now, all that need be done is to add a scale calibrated in such a manner that the needle deflection will indicate the amount of applied voltage — and we have the voltmeter.

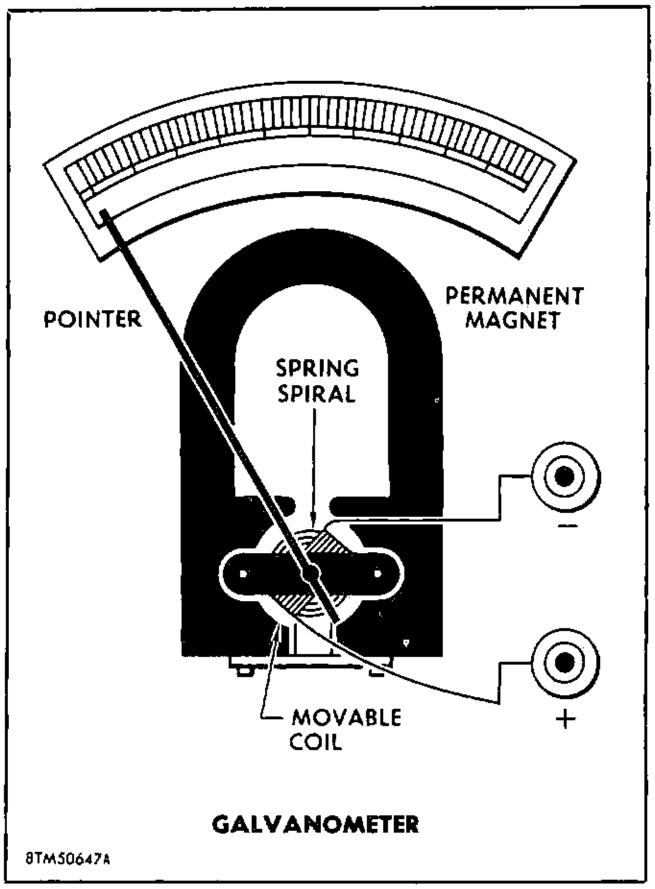


Figure 3-23. Internal Wiring of a Galvanometer

THE VOLTMETER.

The voltmeter used on airplanes is a galvanometer movement connected in series with a high-resistance unit. The voltmeter is used to measure the potential difference between two points. The purpose of the high resistance is to limit the current flow through the meter movement. Most voltmeters give a full scale deflection with less than 0.01 ampere flowing through the movement. Because the resistance of the meter is fixed, the current flow and the amount of deflection of the needle, or pointer, are dependent on the voltage applied to the terminals of the meter. The pointer indicates on the scale the voltage across the unit being measured. Figure 3-24 shows the internal wiring of the voltmeter.

When trouble shooting or checking a system with a voltmeter, always connect it in parallel with the electrical unit to be tested. For example, if you were checking the voltage at the d-c voltage control panel of the F-102A, you would connect the plus (+) terminal of the voltmeter to the positive (+) side of the panel, and the minus (−) terminal of the voltmeter to the negative (−) side of the panel. That is, you would if you wanted to attain a correct direction of pointer deflection. If by accident you connected the voltmeter in series with the circuit, no harm would be done to the meter, but the other units in the circuit would not operate.

In the first instance, *the current is limited* by the high resistance of the meter, and in the second, the *current flow is limited by the meter. This should be clear*

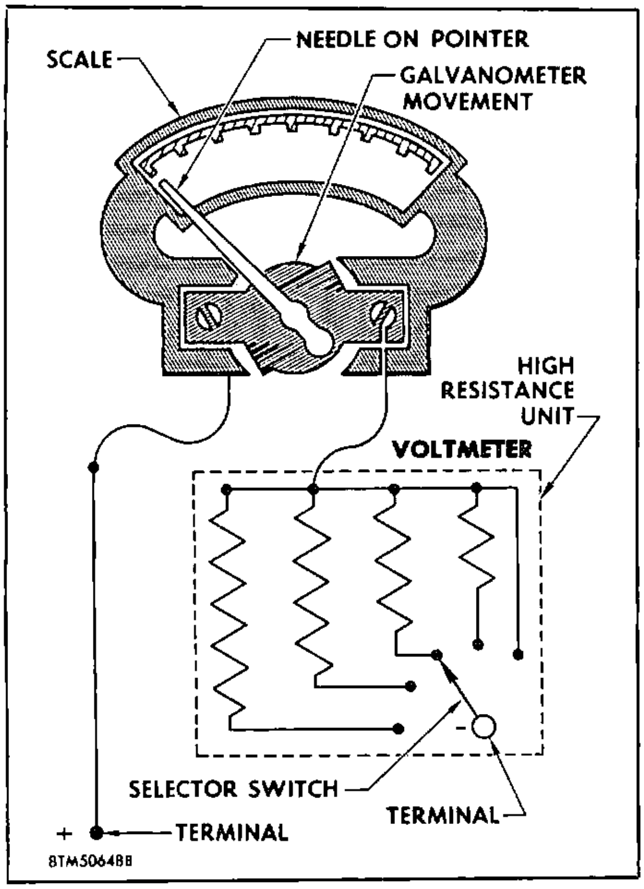


Figure 3-24. Voltmeter Wiring Schematic

enough, if you remember our discussion of series circuits in Chapter II. Short or blow one gadget in the circuit and they all cease to operate. A word of caution—*never connect a voltmeter to a source of voltage which exceeds its calibrated scale.*

Note the selector switch at the lower right portion of the schematic (figure 3-24). This switch permits you to introduce various fixed resistances into the voltmeter circuit, so that wide ranges of voltages can be measured with one small, compact instrument. Otherwise, you would have to use either a meter with a much larger scale, or a number of separate instruments with appropriate internal resistances for the voltage range for which they are calibrated.

When measuring the voltage of any circuit, it is always advisable to select the highest range indicated by the selector knob. According to the deflection of the needle, you can then move the knob to lower ranges to obtain more sensitive readings. This method of starting at the highest range provided by the selector prevents violent deflection of the needle and subsequent damage to the internal mechanisms and circuits of the meter, which would happen in case you accidentally measured a high voltage with the switch at a low indicated range.

THE AMMETER.

An ammeter uses essentially the same internal movement as the voltmeter we have just discussed. Instead of a current limiting resistor in series with the movement, the ammeter uses a shunt, or current bypass, in parallel with the movement. This arrangement is shown in the accompanying internal wiring diagram, figure 3-25.

As you learned in electrical school, a shunt is a resistor made of a special alloy which changes very little in electrical resistance even though there may be decided temperature changes. The movement of the meter and the resistances of the shunt are so proportioned that most of the current flows through the shunt. Only a small fraction of the current flows through the movement—just enough to give a full-scale deflection when the total current is equal to the maximum range of the ammeter.

An example of this can be seen in the illustration of an ammeter with an external shunt (figure 3-26). This is a 300-ampere ammeter, typical of the one used in airplanes. If a movement is used which requires 0.01 ampere to give full-scale deflection, only 0.01 ampere will flow through the meter when 300 amperes is measured. As can be seen, the remaining 299.99 amperes will flow through the shunt.

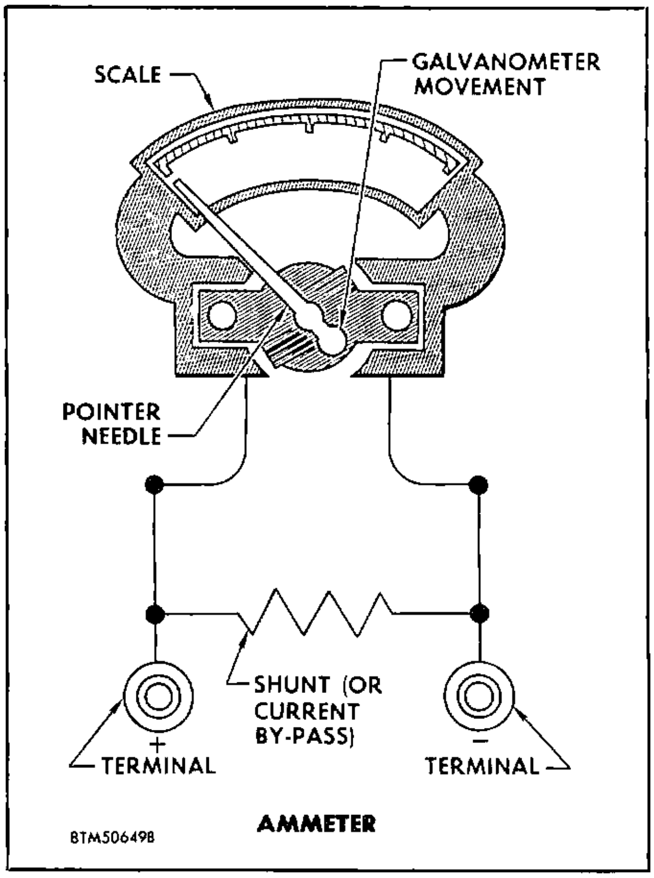


Figure 3-25. Ammeter Wiring Schematic

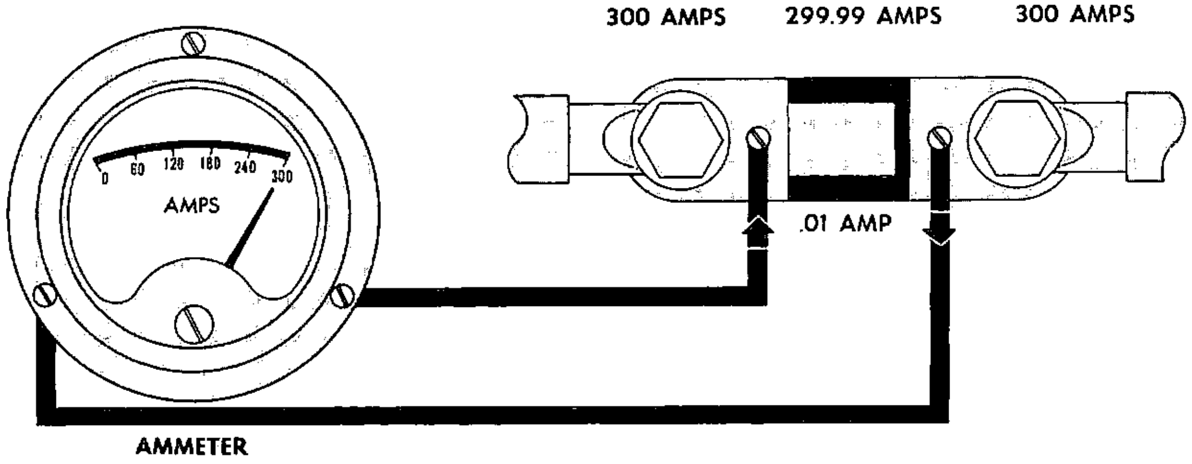


Figure 3-26. Ammeter With External Shunt

While we are on this subject, most airplanes that have an ammeter mounted on their instrument panel, have the shunt part of the ammeter installed in a junction box which is connected to either the plus (+) or minus (−) generator lead. When you find an installation of this kind, or if you ever have to make such an installation, you must always use the leads furnished with the meter. Never alter their length. The reason for this warning should be obvious.

An ammeter is used to measure electrical current flowing in a circuit—and as you learned in Chapters I and II, there is such a thing as "drop." Also, unlike our voltmeter, *an ammeter is always connected in series with the load.* The positive (+) terminal of the ammeter is connected to the positive side of the load. Under no circumstances should the leads be connected across the terminals of a battery or generator—you would immediately burn out the meter. An ammeter is never used, for example, in continuity testing, since it has very low resistance and would thereby short-circuit the battery.

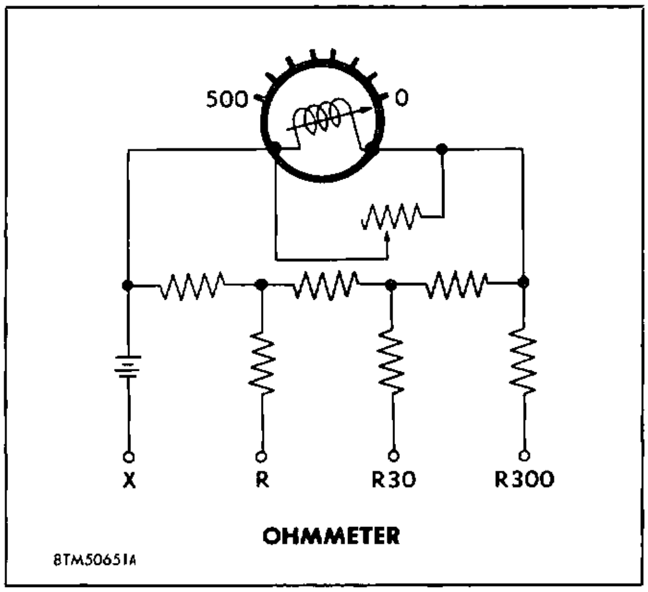


Figure 3-27. Internal Wiring of an Ohmmeter

THE OHMMETER.

An ohmmeter is composed of a meter movement, a network of resistors, and a small dry cell battery as a source of voltage. The internal wiring is shown in figure 3-27. An ohmmeter is used to measure the resistance of a unit or circuit. Since the voltage supplied by the ohmmeter is constant (the dry cell), the current through its meter movement, and hence the deflection of the pointer, depends on the resistance of the unit being tested. The meter, therefore, indicates the resistance of the wire, or cable, or unit measured. It is calibrated to read this resistance directly.

In trouble shooting, the ohmmeter, although primarily designed to measure resistance is useful for checking continuity. We will discuss continuity testing in a moment, but first let's take another look at the internal wiring diagram of our ohmmeter (figure 3-27).

Remember, the ohmmeter uses a small battery for its source of voltage. The fixed resistors are of such value that if the test probes are shorted together, the meter will read full scale. The variable resistor, in parallel with the meter, and the fixed resistors serve to compensate for changes in the voltage of the battery. The variable resistor provides for a zero adjustment of the indicator which is made on the meter control panel.

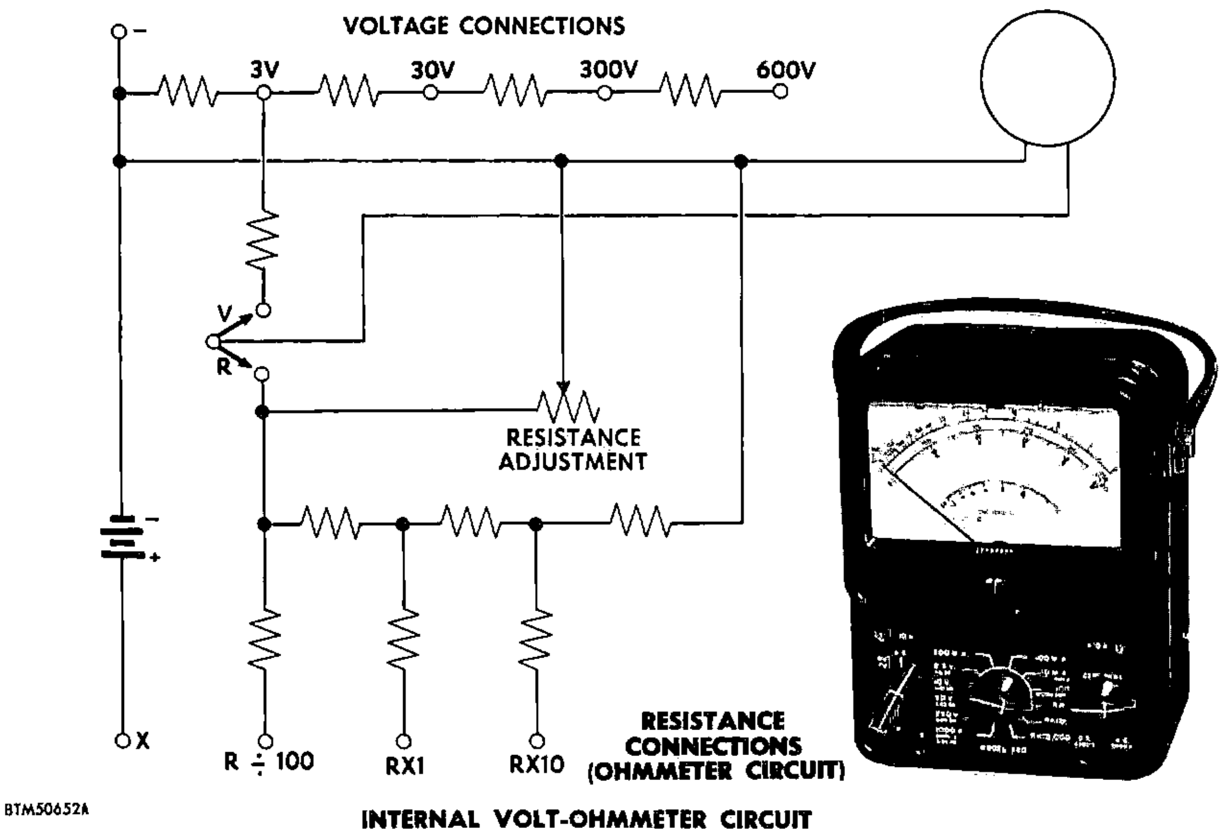


Figure 3-28. Simple Wiring Diagram of a Typical Volt-Ohmmeter

On the face of the meter there may be several calibrated scales. These are made possible by the various values of resistances and battery voltages. When using the ohmmeter you first select the desired scale by a selector switch on the meter panel. Each scale will read low resistances at the upper end. The greater the resistance in a circuit, the less deflection will be seen on the indicator.

THE VOLT-OHMMETER.

The instrument most used in testing and trouble shooting circuits is known as a volt-ohmmeter. As the name implies, this is a combination of the voltmeter and ohmmeter. The schematic drawing (figure 3-28) shows the internal wiring of a typical instrument. As is the case with most manufactured products, there may be variations in the circuits and the connections. The one illustrated shows the fundamental principles behind all volt-ohmmeters; whether it is a Simpson, model 260, or any other instrument; only the values on the face of the meter change.

As we have said, the volt-ohmmeter is a combination instrument. A rotary switch on the instrument case is moved to the position marked "V" when the voltmeter is used, and to "R" when the ohmmeter is used. One of the test leads is plugged into the jack "X," the other into the jack labeled "R" when it is to function as an ohmmeter; the "V" jack of course is used for voltage. To eliminate any resistance in the leads themselves, the free ends of the test probes are touched together, and the adjustment knob is moved until the needle indicates zero on the right side of the scale.

By using our simplified volt-ohmmeter above, let's measure the resistance of an unknown resistor. This is done by touching the two test leads at each end of the resistor. When the needle comes to rest on the scale, the value in ohms will be indicated. But let's suppose that we have not selected the correct resistance range and the needle moves completely off the scale. To get a proper reading you would simply disconnect the test lead from "R" and insert it in "R X 10." Any reading you now get on the scale must be multiplied by 10 to obtain the correct resistance. Note that the volt-ohmmeter illustrated has a rotary switch and dial with several scales for this selection.

THE CONTINUITY TESTER.

Continuity testers vary in construction and operational design. One type consists of a small lamp connected in series with two dry cell batteries and two leads with alligator clamps. Another type contains two batteries conneced in series with a d-c voltmeter. A completed circuit will be registered by the voltmeter.

CONTINUITY IN AN ELECTRICAL CIRCUIT.

An electrical circuit must possess continuity. Mr. Webster says that continuity is the state or quality of being continuous; something that has or gives *continuousness* or sequence. The word *continuousness* is a good word, especially when we are talking about the continuity in an electrical circuit, for if we don't have continuousness, we have trouble.

In other words, an electrical circuit, through which our electrons "beaver" so eagerly from the generator on through their maze of elements in a circuit, must have continuousness if it is to do the job demanded of it. As you have learned, an ohmmeter, voltmeter, and continuity tester are used to test the continuousness of an electrical circuit—*an ammeter, never.*

If an electrical circuit lacks continuity, it is generally attributed to three causes: *open circuits,* due to broken leads and wires, tripped circuit breaker, or a blown fuse; *short circuits* in which grounded leads cause the current to be returned by short-cuts to the source of power; and *low power circuits* which make aircraft lights burn dimly and the relays chatter.

Electrical troubles, however, may not always develop in the wiring; it could be within one of the electrical units. Even if there is a failure in any electrical unit when its control switch is turned on, a check must be made of the power lead to the unit, to determine that it is neither shorted to ground, nor is an open circuit. It is most important that trouble in an aircraft electrical system be carefully analyzed, and systematic steps be taken to isolate it. Time and energy can be saved, and damage to expensive testing equipment can be avoided, if you follow a specific procedure. Do not bulldoze ahead without rhyme or reason.

CONTINUITY TESTING FOR "SHORTS" AND "OPENS."

A short circuit will make the protective circuit breaker trip; an overload, because of ground fault within an operating electrical unit, will do the same. A short may be caused by a lead that is shorted to ground or to another electrical lead. Shorts are dynamite to an aircraft electrical system, and unfortunately, in some instances they cannot be avoided. The reason for a short circuit is sometimes nebulous and hazy.

Usually shorts can be attributed to moisture, the direct contact with liquid seepage, punctured insulation, or chafing due to constant rubbing against another lead or against the airframe of the airplane. There are others, but if you will look at any trouble shooting chart, the second column will read, "Probable Cause." To refer to Mr. Webster again, "probable cause" is an expression taken from the law which means a reasonable ground of assumption that a charge is well founded. But, if you have a tripped circuit breaker your "short circuit" cause is well founded.

When you have reached this "reasonable ground of assumption," you will probably be right (that a short circuit is the basis of the trouble); visually inspect the wire from the tripped breaker throughout its entire length to the unit it connects. For example, a short circuit will create heat, and heat will char or brown.

Vinyl tubing, found on wires or around bundles, as a case in point will darken perceptibly even if cigarette smoke is blown through it. Remember—the cause is probable, the effect is not; so if you find the wire visibly free of defects and the connecting plug free of moisture, the electrical unit itself must be checked. Let's go back to our "sure-fire" testing instruments.

The two illustrations in figure 3-29 show you the techniques in tracing "shorts" or "opens" when voltmeters or continuity testers are used for checking circuit continuity. As you can see, it is a process of elimination and isolation—each wire segment is progressively checked until an open or disconnected area is found. The illustration shows when a continuity tester is used. These are the surest and quickest methods for checking the circuit for "shorts" and "opens."

An *open* between an electrical unit and its power source will not cause a circuit breaker to trip or a fuse to blow. The current just won't flow at all; if started, it may flow only partially through the circuit and then directly back to the source. In locating an open circuit, either a voltmeter or continuity tester may be used as we have illustrated.

However, before making a continuity check be sure the wire being checked is isolated electrically. By this we mean, be sure a false continuity reading is not being obtained through an interconnecting circuit. With a continuity tester, the lamp lights when there is a completed circuit. It will not light when there is an open or incomplete circuit. With a voltmeter, the first zero reading during a continuity check will indicate the break in the circuit.

Keep in mind though, when testing a landing lamp for example, that the voltmeter will read the desired voltage when the positive lead is connected to either terminal of the lamp. The reason for such a reading is, of course, that there is no potential difference across the lamp. There is no current flowing through the lamp, except that allowed by the voltmeter, and this would not be enough to light the lamp.

In checking with an ohmmeter, connect the leads across the circuit. A zero ohms reading will indicate circuit continuity. As you learned when we discussed ohmmeters, choose the dial reading (whether it reads R x 1, R x 10, or R x 10,000) which you think will contain the resistance of the element you are measuring. Generally speaking, you should select a resistance value in which the reading will fall midway, or somewhere in the upper half of the scale.

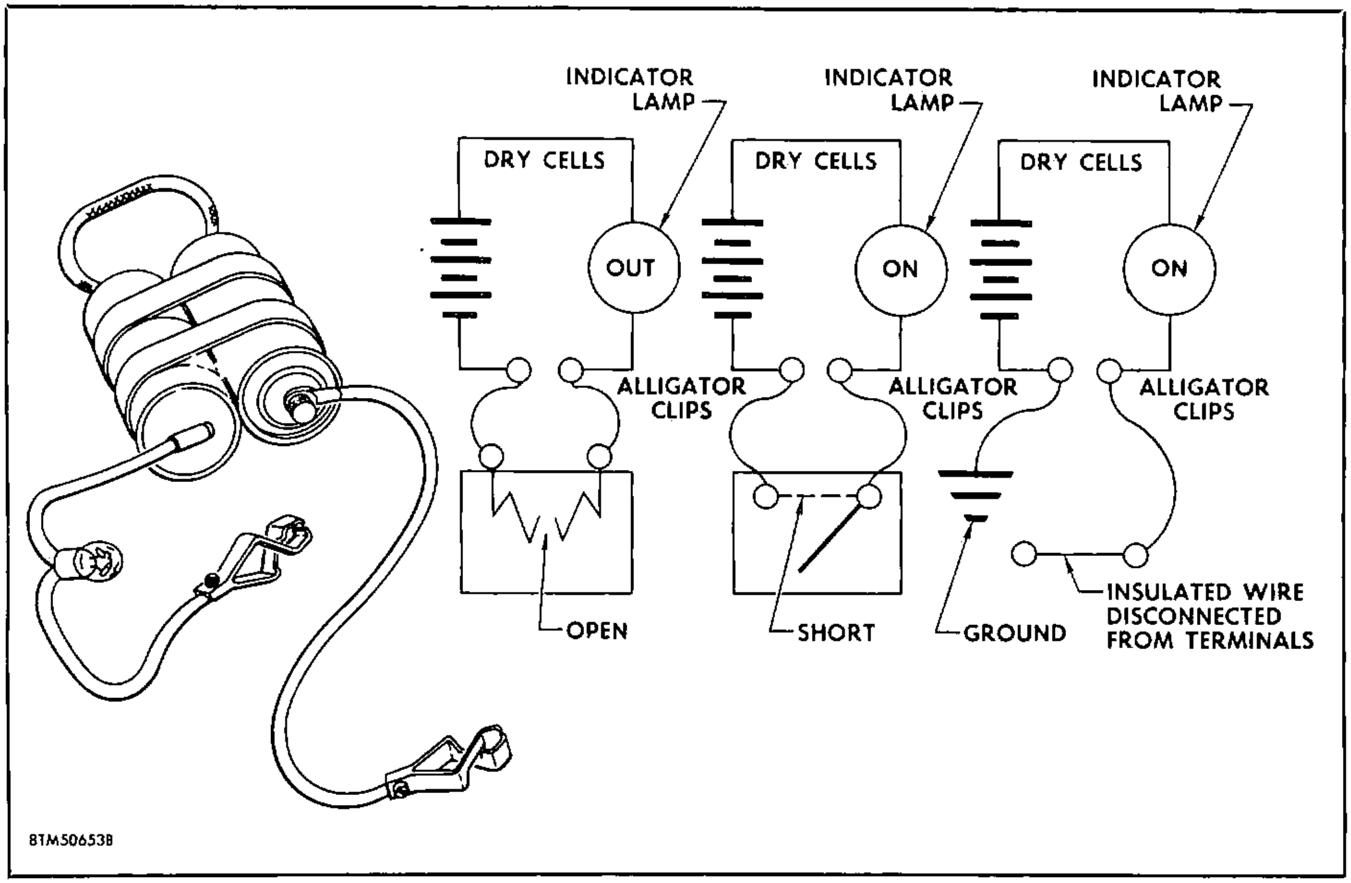


Figure 3-29. Continuity Testing With a Continuity Tester

Here are a few words of caution when using measuring instruments. When testing with a continuity tester, the test lead probes should be inserted at the segment connection points *only.* Under no circumstance should you pierce the insulation at intermediate points with either the probes, or a sharp instrument of any kind. Permanent damage would be the result.

When making a circuit test with an ohmmeter, *never* attempt to check the continuity or measure resistance in a circuit while it is connected to a source of voltage. In addition, disconnect one end of an element when checking resistance; otherwise the ohmmeter will read resistance for the parallel paths. And finally, *never try to be a good guesser.* Always locate the trouble in the positive lead of the circuit, the operating unit itself, or in the negative lead *before you attempt to remove one piece of equipment, or one segment of wire.* Always be sure—then go ahead. We will refer to trouble shooting more specifically in the next section of this supplement, when we analyze the d-c electrical system of the F-102A.

LEADING COMPONENTS OF THE F-102A D-C SYSTEM.

In Chapter I we introduced you to the d-c electrical system of the F-102A. Chapter II was given over primarily to a general discussion of power distribution through a d-c system, the circuits used, the controlling devices, the main power source, and finally the essential and non-essential bus system of the F-102A. In this chapter, we have traced d-c electrical power from the earliest airplanes to the more modern equalizer d-c circuit. Our next topics for discussion were the maintenance and reliability of equipment; their performance at high altitudes, in cold weather, in sand-laden atmosphere, and in the tropics where fungi and moisture play such havoc with electrical systems.

We then discussed the wiring and its applicability to the F-102A—the terminals used, the potted plug, and the care and maintenance of connectors in general. You then learned the instruments used in trouble shooting and how they should be used. Any discussion of trouble shooting in a manual such as this one can only generalize on the subject. It is the "doing" that counts.

As we told you, the trouble shooting charts for the F-102A will be found in T.O. 1F-102A-2-10, and at best they can only be probable in nature. It is you who must ferret out the trouble and this can only be done as we have outlined under the section given over to trouble shooting — by a patient and efficient analysis of the "probable cause." But you can't even do this, if you do not know how the system works, what components it contains, or how it is wired.

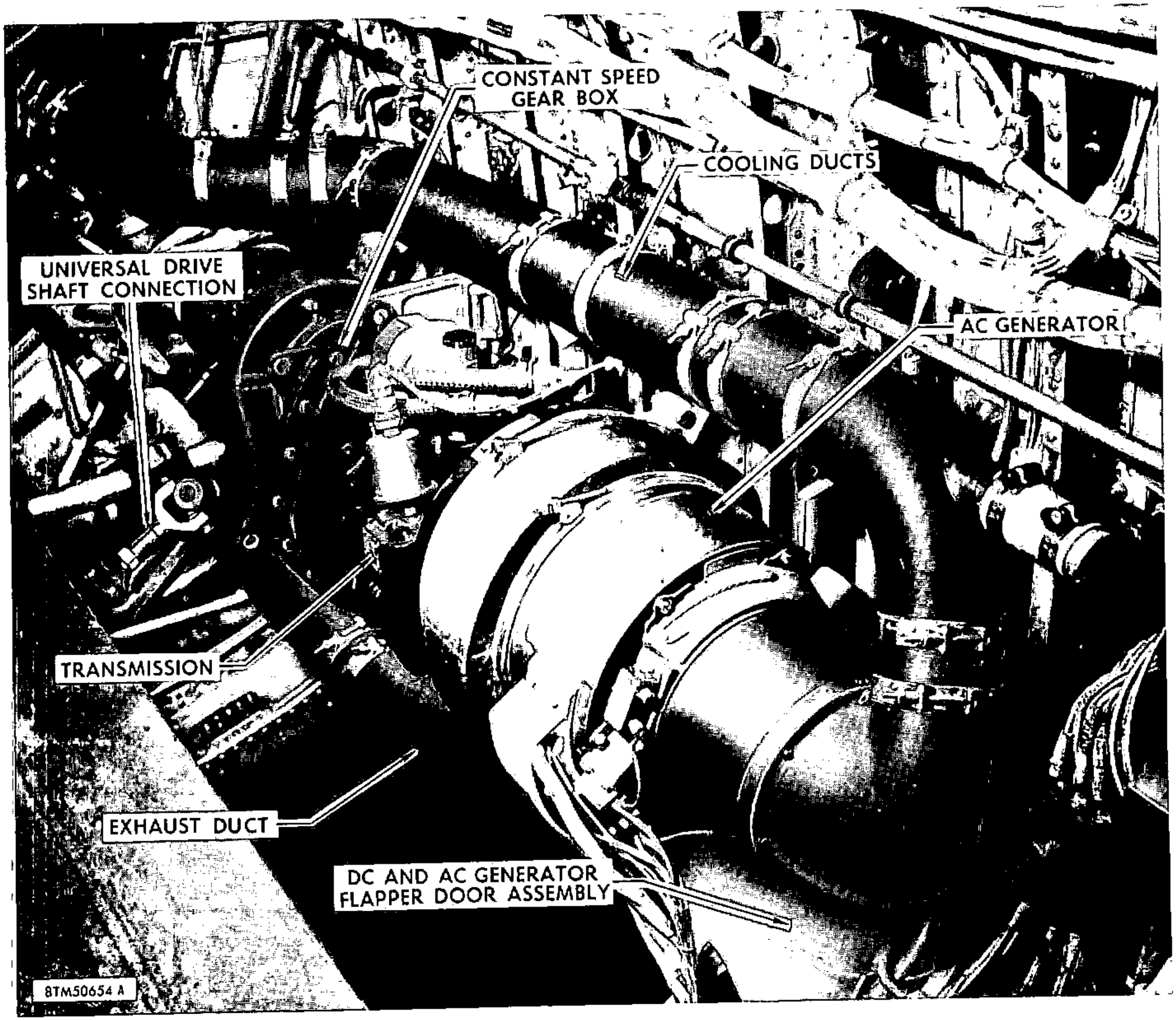


Figure 3-30. Constant-Speed Drive Assembly

The remainder of this chapter is devoted to the actual components and how they are wired into the F-102A d-c electrical power system, from the sources of power to the essential and non-essential buses; and an analysis of the system as a whole. You have been told how the electrical components work and why they are put into a system, and what they do when they are there. As we stated at the beginning of Chapter II, the best power system for any airplane is one tailored to meet the particular demands of the particular airplane. It was stated also that the reliability of an airplane power system, depends on the interconnecting network which distributes the electrical current demanded by this or that circuit. The above statements are basically true, but the prime source of power, for the F-102A d-c electrical system, is mechanical in nature—the constant-speed drive assembly.

THE CONSTANT-SPEED DRIVE ASSEMBLY.

Figure 3-30 shows the constant-speed drive assembly as located in the F-102A. The constant-speed drive system, which is perhaps a better way of describing it, consists of an engine mounted gear box, a universal drive shaft, a transmission and gear box assembly, and the d-c and a-c generators.

As you can see in figure 3-31, the engine mounted gear box is driven directly from the engine. Through its gearing, the double universal drive shaft is rotated, driving the transmission and gear box assembly. This is the largest component of the system; it is mounted on the right-hand side of the airframe, outboard from and beside the engine. Removal or installation of this assembly necessitates the complete removal of the aircraft engine.

On the forward end of the transmission and gear box assembly is the d-c generator mounting pad. When the generator is mounted and the engine is turning over, the generator is driven through its drive system at an input ratio of 1.065 to 1. This means that the input from the engine to the engine mounted gear box, and through the drive shaft to the constant-speed gear box, and then to the generator, is 1.065 revolutions to one revolution of the engine. When facing the gear box mounting pad, the generator turns in a clockwise direction; its speed varies with engine speed at the ratio fixed to the engine input. The output end of the transmission also has a mounting pad. This is provided for the a-c generator which we shall discuss later on in this manual.

Oil received from the engine oil reservoir hydraulically operates and lubricates the transmission and gear box drive assembly. An inverted flight valve, installed between the top and the bottom outlet of the engine oil tank, makes the lubrication and operating oil available to the transmission and gear box assembly during both normal and inverted flight.

For additional information regarding this drive assembly, or system, your attention is called to T.O. 1F-102A-2-10, and T.O. 1F-102A-2-4. For the purpose of this supplement the above brief description and its relationship to the d-c generator is sufficient.

THE F-102A D-C GENERATOR. COOLING SYSTEM.

The F-102A d-c generator mounted on the forward end of the constant-speed drive unit gear box is air-cooled by ram air from the engine cooling air inlet duct scroll. As can be seen in figure 3-32, the air passes from the scroll and moves back through the ducting to the generator cooling air inlet ducts. The cooling air enters the generator through its air inlet housing. It then circulates through the air blast cover, past the commutator, back over the armature and out through the cooling access holes at the drive end of the assembly. Then the air exits through a short duct and flapper valve to the engine accessory area.

The Two Cooling Conditions.

There are two cooling conditions in the system. In flight, with landing gear retracted and an airspeed of 150 knots (175 mph) and above, the pressure in the engine air inlet duct is such that the flapper door is closed (detail A). Below 150 knots and with the landing gear extended, a low-pressure area exists in this duct and the spring-loaded flapper door opens (detail B). (You can find a detailed account of these two conditions in the Power Plant Supplement, Chapter II.) This allows uncontaminated air to be drawn through the flapper door which is in the side of the fuselage next to the constant-speed drive unit installation, causing what might be called a *backdraft.*

The bleed air shutoff valve opens to allow supplemental air for ground cooling, and to maintain engine speed during deceleration. As shown in figure 3-32, the bleed air shutoff valve is an electrically operated sensing valve. It opens when the landing gear is extended and closes when it is retracted. In other words, as soon as we have a reduction of ram air pressure, we have supplemental air which causes backdrafts—not only for the generator cooling system but also for engine accessory section cooling air. This is very much the same as what happens when an ordinary fan or blower is shut off. After it has stopped twirling in one direction, we have a windmill effect as it begins to reverse its direction.

THE F-102A GENERATOR.

As you have learned, power to operate the 30-volt d-c generator, which delivers the rated current of 200 amperes at minimum engine speed, is supplied from the variable speed pad on the forward end of the constant-speed drive unit. The generator is tested and retested prior to installation. If a generator is not loaded beyond its capacity rating and if simple maintenance action is taken, it will give many hours of satisfactory service.

Inspections Are Critical.

It should be clear at this point just how important a clean, secure generator is; and how vital a constant terminal voltage is to any electrical system. For this reason, periodic inspection schedules are set up to insure top performance of all components of Air Force aircraft. In the case of an aircraft such as the F-102A, which must be in a constant state of readiness to repel a sudden attack, these inspections are even more critical.

Just what are these inspections? First of all there is the preflight inspection. This inspection is performed under two categories—with electrical power on, and with electrical power off. The *postflight inspection* is carried out after the airplane has returned to the base. This inspection is also performed with electrical power *on* as well as *off*. Both are important functions in your role as a flight-line maintenance mechanic.

Periodic inspections are made after a prescribed number of flying hours. These inspections are thorough, searching inspections of the entire airplane and its systems. Another inspection category is known as *special inspection requirements.* This phase of aircraft inspection is supplemental to the requirements of preflight, postflight, and periodic inspections. These inspections fall due at regular intervals and are additions to the regular inspection schedules. For example, it is specified in the T.O. 1F-102A-6, which lists these inspection requirements, that after every eighth *periodic inspection* the following check will be made:

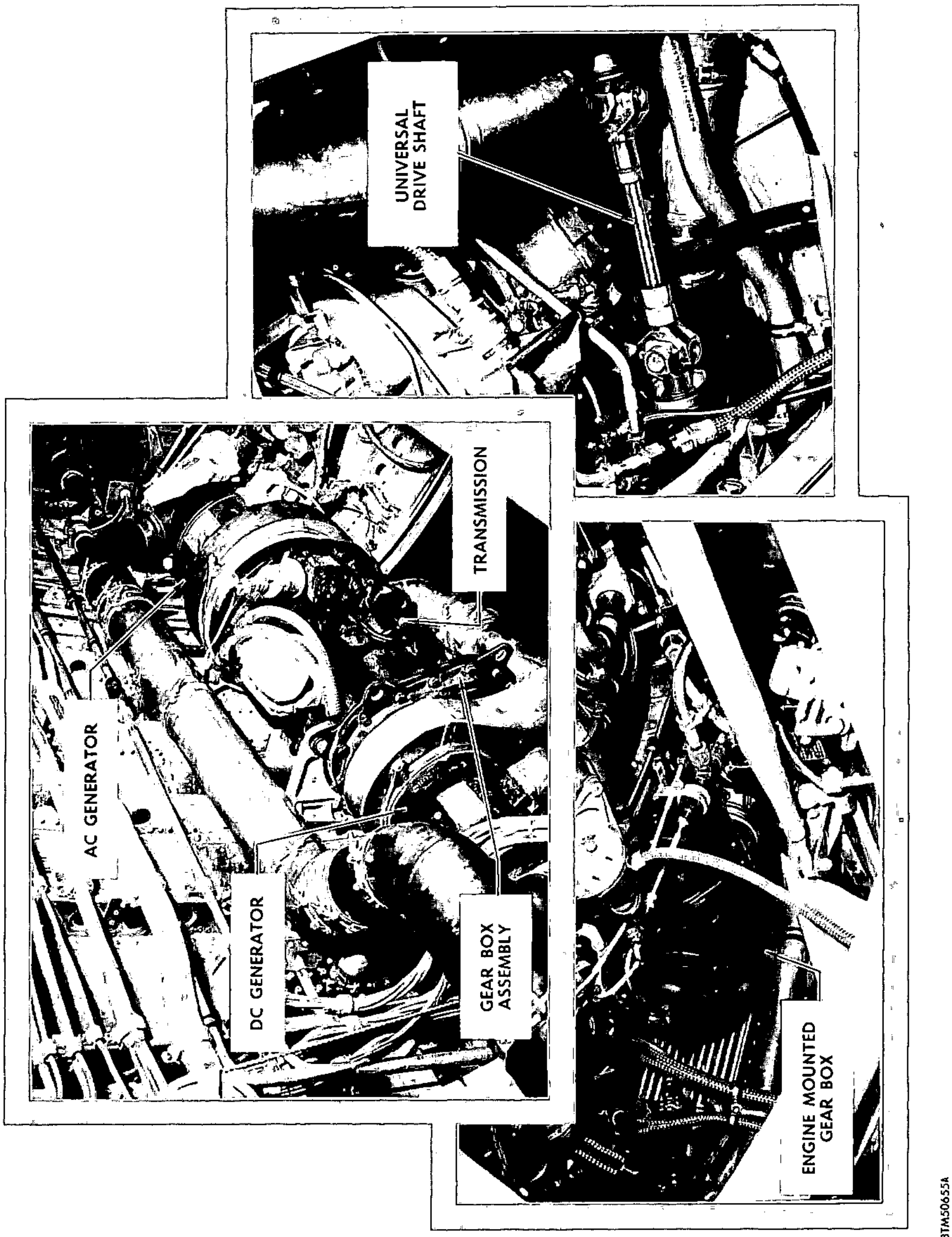


Figure 3-31. Constant-Speed Drive System

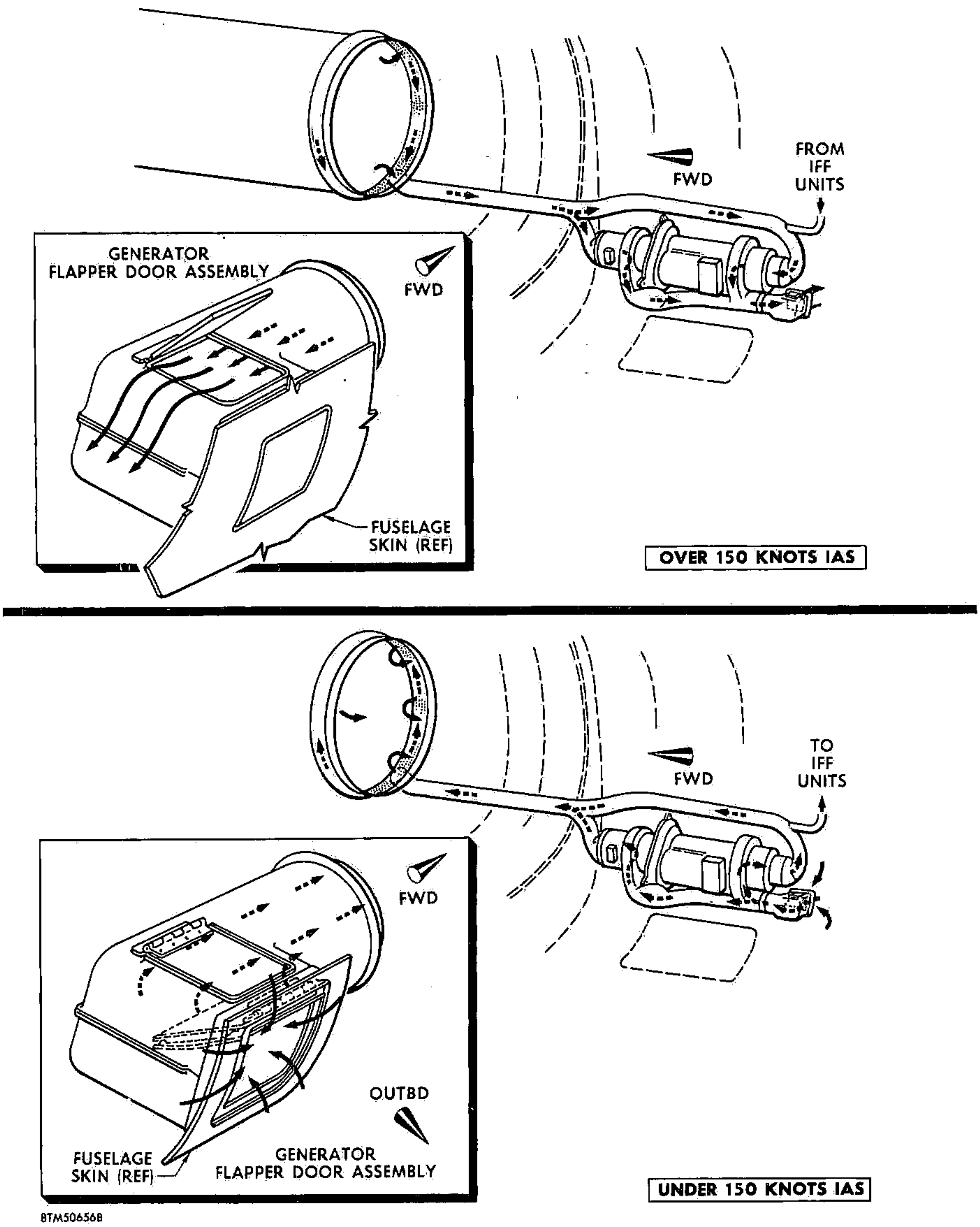


Figure 3-32. F-102A D-C Generator Cooling System

Both electrical systems (this means d-c and ac), from the generating source of power to each electrical component, must be checked for wire deterioration (chafing and fraying); specified support (harness and bundle route security); and for evidence of *overheating*. The connector plugs must be checked and their exteriors examined for security, cracks and *overheating*. Wire shielding must be checked for fraying, crimping, corrosion, and damage. The junction boxes must be checked for cracks, drainage, cleanliness and security. Plastic tubing must be checked for specified drain holes, damage, and security. Terminal strips, connections, bonding jumpers, and ground connections are checked for damage, corrosion and security. The above check *excludes* only the instrument, radio, radar, and armament wiring from the main power buses to the operating units.

The Alert Habit.

The "alert habit" is a very important and critical inspection practice and great care and patience should be used when carrying it out. While we are on this subject (and this is one of the reasons why special inspection was brought into the discussion) don't wait for these eighth, sixth, or fourth periodics to come around. Everytime you poke your head into a compartment, access door, wheel well, or the cockpit of the airplane, *keep your eyes open*. Such alertness should become a habit with you — a part of your general personality which you could well carry anywhere you go, whether you are in the nose wheel well of the F-102A, or strolling down the main drag in Hollywood, or in Kokomo, Indiana. Get the point? You'll never know what you might have missed, by looking the other way.

During each *fourth periodic inspection* of the F-102A—after the power plant is removed—the generator brush cover and the air inlet duct are removed. The inside surfaces are wiped out with a clean, lintless cloth. If you find indications of oil, grease, or dirt; clean the surfaces with a prescribed commercial grade of ether, or dry thoroughly with compressed air. *Under no circumstance* should you use carbon tetrachloride for this method of cleaning. Serious damage to the generator will result. If oil or grease is found on the generator brushes, a new set of brushes should be installed; or if the brushes are worn to the minimum length of 11/16 inches they must be replaced.

The Inspection and Replacement of Generator Brushes.

The inspection of generator brushes is extremely difficult, without engine removal. It can be accomplished, however, with the aid of mirrors and by disconnecting certain elements in and around the generator. This is a procedure which you can learn only by the actual "doing," so it will not be covered here. Actually reading or discussing maintenance action familiarizes you with the method of "doing"—it does not make you an efficient electrical maintenance man. You must "do" to make your familiarization achieve the end result; which is working knowledge and the self-confidence that comes from doing well the job at hand.

Now, let's cover the inspection and replacement of generator brushes, when there is *easy access* to the constant-speed drive assembly—(when the engine is not installed in the aircraft). Access to the d-c generator, is through the right-hand engine accessory compartment door in the belly of the plane, or from the rear of the plane through the engine compartment. At the present time, the engine is removed at each periodic, therefore, you will have easy access from both directions. The inspection of the generator brushes, as we have mentioned, is specified for each fourth periodic check. A periodic check on the F-102A is performed every 25 flying hours. This means that the generator and brushes will be checked every 100 hours of flying time. This speaks well of the modern airplane generator brush for high altitude aircraft; especially, when we think back a few years and realize that it wasn't so long ago that brushes completely disintegrated in a matter of minutes, at altitudes much lower than the present ceiling of the F-102A.

There are such things as "preventive maintenance" and "common sense." You would not have been one of those chosen to maintain a jet interceptor, if it weren't assumed that you had common sense (an instinctive sense also known as "horse sense"). Use it! It will not take you very long to check the generator, after the second or third periodic, especially the brushes. But let us say this, a component can be inspected into a malfunction—threads and air duct clamps wear, as do the ducts themselves. In other words, if the system is in good working order, the generator is secure and shows no signs of excess oil or grease (and if your "horse sense" is working), pass it by—but with your eyes open.

Let's discuss the inspection of generator brushes at the prescribed time interval (every fourth periodic). Once you have gained access to the generator and the air ducts have been removed, and the air inlet duct has been cleaned; unsnap the spring catch and remove the window strap or brush cover band assembly. This accomplished, lift the brush spring arm with a brush spring lifter tool (as shown in figure 3-33) as far as is necessary and pull the brush from its holder. Examine the brush and look for cracks near the rivets —then measure it for the minimum required length tolerance of 11/16 inches. As shown in figure 3-34, brushes are measured from the top to the longest edge of the brush seat. This is the time for a little of that "horse sense" we were speaking about.

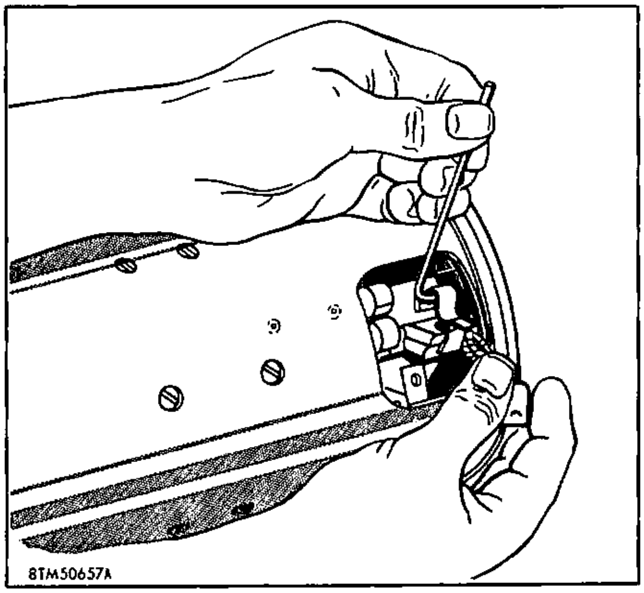


Figure 3-33. Removing F-102A Generator Brushes

This is actually a 100-hour check, and based on a periodic occurring every 25 hours. The next inspection period is 100 hours away. Let's assume the brush measures just within tolerance, and you know approximately how much it has worn since the last check; this then calls for your maintenance experience—"know-how." (Should you change it or not?) If there is any doubt in your mind, ask your crew chief, or if you are the crew chief—it's your decision. Remember, generator brushes are critical items in high-altitude aircraft. Yes, even "halide" treated brushes. What would you do? Would you refer to the applicable Air Force directive, or would you change it?

Oil and grease in a generator is liquid dynamite, for two reasons—it will form a scum on the brushes that acts as an insulator causing failure of the electrical system; and it constitutes a fire hazard, if arcing takes place.

If the commutator is found free of tell-tale burns, grease, oil, and carbon dust concentrations, or particles of carbon in the undercut mica spacing; there is no reason for you to change the generator. If however, the brushes are worn and the commutator appears dirty; replace the generator with a new or reconditioned one. This replacement procedure is given in T.O. 1F-102A-2-10.

An additional word concerning the commutator. A short circuit or open armature coil (disconnect or break) will have occasional burned commutator bars. If a whole series of consecutive bars are burned, the trouble may be due to eccentricity (out of round); oil or grease on the commutator; or it could be caused *by worn or sticky brushes,* or possibly rough bearings. In any case, if any of the above causes for malfunction are in evidence, a new generator must be installed.

When it is found that the generator brushes must be replaced, another generator should be installed. There is reason behind this statement. First of all, it is good maintenance practice, and that is what we are interested in promoting. Secondly, as you probably know, the prescribed time for generator brush run-in is four hours. By run-in, we mean the proper seating of the brushes in the direction of the commutator rotation. This run-in must take place, and the brushes must be seated 100% *before any load is applied to the generator.* Failure to comply with this will result in sparking of the unseated brushes, and the burning and pitting of the commutator. For this reason alone, brush run-in with the aircraft's engine would not be good practice. There would be too great a momentary load thrown on the generator. After all, four hours is average run-in time for proper brush seating. It takes only an hour at the most to replace a generator. In addition, to properly seat brushes the generator must be operated under motorized conditions—this cannot be done with the generator in the aircraft. This is a bench job and out of your bailiwick. So when it is found that the brushes need replacing, it is recommended that you install a new generator.

Generator Brush Spring Tension.

When you have examined the brushes and found them in good condition, replace them in their holders. Be careful in this replacement and do not allow the brush springs to snap back against the brush, since the spring tension might crack the brush. This brings up another important check of the generator, *brush-spring tension.* Brush-spring tension is accomplished by using a spring-type hand scale as shown in figure 3-35. Insert the scales hook, under the brush spring, and lift the scale until the brush spring just clears the top of the brush. The reading should be between 32 and 48 ounces, or two to three pounds. In lowering the spring to the top

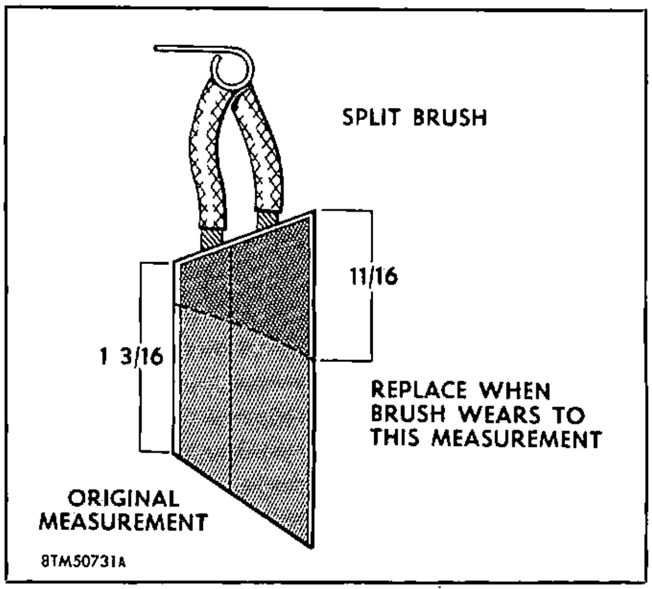


Figure 3-34. Generator Brush Measurement

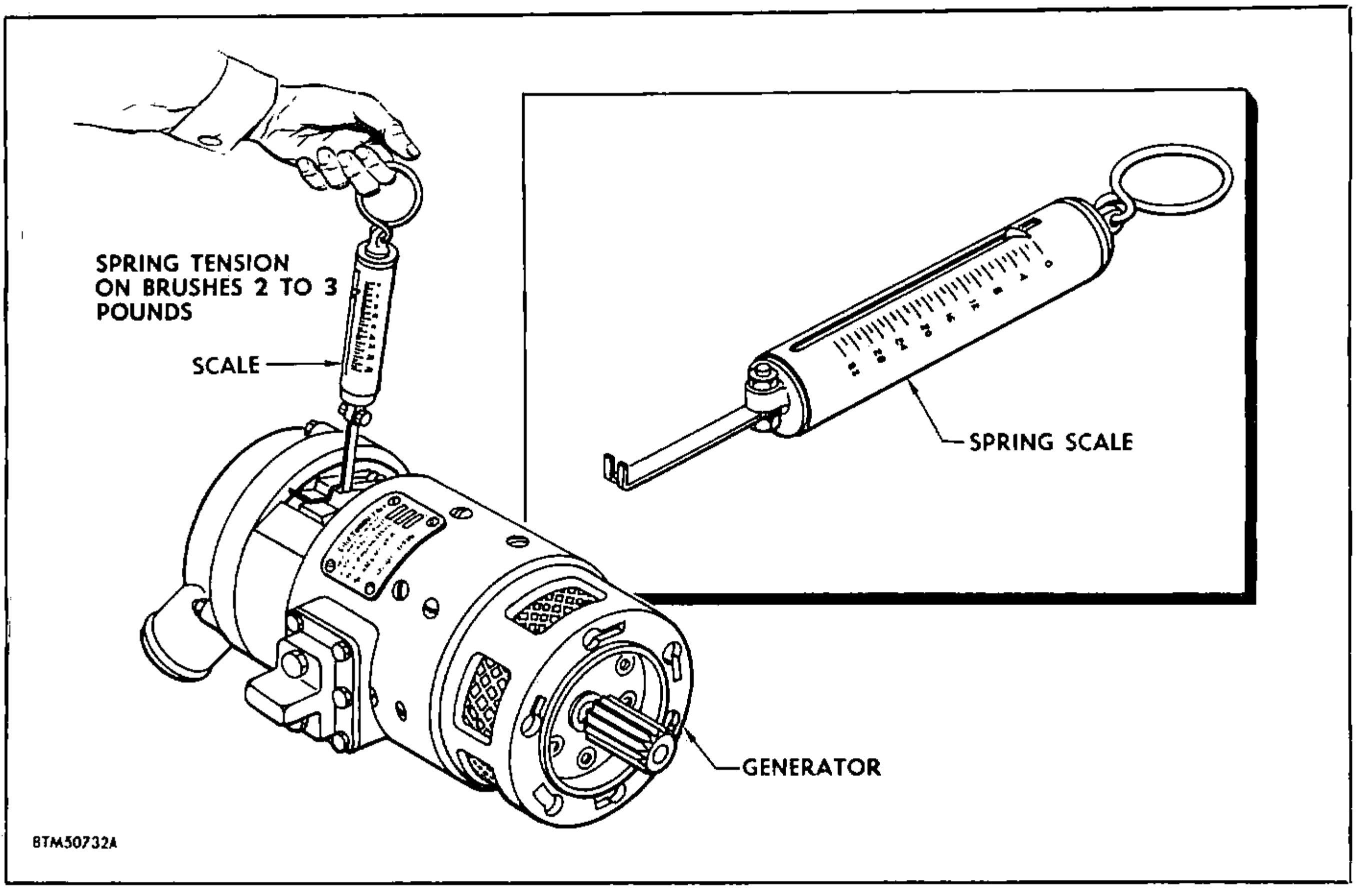


Figure 3-35. Testing Spring Tension

of the brush, do it gently. Another warning: *under no circumstance should you attempt to adjust the spring tension.* This requires the use of special tools and should only be done during generator bench-checks.

F-102A Generator Field Flashing.

Early in this chapter we mentioned the term "field flashing." In that instance, we referred to it when discussing early two-generator battery systems. A great deal has been added to "field flashing" techniques since then. Why is "field flashing" necessary? If an airplane has been parked for long time intervals, the field windings of the d-c generator may lose their residual magnetism and fail to build up the rated terminal voltage. It is also possible that reverse currents may be induced into these windings on engine shutdown; reversing the residual magnetism, so that voltage buildup in the generator will be exactly opposite to its normal polarity. In either case, the trouble is corrected by "flashing the field" with an outside source of power usually a 12-volt battery.

In the F-102A the field is flashed by placing the d-c generator switch momentarily in RESET before placing the switch to the ON position. The d-c control panel which we discussed in Chapter II contains a device whereby the field windings are automatically flashed when the generator switch is placed in RESET.

During RESET, the generator output is connected momentarily to the voltage relay coil in the differential relay until the voltage builds to 22 volts. When this voltage is reached, the correct polarity has been reached in the field windings. It is the field relay interlock which provides this trip-free reset action; this circuit is energized from the essential bus when the reset switch is actuated from the cockpit.

CARE AND ADJUSTMENT OF THE D-C CONTROL PANEL.

In Chapter II, we thoroughly discussed the d-c control panel. In the next section of this chapter, the control panel will be further discussed when we analyze the d-c circuit.

In some early models of the F-102A, the d-c control panel was removed from the airplane by first removing the safety wire from the chrome locking handle. The handle was then pulled out from the mount, and the unit slid straight forward toward the locking handle,

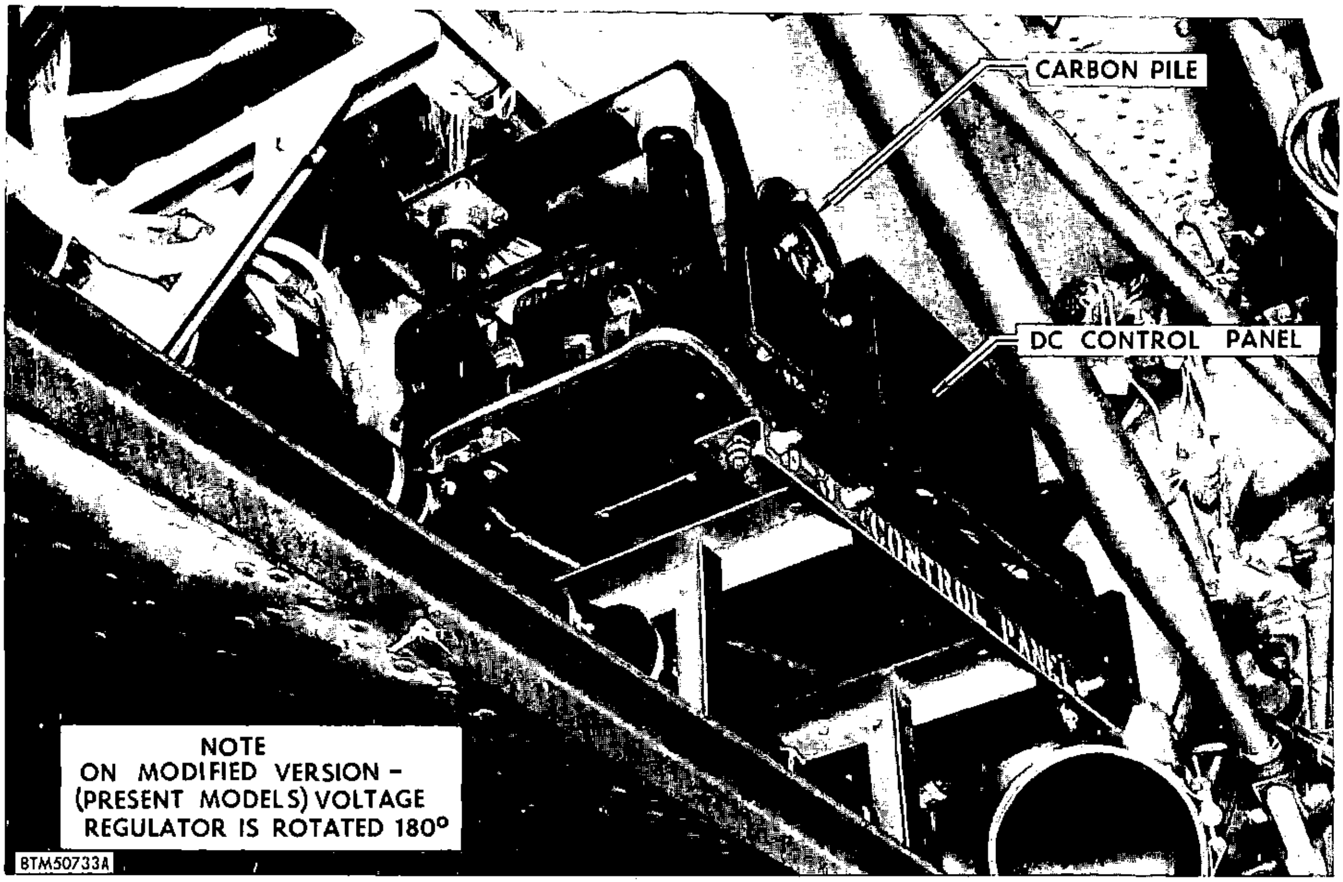


Figure 3-36. F-102A D-C Control Panels

so that it would clear the two fittings on the mount and the male fittings of the quick-disconnect plug. If by any chance you should be called upon to maintain an F-102A not yet modified, be careful not to twist or move the unit sideways on removal or on installation. To do so would either distort or bend the plug pins, or crack the insulation in the plug.

The modified version of the F-102A d-c control panel, as shown in figure 3-36, is turned 180° so that the front of the panel faces to the rear of the airplane. In this way the carbon pile voltage regulator is nearer the airplane's "cg," (center of gravity). Instead of a quick-disconnect plug, the panel is now wired directly. As you learned in Chapter II, your only concern with the control panel is inspection; to check the regulated voltage, and at times, to adjust this voltage.

The inspection of the control panel entails checking: terminals and connections for excess dirt, the cooling fins for overheating (cracking of the enamel), the shock mounts for sagging or deterioration, and last but not least the panel for security in its mounting, and its electrical connections. The T.O. 1F-102A-6 (Inspection Requirements Handbook) will give you the time interval for these inspections. But, as we have stated previously, the only efficient maintenance is preventive and continuous. Keep your eyes open.

How to Check the Voltage in the F-102A D-C Control Panel.

To check voltage, there must be a voltmeter. Therefore, a test voltmeter is connected between the test jacks on the front of the panel. The voltmeter test probes are inserted into the corresponding test jacks VJ− and VJ+ (see figure 2-36). The engine is then started and operated; the generator switch in the cockpit is actuated momentarily to RESET and then to ON, and external power is disconnected from the airplane.

The nose wheel well of the F-102, where the d-c control panel is located, is neither the roomiest of compartments, nor the most brightly lighted. So to insure an exact reading of the voltage, it is recommended that a flashlight or other lighting facility be used. Be very careful in the placement of the meter so that wires and other components in the compartment are not disturbed. During this check, operate the engine at varied speeds, from idling to normal cruising and above. The voltage should remain constant, 27.5 volts.

Voltage Adjustment.

Figure 3-37 shows a mounted control panel, an attached voltmeter, and the proper adjusting technique. The d-c control panel and its relay and carbon stack, are calibrated and adjusted before being installed in the aircraft; or if it is a case of replacement—before it comes

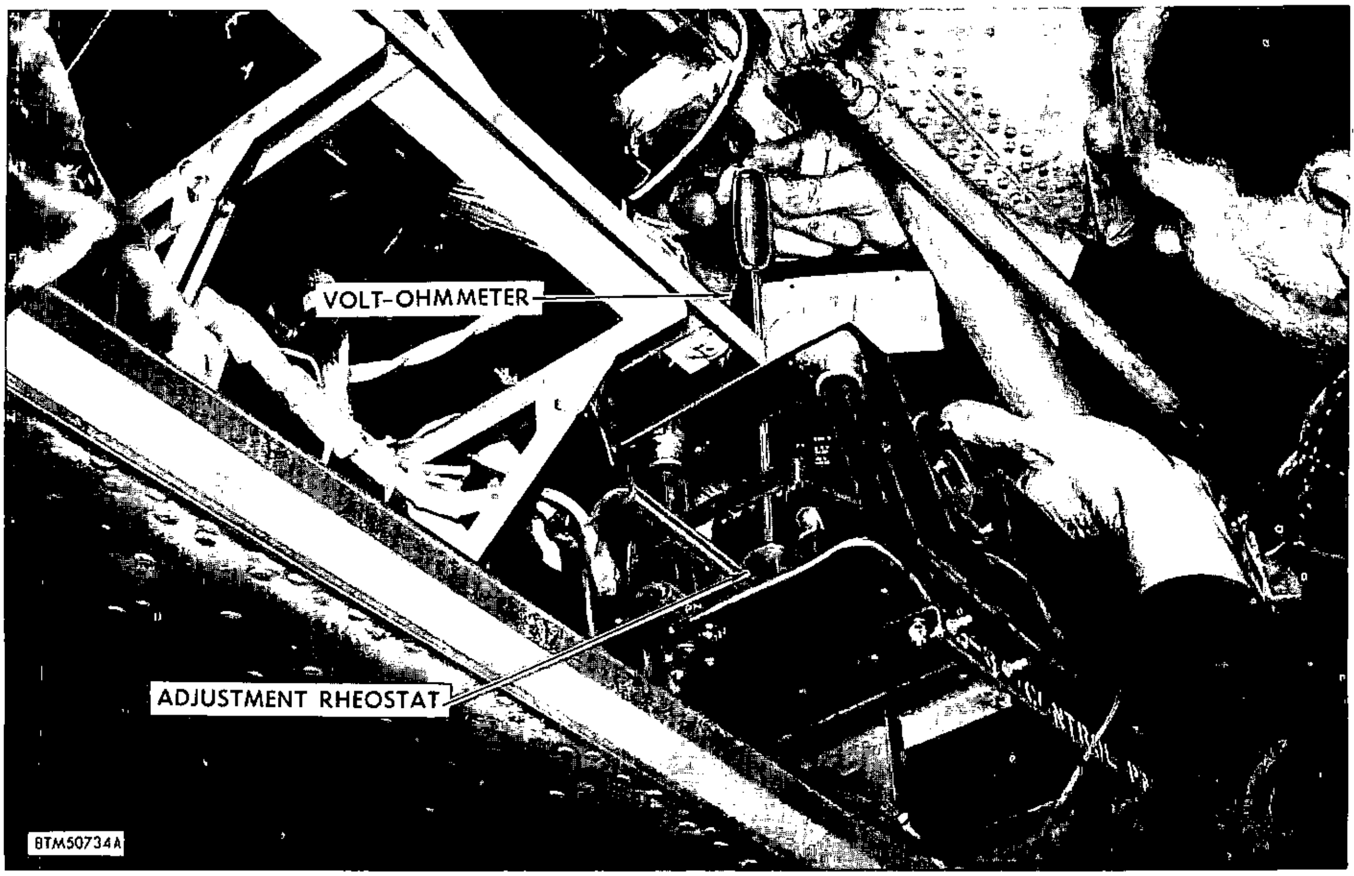


Figure 3-37. Regulating D-C Control Panel

into your hands. Let's suppose that the d-c control panel in the airplane you are maintaining has gone on a "snort" and is in need of replacement. Let us say that you have referred to the applicable directive for disconnecting and the replacement of a d-c control panel, and a new panel is now in the airplane. Your duty now concerns its adjustment. And it must be adjusted to the desired voltage of 27.5 volts.

As shown in figures 3-37, connect a test voltmeter between the test jacks, VJ— and VJ+ (the plus jack is red, the minus jack is black). With the engine turning over at normal cruising rpm, insert the screwdriver in the adjusting screw of the voltage regulation rheostat, and turn slowly in the direction of the required voltage reading. A clockwise turn will *up* the voltage reading, a counterclockwise turn will *lower* it.

When the voltmeter reads 27.5, remove the screwdriver from regulating screw. Watch the voltmeter for one or two minutes for steadiness, and then vary the speeds of the engine. The indicator reading of 27.5 should remain constant. When a new generator has been installed in the aircraft, the voltage must also be checked at the control panel and adjusted accordingly.

As a final word—caution should always be used when handling a control panel, whether it is d-c or a-c. This panel is a precision instrument, just like the precision type portable voltmeter you used to check its regulated voltage output, so be sure to handle it with the utmost care.

THE F-102A BATTERY.

The battery of the F-102A was partially covered in Chapter I, where you reviewed the theory and action of an aircraft battery—the F-102A battery or any aircraft battery. The differences encountered will be its rating, locating, and venting. As a review it might be well for you to turn back to the pages covering storage batteries in Chapter I; if there is any doubt in your mind concerning the theory, the construction, or those factors governing battery life. This, we repeat, is the blood bank of the F-102A d-c electrical system.

Venting System of the F-102A Battery.

The battery in the F-102A is located on the right-hand side of the nose wheel well. It is held in place on its support by two tie-down rods and two quick-disconnect clamps, as shown in figure 3-38. In the upper left-hand corner of the illustration is the battery as it might be seen on a shelf in the battery shack.

Outside air is vented into the fully enclosed metal case through an intake fitting, located slightly behind the

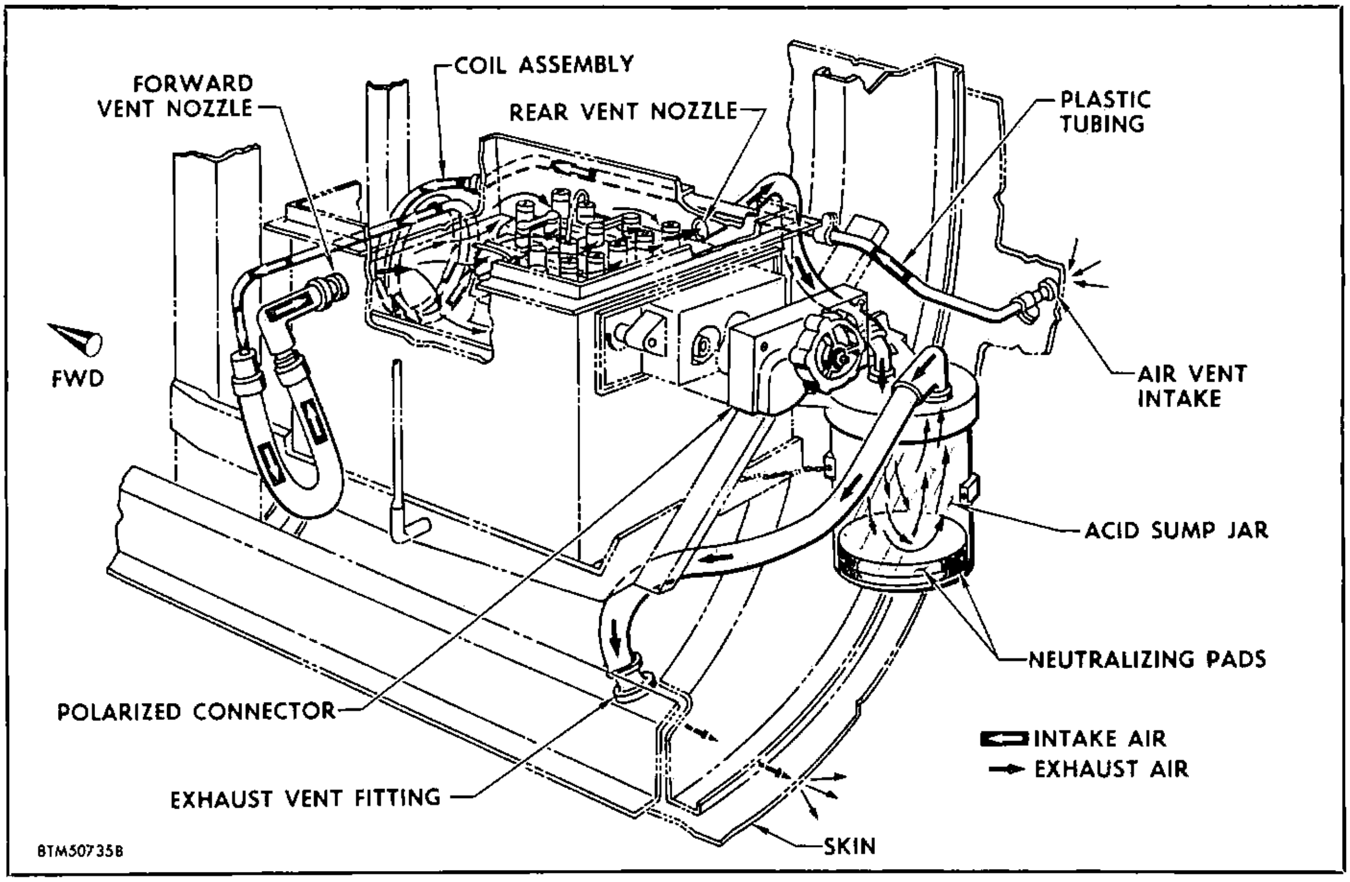


Figure 3-38. Venting System of the F-102A

battery. We mentioned in Chapter I, that the F-102A venting system is slightly different than those found on most airplanes. There is a reason for this deviation. At sonic speeds (Mach 1 and above), an ordinary venting system would not adequately take care of the pressure buildup of the high velocity outside air, and might cause serious damage to the battery. There is also the problem of moisture; the lack of it—at high altitudes; and the abundance of it—at normal flying altitudes. For these reasons—in the F-102A, the outside air is routed from the intake vent in the side of the airplane, through plastic tubing, to an aluminum coil assembly installed at the forward end of the battery. This coil assembly has a smaller circumference than the plastic tubing; and it coils two and one-half turns before the air is dumped into the much larger tube, which ventilates the battery.

Actually the coil assembly functions on the fundamental principles of an air-conditioning system—where the captured air is condensed (thus becoming hot), cooled, and conditioned for the job to be done. When the *conditioned* air has passed through the forward vent nozzle and into the metal battery case; the contaminated air inside the battery case is forced out the rear battery case nozzle, through a short tube, and into the glass acid sump jar. This sump jar contains a pad that has been saturated with sodium bicarbonate and then dried. The sodium bicarbonate pad neutralizes the dangerous gases and any spillage of electrolyte from the vent plugs on the battery proper. This air is now routed through another, shorter plastic tube and expelled into the airstream, through an exhaust fitting on the under side of the airplane. Again a word of caution—when an airplane has returned to the field and a streak of corrosion is detected aft of this exhaust vent; it is a sign of excessive voltage in the system. You must immediately check and adjust the voltage regulator and, if necessary, replace the control panel.

The Inspection and Maintenance of the F-102A Battery.

Before *each flight* the battery is inspected for security and the electrical connector for tightness. This means that you must check the tie-down rods and the two disconnect clamps which hold the battery in position. It also means, that a check must be made of the acid sump jar and the air routing tube holding clamps—for snug and secure fitting. It means, too, that the polarized connector (by which electrical connections are made to the battery), must be tested for tightness. This polarized connector is connected to the battery by turning the knob clockwise; and disconnected by turning counterclockwise. All this is accomplished in one

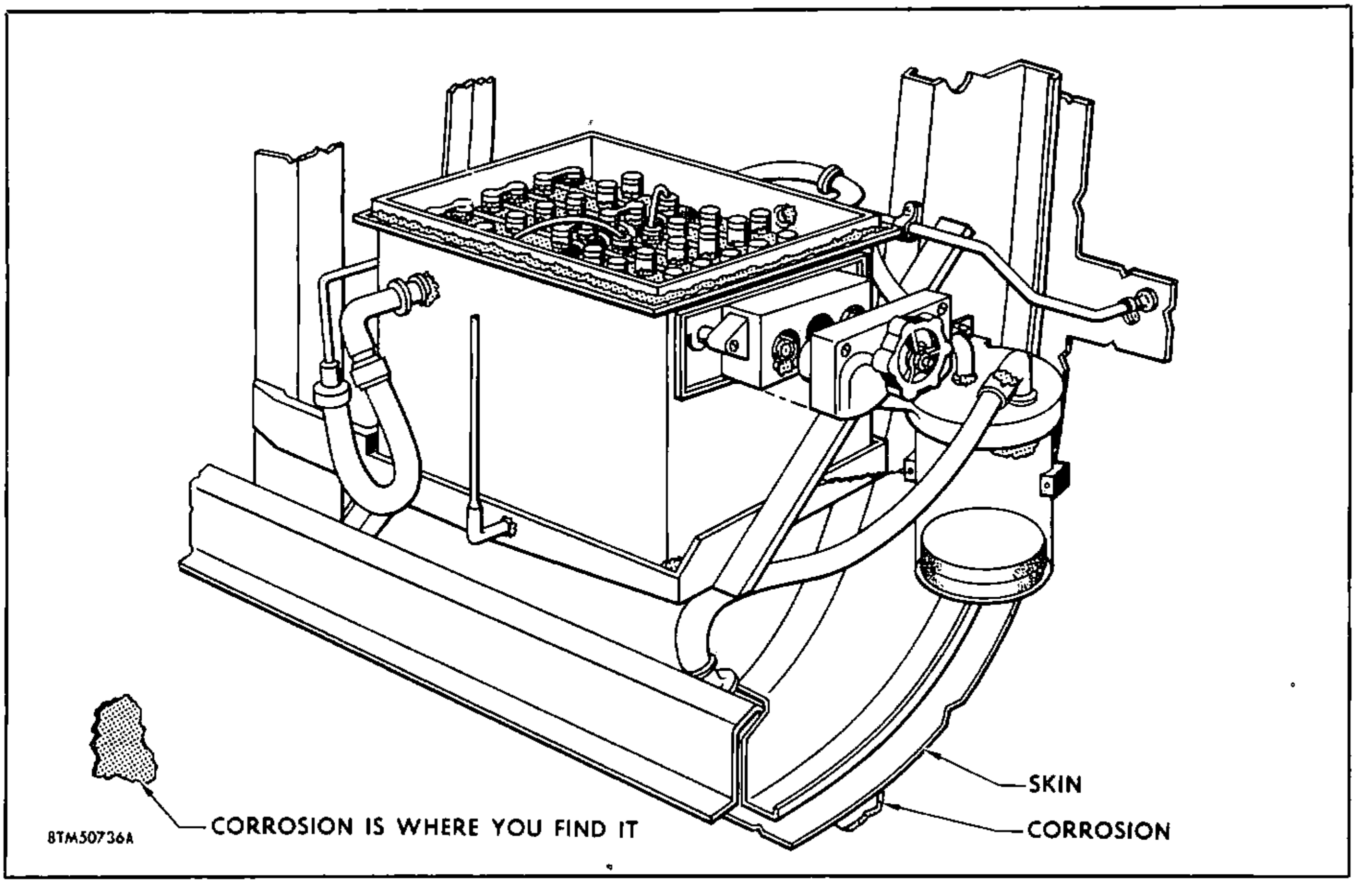


Figure 3-39. Battery Corrosion

man-minute. This means you must work fast but efficiently. There are other checks to be made. Time is precious.

When the airplane you are maintaining returns to the base, there are added battery checks to be made. Number one is for cleanliness. Battery exteriors are subject to corrosion because of the acid electrolyte they contain. For this reason you must look for any evidence of leakage or overflow of electrolyte. The critical areas are shown in figure 3-39. To accomplish this all-important examination: the acid sump is checked; the joints where the plastic tubing mates with the battery box are inspected; and then the cover is removed and the surface of the battery proper is examined for those telltale grey discolorations—*corrosion.* Finally the battery is inspected again for security.

Every seven days comes the big check when the battery cells are inspected for the proper electrolyte level and the electrolyte is checked for its specific gravity. Additional seven-day inspections are given in T.O. 1F-102A-6.

The Specific Gravity Test.

As you learned in Chapter I, the state of charge of a storage cell depends upon the condition of its active materials—primarily the plates. This state of charge is indicated by the density of the electrolyte. The density is checked by an instrument which measures the specific gravity of liquids; it is known as a *hydrometer.* If you are not familiar with a hydrometer, the drawing (figure 3-40) is typical of the one you will be using on the "line."

THE HYDROMETER. The aircraft battery hydrometer syringe consists of a small sealed float-type glass tube, weighted at one end and with a calibrated scale along its side, from which the specific gravity readings can be read. The scale usually ranges from 1.175 to 1.325 and looks much the same as those found on a common everyday thermometer. The weighted end of the hydrometer float is calibrated in relation to the known densities of electrolyte under most conditions.

READING THE HYDROMETER. When electrolyte is drawn into the syringe, the weight holds the hydrometer float perpendicular and at the correct density level. The scale value indicates (at the liquid level of the electrolyte), the specific gravity and thus the state of charge in the cell. The denser the electrolyte is, the higher the hydrometer float will rise, and the brighter will be the charge of the battery. The highest number on the scale (1.325) is therefore at the lower end of the scale.

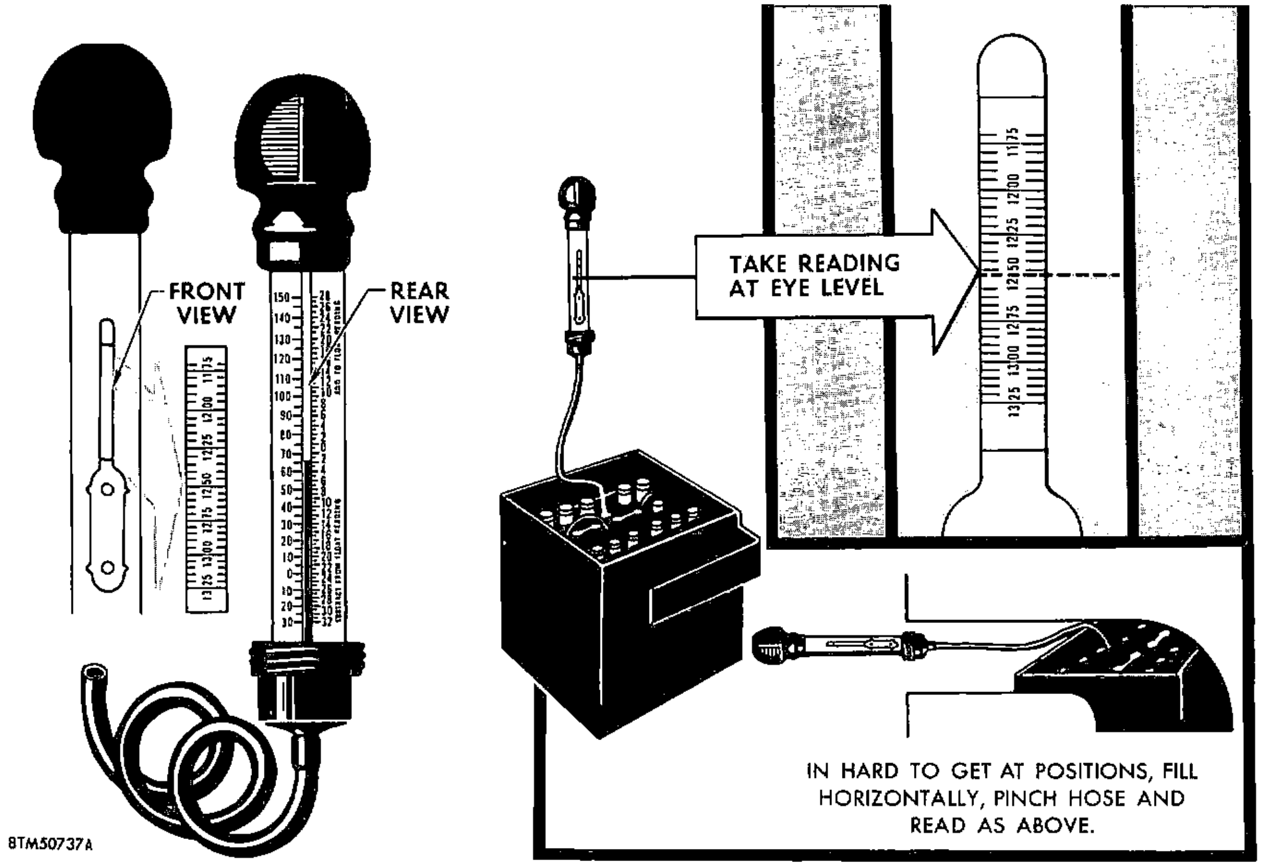


Figure 3-40. Battery Hydrometer Syringe

In a new, fully charged storage battery, the electrolyte is approximately 30% acid and 70% water (by volume) which is 1.300 times as heavy as pure water. During discharge this solution becomes lower in density and its specific gravity drops below 1.300. Specific gravity (in case you have forgotten some of your high school physics) is simply a comparison of a body's density (or weight), as compared with water. A body is merely a definite piece of matter—a magnet, a quart of water or an atom of oxygen. A specific gravity reading between 1.300 and 1.275 indicates a high state of charge; 1.275 to 1.240 a medium state; and 1.240 to 1.200 a low state of charge. Therefore the hydrometer reading in each of the 12 cells of the F-102A battery should be 1.275 to 1.300. If the hydrometer indicates 1.240 or below in any cell, the battery must be recharged. This means a replacement, as the charging of batteries is out of your present sphere of endeavor.

HOW TO TEST FOR SPECIFIC GRAVITY. First you must unlock the overcenter tie-rod fasteners and remove the battery case cover. Next, you remove the non-spilling vent plugs, being careful they are in the right sequence for replacement. For example, if you start at the forward end of the battery, the plugs should be replaced in that order, with the same plug that you removed from that particular hole. This is simply good maintenance practice. Another plug may screw into that particular opening but even threaded holes have their peculiarities and tolerances, so to be on the safe side, re-install them in the same order they were removed.

To take a hydrometer reading, insert the nozzle of an approved temperature-corrected hydrometer syringe (shown in figure 3-41) through the opening and into the electrolyte. Squeeze the bulb and release it slowly, to allow enough electrolyte into the syringe glass to float the hydrometer freely. Withdraw the syringe, and holding it vertically, check the reading on the thermometer-like stem of the hydrometer float. As a precautionary note—just as you must return the vent plug to the same access hole it came out of—always return the electrolyte to the same cell from which it was originally taken. This is good maintenance practice and at the same time removes the danger of one cell having a greater electrolyte content than another. A loss of electrolyte from a cell could lower the specific gravity of that particular cell. The battery hydrometer syringe should be washed and cleaned often. *Dirty hydrometers read inaccurately*. If any electrolyte is spilled during a specific gravity test, you must neutralize it immediately with a solution of bicarbonate of soda (sodium bicarbonate).

The Level of Electrolyte.

From time to time it may be necessary for you to add distilled water to a cell to bring up the electrolyte to its desired level. This is accomplished with a self-leveling syringe. Figure 3-42 shows how a self-leveling syringe should be used. The self-leveling syringe is filled with water and inserted into the cell. When using the syringe, be sure to hold it in a vertical position as shown (regardless of battery level) and fill the cell. Then withdraw the excess water back into the syringe until air is sucked in. This leaves the electrolyte at the proper level. In an emergency, and only in an emergency, ordinary tap water or clean drinking water may be substituted.

The normal level of electrolyte is approximately ⅜ inch above the tops of the separators, or level with the bottom of the tube which extends down from the vent opening in the cell cover. *Under no circumstances* should you allow the electrolyte level to fall below the tops of the plates. By the same token, extreme care must be taken to prevent overfilling of the cells. When the aircraft is operating in freezing temperatures, and this is extremely applicable to the F-102A, the battery should be charged immediately after the addition of any water. A 45-minute charge is sufficient to mix the added water with the electrolyte. *If charging is impossible at the moment, do not add water or it will freeze.*

Figure 3-41. Reading the Hydrometer

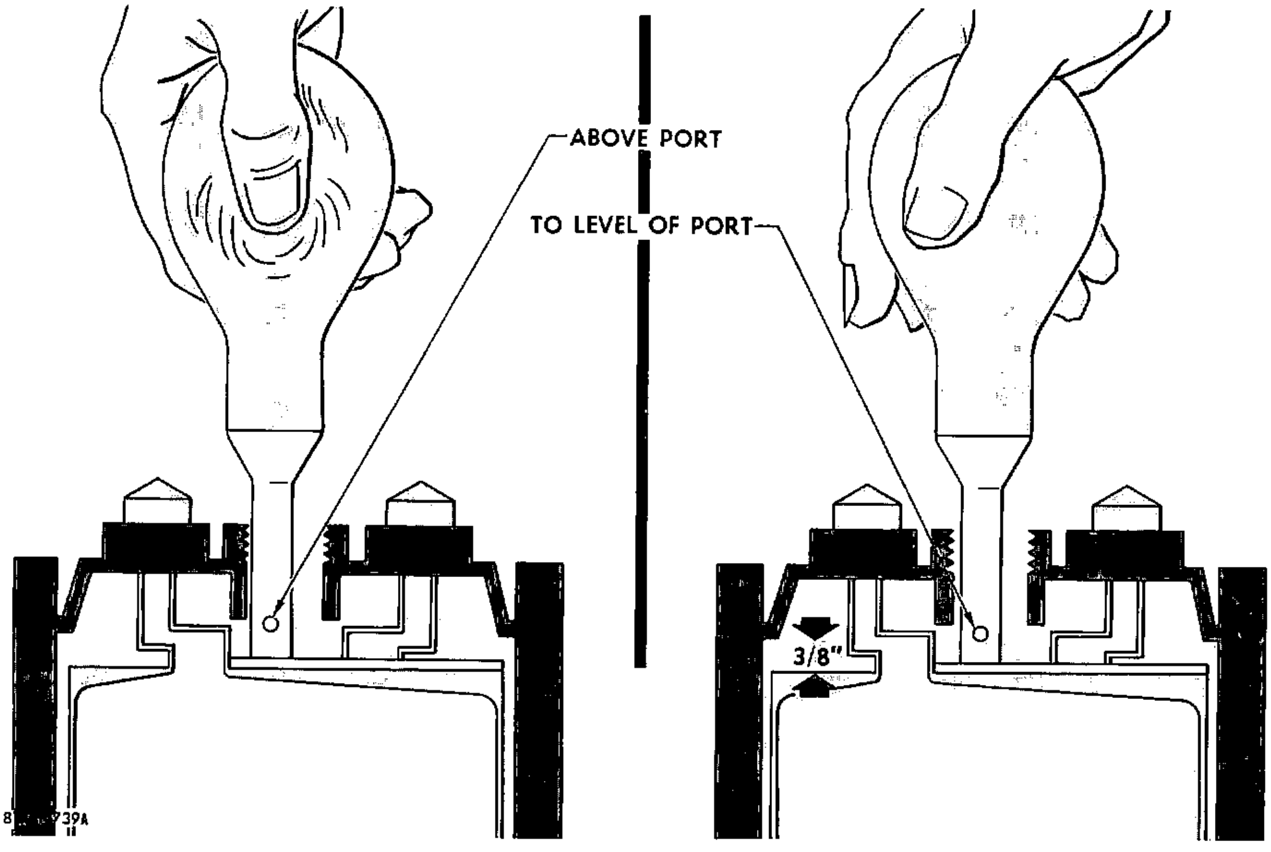


Figure 3-42. Using the Self-Leveling Syringe

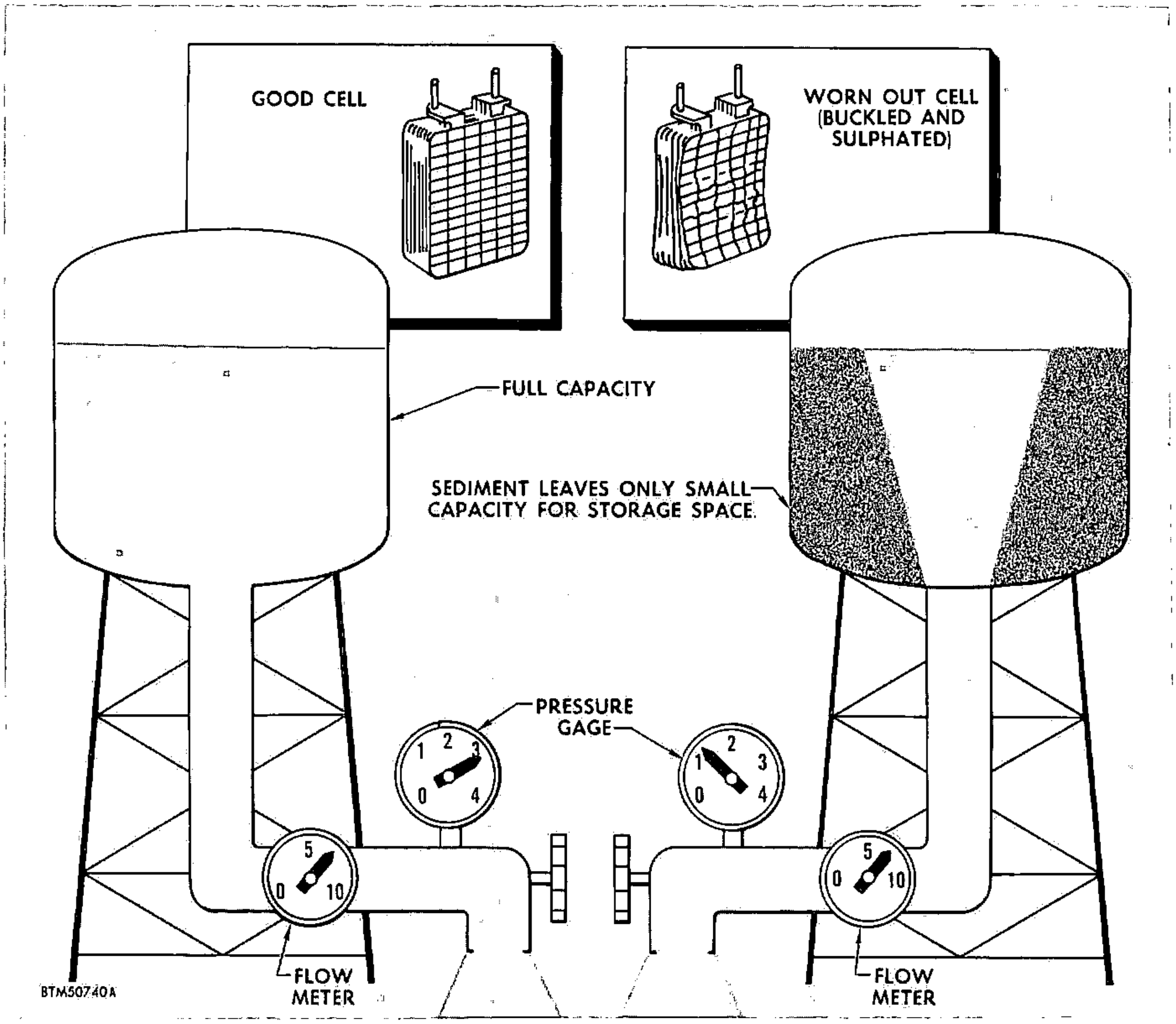


Figure 3-43. Battery Capacity Decreases With Age

A look at a freezing temperature chart for electrolyte might be helpful to you. This chart is given in T.O. 1F-102A-2-10.

Removal of the F-102A Battery.

The battery in the F-102A is removed from the aircraft every four months for a capacity inspection. This check is run in the battery shop and is actually a remaining-life-expectancy check. As you learned in Chapter I, batteries are rated by ampere-hour capacity. This capacity decreases with age.

CAPACITY CHECK. The battery in the F-102A has a 24-ampere-hour capacity. This means that the particular battery will, in theory, furnish 24 amperes continuously for five hours before it is discharged. While in all probability, you will not be called upon to run a capacity check, you should understand why it is so necessary.

Battery capacity and the condition of its charge, however, should not be confused. The capacity of a cell, as we have already mentioned, depends on the amount of useful active material on the plates during normal operation. If half the active material has been shed, the battery is equal to only ½ its original size. In this way a 24-ampere-hour battery becomes only a 12-ampere-hour battery. This capacity is inadequate to handle the job; so the battery must be replaced. The shedding we spoke of occurs slowly at first and then more rapidly as the cell, with its reduced capacity, is called upon to handle the same load as it did when it actually had a 24-ampere-hour capacity. Figure 3-43 will help you to understand this reduction in capacity. Mathematically, ampere-hours is the product of the electrical current in amperes, multiplied by the discharge time in hours. Let's take the example of a typical heavy-duty airplane battery which has a capacity at the 5-hour rate of

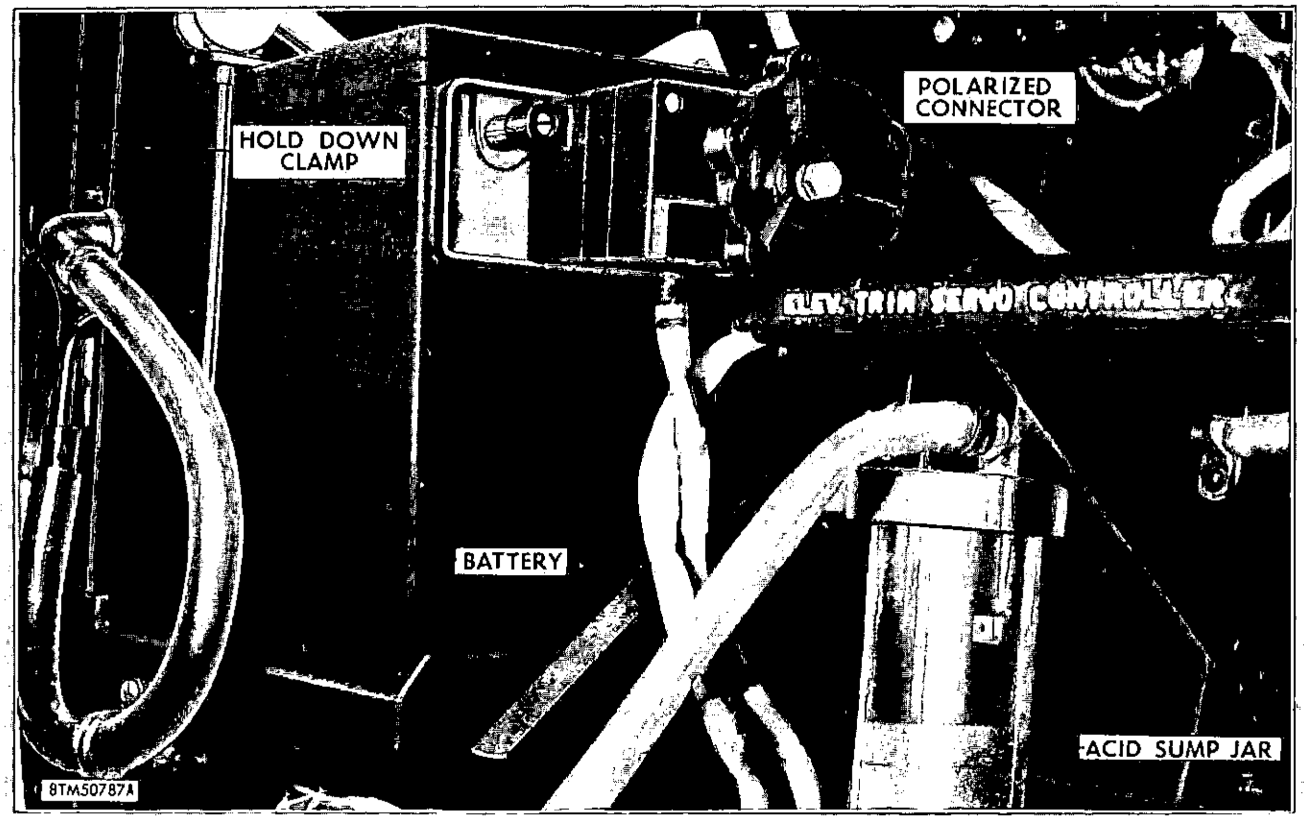


Figure 3-44. The F-102A Battery Installed

88 amperes. By dividing five into 88, we arrive at a product of 17.6 amperes. This means that this battery will operate continuously for five hours and put out 17.6 amperes; or for 1 hour, 88 amperes. Although current may be obtained at the end of this time interval discharge period, the voltage output may have dropped to a point beyond which the battery's future usefulness is very much in doubt. So from time to time the battery must be checked for its capacity.

ELECTROLYTE SPILLAGE. Another reason for removing the battery from the airplane is in case of an excess spillage of electrolyte. If this occurs the battery is returned to the battery shop to be refilled. The battery is then replaced by a new or fully charged battery which you have picked up in exchange. The battery must also be removed for cleaning. Figure 3-44 shows the F-102A battery as it is located in the airplane. For removal and installation procedures, see T.O. 1F-102A-2-10.

CLEANING THE F-102A BATTERY. As you know, battery exteriors are subject to corrosion, and as mentioned earlier in this chapter, corrosion is the electrochemical destruction of metal—*ionization*. Corrosion is removed from the battery by brushing with a stiff bristled brush. Do not use a wire brush as this will injure the casing. During this cleaning process, the vent plugs must be in place. The reason for this should be clear enough. When the corrosion has been brushed clean, the battery is washed with a solution of sodium bicarbonate and warm water—one pound of sodium bicarbonate to one gallon of water. This neutralizes any remaining electrolyte.

The battery is then rinsed off with clear warm water. At this time, the electrical connections, the vent plugs, the battery rack, vent lines, and the acid sump jar are also checked for cleanliness, and if dirty, they are cleaned. The examination of the non-spill vent plugs is especially critical. Make doubly sure the gas escape holes are unobstructed.

REMOVAL AND CLEANING OF THE BATTERY SUMP.

When removing the sump jar, take every precaution against spilling any acid it may contain. To remove the jar, remove the safety wire and unscrew the jar. Figure 3-45 shows this procedure. If you find acid in the sump, clean it out and dry it thoroughly. Remove the old pad and replace it. Remember that the pads in an acid sump jar have been saturated with sodium bicarbonate and then allowed to dry. Make very sure this is true of the ones you replace.

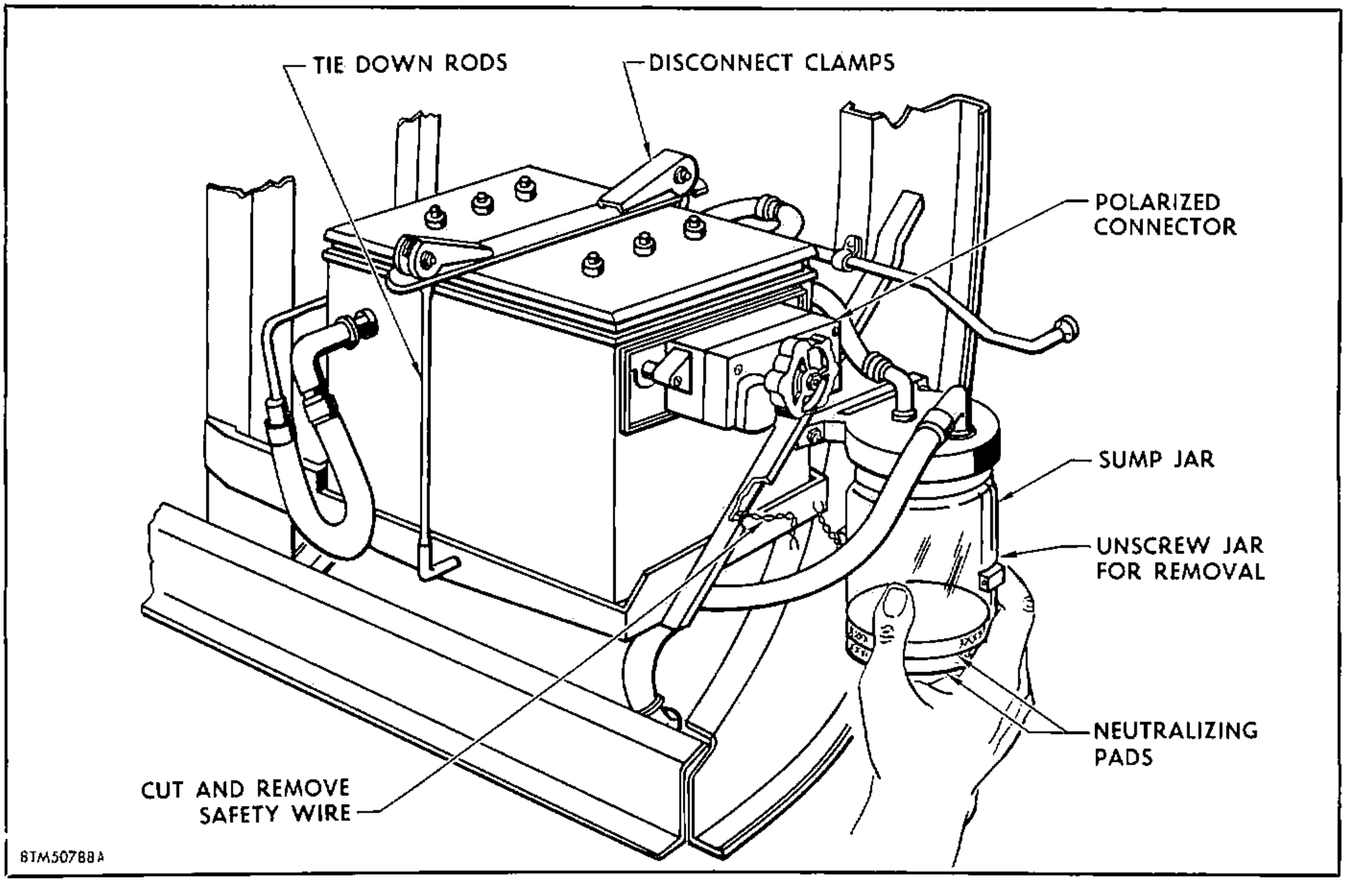


Figure 3-45. Removal of Battery Sump Installation

A more complete explanation of the storage battery, its maintenance and care is given in T.O. 8D2-1-31, which deals comprehensively with all aircraft storage batteries and their venting systems.

D-C POWER LOADING IN THE F-102A.

From time to time, a great deal has been said in this manual concerning "load." Load merely means the sum total of the current drain of all pieces of the aircraft electrical equipment installed in the airplane for a given period of operation under various conditions. The first step in the development of any electrical system, is to establish the electrical loads which will be applied to the system. The power demanded by the load, is the next consideration.

In the case of the F-102A, the power supplying the d-c load is provided by a 200-ampere, 30-volt generator while in flight, and by a 24-ampere-hour storage battery during emergency operations—when the load is considerably less due to the automatic disconnect from the non-essential bus. But bear in mind that even with this lessening of the demanded load of the airplane, the battery's output capability is reduced to a matter of minutes. For example: the load on the generator in the F-102A for ½ minute at normal cruising speed is 168.7 amperes; during an emergency, the d-c system demands only 73.7 amperes for the same estimated cruising speed and for the same time interval. See figure 3-46.

Figure 3-47 shows the major components in the d-c electrical system; the essential and non-essential bus bars, the systems connected to those bus bars, and the current demanded by those systems during normal operations. The current consumption of the individual components comprising the F-102A's functional electrical system is given in T. O. 1F-102A-2-10.

LOAD ESTIMATES UNDER OPERATING CONDITIONS.

The load estimate for an airplane is based on the maximum continuous load during flight. The electrical power consumed, of course, is computed in amperes. These computations are the loads on the generator, or battery, predicated on the systems which are *expected* to be in operation under certain conditions such as: standby (start and warmup), taxi, takeoff and climb, cruise, combat, descent and landing, and emergency. The load values are the average for a given time interval of operation. For example, the amperes the F-102A will use,

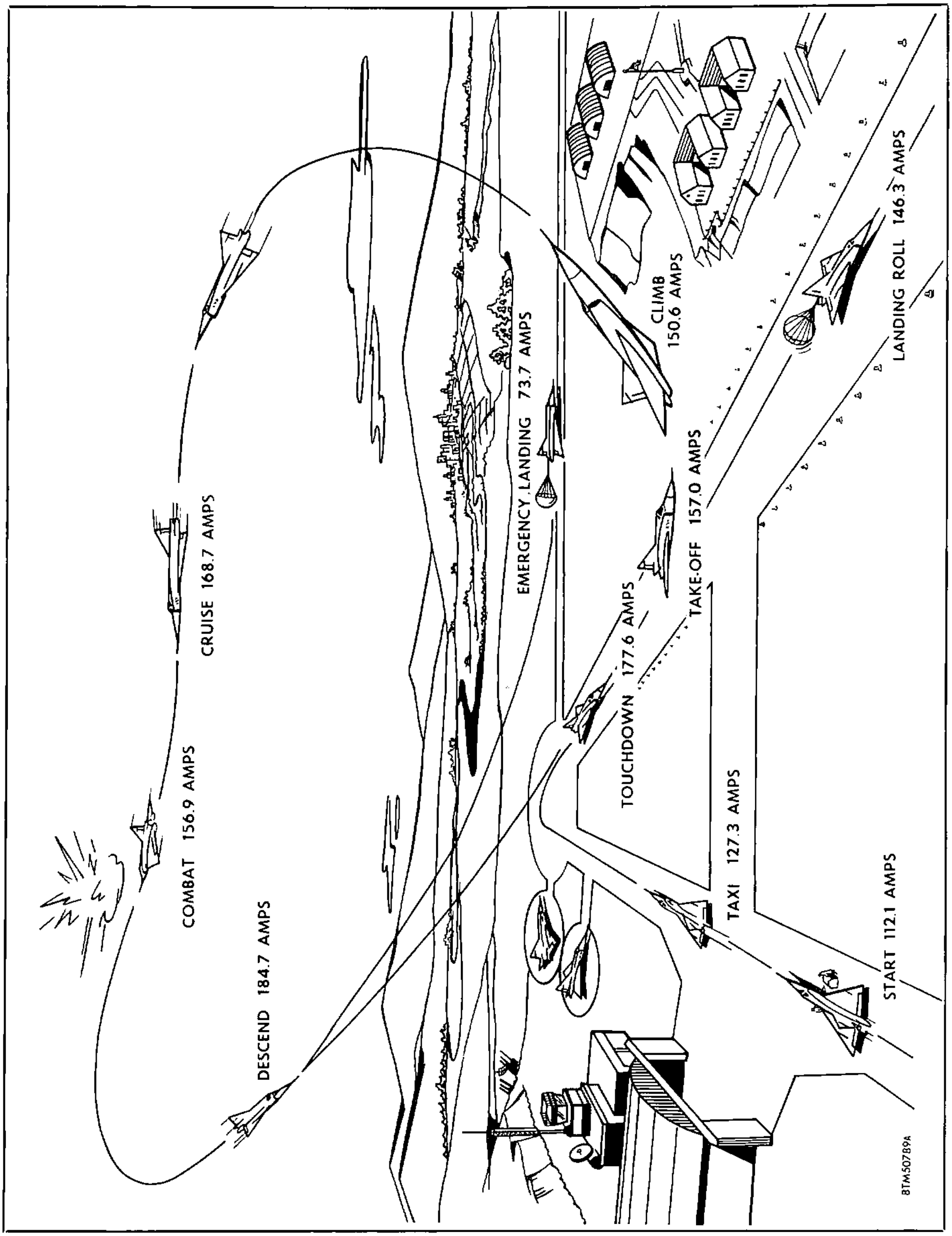


Figure 3-46. Power Loading on the Ground and in the Air

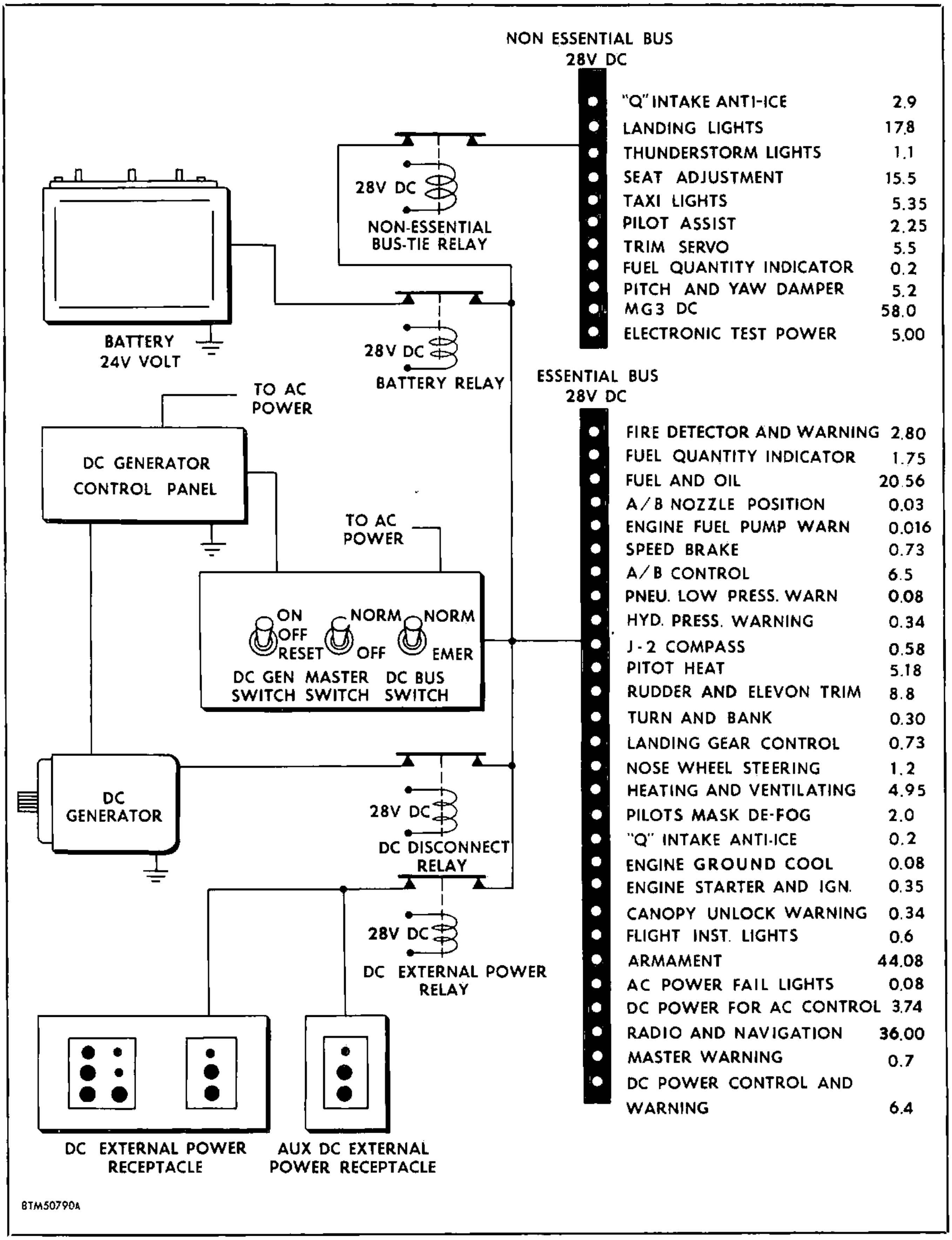


Figure 3-47. The D-C System Under Load

under certain conditions, will vary as shown in the table below:

Standby (start and warmup) from:

0 to ½ minute	112.1 amperes
½ to 2 minutes	97.8 amperes
2 to 30 minutes	67.2 amperes

Taxi from:

0 to ½ minute	127.3 amperes
½ to 2 minutes	123.0 amperes

Takeoff and climb from:

0 to ½ minute	157.0 amperes
½ to 2 minutes	150.6 amperes
2 to 10 minutes	136.8 amperes

Cruising from:

0 to ½ minute	168.7 amperes
½ to 2 minutes	161.2 amperes
2 to 30 minutes	133.0 amperes

Combat from:

0 to ½ minute	156.9 amperes
½ to 2 minutes	147.3 amperes
2 to 5 minutes	140.6 amperes

Descent and landing from:

0 to ½ minute	184.7 amperes
½ to 2 minutes	177.2 amperes
2 to 10 minutes	146.3 amperes

In an emergency (non-essential bus disconnected) the maximum usable current under all conditions would be from:

0 to ½ minutes	73.7 amperes
½ to 2 minutes	62.0 amperes
2 to 30 minutes	35.3 amperes

As we have already mentioned, the above values are computed as an average for the given period of operation. During normal operations, some of the electrical components will be functioning while others will be idle; therefore, the power loads draining the d-c system will vary according to the number of electrical units being operated at any particular time.

F-102A D-C POWER CIRCUIT ANALYSIS.

Just what is meant by analysis? According to our friend Mr. Webster, it is the separation of anything into the parts which aid in making that particular thing function. Analysis, is also the examination of anything to determine its component parts; separate them from the whole "ball of wax;" and establish their relationship to one another. This last phrase "their relationship to one another" is our prime interest in this particular section of this training supplement.

Up to this point we have examined the distinguishing characteristics of the individual component parts—how they function, why they are necessary to an electrical system, and to a certain extent, where and why they are in the system. In a table of organization, as shown in figure 3-48, the chain of command is shown through a series of symbolic rectangular boxes joined together by a network of fine lines. These lines show the relationship of one level of command to another. In the same manner the components of an electrical system are organized through an interconnecting network — the circuits. The components are shown symbolically (through symbols); their relationship to one another is indicated by the interconnecting lines — a wiring network. Electrically, this organizational drawing is called a schematic.

A circuit analysis of an electrical system of a new and modern aircraft must, in its final form, be only a ghost of the end product. The reason for this is that engineers and designers are constantly using their knowledge to build toward finer and more brilliant performance in the aircraft. Performance reports arrive daily from the field. Changes—modifications—occur daily, sometimes within the hour.

So at best, until the absolute reliability of the whole is tested and retested, a circuit analysis can only be typical. Unfortunately this "absolute reliability" will never be achieved. Nothing is perfect in engineering and scientific fields, or in any other field for that matter. But a goal must be set to strive toward—thus, the expression "absolute reliability"—a reliability as nearly perfect as man is capable of achieving.

However, if you understand the basic characteristics of the individual electrical components and know their symbol designations, as shown in figure 3-49; you will have no trouble following subsequent modifications on the F-102A electrical systems. These symbols are standard. For further knowledge of them refer to the National Military Establishment Standard for Electrical and Electronic Symbols, Jan-Std-15, dated 19 October 1948.

The present analysis of the F-102A d-c electrical power system is based on earlier models of the airplane. If you understand one, you will understand the other. This analysis will cover the d-c power system from its source of power to the essential and non-essential distribution buses. The systems energized from these distribution points, are discussed and analyzed under the applicable supplements dealing with the particular system. The lighting systems in the airplane are discussed in Chapter V.

Remember, everything that has been said so far in this supplement can be applied directly to the F-102A. Every component or gadget discussed is used somewhere in some system within the airplane. Now all we have to do is simply "wire" them together and find out their relationship to one another.

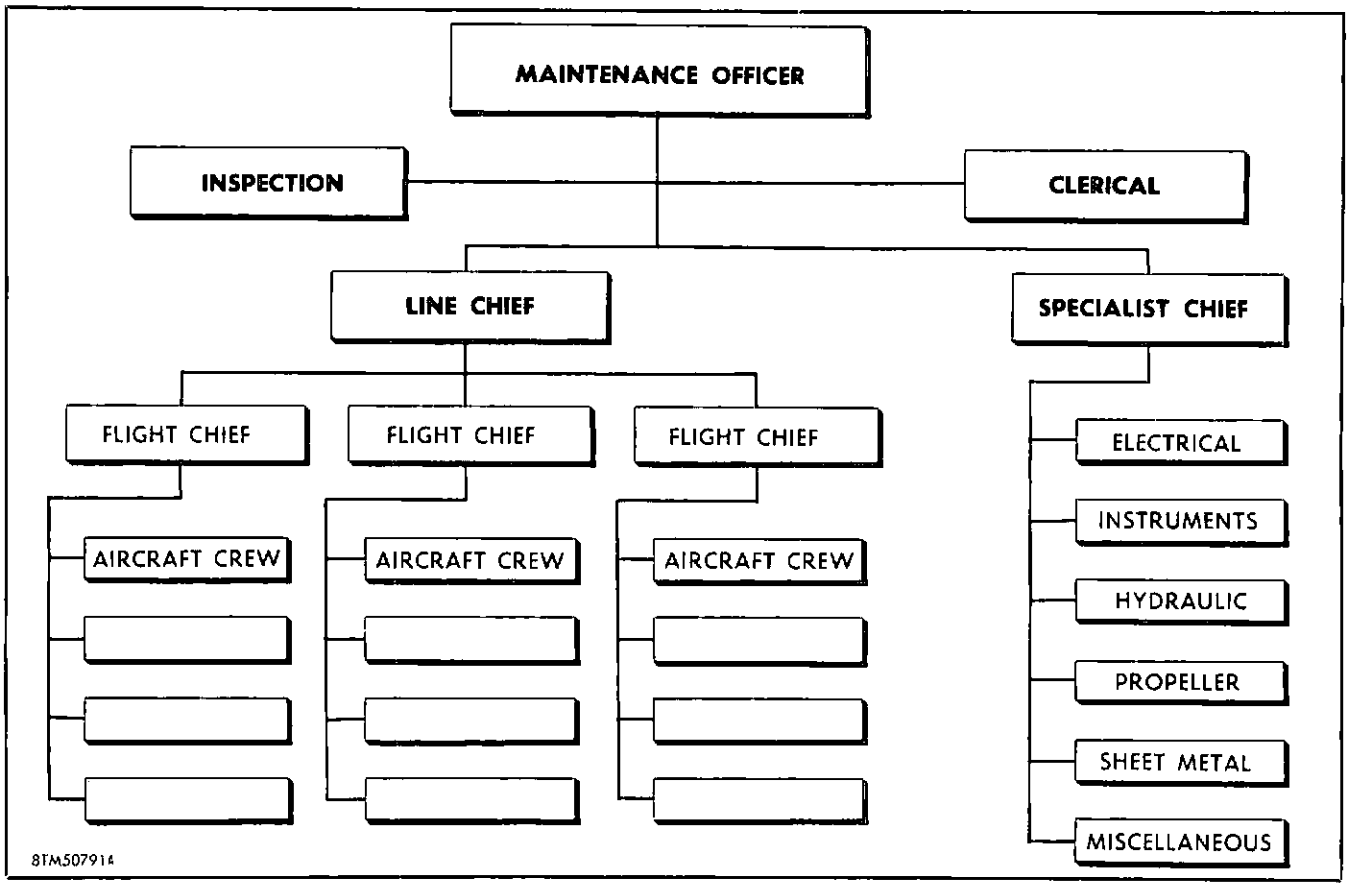


Figure 3-48. Typical Field Maintenance Organizational Chart

THE DIRECT CURRENT SYSTEM.

To bring you up to date on the F-102A direct-current system, let us review briefly what it contains. First of all, it is a conventional 28-volt ground-return system, which during normal in-flight operations is powered by a 30-volt, type G35-5, air-cooled, d-c generator, mounted on the forward end of the constant-speed drive assembly. A 24-volt, 24-ampere-hour, lead-acid storage battery supplies the power for limited ground and emergency operations. In Chapter II, we discussed the power distribution to the buses, as controlled by a generator control panel which gives the system voltage regulation through the carbon pile regulator; and overvoltage and reverse protection through relays.

The d-c external power receptacle in the main wheel well provides the system with a connection for a 28-volt ground-support power cart (used in ground operational checks and inspections). This receptacle is connected into the system through a relay; the receptacle also provides a connection for a d-c load bank.

In the d-c system all sources of power are connected to the essential and non-essential buses through relays. The power is distributed through protective circuit breakers at the buses, to other switches and relays; these control and protect the d-c powered electrical devices located throughout the airplane. As we explained in Chapter II, a d-c power failure is sensed by means of a master warning indicator light on the instrument panel; it is also sensed by the d-c power failure light on the master warning indicator panel. The master warning system is fully analyzed in the Instrument supplement of this series. The wiring schematic of the d-c power system is shown in figure 3-50, with a location diagram of the electrical components. It is suggested you refer to this diagram now for a good visual concept of the ensuing d-c system analysis.

The D-C Generator and Control Panel Wired Into the System.

The simplified diagram (figure 3-51) shows how the d-c generator and its companion control panel are wired into the system. This is the d-c system as found on earlier versions of the F-102A. Let's start with the positive power lead, B. Note that this lead is connected directly to terminal 17 of the panel. Another lead is connected through contact A2 of the d-c disconnect relay to terminal 1. (Also refer to figure 3-50 for this discussion.) Contact A1 of this relay is tied into the d-c essential bus. This circuit is protected through the d-c power control circuit breaker on the forward auxiliary circuit breaker panel in the cockpit, and completed when the d-c generator switch is actuated. The lead from terminal 15 of the control panel is

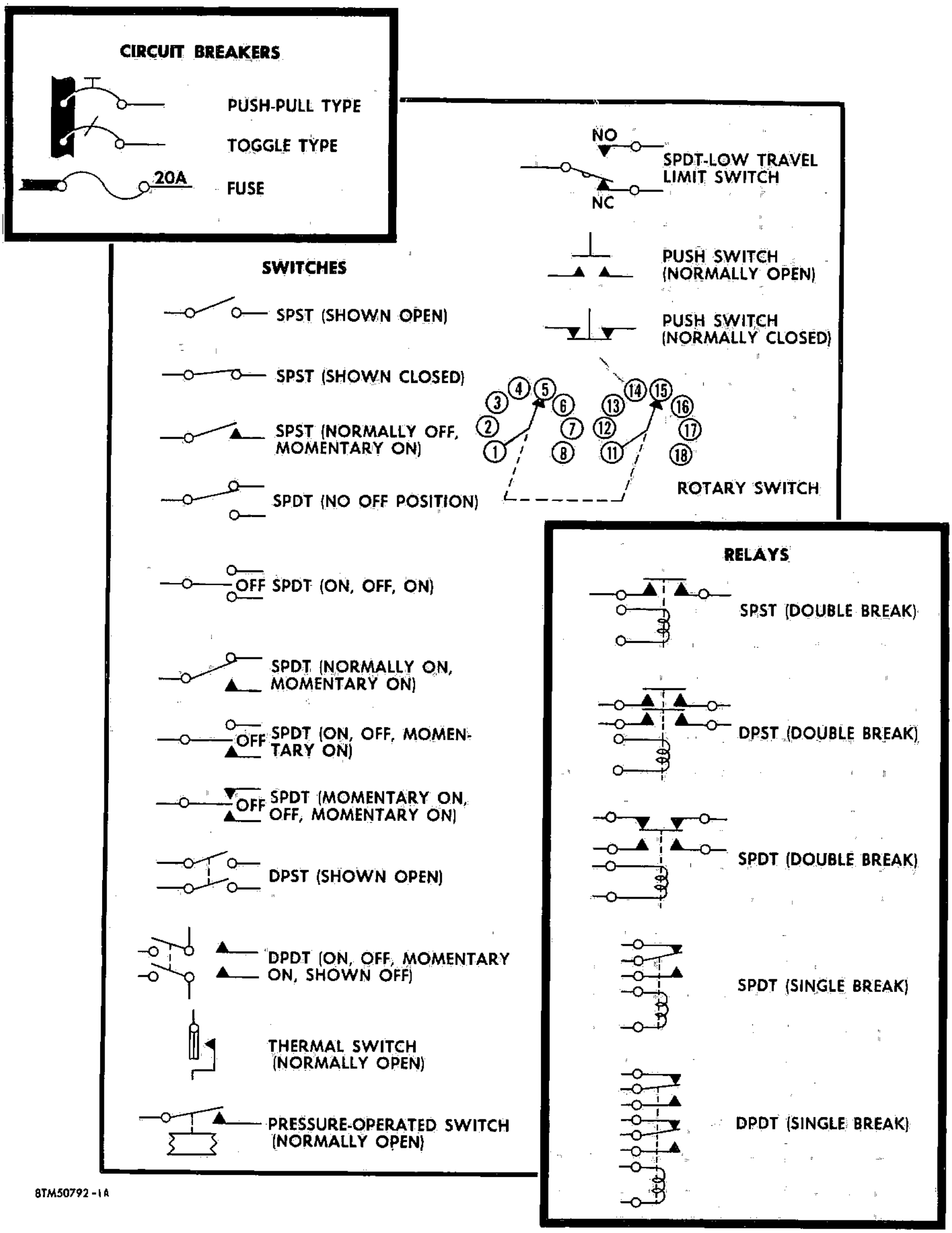


Figure 3-49. Electrical Symbol Chart (Sheet 1 of 2)

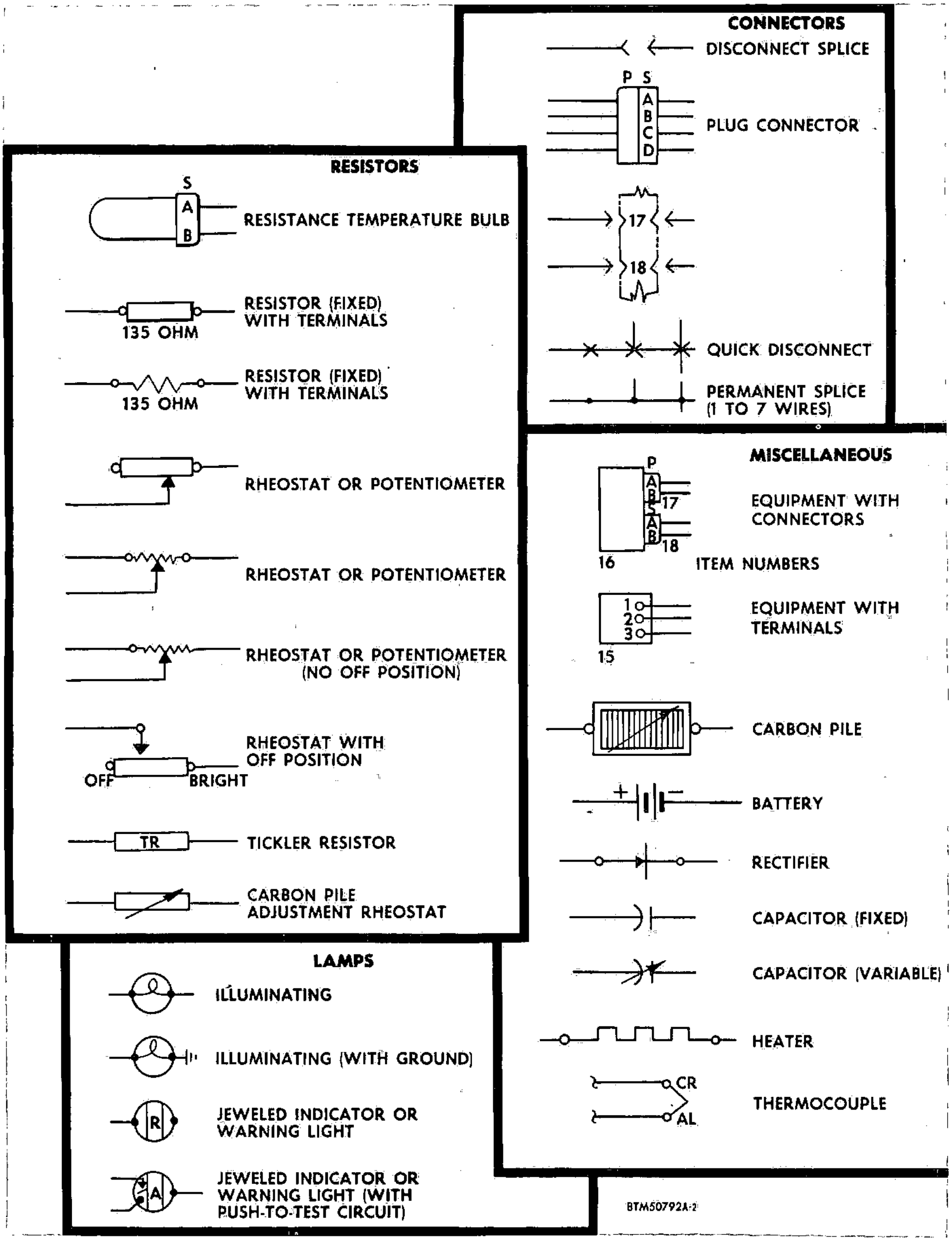


Figure 3-49. Electrical Symbol Chart (Sheet 2 of 2)

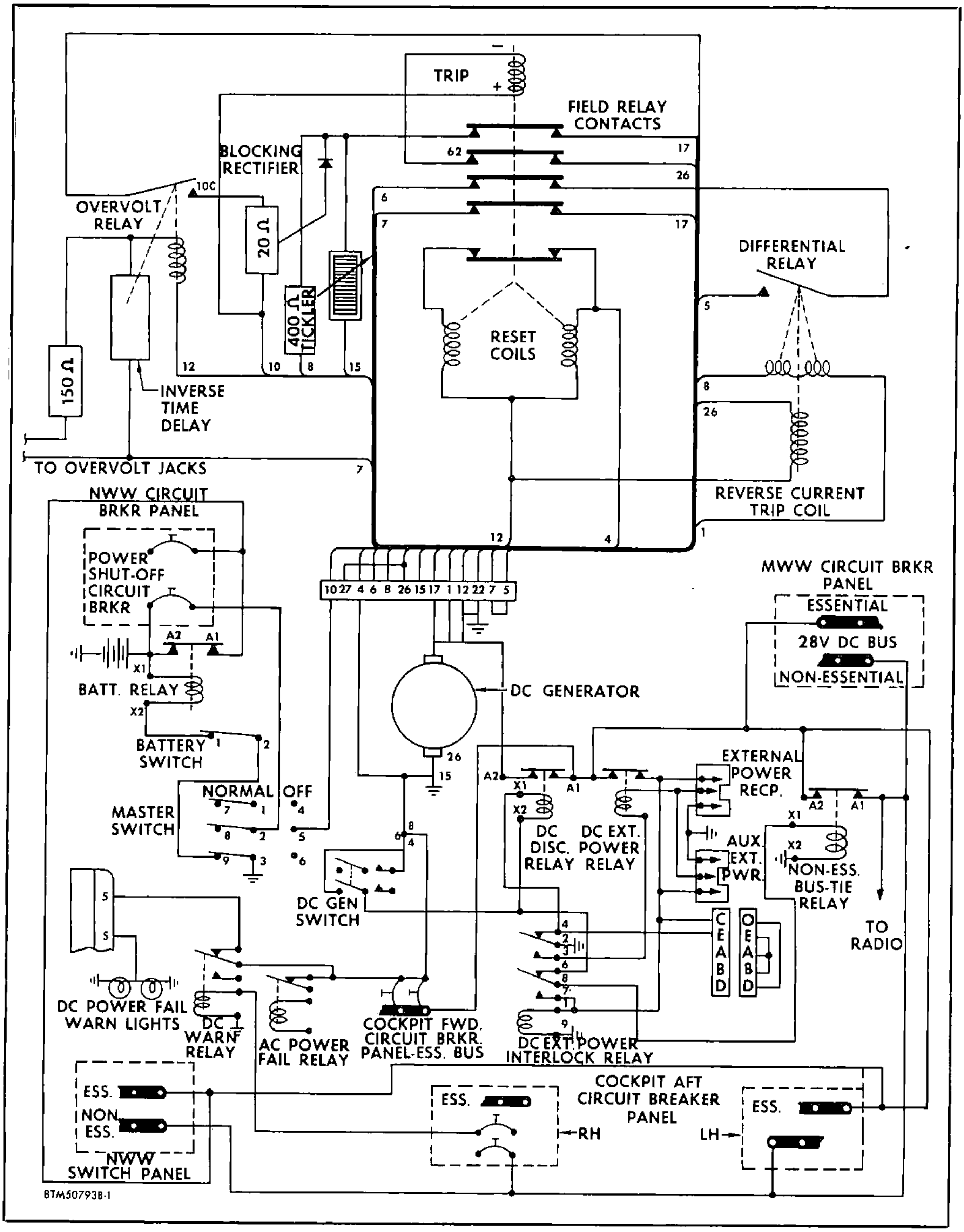


Figure 3-50. Schematic Diagram of D-C Electrical System and Location of Components (Sheet 1 of 2)

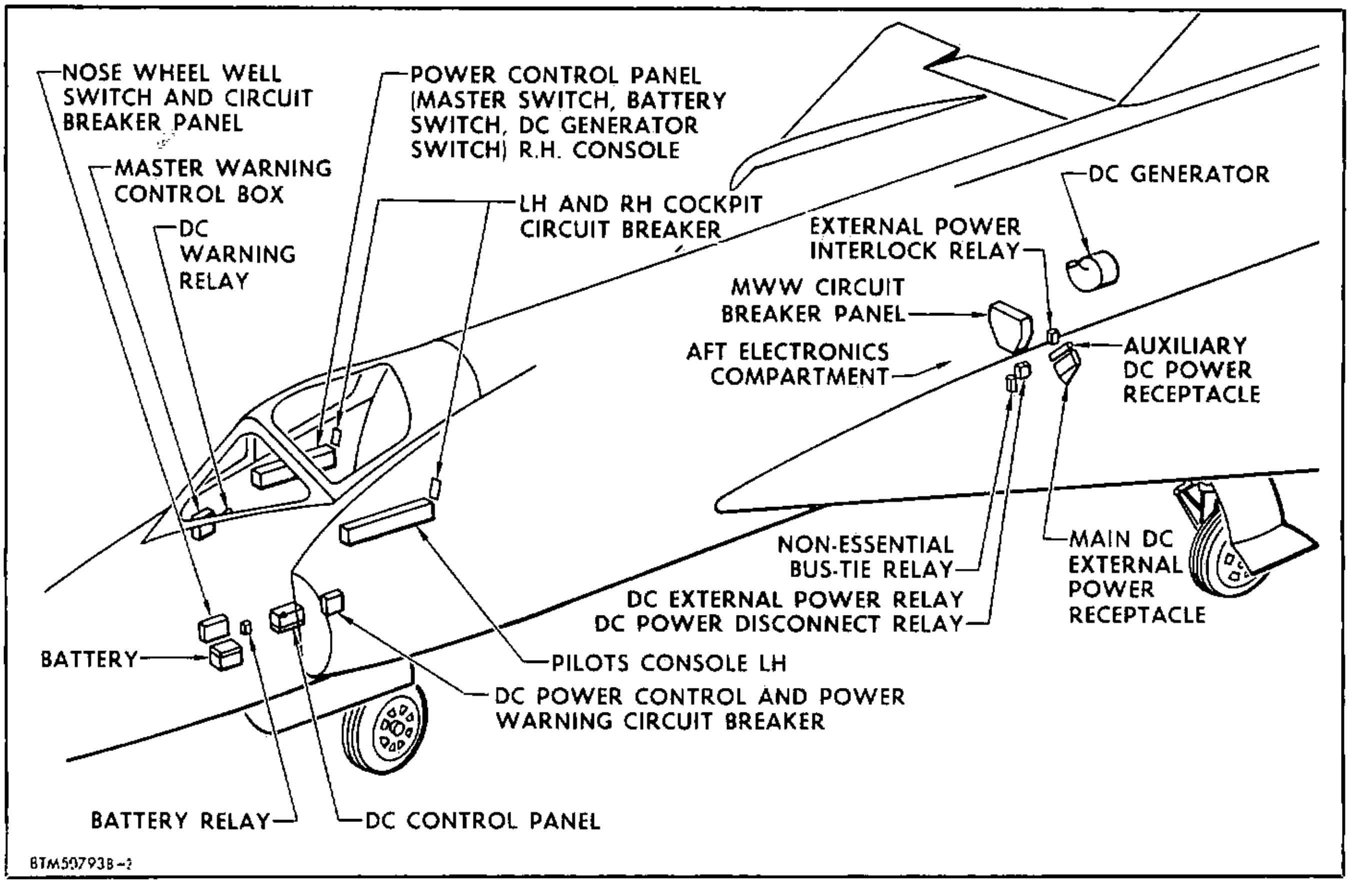


Figure 3-50. Schematic Diagram of D-C Electrical System and Location of Components (Sheet 2 of 2)

connected to the generator terminal A. This is the positive shunt field winding connection of the generator, which is connected through terminal 15 direct to the carbon pile voltage regulator.

The "equalizer" terminal of the generator, D, is wired to terminal 26 and 27 on the panel. An internal jumper wire ties these two terminals together. These, in turn, are connected to the negative side of the differential reverse current trip coil (26 on figure 3-50). This relay as you have already learned in Chapter II is actuated by the combined efforts of the differential voltage coil, reverse current coil, and the voltage relay coil. Generator terminal E is connected directly to ground. The control panel is grounded at terminals 12 and 22.

NOTE

It might be a good thing, before pursuing this further, to remind you that this is a ground return system, which means: when the airplane is properly bonded (the bonding straps tight and secure) the metallic structure of the airplane acts as a return path for most of the circuits. When it is necessary for you to replace or disconnect the generator from the system, extreme care must be taken to replace these wires to their proper terminal connections.

The D-C Control Panel in the System.

Since the control panel is located in the nose wheel well, all voltage adjustments must be made during ground operations. There is no voltage regulation in the cockpit. The system has reverse polarity protection because the generator is automatically connected to the d-c bus. This prevents connecting the generator to an energized d-c bus when polarity of the generator is reversed. There is also reverse current protection through the differential relay which senses all polarity changes from generator terminals D to E due to any reverse current.

Then there is overvoltage protection (upper left of figure 3-50). The overvoltage relay has an inverse time voltage characteristic. A sustained voltage (1 to 3 seconds) of over 32 volts energizes the overvoltage relay, which in turn energizes the trip coil of the field relay thus opening the field relay contacts in the control panel. It is this action which takes the generator off the bus.

The panel performs another function by providing a means for manual *reset* through the d-c generator switch. This applies battery voltage to the field relay reset coil, energizing it and closing the field relays. On its front panel are located the voltmeter test jacks and

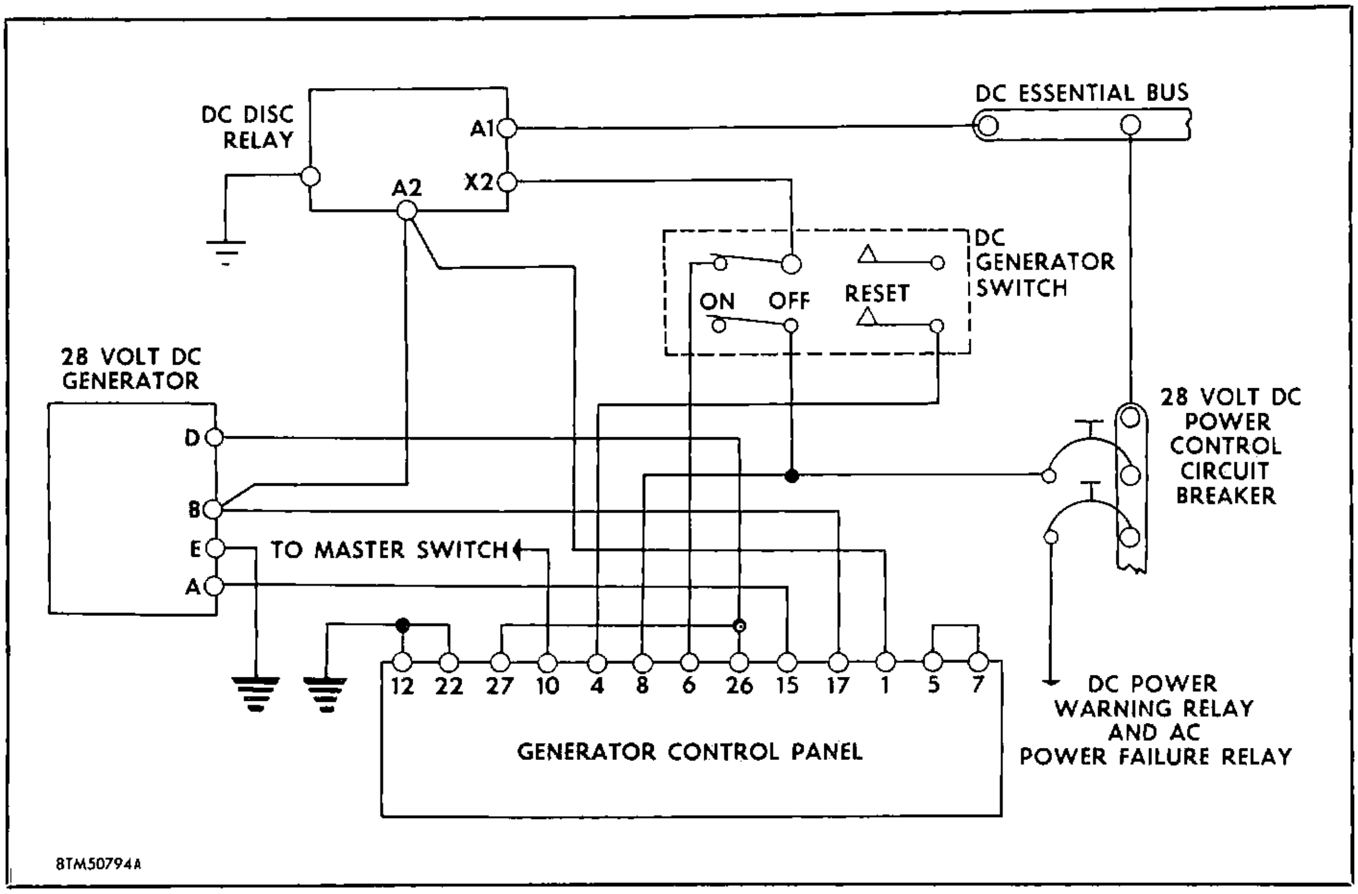


Figure 3-51. D-C Generator and Control Panel Wired Into System

an overvoltage testing connection for tripping the overvoltage relay. Finally, this panel connects the generator to, and disconnects it from, the essential bus through the d-c disconnect relay.

The Battery Wired Into the System.

The positive terminal of the battery is connected to terminal A2 of the battery relay. The negative terminal is connected direct to ground. The simplified diagram (figure 3-52) shows the 24-ampere-hour battery connected into the d-c power system; its relationship to the control panel; and its controlling and protective devices.

As you can see, terminal X1 of the battery relay is connected to contact A2 of the battery relay; while terminal X2 is tied to terminal 1 of the battery switch. In the closed position, the battery switch completes the circuit to contact 3 of the master switch when the master switch is in the normal position. The master switch is grounded to the metallic structure of the airplane through contact 3.

In this way, when the battery switch is closed and the master switch is in the normal position, terminal X2 of the battery relay is connected to ground. So, connecting terminal X2 to ground energizes the solenoid of the battery relay, thus closing contacts A1 and A2 and allowing current to flow to the buses shown in figure 3-52. It is contact A1 of the battery relay which is connected to the d-c essential and to the non-essential bus, through the non-essential bus tie-relay.

Let's go back to the master switch for a moment. In figure 3-50, contact A2 is connected directly to the power shutoff, 5-amp, circuit breaker, located on the nose wheel well circuit breaker panel. Therefore, when the circuit breaker is closed, a connection to contacts 1 and 2 of the battery switch exists, regardless of the position of the battery relay contacts.

In the simplified wiring schematic of the d-c system, figure 3-50, you can see that the positive side of the trip coil is connected to terminal 10 of the panel, and that terminal 10 is connected to the battery through contacts 2 and 5 of the master switch, when it is in the OFF position. When battery voltage is applied to the trip coil in an emergency (when the generator output falls below battery voltage) the trip coil is energized and the field relay contacts are opened. Terminal 10 is also connected to both sides of the blocking rectifier and to contact 10C of the overvoltage relay.

The Master Switch and the F-102A D-C Power System.

The master switch is located on the electrical power switch panel. This panel, as you can see in figure 3-53,

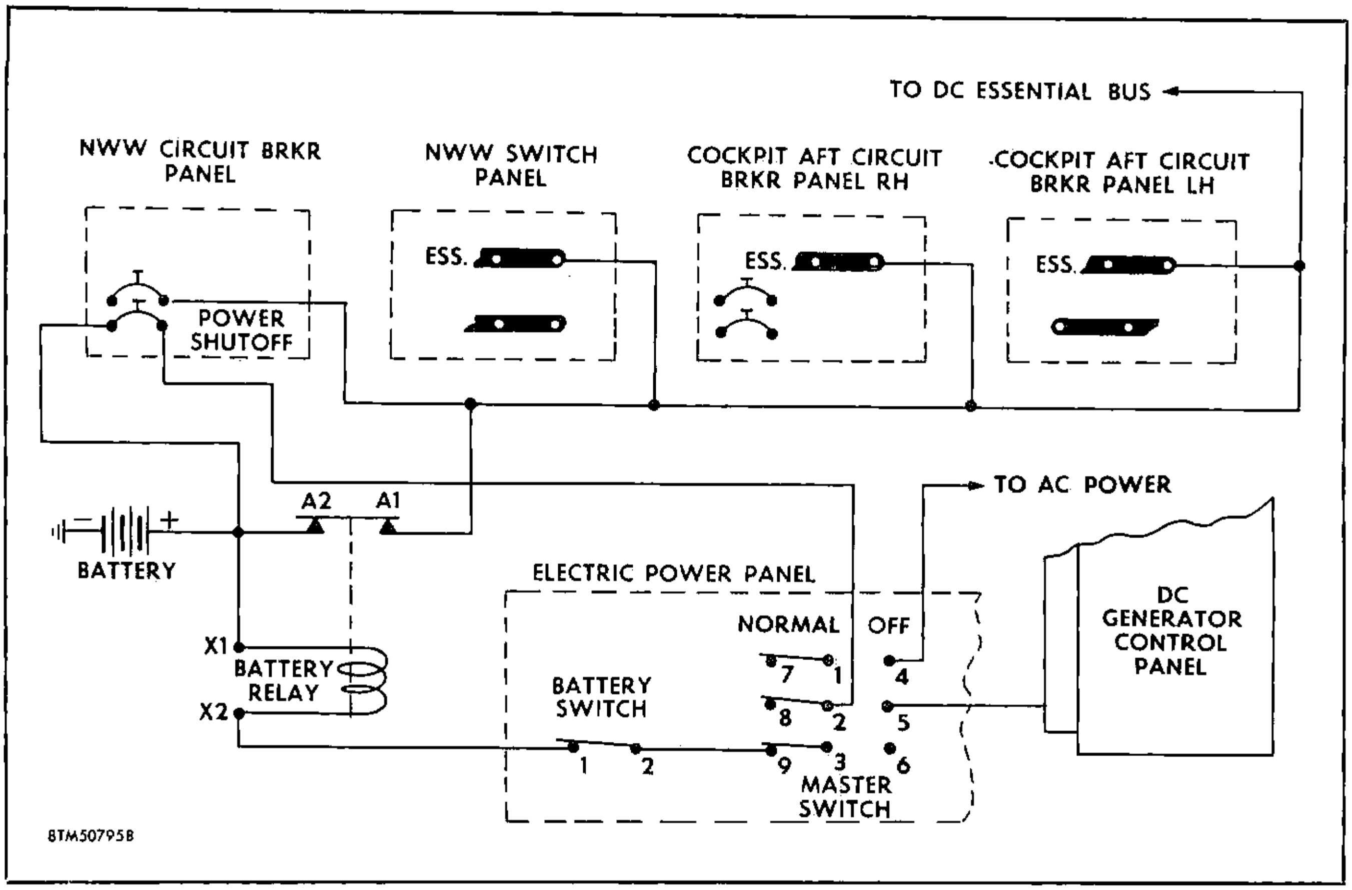


Figure 3-52. The Battery Wired Into the System

is located at the forward end of the right-hand console in the cockpit. The panel consists of the master switch, the battery switch, and the generator ON-OFF-RESET switch. Forward of these switches, and on the same panel, is the switch gear of the a-c system.

The master switch is placarded MASTER-NORMAL-OFF. The switch must be in the NORMAL position to energize either of the distribution buses in the system. At the beginning of this chapter, under Basic Power Supplies, you learned that when a master switch was incorporated into direct power systems, it was arranged so that in case of a crash the pilot can *open* the circuit by hitting the master switch to "off" and eliminate the "live" or "hot" wire condition on the airplane.

Essentially, the master switch in the F-102A is in the system for the above reason. As you know there is a battery and generator switch which deenergizes the battery-generator circuits independently of the master switch—once the master switch is actuated to NORMAL. There is no reason to move the master switch to OFF once it is in the NORMAL position. The switch was designed this way and should be used in this manner. Not only this, but upon touchdown and when the airplane is parked, if the master switch is accidentally or thoughtlessly flipped to the OFF position, the battery will drain through the switch to the trip coil of the field relay in the d-c system. If the switch is left in the OFF position for any length of time, even the meager load thrown on the battery in this manner will cut down on its charge and in time you will find a dead battery. So, under no circumstance should the master switch be moved from ON to NORMAL while the plane is grounded and in a normal readiness status. In the air, only when a crash landing is indicated, will it be switched to the OFF position; thereby deenergizing both systems in the airplane. This minimizes the hot wire factor and, at the same time, the danger of fire. But let's get back to the way the switch is hooked into the system, and its relation to the other components.

As you learned, when we discussed the circuit wiring of the battery, the battery is connected to the battery switch (contacts 1 and 2) and through the 5-amp power shutoff circuit breaker. When the master switch is in the NORMAL position, contacts 1 and 7 and contacts 2 and 8 are connected. Contacts 7 and 8 are blank. In the OFF position, contacts 1 and 4, and 2 and 5, are bridged. And conversely, with the master switch in the OFF position, 9 and 3 are opened. This deenergizes the battery relay and disconnects the battery from the d-c essential bus. Also, in the OFF position, contacts 1 and 2 are connected to contacts 4 and 5. (Contact 4 is connected into terminals 14 and 16 of the a-c generator control panel.) Contact 5 is connected to terminal 10 of the d-c

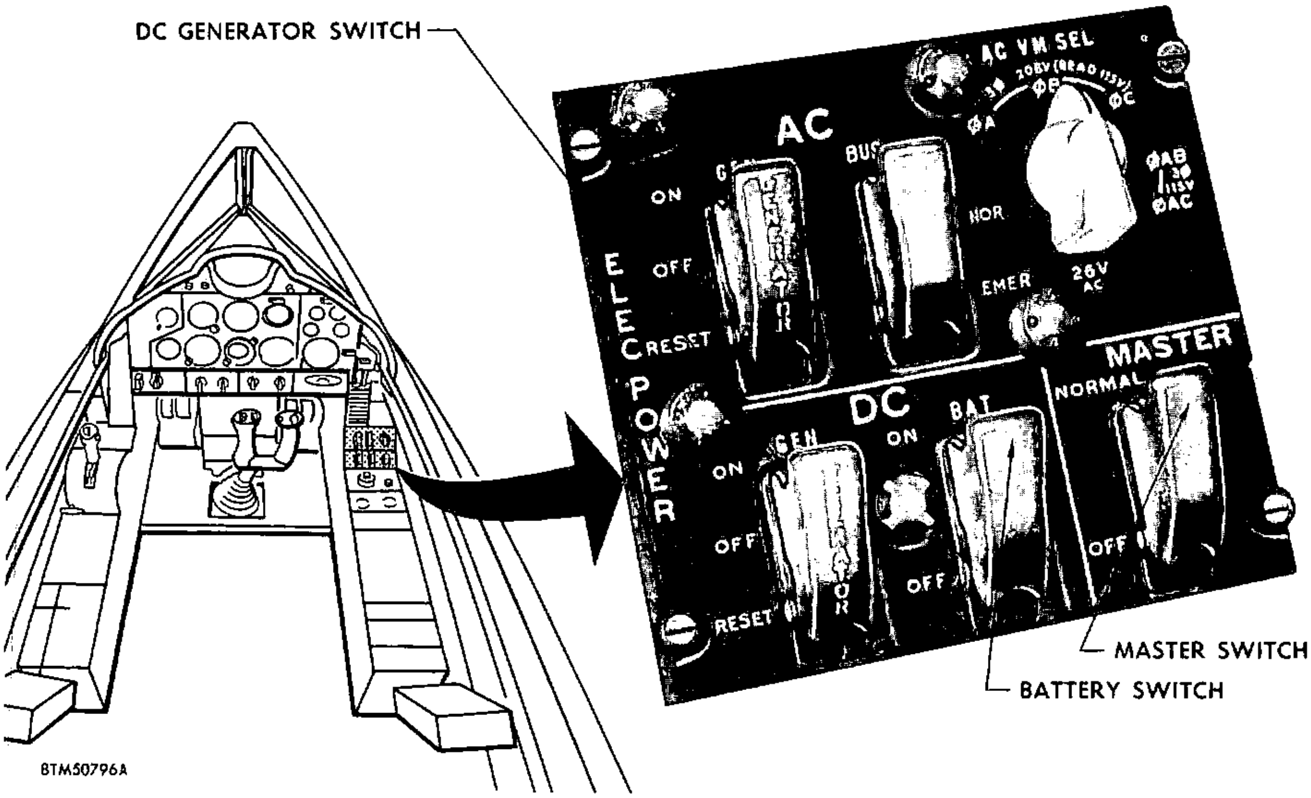


Figure 3-53. Electrical Power Switch Panel

control panel, which in turn energizes the trip coil of the field relay inside the panel. In this way, the master switch controls both the essential and non-essential buses in the systems.

Actually one could say that power from the battery is responsible for deenergizing both power systems in the airplane; because it is battery power which energizes the trip coils in both d-c and a-c control panels, and in so doing it opens the field relays and thus takes power off the buses in both systems. This brings up a point which we mentioned earlier in our discussion—its importance can not be over-emphasized. *Remember at all times,* when the power shutoff circuit breaker is closed and the master switch is OFF, the battery is still connected into the system; and this power is being applied to the trip coils of both the d-c and a-c control panels. To prevent overheating of these coils, and an excessive drain of battery current the master switch is kept in the NORMAL position during normal operation except —as we mentioned before—in the case of a crash, or an emergency dead stick landing.

The Battery Switch and the F-102A D-C Power System.

The battery switch has the simplest function of any of the manually operated switches in the system. It is located slightly inboard of the master switch on the electrical power switch panel. It is simply an ON-OFF switch. When it is ON, the battery relay is energized; when it is OFF, the battery circuit is open and the solenoid in the battery relay is deenergized and the relay contacts A1 and A2 open.

The Generator Switch and the F-102A D-C Power System.

The action of the generator switch was partially covered earlier in this chapter, when we discussed generator "field flashing." However, we did not qualify its relationship to the other components.

Basically the generator switch is a three-position switch —ON-OFF-RESET. The switch is spring-loaded only in the RESET position. *Reset* actually means, momentarily on-off. In other words the *reset* has a pushbutton type action. When the *reset* portion of this switch is actuated, the generator circuit is broken and the d-c disconnect relay coil is deenergized through terminal X2 closing contacts A1 and A2 (figure 3-51), and the field relay reset circuit is energized from the essential bus. You will be given a more complete coverage of this action when we discuss the entire system under normal operations.

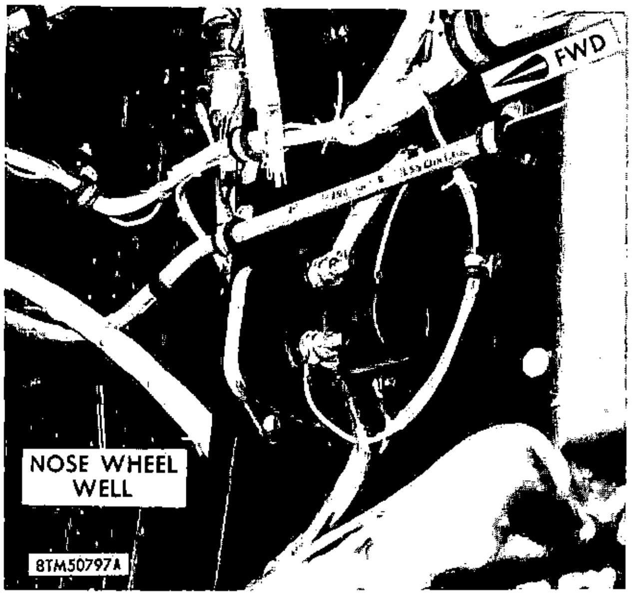


Figure 3-54. Battery Relay

This blow-by-blow description of how the generator, the control panel, the battery and the main switch gear of the d-c system (during normal operation) is related to the other parts of the system, has been necessarily specific. It is hoped that tracing the wires from terminal to terminal and calling out the relay contacts has assisted you, in your efforts to trace the rest of the wires and the current flow through to their points of contact with the other components within the system. From here on out we will refer to pin, terminal, and contact numbers only when it is felt that an explanation is in order. The reason for this should be clear enough: this is a training manual, and you will be called upon time and again to read either wiring diagrams or schematics to learn just how a system works. It may be that you are trouble-shooting the system, or a modification has come through with a new diagram showing the changes. It is imperative that you know how to read these electrical "tables of organization," in order to carry out your maintenance job efficiently. In electrical school you were undoubtedly given instruction in reading these diagrams; so now you are on your own. However, before we go into the complete analysis of the system under normal in-flight operations, as well as ground operations; let's define and locate some of the other components in the system and see how they are related to each other.

The Battery Relay and the D-C Power System.

The battery relay is an spst (single-pole single-throw), double-break relay switch. The accompanying photograph (figure 3-54) shows this relay installed in the F-102A. It is located close to the battery in the forward electronics compartment on the right-hand side of the airplane (see figure 3-50). As you know, when this relay is energized and is in the closed position, it connects the battery to the d-c essential bus.

The Disconnect Relay and the F-102A D-C Power System.

The disconnect relay is also an spst, double-break relay switch. It is provided in the system for the purpose of connecting the d-c generator output to, and disconnecting it from, the d-c essential bus (see figure 3-50). Its coil is energized by power from the generator shunt-field, terminal A.

The Non-Essential Bus Tie Relay and the D-C Power System.

Here again we have an spst, double-break relay switch. It is located inside the main wheel well circuit breaker panel (see figure 3-50). A photograph of this panel is shown in Chapter II, of this manual. The non-essential bus tie relay connects the non-essential bus to the d-c essential bus. In the case of a d-c power failure this relay is deenergized, disconnecting the non-essential bus from the essential bus, and thus taking the non-essential systems in the airplane out of the d-c system. In figure 3-50, terminal X2 of this relay is connected direct to ground; X1 to pin 8 of the d-c external power interlock relay. If you will trace the wire from contact A1, you will readily see how the non-essential circuits in the system are taken completely out of the system once the non-essential relay contacts are open.

THE F-102A D-C SYSTEM WHEN UNDER NORMAL OPERATION.

In-flight operations and normal operations have been mentioned more than once in this manual. So, for the record they both mean the same thing. There should be no mystery concerning the mechanical operation of a relay, or any switch for that matter, when wired electrically into the d-c system. There should also be a clear understanding of how the generator works in the system; how the battery functions; and why a system has a control panel, and other controlling and protective devices. You should know also that when the master switch is in NORMAL, the battery switch to ON and the generator switch momentarily to RESET and then to ON; that power is circulated from the battery and the generator through the power network to the essential and non-essential buses. At these distribution points, the current simply waits for an additional switch to be closed—or a dial to be turned—to do the various jobs demanded of it by the other systems in the airplane.

In the following explanation of the d-c power system under normal operating condition, pins, terminals, and contacts have been called out, for the purpose of helping you to visualize the system in operation. Some of the operation will be repetitious, but we learn by repeating.

THE D-C SYSTEM ENERGIZED.

Before actuating either the battery or generator switch (the master switch will, of course, be ON or in its NORMAL position) there are several d-c circuit breakers which you must be sure are in the engaged position.

Then there are the D-C Power Control, the D-C Power Warning, the D-C Power Failure and the Power Shutoff circuit breakers. Two of these, the d-c power control (PWR CONT DC) and the d-c power warning (PWR WARN), circuit breakers, are located on the Forward Auxiliary Circuit Breaker Panel. This panel and its circuit breakers are shown in Chapter II of this manual. The d-c power failure circuit breaker (DC FAIL) is located on the Right-Hand Cockpit Aft Circuit Breaker Panel. This panel is also shown in Chapter II, figure 2-40 and figure 2-46. The power shutoff circuit breaker (PWR SHUTOFF) is located in the Nose Wheel Well Circuit Breaker Panel and is shown in figure 2-42. When these circuit breakers have been checked and you have determined they are engaged, the battery switch is placed in the ON position. The D-C Power Failure light on the master warning indicator panel (see figure 3-55) should come on. If it fails to light—you have a malfunction.

You Have A Malfunction.

If the d-c power failure light fails to come on, it indicates that either the battery system or the warning system is not functioning. But, perhaps the PWR WARN circuit breaker has tripped? You check that one first and find it in the proper position. This could mean then that the battery relay is not functioning. So, next, you check the battery circuit and if this is found to be the cause of the trouble, the battery relay will be replaced. However, this time you found that it was not in the battery circuit. So the warning system is next on your check list. This is accomplished by a thorough check of the warning circuits using the diagram in your F-102A Maintenance Manual. These circuits are terminated at the master warning relay box located on the right-hand side of the nose wheel well. If the trouble is indicated as being in this relay box, the box must be replaced.

The foregoing, of course, was all supposition, because when you placed the battery in the ON position in the first place the DC POWER FAILURE warning light actually illuminated. This indicated that the battery relay was energized by the battery switch and that battery power was applied to the d-c essential bus.

The Generator on the "Line."

To put the generator on the line, the generator switch is momentarily held in the RESET position. While the generator switch is being held in this position, d-c power—through the essential bus and the power control circuit breaker—is being applied to the field relay coil. This power places the multiple contacts of the field relay in their normal operating position (they may have tripped through a reversal of current on shut down), and at the same instant voltage is applied to the generator field windings to change them. The voltage applied to the generator field assures a voltage buildup of the generator and a voltage output of the correct polarity.

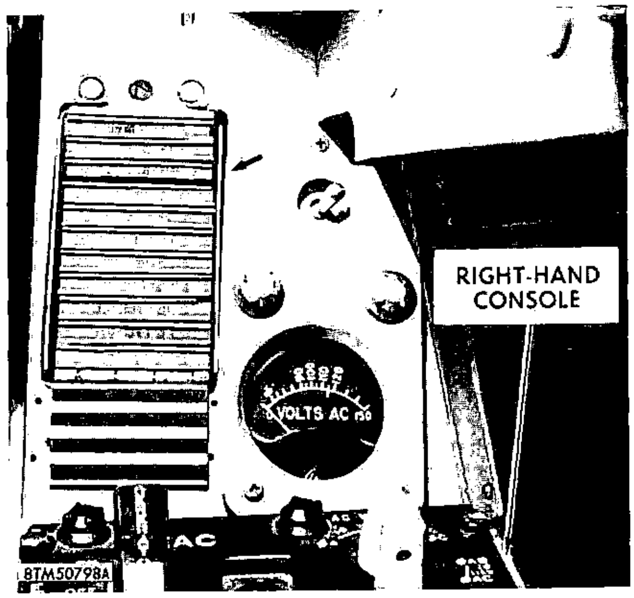


Figure 3-55. Master Warning Indicator Panel

The System Operates Normally.

When the generator switch is positioned to ON, voltage is applied through the generator control panel to the d-c power disconnect relay and the non-essential bus tie relay. By tracing the wires in the diagram (figure 3-50) you should be able to determine just how these last two relays are energized. Remember, we have just positioned the generator switch to ON. This completes the circuit through the switch contacts to terminal X2 of the coil of the disconnect relay; also to pin 6 of the d-c external power interlock relay (figure 3-50) and from pin 8 of the same relay into the coil of the non-essential bus tie relay, thus closing the contacts of both relays. The external power interlock relay, when external power is not being applied, is in a deenergized position. Therefore, terminal X1 of the d-c disconnect relay coil is connected to ground through terminals 4 and 2. This completes the circuit to the d-c disconnect relay and closes the contacts. It is in this manner that power is applied to the non-essential bus tie relay through the above mentioned pins 6 and 8. It should be noted that this relay is also tied to ground. As soon as the contacts of these relays are closed generator output power is applied to both the essential and non-essential buses.

Power on the non-essential relay opens the circuit to the master warning box and the D-C FAIL light goes out. The d-c warning relay is a single-pole double-throw (single-break) relay switch and its solenoid is energized through terminal X1 from the essential bus; X2 is connected to ground closing the circuit.

The above explanation tells how normal operation of the d-c power system is obtained. There is nothing difficult about it, but so that you will have an even more concise understanding, let's take the d-c generator out of the system by the numbers. This will not only give

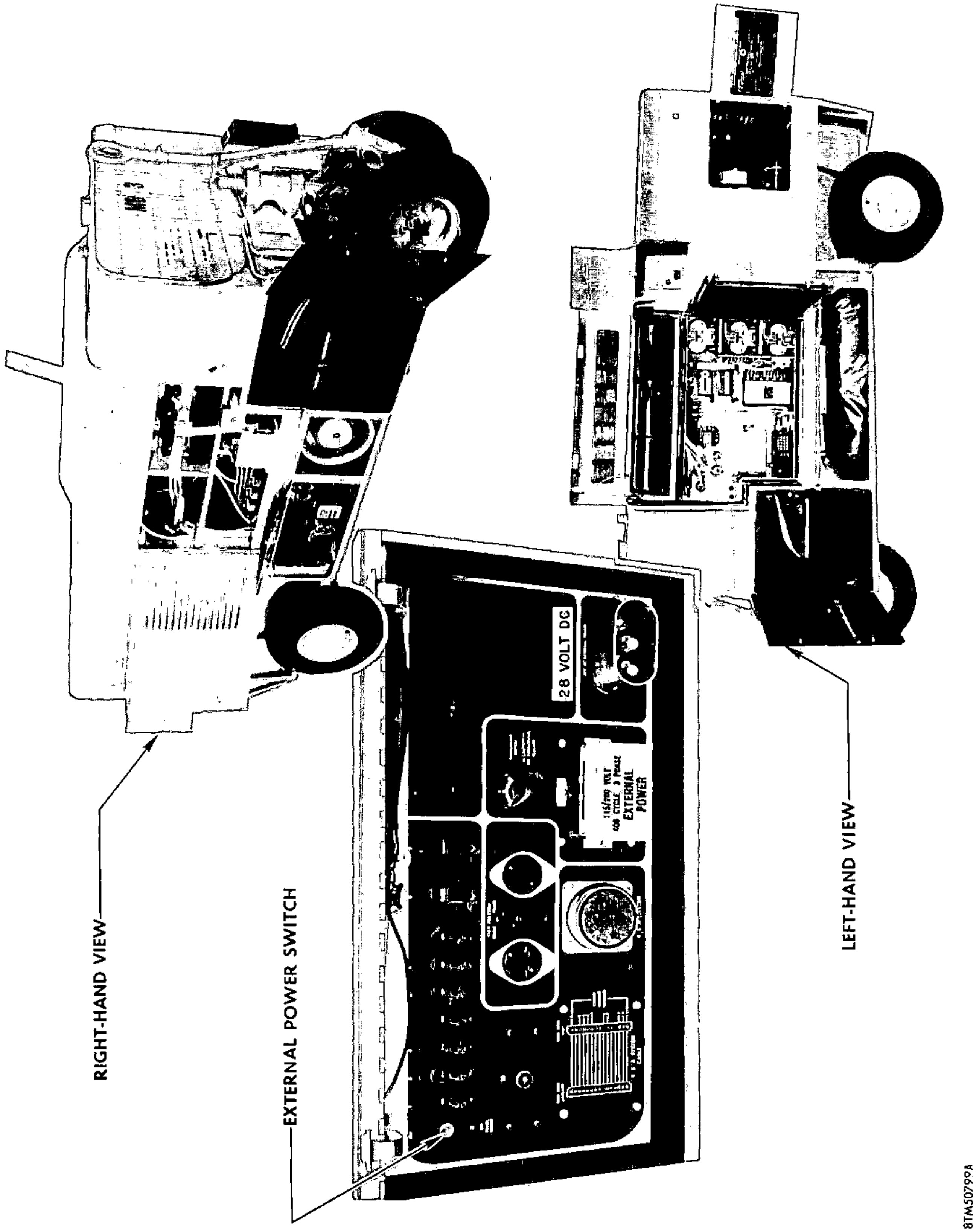


Figure 3-56. MD-3 External Power Ground Support Unit

you a clearer understanding of what is meant by reverse current, overvoltage, and connecting the generator to an energized bus when the generator's polarity is reversed; but it should show you why the control panel has been called the mainstay of the d-c power system.

WHEN A GENERATOR MALFUNCTIONS.

By tracing through the diagram, figure 3-50, you will readily see that during normal d-c operation; terminal 6 of the d-c generator control panel is connected to contact 62 of the field relay; that is, when this relay is energized. It is in this manner, the d-c generator output to the d-c generator control panel appears indirectly at terminal 6, and directly at terminal 1 of the d-c generator switch. And, as we have pointed out in our previous discussion, terminal 6 of the panel is connected to contact 6 of the d-c generator control switch. The d-c generator switch—when in the ON position—connects terminal 6 to terminal X2 of the d-c disconnect relay. This connection is in parallel to terminal X2 of the d-c non-essential bus tie relay through contacts 6 and 8 of the d-c external power interlock relay when external power is not being used. This relay is deenergized until ground power is plugged into the airplane.

By placing your finger on terminal X1 of the d-c disconnect relay and another finger on terminal E of the generator test receptacle; it can be clearly seen that this circuit is connected in parallel to ground through contacts 2 and 4 of the interlock relay when (and we repeat), when it is not energized by external power. In this same condition, contact 6 of the d-c external power interlock relay is connected to contact 8. Conversely, contact 8 is connected to terminal X1 of the d-c non-essential bus tie relay. Terminal X2 of the d-c non-essential bus tie relay is connected directly to ground. It is by this route that generator terminal output voltage turns up at terminal 6 of the d-c generator control panel and energizes the d-c disconnect and the d-c non-essential bus tie relays.

The Battery Takes Over.

The d-c generator control panel provides protection against reverse current, overvoltage, and connects the generator to an energized d-c bus when the generator polarity is reversed. Should any of these malfunctions occur, the generator output appearing at terminal 6 of the d-c generator control panel will be reduced to *no voltage;* and, the d-c disconnect and the d-c non-essential bus tie relays will be deenergized and their contacts will open. By the same token, if the d-c generator output falls below the hold-in voltage (which is approximately 7 volts), of the d-c disconnect and the d-c non-essential bus tie; these relays will also be deenergized. In either event, contacts A1 and A2 of the d-c disconnect relay and the d-c non-essential bus tie relay will open. Opening the contacts of the d-c disconnect relay, disconnects terminal B of the generator and terminal 1 of the d-c generator control panel from the d-c essential bus. Opening the contacts of the d-c non-essential bus tie relay, of course, disconnects the d-c non-essential from the d-c essential bus. When the generator output is removed from the d-c essential bus during flight, it is the battery which takes over and supplies power for limited emergency operations (6 to 12 minutes).

THE F-102A POWERED FOR D-C GROUND OPERATIONS.

During ground operational checks and testing of either the d-c or a-c electrical power systems, an external source of electrical power is connected to the airplane. This source of power is received from a ground support multiple power plant. We qualify it as *multiple,* because it contains both a d-c and an a-c source of power. The unit you will use on the flight-line for the F-102A is designated as an MD-3 ground power unit. It is shown in figure 3-56. It is gasoline-driven and provides 28 volts for d-c power; and 115/200 volts, 3 phase, 400 cycles for a-c power. Another unit, especially designed for hangar operations, is the MC-1. This version is electrically driven. For further information on the MD-3 unit refer to T. O. 35C2-3-249-1.

CONNECTING D-C EXTERNAL POWER.

Figure 3-57 shows the location and the appearance of the main external power receptacle which provides connections for both d-c and a-c power. The inset in the lower right-hand corner is the auxiliary d-c receptacle. This receptacle is used when landing gear tests are in progress. As you can see, the main d-c receptacle is inboard of the a-c receptacle and contains but three contacts.

There are a number of precautions that must be observed when connecting external power to the airplane. You will remember that we said, the master switch — when once placed in the NORMAL position, remained in that position—and only when the airplane crash landed was the switch to be moved to the OFF position. For actual flight readiness this was quite true, but when the aircraft is undergoing ground tests and repair with the use of external power, the master switch is actuated to its OFF position. Before connecting external power to the airplane be sure that this switch is OFF. Also be sure that the battery switch is OFF. With external power, the generator output is controlled from the MD-3 ground support unit; therefore, the generator switch must at all times be in its OFF position. External power impressed on the coils or brushes of the aircraft-installed generator will cause serious damage.

Another precaution—and this must be strictly adhered to when either normal or external power is on the aircraft—be extremely cautious when working around terminal strips, junction boxes, circuit breaker panels, relays or limit switches. You could be badly burned—even killed. Never attempt to remove any of the components in an electrical system without first turning off the electrical power.

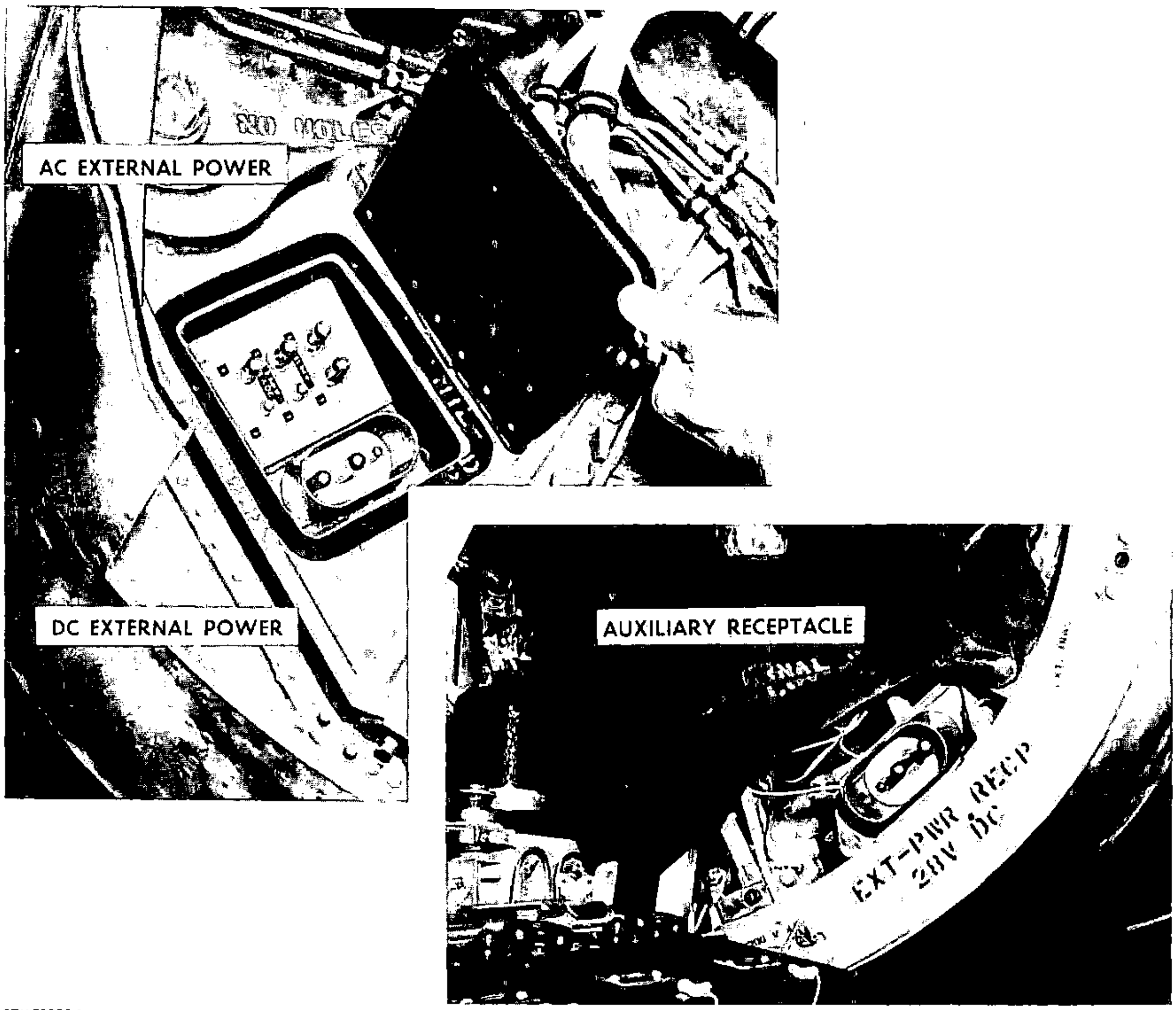


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Figure 3-57. The D-C External Power Receptacle and D-C Power Auxiliary Receptacle

EXTERNAL D-C POWER ON THE F-102A.

When you insert the external power cable connector in the d-c external power receptacle, and the d-c external source is turned on; 28 volts are connected to terminal X1 of the d-c external power relay from the negative side of the receptacle. At the same time, d-c external power is connected in parallel to terminal 1 and contact 7 of the d-c external power interlock relay and to terminal C of the generator test receptacle (see figure 3-50).

You can see that there are three connections in both the external power receptacle and the auxiliary receptacle. One appears as positive (+), the other as negative (−). The d-c external power appearing at the center terminals of the d-c external and the d-c auxiliary external power receptacles, is provided to energizes the d-c external power relay, the d-c external power interlock relays, and the d-c non-essential bus tie relay. Terminal 9 of the d-c external power interlock relay is connected direct to ground. Thus, the d-c external power interlock relay is energized, and it opens contacts 2, 4, 6, and 8; and closes contacts 2, 3, 7, and 8. Opening contacts 2 and 4 disconnects terminal X1 of the d-c disconnect relay and terminal E of the generator test receptacle from ground. Opening contacts 6 and 8 disconnects contact 6 of the d-c generator switch from terminal X2 of the d-c non-essential bus tie relay. Closing contacts 2 and 3 of the d-c external power interlock relay connects terminal X2 of the d-c external power relay to ground, thus energizing the relay and closing contacts A1 and A2. This pattern of electrical action connects d-c external power to the d-c essential bus.

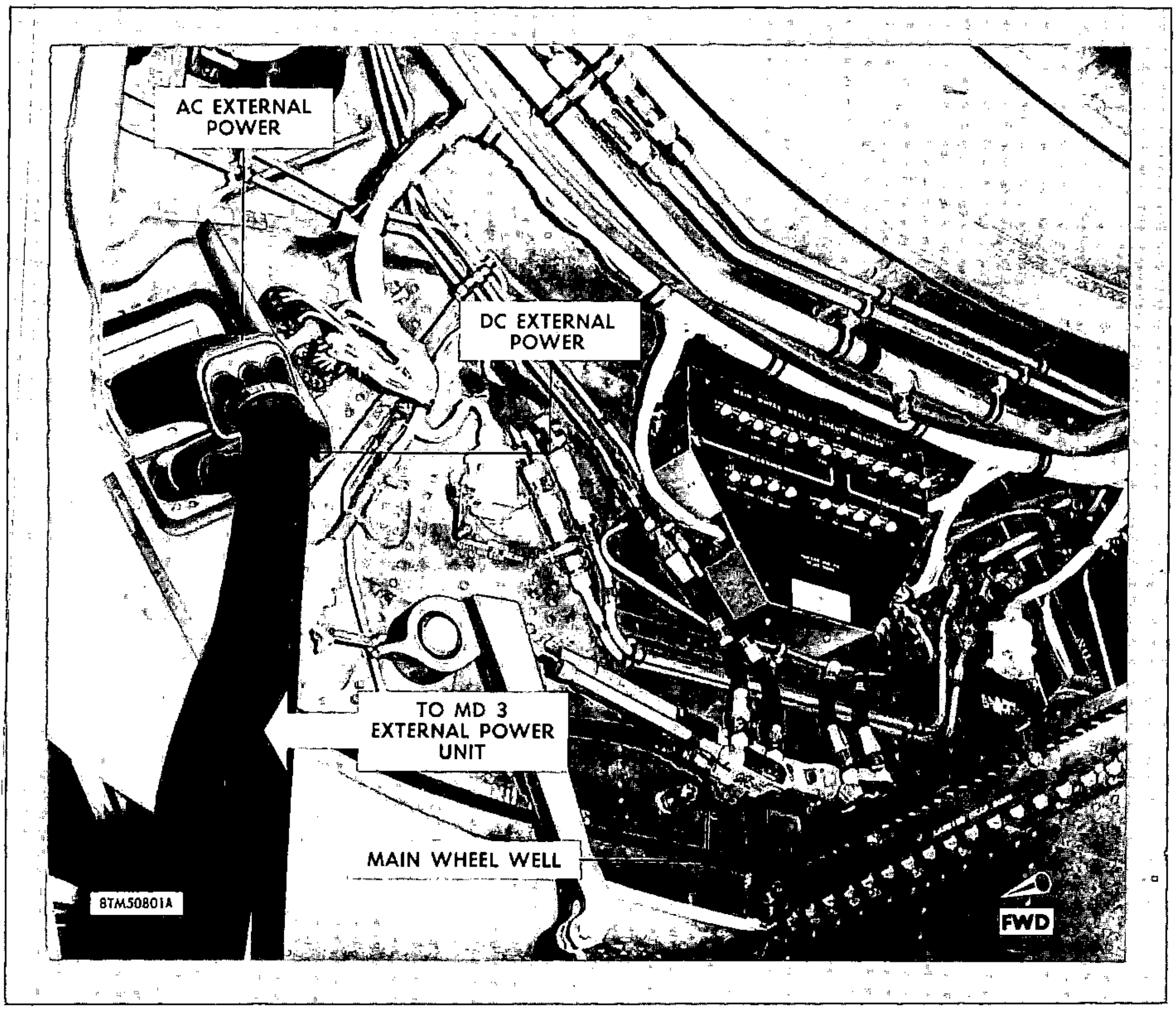


Figure 3-58. External Power on the F-102A

The closing of contacts 7 and 8 of the d-c external power interlock relay connects d-c external power to terminal X2 of the d-c non-essential bus tie relay. This energizes the relay, closing contacts A1 and A2, and connects the d-c non-essential bus to the d-c essential bus.

A WORD ABOUT OPERATIONAL CHECKS AND TESTING.

The operational checks and the testing of the d-c power system are fully covered in T. O. 1F-102A-2-10, the F-102A Handbook of Maintenance Instructions; therefore they will not be covered in this supplement. Electricity, as you know, is instantaneous. When a switch is turned on, relays are activated, the system is energized, and certain lights illuminate and extinguish as we explained when talking about the D-C Power Failure light. The failure of this light to illuminate, if you remember, necessitated our checking both the battery system and warning system for continuity. If the light had failed to extinguish with the d-c battery switch ON, the master switch in NORMAL, and when generator output was impressed on the system by the generator switch being actuated; our check would have been longer and more involved. But a routine continuity check, which we have discussed in this chapter, would have obtained the same results.

If such a malfunction had occurred, any one of six possible reasons, or perhaps a combination of several, could have been responsible. It could be that the d-c warning relay was not functioning or perhaps the battery relay had stuck in a closed position—perhaps the non-essential bus tie relay was faulty. Another ***probable*** cause could be the d-c disconnect relay's failure to close the circuit, a defective d-c control panel, and if the d-c generator terminal voltage was below its rated output or not functioning at all. In each of the above cases the component must be replaced if it is found defective.

As you can see, unless the probable cause of the master warning light's failure to extinguish is found immediately, you will have some extensive checking to do. Where to begin? Well, perhaps the D-C Fail circuit breaker is not engaged. You've checked it and found it O.K.? How about the d-c control panel? Have you checked the voltage at the test jacks? You have! Then let's start checking the battery circuit, and after that let's isolate the trouble to the specific relays mentioned above. This is accomplished by following the test procedures as outlined in T. O. 1F-102A-2-10.

If you will think through what has been "fictionalized" above, and have a clear picture of the d-c system as a whole unit of operation, some of that "plain old common sense" we spoke of earlier should help you to analyze any trouble confronting you. For example, if it is found that the generator voltage will not build to a proper value, the trouble must be at the generator; it is either overheating, which might mean a dirty generator, or there is resistance in the external wiring, or load. Where would you begin in this case? Why at the generator, where else? And how would you go about it? First, check the electrical connections, test them for tightness and after this you would check the entire circuit for continuity. It's as simple as that.

A LAST WORD.

The analysis of the d-c system just covered, has been nothing more than a summary of all we have discussed so far in this manual concerning the direct-current system in the F-102A. It is hoped, with the help of the Technical Orders (T. O.'s) you have been referred to, your knowledge at this point is more than just familiarity—that "false sense of knowledge." And remember, you only learn and become efficient in your job as a maintenance man by the actual "doing."

In the next chapter of this supplement, we will discuss the alternating-current system in the F-102A and its related components. In some cases, such as circuit protective and controlling devices which relate to both systems, and subjects such as high altitude and climatic effects, wiring, and others of this nature, which were covered in the preceding chapters, you will simply be given reference to these portions of the supplement when they are applicable. *They will not be covered again.* But they are in the supplement, and will be continual reference sources for you, and we hope valuable to your knowledge of electricity and the F-102A.

Chapter IV

THE F-102A A-C ELECTRICAL POWER SYSTEM

Contents *Page*

In this chapter we will discuss some of the basic principles of alternating current, its protective and controlling devices, and how it is distributed and used in the F-102A interceptor. In the opening pages of the last chapter, we traced direct current as it was, and is used in aircraft, from that icy-cold morning near Kitty Hawk, North Carolina, up to the present day. In the early days, d-c electricity composed 90 percent of the electrical power used in an airplane; any a-c electricity necessary for lighting systems and instruments was converted from the d-c system or manufactured by a cumbersome inverter. Today, there has been almost a complete turnabout. Most of the critical electrical equipment installed in aircraft uses alternating current.

There is nothing particularly difficult or mysterious about the a-c power system in the F-102A. It is simply a little more involved than the direct current system. And for this reason, perhaps a little general information concerning alternating current, as such, should be the order of the day. After all, it should not be too difficult to venture the prediction that in the very near future a-c electricity will be the primary and major power system in most aircraft.

WHY ALTERNATING CURRENT IN AIRCRAFT.

That electric coffee pot, toaster, waffle iron, the iron used to press the shirt you are wearing, the electric shaver you used this morning, all derive their electrical energy from alternating current. Many years ago, public utility companies learned that alternating current could be transmitted over long distances more readily and more economically than direct current. Alternating-current devices are simpler and less subject to trouble and they are smaller and therefore light of weight. Not only this, but a-c voltages can be increased and decreased, without loss, through step-up or step-down transformers. The transformers used in the F-102A a-c power system are a good example.

In basic electrical school, you learned that in conventionally equipped airplanes, electrical systems are usually divided into four major groups: the ignition system, the starting system, the d-c generator and its control system, and the lighting system. What you learned is fundamentally true, but the conventional airplane is fast disappearing from the sky, especially when we relate it to the military service. Let's find out why; let's take a typical high performance modern aircraft and find out why its power demands have increased to a nondimensional one of great scope. Let's take the F-102A as our example.

First of all, the F-102A, as an interceptor, has layers of altitude to traverse before its combat objective is reached. To carry out its mission effectively, it must travel at extreme speeds. And under its sleek skin are installed the units that make this mission possible. The critical electronic, radar and armament installations, which would fill a good-sized truck, are installed in its compartments. These (together with the machinery for the manufacture of the required electrical energy and its measurement, application and control) make an alternating-current system, with its simplicity, its weight and space-saving characteristics, an absolute "must."

The amount of useful load is highly important to the modern high-speed aircraft, and each pound saved without impairing performance can be added to the aircraft's useful load. Reliability, however, cannot be sacrificed to save weight. Actually, to say that the size and weight of any equipment installed on an airplane are "important" would be a gross understatement of fact. They are critical. This is especially true of the modern supersonic airplane.

It is for this reason that in some airplanes today, direct current has been eliminated almost completely. Batteries, for example, are in some cases things of the past. In a case such as this, the small amount of d-c electricity needed for some instruments and communication equipment is obtained from transformer-rectifiers. This method of obtaining direct current will be discussed later in this manual.

ALTERNATING CURRENT HISTORICALLY.

Probably the first important application of a-c systems was in 1941 on the XB-19. This system provided a total of 25-kva from two generators operated in parallel at 400 cps (cycles per second) and driven by auxiliary gas engines. But the first large-scale use of an a-c power system was on the B-36. The power derived from its four, 40-kva generators, operating in parallel, far exceeded that of any previous airplane. Alternating-current generators are rated in KVA, which stands for Kilovolt Amperes.

Actually, a-c power on airplanes dates back to World War I when wind-driven a-c generators powered the first spark transmitters on Air Force airplanes. These alternators, as they were then designated, produced high frequency power for radio transmitters. The transmitters were built into the mechanism. A number of years went by before an improved modification was initiated. As airplanes became faster and demanded more maneuverability, the high parasitic drag of these wind-driven alternators caused them to be abandoned. Parasitic simply means, of the nature of a parasite.

In 1941, the Army Air Force developed a rectified a-c system. The reason for this was two-fold: the first of these we have already discussed on page 61 — the unexpected short life of carbon brushes on d-c generators when operating at high altitudes; and second, airplanes became so large that the electrical power demanded by them was greater than the ordinary generator could provide. This power could not be economically provided with the commonly used 28-volt d-c system because of the excessive weight of the cabling and the large number of generators required to supply the demand. At first, many aircraft designers were reluctant to use a-c power because of its relatively complicated control and protective devices as compared with the tried and proven d-c devices. But, in a comparatively short time, these arguments against turned to arguments for, and that was due mainly to centralized a-c control and protective panels; to accurate voltage regulators; and to the development of a light weight constant-speed drive similar to the one used today on the F-102A.

A COMPARISON OF ALTERNATING AND DIRECT CURRENT.

Included in Chapters II and III of this supplement are many electrical devices which not only pertain to direct-current systems but are also peculiar to alternating-current systems. In Chapter II, as an example, circuit protective and controlling devices such as switches, circuit breakers, and various types of relays were discussed at length. Those devices are also used throughout the F-102A a-c electrical power system. In Chapter III, the general information contained under those sections dealing with the maintenance and reliability of electrical equipment and with natural and the induced environment and its effects on an electrical system, can also be applied to alternating-current systems. In the same degree, wiring practices, the potted component, and trouble shooting in general can all be applied to the a-c power system of any airplane. In fact, many of the principles, the characteristics, and the effects of alternating current are similar to those of direct current. But, there are differences between d-c and a-c power. Some of these we have covered—for example, the weight and space-saving characteristics of a-c. Further examples are that in many a-c motors no brushes are required,

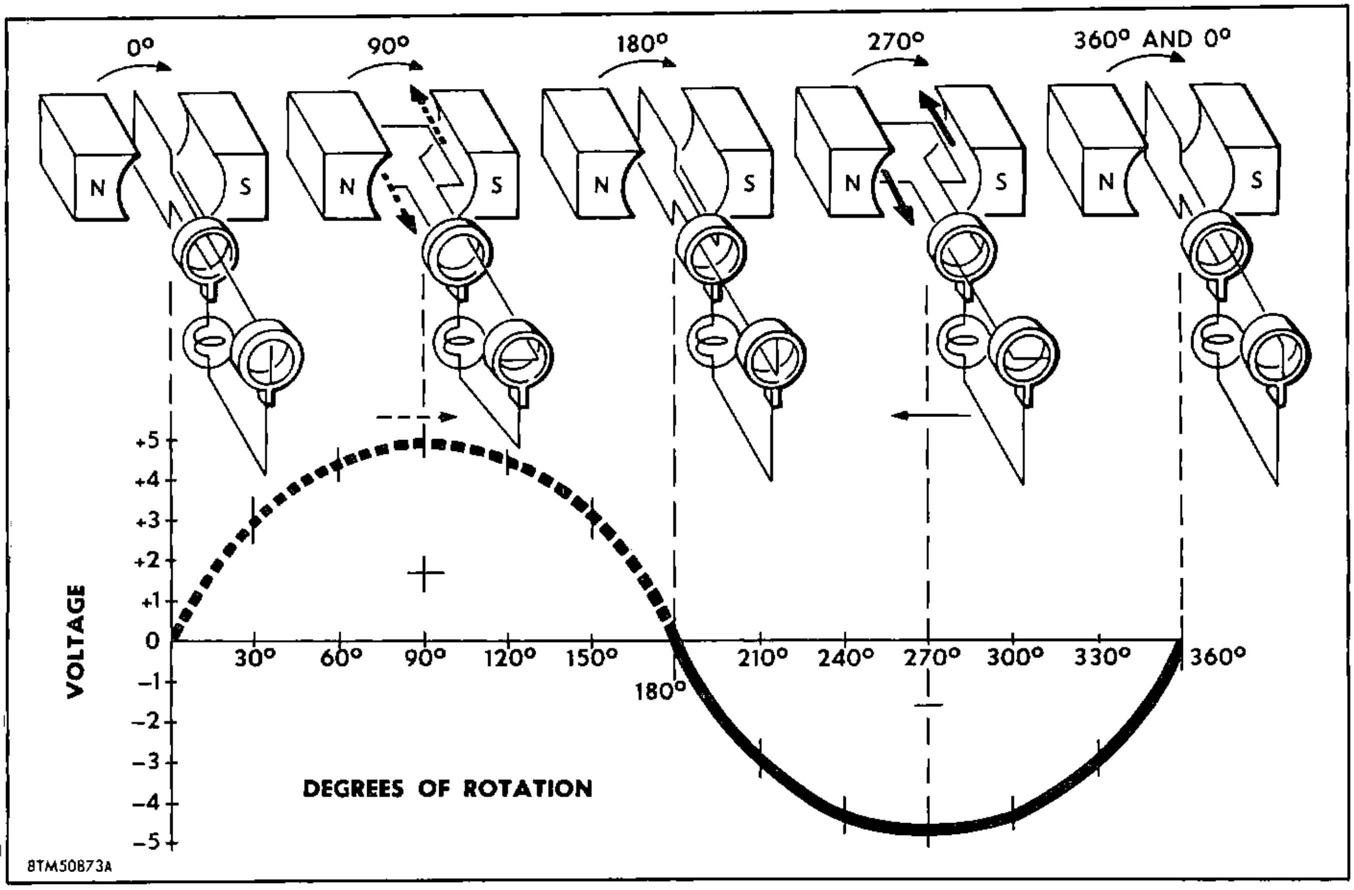


Figure 4-1. Generation of Alternating Voltage

and commutation trouble at high altitudes is thereby eliminated; circuit breakers will operate satisfactorily under load at high altitude in a-c systems—circuit interruption is a rare fault. In some cases, in d-c systems, arcing is so excessive at high altitude that circuit breakers in d-c systems must be replaced frequently. And there are other differences peculiar to alternating-current systems which you will recognize as we progress in this chapter.

In Chapter I, the principles of generating a-c emf are discussed at length. You learned that the principal difference between an a-c generator and a d-c generator is the method used to connect their external circuits—the a-c generator is connected to slip rings the d-c generator to the commutator; and that either type operates by the induction of a-c voltage in the coils. For a review of these principles, you should turn to Chapter I and re-read this material, and if necessary, study those pages dealing with the generation of alternating current.

Pehaps at this point in the discussion, we should further compare these currents by bringing back into focus the movement of electrons as they behave in an a-c circuit. As you already know, direct current is current that always flows constantly in one direction—it is constant also in its amplitude.

Alternating current, on the other hand, changes its direction constantly and at regular intervals. This is pictorially summarized in figure 4-1. In an alternating-current circuit, electrons move through the circuit in one direction for a short period of time, when our a-c generator is on its positive alternation. Then, as the generator starts its negative alternation, the electrons change their direction and move back over the path they formerly covered for a corresponding time interval. This is best illustrated in the following simple a-c circuit, figure 4-2.

THE PECULIARITIES OF ALTERNATING CURRENT.

Direct current consists of electrons moving from a point of low potential to a point of high potential. This is also true in alternating current with but one difference—they have their own peculiar method of travel. This difference is best demonstrated in figure 4-2 above. As indicated by the solid arrows, electrons move toward their point of high potential. But unlike our electron flow in direct current, which has continuousness, the electron flow in a-c changes at this high pont, from one side of the circuit to the other. The polarity of the applied alternating emf (electromagnetic force) has changed, causing the direction of current flow to change

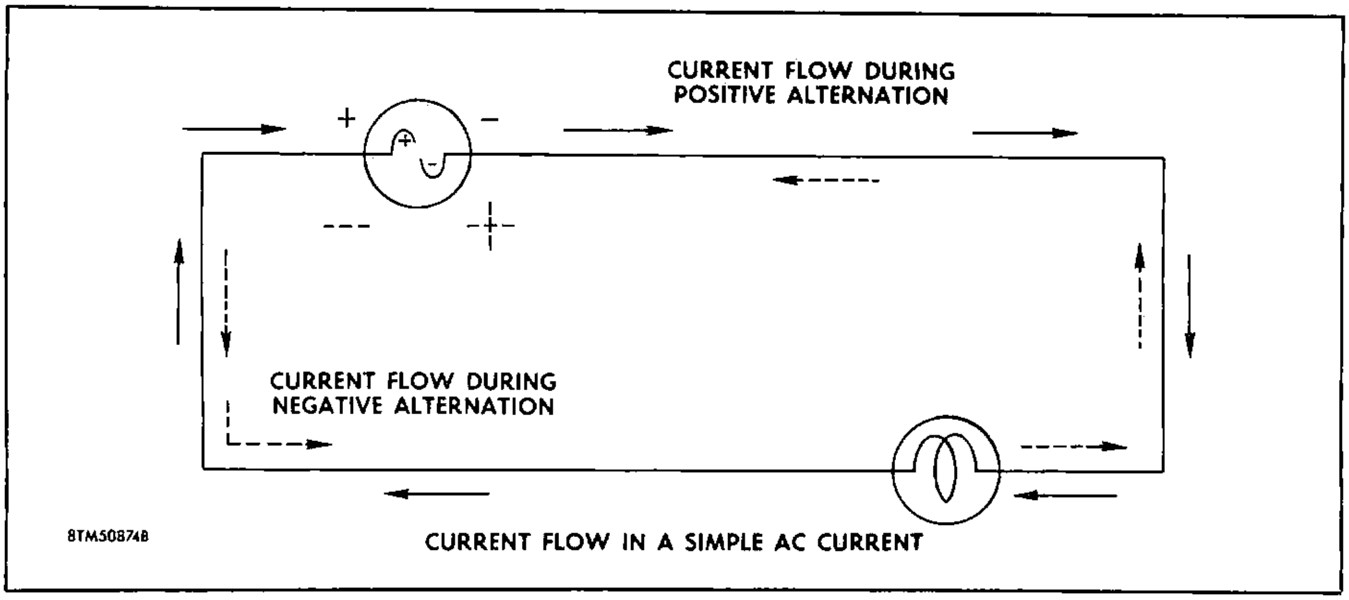


Figure 4-2. Electron Flow in an A-C Circuit

as indicated by the dotted arrows. In other words, alternating current is a current that has continuousness only in its varying in magnitude and its periodic change of direction.

CYCLE AND FREQUENCY.

One completed positive and negative alternation of an a-c generator is termed a cycle. Whenever a voltage or a current passes through a series of changes and then returns to its starting point and starts the same series of changes all over again, the series is called a cycle. In electricity, the cycle is represented by the symbol ∿. We have said that there are two alternations in the complete cycle of an a-c generator. The number of times each cycle occurs in a given period of time is called frequency. Electrically speaking, the frequency of an electrical current or voltage indicates the number of times a complete cycle recurs in one second. This is demonstrated in the cycle of voltage and its corresponding current shown in figure 4-3.

In a generator, the voltage and current pass through a complete cycle of values each time a coil passes under the north and south pole of the magnet. The number of cycles for each revolution of the coil is equal to the number of cycles in one complete revolution multiplied by the number of revolutions per second. For clarity, let's express this mathematically as follows:

F (the frequency)

$$= \frac{\text{P (the number of N and S poles)}}{\text{2 (the number of pole pairs)}} \times \frac{\text{RPM}}{60}$$

Sixty represents the number of seconds in one minute. To find the revolutions in one second, we simply divide the revolutions per minute by 60. Thus, if we have an a-c generator that turns at a speed of 6000 rpm, such as the one used in the F-102A, we would find that it turns 100 revolutions in one second; since it is an eight-pole a-c generator, the frequency would be four times 100 or 400 cycles per second. Expressed in the form of the above equation, it would look something like this:

$$F = \frac{8}{2} \times \frac{6000}{60}$$

$$F = 4 \times 100$$

$$F = 400 \text{ cps (cycles per second)}$$

In other words, to generate the 400-cycle alternating current used in the F-102A, it is necessary that a coil rotate at 6000 constant revolutions per minute. Of course, there is always a plus or minus tolerance allowable. In the case of the F-102A, it is 6000 rpm $\pm$ 60 and 400 cycles $\pm$ 4, or a tolerance of 1 percent.

ALTERNATING CURRENT VALUES.

Four values must be considered when discussing alternating current and voltage. We will discuss them in the following order: the maximum or peak values, the instantaneous values, the average value, and effective value. These are shown in the sine-wave form, figure 4-4.

Peak and Instantaneous Values.

The maximum or peak value is indicated as the highest voltage or the highest current reached on either the positive alternation or the negative alternation. The instantaneous values of current and voltage in alternating current vary constantly. Also, there may be any number of instantaneous values between zero and the maximum or peak value in either the positive or negative alternation. Instantaneous value is the value of the a-c voltage or current at one particular instant. It may

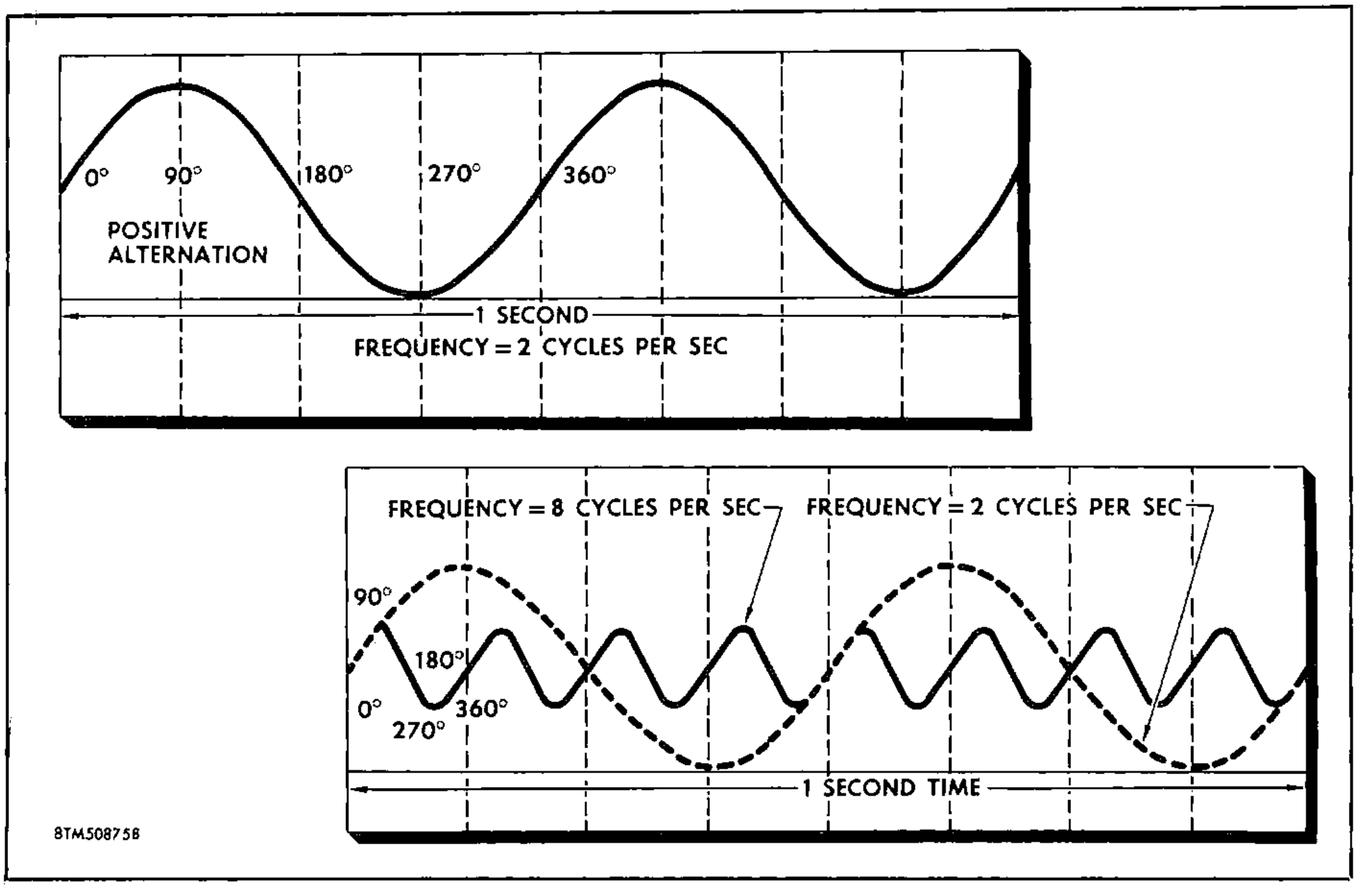


Figure 4-3. Frequency and Cycle in A-C

be the same as the maximum or peak value if the selected instant is at the time the voltage or current stops increasing and starts decreasing. The instantaneous value could be zero if the selected instant is the time during which the polarity of the voltage is changing. It is for this reason that in alternating current, the maximum or peak voltage or current values cannot be used directly in solving for power consumption as they can be in direct current. In the curve, any point we wish to pick out could be the instantaneous value in either alternation. It could be the peak value, it could be zero, or it could be the value between these two extremes.

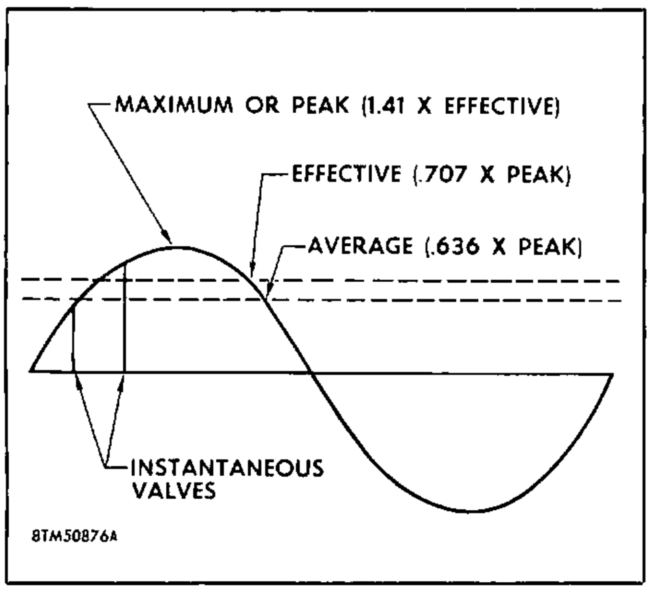
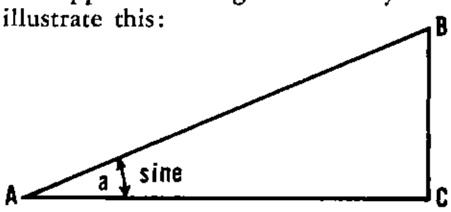


Figure 4-4. A-C Values

The Sine of an Angle.

To carry this discussion into the sphere of mathematics, the sine of the angle, through which a revolving coil of wire has moved from zero to any point in its orbit, may be used graphically to represent the voltage induced in the coil at any particular instant. Each curve or wave shown thus far in this chapter is what is known as a sine wave. They are plotted from the sines of the various angles of rotation. Let's find out more about them.

In a right triangle the sine of an angle is equal to the side opposite the angle divided by the hypotenuse. Let's illustrate this:



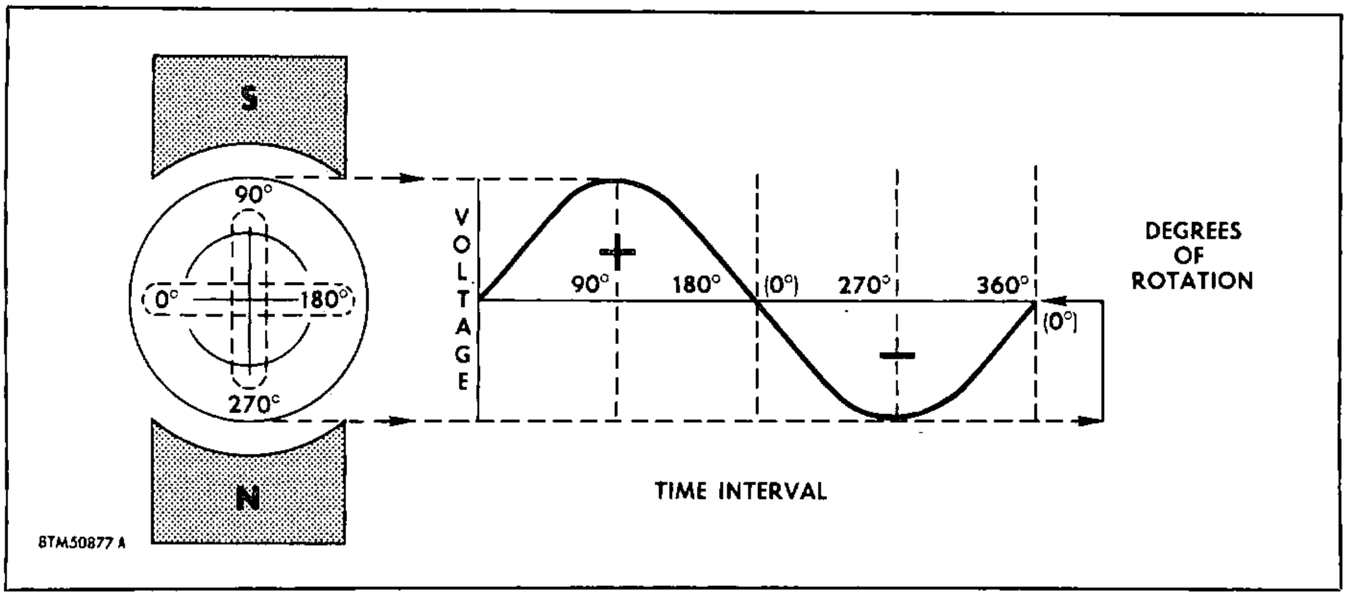


Figure 4-5. Voltage Sine-Wave

From high school, you undoubtedly remember the definition we have given, but if not, the right triangle drawn above will refresh your memory. In this triangle ABC, AB is the hypotenuse, BC is the side opposite the angle a or the sine, so we would write it something like this:

$$\text{Sine a} = \frac{\text{BC (the opposite side)}}{\text{AB (the hypotenuse)}}$$

For example: 0° has a sine of zero, for 30° it is 0.50; 60° it is 0.866; 90° it is 1.0, and for 120° it becomes 0.866 (as it was for 60°) and at 180° it becomes zero again. As you can see in the plotted sine wave, figure 4-5, as the values move to the negative side of the horizontal line, they become minus quantities; at 210° the sine is −0.50, 240° is −0.866, 270° is −1.0, 300° is −0.866, and at 360° we find that we are back at zero again.

Finding Instantaneous Values.

The formula used to find the instantaneous values of an emf in terms of the sine of the angle, through which the conductor has been rotated from its neutral position (0°) to the maximum a-c voltage value is written:

$$\text{sine a} = \frac{e}{E_{max}}$$

a is the angle through which the conductor has moved from its neutral position,

e is the instantaneous value, and

E_{max} is the maximum value.

The formula to find the instantaneous value of current would simply mean substituting i and I_{max} for e and E_{max}. In your capacity as a flight-line maintenance mechanic, you will probably never be called on to use the formulas given above. It is hoped, however, that by knowing the method of plotting and of finding the values of alternating current that the elementary mathematical facts just given you will help to clarify the values found as you learn basic a-c circuits.

Average Values.

The average value in alternating current is the average of all the instantaneous values during one alteration. And, except from a mathematical viewpoint, it is of no great significance. It is merely a numerical average of all the sine values for all the angles, and it is equal to 0.636 times the maximum value.

What Is Meant by Phase.

A sine-wave represents a series of a-c values, or quantities plotted by vector representation. In the same manner that the positioned hands of an ordinary wrist watch accurately represent the time of day, an interval of time, or the quantity of time left in a working day, the vector is an imaginary indicator whose length, direction, and position accurately represents a given electrical quantity of current, voltage, or power. Vector quantity is computed mathematically and is mentioned here simply to familiarize you with the term. For our immediate understanding, let's say that a vector represents an electrical amount and phase. And electrically that phase is an interval of time.

When a circuit contains pure resistance, the current and voltage will pass though zero and reach maximum value at the same time. The current and voltage are then said to be in phase. This is shown in the illustration of the output voltage characteristics of a two-phase a-c generator, figure 4-6. The phrase "pure resistance" means a circuit containing little or no reactance. If, however,

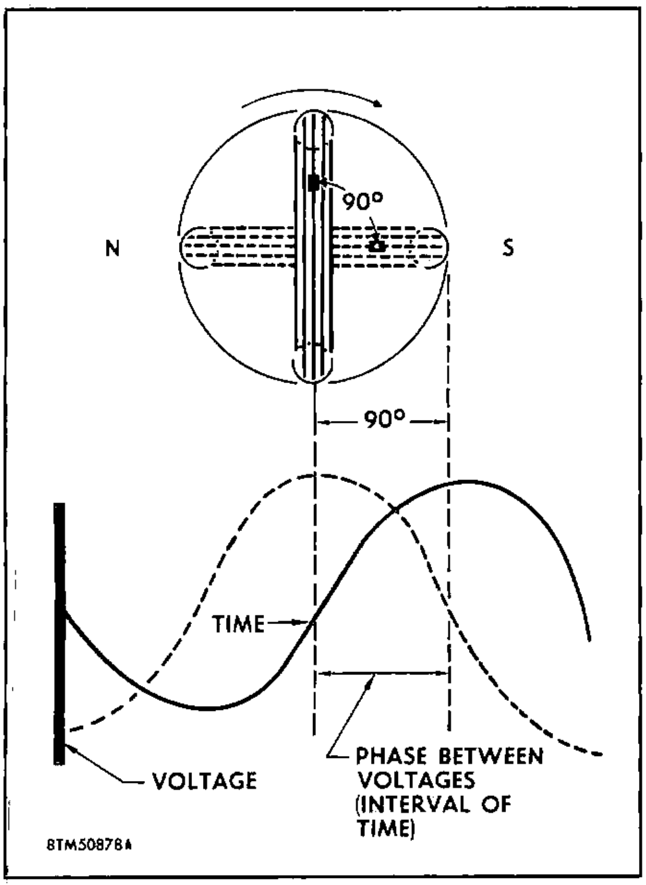


Figure 4-6. Two-Phase A-C Generator

there is resistance, and the current and voltage pass through zero and reach maximum values at different time intervals, the current and voltage are said to be out of phase. By measuring the number of degrees of rotation that has taken place between the time the leading curve crosses the zero point—or any other reference point for that matter—the number of degrees difference in the two curves may be determined, as shown in figure 4-7. The difference in degrees between the two curves is called "the phase angle."

Example "A" shows voltage and current in phase. In sine-wave "B," the current is lagging the voltage by 90°, or ¼ of a cycle. This is the effect of a pure inductive circuit. In sine-wave "C," the current has reached its maximum value ¼ of a cycle, or 90° ahead of the voltage. When this occurs, it means that we have a circuit which contains only capacitance. Therefore, the amount that the current "lags" or "leads" the voltage in a circuit depends on the amount of resistance, inductance, and capacitance in the circuit.

Effective Values of Current and Voltage.

In alternating current, any values given for current or for voltage are assumed to be effective values unless otherwise specified. But in practice, you will be dealing only with effective values of voltage and current. This should not be confused, as it often is, with the "effective value." The effective value is the actual rating of the useful power available to do work. And because it is the actual rating of the useful power to do work, perhaps a discussion of what is meant by the phase of current and voltage should be discussed before taking up the most important of alternating-current values, effective value.

As you know, in any d-c circuit, the voltage across the circuit and the current through the circuit have certain instantaneous magnitudes. As you learned earlier in Chapter II, these are determined by the actual values of the voltage and the resistance present in the circuit.

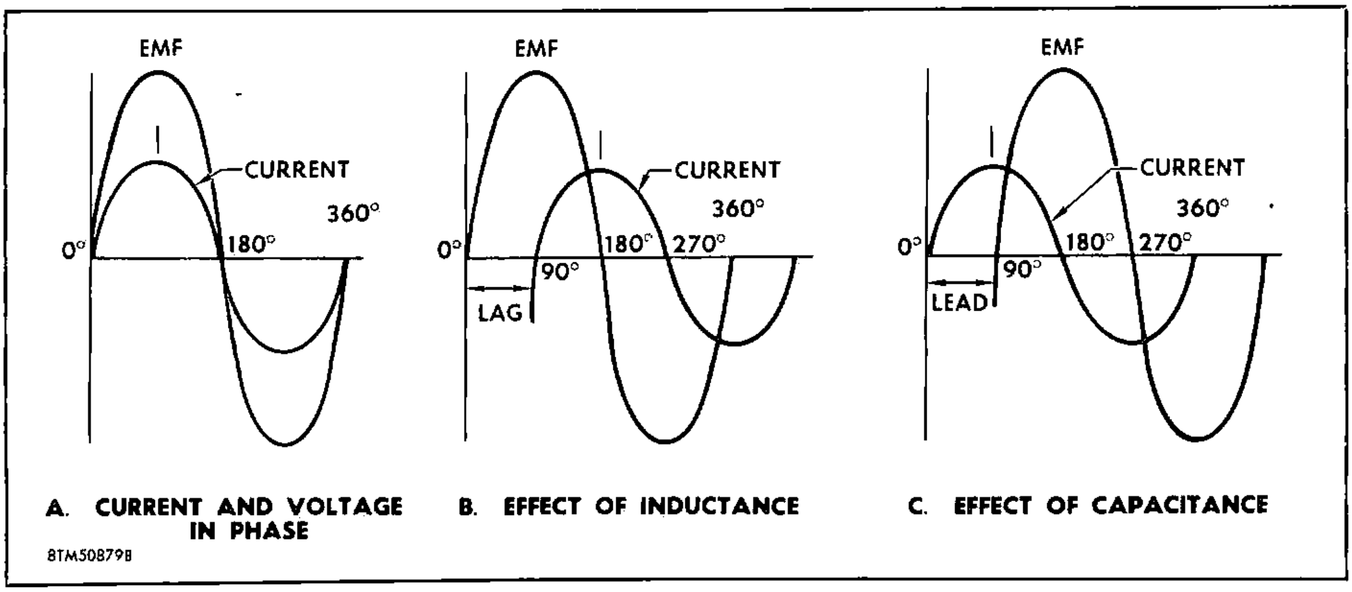


Figure 4-7. Sine Waves of A-C Voltage and Current

It should be clear then why the term "effective value" had no particular significance in our discussion of d-c circuits. After all, if 10 volts or 1000 volts are applied to a d-c circuit, the effective d-c voltage would obviously be 10 volts or 1000 volts. If this statement is puzzling, it is suggested that you turn to those pages dealing in d-c circuits and re-read them. But, since in a-c circuits the instantaneous values of current and voltage vary, there must be some basis on which to judge them. The basis used is d-c current.

In an a-c circuit, when the voltage and current are in phase, the effective values of voltage and current are expressed as values of d-c voltage and current. In other words, the effective value of alternating current is the *same value as direct current* in the *same circuit* that would cause the *same amount of electrical energy* to be dissipated or produce an equal heating effect.

To get back to our sine-wave voltages for a moment, if the a-c power under consideration is of a sine-wave nature, it would not be necessary to resort to any experimental computations to arrive at its effective value. It is proven mathematically that for sine-wave current or voltage, the effective value is equal to 0.707 times the maximum or peak value. As an example, let's say 10 volts of a-c are applied to a circuit; the effective voltage present would then be equal to 0.707 times 10 volts, or 7.07 volts. This is also true of current. If the current in this same circuit measured 10 amperes peak or maximum, the effective current flowing would be 7.07 amperes.

The effective value then in an a-c circuit is equal to 0.707 times the peak value ($E_{eff} = .707E_{peak}$). The effective value is also called the root mean square or RMS value.

Let's find out how this particular figure above — this constant — is arrived at, and why it is referred to as the RMS.

The effective value of an emf is the square root of the average of the sums of the squares of the instantaneous values which make up a cycle or one complete alternation. And as we have shown above, the effective emf is represented by the symbol E_{eff} and its relation to the peak emf is expressed in the formula:

$$E_{eff} = \frac{E_{peak}}{\sqrt{2}}$$

or

$$E_{eff} = 0.707E_{peak}$$

and for current,

$$I_{eff} = 0.707I_{peak}$$

So that these values for sine-wave current and voltage can be arrived at without lengthy mathematical computations a list of constants has been calculated. The relationships between these values apply, however, only to sine-wave AC, and are used for both voltage and current.

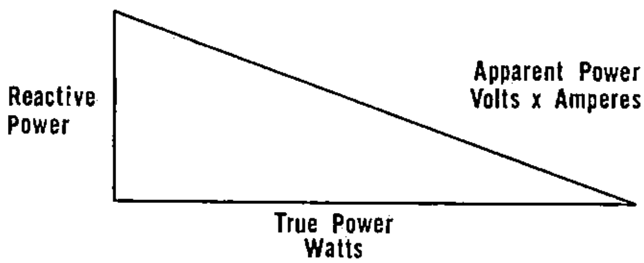
The effective value is equal to 0.707 times the peak or maximum value. The other values which have been discussed generally, follow along in line: The peak or maximum value is equal to 1.41 times the effective value; the average value is equal to 0.636 times the peak or maximum value; and the maximum value is equal to 1.57 times the average value.

APPARENT POWER.

While we are on the subject of values, it might be well to discuss what is known as "apparent power." Apparent power or volt-amperes is of considerable importance because it is the volt-amperes, and not volts, which determine the operating limits of an a-c generator. The a-c generator used in the F-102A, for example, has a rating of 30-kva. Kva is a measurement of the apparent power of an electrical system — the number of volt-amperes divided by 1000. In other words, it is the product of one impressed kilovolt and the resulting current in amperes — a kilovolt being equal to 1000 volts. It is from this that we get the kilovolt-ampere and its abbreviation, kva.

The apparent power in an a-c circuit is equal to the product of the effective, the rms, values of voltage and current. This product, however, is not equal to the true power, except when the voltage and current are in phase.

On page 47 of this supplement, you learned that the unit of electrical power is the *watt*. Electrical power is the rate at which electrical energy in a circuit is expended. Another way of expressing the same thing: power is the rate of doing work; in electricity it is equal to the voltage multiplied by the current in the circuit. Written in equation form for obtaining the power factor in a d-c circuit, the formula reads: $P = EI$, or watts equals volts times amperes. In other words, if 1 ampere flows in a circuit with a pressure behind it of 200 volts, the power is 200 watts. The product of the volts and the amperes is what is known as the "true power" in the circuit. But on page 47, we were talking about d-c circuits and here we are discussing a-c. Well, actually, they parallel each other. In an a-c circuit, a voltmeter indicates the effective power while an ammeter indicates the effective current. Apparent power is the product of these two readings; so in an a-c circuit, it is only when an a-c circuit consists of pure resistance that the apparent power is equal to the true power. Let's take a look at the power relations in an a-c circuit:



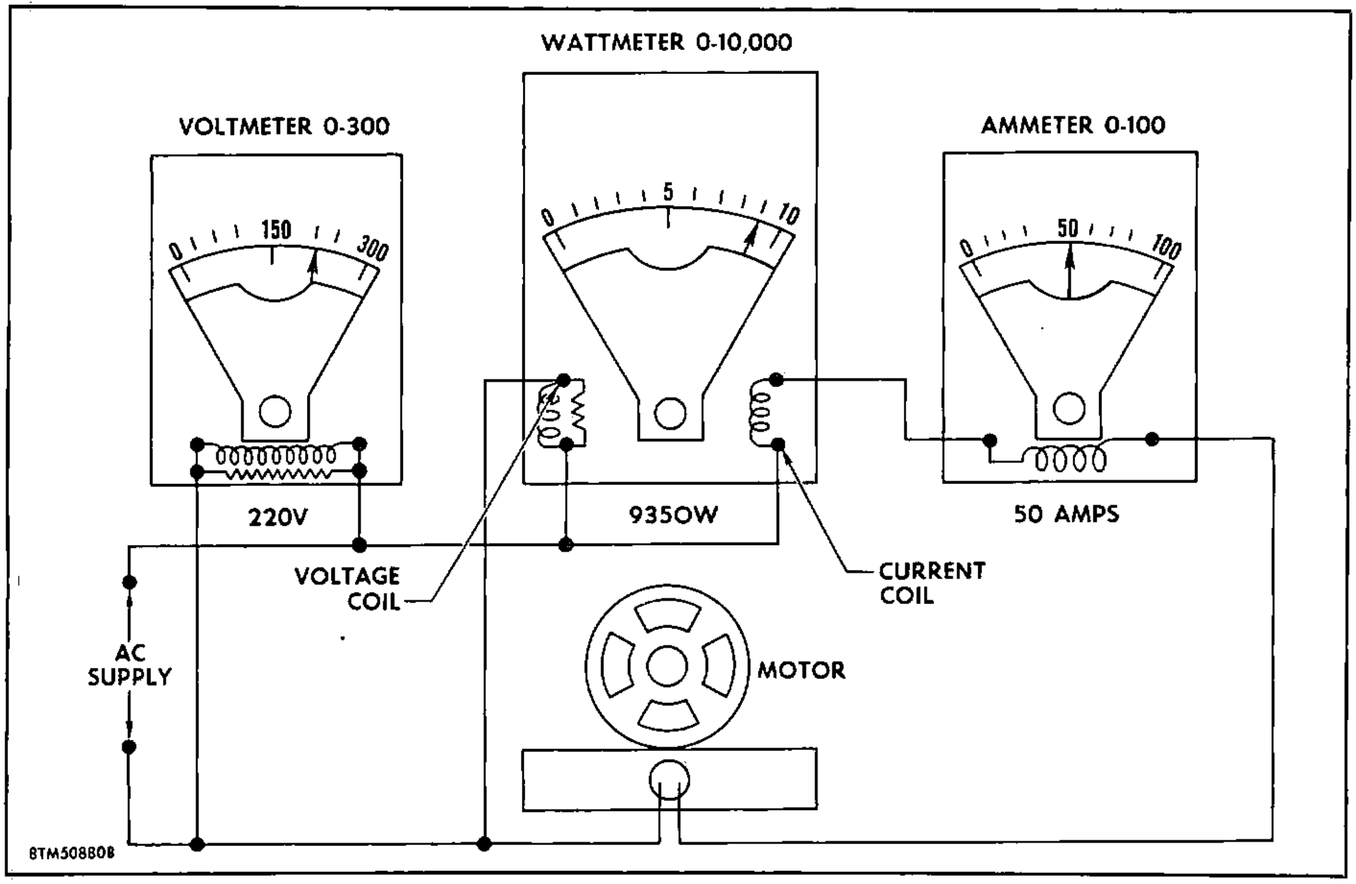


Figure 4-8. Measuring the Power Factor

In an a-c circuit, the true power is less than the apparent power when the current and voltage are out of phase. And this, as you know, is caused by the amount of capacitance or inductance in the circuit. The true power in an a-c circuit is therefore obtained by a wattmeter reading and not by multiplication as it is in a d-c circuit. The wattmeter will be discussed a little later in this chapter when measuring instruments for the a-c system are discussed. It is the ratio in an a-c circuit of the true power to the apparent power that is called the power factor. It is usually expressed in percent. Let's look at this relationship in equation form:

$$\text{Power Factor} = \frac{100 \times \text{Watts (The True Power)}}{\text{Volts} \times \text{Amperes (The Apparent Power)}}$$

Figure 4-8 should clarify the measurement of the power factor in an a-c circuit. As you can see, the 220-volt a-c motor is apparently taking 50 amperes from the line. The wattmeter, however, in the line shows that only 9350 watts are taken by the motor. To find the apparent power and the power factor in a case like this we would use our equation. We know that the apparent power is equal to the volts times the amperes,

$$\text{Apparent Power} = 220 \times 50$$

or

$$11{,}000 \text{ volt-amperes}$$

therefore

$$\text{Power Factor} = \frac{9350 \times 100}{11{,}000}$$

$$PF = \frac{935{,}000}{11{,}000}$$

$$PF = 85 \text{ or } 85\%$$

As we have already pointed out, volt-amperes in alternating-current is important as it determines the operating limits of an a-c generator. And since it is important, perhaps a further word should be said in regard to these operating limits. These output limits of an a-c generator are determined chiefly by the temperature rise which is produced in the windings. This increase in temperature is caused principally by the core and copper losses. Core losses depend on the frequency and the flux density and are fixed by the operating voltage and frequency. Copper losses, on the other hand, are determined by the amount of current. Full load is reached when the equipment is carrying the full rated current at the rated voltage and frequency.

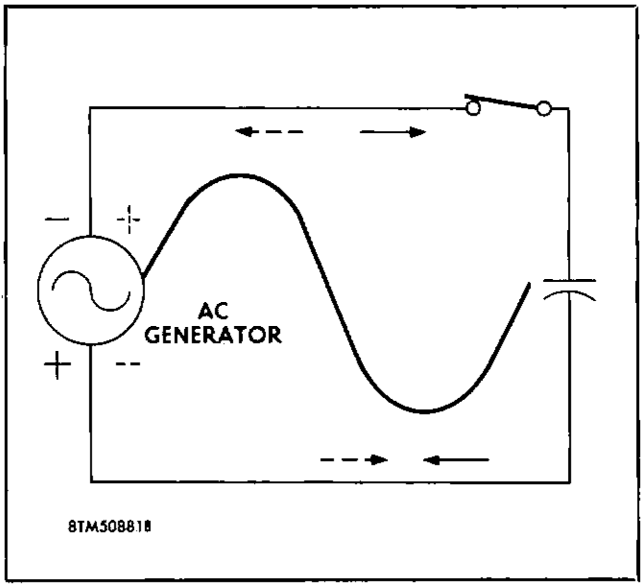


Figure 4-9. Capacitance in an A-C Circuit

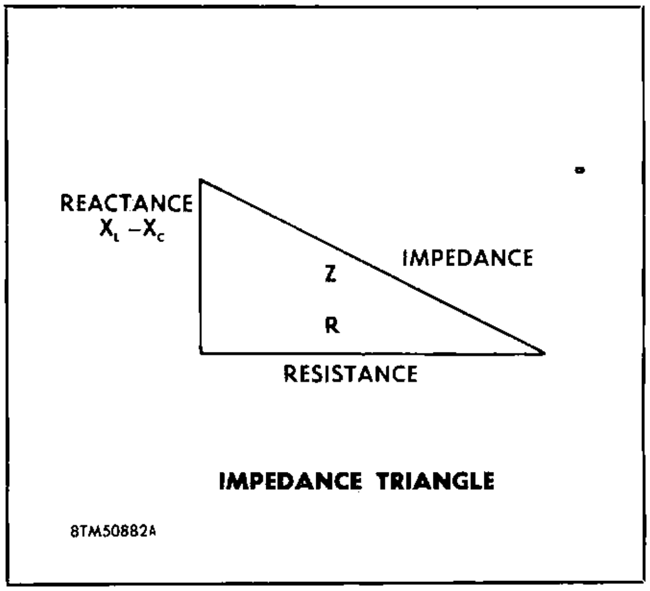


Figure 4-10. The Impedance Triangle

Before going any further, it is suggested that it might be a very good idea if you turn back and re-read Chapter I, paying special attention to Lentz's Law found on page 23. This is especially applicable when studying alternating-current circuits which contain inductance.

INDUCTANCE.

We explained in Chapter I, when we were talking about the fundamental a-c generator, that when an alternating current flows through a coil of wire, the rise and fall of the current flow, first in one direction and then in another, sets up expanding and collapsing magnetic fields about the coil. This induces a voltage in the coil which is opposite in direction to the applied voltage and the flow of current, and opposes any change in the alternating current. It is sometimes called back emf, or more familiarly counter-voltage. Let's carry this a step farther.

You learned how two windings adjacent to one another will react upon each other by a process known as mutual inductance. You also learned how a single coil will react upon itself by a process known as self-induction and produce a counter voltage. This was illustrated in figure 1-24, on page 24. This property of a coil to oppose any change in the current flowing through it is called *inductance.*

Inductive Reactance and the "Henry."

Inductance is indicated by the letter L — its unit of measurement is called the "henry."

In an a-c circuit containing inductance there is opposition to the flow of current in addition to the resistance normally present. The extent of this opposition depends on two things, the frequency of the applied voltage, and the amount of inductance that is present in the circuit. This opposition is what is known as "inductive reactance." Inductive reactance is indicated by the symbol X_L. The formula used for finding the inductive reactance is:

$$X_L = 2\pi (f \times L)$$

where

$2(\pi) = 6.28$

f = the frequency in cycles per second

L = the inductance in henrys

As you have learned in Chapter I, because of the nature of a counter emf there is no actual loss of electrical energy. Therefore, even though inductive reactance is an opposition to a-c current flow, its result is not loss. But it does require a greater applied voltage to overcome this additional opposition. It is due to the opposition of inductive reactance that current lags the voltage in an a-c circuit. But it should be clear that by multiplying the instantaneous values of the voltage and current together, when this out-of-phase condition exists, that the power output is greatly diminished. And if the circuit is purely inductive, the current would lag the voltage by 90°.

Inductance in Series and Parallel A-C Circuits.

You remember that in d-c circuits resistances in series are added to find the total resistance in the circuit. This will be found beginning on page 40 if you wish to review or compare methods of computing values in a-c and d-c circuits. In the identical manner, the total reactance in an a-c circuit is found by adding the individual reactances. Let's illustrate this:

$$(X_L)_T = (X_L)_1 + (X_L)_2$$

To find the total reactance of inductors in a-c parallel circuits, the following formula is applied.

$$(X_L)_T = \frac{1}{\frac{1}{(X_L)_1} + \frac{1}{(X_L)_2} + \frac{1}{(X_L)_3} + \frac{1}{(X_L)_4}}$$

$(X_L)_T$ = Total Reactance

$(X_L)_{1,\ etc.}$ = Individual Reactances

CAPACITANCE.

In Chapter I, page 22, you were introduced to a capacitance, and on page 27 capacitors were discussed at length. But because capacitance is such an important property in a-c circuits, it is felt that you should have more knowledge of it. For that matter, most of what we have been discussing thus far in this chapter has been covered in some degree in Chapter I, or II or III.

Inductance is the property of a coil in an a-c circuit. Capacitance is the property of a capacitor. The unit of capacitance is called the farad. As previously explained in Chapter I, page 27, a capacitor is a device having the ability to store, or hold, a charge of electricity. When placed in an alternating-current circuit, it stores electricity on one alternation (½ cycle) and when the current reverses polarity on the other alternation, the capacitor becomes momentarily discharged and then recharges in the other direction on the second alternation. Figure 4-9 is a simple visual explanation of an a-c circuit containing a capacitor.

As you can see, the plate of the capacitor alternately changes polarity. In a circuit where there is only capacitance, the current leads the impressed voltage. This is in direct contrast with a circuit containing pure inductance, when the current lags the voltage. This is illustrated on page 22.

Capacitive Reactance.

Capacitance, like inductance, offers opposition to the flow of an alternating current. This opposition is called *capacitive reactance.* Capacitive reactance is measured in ohms, just as inductive reactance is measured. It is designated, however, by the symbol X_c.

$$X_c = \frac{1}{2\pi \times f \times C}$$

where

$2\pi = 6.28$

f = The frequency in cycles per second

but

C = The capacitance in *farads*

The Effects of Inductive and Capacitive Reactance.

Let's get back for a moment to our formula for capacitance reactance and further compare it with our formula for inductive reactance.

By comparing the two formulas, we can see that while X_L is directly proportional to the frequency and inductance, X_c is inversely (the exact reverse) proportional to the frequency and capacitance. Another way of saying the same thing, X_L increases as the frequency and inductance increase, and X_c decreases as the frequency and capacitance increase. This simply means that since inductive and capacitance reactance act in opposite directions, one can be used to cancel out the effects of the other. How is it accomplished? Well, we know that if a power circuit contains a large value of inductance, it will cause the current to lag the voltage; and by the same token, we know that too much capacitance will cause the current to lead the voltage. Therefore, by adding just enough capacitance to the circuit to counteract the effects of the inductance we can bring the current and voltage back in phase. This is usually done in a-c circuits and causes the apparent power and true power to be equal.

IMPEDANCE.

Impedance is the total opposition to the flow of alternating current in a circuit — the combined effect of the total reactance and the resistance. The total reactance is the difference between X_L and X_c. The symbol for impedance is Z. Because it opposes current flow, it has the same unit of measurement as resistance — the ohm. This is perhaps more clearly shown in the impedance triangle in figure 4-10.

ALTERNATING-CURRENT CIRCUITS.

This section, as its heading implies, deals with Ohm's law as it relates to a-c circuits; and at the same time, it will be a sort of resume of what we have learned so far concerning the peculiarities of a-c circuits. As a whole, we will be dealing purely with simple mathematical equations for finding the various values peculiar to a-c circuits. If you know these, and understand them, you will have no trouble in understanding of alternating current. It is recommended that you work these problems out yourself, even though the method and answers are shown. A complete manual, and longer than this one, could be written about "trouble-shooting," but if you didn't get out and actually trouble-shoot, the manual would be worthless, and you would be worthless as a flight-line mechanic. So, it is the "doing" that counts, and with it comes interest in what you are doing and interest builds efficiency, and with efficiency comes promotion.

Problems In A-C Circuits.

As a beginning, let's take our formula for impedance:

$$I = \frac{E}{Z}$$

And let's say we have a series circuit containing a lamp with 11-ohms resistance connected across a source of voltage, figure 4-11.

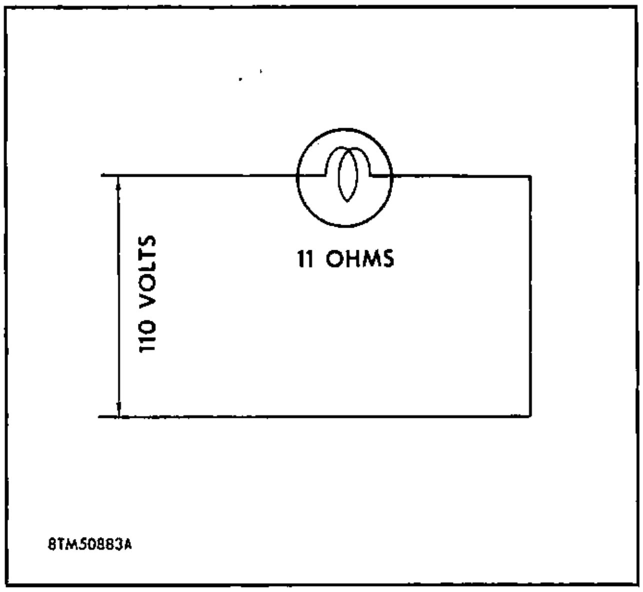


Figure 4-11. A-C and D-C Voltage in a Series Circuit

For the sake of comparison, let's find how much current will flow if 110-volts DC is applied and how much current will flow if 110-volts AC is applied.

If you will turn to page 39, and take a look at the Ohm's law chart for d-c circuits you will see that to find I (amperes) the formula is:

$$I = \frac{E}{R}$$

So in this case:

$$I = \frac{110}{11}$$

or

$$I = 10 \text{ amperes DC}$$

If AC is applied the formula would be:

$$I = \frac{E}{Z}$$

where Z (impedance) is equal to R (resistance, by substituting):

$$I = \frac{110}{11}$$

or

$$I = 10 \text{ amperes AC}$$

As you can see, this simple circuit contains resistance only, which bears out a previous statement made: that if a circuit contains resistance only, the current flow is the same regardless of whether the applied voltage is alternating current or direct current.

Now, let's apply Ohm's law to an a-c circuit containing inductive reactance whose symbol is X_L. The formula would then look like this:

$$I = \frac{E}{X_L}$$

And suppose we have an a-c series circuit, figure 4-12, in which the inductance is 0.146 henrys and the voltage is 110 volts at a frequency of 60 cycles per second.

Now, just what would be the flow of current in this circuit? Well, first of all, the inductive reactance must be found, and we know that the inductance as expresed in henrys is $L = 0.146$; so by substitution in our formula for finding the inductive reactance we would have

$$X_L = 2\pi \times f \times L$$

$$X_L = 6.28 \times 60 \times 0.146$$

$$X_L = 55 \text{ ohms}$$

so to find the current,

$$I = \frac{E}{X_L}$$

$$I = \frac{110}{55}$$

$$I = 2 \text{ amperes}$$

In the same manner, to find the capacitive reactance and the current flow in a circuit we would find the reactance first and then the current.

Let's assume that in another series circuit, figure 4-13, there is an impressed voltage of 110 volts at 60 cycle (~) per second and that in the circuit there is a condenser whose capacitance is 80 uf.

You learned that one million microfarads are equal to one farad. This means that to change 80 uf to farads we divide by 80 by 1,000,000. The quotient, or the answer to this problem in long division, is 0.000080 farads. All you must do to arrive at this, is to watch your decimal points.

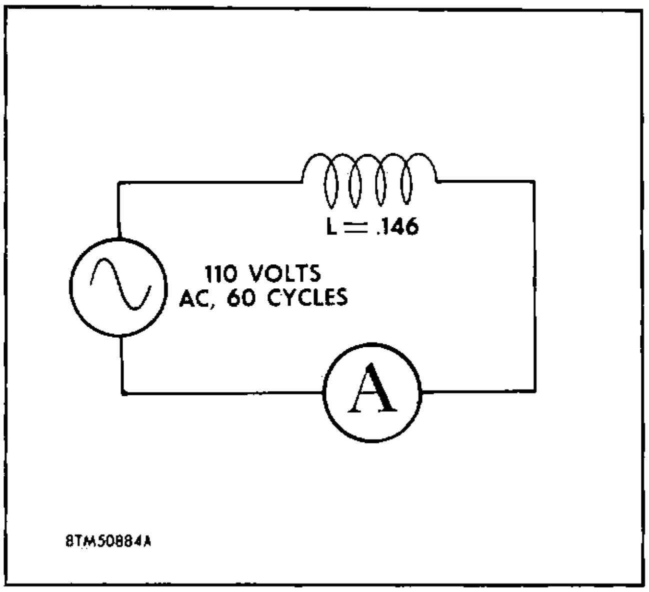


Figure 4-12. A-C Circuit and Inductance

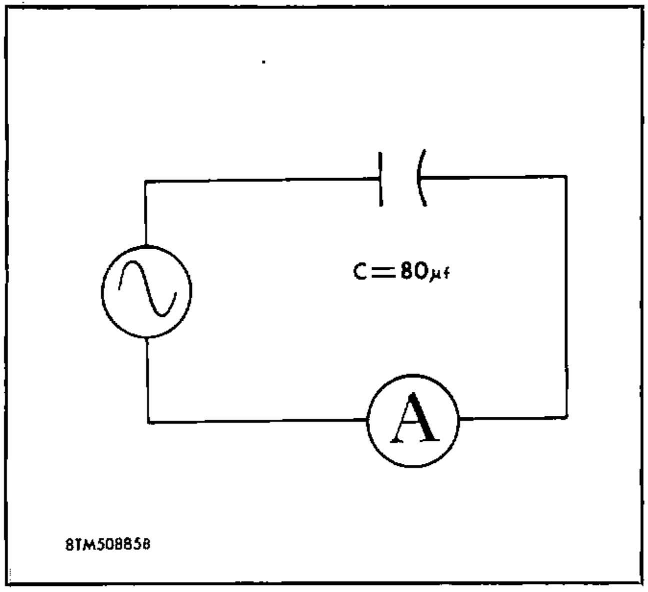


Figure 4-13. Capacitance in a Circuit

By substituting in our equation we find that

$$X_c = \frac{1}{6.28 \times 60 \times 0.000080}$$

which is equal to $X_c = 33.2$ ohms of reactance.

Knowing the capacitive reactance and by checking our formula chart, we find that to solve for the current flow in a capacitive reactance circuit we use

$$I = \frac{E}{X_c}$$

and by substitution,

$$I = \frac{110 \text{ volts}}{33.2 \text{ ohms}}$$

therefore,

$$I = 3.31 \text{ amperes}$$

The circuit problems used above have been comparatively simple, but suppose an a-c circuit contained inductance or capacitance and Z the impedance could no longer be simply substituted for the resistance R in Ohm's law for computing current flow.

In cases such as these, two entirely new formulas are used to find the impedance. In an a-c circuit containing resistance and inductance the formula is:

$$Z = \sqrt{R^2 + X_L^2}$$

And if the circuit contains resistance and capacitance,

$$Z = \sqrt{R^2 + X_c^2}$$

To illustrate our first formula, suppose we draw an a-c circuit, figure 4-14, which contains both resistance and inductance.

In this circuit resistance (6 ohms) and inductance (0.021 henrys) is connected in series with a 110-volt-at-60-cycle supply. Our problem is to find the impedance and the current through the lamp and the coil.

In order to do this we must first compute the inductive reactance of the coil. Why? Because the impedance is the combined effect in this case of resistance and the inductive reactance. So, we use our formula for computing inductive reactance,

$$X_L = 2\pi \times f \times L$$

and substitute accordingly:

$$X_L = 6.28 \times 60 \times 0.021$$

therefore,

$$X_L = 8 \text{ ohms of inductive reactance}$$

Our next step is to compute the total impedance, and by using our first formula,

$$Z = \sqrt{R^2 + X_L^2}$$

we substitute

$$Z = \sqrt{6^2 + X_L^2}$$

therefore,

$$Z = \sqrt{6^2 + 8^2}$$

$$Z = \sqrt{36 + 64}$$

$$Z = \sqrt{100}$$

and by taking the square root,

$$Z = 10 \text{ ohms of impedance}$$

The current flow would then be solved by Ohm's law method,

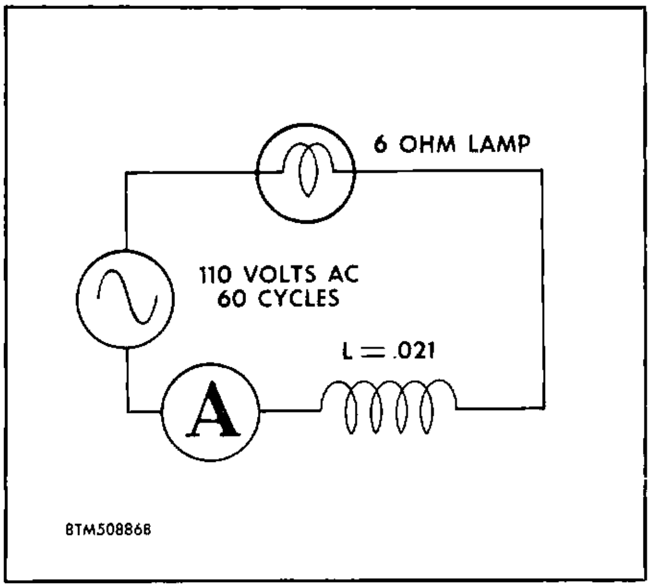


Figure 4-14. Resistance and Inductance in an A-C Circuit

$$I = \frac{E}{Z}$$

$$I = \frac{110}{10}$$

$$I = 11 \text{ amperes}$$

But let's find out how much of a voltage drop there is across the resistance in this circuit. And after that how much there is across the inductance.

In Chapter II you learned that when there is a voltage drop in a d-c circuit it is equal to the sum of the voltages across each of the resistances and that the method of solving such a problem was by using the formula taken from the Ohm's law chart for d-c circuits,

$$E = I \times R$$

In an a-c circuit, then E_R would represent the voltage drop across the resistance. Therefore, E_R would be substituted for E in the above formula, and we would have

$E_R = I \times R$

$E_R = 11 \times 6$

or

$E_R = 66$ volts, the voltage drop across the resistance

The drop across the inductance would be expressed, E_{XL}. So, our formula would now be

$E_{XL} = I \times R_{XL}$

therefore,

$E_{XL} = 11 \times 8$

or

$E_{XL} = 88$ volts, the voltage drop across the inductance

Now just what does this mean? Here we have a 66-volt drop across the resistance and an 88-volt drop across the inductance. The sum of these two voltages, we find by simple addition, is 154 volts—44 volts greater than the impressed voltage! Actually the reason for this is quite simple. It is the result of the two voltages being out of phase.

The mathematical analysis of the circuits shows clearly that in a-c circuits, unlike d-c, the amount of current flow depends not only on the resistance but also on the inductance or the capacitance in the circuit. As a matter of fact, in an a-c circuit either or both of these electrical properties have far more influence on the current flow than resistance, and as you have seen, in some cases, almost entirely control the current.

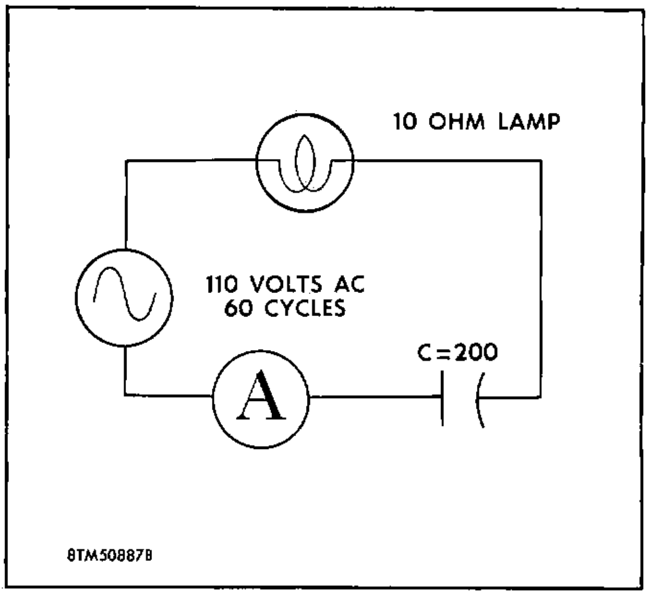


Figure 4-15. Capacitance and Resistance in an A-C Circuit

If you will compare our computations here with those relating to d-c circuits in Chapter II, pages 38 through 48, it should be plain enough that resistance has practically the same effect in an a-c circuit as it has in a d-c circuit. The difference found in the effects of inductance and capacitance in d-c and a-c circuits results largely from the fact that a-c constantly changes in direction and value. This constant change causes a varying in the magnetic and electrostatic fields. The magnetic field is, of course, associated with inductance, the electrostatic field with capacitance.

When there is capacitance in an a-c circuit as well as resistance, we use our second formula to find the total impedance:

$$Z = \sqrt{R^2 + X_C^2} \text{ and the formula } X_C = \frac{1}{2\pi \times f \times c}$$

to find the capacitive reactance. In the series circuit in figure 4-15, there is a capacitor of 200 uf connected in series with a 10-ohm lamp:

Let's find the total impedance, the amount of current, and the voltage drop across the resistance.

First of all, just as you did in the last problem, the uf quantity must be changed back into farads:

$$200 \text{ microfarads} = \frac{200}{1{,}000{,}000} = .000200 \text{ farads.}$$

By substitution in our formula, to find the capacitive reactance:

$$X_C = \frac{1}{2\pi \times f \times c}$$

$$X_C = \frac{1}{6.28 \times 60 \times .000200}$$

$$X_C = \frac{1}{.07536}$$

$X_C = 13$ ohms, the capacitive reactance

To find the total impedance, we now use the impedance formula for circuits containing capacitance and resistance, which, as you know, is

$$Z = \sqrt{R^2 + X_C^2}$$

and by substituting,

$$Z = \sqrt{10^2 + 13^2}$$

$$Z = \sqrt{100 + 169}$$

$$Z = \sqrt{269}$$

$Z = 16.4$ ohms, the total impedance

To solve for current flow in this circuit, we return to the Ohm's law:

$$I = \frac{E}{Z}$$

$$I = \frac{110}{16.4}$$

$I = 6.7$ amperes

And to find the voltage drop across the lamp,

$$E_R = I \times R$$

$E_R = 67$ volts

The voltage drop across the capacitor would in this case be expressed E_{XC};

therefore,

$$E_{XC} = 1 \times X_C$$

$$E_{XC} = 6.7 \times 13$$

$E_{XC} = 86.1$ volts

As in our circuit containing resistance and inductance but no capacitance, we find that these two voltages do not equal the supposed applied voltage of 110 volts. Here again you have a circuit in which E and I are out of phase, but instead of lagging, the voltage being a capacitive circuit, the current is leading the voltage. The voltage would be expressed E_T. So to find the applied voltage you would use the equation

$$E_T = \sqrt{E_R{}^2 + E_{XC}{}^2}$$

$$E_T = \sqrt{67^2 + 86.1^2}$$

$$E_T = \sqrt{4489 + 7413}$$

$$E_T = \sqrt{11902}$$

$E_T = 110$ volts

Now, how about a circuit which contains not only resistance and capacitance but also inductance? See figure 4-16. Since X_L and X_C tend to cancel each other out, they would be expressed $(X_L - X_C)$. Therefore, the equation for a circuit containing resistance, inductance, and capacitance is:

$$Z = \sqrt{R^2 + (X_L - X_C)^2}$$

By substituting, you would, in this case, solve for the total impedance. As an example, suppose we have a series circuit which looks something like figure 4-16. To find the impedance,

$$Z = \sqrt{R^2 + (X_L - X_C)^2}$$

$$Z = \sqrt{4^2 + (10 - 7)^2}$$

$$Z = \sqrt{4^2 + 3^2}$$

$$Z = \sqrt{16 + 9}$$

$$Z = \sqrt{25}$$

$Z = \quad 5$ ohms

As you should now understand, many of the ideas you absorbed in the study of d-c circuits are applicable to a-c circuits. Those features that you have found different are caused by the constant changing of the direction and the magnitude of alternating current.

The rules and equations for d-c circuits as you have discovered apply to a-c circuits when those circuits contain resistances alone, such as lamps, a coil, or heating element. So, in order to use the effective values of voltage and current in a-c circuits, the effect of inductance and capacitance must be taken into consideration as well as the resistance. And the combined effect of resistance,

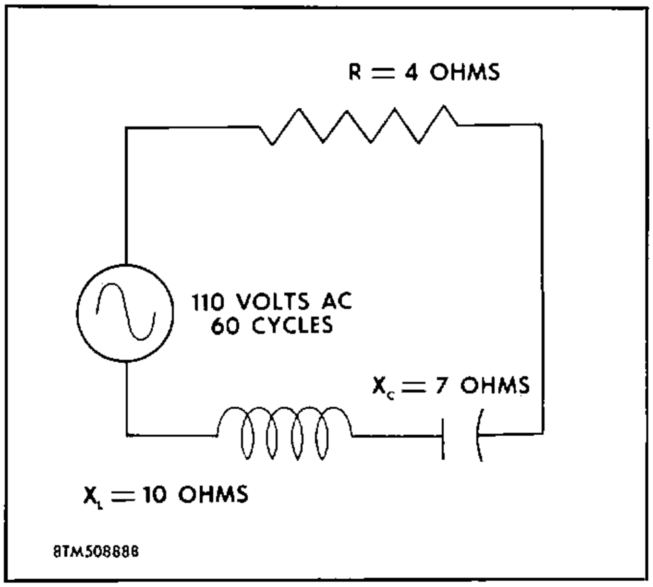


Figure 4-16. Resistance, Inductance, Capacitance in an A-C Circuit

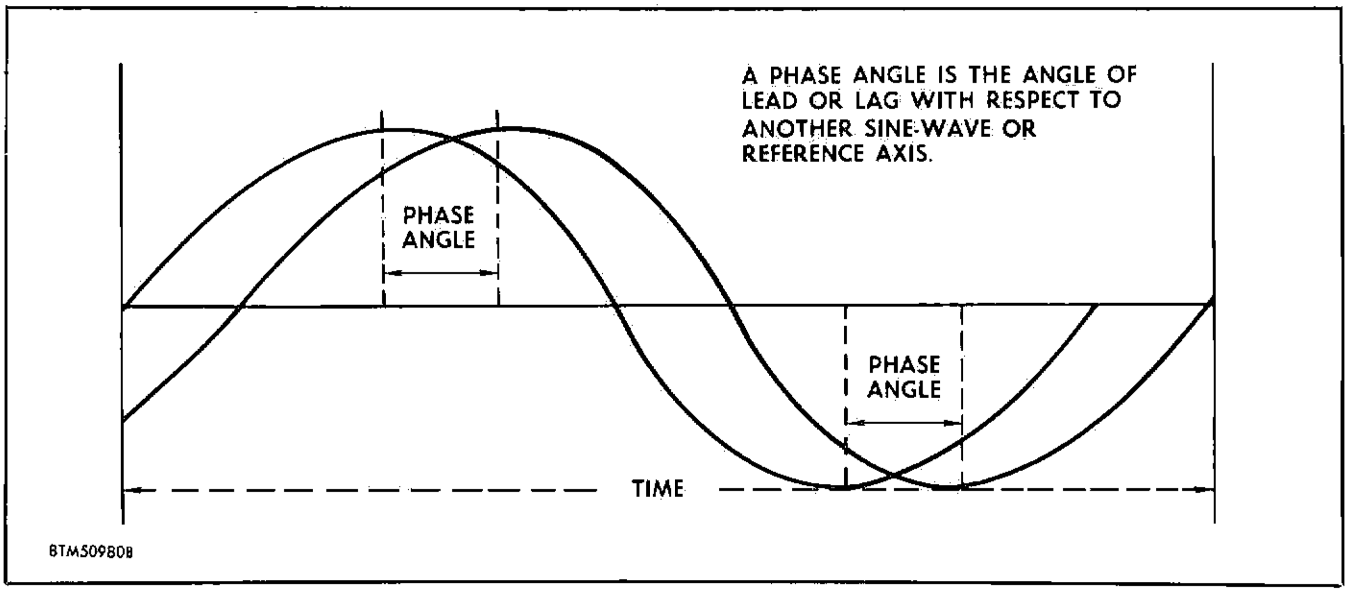


Figure 4-17. Phase Angle

inductive reactance, and capacitive reactance makes up the total opposition to the flow of our electrons in an a-c circuit.

But as you have gathered through your computations of the above circuit problems, it is either capacitive reactance or inductive reactance which predominates in the alternating-current circuit.

When we discussed apparent power, you learned that in an a-c circuit the ratio between the apparent power and the true power is what is known as the power factor; that it is usually expressed in percent; and that the formula is written

$$\text{Power Factor} = \frac{\text{true power}}{\text{apparent power}}$$

The reason for this equation being written thus is that since the difference between true power and apparent power is directly caused by the phase separation of the voltage and the current in an a-c circuit, it would seem logical to expect a very definite relationship between the phase angle, the true power, and apparent power. And to find the phase angle, which is illustrated in figure 4-17, we must divide the true power by the apparent power — the apparent power (volt-amperes) being the power delivered (watts) to the circuit, the true power or effective power being the power actually consumed by the circuit.

WATTLESS POWER—VARS.

In Chapter II we touched on power as it relates to the direct-current circuit. The heating effect of electrical current is electrical power at work, and the unit of electrical power is the watt. Electrical power is the time rate at which electrical energy in a circuit is expended. In AC as in DC we have the watt as a unit of power, but we have also another type of power called "wattless power," which is expressed in vars, or the *volt-amperes reactive.* For example, when we get to A-C Load Analysis in this chapter you will find that in takeoff and climb the F-102A uses 20543 watts, 9236 vars, and 22.48 kva.

The presence of either capacitive reactance or inductive reactance, as you have learned, will cause the current and voltage to be separated by some angle between 0 and 90°. But, and remember this, it will always be less than 90° because all circuits contain some resistance to current flow.

The power which results from the applied voltage and the particular element of the current in phase with it is the actual power dissipated in the circuit. It should be noted, that all applied power dissipated by an electrical device is not, however, delivered in the form of useful power. Some of the applied energy is lost in heat; this loss is expressed I^2R; it is caused by the current flowing in the resistance of the circuit. The loss represents a direct conversion of electrical energy into heat energy in the resistance.

That part of the current not transformed into heat is in this manner 90 degrees out of phase with the applied voltage. It is this current which produces the magnetic field in inductive devices such as coils. And this degree of power which produces this field is what is known as our "wattless power," "reactive power," and is expressed in vars. Actually it is not power at all because it consumes no energy, since all the energy stored in magnetic or electrostatic fields is returned to the source of power when the fields collapse. And since none of this energy was converted into kinetic energy, except those I^2R

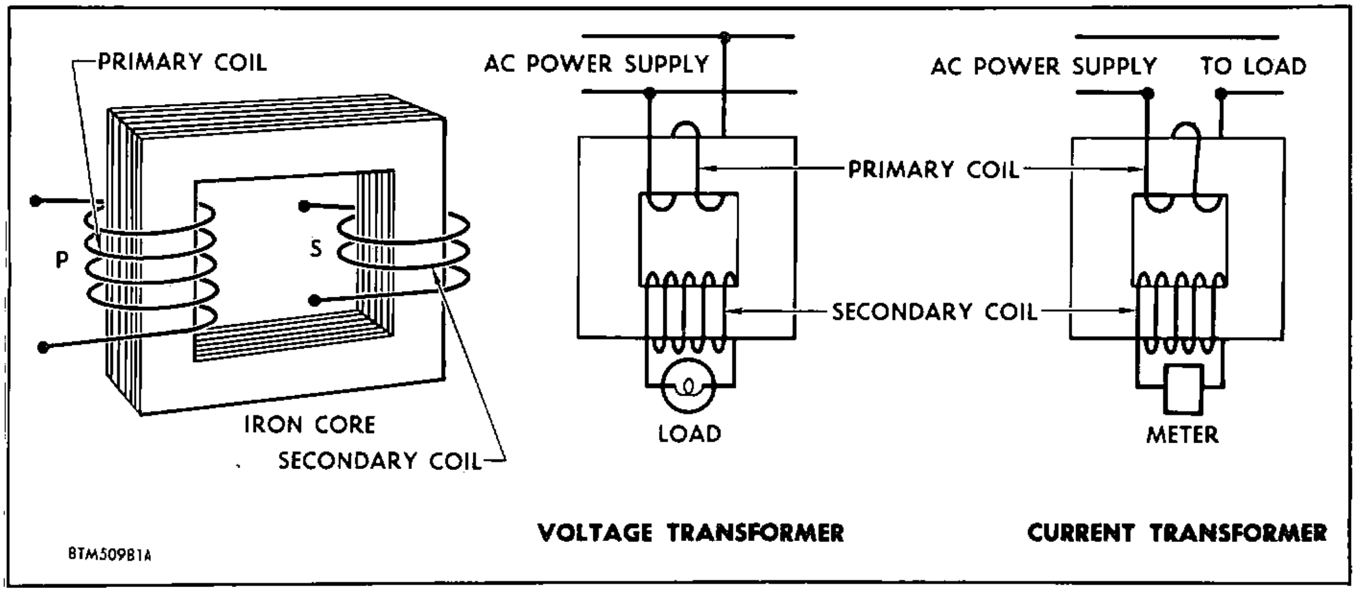


Figure 4-18. Basic Windings of Transformers

losses associated with the induced current flow, there has been no consumption of power. It is for this reason that vars or volt-amperes reactive are spoken of as "wattless power."

TRANSFORMERS.

A transformer is a device which steps up or steps down an a-c voltage. It consists of two coils which are not electrically connected, but are so arranged that the magnetic flux surrounding one coil threads through, or cuts, the other coil. In other words, when an alternating current flows through one coil, the varying magnetic flux creates an alternating voltage in the other winding by mutual induction. Mutual induction was covered in Chapter I.

THE PRIMARY PARTS.

A transformer consists of three primary parts — an iron core which provides a circuit of low reluctance for the magnetic flux, a primary winding which receives the electrical energy from the supply source, and a secondary winding which receives electrical energy by induction from the primary and delivers it to the secondary circuit.

To obtain maximum inductive effect between them, the primary and secondary coils are usually wound one upon the other on a closed core. The turns of insulated wire and layers of the coil are well insulated from each other by layers of paraffin-impregnated paper or mica. The iron core is laminated to minimize magnetic current losses, known as eddy losses, and is usually made of specially prepared silicon steels, since these steels have a low hysteresis loss. And if you remember, hysteresis loss is due to heat caused by molecular friction when a magnet reverses its polarity.

CLASSES OF TRANSFORMERS.

There are voltage transformers for stepping up or stepping down voltages, and current transformers, which are generally used in instrument circuits. In voltage transformers, the primary coils are connected in parallel across the supply voltage, as shown in the illustration below. In current transformers, however, the primary windings are connected in series in the primary circuit. Examples of these internal windings are shown in figure 4-18.

Of the two types, the voltage transformer is the more common. There are also power distributing transformers, such as the one found in the voltage regulator of the F-102A, which are used with high voltages and heavy loads. Transformers are usually rated, just as a-c generators are rated, in kilovolt-amperes.

How a Transformer Works.

When an a-c voltage is applied across the primary terminals of a transformer, an alternating current will flow and self-induce a voltage in the primary coil which is opposite and nearly equal to the connected voltage. The difference between these two voltages will allow just enough current to flow in the primary coil to magnetize its iron core. This is called the exciting, or magnetizing current. The magnetic field caused by the exciting current cuts across the secondary coil and induces a secondary voltage by mutual induction. If a load is connected across the secondary coil of the transformer, the load current flowing through the secondary coil will produce a magnetic field which tends to neutralize the magnetic field produced by the primary current. This reduces the self-induced, the opposition voltage, in the primary coil and allows more primary current to flow. The primary current increases as the secondary load current

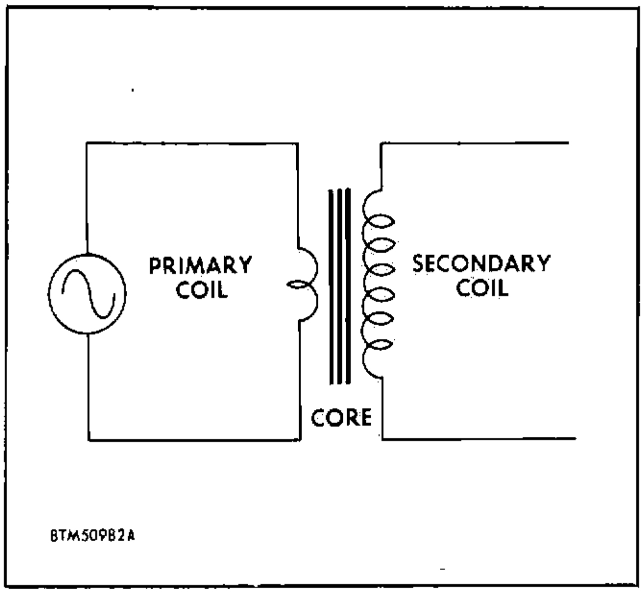


Figure 4-19. Step-Up Transformer

increases; it decreases as the secondary load current decreases. When the secondary load is removed, the primary current is again reduced to the small exciting current sufficient to magnetize the transformer iron core.

Turns Ratio.

To accomplish changing voltage from one value to another, transformers are designed so that there are more turns on one coil than on the other. If the primary coil, for example, has 200 turns and the secondary 1000 turns, the voltage available in the secondary terminals will be $\frac{1000}{200}$, or 5 times as great as the voltage impressed on the primary. This ratio of the number of turns in the secondary coil (N_2), to the number of turns in the primary (N_1), is called the ratio of transformation and is expressed mathematically like this:

$$r\ (\text{ratio}) = \frac{N_2}{N_1} = \frac{E_2}{E_1}$$

where E_2 and E_1 are the respective voltages of the two windings.

Step-Up Transformers.

When a transformer delivers a higher voltage than the applied voltage, it is called a step-up transformer. Using the ratio of transformation equation, you can determine the output voltage of a step-up transformer for any value of input voltage. A simple wiring schematic of a step-up transformer is shown in the shaded portion of figure 4-19. Thus, if 110-volt AC is impressed on the 200-turn primary of a transformer with a 1000-turn secondary, you can determine the output voltage by the equation:

$$r = \frac{N_2}{N_1} = \frac{E_2}{E_1}$$

$$r = \frac{1000}{200} = \frac{E_2}{110}$$

$$r = 5, \text{ the turns ratio}$$

since,

$$E_2 = E_1 \times r$$

$$E_2 = 110 \times 5$$

$$E_2 = 550 \text{ volts}$$

Step-Down Transformers.

When a transformer lowers the voltage, it is called a step-down transformer. Its basic windings are shown in the unshaded part of figure 4-20. The ratio of transformation equation applies also to step-down transformers. Thus, if 110-volt AC is impressed on the 1000-turn primary of a transformer with a secondary of 200 turns, you can determine the voltage output by substituting in the equation:

$$r = \frac{N_2}{N_1} = \frac{E_2}{E_1}$$

$$r = \frac{200}{1000} = \frac{E_2}{110}$$

$$r = \frac{1}{5} = \frac{E_2}{110}$$

since the turns ratio is 1 to 5,

$$E_2 = \frac{1}{5} \text{ of } 110$$

$$E_2 = 22 \text{ volts output}$$

Any desired amount of alternating voltage can be obtained by properly proportioning the number of turns

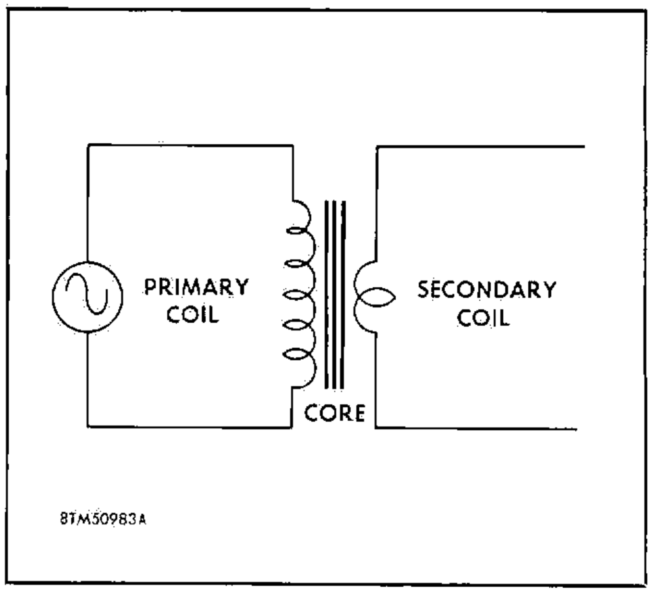


Figure 4-20. Step-Down Transformer

on the windings of a transformer. In transformers supplying more than the applied voltage, there are more windings on the secondary than on the primary. In transformers in which the output voltage is less than the applied voltage, there are fewer turns on the secondary winding than on the primary winding.

Power Losses In Transformers.

There are several losses in transformers. These losses are due to eddy currents and hysteresis losses in the core and I^2R losses in the primary and secondary windings. Eddy current losses result from currents induced in the iron core by an alternating current. Hysteresis losses result, as you know, because the flux in the core is constantly alternating and power is required to effect these reverses of polarity in the iron core. The I^2R losses are a product of the heat created in the windings, as you were told on page 162, is due to the resistance in the windings themselves. For most practical purposes, since the efficiency of a transformer is unusually high, these losses are not taken into consideration in calculating the voltage output of a transformer.

Power In Transformers.

Since a transformer does not add any electricity to the circuit but merely changes or transforms the electricity that already exists in the circuit from one voltage to another, the total amount of energy in a circuit remains the same. If it were possible to construct a perfect transformer, power would be transferred from one voltage to another with no loss in power. Power, however, is the product of volts times amperes; therefore an increase in voltage by means of the transformer must result in a decrease in current or vice versa. There cannot be more power in the secondary side of a transformer than there is in the primary. The product of amperes times volts remains the same. Thus the primary power is $E_p \times I_p$; the secondary power $E_s \times I_s$. And since there is no loss or gain in power, $E_p \times I_p$ is equal to $E_s \times I_s$. From this you can derive the equation

$$\frac{I_p}{I_s} = \frac{E_s}{E_p} = \frac{N_2}{N_1}$$

How Transformers Are Connected Into an A-C System.

Three-phase a-c generators and other three-phase loads are connected with their coils or load elements so arranged that three transmission lines are required for the delivery of power. Transformers that are used for stepping the voltage up or down in a three-phase circuit are electrically connected so that power is delivered to the primary and taken from the secondary by this standard three-wire system.

Single-phase transformers, however, and single-phase lights, as well as single-phase motors, may be connected across any one phase of a three-phase circuit. In order to balance the load on the three alternator coils, when single-phase loads are connected to three-phase circuits, the loads across all phases are made as nearly equal as possible.

A single-phase transformer can also be used with a two-wire single-phase system to step-down voltages fed to it from feeder lines. The stepped-down voltage is used to operate various electrical devices. The center wire in this transformer is grounded, as shown in the simplified schematic, figure 4-21.

A single-phase, three-wire system, figure 4-22, is used either in lighting circuits or in mixed lighting, motor combination; it is also used to step down voltage. The secondary coils may be tapped to obtain more than one

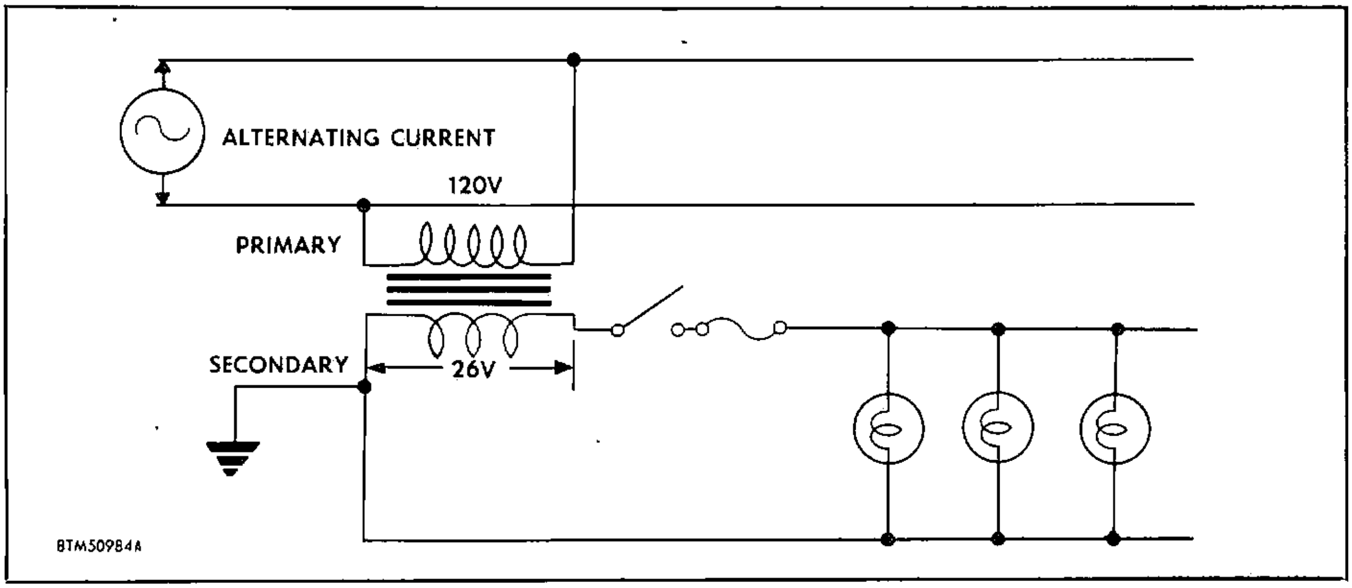


Figure 4-21. Step-Down Transformer—Two-Wire System

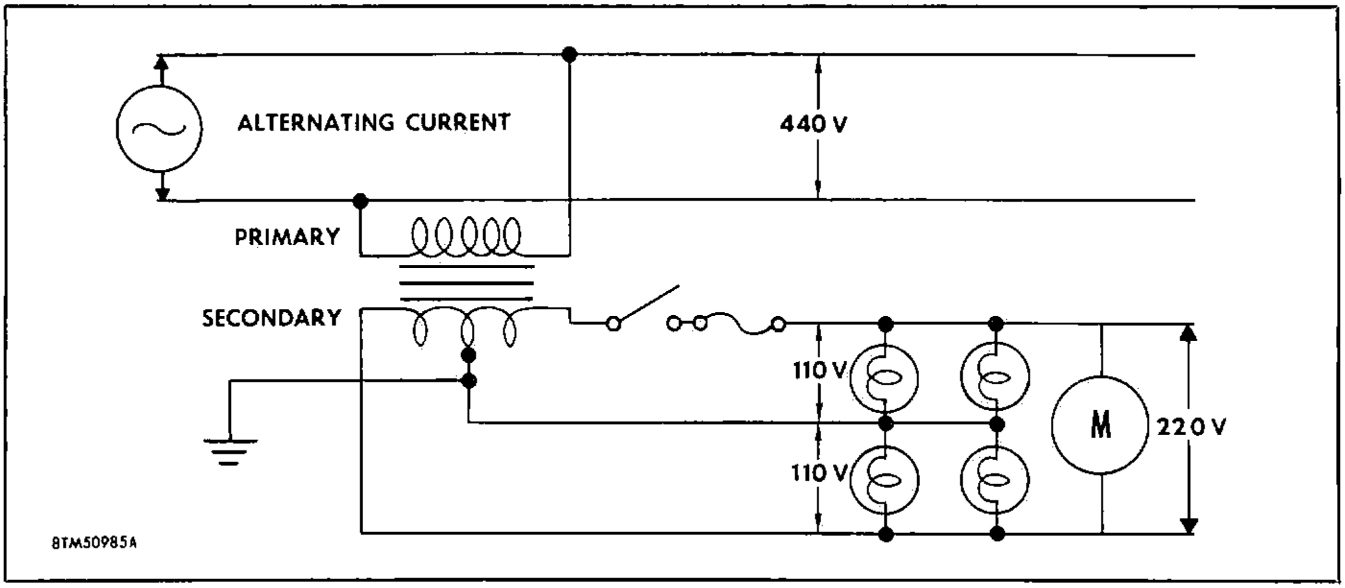


Figure 4-22. Step-Down Transformer in a Three-Wire System

voltage — the center wire is grounded. This type of hook-up is used in the F-102A instrument transformer, which steps the voltage down to 26 volts.

Transformers for three-phase circuits can be connected in one of several combinations of the wye (y) and delta (Δ) connections. The connection used depends on the requirements for the transformer.

The Wye Connection.

When the wye connection is used in three-phase transformers, a fourth or neutral wire may be used. This is shown in figure 4-23. The neutral wire serves to connect single-phase equipment to the transformer. Voltages (115v) between any one of the three-phase lines and the neutral wire can be used for power for devices such as lights or single-phase motors. In combination, all four wires can furnish power at 208 volts, three-phase, for operating three-phase equipment such as three-phase motors or rectifiers. When only three-phase equipment is used, the ground wire may be omitted. The system is then a three-phase, three-wire system.

The Delta Connection.

In figure 4-24, showing the primary and secondary with a delta connection, the transformer has the same voltage output as the line-to-line voltage. Between any two phases, the voltage is 240 volts. This type of connection,

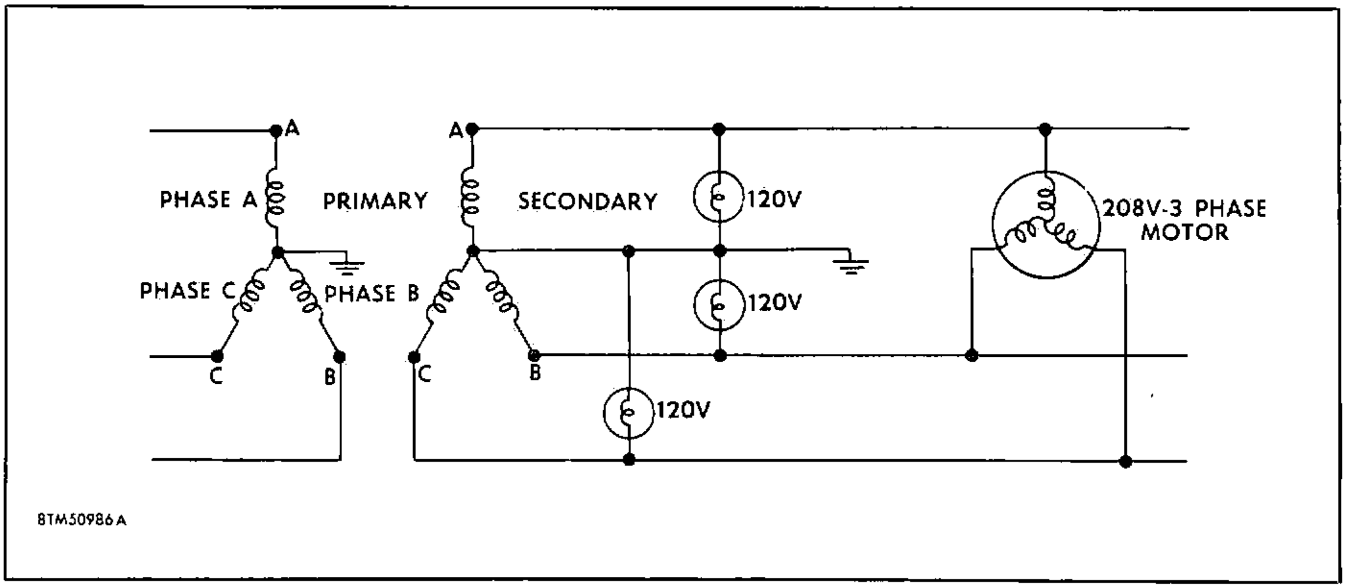


Figure 4-23. Wye Systems

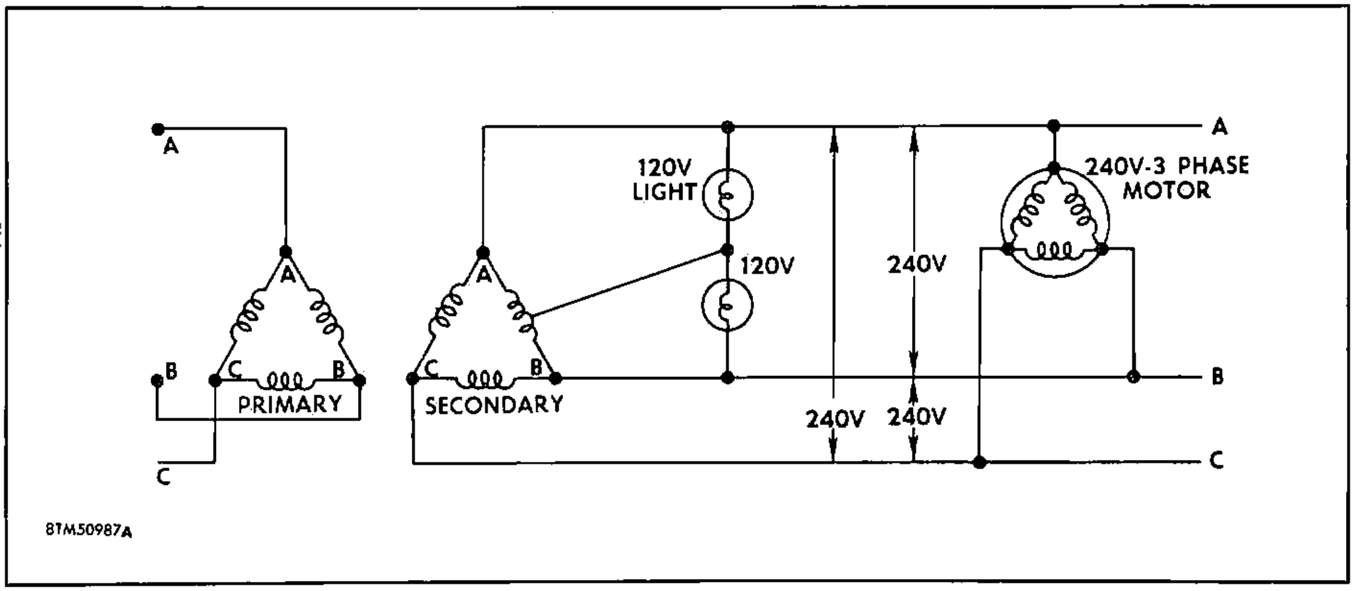


Figure 4-24. The Delta Connection

using the three wires — a, b, and c — can furnish 240-volt, three-phase power for the operation of three-phase equipment. To obtain single-phase, 120-volt power in a delta type distribution, the center taps may be taken off the individual phases. This system is not used as much as the wye system for single-phase power.

The Wye and Delta Connections.

The type connection used for the primary coils may or may not be the same as the type of connection used for the secondary coils. For example, the primary may be a delta connection and the secondary, a wye connection. This is called a delta-wye (Δ-y) connected transformer. Other combinations are delta-delta, wye-delta, and wye-wye. The wye-delta combination is one of those used in the F-102A and is shown in figure 4-25.

DRY-DISC RECTIFIERS.

The dry-disc rectifier is a source of low voltage at high amperage. A rectifier is a device which transforms alternating current into direct current by limiting or regulating the direction of current flow. The principal types of rectifiers used are dry-disc rectifiers and vacuum tube rectifiers.

Dry-disc rectifiers commonly used in airplane electrical systems are based on the principle that electrical current flows through a junction of two dissimilar conducting materials more easily in one direction than it does in the opposite direction. This is due to resistance to current flow in one direction being low, while in the other direction it is high. Depending, of course, on the material used, several amperes may flow in the direction of low resistance but only a few milli-amperes in the direction of high resistance.

Two of the types of dry-disc rectifiers most commonly used in aircraft are the copper-oxide rectifier, and the selenium rectifier. These are shown in figure 4-26.

The copper-oxide rectifier consists of a copper disc upon which a layer of copper oxide has been formed by heating. It may also consist of a chemical copper-oxide preparation spread evenly over the copper surface. Metal plates, usually lead plates, are pressed against the two opposite faces of the disc to form a good contact. Current flow is from the copper to the copper oxide.

The selenium rectifier consists of an iron disc similar to a washer, one side of which is coated with selenium. Its operation is similar to that of the copper-oxide rectifier. Current flows from the selenium to the iron.

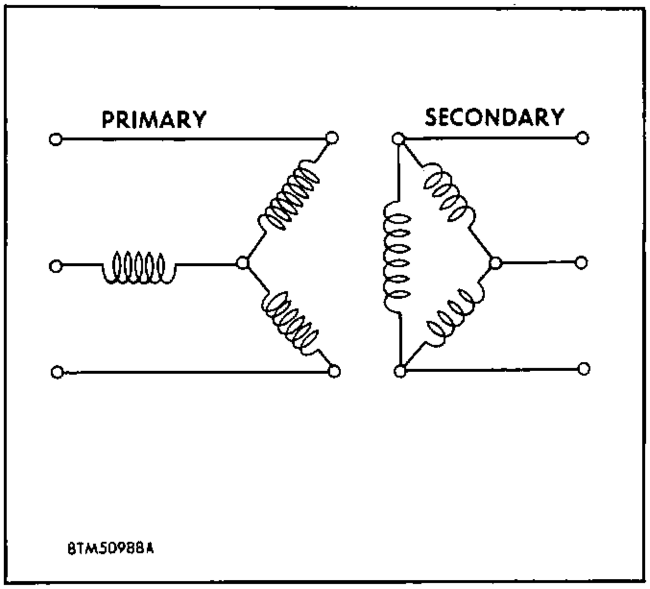


Figure 4-25. The Wye-Delta Connection

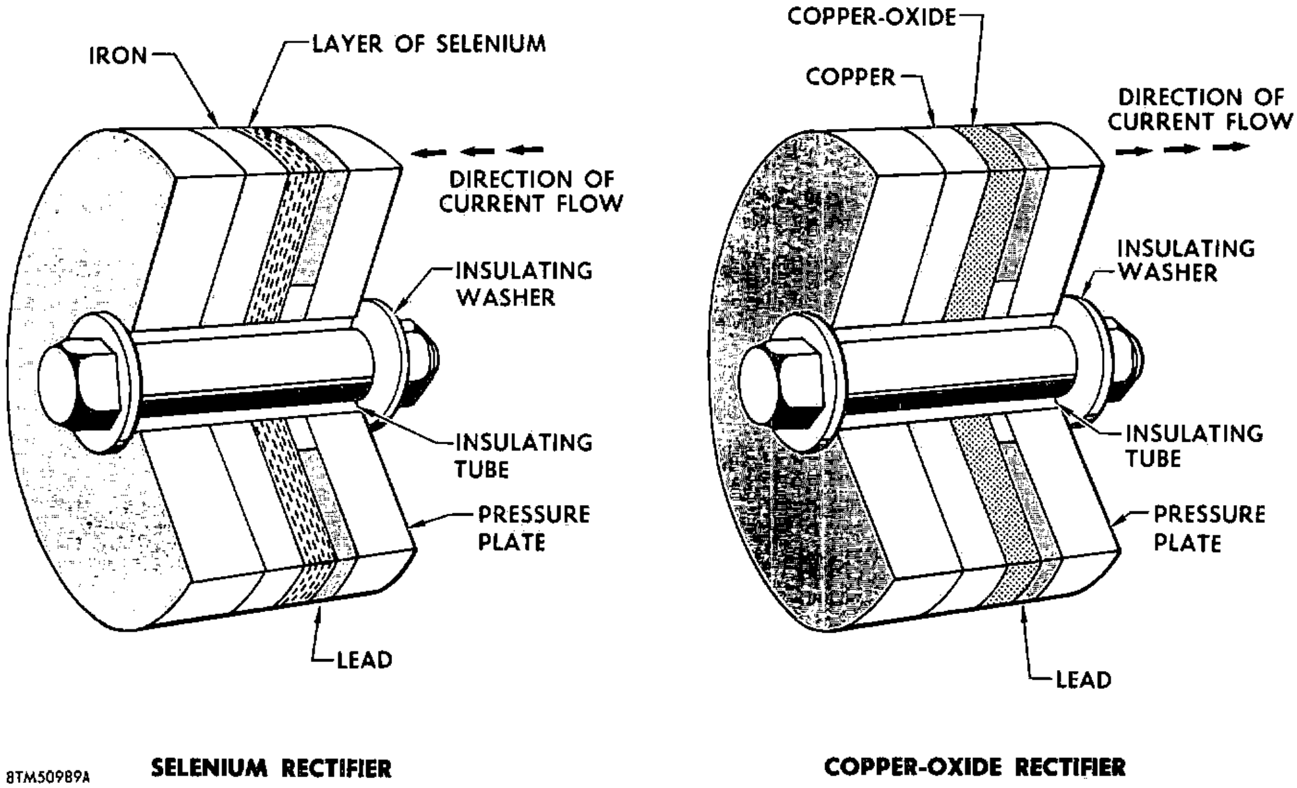


Figure 4-26. Dry-Disc Rectifiers

CHARACTERISTICS.

In practice, a number of discs may be placed together in series or parallel to form a rectifier assembly. When assembled, the pressure on the assembly plates is from 500 to 2000 pounds per square inch. The resistance of the rectifier increases about 25% during the first three months of operation; this increase in resistance of a new unit is known as aging. The maximum life of a dry-disc rectifier is undertermined. Many have operated for a number of years without any loss in efficiency. Metal plates much larger than the discs are usually assembled between each disc section to increase the heat-dissipating capacity.

To increase the voltage rating of a rectifier, a number of discs are connected in series. To increase the ampere capacity of a rectifier, a number of discs are connected in parallel. A series-parallel combination increases both the voltage rating and the ampere capacity of the rectifier.

THE DRY-DISC RECTIFIER CIRCUITS.

Dry-disc rectifiers are connected as half-wave or as full-wave rectifiers. It is most important that you understand the function of these circuits if you are to understand the voltage regulation of the F-102A a-c electrical power system.

In the half-wave circuit, current flows in the desired direction during only half of the a-c cycle. During the other half-cycle, the current is almost completely blocked.

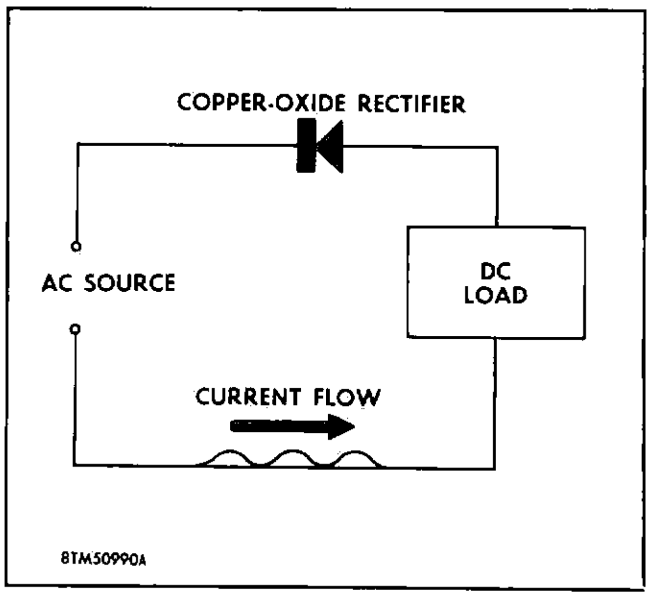


Figure 4-27. Half-Wave Dry-Disc Rectifier Circuit

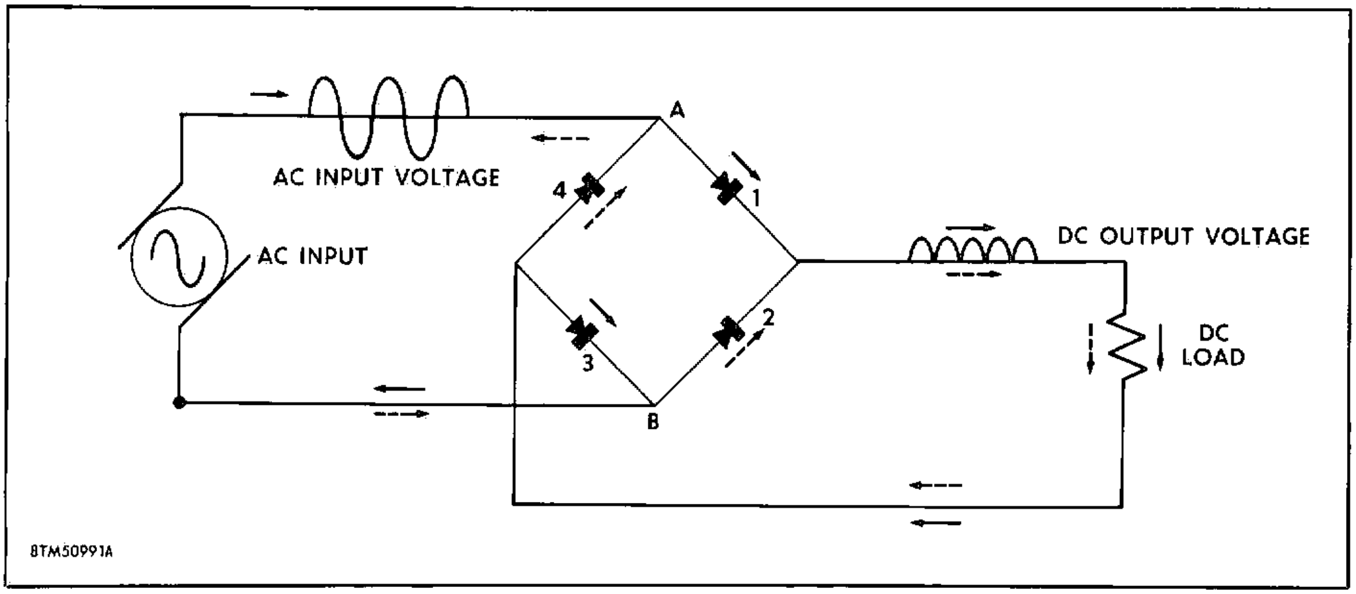


Figure 4-28. Full-Wave Dry-Disc Rectifier Circuit

The output of the half-wave rectifier is as shown in figure 4-28. The bottom half of the a-c wave has been cut out, and a series of pulses remain. The number of these pulses is equal to the frequency of the applied voltage—60-cycle voltage produces 60 pulses, a 400-cycle voltage produces 400 pulses per second.

A dry-disc rectifier for full-wave rectification is a bridge arrangement, as shown in figure 4-28, which is comparable to the arrangement in the F-102A voltage regulator. When point A in the rectifier is negative with respect to point B, sections 1 and 3 of the rectifier conduct current in the direction indicated by the solid arrows. A half cycle later when B is negative with respect to A, sections 2 and 4 conduct in the direction indicated by the dotted arrows. On both halves of the cycle, current flows in the same direction through the load.

In full-wave rectification, the number of pulses per second is equal to twice the frequency. Thus, for a 60-cycle current there are 120 pulses, for a 400-cycle current there are 800 pulses. Since the number of pulses, or the ripple frequency, is high, it is much easier to filter out the full-wave rectification pulses than the half-wave rectification pulses.

HOW DRY-DISC RECTIFIERS ARE USED IN A-C CIRCUITS.

Dry-disc rectifiers, particularly the copper-oxide and selenium types, are used in either single or three-phase circuits. Current to them is supplied by a transformer.

Figure 4-29 shows six rectifier units connected as a bridge circuit with a three-phase, step-down transformer, and a cooling fan. The resistor connected in parallel with the rectifier is used to iron out excess voltages during cycle change-over. The transformer rectifier assembly is used to supply 28-volt d-c to the fan from a 400-cycle 208-volt a-c system. You will find a similar hook-up in the voltage sensing circuit of the F-102A voltage regulator.

TROUBLE SHOOTING THE A-C POWER CONTROL SYSTEM.

In Chapter III, you remember, we discussed trouble shooting in general, as it applied to d-c power systems. In a-c systems there is little difference in the procedure. Trouble shooting charts for the a-c system will be found in T.O. 1F-102A-2-10. Procedures followed for operational check-outs will also be found in the T.O. 1F-102A-2-10 Handbook. There are few differences, however, in the instruments used to check out an a-c system, and for that reason they are included in this section.

A-C INSTRUMENTS.

If a d-c measuring device—say an ammeter—were connected into an a-c circuit, it would indicate zero. Why? Because a d-c ammeter reads only average values, and the average of an a-c circuit is zero. In an a-c circuit, the current attempts to go in one direction during one-half of the cycle and in the reverse direction during the other half. The direction of current flow, however, reverses too rapidly for the coil in the d-c ammeter to follow, and the coil takes an average position—zero. A meter with a permanent magnet cannot be used to measure either an alternating voltage or an alternating current.

For airplane electrical systems, there are several types of meters suitable for measuring a-c voltage and current —the electrodynamometer, the moving iron-vane, and the rectifier type. Power measuring instruments, such

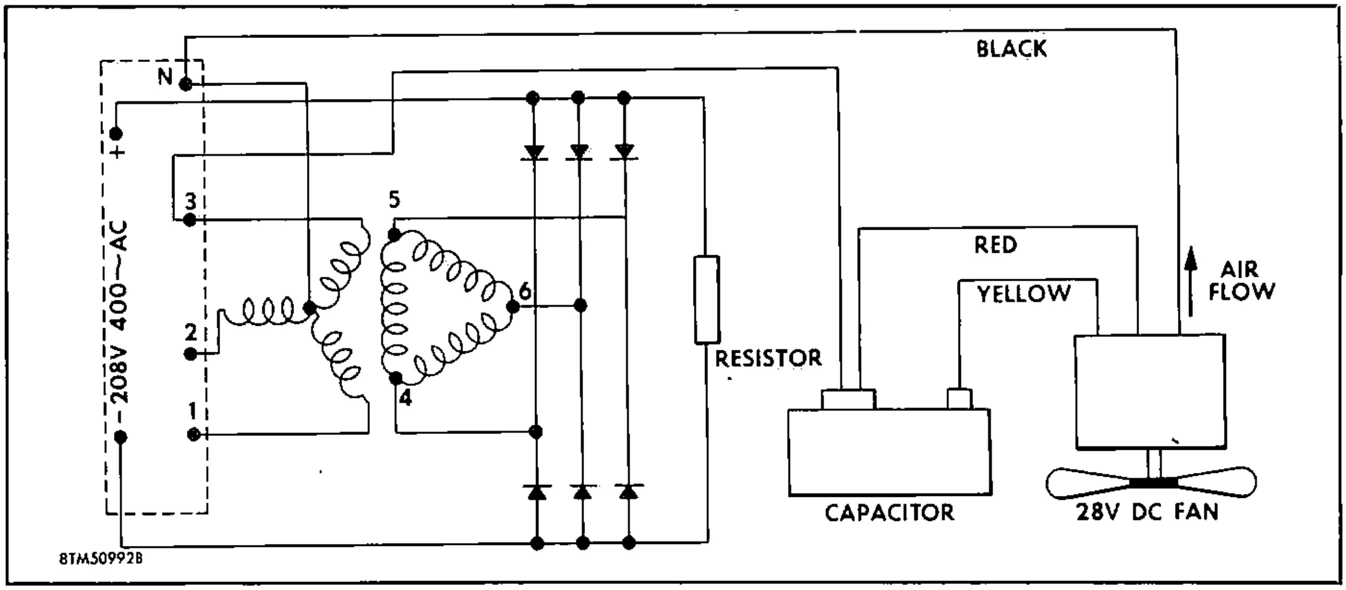


Figure 4-29. Three-Phase Transformer Rectifier Assembly

as wattmeters and varmeters, are usually electrodynamometer type meters. Frequency meters may be electrodynamometer or electromechanical. Electrodynamometer type ammeters and voltmeters are not only very accurate, they are sensitive. Moving iron-vane meters, although not quite as sensitive nor as accurate, are less expensive.

Electrodynamometer Meters.

The electrodynamometer type meter depends on the principle that whenever a current is caused to flow in a conductor, there is a magnetic field around the conductor which is proportional to the strength of the current flow.

Let's take a look at the simple drawing in figure 4-30. As you can see in the electrodynamometer type meter, the conductors are field coils and a movable coil. The field coils are split and fixed in position. The movable coil pivots between the two halves of the field coil so that its only movement is rotation—this rotation is opposed to a spring arrangement. In this way, when current flows in both the field and movable coils, the magnetic field polarities set up are such that the movable coil turns in a clockwise direction. Naturally, the more current that flows through the coils, the more easily the moving coil can overcome the opposition to the spring, and the farther it will rotate. With a pointer connected to the movable coil and a calibrating scale over which the pointer will move, the instrument can be used for measuring the reaction effect of the coils upon one another, and the current through the coils.

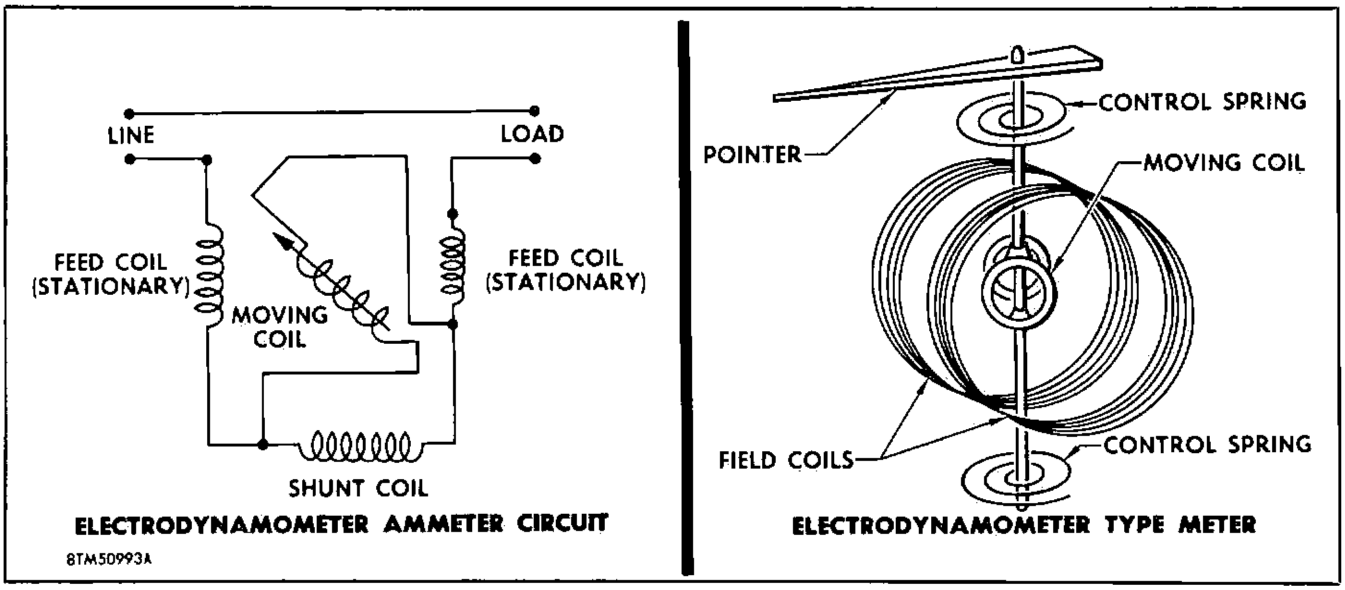


Figure 4-30. Electrodynamometer Type Meter and Circuit

Electrodynamometer Ammeter.

In the electrodynamometer type ammeter, a meter used for measuring alternating current, the coils are of low-resistance wire so that there will be a minimum voltage drop in the circuit measured. An inductive shunt is connected in series with the field coils, as shown in figure 4-30. This shunt, if you remember, is similar to the resistor shunt used in d-c ammeters, permitting only part of the current being measured to flow through the coils. As in the d-c ammeter most of the current in the circuit flows through the shunt, but the scale is calibrated accordingly and the meter, thereby, reads the total current. An a-c ammeter, like a d-c ammeter, is connected in series with the circuit in which current is measured. Effective values are indicated by this meter.

Electrodynamometer Voltmeter.

In the electrodynamometer voltmeter, the field coils are wound with many turns of small wire. Approximately 0.01 ampere of current flow through both coils is required before the meter will operate. The resistors are usually carbon, which, as you know, are of non-inductive material, are connected in series with the coils and provide for different voltage ranges. Voltmeters are connected in parallel across the unit in which voltage is to be measured. The values of the indicated voltages, in this case, are the effective values. The internal circuit of this meter is shown in figure 4-31.

Wattmeters.

You learned during our discussion of a-c circuits that power in an a-c circuit is not always found by multiplying voltage by amperage, as in a d-c circuit. A simple power computation can be made for an a-c circuit only when the voltage and current are in phase; that is,

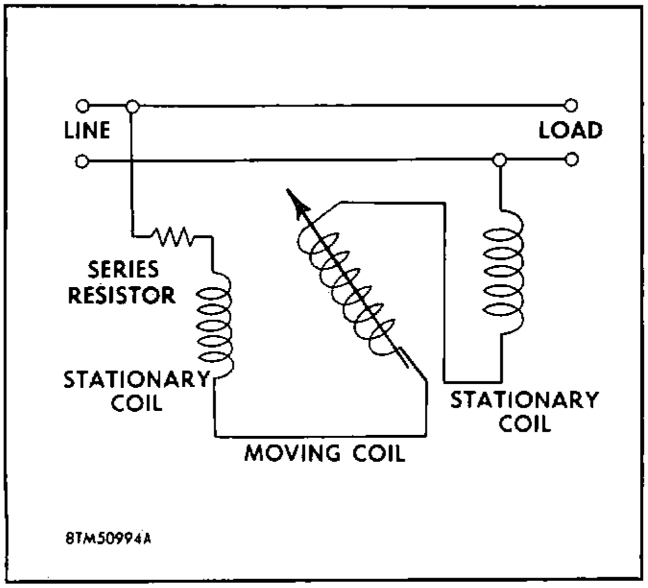


Figure 4-31. Electrodynamometer Voltmeter Circuit

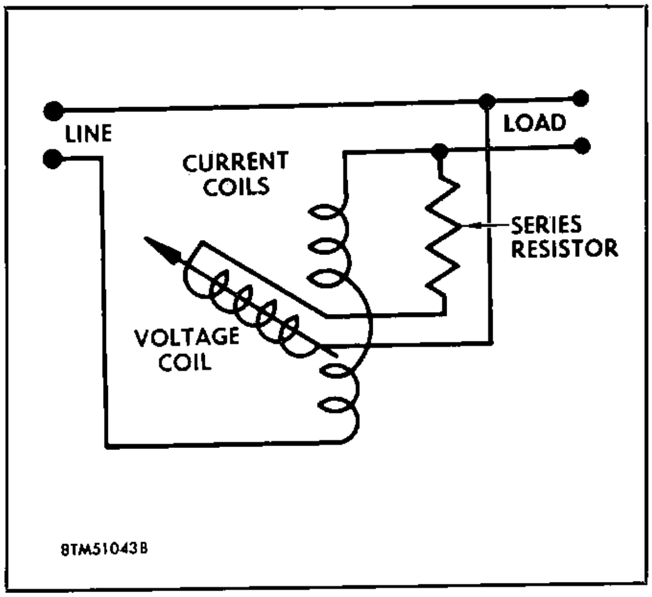


Figure 4-32. Electrodynamometer Wattmeter

when there is a purely resistive load. As you have found out, this condition seldom exists in practice. As in almost all a-c circuits, the load is reactive because of the presence of inductance and capacitance. The wattmeter, however, measures the true power consumed in a circuit by all electrical devices, regardless of the type of load.

The dynamometer wattmeter shown in figure 4-32 is a combination voltmeter and ammeter, and measures true power in watts. In the dynamometer wattmeter, there are two fixed coils and a movable coil. The fixed coils, the current-measuring element, are connected in series with the load. The movable coil, the voltage-measuring element, is connected in parallel with the load. Since the force acting upon the movable coil is the result of the magnetic fields of both the fixed and movable coils, the deflection of the pointer attached to the movable coil is proportional to the value of the voltage times the current.

Wattmeters may be used to measure power consumed in either single-phase or three-phase circuits in which the load is balanced.

The single-phase wattmeter has a high-resistance voltage coil of many turns of fine wire, and stationary coils, called current coils, of low resistance with a few turns of heavy wire. Connect the current coils in the line in series with the load and the voltage coil across the line. A single-phase wattmeter may be connected to measure the power in a three-phase circuit. To do this, you connect the current coil in one load line and the voltage coil between the line and ground. This will give the power in one phase. You simply multiply this by 3 to get the total power.

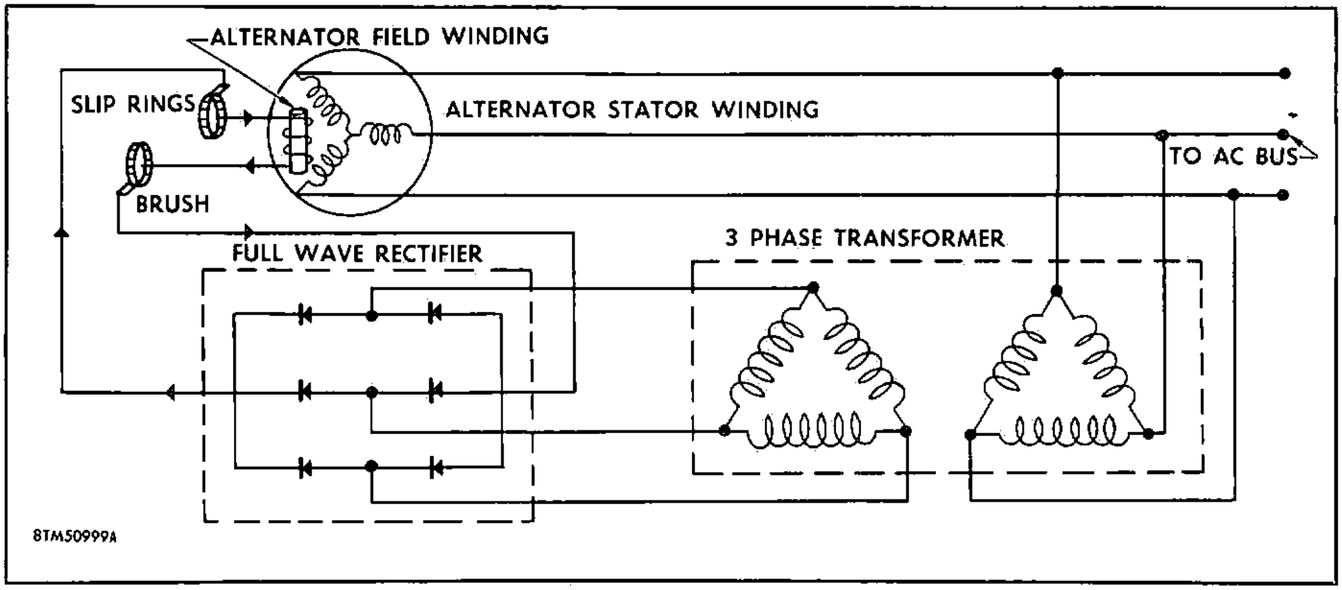


Figure 4-33. A-C Generator With Excitation Current Furnished by Rectifier

Three-phase wattmeters consist of two or more single-phase movements with all the moving elements mounted on one shaft. Separate single-phase wattmeters can be used to measure power in three-phase circuits. In this case, add the two wattmeter readings if the power factor of the load (motor) is greater than 50 percent. If the power factor is below 50 percent, the power input to the load (motor) is the difference between the two readings. You can determine whether to add or subtract the readings by the following—if both of the scale pointers deflect toward the top of the scale, add the readings; if one tends to indicate a negative value, reverse either the voltage or current connections and subtract the reading of one wattmeter from the reading of the other.

Precautions When Using A-C Measuring Instruments.

Never connect a voltmeter or ammeter in a circuit carrying a higher voltage or current than the rating of the meter.

Always check the rating of the meter before you use it.

Connect an ammeter in series with the circuit.

When using a wattmeter, always connect the voltage coil to the supply side of the current coil.

A-C GENERATOR POWER SYSTEMS.

You know that an a-c generator which generates a single a-c voltage is a single-phase alternator. You learned that such a machine has a single coil which rotates between two poles and more recently you learned that only a single voltage sine-wave is produced.

The practical a-c generator is a much more complicated machine as we explained in Chapter I. To provide a strong and controllable magnetic field, the permanent magnet is replaced by an electromagnet in which a direct current flows. This current is called the exciting current and is supplied by either an auxiliary d-c generator or by rectified d-c voltage. In the a-c system of the F-102A, this power is received from the d-c essential bus. As you know, when a rectifier is used, it changes a-c output into d-c current—this current being fed back to the generator field. The voltages developed by aircraft a-c generators are much higher than is necessary for this field excitation, so the input to the rectifier is reduced by means of a step-down transformer. In the F-102A, for control functions, d-c power can be selected either from the d-c essential bus or from transformer rectifier power. We shall cover this more completely further along in this chapter.

An a-c generator with excitation current furnished by a rectifier is shown in figure 4-33. It is the rotating element, or member, of an a-c generator which is called the rotor, the stationary element, or member, the stator. It is such a machine which is referred to as a polyphase alternator system. Let's find out why a-c generators are referred to as "two-phase" or "three-phase" or polyphase machines.

THE POLYPHASE GENERATOR SYSTEM.

The simplest of polyphase systems is the two-phase a-c generator. If a second coil is added to our two-pole single-coil generator, at right angles to the first coil —90° apart—and if another pair of slip rings is added, an alternating current will be induced in each coil as the two pass through the magnetic field in *succession.*

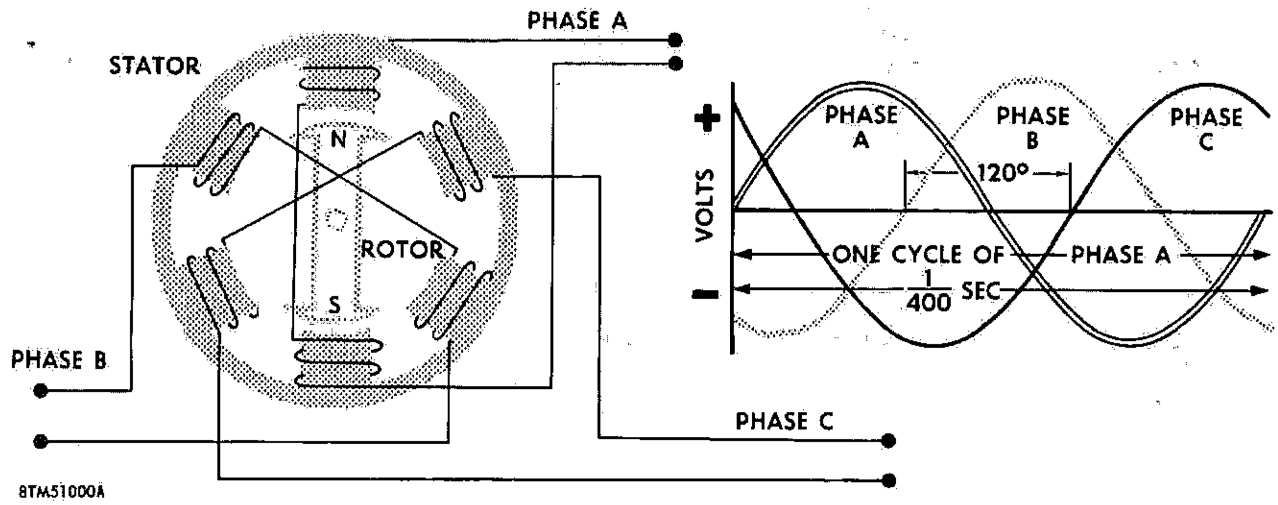


Figure 4-34. Three-Phase, 400-Cycle Alternator and Voltage Sine-Wave

Now, since they do not pass through the field together, the peak voltage in each will be induced at different times. This, of course, depends upon the position of the magnetic field at the time. In other words, while one coil is directly in the field, the peak voltage is being induced as the other coil is entering the field. This means that as the first coil is leaving, the second coil is approaching the field pole. The voltage output from each coil as you learned earlier is called a "phase," and that these two voltages always retain a very definite relationship. This relationship should be clear enough: one coil is displaced physically by 90°, and as the rotor turns, the induced voltage first rises to its peak in the first coil, or first phase; and when the two coils have turned 90° further, the induced voltage in the second coil, or phase two, has reached its peak.

And this brings us to the three-phase, 400-cycle a-c generator which takes 1/400 of a second to generate one complete cycle. This is the generator found in the F-102A interceptor. A generator such as this generally has a multi-pole rotating magnet and a multi-pole stator. The windings are so spaced and wound that the voltage generated is 120 electrical degrees apart regardless of the number of poles. In other words, as shown in figure 4-34, it has three sets of stator windings spaced equally 120 degrees apart. If the magnetic field as shown is rotated within the stator windings, three voltages are induced—one in each stator. It takes an a-c generator like this 1/400 of a second to generate one cycle. And the peak voltages in each phase being 120 degrees apart makes them 1/3 of 1/400 of a second or 1/1200 of a second apart.

THE WYE CONNECTED GENERATOR.

The three-phase a-c generator having three complete stator windings is actually equal to three single-phase alternators, the difference being that it maintains 120-electrical degrees separation between phases. In order that the two separate leads for each phase, as shown in figure 4-35, do not interfere with each other, two methods of connection have been devised—the wye connection and the delta connection. It is the wye connection, however, in which we are most interested at the moment, as this is the connection used in the F-102A a-c generator. Figure 4-35 shows the two methods of connections used in wye connected generators.

WHAT IS LINE-TO-LINE VOLTAGE.

The voltage between any two wires is called line-to-line voltage. Line-to-neutral voltage, as shown in the upper section of figure 4-35, simply means that the common connection, or neutral wire is the return wire for all three phases, and if the load is properly in balance—among all three phases—the current in this particular wire is zero. In many cases, this wire is omitted entirely, or, as in the case of the F-102A, it is connected to ground.

If 120 volts, in a wye-connected a-c generator, are generated in one phase, then the line-to-line voltage must be equal to 208 volts. How do we arrive at this? The line-to-line voltage = 1.73 times the line-to-neutral voltage; so by multiplying

$$1.73 \times 120 \text{ volts} = 207.6$$

or

208 volts, the line-to-line voltage

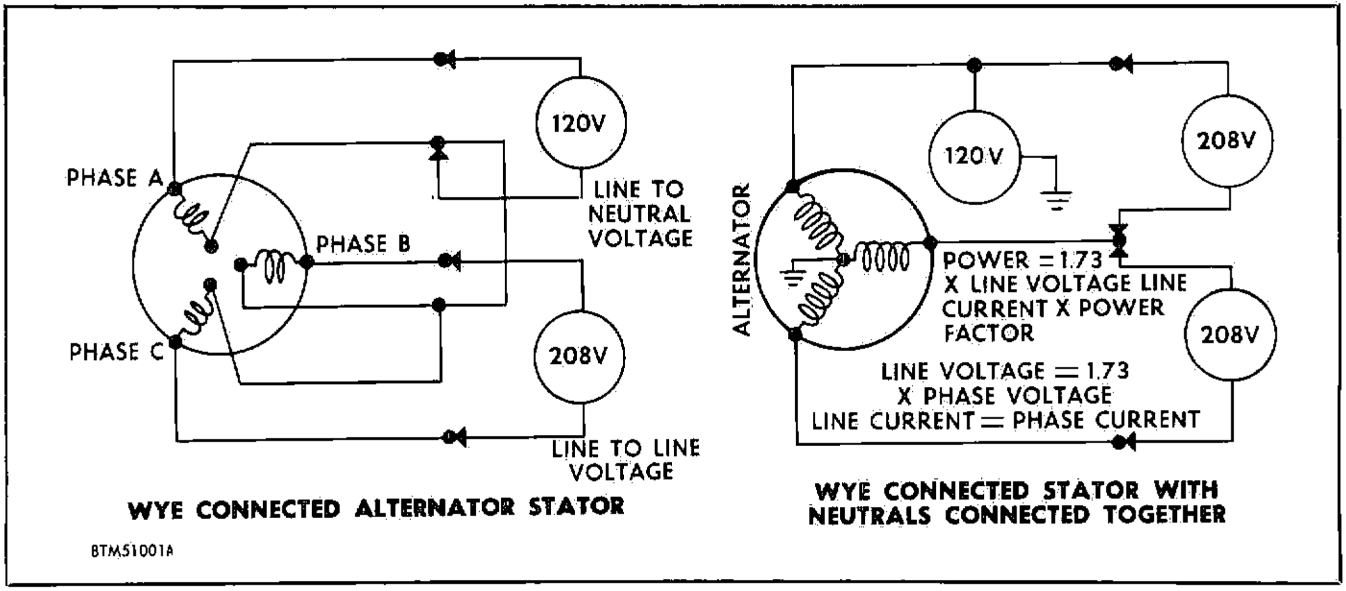


Figure 4-35. Wye Connected Stator With Neutrals Connected Together

TERMINAL CONNECTIONS.

The terminal block connections of an a-c generator are essentially connected as shown in figure 4-36. The neutral wires T_4, T_5, and T_6 are brought out and connected to an external ground. In the F-102A, T_4, T_5, and T_6 are connected to a common ground and T_1, T_2 and T_3 are connected to the "T" terminals of the voltage regulator and through the a-c power disconnect relay to the three-phase load and then to ground.

And speaking of load, one of the main advantages of a wye-connected a-c generator is that a higher voltage is obtained by connecting the load across two of the phase coils. For this reason, it is not necessary to generate a high voltage in each phase group because the current through each phase group will be equal to the current flowing through the load.

THE F-102A 30-KVA GENERATOR.

Figure 4-37 shows an exploded view of the a-c generator used in the F-102A. At the top of the illustration is the rotating member, the rotor; and across the bottom is the stationary member, the stator. The picture in the center is the generator assembled.

In direct-current generators, the alternating current which is induced in the armature coils is rectified by the action of the commutator. In alternating-current generators, as alternating current is already present in the armature coils, the output is taken from the fixed stator windings by means of a rotary collector which does not rectify. This collector, or collectors as the case may be, is the slip ring. The slip rings are mounted on, and insulated from, the rotating shaft. The brushes, which contact the slip rings, are mounted on the stationary member of the assembly. A-C generators have a great many armature coils and each coil has one lead connected to one slip ring and another lead connected to the other ring. D-C excitation current is applied through the exciter stator and slip rings.

The armature coils in a-c generators develop high voltages while the armature rotating field carries relatively low ones. In this way, by using the stationary armature type construction, direct connections can be made to the high voltage coils and sliding contacts can be safely used to transfer low exciting voltage to the field armature rotating field. The low exciting voltage required for the field permits the use of less bulky field coil insulation and thereby greatly reduces weight.

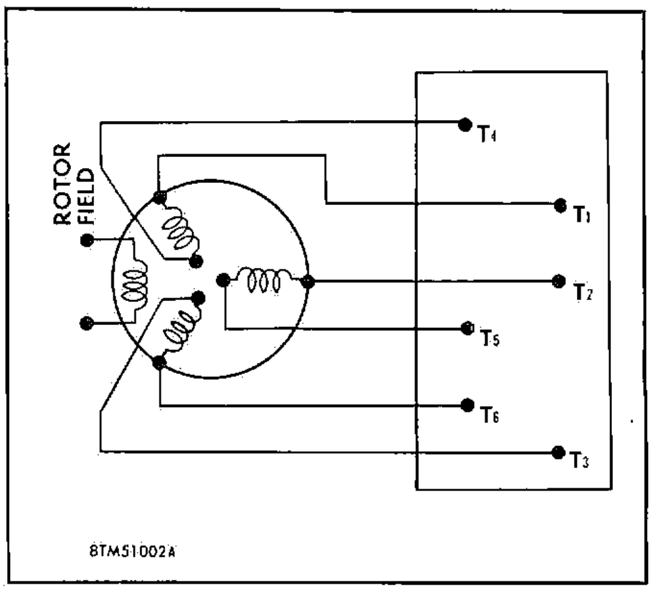


Figure 4-36. Terminal Block Connections of Alternator

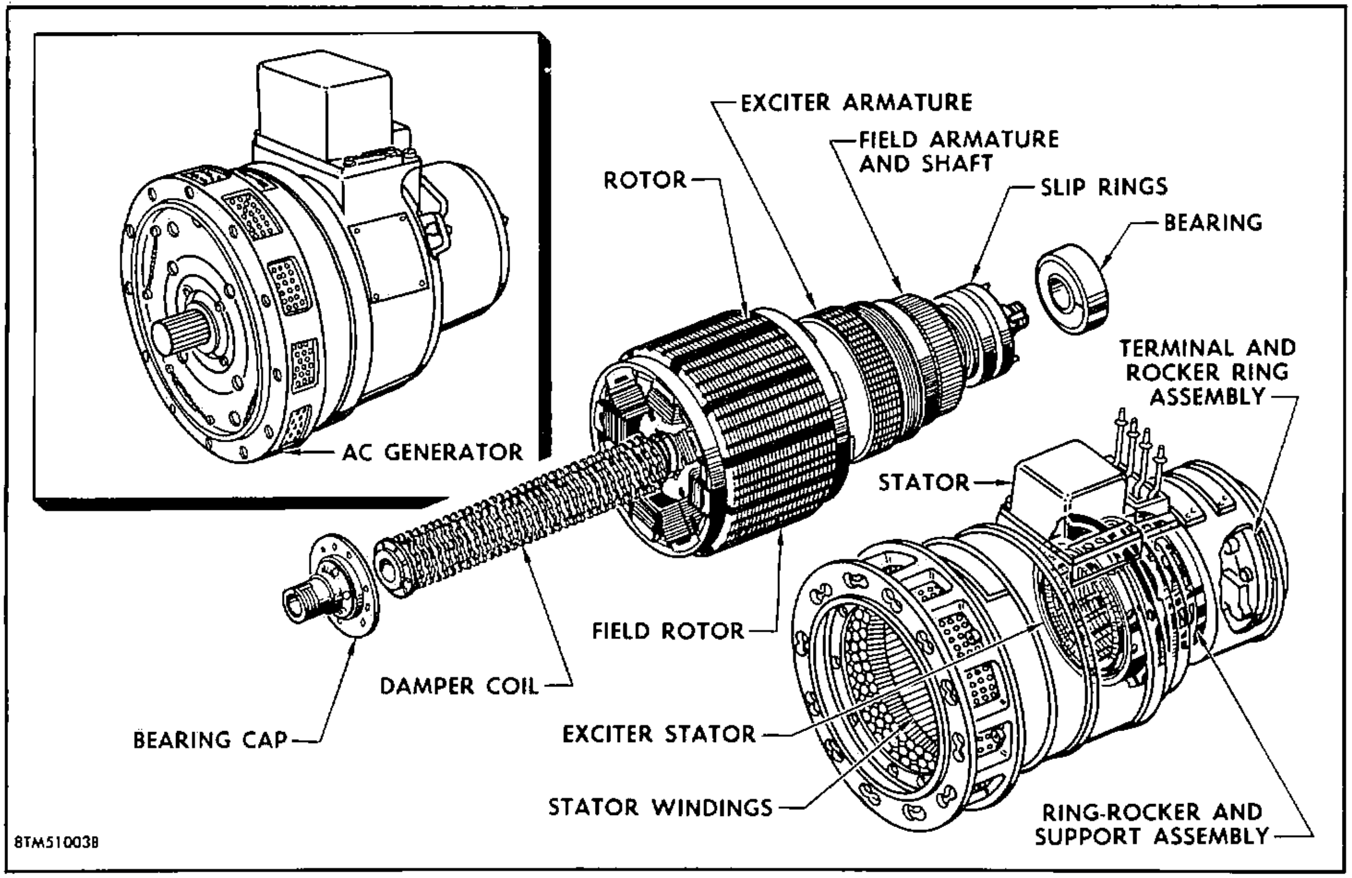


Figure 4-37. The F-102A A-C Generator

The Generator Damper Assembly.

The damper assembly seen in the illustration bears some explanation. Damper windings as they are called stabilize the speed of an a-c generator. For example, very simply, if the speed of the generator tends to increase, a counter emf or voltage action takes place in these windings and places a load on the rotor which tends to slow the machine down. If the speed tends to decrease, another action takes place in the damper windings, and the speed of the rotor increases. This is called induction-motor action.

INDUCTION-MOTOR ACTION. The basic principles of induction-motor action are quite simple. A motor is a machine that converts electrical energy into mechanical energy. And when a magnetic field exists about any current-carrying conductor, the strength of this field depends upon the amount of current. So when a current-carrying conductor is placed in a magnetic field, a force is exerted which tends to move the conductor out of the field. If the field strength of a motor is reduced, the value of the counter emf, which depends on the strength of the flux field strength, is reduced. This drop in counter emf causes greater current than is required by the load, and motor speed increases. But an increase in field current, increases the counter emf, and the armature current decreases. The result: the amount of current is not sufficient to maintain the load —the motor speed decreases.

This particular a-c generator is typical of many alternators used on modern high performance aircraft. It supplies 30-kva, three-phase a-c power which is automatically maintained at ± 4 cycles per second. It is air-cooled, wye-connected, and supplies two voltages. For this reason, it can be referred to as an 120/208-volt alternating-power system. In other words it supplies 120 volts in each phase, A, and B, and C. And if you remember if 120 volts are generated in one phase of a system such as this, then the line-to-line voltage must be equal to 1.73 times 120 volts or 208 volts. At 6000 rpm it has a minimum lagging power factor of 75% or 0.75. The power factor of an a-c generator was discussed earlier in this chapter. Also, this is a self-excited a-c generator using 9.5 volt DC with a current of 46 amperes. It weighs 95 pounds.

A great deal has been said in this manual about weight —especially in comparing a-c and d-c equipment. It will take only a casual glance at the a-c generator in the F-102A to see that it is considerably larger than the d-c generator. This is true; the d-c generator weighs only 51 pounds against the 95 pounds weight of the a-c generator. Remember, however, that the d-c generator has a rated output of only 6 kilowatts while the

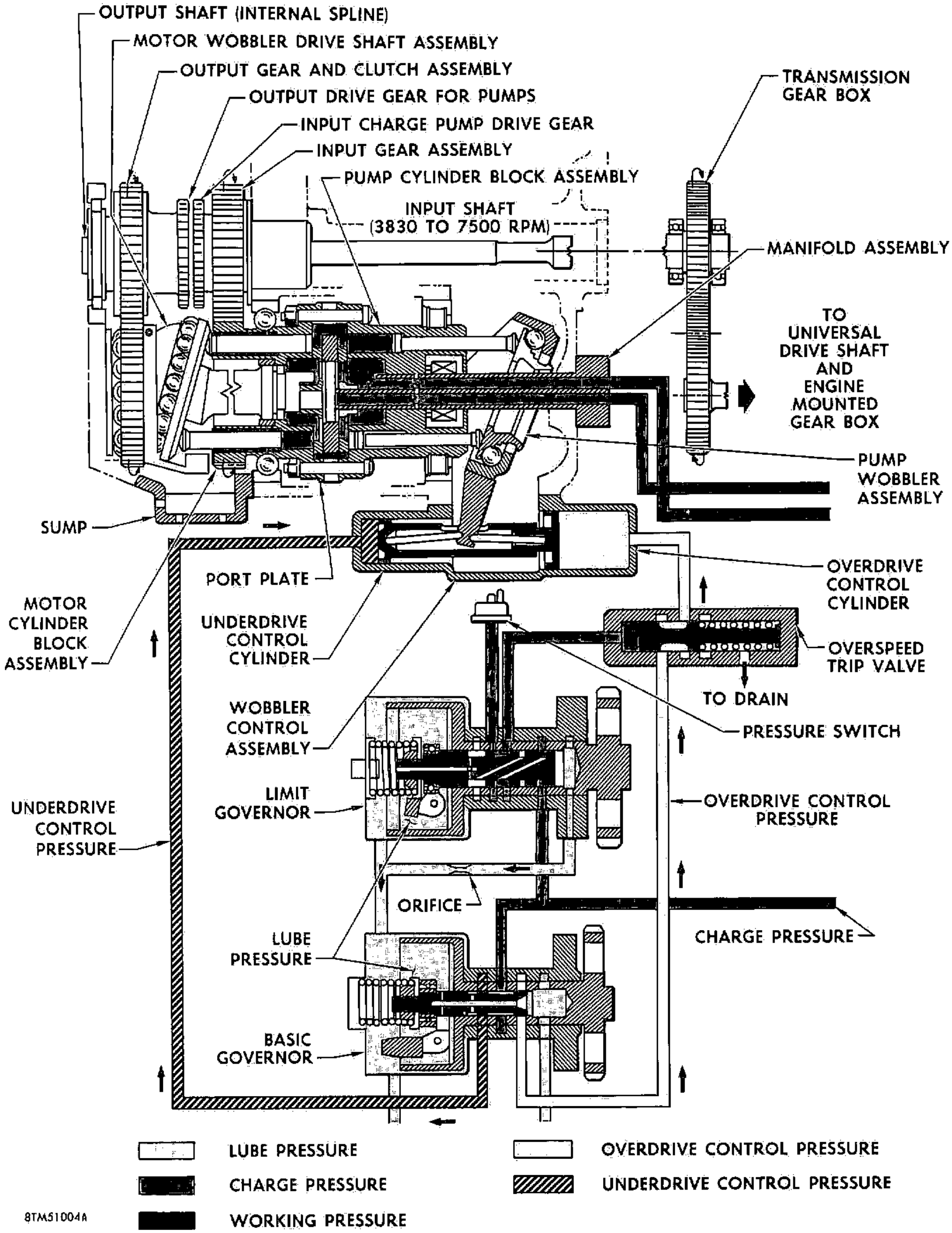


Figure 4-38. Constant-Speed Drive Unit

a-c generator has an output of 30 kilowatts, or five times the power of the d-c machine. A kilowatt is one thousand watts.

THE F-102A A-C GENERATOR CONSTANT-SPEED DRIVE.

In the F-102A a-c generator power system, the a-c generator is mounted on the rear mounting pad of the constant-speed drive unit. The generator is driven from the transmission gear assembly of this unit in a counter-clockwise direction, when you are facing the pad, and turns at a constant speed of 6000 $\pm$60 rpm.

A portion of the constant-speed drive unit is shown schematically in figure 4-38. This is not shown in Chapter III, but photographs of the unit and its installation are shown. It is suggested that you turn to these pages now and review their contents. It is not felt that a complete explanation of the constant-speed drive unit's operation is within the scope of this supplement, but you should at least be familiar with its internal control units and the pressures exerted on those units. Its basic principles as they relate to both the d-c and a-c generators are discussed in Chapter III. We have left them for this section as they are more applicable to the a-c system as a whole than to the d-c system. The d-c generator, after all, varies with engine speed; the a-c generator is driven at a constant speed plus or minus 10%—and the unit driving it is called the constant-speed drive assembly.

CONSTANT-SPEED DRIVE OVERSPEED AND UNDERSPEED PROTECTION.

As you can see in the hydraulic circuit of this drive, figure 4-38, there is an overspeed trip valve which automatically prevents the a-c generator drive pad speed from exceeding 7500 rpm in the event of a malfunction of the normal governing unit. This governing unit has a flyweight governor, or the basic governor, which maintains the correct position of the pump wobbler assembly, and the limit governor assembly, which controls the overspeed trip valve.

The overspeed unit trips when the output speed of the transmission gear assembly exceeds 7000 rpm. This tripping is motivated when the flyweights of the overspeed limit governor move apart and exert a lifting force which overcomes the tension of the biasing coil spring. When the gear box ceases to function, this overspeed unit resets automatically for normal operation.

Electrically speaking, an underspeed switch closes its normally open contacts when the a-c generator pad output speed reaches 5550 rpm or 370 cps, and opens these same contacts before the output drops to the minimum of 5250 or 350 cps. You are probably asking yourself at this point: Okay, but how is an electrical underspeed switch activated in a set-up like this? A good question. Let's take a look at our schematic, figure 4-38.

During normal operation, the pressure switch, in the schematic, is in a closed position. This is due to the charging pressure exerted against it. This pressure is regulated by the stem in the limit governor. In case of a malfunction, the pressure bearing on the base of the stem quickly builds up with enough force to compress the coil spring, moving the stem. This changes the position of the stem and in so doing closes off the charge pressure from the pressure switch and the base of the trip valve. At the same time, pressure is drained from the pressure switch through tiny ports to the flyweight housing. It is in this way that the pressure switch is opened.

The electrical underspeed switch is connected to pin X2 of the a-c power disconnect relay of the auxiliary power relay and to pin A2 through A5 of the same relay to ground. So when pressure is drained from the pressure switch, the overspeed switch in the electrical system is actuated, disconnecting the generator from the a-c essential bus through the action of the a-c power disconnect relay.

The output gear and its clutch assembly in the constant speed drive prevents the gearing to the a-c generator pad from being driven by the deceleration of the a-c generator when an automatic disconnect occurs. In the event of a serious a-c power overload, a shear section in the transmission will shear and disengage the hydraulic transmission coupling to the a-c generator. This action, however, in no way interferes with the normal function of the d-c generator. Further information regarding the constant-speed drive and complete schematic of the system will be found in the supplement entitled Power Plant Installation.

THE F-102A A-C ELECTRICAL POWER SYSTEM.

The schematic diagram in figure 4-39 shows the F-102A a-c electrical power system which we will analyze in the following pages. As you know, there are two alternating-current generators in the a-c electrical power system of the F-102A. There is one a-c control panel for protection against overvoltage, and the general control of a-c generator operation such as phase regulation, and the like. We will discuss the various duties of this control panel when we cover the 30-kva control system. Voltage regulation of the system is maintained through a magnetic amplifier. During normal operations, the a-c system is supplied by a 30-kva generator rated at 120/208 volts and regulated through the magnetic amplifier to 115/200 volts. It is a three-phase, 400-cycle generator. It is an air-cooled "wye" connected unit which generates 120 volts in each of its phases with a 208-volt line-to-line (phase-to-phase)

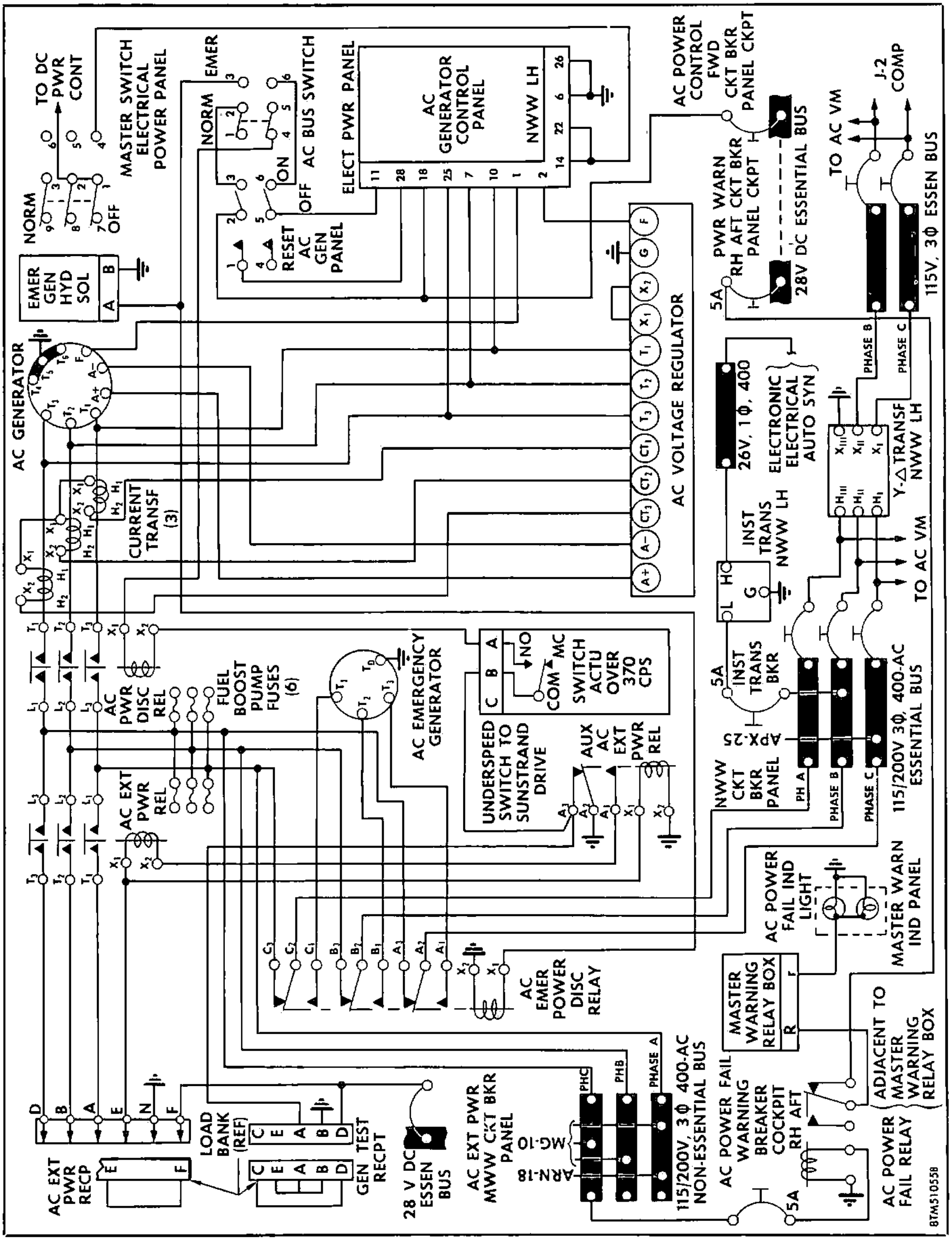


Figure 4-39. A-C Electrical System Schematic

voltage. Mounted on the rear pad of the constant-speed drive assembly, this a-c generator supplies, through the drive's transmission gearing, a regulated power through an a-c disconnect relay. The cooling system of the 30-kva a-c generator is shown (in figure 4-37) and discussed on page 174.

For an emergency, the system is provided with a 1-kva, 120/208-volt generator that is internally regulated to 115/200 volts. This is a three-phase, 400-cycle, air-cooled generator. The a-c generator is driven by a hydraulic motor which is connected into the secondary hydraulic system through an electrical-sensed hydraulic solenoid. The generator furnishes limited emergency power to the airplane through an a-c emergency power disconnect relay system. Both the 30-KVA and the 1-KVA generators are controlled by an a-c generator switch and an a-c bus switch.

As you have learned, one of the reasons that alternating current adapts itself so easily to aircraft is the pliant way it can be stepped down or up to meet the demands of the various circuits by means of transformers. In the F-102A, a step-down transformer reduces the a-c voltage from 115 volts to 26 volts during both normal and emergency operations. This 26-volt, single-phase, a-c power is used for certain instruments; for this reason, this particular transformer is commonly referred to as an instrument transformer. The simplified diagram in figure 4-40 shows the extent of the a-c power system in the F-102A, its three phases, and its general bus systems. Figure 4-41 shows the a-c system components in the F-102A.

The three-phase output of the a-c generator during normal operation is stepped down to 115 volts by using phase B and C as single-phase power, through a wye-delta transformer. This power is used during both normal and emergency operations. A selector switch on the electrical power control panel in the cockpit controls the a-c voltmeter which provides voltage readings of each succeeding phase as ØA, ØB, ØC, as well as the 115 volts, ØAB and ØAC, and the 26 volts, to ground.

In the a-c electrical system, circuit breakers are provided for the protection of all circuits except the attitude gyro and those instruments supplied from the 26-volt a-c essential bus. In place of individual circuit breakers, phase B is connected to the primary of the instrument transformer through the instrument transformer circuit breaker with a 5-amp rating.

For external a-c power, an a-c external power receptacle is located on the left-hand side of the main wheel well below the d-c external power receptacle.

The operation of either the 30-kva or 1-kva a-c generator is controlled by the a-c generator switch and the a-c bus switch. An a-c power failure is indicated by a master warning system. As in the d-c system, a power failure is indicated through the master warning system which illuminates the master warning indicator light on the instrument panel itself, and the a-c power failure indicator on the master warning indicator panel, just above the d-c indication of failure.

Before analyzing the a-c system, as energized for normal in-flight operations, let's review briefly the components which will function within the system under this particular flight condition. First of all, the 30-kva control system must operate or we would have no system. This embraces the a-c generator itself, the voltage regulator, and control panel. It is assumed, of course, that the constant-speed drive is operating smoothly, and that the d-c side of the system is also functioning normally.

In the power distribution system to the buses, we have the a-c power disconnect relay, which, when energized, connects the a-c power to the buses. Then, there is the a-c emergency power disconnect relay, through whose contacts normal a-c power is fed to phases A, B, and C of the essential bus, and which disconnects normal power and feeds emergency power to this bus if a malfunction occurs—when the 30-kva generator is taken off the line. The a-c power failure relay, the underspeed switch, and various circuit breakers, stand as guards over the system as a whole—just as the fuel boost pump fuses guard the fuel boost pump system. The step-down transformers in the system function as the needs of the various subsystems require them to function. These components then are the devices which channel power for flight conditions—these are the primary devices on which the airplane depends for carrying out its mission, for making it do what it was designed to do.

It is a rather simple system when you analyze it, when you know how a component works, and why it works; actually with proper maintenance, with proper care and treatment, it is no more complicated—or temperamental—than the electrical system in your automobile, or the system which brings current for that television set, dishwasher, or the light you shave by. Sure, for air-borne electricity there are "G" conditions, stress and strain, and high altitudes to put up with, but leave these complications to engineering "know-how" and simply and efficiently do the job required of you. Electricity at times can certainly be temperamental—and this is expected—but as we have mentioned before, it is highly conveyable, controllable, and convertible. In the F-102A it is conveyed, controlled and converted, in a tailor-made fashion. And as we said at the beginning of Chapter II, "The best power and distribution system for any airplane is one tailored to meet the demands of the particular airplane." This is the type system found in the F-102A.

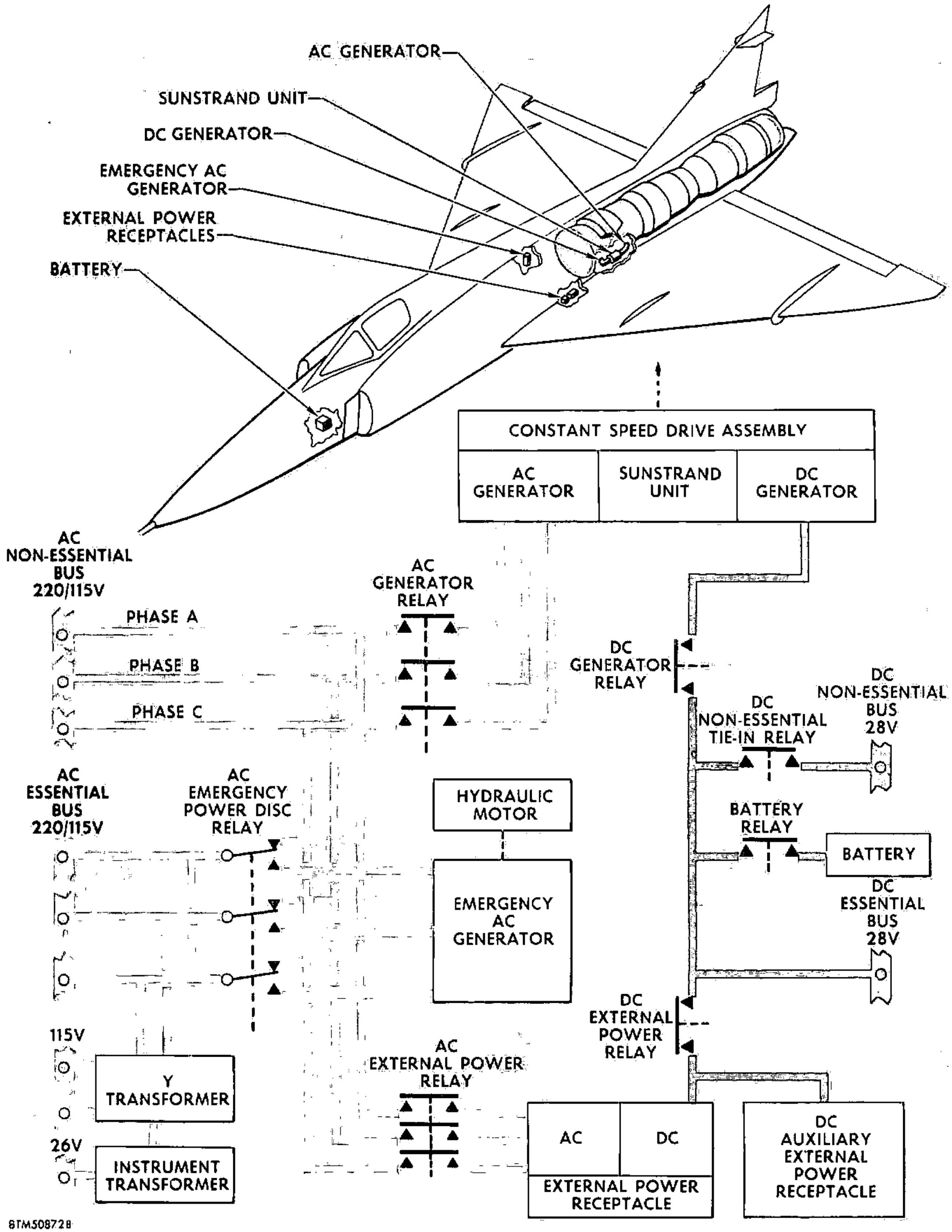


Figure 4-40. The F-102A A-C Electrical Power System Diagram

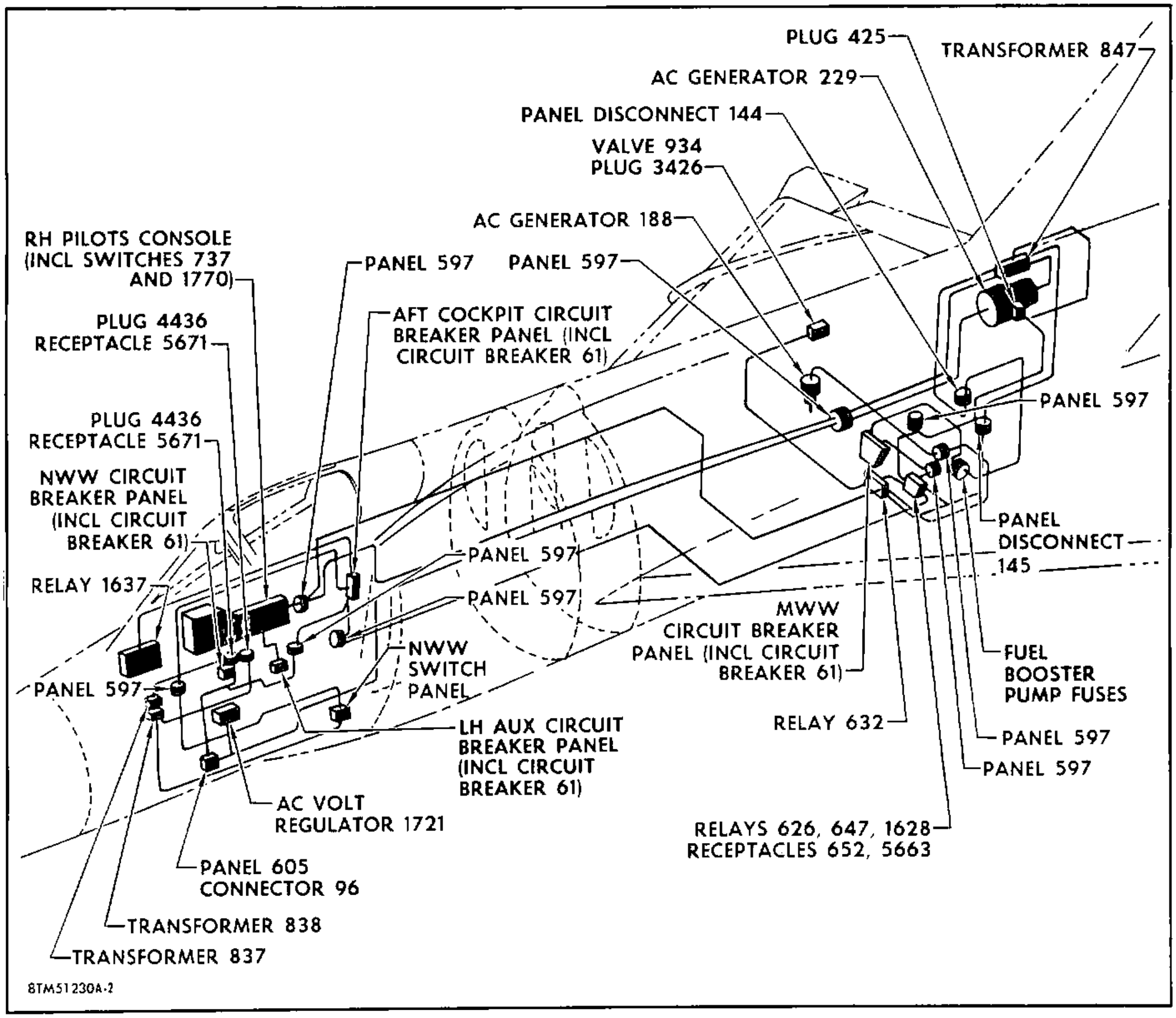


Figure 4-41. F-102A A-C Electrical System Components

THE F-102A 30-KVA A-C CONTROL SYSTEM.

In the d-c system, if you will recall, we had a single control panel consisting of a carbon-pile voltage regulator and numerous relays enclosed in the covered portion of the panel. You were told that except for flight-line voltage regulation, at the rheostat, and the utilization of the test jacks for checking the voltage, you would have little to do with the control panel except to disconnect it from the system, in case it becomes inoperative, or to connect its replacement into the system.

In the a-c system, not only do we have a control panel, but a magnetic amplifier. The magnetic amplifier does for the a-c system what the carbon pile does for the d-c system — it regulates the voltages. The a-c control panel, in like manner, is in the system to provide control and protective equipment for the 120/208-volt a-c system. Here again, except for checking the magnetic amplifier or the control panel, as a flight-line mechanic, you will not be called on to overhaul or service these devices. When you have checked through the system and found one or the other to be functioning improperly it will be a case of remove and exchange for one that you know has been serviced and tested and is known to be in proper working order.

It is important, however, that you understand their function in the system to understand and have complete knowledge of the a-c system, just as it was important to understand the function of the d-c control panel in the d-c power system. For that reason, we will take up their function in the system in the same order that we discussed the carbon pile and the enclosed relays

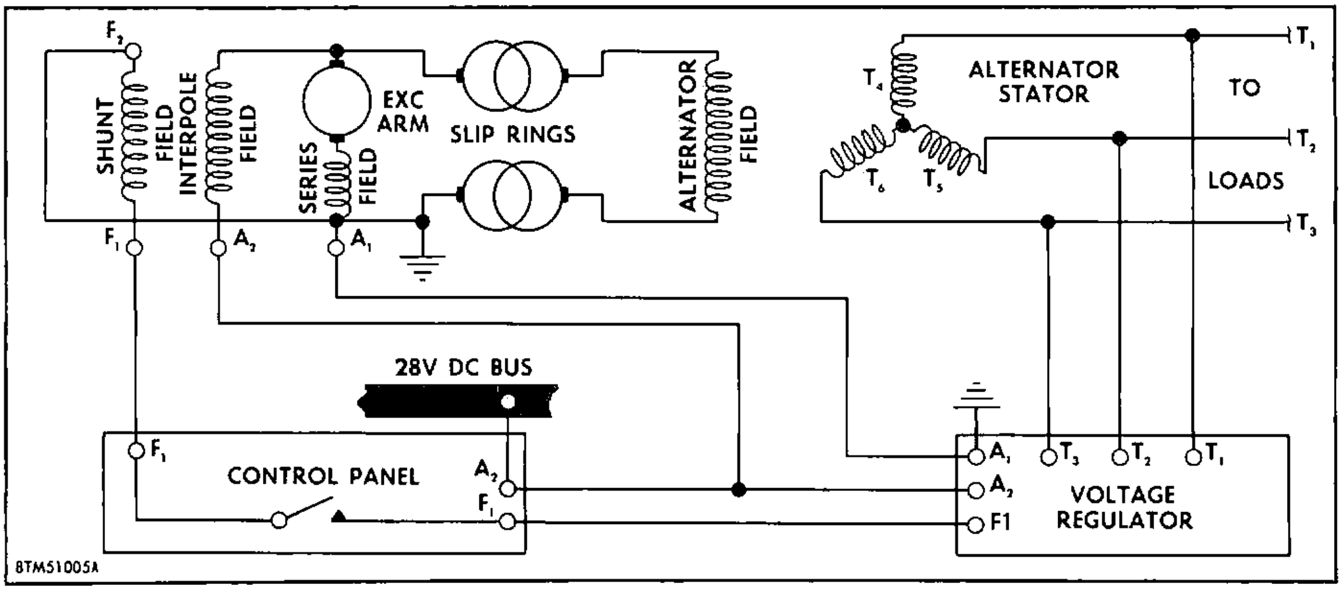


Figure 4-42. The 30-KVA A-C Control System

in the d-c control panel—the voltage regulator and then the control panel. The a-c generator and its companion units are shown in figure 4-42.

PROTECTION.

High performance airplanes demand the ultimate in electrical supply and control systems. The a-c generator, the voltage regulator, and the control panel found in the F-102A may be described as a tailor-made system for the high performance demands of this interceptor. But these precision-made control and protective devices would be worth very little to the overall system if it were not for the other relays and circuit breakers also found in the system. These relays and circuit breakers were thoroughly discussed in Chapter II. It is in the a-c system that the 12 fuses used in the F-102A are found. These are utilized for protection in the fuel boost pump circuits which are discussed in another supplement of this series, Airframe Fuel System.

Alternating-current power systems in aircraft were originally thought of as the juvenile delinquent of electrical power systems. The reason it was occasionally referred to in this manner is clearly shown by the seemingly endless array of protecting and controlling devices found in an a-c system, from the very source of power and on up to the essential and non-essential buses.

Some examples of what might be called the "silent sentinels" of an a-c electrical system have already been covered. You have learned how the constant-speed drive prevents the a-c generator drive speed from exceeding 7500 rpm. How the overspeed unit trips when the output speed of the transmission gear exceeds 7000 rpm; and how the pressure switch controls the electrical underspeed switch which in turn disconnects the a-c generator from the a-c essential bus. And how in the event of a serious power overload, the shear section in the transmission shears and disengages the a-c generator from the system completely. And then there are the damper windings in the rotor shaft, which help to regulate or stabilize the speed of the a-c generator. It takes all these devices, including the regulation and control of the voltage and current developed by the a-c generator, and the other relays and circuit breakers throughout the system, to insure the safety and protection needed by the a-c generator, and all the other components of an a-c electrical system.

SIMPLICITY AND SAFETY.

These protecting and controlling devices have been the result of the difficulties encountered due to the comparatively recent development of a-c load demands. Let's face it. From an engineering standpoint, the complexities of a-c power systems in aircraft are many. It should be unnecessary to tell you, a flight-line mechanic, that conditions are far different in supplying electrical power for television sets and washing machines than for aircraft. The lives of the people in a community do not depend completely upon an electrical power system as is the case when dealing with aircraft. This places a very special premium on the safe and the effective operation of aircraft electrical control and protective devices. Thanks to engineering "know-how" and the rigid specifications laid down by your Air Force, the reliability and safety factors governing an alternating current system have, with proper and efficient maintenance, become fully as good as those used in d-c systems—in some cases much better.

Simplicity and safety are desirable not only from an operating and maintenance standpoint. Simplicity enables you as a crew member to understand the overall operation of the a-c system—or any system—in a shorter period of time, and reduces that very important factor —the possibility of "human error" while operating or trouble shooting the system. The reduction of maintenance time increases the available flying time. For this reason, it is important that the number and the types of protective devices be kept to a minimum, consistent with the desired reliability of the system. So, when you hear people talking about the safety and the reliability of an airplane's electrical system—they are talking indirectly about you. Safety and reliability mean simplicity of understanding and therefore more efficient maintenance.

There is nothing complex or mysterious about the a-c system in the F-102A. In looking at the overall schematic diagram, it might look rather complicated but if the voltage regulator and control panel were taken completely out of the system it would be quite as simple as the d-c system; that is, as long as you recognize that it is d-c electricity which actuates the various relays and excites the generator field and that it is a-c electricity which actually energizes the system.

Let's start by finding out why the voltage regulator is in the system and what it does in the system—and why it is called a magnetic amplifier. As a flight-line maintenance man, it is only necessary that you understand why it is in the a-c electrical system and what it does in the system. The following explanation is simply to bring out in the open what this particular "black Box" consists of and how it works. If a malfunction is traced to it, you will replace it.

THE MAGNETIC AMPLIFIER AND THE F-102A.

The magnetic amplifier is what is known as a static device—there are no moving parts. And since there are no moving parts, it is not subject to mechanical wear and should have long life expectancy.

In figure 4-43, which shows the magnetic amplifier used on early F-102A airplanes, the magnetic amplifier as a unit has a mesh screen type housing for cooling purposes. The terminal board has fourteen terminal connections, twelve of which are used. At the top of the board, and inset as shown in figure 4-48, is the a-c voltage adjustment. This adjustment is made by loosening the locking nut on the voltage regulating adjusting screw and turning the adjustment screw until the voltmeter reads 115 volts. This is a simple operation and is covered more specifically in the T.O. 1F-102A-2-10.

The magnetic amplifier goes back quite far in electrical history, when they were more commonly known as a type of iron-core reactor. These reactors were, and still are, used for theater dimmers, to reduce the voltages in spotlights, footlights and the like, in filters for smoothing rectified alternating current, and in selective filters for communications and power-relay circuits. But, until the last few years, their application to aircraft was limited because of unsatisfactory performance under certain environmental conditions. Recent improvement, however, in dry-disc rectifiers, which we have already discussed, and certain new magnetic materials have increased the efficiency and sensitivity of this particular type of reactor. In the F-102A, with its improvements and additions, the reactor, today more commonly known as a magnetic amplifier, is referred to as the voltage regulator. The block diagram in figure 4-44, of a regulator circuit is typical of the average regulator circuits. The current transformers act as excitation boosters during overload.

HOW THE VOLTAGE REGULATOR WORKS.

The voltage regulator, through its control circuits and its output to the exciter shunt field, maintains a constant generator voltage. It senses the a-c generator output voltage through the voltage sensing circuit and controls the proper amount of d-c current which is supplied to the exciter field in order to maintain the correct output voltage at the generator terminals. In the F-102A, the regulator maintains the a-c generator output voltage within $\pm 2.5\%$ of the rated line voltage.

When the voltage is adjusted through the voltage control rheostat, it directly controls the current supplied to the shunt field of the generator's exciter. The control of the exciter current controls not only the a-c input but also its output voltage.

Two-Stage Amplification.

The diagram in figure 4-45 shows the internal circuitry of the voltage regulator in the F-102A. As you can see, the regulator is divided into two stages; therefore, it can also be called a two-stage magnetic amplifier.

The sensing circuit, which you see in the first stage, supplies a signal proportional (in quantity) to the average of the three line-to-line a-c voltages, to the direct-current control winding. This signal, in other words, is taken from all three a-c line-to-line voltages, rectified, and compared magnetically to the first-stage amplifier, with an added constant signal from the voltage reference. The value which results from these two signals acts on the first-stage magnetic amplifier and controls the level of the first-stage output.

It is the output from the first stage which is fed into the second-stage d-c control winding. This output is a d-c control current which is biased to a proper operating level through a full-wave rectifier designated in figure 4-45 as the bias supply. Actually this is a bias signal, which acts similarly to the constant reference signal used in the first stage. The difference is that it is variable at this stage because it is proportional to the a-c line voltage. It is the magnetic result of the full-wave rectifier and the first stage output signal which controls the output of the second-stage magnetic amplifier.

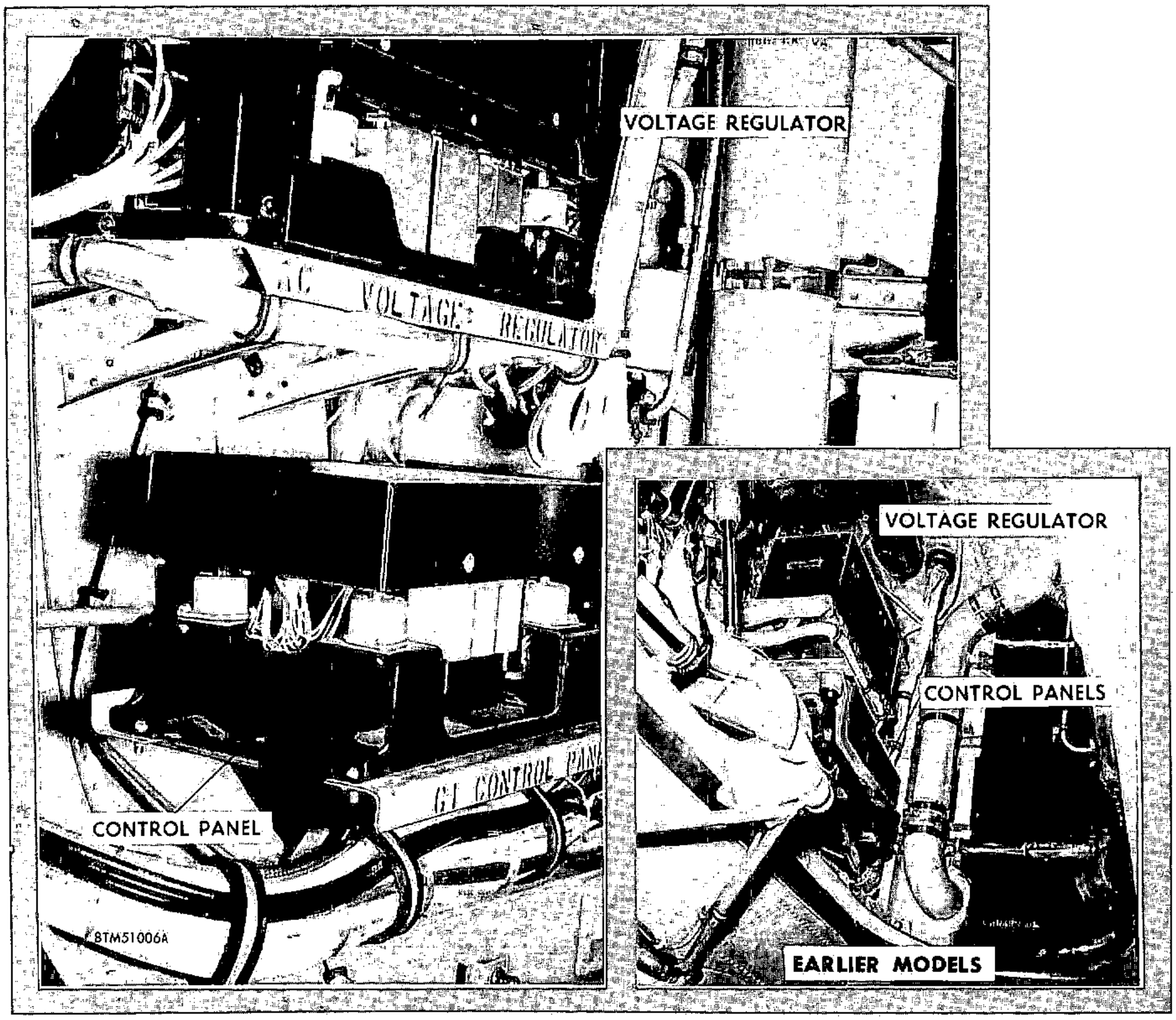


Figure 4-43. The A-C Voltage Regulator

The second stage of this regulator is the power stage and, together with rectified d-c power from the control panel, supplies d-c current to the shunt field of the exciter. It has been mentioned before that it is the amount of this sensed excitation which controls the output level of the a-c generator.

THE CLOSED LOOP SYSTEM. To maintain the stability of this unit and the system as a whole, the feedback network, as shown in figure 4-45, takes any rate of change signal from the exciter output voltage, and feeds it into the rheostat control winding of the first stage. This signal is in such a direction as to oppose any change which might occur due to a transient voltage in the a-c line voltage. Because of this feedback network, the magamp is sometimes referred to as a negative feedback closed loop system.

THE STARTING RELAY. The relay shown in the upper right-hand corner of figure 4-45 is what is known as a starting or start relay. As you know, to obtain output power for the exciter field from the voltage regulator, a-c generator voltage is required as input power to the voltage regulator. A starting relay is used to permit generator buildup without voltage regulation. For this reason, the contacts of the starting relay are shown normally closed—normally closed simply means that this is connected from the A+ terminal to F1, through the field flashing network and generator control relay of the control panel. Connected in this way, it allows the generator exciter to use self-excitation to build up the a-c generator voltage so that power is available for use by the static regulator. When the a-c generator builds itself to a predetermined voltage level, the contacts of the start relay open. This disconnects

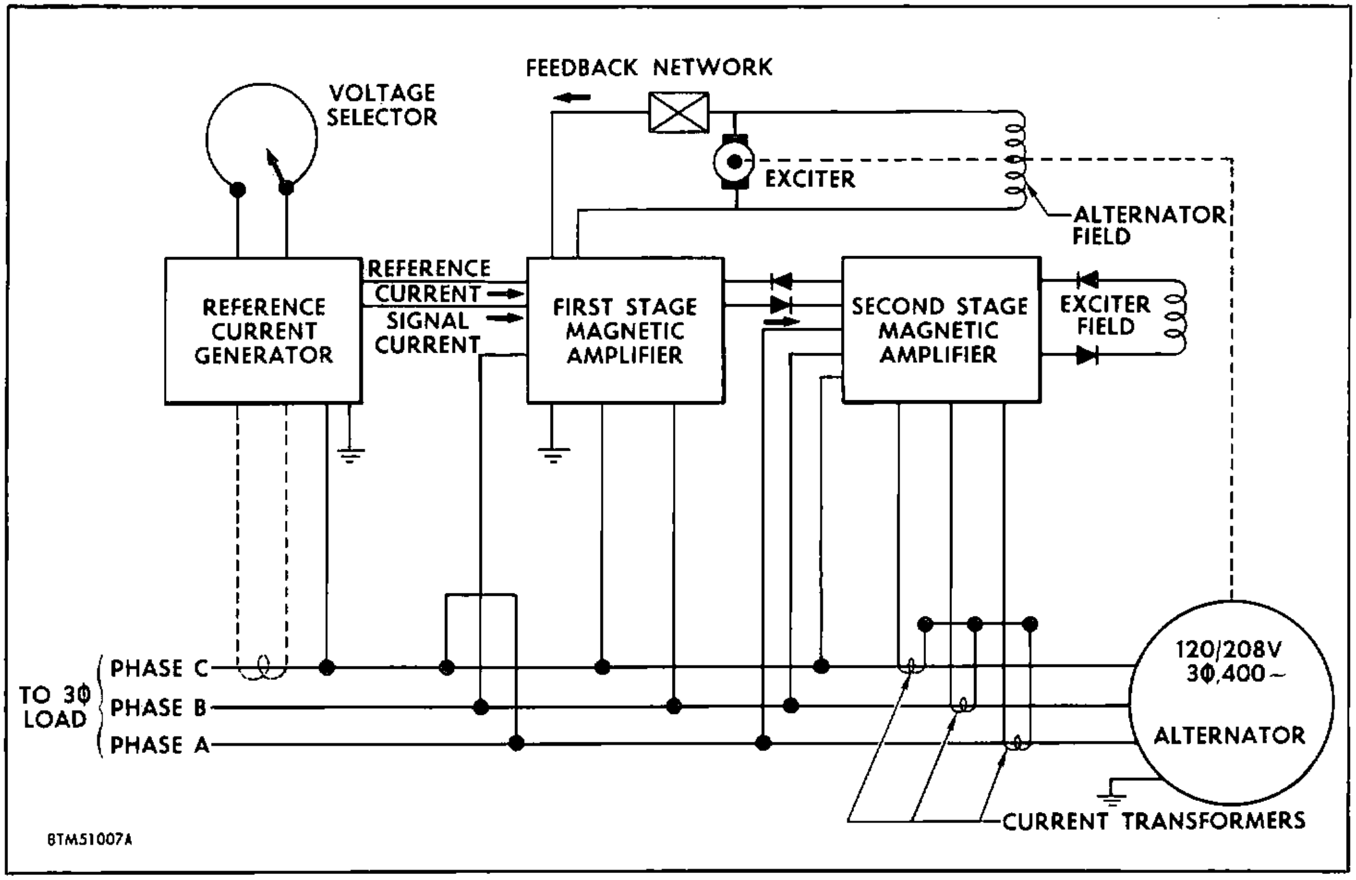


Figure 4-44. Block Diagram of A-C Voltage Regulator

the exciter shunt field and the field flashing network in the control panel. The exciter field current is then furnished solely from the output of the second stage or power stage of the regulator during normal operation.

Current Transformers in the Voltage Control System.

The current transformers, one located on each phase, are used to boost the regulator output during overload and short circuit conditions. See figure 4-46. Since rated voltage output of the generator is required under all conditions, including that of overload, the boost current transformers increase the regulator maximum output limit by increasing the voltage supply to the second-stage magnetic amplifier output circuit. When the system is operating at rated voltage, the regulator, as you have been told, controls the excitation of the exciter field as required. This is to maintain the system's voltage at a constant level. The current transformers are used only to extend the maximum output limit of the regulator.

During three-phase short-circuit conditions, the voltage drops to a low value which is beyond the control of the voltage regulator. When this occurs, there is *no control of the a-c system except by the current transformers,* which have been designed to supply the rated short-circuit current. Three-phase short circuits are primarily controlled by the design of the current transformer and the generator. The voltage developed across the transformers during three-phase short circuits maintains voltage on the start relay coils, so that the contacts remain open when the line voltage drops to a low value.

The current transformers in the F-102A system are located on brackets adjacent to the a-c generator a little behind the constant-speed drive unit. As you can see in figure 4-47 these transformers are of the simple ring-type construction, and look somewhat like three doughnuts in search of a cup of coffee. The primary coil, as in all transformers, receives the electrical energy whose current is to be transformed. The secondary coil is the sensing coil and is wound into a ring around the iron core. During the operation of the system, current flowing through the particular phase lead sets up a magnetic field in the core. It is this field which sets up a current in the secondary winding which is used as the sensing current in the operation of our various protective relays.

The three output a-c generator phase leads, T_1, T_2, T_3, are fed through the rings at the time of installation—phase A through one, phase B through another, and phase C through the other. Now, because there are no

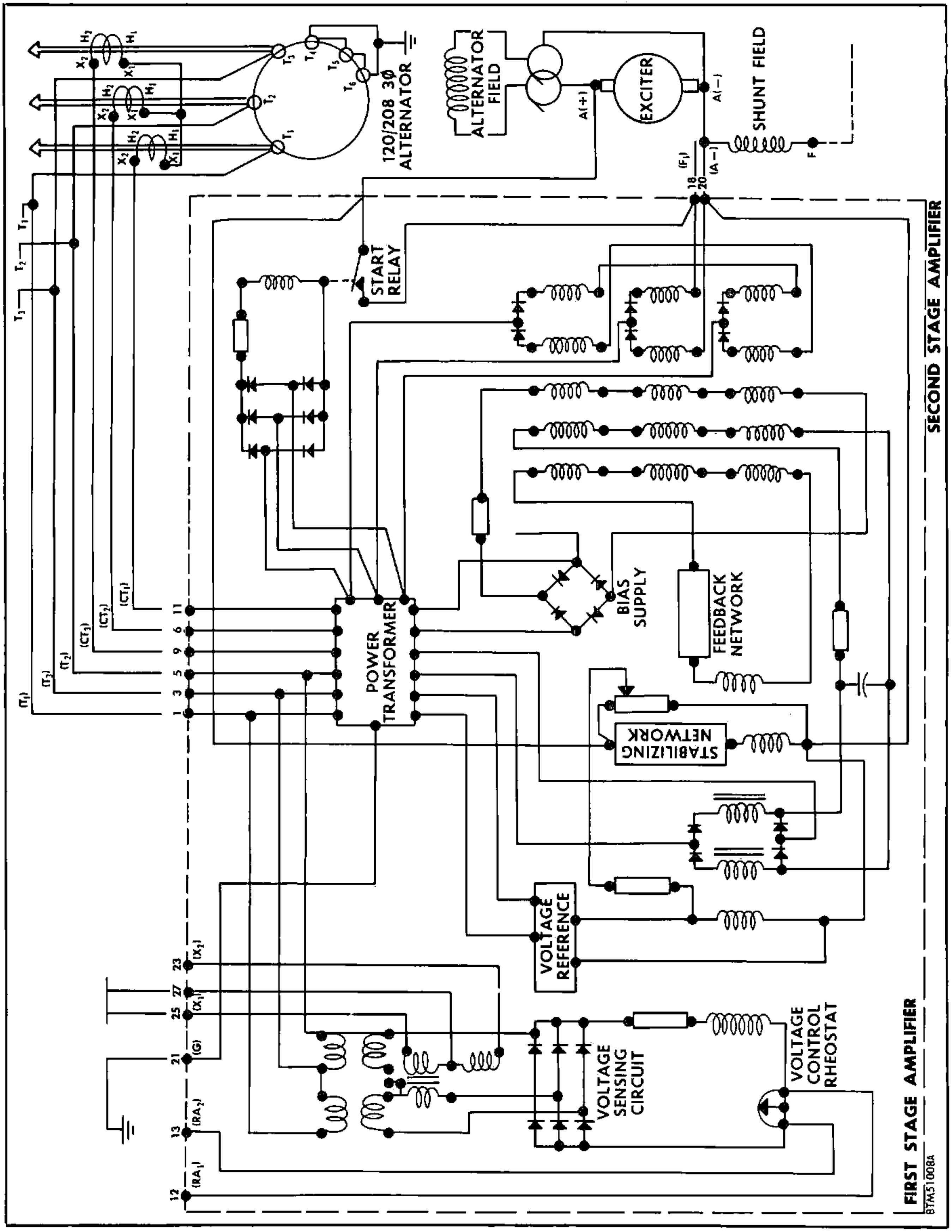


Figure 4-45. Schematic of Internal Wiring of Two-Stage Magnetic Amplifier

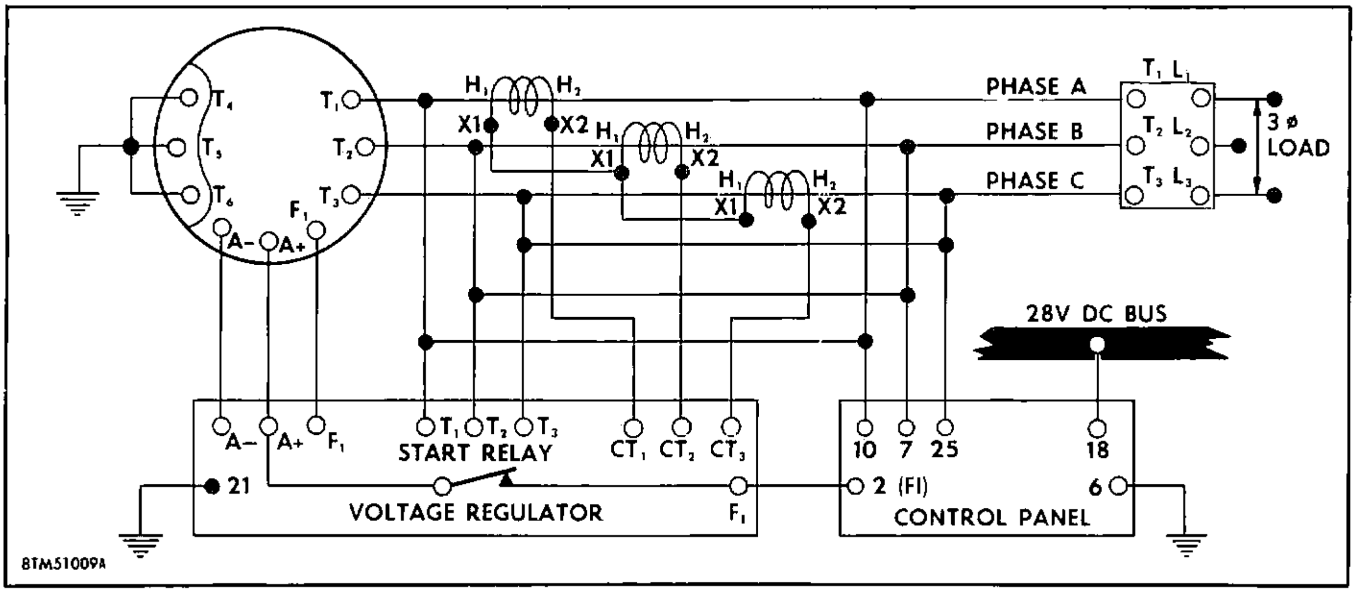


Figure 4-46. Current Transformers in Voltage Control System

terminals for the primary windings in this particular transformer, it is most important that the sides labeled H_1 and H_2 as shown in figure 4-47 be correctly located with respect to the direction of current flow in the generator leads. If there is any question in your mind concerning this statement, you are referred to Chapter I. The current transformers in this manner furnish the voltage regulator with a reduced voltage, or signal if you prefer, proportional to the output voltage of the a-c generator. In other words it is the combined effort

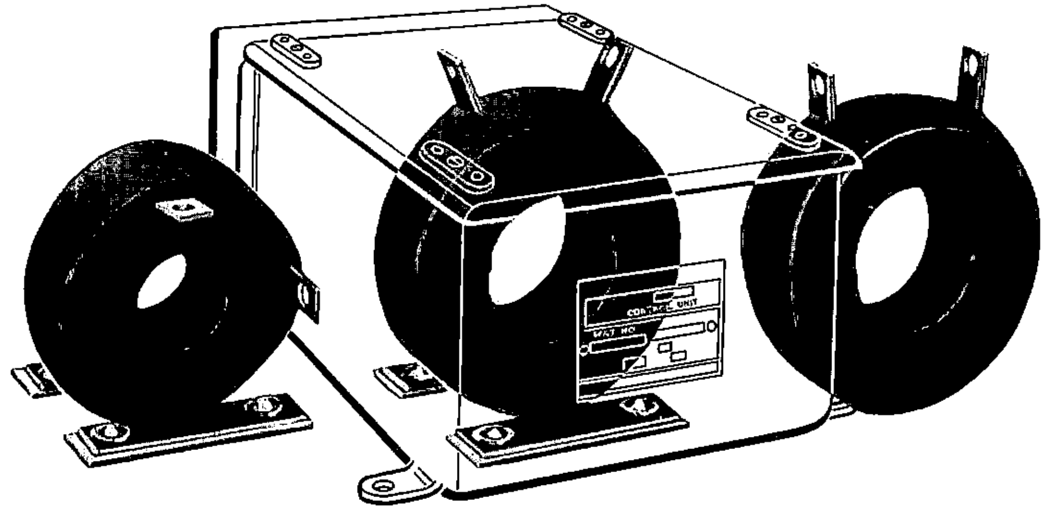


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Figure 4-47. Current Transformers

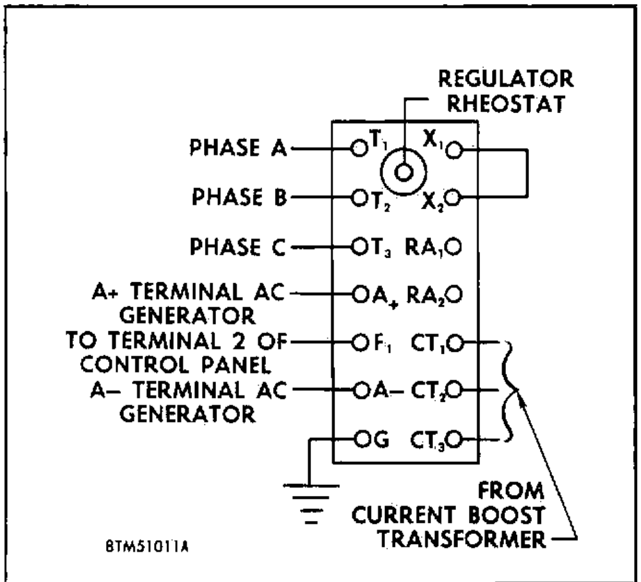

Figure 4-48. Terminal Connections of Voltage Regulator

of the output from these external boost transformers and the second-stage, or power-stage, amplifier which is supplied to the shunt field of the exciter. This is achieved, as a-c generated voltages deviate from their rated values, through the "error signal" to the first stage which is amplified, causing the power stage to adjust the exciter shunt current and thereby restore the rated line voltages.

MAINTENANCE AND THE VOLTAGE REGULATOR.

With the exception of the small spst relay switch, with normally closed contacts, which we have called the start relay, the voltage regulator has no moving parts. And since there are no moving parts subject to mechanical wear, maintenance is simply a matter of post-flight and periodic inspection for security, damage, and cleanliness.

This does not mean that the voltage regulator is foolproof, or that the regulator can be checked off haphazardly. To the contrary, special care should be taken in checking its connections for corrosion and tightness and for a generally clean condition.

The terminal board is shown in figure 4-48, along with the wires as they are connected to it.

The statement has been made that the voltage regulator is ot foolproof. This is perfectly true, just as it is true of any component found in the airplane. And when you have found, by checking the circuit, that it is the regulator which is not functioning correctly (perhaps you have found a fluctuating bus voltage or the a-c failure warning light fails to go out, but that normal voltage is indicated at terminals T_1, T_2, and T_3 on the a-c generator) you simply replace the unit. On the other hand, if the indicated voltage is too high or too low, our magamp regulator is out of adjustment. This means you must readjust the voltage at the regulator rheostat. Here again, however, if it fails to adjust, and the circuit has been thoroughly checked, as described in T.O. 1F-102A-2-10, or as suggested under our trouble shooting section in Chapter III, replace the regulator

Complete operational check-outs of the a-c electrical power system will be found in the maintenance instruction handbook, previously referred to in this supplement, T.O. 1F-102A-2-10.

The explanation of the operation of and maintenance problems encountered in the voltage regulator has been necessarily brief. When the a-c power system is analyzed, there will be further references to it which should round out your present knowledge. Also, if you have applied your present knowledge of electricity, of magnetic principles, of transformers, of rectifiers to the above explanation, you should have no trouble in understanding the function of the a-c voltage regulator as it is applied to the F-102A a-c power system. And for that matter, any magamp regulator which you might be faced with—yes, even if voltage reference tubes and "high or low phase voltage" have been incorporated. And on later models of the F-102A, electronic tubes will be found within the voltage regulator circuitry.

THE F-102A A-C CONTROL PANEL.

The a-c control panel is located in the nose wheel well, opposite to, and little forward of the d-c control panel on the left of the well. The a-c control panel in the F-102A is shown in figure 4-49. The purpose of this panel is to provide complete control and protection to the 115/200-volt, three-phase a-c electrical power system. This means that after the 120/208-volt, three-phase system has been regulated to the prescribed 115 volts at the voltage regulator, the system will become a wye-connected system, with 115 volts generated in a single phase instead of the rated 120 volts. Therefore if the line-to-neutral voltage is 115 volts, the line-to-line voltage must be 1.73 times 115 or approximately 200 volts. This is based on the generator's frequency range of 320 to 1200 cycles (wide-speed range) or 380 to 420, regulated 400, ±4 cycles, per second.

The entire control panel is enclosed. Contained within its black-box construction is a full-wave three-phase transformer rectifier system for sensing overvoltage, overvoltage relay, a trip-free overvoltage lockout relay, a latch-type generator control relay with its trip coil, and a power selector rectifier, which selects either 28-volt d-c power from the d-c essential bus, or rectified d-c power for control functions and which acts as a blocking rectifier in case of a power failure. The relays are energized by d-c power. Fundamentally there is very little difference in the theory behind the relays — or the construction — contained in this panel and those found in the d-c control panel that we covered in Chapter II. A schematic wiring diagram of this unit is shown in figure 4-50. The terminal connection drawing shown in figure 4-51 is to help you to follow more clearly the explanation of this panel. Let's talk about the Generator Control Relay first.

Generator Control Relay.

It is this relay which controls the a-c generator. Control is granted through its contacts which are in series with the generator field windings. Perhaps the easiest way to trace this out is to start with the low side of the shunt coil, F. The generator control relay, as you can see, is of the latch type with a reset (close) coil and a trip (open) coil. The mechanical operation of control relays, are described in Chapter II. The latch-type relay stays in position if d-c control power is lost.

This d-c power is provided by d-c ground return power and d-c power through terminal 4 of the master switch. By staying in its normally closed position the latch-type relay prevents simultaneous loss of a-c power. Also, in this way, it has a greater interrupting capacity, if required, to open the large field currents and voltages encountered during fault conditions. Usually a number of auxiliary contacts which are used for interlocking purposes are also controlled by this relay. As you can see in figure 4-51, these contacts in this case are designated as a trip-free lockout relay which controls the field flashing circuit.

Most generator control relays are electrically and mechanically trip-free. Another term which is used is anti-cycling. This simply means that if the reset coil is energized continuously, and if a signal is applied to the trip coil, the relay will trip and stay in this position. Without this anti-cycling or trip-free feature, the relay would cycle back and forth—from open to close, open to close—until some part of the system went completely on the fritz, in which case the airplane would be in real trouble. A good example of a signal being applied to the trip coil is in the F-102A a-c system.

You have been told that this particular control panel senses line-to-line voltages of all three-phases, through the three-phase, full-wave transformer rectifier. In figure 4-50, we have designated this as the rectifying system for overvoltage sensing. Should any one of the phases reach a sustained line-to-neutral (115-volts normal) of approximately 135 volts, the overvoltage relay coil will be energized and the generator control relay will trip.

To sum up, the generator control relay is absolutely essential to every type of aircraft electrical power sys-

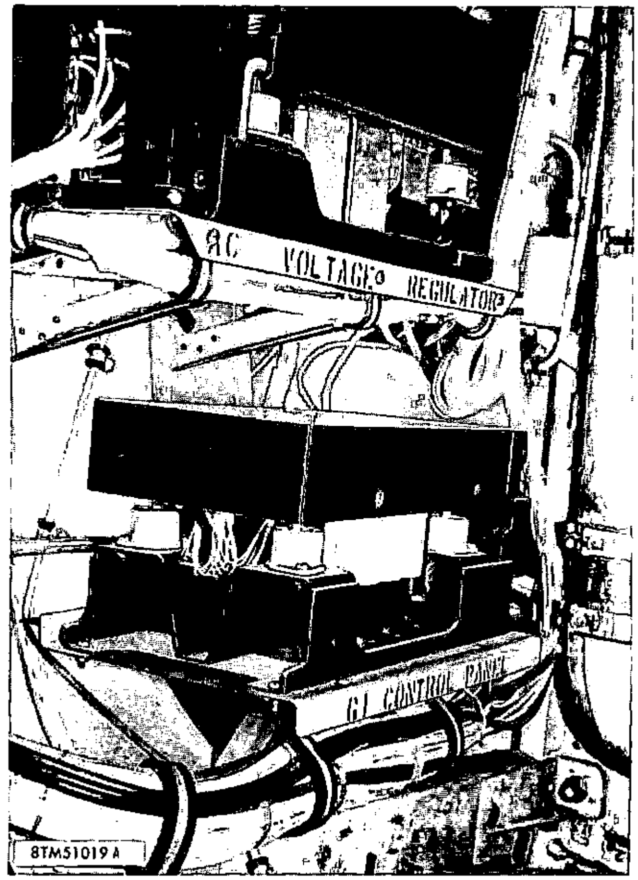

Figure 4-49. F-102A Control Panel

tem. When we discussed the d-c system, we called it out as the field relay—it controlled the generator field in this system. In the a-c system, it controls the excitation of the a-c generator and is therefore connected to trip whenever a generator or voltage regulator fault should occur.

Field Flashing the A-C Generator.

To review briefly this important phase in your electrical knowledge, field flashing circuits in self-excited a-c generators are used to insure generator build-up when the field circuit is closed. This portion of the system is discussed under the voltage regulator—the start relay if you remember, controls this build-up process. Under normal conditions, enough residual magnetism is present in the field of the generator to allow voltage build-up. If this is the case, no problem exists. There are, however, several and very possible conditions which can prevent build-up. For example, in a generator not used for a long time, or in a replacement, or in a generator which has undergone a shock of some kind, the necessary residual magnetism will, in all probability, not be present in sufficient strength. Also, environmental conditions, including aviation-fuel vapors, can and do cause film to form on commutators or on the slip rings and thus prevent build-up. So, field-flashing voltages must be sufficient to overcome all or any of these conditions.

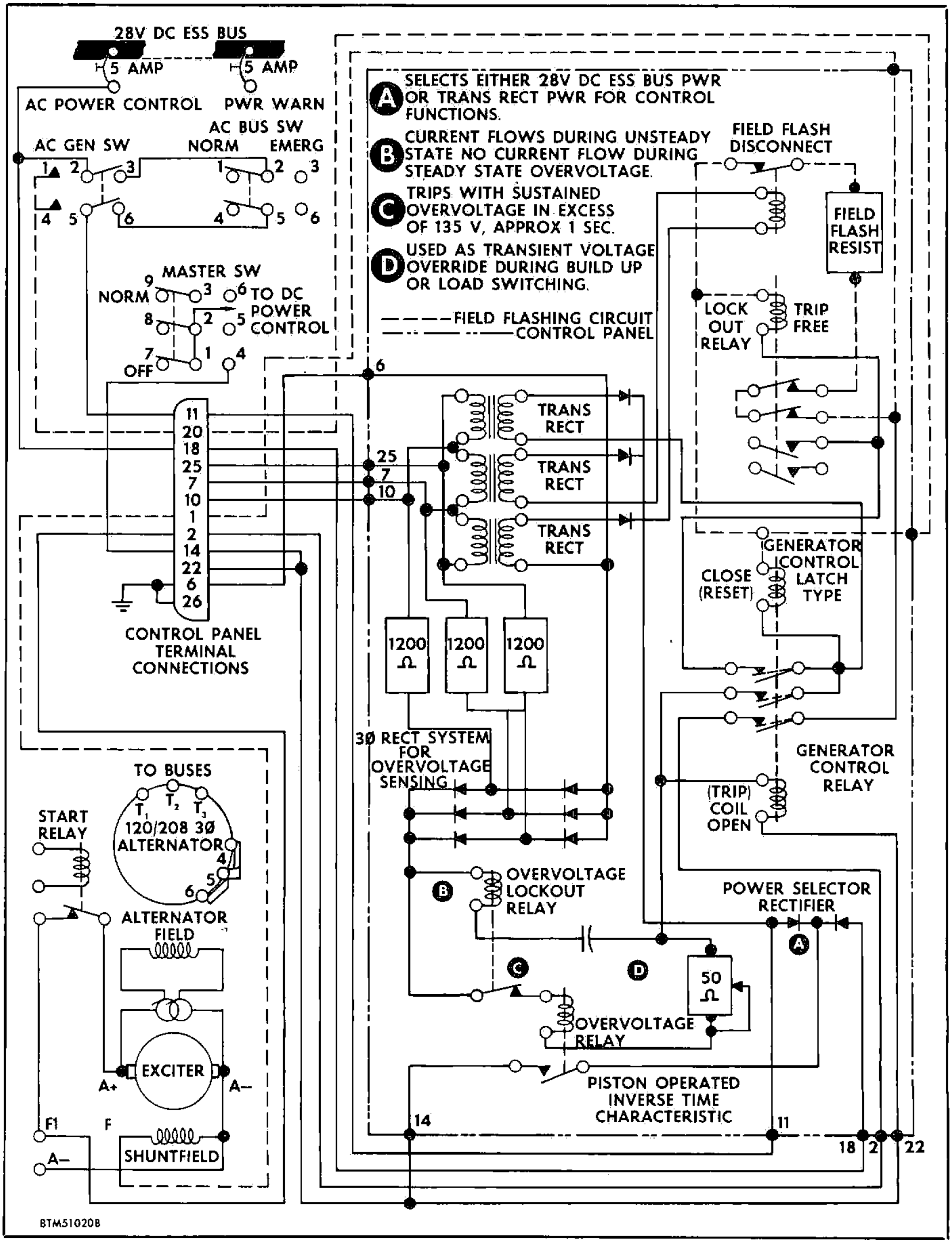

Figure 4-50. Schematic Wiring Diagram of the A-C Control Panel

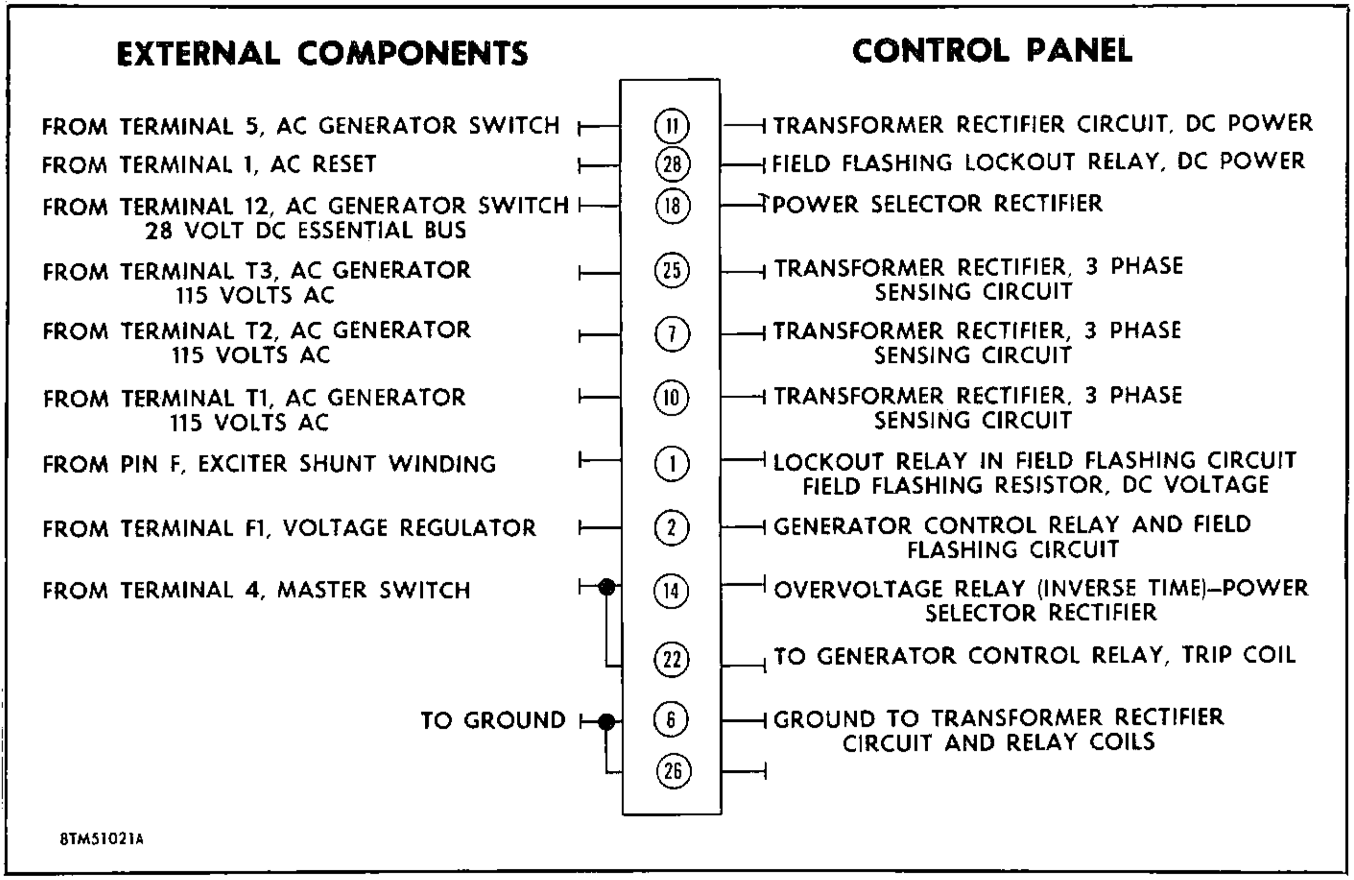

Figure 4-51. Control Panel Connection Diagram

The field flashing circuit you see in figure 4-50, which is represented by the broken lines, you might say, is held constant by the starting relay in the voltage regulator. The circuit itself is connected from the 28-volt d-c voltage supply through terminal 2 and 28 of the control panel. From here, it travels through the field flash resistor, the contacts of the lockout relay to terminal 1 of the panel and then to the F connection of the exciter shunt field. The mechanical resetting action of the a-c generator switch means that field flashing is accomplished every time the generator relay resets. The 28-volt d-c which flows assures exciter field build-up.

The Overvoltage Relay.

Here again, just as in the d-c system, we have an overvoltage relay with an inverse-time characteristic. You must remember, however, that we are now dealing with a-c voltage, so, even if the mechanical action and basic theory behind this relay are the same, there are differences. This relay is perhaps the most essential of all relays in any aircraft power system, because, as in the case of an a-c system, it not only smooths the ruffled features of an over-excited generator but also protects the connected loads and the power system as a whole from serious damage. Let's study this relay as it pertains to the a-c system of the F-102A.

HOW THE OVERVOLTAGE RELAY WORKS IN THE A-C SYSTEM. As you know, loads connected to the a-c generator are designed to operate at a certain maximum voltage and a minimum frequency. In the F-102A a-c control panel, the overvoltage relay protects these loads from sustained high voltages but allows normal transient overvoltages to pass. Any damages occurring from voltages higher than normal are mostly due to the effect of heat in the relay coils—this, of course, is directly attributed to the voltage involved and time: the higher the voltage, the less time the components in the system can withstand it. For this reason, just as it is in the d-c system, the a-c overvoltage relay must have an inverse-time characteristic.

As we have mentioned, the overvoltage relay senses the average value of the three-phase voltage. This is obtained by rectifying and thereby obtaining a d-c voltage.

The three 1200-ohm resistors which you see in figure 4-50 are known as dropping resistors and are actual electrical sensing devices used to sense the output voltage of the a-c generator and indirectly operate the overvoltage relay. With this type sensing, a simple-phase line-to-ground fault will not operate the relay, for when a fault of this kind arises, it will cause at least one of the other two-phase voltages to rise also. If the relay sensed only one-phase voltage, it would operate on a single-phase fault and not allow the fault to clear itself.

It is the inverse-time characteristic of the overvoltage relay which protects the system from nuisance trips.

In the case of a sustained overvoltage of 135 volts or above, the overvoltage relay will operate in approximately 1 second; at 200 volts approximately .2 seconds; and at 300 volts, the operating time of the inverse-time characteristics will be 0.05 seconds, with the overvoltage lockout relay disconnected.

THE OVERVOLTAGE LOCKOUT RELAY. An overvoltage lockout relay, as shown in figure 4-50 is used when it is necessary to override voltage transients which occur on generator build-up. In a single a-c generator system with a wide speed range, as used in the F-102A, these transient voltages can be extremely high at the upper end of the speed range. The fundamental theory behind this relay is simply that the ordinary overvoltage relay will trip under these transients. It is for this reason that the lockout relay is incorporated into the system, to provide an extra time delay when certain high transient voltages occur during build-up or load switching. This tripping is known as nuisance tripping.

To overcome this, the overvoltage lockout relay coil is connected in series with a capacitor, so that under steady voltage conditions, the lockout coil will have no current. But under high transient voltage, current will flow through the coil. And when this current is of such a magnitude, the surge of emf causing it will trip the overvoltage relay, opening the contacts of the overvoltage lockout relay and thereby opening the circuit to the generator control relay coil. In other words, this extra time delay is energized from a generator control relay which holds the overvoltage relay circuit open for a small fraction of a second to override this transient voltage. Once the system is in operation, the overvoltage lockout relay has no effect on the operation of the overvoltage relay.

The instant the voltage returns to a steady stage, the overvoltage relay closes and the overvoltage relay coil is reconnected. If this steady voltage condition is high, the overvoltage relay will disconnect the generator from the system through the external a-c power disconnect relay.

The Power Selector Rectifier.

In order that the a-c system be independent of the d-c system during normal operation, the transformer rectifier supplies sufficient d-c power to operate the a-c power disconnect relay and the relays we have just covered. The power selector rectifier acts in the capacity of a blocking rectifier, and through its blocking action selects either 28 volts from the d-c essential bus, channeled through terminal 18 of the control panel, or rectified d-c power for performing these control functions. In other words, during normal operation of the a-c generator, the transformer rectifier system is the source of d-c power, but it is the d-c essential bus which is the power source for reset action and field flashing.

You and the A-C Control Panel.

As in the case of the voltage regulator, when the control panel is found inoperable or erratic in its operation, the panel will be removed and replaced. And keep in mind that this unit contains a number of mechanically-activated electrical devices which are subject to wear. Remember, also, that the control panel is a precision instrument and is sensitive to rough handling. It is likewise an instrument of proven reliability. If the proper precautions are taken, if you perform your inspections for its security and cleanliness, and if you treat it with the respect due it—as one of the most important electrical components in the airplane—it should give long and efficient service. So much for the 30-kva control system of the F-102A.

THE F-102A A-C POWER SYSTEM ANALYSIS.

This supplement, as you know, is essentially a supplement which concerns the a-c and d-c electrical power systems from their sources of power to the essential and non-essential bus systems as used in the airplane. We are now ready to distribute this controlled power to the bus systems.

In the following discussion, we will start with the switch gear which motivates this distribution. When these have been wired into the system, you will learn those additional relays and electrical devices which make up the 115/200-volt a-c electrical power system. A location diagram of the various a-c components found in the system is shown on figure 4-41, and an a-c schematic wiring diagram on figure 4-40. These diagrams will be used throughout this discussion.

THE A-C ELECTRICAL POWER PANEL.

The a-c electrical power panel is shown in figure 4-52. It consists of the a-c generator switch, the a-c bus switch, and the a-c selector switch. Above the a-c portion of the electrical-power panel, and to the right of the warning light panel, is the a-c voltmeter which will be discussed in conjunction with the a-c selector switch. It might be well to remind you that when *any switch* is positioned to the "on" position, *current flows,* and, with few exceptions, when a relay coil is energized and the relay contacts close in a circuit, *current flows.* So, in the following explanation of the wiring pattern, keep in mind that when terminal such and such, is connected through a cable to another such and such terminal, this is the trail taken by our electrons.

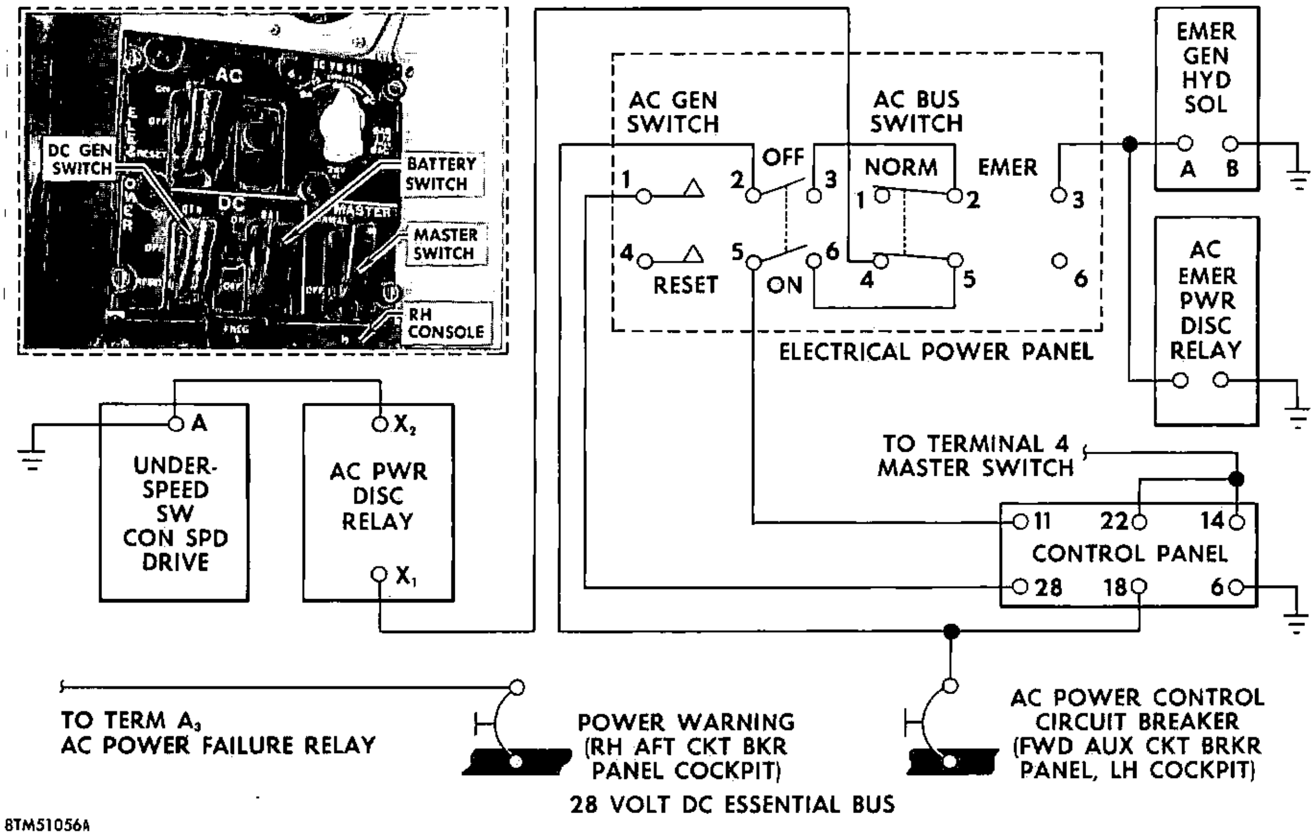

Figure 4-52. Electrical Power Panel

The A-C Generator Switch Wired Into the System.

The a-c generator switch is a dpdt, momentary on-off switch. The simplified wiring schematic, figure 4-53, shows this switch as well as the a-c bus switch as they are wired into the system.

In the reset position, contacts 1 and 2 connect the d-c essential bus momentarily to terminal 28 of the a-c control panel. This, as you have been told, sends d-c voltage to the control panel for flashing the field of the a-c generator. When the switch is pushed to the ON position, contacts 2 and 3 connect the d-c essential bus to contact 2 of the a-c bus switch. Contacts 5 and 6 of the generator switch are connected to terminal 11 of the panel and to contact 5 of the bus switch, which through terminal 4 is wired to pin X1 of the a-c disconnect relay. X2 of this relay is attached to ground through the underspeed switch of the constant-speed drive.

The A-C Bus Switch.

This switch, as you can see in the illustration, is to the right of the a-c generator switch. It is designated as a dpdt and is shown in the closed or normal position.

The duty of this switch in the system is to connect the transformer rectifier output to the a-c power disconnect relay when it is in its normally closed position. When it is activated to the emergency position, it connects the d-c essential bus in parallel to the emergency, a-c generator hydraulic solenoid, and to pin X1 of the a-c emergency power disconnect relay. Pin X2 of this relay is connected to ground. The a-c emergency system and its components will be discussed later on in this chapter; but for the time being, when the a-c bus switch is in the normal position, contacts 4 and 5 connect terminal 11 of the a-c control to terminal X1 of the disconnect relay, providing the generator switch is in the ON position.

In the EMER position, contacts 2 and 3 of the a-c bus switch connect the d-c essential bus in parallel to terminal A of the emergency generator hydraulic solenoid, and, as we have said, to pin X1 of the a-c emergency disconnect relay; that is, if the a-c generator switch is in the ON position. It might be well to remember at this point that in tracing through circuits, it is always well to find the ground connection and to start from there. Pin X2 of the a-c emergency disconnect relay, for example, is connected to ground, as is terminal B of the a-c emergency generator hydraulic solenoids.

The A-C Selector Switch and Voltmeter.

The schematic diagram in figure 4-54 shows the a-c voltmeter and a selector switch connected into the power system. These two components are provided as a means of checking all a-c voltages.

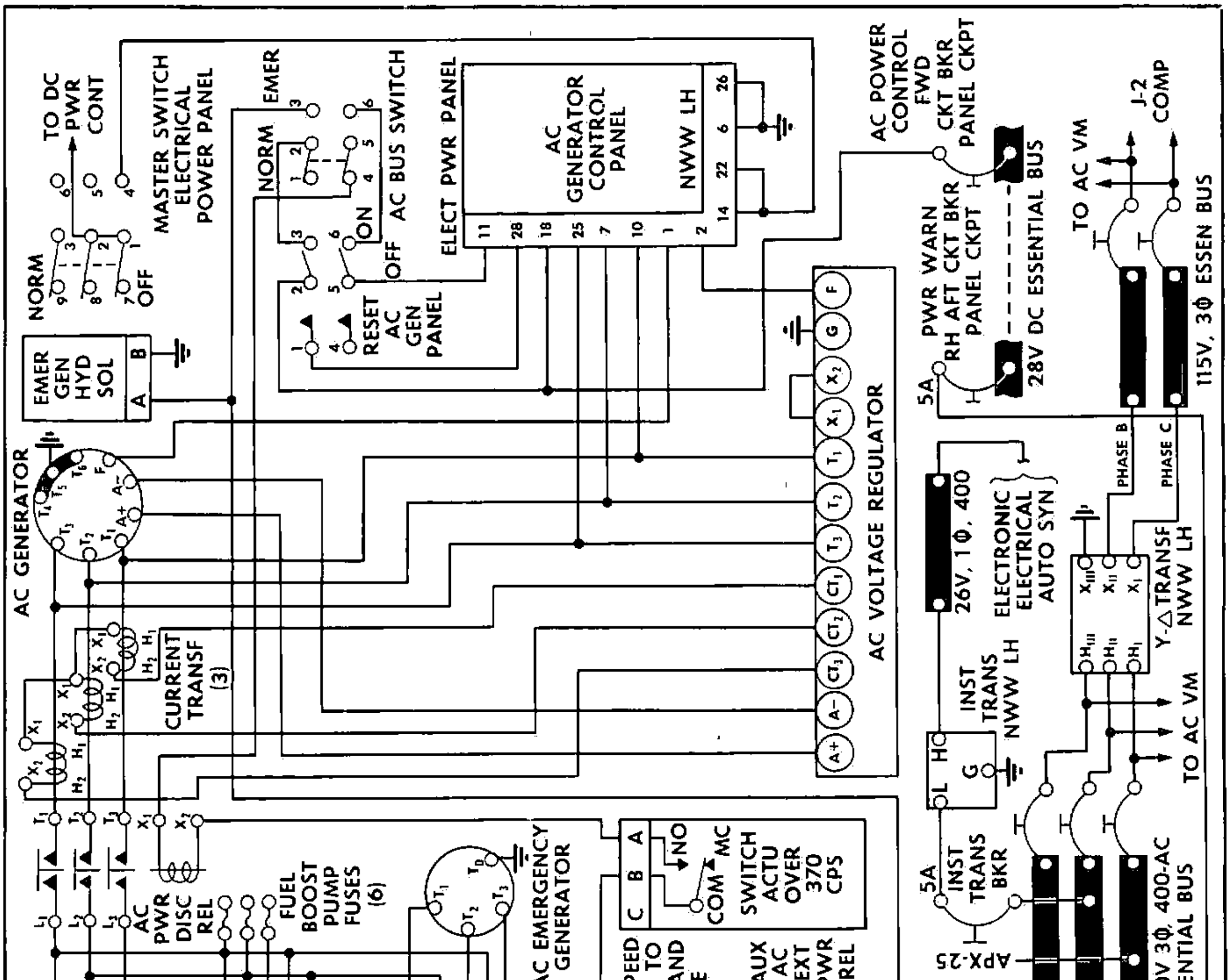

Figure 4-53. A-C Electrical System Schematic

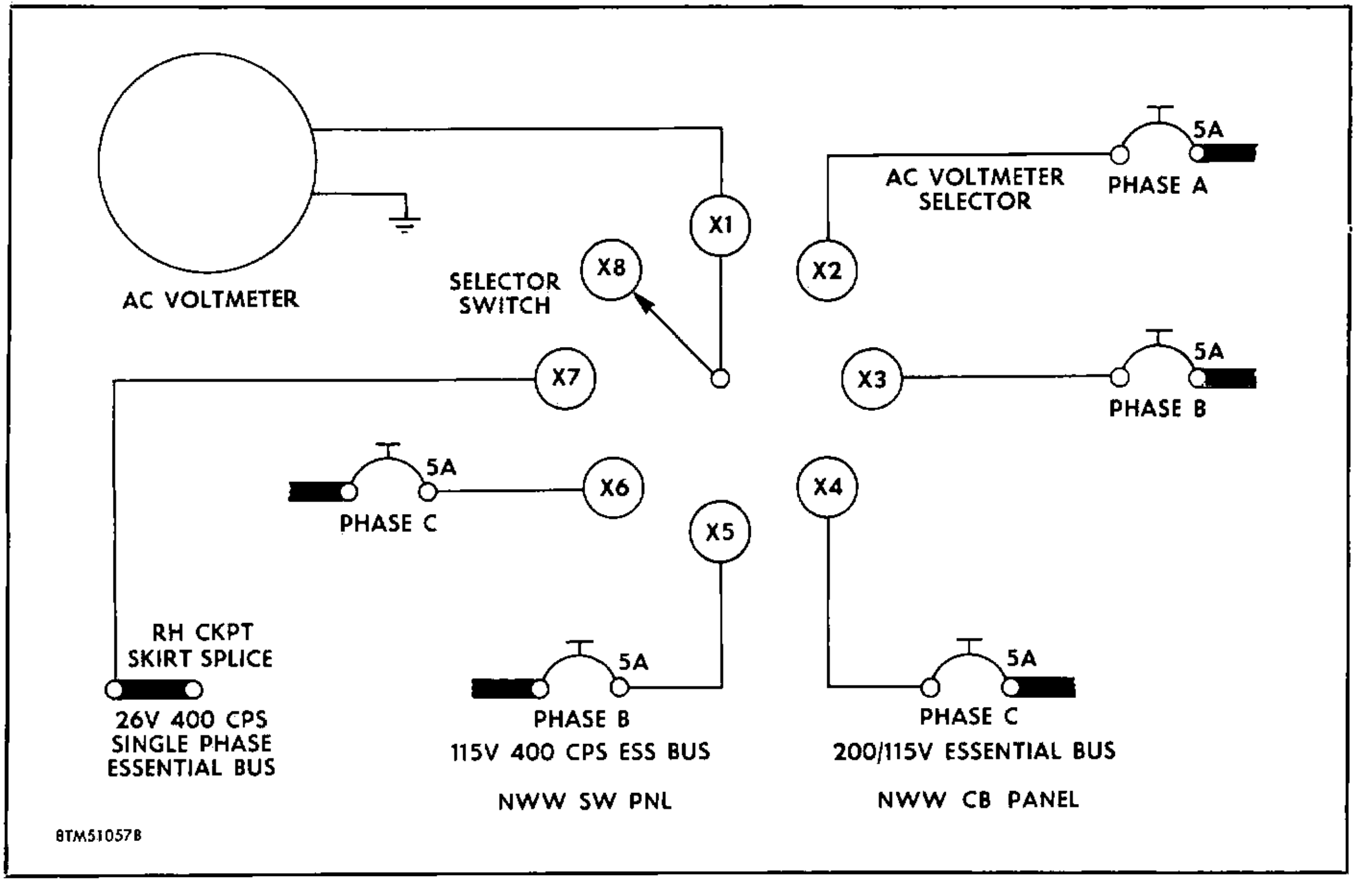

Figure 4-54. Selector Switch and A-C Voltmeter

For example, placing the selector knob in any of the phase positions ϕA, ϕB, or ϕC, connects the meter first across Phase A, then Phase B, and finally Phase C to ground. The correct reading in any of these phases should be 115 volts. If it is not, the system must be checked. Perhaps it is too high?—or too low? This would, of course, mean the voltage regulator is out of adjustment; or you may find the bus voltage fluctuating. This condition, too, would mean a complete check-through of the circuits; either the current transformers are not functioning, the voltage regulator is out of whack, or there may be a defect in the generator. At any rate, something is wrong; and whenever a malfunction of this sort occurs, the entire system must be checked.

By placing the selector knob in the 26-volt position, you connect the a-c voltmeter between the 26-volt output of the instrument transformer and ground. The voltmeter is connected directly to the 26-volt, 400-cycle, single-phase essential bus. The instrument transformer is connected through a 5-ampere circuit breaker to phase B of the 115/200-volt, 3-phase, 400-cps, a-c essential bus. The meter reading in this position should indicate 26 volts. If there is no voltage indication, then your instrument transformer has probably been shorted out, or the instrument transformer's circuit breaker has tripped. This circuit breaker is found in the nose wheel well circuit breaker panel.

A further check on the a-c voltage is provided at positions ϕAB and ϕAC. When we checked the voltage across phase A, B, or C to ground, this was an indication of the voltage at the input to the wye and delta transformer. Positions AB and AC on the voltmeter are provided to check these voltages, to ground, on the output side of this same wye-delta transformer. Both the instrument and wye-delta transformer as they are hooked into this system will be discussed after certain relays have been wired into the system. Transformers—their windings, their theory of operation—are given a brief discussion earlier in this supplement. Voltages, of course, are checked when the system is energized.

A-C POWER DISCONNECT RELAY.

The a-c power disconnect relay is designated as dpst (triple break) relay. If you will turn back to the location diagram of a-c electrical components, figure 4-41, you will find this relay is bracket-mounted to the rear of the constant-speed drive assembly. Its duty in the system is to connect the a-c generator to, and to disconnect it from, the a-c buses. A glance at the wiring diagram will show you that terminal X1 of its coil is connected to terminal 4 of the a-c bus switch; terminal X2 of the coil is connected to ground through the normally closed contacts of the constant speed underspeed switch and contacts A2 and A3 of the auxiliary external power relay which during in-flight operations is de-energized. A2, as you can see, is grounded. The other connecting wiring is plain enough and should not require explanation.

The a-c external power relay is de-energized during normal in-flight operations. Its contacts are open, and a-c power is distributed to phase A, B, and C of the 115/200-volt, 3-phase, 400-cycle, a-c non-essential bus, and to phase A, B, and C of the essential bus. The a-c emergency power disconnect relay is utilized for this distribution.

THE CONSTANT-SPEED DRIVE UNDERSPEED SWITCH.

When discussing the constant-speed drive and its relationship to the a-c power system, the action of this switch in the system was explained. It is felt, however, that a further word concerning it is necessary—before we discuss the a-c system energized for normal in-flight operations—in its relationship to the other components in the system.

As you know, this switch is located at the constant-speed drive unit on the transmission gear housing, and is centrifugally actuated through the action of charge pressure from the limit governor in the drive unit. Terminal B of this switch is connected in parallel to terminal A3 of the auxiliary external power relay which, in turn, is wired into the generator test receptacle at terminal A. You will learn further of these two components when external power is discussed. If you will glance at the schematic diagram of the system, figure 4-40, you will see that terminal A of the underspeed switch is tied to X2 connection of the a-c power disconnect relay.

Contacts A and B (C is not used) of the underspeed switch are normally open until the constant-speed drive unit obtains a speed slightly under 5550 rpm, which is 370 cps; at this time, the contacts close and remain closed at speeds up to 6000 rpm ±60, or 400 cps. On shut-down, the contacts will open just before the generator speed reduces to 5250 rpm, or 350 cps.

THE A-C EMERGENCY POWER DISCONNECT RELAY.

This relay is a dpdt, single-break relay switch. And in the de-energized position, that is, during normal in-flight conditions, its contacts carry the necessary a-c power to the essential bus. If you will trace this power in the wiring diagram, figure 4-40, you will find that when an emergency arises, the solenoid coil of the emergency disconnect relay pulls the contacts into their emergency position, disconnecting the non-essential bus and allowing a-c power from the 1-kva emergency generator to feed the essential bus. In other words, whenever it is energized, this relay is exactly what its name implies. It disconnects the a-c generator from, and connects the a-c emergency generator to, phases A, B, and C of the a-c essential bus. Terminal X1 of the coil is connected in parallel to terminal A of the emergency generator and to terminal 3 of the a-c bus switch, which is the emergency contact. Terminal X2, as you can see, is grounded. The a-c power leads of the emergency generator, T1, T2, and T3 are connected to this relay at contacts C1, B1, and A1.

THE A-C POWER FAILURE RELAY.

This relay is a spdt, single-break, relay switch and is located forward of the cockpit instrument panel close by the master warning relay box. The relay is present in the system to connect the d-c essential bus to, and to disconnect it from, the master warning system. Terminal X1 of the coil is grounded; X2 is connected to phase C of the 115/200-volt non-essential bus through the a-c power failure warning circuit breaker. This circuit breaker is located in the right-hand aft circuit breaker panel in the cockpit of the airplane. Contact A3 of this relay is connected to the d-c essential bus through the 28-volt d-c power warning circuit breaker. Contact A2 connects with terminal R of the master warning relay box. A2 and A3 are in contact when the a-c power failure relay is de-energized.

THE WYE-TO-DELTA TRANSFORMER WIRED INTO CIRCUIT.

The wye-to-delta 115/200-volt transformer is located in the left-hand side of the nose wheel well. The transformer steps down the 115/200-volt, 3-phase, 400-cycle a-c generator output of both the 30-kva and 1-kva machines to a regulated 115-volt, 2-phase output in phases B and C on the 115-volt essential bus. The high side of this particular transformer is wye-connected, with the coils connected from line-to-line. The low side of the transformer is connected in a delta configuration with a two-wire ground system. Phase A is grounded. (The basic principles behind *step-down* and *step-up* transformers are discussed on pages 164 and 165; refer to this material if you do not understand the above discussion.)

THE INSTRUMENT TRANSFORMER WIRED INTO CIRCUIT.

The 115-volt to 26-volt, 400-cycle, step-down transformer, is located in the left-hand side of the nose wheel well, and provides power for the 26-volt a-c essential bus. The 26-volt, 400-cycle power is used for instruments. The primary of the transformer is connected to the a-c essential bus, phase B, through the 5-amp instrument transformer circuit breaker, which is located on the nose wheel well circuit breaker panel.

THE A-C POWER SYSTEM ENERGIZED FOR NORMAL FLIGHT CONDITIONS.

Since the field flashing of the a-c generator depends upon d-c voltage, it stands to reason that the d-c power must arrive in the a-c generator field-flash circuit by some means. It arrives the instant you place the d-c generator switch to the ON position and the a-c generator switch momentarily to RESET. It is d-c battery power, then, which is used for field flashing the a-c generator. It is taken directly from the d-c essential bus. Let's review, and find out how d-c power arrives at the d-c essential bus and how the a-c system utilizes this d-c electricity.

D-C Battery Voltage in the A-C System.

When the battery switch is placed in the ON position (see figure 3-49 in Chapter III), terminal X2 of the battery relay is connected to contact 9 of the master switch. And the master switch being in the normal position, contact 9 is connected to contact 3 and contact 3 is connected directly to ground. Thus, the battery relay is energized, contacts A1 and A2 close and connect the battery to the d-c essential bus.

The initial battery power has other functions in the system, as well. For example, by closing the power shutoff circuit breaker, located in the nose wheel well circuit breaker panel, the battery is connected to terminals 1 and 2 of the master switch. And by closing the power warning circuit breaker, located in the forward auxiliary circuit breaker panel, you connect A3 of the a-c power failure relay to the d-c essential bus. Contact A3 is wired to terminal R of the master warning relay box. Thus, the master warning and the a-c power failure indicators will be illuminated. They extinguish when the system is energized, and come on again only when there is a malfunction of the a-c 30-kva generator. This is essentially the same as the generator low warning light on the dashboard of one of the newer automobiles. This a-c power control circuit breaker is most important to our present discussion. When closed, it connects in parallel terminal 18 of the a-c generator control panel and contact 2 of the a-c generator switch to the d-c essential bus. When the generator switch is placed momentarily in reset position, contacts 1 and 2 of the a-c generator switch are closed, and the d-c essential bus is once again connected into the a-c system through terminal 28 of the control panel.

The A-C Control Panel and D-C Voltage.

Terminal 28 of the control panel is now connected in parallel to the generator control relay's reset coil, to the lockout relay coil, and indirectly to terminal F of the exciter shunt field of the a-c generator through the normally closed contacts of the de-energized lockout relay in the field flash circuit. These contacts of the lockout relay are normally closed, and power is brought through the above parallel circuit to terminal 1 of the a-c control panel. A 30-ohm, 25-watt resistor is connected in series between the closed contacts of the field flash disconnect relay, and the normally closed contacts of the lockout relay. The exciter shunt field, thereby, limits the exciter shunt field current flowing through terminal 1 to approximately 10 amps when the a-c generator switch is pressed; and the reset cycle is set in motion for the purpose of flashing the alternator field. A word concerning this field flash disconnect relay: in many control panels this relay is called a power indicator relay. In this case, however, there is no power indicator attached, and its sole purpose is to bridge the field flash circuit.

The A-C Generator Switch, On.

When the a-c generator switch is placed in the ON position, voltage is removed from the generator relay close coil, from the lockout relay coil, and from the exciter shunt field of the generator. "How?" you may ask. Well, as you can see by taking a look once more at the schematic diagram of the control panel, figure 4-50, the low side of the generator control relay reset coil is connected to terminal 6 which is grounded. In this manner, when the generator control relay is energized and mechanically latched in, it connects the low-side of the lockout relay coil and the low-side of the generator control relay open coil to terminal 6 which is grounded. Therefore, the d-c essential bus is connected to terminal 28 of the a-c generator control panel only during that period when the reset switch is closed.

A-C Power on the Line.

As you know, or should know, the transformer rectifier output in the a-c control panel is connected in parallel to the power selector rectifier and to terminal 11 of the a-c control panel which is connected to terminal X1 of the a-c power disconnect relay. This connection is made through contacts 5 and 6 of the a-c generator switch when it is in the ON position and through contacts 4 and 5 of the a-c bus switch in its NORMAL position. Terminal X2 of the a-c power disconnect relay is connected to ground through the underspeed switch and contacts A2 and A3 of the auxiliary a-c external power relay. When the a-c generator attains a speed of 5550 rpm (370 cps), and the d-c ouput of the transformer rectifier unit of the a-c generator control panel is 18 volts, or more, the a-c power disconnect relay will be energized and close its contacts, and the a-c generator's regulated output will appear at the a-c essential bus through contacts A2 and A3, B2 and B3, and C2 and C3 of the a-c emergency power disconnect relay. As you can see, phase C on the a-c non-essential bus is connected to terminal X1 of the a-c power failure relay through the a-c power failure warning circuit breaker and terminal X2 completes the circuit to ground. In this manner, the a-c power failure relay is energized: opening contacts A2 and A3 disconnects the d-c essential bus from terminal R of the master warning box, and the master warning and the a-c power failure lights are extinguished. Let's find out what happens to bring them on again.

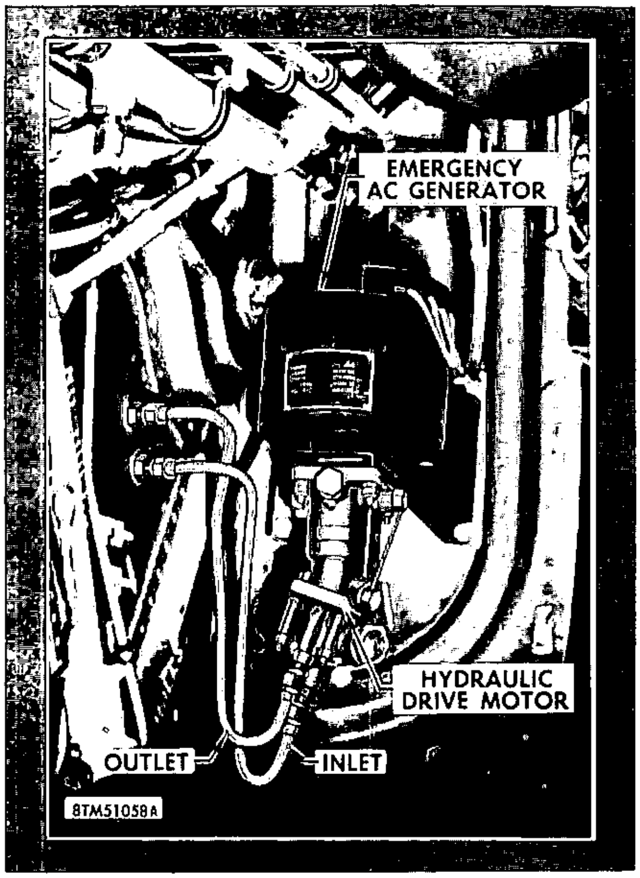

Figure 4-55. F-102A Emergency Generator

30-KVA Generator Taken Off the Line.

It is the a-c generator control panel which provides overvoltage as well as undervoltage protection for the a-c generator. And it is the three-phase sensing rectifier —connected to the a-c generator output through the three 1200-ohm resistors—which is used to sense the output voltage of the a-c generator for operation of the overvoltage relay. This sensing rectifier system supplies sufficient d-c voltage to energize the overvoltage relay when a sustained voltage of over 135 volts appears at the a-c generator output. When the overvoltage relay is energized, its normally open contacts close. This action connects the power selector rectifier d-c output to terminal 14 of the control panel, which is connected externally to terminal 22. Terminal 22 is connected internally to the generator control relay open coil. It is through this connecting network that the generator control relay contacts are opened, which removes the d-c exciting voltage from the exciter shunt field.

The generator control relay's mode of operation was covered when we discussed the a-c control panel on page 196. However, when the d-c output of the transformer rectifier appears at terminal 11 of the panel, it is reduced below the drop-out voltage of the a-c power disconnect relay, which, incidentally, is an approximate 7 volts, and the relay will be de-energized, and its contacts will open. When the 30-kva a-c generator is taken out of the system and the a-c generator output is removed from the a-c buses, the a-c power failure relay is de-energized. In its de-energized position, contacts A2-A3 are closed, and the d-c essential bus is connected to terminal R of the master warning relay box, illuminating the master warning and the a-c power failure indicator lights. Simple?

THE F-102A AND A-C EMERGENCY POWER.

Figure 4-55 shows the a-c emergency generator and its hydraulic motor as it is mounted on the right-hand side of the main wheel well. The line drawing shows the generator and its component parts as they are installed. As you can see, the combined unit is mounted in a vertical position. The hydraulic motor which drives it is installed at the bottom.

The 1-kva, 120/208-volt, three-phase, 400-cycle, a-c generator supplies limited power to the a-c essential bus for emergency operations.

The A-C Emergency Generator.

When available, a more complete exploded view of this generator will be incorporated into this supplement along with an explanation of its operating parts. Until more information is available a brief explanation of its operating qualities and design features should suffice.

The machine has a minimum lagging power factor of 0.8. It is a permanent magnet variable, axial air-gap type, with wye-stator configuration. The three phases and neutral are brought out to four terminals on the base, T0, T1, T2, and T3—T0 is connected to ground. Six magnets are mounted on the face of the rotor, 60° apart. The air gap is adjusted by varying the position of the stator, and by this variance in the air gap it is possible to change the output voltage. The theory behind this air gap adjustment is thoroughly explained in Chapter II, in covering the carbon-pile voltage regulator.

These adjustments are critical, however, and should not be attempted except by technicians trained for the job who are equipped with the required shop level equipment. In other words, these adjustments are not performed on the flight-line, any more than the servicing of a magnetic amplifier is accomplished on the line.

The 1-kva emergency generator is designed to maintain an output of 120 volts ±10% between one-half load and full load at 400 cycles ±5%, and has a speed range of 380 to 420 cycles per second. It is also designed to operate in an ambient temperature of 38°C, and with a temperature rise of 40° above ambient for a period of ten minutes. The movement of the rotor provides cooling and gives considerable aid in curculating air through the openings around the top of the housing proper.

THE EMERGENCY GENERATOR HYDRAULIC MOTOR. The hydraulic drive motor installation drawing is shown in figure 4-56. As you can see, this motor consists of a main housing generator pad with connections for the inlet and outlet lines which port hydraulic fluid from the secondary hydraulic system. This hydraulic drive motor is a constant-speed, piston-type unit. A self-contained speed control valve maintains the driving speed at 8080 rpm ±400, with an inlet pressure from the secondary hydraulic system which varies from 2000 to 3000 pounds per square inch (psi). It is the emergency hydraulic solenoid, which you see in the schematic wiring diagram of the a-c power system, figure 4-40, which, when energized by the a-c bus switch, ports hydraulic fluid to the a-c emergency hydraulic motor. A more complete explanation of this emergency a-c generator hydraulic system will be found in the Hydraulic System Supplement of this training series.

THE EMERGENCY GENERATOR HYDRAULIC SOLENOID. This solenoid valve is electrically operated and controls the flow of hydraulic fluid to the emergency a-c generator's hydraulic motor just discussed. When emergency power is needed, the a-c bus switch is switched to the EMER position. This directs electrical power to the solenoid valve, opening the valve, and releasing pressurized hydraulic fluid.

Terminal A of this solenoid is connected in parallel to terminal 3 of the a-c bus switch and to terminal X1 of the a-c emergency power disconnect relay. Terminal B of the hydraulic solenoid is connected to ground.

When the 30-kva generator goes off the line, the master and the a-c power failure lights come on; unlike the d-c system, in which the emergency power automatically takes over, the a-c bus switch must be actuated to the emergency position (EMER).

This action connects the d-c essential bus in parallel to terminal A of the emergency hydraulic solenoid and to terminal X1 of the a-c emergency power disconnect relay. Terminal X2 of this relay is connected to ground.

1-KVA EMERGENCY POWER ON THE LINE.

The a-c emergency electrical power system is shown in figure 4-57. The d-c power is shown as a comparative system and to give you a clearer concept of the emergency system as a whole.

For our analysis, the overall schematic, figure 4-40, will be used in conjunction with the a-c - d-c composite schematic, figure 4-57.

The loss of a-c power is indicated by the illumination of the master warning indicator light in the upper right corner of the instrument panel and by the a-c power failure warning indicator light in the warning indicator light panel. During flight operation, if a situation like this occurred, the pilot would attempt to restore power to the a-c bus by placing the a-c generator switch in the reset position momentarily, then to the ON position. If this action failed to extinguish the a-c power failure warning indicator light, and the a-c voltmeter indicated no voltage, he would place the a-c bus switch in its EMER position. This also holds true during operational checks and testing. These procedures will be found in T. O. 1F-102A-2-10.

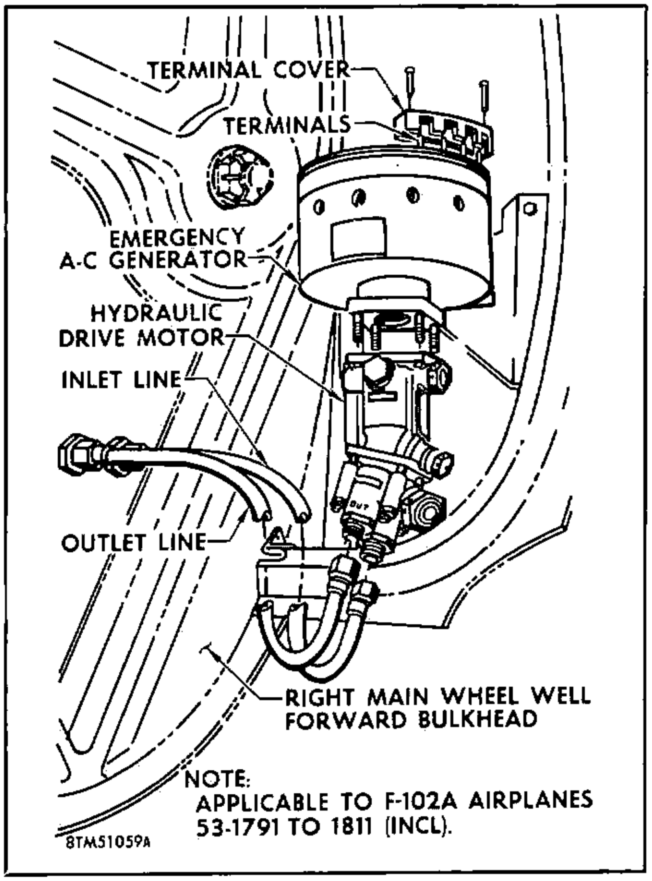

Figure 4-56. Emergency Hydraulic Motor and Solenoid

When the a-c bus switch is placed to EMER position, the switch contacts 4 and 5 open, and contacts 2 and 3 close. The d-c essential bus is now connected in parallel to terminal A of the emergency generator hydraulic solenoid, whose terminal B is connected directly to ground, and to terminal X1 of the a-c emergency power disconnect, X1 being also grounded. In this manner, the emergency generator hydraulic solenoid and the a-c emergency power disconnect relay are energized. And as you have learned, energizing the emergency generator hydraulic solenoid ports hydraulic fluid from the secondary hydraulic system directly to the hydraulic motor. The energizing of the a-c emergency power disconnect relay opens contacts A2 and A3, B2 and B3, and C2 and C3, closing contacts A1 and A2, B1 and B2, and C1 and

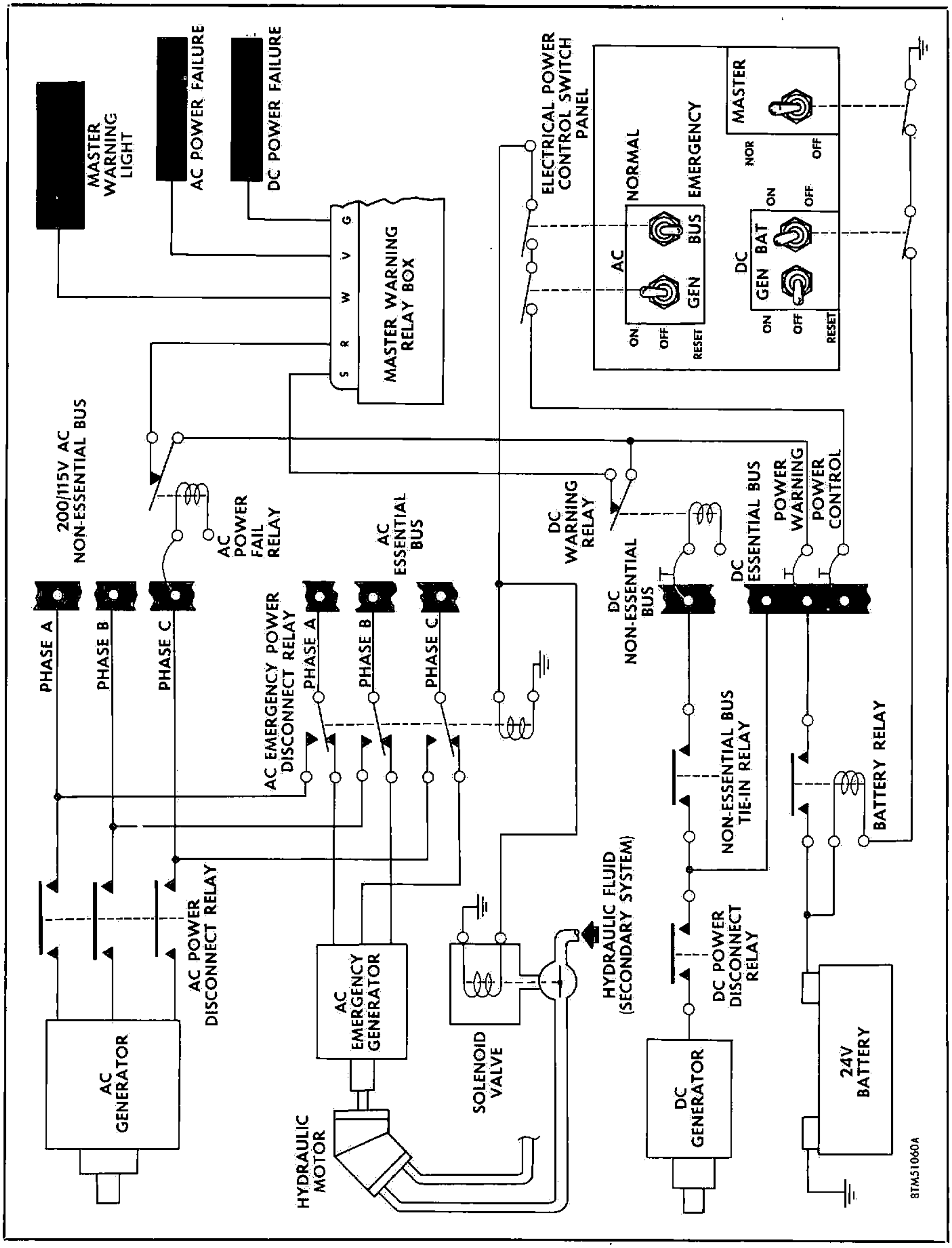

Figure 4-57. Emergency Power on the F-102A

C2. As you can see in figures 4-40 and 4-57, the opening of contacts A2 and A3 disconnects contact L3 of the a-c power disconnect relay and connects terminals A2 and A1. A1 is connected directly to emergency generator terminal T3. Phase C of the a-c non-essential bus and phase C of the fuel boost pump bus, as you can see, are, in this manner, disconnected from phase C of the a-c essential bus. In like manner, the opening of contacts B2 and B3 disconnects contact L2 of the a-c power disconnect relay and connects B1 and B2 to T2 of the generator terminal block. Phase B of the a-c non-essential bus and phase B of the fuel boost pump bus are now disconnected from phase B of the a-c essential bus. And so it goes right on down the line, C2 and C3 disconnect contact L1 of the a-c power disconnect relay, contact L1 of the a-c external power relay, phase A of the a-c non-essential bus, and phase A of the fuel boost pump bus from phase A of the a-c essential bus. The closing of contacts C1 and C2 completes the connection at terminal T1 of the a-c emergency generator, and the 1-kva generator now takes over the system. By tracing through the system, you can see that all three phases are now connected—phase A, B, and C—to the a-c essential bus. The non-essential bus has been completely cut out of the system. The a-c power failure indicator light will remain lighted, however, since phase C of the a-c non-essential bus is not connected to the output of the a-c emergency generator. So much for the F-102A under emergency power.

THE F-102A AND A-C EXTERNAL POWER.

The MD-3 unit used to provide the necessary electrical current is illustrated in figure 3-56, Chapter III of this supplement. This ground support unit supplies the a-c external power for use in ground operations of the F-102A. For complete information regarding this equipment you are referred to T. O. 35C2-3-249-1.

Ground power, of course, is used mainly for the proper servicing during maintenance operations on the F-102A. In this manner, the airplane is properly checked out under various modes of operational flight. Operational checks and tests for ground operations will be found in the applicable maintenance instructions manual T. O. 1F-102A-2-10. A trouble shooting chart with possible trouble, its probable cause, and the correction of the malfunction will also be found in that manual.

Connecting In A-C External Power.

The d-c external power receptacle will be found on the forward bulkhead of the left-hand side of the main wheel well. This receptacle is above the d-c external power receptacle and is illustrated with external power connected in Chapter III (figure 3-57).

Before connecting a-c external power to the airplane, be very sure that the a-c generator and the d-c battery switches are in their OFF positions and that the a-c bus switch is in the NORMAL position.

If external power is to be used on the airplane for long periods of time during your maintenance operations, there are certain circuit breakers which should be pulled to their tripped position. Four of these are in the upper electronics compartment circuit breaker panel and are labeled: intph AIC-10DC, Glide Path ARN-18DC, Glide Path ARN-115VAC, and MARKER BEACON ARN-12DC. Three more will be found in the nose wheel well: J-2 comp., J-2 compass and voltmeter; PHB and PHC; and the one labeled PILOT ASSIST. Three others which must be pulled will be found in the main wheel well circuit breaker panel and are labeled: FLIGHT CONTROL, FLIGHT CONT DAMPER, and PITCH. The reason for pulling these breakers is that the above circuits are energized whenever electrical power is impressed on the airplane.

In Chapter II, pages 72 through 80, the circuit breaker panels found in the F-102A are illustrated. At the end of this chapter, the a-c circuit breakers found in these panels are listed with decal, their amperage rating, the phase voltage, bus, and the circuit they protect.

The A-C External Power Receptacle.

If you will take another look at the schematic wiring diagrams, figures 4-40 and 4-57, you will find that terminals A, B, and C, of the external power receptacle are connected to contacts T1, T2, and T3 of the a-c external power relay. Terminal E is connected in parallel to terminal X1 of the a-c external power relay and to terminal X1 of the auxiliary a-c external power relay. Terminal F is connected in parallel with terminal D of the generator test receptacle to the d-c essential bus through the a-c external power circuit breaker. Terminal N is connected to ground.

The Generator Test Receptacle.

The generator test receptacle, located on the forward bulkhead on the left-hand side of the main wheel well, is used in conjunction with the a-c external power receptacle to energize the a-c external power and auxiliary a-c external power relays without interfering with the operation of the a-c power disconnect relay. Terminal A of this test receptacle is connected to terminal X2 of the a-c power disconnect relay through contacts B and A of the constant-speed drive, underspeed switch when this switch is energized. Terminal A is also connected in parallel to ground through contacts A2 and A3 of the auxiliary a-c external power relay when this relay is de-energized. To return to the test receptacle, terminal B, as you can see, is connected directly to ground while terminal D is connected to the d-c essential bus through the a-c external power circuit breaker; terminal D is also connected in parallel to terminal F of the a-c external power receptacle. By turning to the schematic wiring diagram of the d-c system, figure 3-50, Chapter III, you will see that terminal C is connected in parallel to terminal 1 and contact 7 of the d-c external power interlock relay. Terminal C is also connected to the d-c external power relay and to the common terminal of the d-c external power and the d-c auxiliary power receptacles. Terminal E of this test receptacle, in turn, is connected in parallel to contact 4 of the d-c external power interlock relay and to terminal X1 of the d-c disconnect relay. Contact 4 of the d-c external relay is connected to ground through contact 2. All this sounds rather complicated, but if you will trace this through, you will see that it is in this manner that the a-c power disconnect relay is isolated, and is not interfered with during ground operations when the a-c external power and auxiliary external power relays are energized.

The A-C External Power Relay.

The function of the relay in this system is to connect the a-c external power to the a-c buses. The coil terminal X2 of this relay is connected to contact A1 of the auxiliary a-c external power relay; terminal X2 is connected in parallel to terminal E, of the a-c external power receptacle, terminal D of the generator test receptacle, and to terminal X1 of the auxiliary a-c external power relay. Contacts T1, T2, and T3 are connected to terminals A, B, and C, in that order, of the a-c external power receptacle. Contacts L1, 2, and 3 are, of course, connected to contacts L3, 2, and 1 of the a-c power disconnect relay.

The Auxiliary A-C External Power Relay.

The function of this normally closed relay in the system is to disconnect terminal B of the underspeed switch and terminal A of the generator test receptacle from ground. In the energized position, as you can see, it provides a ground connection for terminal X2 of the a-c external power relay. Terminal X1 of the auxiliary a-c external power relay and terminal X1 of the a-c external power relay are connected in parallel with terminal D of the generator test receptacle, terminal E of the a-c external power receptacle, and finally to the d-c essential bus through a jumper wired between connector pins E and F in the a-c external power cable connector. Terminal F is connected directly to the 5-amp external power circuit breaker in the main wheel well circuit breaker panel. Terminal X2 of this auxiliary relay is connected directly to ground. As we mentioned earlier, contact A1 of the auxiliary relay provides a ground for terminal X2 of the a-c external power relay through contact A2. Contact A3, as you can see, is connected to terminal B of the underspeed switch and terminal A of the generator test receptacle.

EXTERNAL POWER ON THE LINE.

When external power has been connected to the airplane, current moves through the power warning circuit breaker, connecting the d-c essential bus to contact A3 of the a-c power failure relay. Contact A2 is connected to terminal R of the master warning relay box. Contacts A2 and A3 are closed in the de-energized position, connecting the d-c essential bus to terminal R. The master warning and the a-c power failure indicators will be lighted, assuming that the master warning system has been energized by the d-c essential bus.

To connect the d-c essential bus in parallel to terminal D of the generator test receptacle and to terminal F of the a-c external power receptacle, the a-c external power circuit breaker must be closed. Because the a-c external power cable connector has a jumper connected between sockets E and F when the connector is plugged into the a-c external power receptacle, the d-c essential bus is connected to terminals X1 of the a-c external power relay and the auxiliary a-c external power relay. Terminal X2 of the auxiliary a-c external power relay is connected directly to ground. The auxiliary a-c external power relay through this network is energized, opening contacts A2 and A3, and closing contacts A1 and A2. Opening contacts A2 and A3 disconnects terminal B of the underspeed switch and terminal A of the generator test receptacle from ground. This prevents inadvertent energizing of the a-c power disconnect relay. The isolation network of this relay was explained when we wired the generator test receptacle into the system.

Closing contacts A1 and A2 of the auxiliary a-c external power relay connects terminal X2 of the a-c external power relay to ground, which energizes the relay, closing its contacts. This connects terminals A, B, and C of the a-c external power receptacle to the a-c buses. Terminal X1 of the a-c power failure relay is connected to phase C of the a-c non-essential bus. Terminal X2, being connected directly to ground, energizes the a-c power failure relay, and opens contacts A2 and A3, disconnecting the d-c essential bus from terminal R of the master warning relay box. This action extinguishes the master warning and the a-c power failure lights, and the a-c power system is under normal ground operation for operational tests and servicing.

A-C ESSENTIAL AND NON-ESSENTIAL BUS SYSTEM.

You have learned that in the a-c electrical power system, the non-essential buses are connected directly to the 30-kva a-c generator through the a-c power disconnect relay when the a-c generator and a-c bus switches are activated. The essential buses are connected in essentially the same manner except that in this case current is routed through the emergency a-c power disconnect relay. In the event of an a-c generator failure, the emergency a-c power disconnect relay is energized and routes 1-kva emergency power to the essential buses. The relay is energized when the a-c bus switch is pushed to the EMER position. In more or less the same manner, the a-c emergency power disconnect relay is de-energized when external power is connected to the airplane and power is channeled to the non-essential buses.

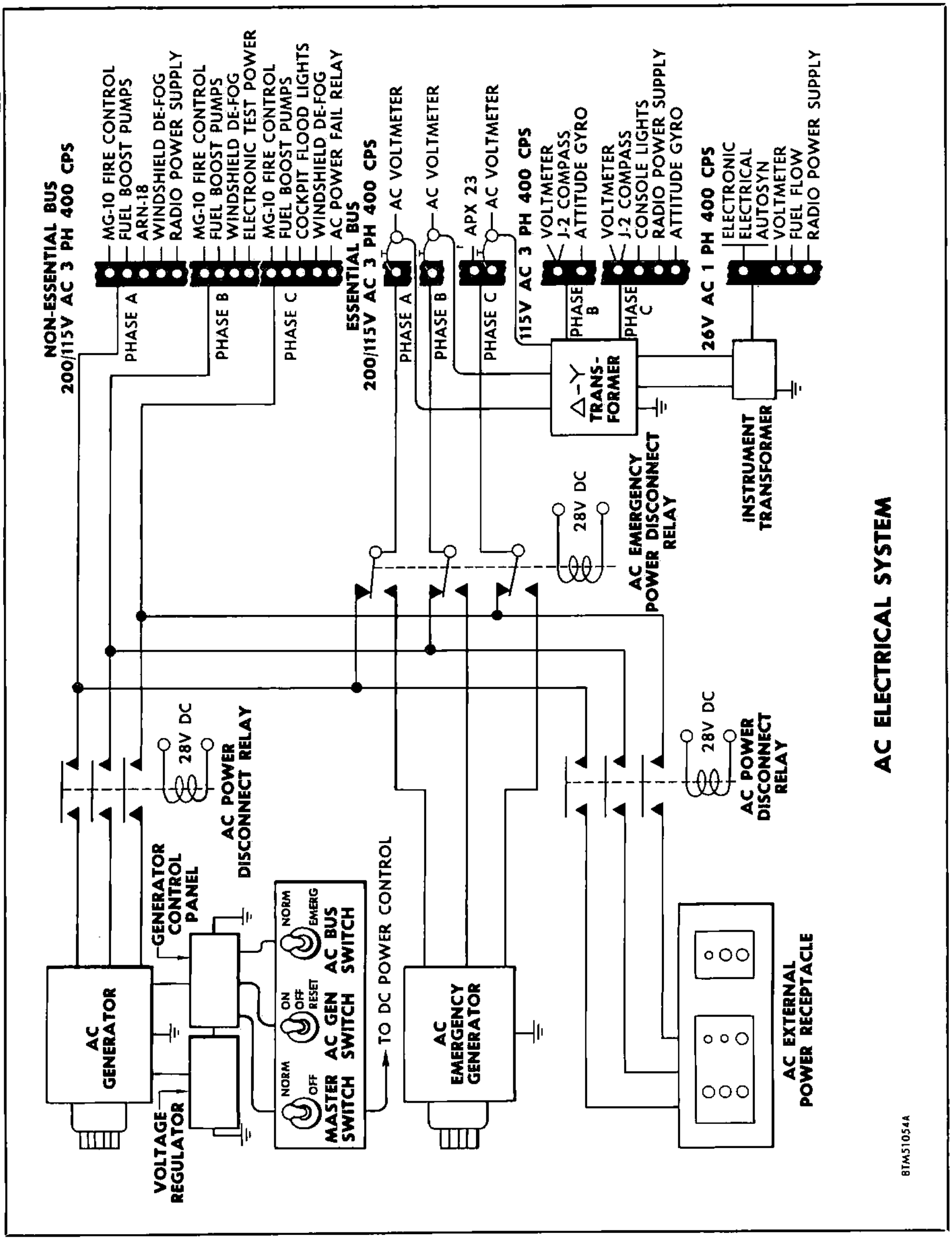

Figure 4-58. The A-C Bus System

There are approximately 64 circuits supplied by both the d-c and a-c bus systems. An analysis of these various circuits and of their components will be found in the applicable system training supplement of this series. Complete schematic and wiring diagrams will be found of all circuits in the maintenance manuals which directly pertain to the F-102A airplane, T. O. 1F-102A-2-10, T. O. 1F-102A-2-12, and T. O. 1F-102A-2-13. These should be consulted at every opportunity. In fact, they are an invaluable part of your equipment.

A-C CIRCUIT BREAKER PANEL LOCATIONS AND CONTENTS.

As you were told in Chapter II when the d-c system was discussed, there are nine individual circuit breaker panels scattered throughout the F-102A. Each of these panels contains a segment of either the essential or non-essential buses. These buses are located in the rear of the various circuit breaker and switch panels. These panels contain both the d-c and a-c buses and the circuit breakers. But for our purpose in this chapter, when we are discussing the a-c system, only the a-c buses and circuit breakers and their location will be listed. The larger of these panels are the main wheel well circuit breaker panel, the nose wheel well switch panel, and the panel in the upper electronics compartment. In the left-hand side of the cockpit is the forward auxiliary circuit breaker panel, located slightly above the cockpit floor at the forward end of the left-hand console and below the cabin altimeter. The left-hand aft circuit breaker panel is located immediately below the cowling angle and above the "G" suit control valve at the rear of this console. Opposite this panel, and on the right-hand side of the cockpit in relatively the same position, is the right-hand aft circuit breaker panel. Just below this panel, on the aft end of the right-hand console, is the right-hand console communication and radar circuit breaker panel. At the forward end of this console is the electrical power switch panel. The utility switch panel is found on the skirt below the pilot's instrument panel.

The following tables locate each circuit breaker panel for you and tell you what it contains. The load capacity of the individual breaker is given, its bus, and the voltage. While there are bus connections in these panels expressly for the d-c electrical system, those located in the following charts are for the a-c electrical system only. The D-C was covered in Chapter II. For illustrations of these panels it is suggested you turn back to Chapter II, from page 72 to page 80.

In the first column, "Circuit Breaker Decal," you will find, in some cases, two different decal designations. The first one quoted is on earlier airplanes, the second on later models.

MAIN WHEEL WELL A-C CIRCUIT BREAKER PANEL.

CIRCUIT BREAKER DECAL	AMPERAGE, VOLTAGE AND BUS
ROLL RATE GYRO ϕC ROLL RATE GYRO AC ϕC	5-amp, 115/200-v AC, essential
ROLL RATE GYRO ϕB ROLL RATE GYRO AC ϕB	5-amp, 115/200-v AC, essential
ROLL RATE GYRO ϕA ROLL RATE GYRO AC ϕA	5-amp, 115/200-v AC, essential
FLIGHT CONT	5-amp, 115-v AC, ϕB, non-essential
FLIGHT CONTROL DAMPER AC ϕB	5-amp, 115/200-v AC, non-essential

NOSE WHEEL WELL A-C CIRCUIT BREAKER PANEL.

CIRCUIT BREAKER DECAL	AMPERAGE, VOLTAGE AND BUS
VOLTMETER AND TRANSFORMER ϕA, ϕB, ϕC	5-amp, 115/200-v AC, essential
FUEL QUANT	5-amp, 115-v AC, essential
115/26V TRANS	5-amp, 115/200-v AC, essential
PRESS RATIO INST	5-amp, 200/115-v AC, essential
ATTITUDE GYRO CONT 115V ϕB	5-amp, 115-v AC, essential
ATTITUDE GYRO CONT 115V ϕC	5-amp, 115-v AC, essential

NOSE WHEEL WELL A-C SWITCH PANEL.

CIRCUIT BREAKER DECAL	AMPERAGE, VOLTAGE AND BUS
J-2 COMPASS & VOLTMETER ϕB	5-amp, 115-v AC, essential
J-2 COMPASS & VOLTMETER ϕC	5-amp, 115-v AC, non-essential
TRIM SERVO AC	5-amp, 115-v AC, non-essential
ELECTRONIC TEST POWER AC	10-amp, 115/200-v AC, 3ϕ, non-essential

UPPER ELECTRONICS COMPARTMENT A-C CIRCUIT BREAKER PANEL.

CIRCUIT BREAKER DECAL	AMPERAGE, VOLTAGE AND BUS
GLIDE PATH ARN-18 115 AC	5-amp, 115/200-v AC, non-essential
MG-3 ϕA 115 AC	35-amp, 115/200-v AC, non-essential
MG-3 ϕB 115 AC	35-amp, 115/200-v AC, non-essential
MG-3 ϕC 115 AC	35-amp, 115/200-v AC, non-essential
AUTO FLT CON SYS AC	5-amp, 115/200-v AC, non-essential

FORWARD AUXILIARY A-C CIRCUIT BREAKER PANEL (COCKPIT).

CIRCUIT BREAKER DECAL	AMPERAGE, VOLTAGE AND BUS
PWR CONT AC	5-amp, 115/200-v AC, essential

LEFT-HAND AFT A-C CIRCUIT BREAKER PANEL (COCKPIT).

CIRCUIT BREAKER DECAL	AMPERAGE, VOLTAGE AND BUS
WINDSHIELD PHB	20-amp, 115/200-v AC, non-essential
WINDSHIELD PHC	10-amp, 115/200-v AC, non-essential

RIGHT-HAND AFT A-C CIRCUIT BREAKER PANEL (COCKPIT).

CIRCUIT BREAKER DECAL	AMPERAGE, VOLTAGE AND BUS
A-C FAIL	5-amp, 115/200-v AC, non-essential
LIGHTING CONSOLE	5-amp, 115-v AC, essential
LIGHTING INST PNL	5-amp, 115-v AC, essential
FLOOD	5-amp, 115/200-v AC, non-essential
POSITION PWR	5-amp, 115/200-v AC, essential

RIGHT-HAND CONSOLE COMMUNICATION AND RADAR A-C CIRCUIT BREAKER PANEL (COCKPIT).

CIRCUIT BREAKER DECAL	AMPERAGE, VOLTAGE AND BUS
GROUND TO AIR 115 AC	5-amp, 115/200-v DC, essential
AIR TO AIR 115 AC	5-amp, 115/200-v AC, non-essential

A-C POWER LOADING.

For the complete charts on power loading of the F-102A a-c electrical system you are referred to the applicable maintenance instruction manual, T. O. 1F-102A-2-10. For our purpose in this section of the manual, it is felt that the accompanying load analysis chart, and what you learned of power loading when studying the d-c electrical power system, will give you a clear understanding of the a-c power loading in the F-102A.

A-C LOAD ANALYSIS.

The chart that follows is the computed loads on both the a-c generator and the a-c emergency generator. The values given are predicted on the systems expected to be in operation under the flight conditions shown on the chart. These values, of course, just as they were in the d-c system, are simply an average of the watts, the vars and the kva used for certain given periods of time as shown.

A-C LOAD ANALYSIS CHART.

STANDBY			
0 - 1/12 min.	6609 watts	1400 vars	6.75 kva
1/12- 2 min.	6559 watts	1380 vars	6.71 kva
2 -30 min.	2669 watts	1104 vars	2.89 kva
TAXI			
0 - 1/12 min.	19544 watts	8843 vars	22.48 kva
1/12- 2 min.	19509 watts	3823 vars	21.1 kva
TAKEOFF AND CLIMB			
0 - 1/12 min.	20543 watts	9226 vars	22.4 kva
1/12- 2 min.	20325 watts	9116 vars	22.22 kva
2 -10 min.	20203 watts	9061 vars	22.18 kva

CRUISE

0 - 1/12 min.	20025 watts	9018 vars	21.95 kva
1/12- 2 min.	20025 watts	9018 vars	21.95 kva
2 -30 min.	20025 watts	9018 vars	21.95 kva

COMBAT

0 - 1/12 min.	22173 watts	10375 vars	24.43 kva
1/12- 2 min.	20637 watts	9635 vars	22.76 kva
2 - 5 min.	20293 watts	9475 vars	22.38 kva

DESCENT AND LANDING

0 - 1/12 min.	18687 watts	8342 vars	20.45 kva
1/12- 2 min.	18687 watts	8342 vars	20.45 kva
2 -10 min.	18647 watts	8263 vars	20.4 kva

EMERGENCY

0 - 1/12 min.	871 watts	316 vars	0.911 kva
1/12- 2 min.	871 watts	316 vars	0.911 kva
2 -30 min.	763 watts	238 vars	0.785 kva

SUMMARY OF D-C AND A-C ELECTRICAL POWER SYSTEMS.

THE D-C POWER SYSTEM.

The d-c power system buses may be energized from three possible sources: the 24-volt, 24-ampere battery; the 30-volt, 200-ampere generator; or an external source through the d-c external power receptacle.

The battery alone will energize only the d-c essential bus, but the generator or external source will energize both the essential and non-essential buses. The battery may be paralleled with the generator or the external source, but the generator is automatically disconnected when the external source is connected. The battery is connected to the bus by operating the MASTER switch to NORMAL and the battery (BATT) switch to ON. With the battery energizing the essential bus, the generator is connected to the bus by operating the MASTER switch to NORMAL and the d-c generator switch (GEN) to RESET momentarily, and then to ON. An external source of power is connected to the bus by connecting it to the main or the auxiliary external power receptacles. An illuminated warning light (D-C POWER FAILURE) on the master warning light panel indicates a de-energized non-essential bus and a corresponding generator failure.

A combination control panel and voltage regulator functions as follows: it provides protection against overvoltage, reverse current and reverse polarity; senses differential voltage (generator versus bus); regulates voltage; connects the generator to, and disconnects the generator from the bus; and provides a means for manual reset, voltmeter connection, and testing of overvoltage protection. The generator is tested by connecting the appropriate jumper plug to the generator test receptacle and connecting the test unit (load bank, etc.) to the external power receptacle.

THE A-C POWER SYSTEM.

The a-c power system buses may be energized from three possible sources: the 30-kva main a-c generator; the 1-kva emergency a-c generator; or an external power source. Connection of any source of power to the bus automatically disconnects the other sources. The main generator and external source energize both the essential and non-essential buses, but the emergency generator energizes only the essential bus. The main generator is connected to the bus by operating the MASTER and A-C BUS switches to NORMAL and the a-c generator (A-C GEN) switch to RESET momentarily, and then to ON. The emergency generator is connected to the bus by operating the A-C BUS switch to EMERGENCY and the a-c generator switch to ON. The external source of power is connected to the bus by connecting it to the external power receptacle. An illuminated warning light indicates a de-energized non-essential bus and a corresponding main generator failure. The voltage regulator and current transformers control the main generator output voltage under varying load conditions. The control panel connects the main generator to and disconnects the main generator from the bus, protects against overvoltage, and provides a means for manual reset and testing of overvoltage protection. The main generator is tested by connecting the appropriate jumper plug to the generator test receptacle and connecting the test unit (load bank, etc.) to the external power receptacle. The Sundstrand underspeed switch allows connection of the main generator to the bus when the generator frequency has increased beyond 370 cps and disconnects the generator from the bus when the frequency drops below 350 cps.

YOU AND THE F-102A ELECTRICAL POWER SYSTEMS.

Pre-flight, post-flight, periodic, and special inspection requirements are perhaps the most important duties you will perform as a flight-line mechanic. The efficient manner in which you prepare them will prevent undue malfunctions and keep the airplane in the air and in a state of readiness for its all-important mission—the protection of the homes and industry of America.

These inspections are found in the Handbook of Inspection Requirements, T.O. 1F-102A-6. The final section of this supplement deals with what is called Replacement

Schedule. Replacement periods specified can be extended by proper authorization. But, for the most part, equipment will be replaced by a newly serviced unit or a completely new item. For example, every 500 flying hours, the a-c regulator, a-c control panel, and a-c 30-kva generator will be replaced. In the case of the a-c emergency generator, 1000 hours is the specified period for removal and replacement.

All this is simply preventive maintenance—you lock the garage door before your automobile is stolen, not after it has been found wrapped around a telegraph pole. The a-c voltage regulator may be in a tip-top performance status until 499th using hour, but on the 501st hour it could go completely hay-wire. So, unless there has been specific authorization to the contrary, any of the items specified for removal should be removed at the Periodic Inspection nearest the time when the replacement is due.

Before closing this portion of the F-102A electrical power systems, from their sources of power to the essential and non-essential bus systems, let's sum up some of the hard cold facts of trouble shooting.

TROUBLE SHOOTING ELECTRICAL SYSTEMS.

When there is an operational failure of any electrical unit or circuit, the airplane is energized and a check will be made of the power lead to the unit to determine if either the unit or the circuit is at fault. To locate the malfunction, you determine first if the electrical power is properly connected to the airplane, and if the circuit breaker protecting that circuit has tripped. A tripped circuit breaker, of course, will be an indication of a short or overload in the circuit or in one of its units. If the circuit breaker is not tripped, it indicates an open circuit.

SHORTED OR OVER-LOADED CIRCUITS.

First of all, find the power lead to the inoperative unit, *by studying the schematic diagram, or the wiring diagram of that circuit.* Wiring diagrams for all circuits in the F-102A will be found in T. O. 1F-102A-2-13.

When you are sure, disconnect the electrical connections at the unit, and energize the circuit with the proper switches. If the circuit breaker trips or a fuse blows, there is a short in the circuitry. If the circuit breaker does not trip, the trouble is in the unit.

If a short appears to be in the circuitry, a general examination of harnesses for damage may locate the most likely area for the short. Inspect for chafing against another lead or structural member of the airplane. If such an area is found, check the continuity of each wire in the harness with a volt-ohmmeter.

If, however, an examination of the harnesses does not reveal any apparent damage, a continuity check of the circuit with a volt-ohmmeter must be made. Follow the wiring diagram of the circuit to determine the proper splice panels and the pin numbers to use. Continuity checks should be made at designated wire breaks only.

When the continuity check reveals the damaged wire, separate the wire from the harness bundle for closer examination. At the same time, other wires in the same bundle should be very definitely checked for continuity. The damaged wire is repaired by splicing or replacement, and the insulation is repaired by wrapping with the specified plastic tape or vinyl. *Never use friction tape.* A splice may be made in the damaged wire only if necessary, and, then only if it does not exceed the specifications outlined by the applicable Air Force technical order, T. O. 1F-102A-2-10.

OPEN CIRCUITS.

To determine an open circuit, the electrical connections at the inoperative unit are disconnected and the circuit is energized.

Disconnect the electrical connections at the inoperative unit and energize the circuit.

Check the voltage between the power lead, where it has been disconnected from the unit, and ground. A no-voltage indication on a voltmeter or test light indicates an open circuit. Inspect the harness involved for damage. If damage is found, remove power and check the continuity of individual wires in the harness with a volt-ohmmeter connected at the terminal ends of each wire. If no apparent damage is found, perform a continuity check on the power lead with a volt-ohmmeter.

When the open lead has been found, it may be necessary to examine the wire closely for breaks under the insulation. Locate breaks of this type by feeling the insulation until a soft spot is found. A break may be repaired by a splice, if such a splice does not exceed specifications outlined in T. O. 1F-102A-2-10.

If the check on the power lead indicates that the trouble is in the unit, check continuity through the unit. Pay particular attention to ground connections.

CONTINUITY CHECKS.

In making continuity checks of circuits with a volt-ohmmeter, the nature of the trouble must first be determined. During continuity checks, pay particular attention to terminal connections. Check to see that cable and line connections are tight and in good condition and that dirt or corrosion is not impairing the circuit.

CHECKING FOR SHORT CIRCUITS.

In checking for a short circuit:

Disconnect all electrical power from the airplane.

Disconnect the terminals at each end of the wire or equipment to isolate it. Do not strip or puncture insulation to make tests.

Connect one lead of a volt-ohmmeter to one end of the wire. Connect the other lead to ground.

Measure the resistance. A reading of zero ohms indicates that wire or equipment is shorted to ground.

Connect volt-ohmmeter between wires in the cable. A reading of zero ohms indicates a short between wires.

CHECKING FOR OPEN CIRCUITS.

In checking for an open circuit:

Energize the circuit.

With a voltmeter or test light, check the voltage between the terminal and ground at each terminal along the power lead.

A no-voltage reading indicates the open circuit is between the terminal and the power source.

An open circuit may also be tested with a volt-ohmmeter as follows:

Disconnect electrical power from the airplane.

Connect the leads of the volt-ohmmeter at adjacent terminals along the wire.

Measure the resistance. A high resistance indicates the open circuit.

BONDING MAINTENANCE.

Maintenance of the bonding to insure good electrical conductivity is of the utmost importance. The following general instructions are adhered to when repairing or replacing bonding:

Bonding jumper replacement must be identical with the original bonding jumpers.

Bonding jumpers must be installed so that vibration, expansion or contraction, or relative movement incidental to normal operation will not break the bonding connections or loosen them to the extent that their electrical conductivity will be affected.

Be particularly cautious when installing jumpers so that the bonding jumpers are installed in such manner that normal operation of any movable component is not hindered. Be careful that bonding braid does not rub against other parts, particularly sharp edges, for damage to the bonding and/or its adjacent component may result.

Bonding clamps on flexible metal hose must be installed so as not to crimp or damage the conduit or hose.

To insure good electrical contact, mating surfaces of bonding jumper terminal lugs and connection points must be cleaned prior to installation.

Check bonding resistance. Resistance should be below 27-milliohms.

As a final warning, and as a final word, never remove a cover of an electrical unit while electrical power is on. You will find that all junction boxes and splice panels which contain a hot lead—a wire that becomes energized whenever electrical power is on the airplane—are placarded with a WARNING sign.

Chapter V

THE LIGHTING SYSTEMS IN THE F-102A

Contents *Page*

The lighting system in an airplane serves two purposes —to illuminate outside the airplane and to illuminate the interior. Lights on the exterior provide lighting for night operations such as landing and formation flying. Interior lighting provides illumination for instruments, equipment, and the cockpit. In addition, certain special lights called indicator lights inside the plane indicate the operational status of equipment, such as the position of the landing gear or the armament bay doors.

In the F-102A the lighting installations are also divided into two groups—exterior lights and interior lights. These installations are supplied from the buses by three classes of electrical power.

Some of the exterior and interior lighting circuits are supplied by 28-volt d-c power from both the 28-volt d-c essential and non-essential buses. From the 115/200-volt, single-phase a-c bus, power is supplied to the position lights and a few of the interior lighting circuits. And from the a-c 115/200-volt non-essential bus, a-c power is routed to the cockpit flood lighting.

THE EXTERIOR LIGHTING SYSTEMS.

The exterior lights consist of the position lights, the landing lights, and taxi lights. The position lights are installed in the airplane to aid in recognition and navigation. The landing lights and taxi lights, of course, are put there to aid the pilot in landing or taxiing the airplane after sundown. The perspective drawing, figure 5-1, shows the location of the position lights as they are installed on the F-102A.

POSITION LIGHTS.

As you can see in figure 5-1, the position light system in the F-102A consists of ten position lights: on the fuselage are the two upper and two lower lights; on the left-hand wing tip is the usual red light, while on the right-hand wing tip you will find the conventional green unit; then there are two additional lights—one white and one amber—on each aft wing root fairing.

These three groups of position lights—on the fuselage, the wing tips, and the tail of the plane—are controlled by two switches located on the utility switch panel. A photograph of this panel is shown on page 80, Chapter II, of this supplement.

How the Position Light Circuit Operates.

The schematic wiring diagram, figure 5-2, should be easy to understand and even easier to trace. As you can see, the position lights receive their power from phase A of the 115-volt essential bus. The circuit is protected by a 5-amp circuit breaker (Position Lts Pwr) located in the right-hand aft circuit breaker panel in the cockpit. Power for the flasher control relay is supplied from the 28-volt d-c essential bus. This circuit is also protected by a 5-amp circuit breaker (Position Lts Cont) located on the same panel.

The intensity of all the position lights is controlled by one of two ON positions—BRIGHT or DIM. The other power controlling switch is a three-position switch whose positions are designated as STEADY-OFF-FLASH.

When the 115-volt a-c power is applied through the two-position BRIGHT-DIM switch to the transformer, the switch selects the desired brilliance from the tapped primary input and determines the intensity of the output. When the switch is positioned to BRIGHT, there is a 28-volt output—when positioned to DIM, a

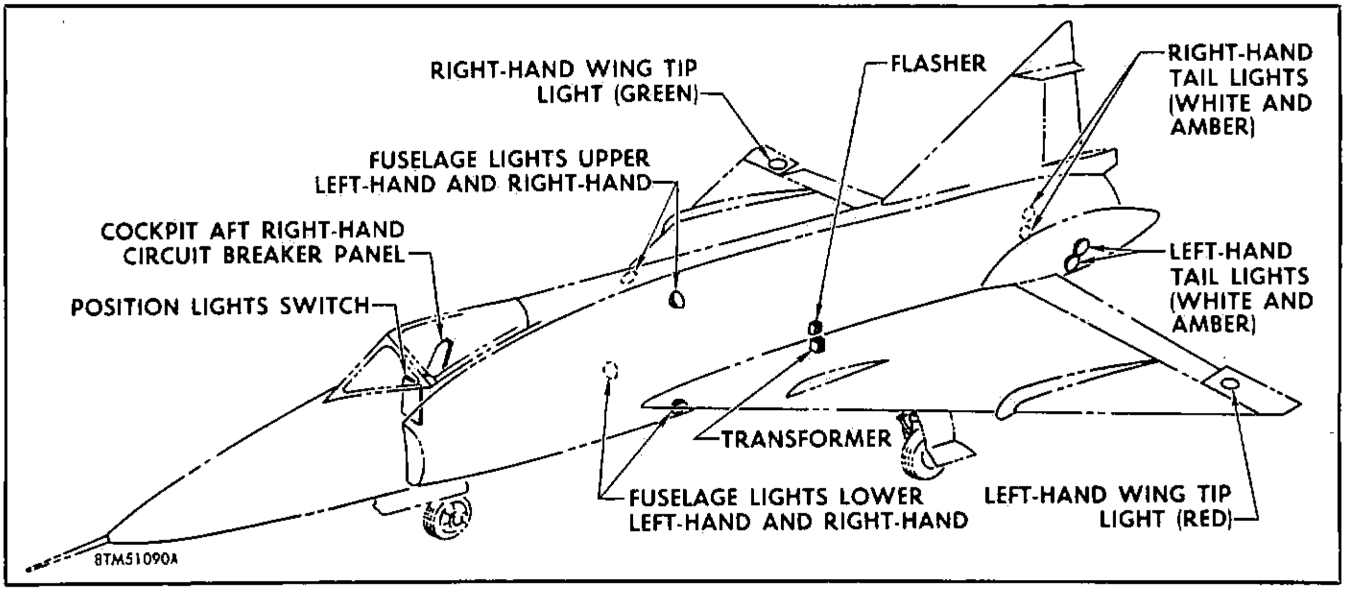

Figure 5-1. The F-102A Position Lights

14-volt output. Very simply, this can be explained as a step-down transformer which works on the same old principle, that when a dpst switch is connected in series with the lamp and since a resistor decreases current flow, the light intensity is reduced. When the series connected switch is on BRIGHT, the resistor is shorted out, and the lamps glow at full brilliance. In this case, the coils of the transformer are in series, and to ground; but the output of the transformer (LV) is connected in parallel to the four fuselage lights—and if you will trace the circuitry on through, connected in parallel to a flasher unit and the wing tip and the tail lights. In this way, while the wing tip and tail lights flash—ON-OFF—the fuselage lights (isolated from the flasher unit) remain steady at all times whenever the position lights are energized.

As we have mentioned, the power for the flashing unit is received from the 28-volt d-c essential bus. The flasher unit energizes and de-energizes the coils of the double throw relay at the rate of 40-cycles per minute ($\pm$15%). It applies power in the energized position to the amber or yellow tail light only through terminal 1. In the de-energized position—the normal position—power is removed from the yellow lights and applied to the wing tip and clear lensed tail lights through terminal 3. The position lights in the F-102A can in this manner be selected to either the STEADY position—where all ten lights in the circuit illuminate steadily, or to the FLASH position, where the four fuselage lights burn steadily and the yellow lights flash on and off alternately with the clear tail lights, and the red and green wing tip lights. In either case, a bright or a dim intensity can be used. Terminal 6 of the flashing unit is connected to ground.

The Flashing Unit.

Formerly, a flasher control mechanism consisted of a small motor-driven camshaft on which two cams were mounted, and a simple switching mechanism of two breaker arms and contact screws. The motor rotated the camshaft through a set of reduction gears, causing the cam to operate the breaker arm from one contact arm to the other. In this way, the position light circuit was flashed from contact to contact. This method of flashing is still found in many flasher systems.

The flasher in the F-102A, however, is a hermetically sealed unit containing two relay switches and a flash control consisting of an electronic network of capacitors. This control component works on the time constant principle. The symbol for this time constant is RC, which may be defined simply as the time required to charge or discharge a capacitor. We discussed this principle earlier in this chapter and in Chapter I, page 27. In this particular flasher control, the capacitors being charged and discharged while wired to the coils (the resistances) of the relays, regulate the opening and closing of the relay contacts. The rate of charge and discharge is 40 cycles per minute. During one complete cycle, each circuit is energized and de-energized once. In other words, during the 20-cycle period, the wing tip lights are energized and de-energized; during the second 20-cycle period, the tail lights are energized and de-energized.

As a final word in this flashing unit, if for any reason the flasher control assembly becomes inoperable, the circuits controlling the wing tip lights and the *clear* tail lights will still be energized. The reason for this is clearly seen in figure 5-2—the two relays are normally in the on-off position, thereby completing these circuits when de-energized.

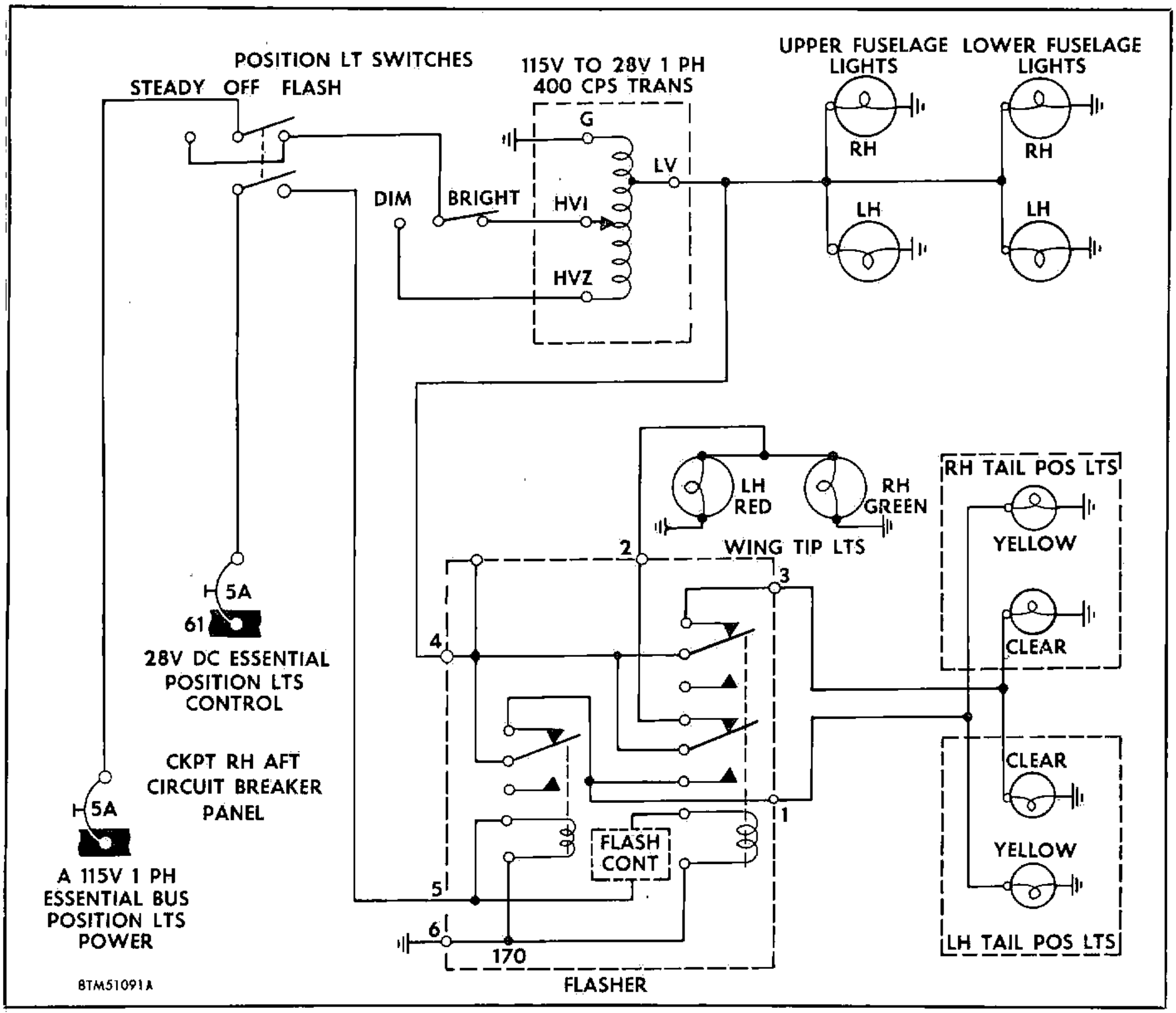

Figure 5-2. The F-102A Position Light Circuit

THE LANDING LIGHT SYSTEM.

Landing lights are installed in aircraft to illuminate runways during night landings. The lights are very powerful and each light is directed by a parabolic reflector at an angle providing a maximum range in illumination.

The landing lights on the F-102A are mounted on the left-hand and right-hand main landing gear struts, as shown in figure 5-3. The light circuits are routed to ground through the nose landing gear down switch. This method of ground routing prevents the completion of the circuit and the illumination of the landing lights until the nose landing gear down switch is actuated by the nose landing gear being extended. When the gear is fully extended, the down lock switch closes the light circuit, allowing illumination when the landing light switch is closed.

In the schematic wiring diagram of the landing light circuit, figure 5-4, note that the circuit is supplied from the 28-volt d-c non-essential bus. The circuit is protected by two 10-amp circuit breakers located in the main wheel well circuit breaker panel. Both circuit breakers are wired to terminal X1 of both the left-and right-hand landing light relays. These relays are located in the forward bulkhead of the main wheel well. When these relays are closed through the action of the nose landing gear down switch, they complete the circuit, and the two 250-watt sealed beam landing lights are illuminated. Except for their power rating, these sealed beam lamps are essentially the same as those used in the headlights of the average automobile.

Briefly then, the d-c non-essential bus is connected to the left- and right-hand landing light relays; and when these relays are energized, to the left- and right-hand

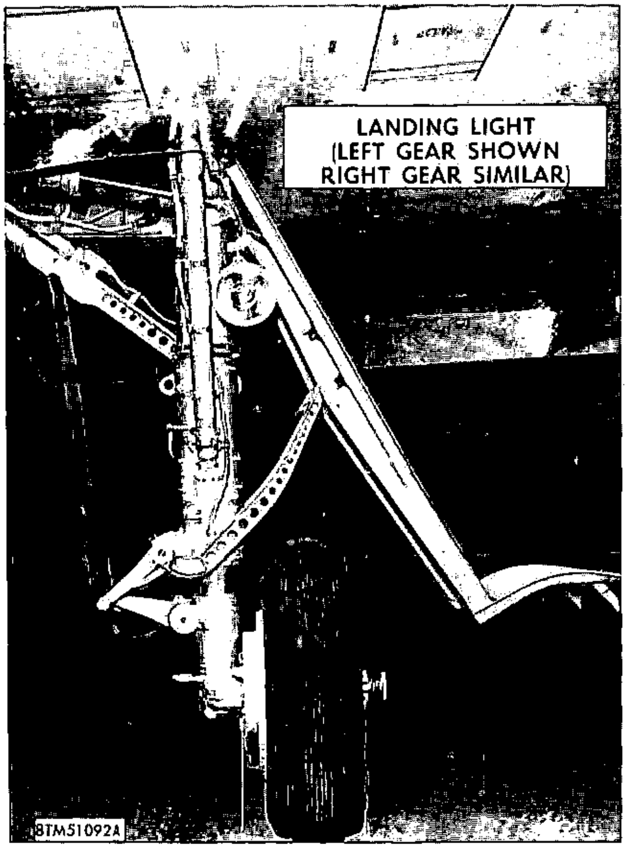

Figure 5-3. F-102A Landing Lights

landing lights. The completed circuit, to energize the relays, is completed through the taxi-landing light switch to terminal E of the nose landing gear down switch and to ground. The taxi-landing light switch is in series with the nose landing gear switch when it is in the down and locked position. Therefore, the nose landing gear must be down and locked before the landing light can be illuminated.

For a further explanation of the landing gear position and warning circuits you are referred to Chapter VII of the Airplane General Supplement of this series.

For the adjustment and operational checks on the lights, you are referred to T.O. 1F-102A-2-10.

THE TAXI LIGHT SYSTEM.

The taxi light system in the F-102A consists of a 150-watt sealed beam lamp and assembly. The light is mounted on a bracket on the inner side of the nose wheel door as shown in figure 5-5. A guard is provided to protect the light from objects picked up by the nose wheel.

The circuit in the wiring taxi light schematic, figure 5-6, is perhaps as simple as any you will find. It contains a simple spst (single break) relay which is energized from the 28-volt d-c non-essential bus through a 10-amp

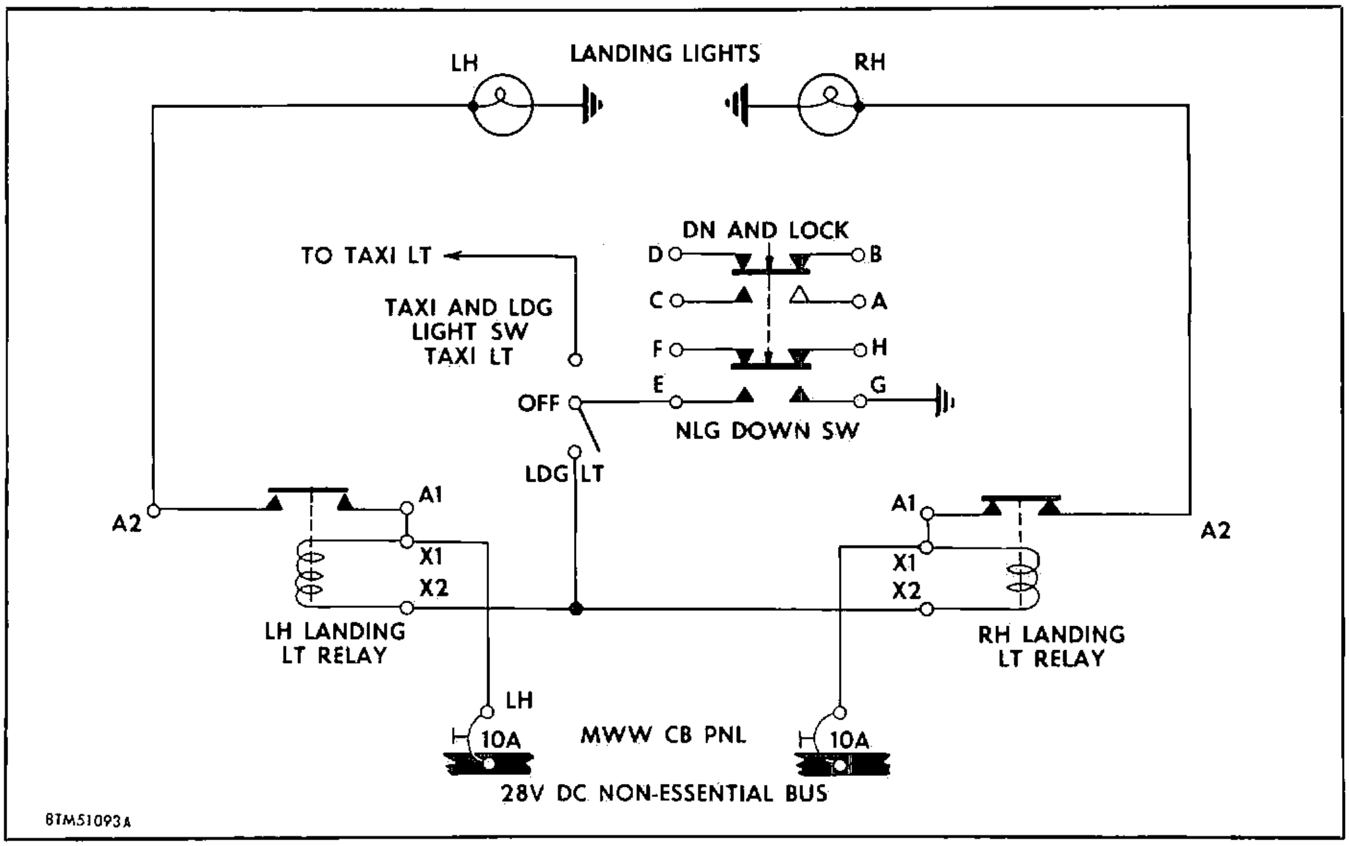

Figure 5-4. F-102A Landing Light Circuit Schematic

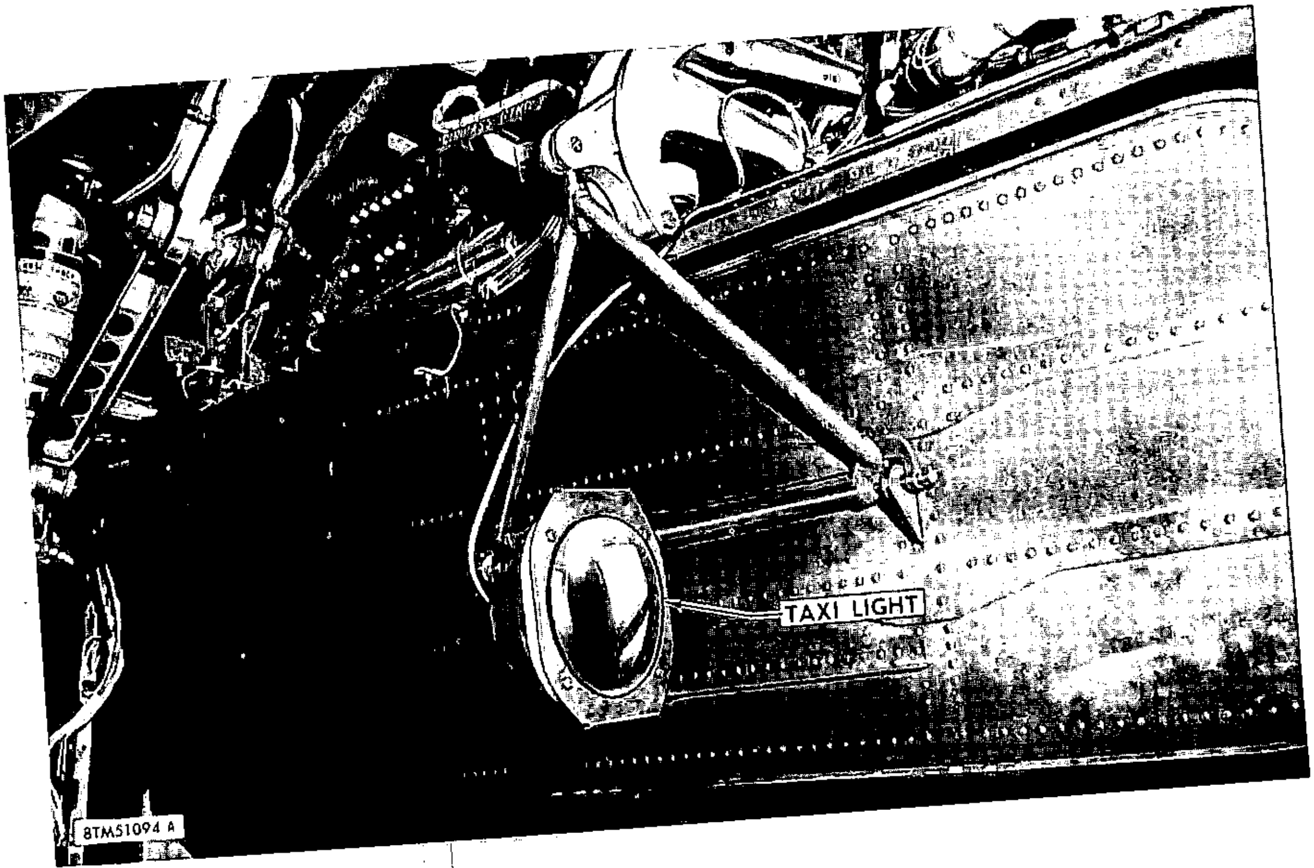

Figure 5-5. F-102A Taxi Light

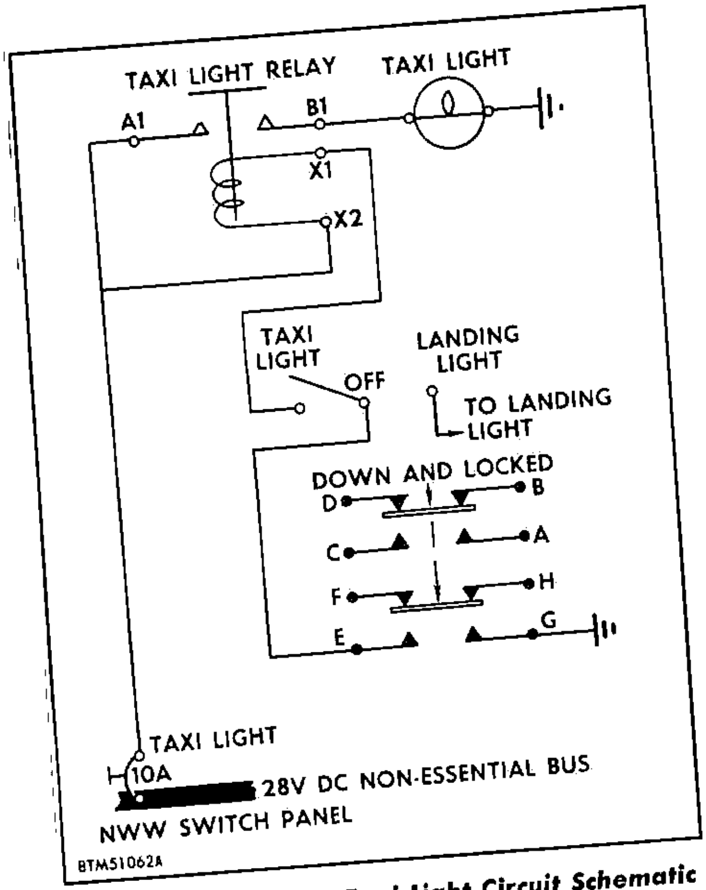

Figure 5-6. F-102A Taxi Light Circuit Schematic

circuit breaker. It is controlled in the same manner in which the landing lights are controlled, remotely by the nose landing gear down switch, and manually by the taxi-landing light switch located in the upper right hand corner of the landing gear control panel shown in figure 5-7.

The taxi light relay and the circuit breaker are located in the nose wheel well. The taxi light assembly is extended and retracted by extending and retracting the landing gear. The circuit to energize the taxi light relay is completed to ground through the taxi light switch, when closed; and terminal E and the contacts of the landing gear down switch when in the down and locked position. So much for the external lighting systems of the F-102A.

INTERIOR LIGHTING SYSTEMS OF THE F-102A.

The interior lighting of the F-102A consists of the instrument panel lights, the pilot's console lights, thunderstorm lights, the flight instrument lights, and the cockpit floodlights. The lighting is divided into two categories—edge lighting and floodlighting. Edge lighting is a type of indirect or a diffused red-edge method of lighting an individual unit through a light transmitting plastic panel. Floodlighting is simply the flooding of the particular area with light. The interior

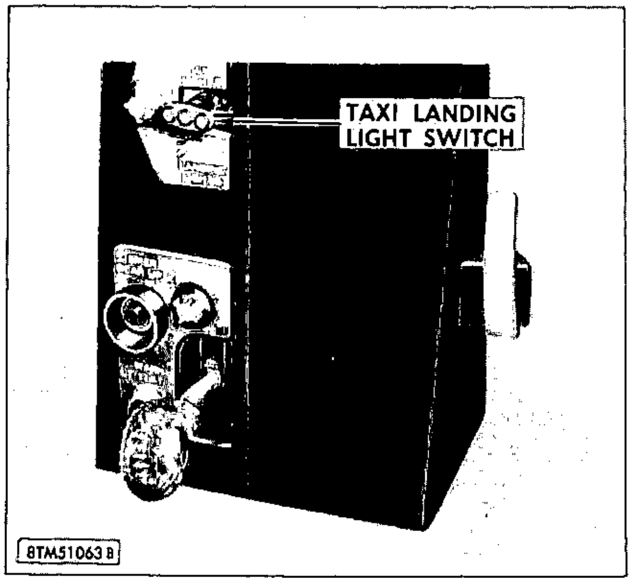

Figure 5-7. Landing Gear Control Panel

lighting is controlled by five powerstats installed on the light control panel. These powerstats are nothing more than variable transformers. This panel is located just below the utility switch panel on the skirt of the instrument panel. The photograph of this panel is shown in figure 5-9. The cockpit lighting arrangement is shown in figure 5-8. Let's discuss instrument lighting first.

THE INSTRUMENT PANEL LIGHTING SYSTEM. Lamps set in the instrument panel, which is covered usually by a plastic reflector panel, provide one method of indirect lighting for instruments. The light from these lamps is distributed over the entire panel. The reflector has openings in it for observing the various instruments.

Another form of instrument lighting uses fluorescent lighting to eliminate the glare so common in other types of lighting. The lamp assembly has a special lens, which will pass only ultraviolet light, and a screen to regulate

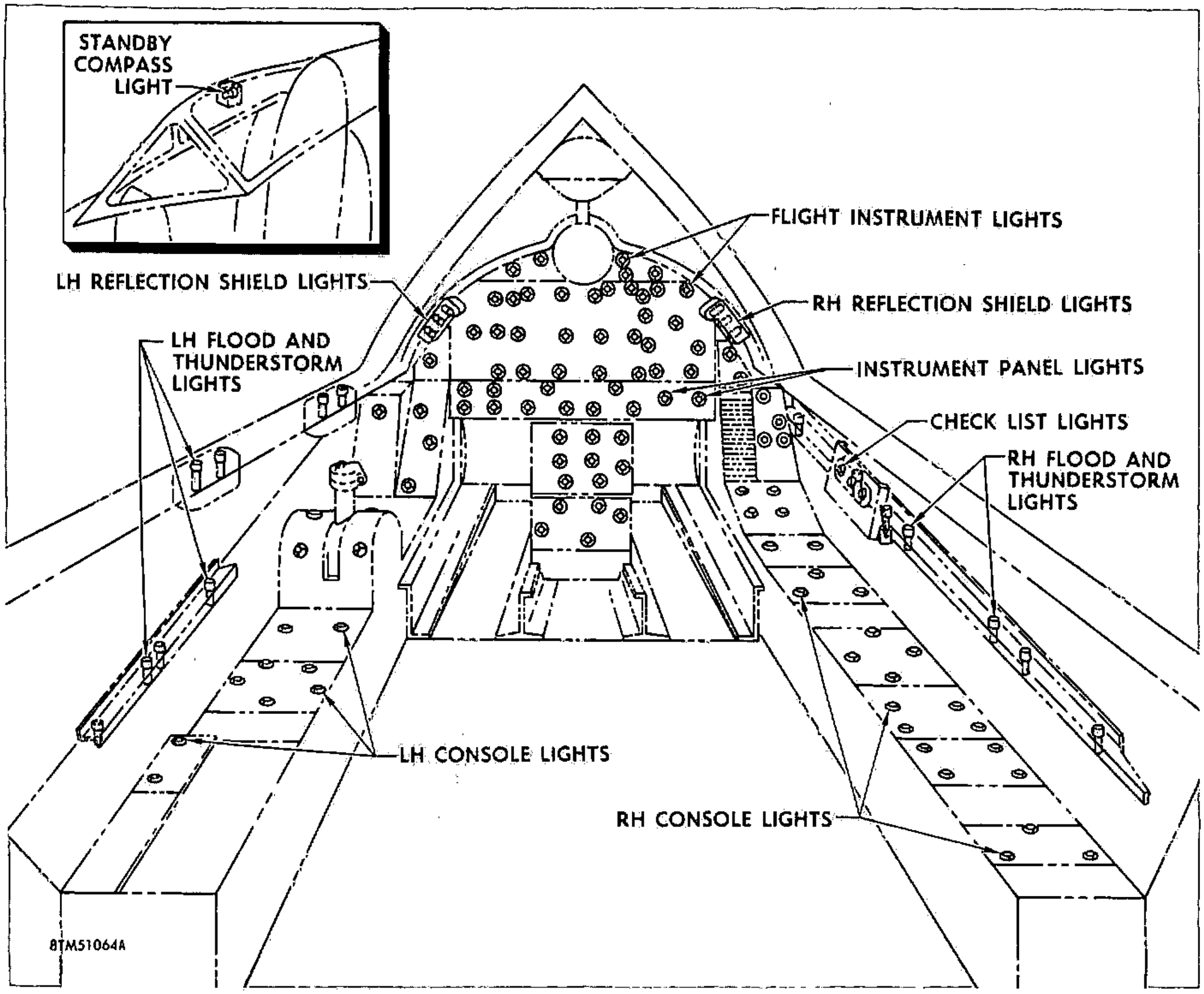

Figure 5-8. F-102A Cockpit Lighting

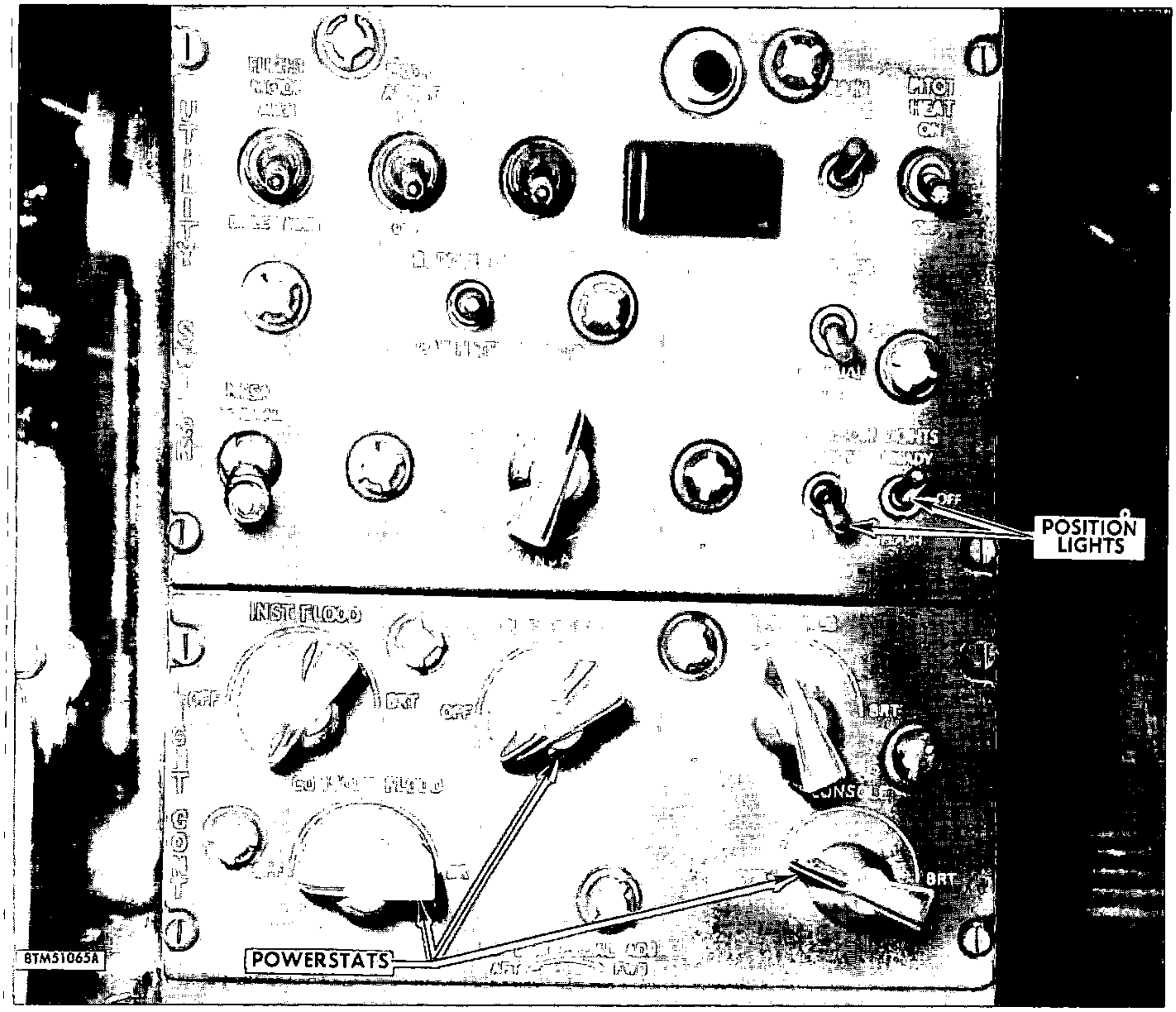

Figure 5-9. F-102A Light Control Panel

the amount of light. Another screen passes the visible light. Instruments used with this type of lighting have the dial figures painted with a material which is sensitive to ultraviolet light. When the invisible ultraviolet light is directed on the instruments, the figures are outlined in a soft glow and become very distinct even though there is no visible light coming from the lamp.

In the F-102A, the instrument panels are made of a translucent plastic. The instrument panel lighting assemblies are imbedded in the panel. The faces of the panels are coated except where illumination is desired. The instrument panel cutouts are chamfered, grooved, or beveled. This permits the transmission of the red-edge light from the edges of the panel cutouts to illuminate the instruments.

As you probably know, an instrument panel is divided into sectional panels, each panel having its individual name. In the instrument panel lighting system, each of these panels has its own lamp assemblies. For example, the lighting control panel has five lamps, the cabin heat control panel, five, the pilot's left-hand instrument panel five; the antenna scan panel, the radar control panel, and the HYD and OAT indicator panel three each; the left-hand and right-hand MG3 control panels, the target attitude panel two each; and the pilot's instrument panel nine—forty-two lights in all.

The powerstat variable voltage step-down transformer is mounted on the light control panel, and controls the intensity of these lights. A 5-amp circuit breaker protects the circuit to the lights from ØB of the 115-volt a-c essential bus. This instrument light circuit breaker is located on the right-hand aft cockpit circuit breaker panel.

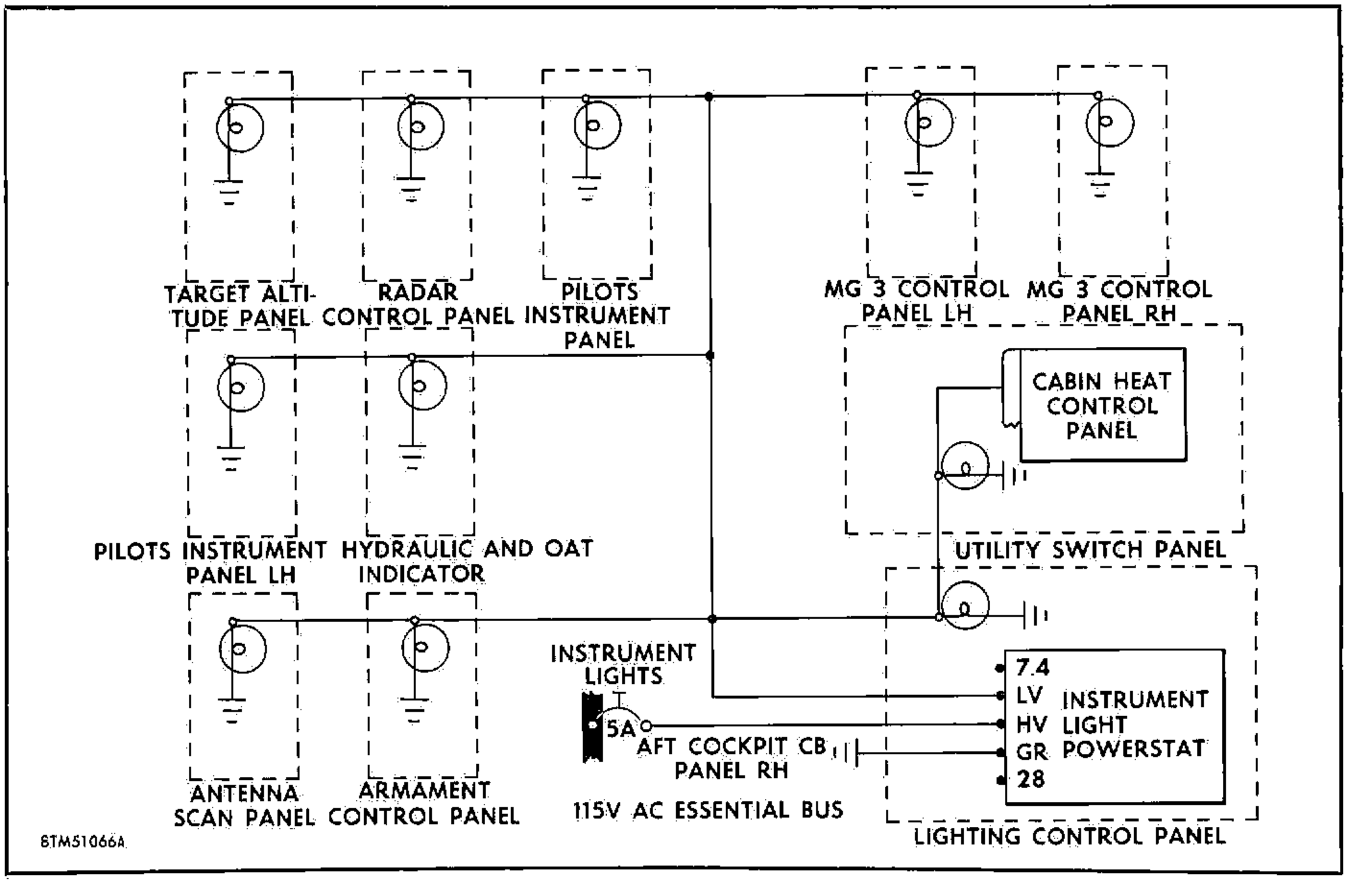

Figure 5-10. Instrument Panel Lighting Circuit Schematic

As you can see in the simplified schematic wiring diagram, figure 5-10, the panel lights are connected in parallel to the low voltage (LV) output terminal of the powerstat. The high voltage (HV) input is obtained from the a-c essential bus. The powerstat is grounded and has minimum output of 7.4 volts to a maximum of 28 volts. When the powerstat control knob is rotated clockwise from the OFF position, the panels are illuminated with increasing intensity until the BRIGHT position is reached. The maximum output of the powerstat at this point is 28 volts.

THE CONSOLE LIGHTS.

The left-and-right-hand consoles are illuminated in the same manner as the instrument panel. Power is taken from the 115-volt a-c essential bus and controlled by the console lights powerstat. This powerstat has the same step-down feature as the instrument panel powerstat and has the same 0-to 28-volt variable control.

There are 25 lights in the console lighting system. The schematic wiring diagram of this circuit is shown in figure 5-11. The console panels frame it. This system is protected by the console light 5-amp circuit breaker, situated in the right-hand aft circuit breaker panel.

Each individual circuit in the system is connected in parallel to the low voltage output of the powerstat. The high voltage input received from the a-c essential bus is regulated by rotating the knob of the powerstat clockwise from OFF to BRIGHT. Here again, as in the case of the instrument panel lights, the intensity is increased until the bright position is reached. The maximum output at this point is 28 volts. The panels are red-edge lighted in the same indirect manner as those in the instrument lighting system.

THUNDERSTORM LIGHTS.

The thunderstorm lighting system consists of five lights and light assemblies, a thunderstorm light switch, and a thunderstorm light circuit breaker. The wiring schematic of this system is shown in figure 5-12.

As the name implies, these lights are provided for use during thunderstorms to reduce the blinding effect produced by lightning. The lights are mounted one on the left-and right-hand reflection shields, one just above the left-hand console, and two above the right-hand console. The thunderstorm light switch is a two-way ON-OFF toggle switch located on the right-hand auxiliary panel. It is labeled STORM LIGHTS. The thunderstorm 5-amp STORM circuit breaker, which protects the circuit, is located on the cockpit right-hand aft circuit breaker panel.

The thunderstorm lighting circuit receives its power from the 28-volt d-c non-essential bus. The circuit is

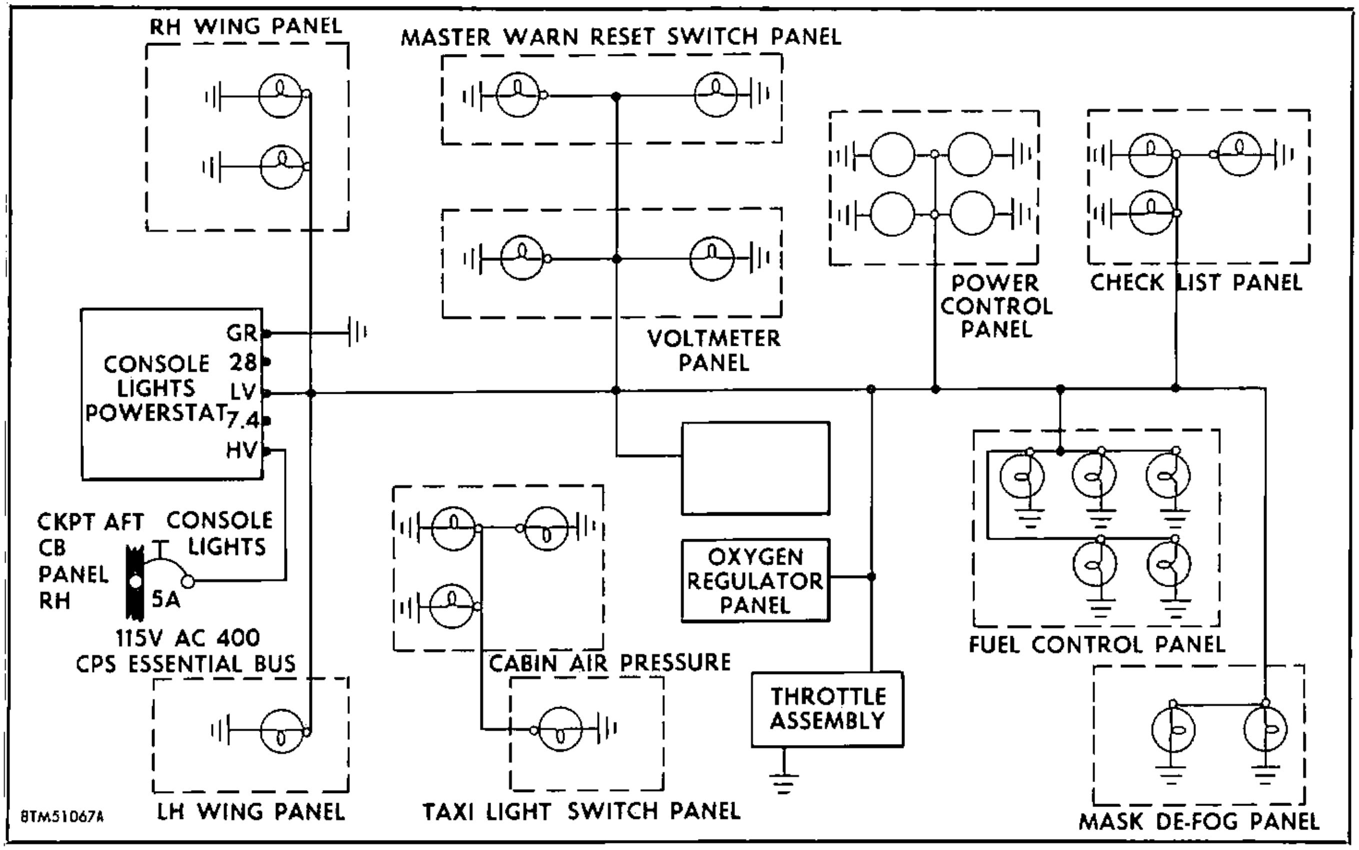

Figure 5-11. Console Panel Lighting Circuit Schematic

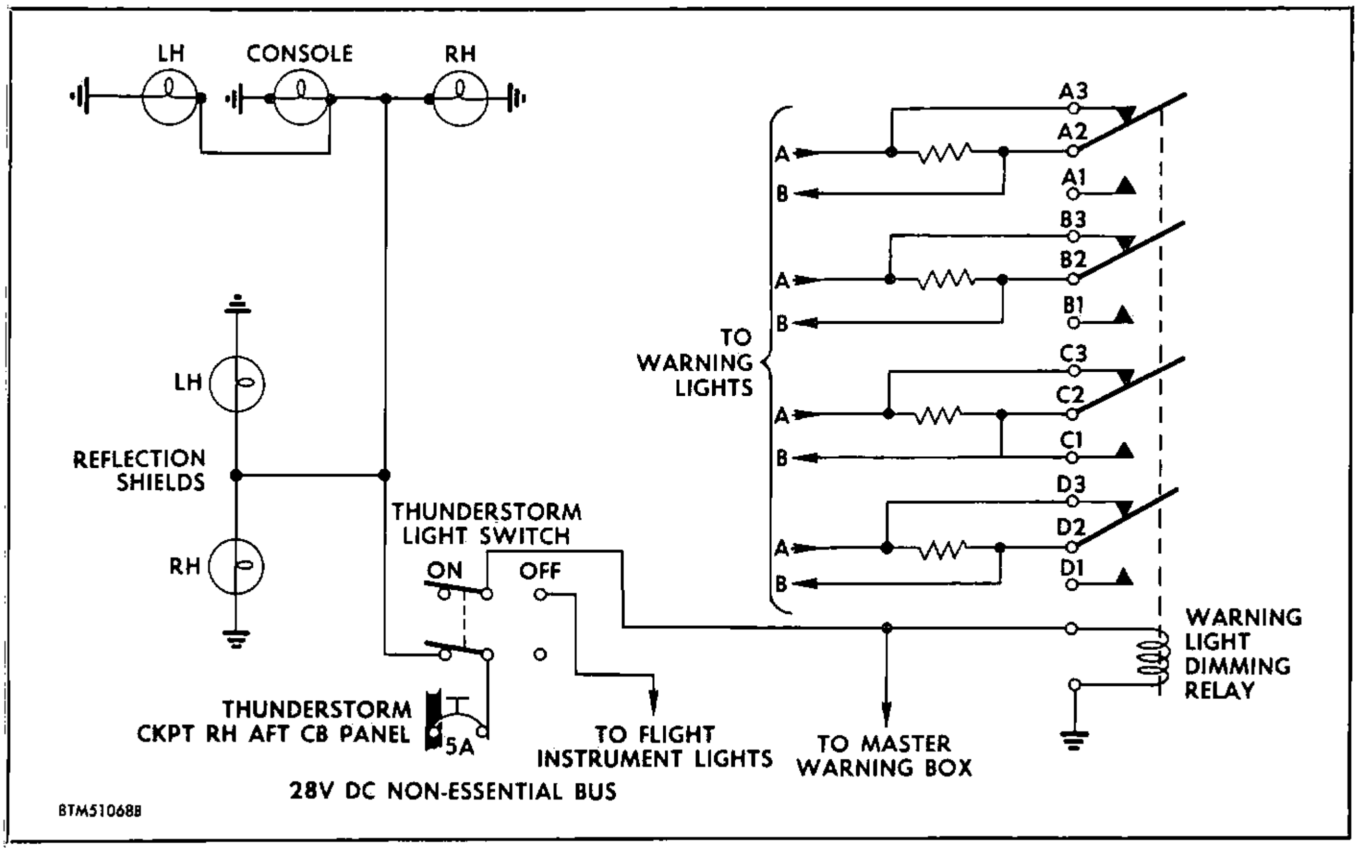

Figure 5-12. Thunderstorm Light Circuit Schematic

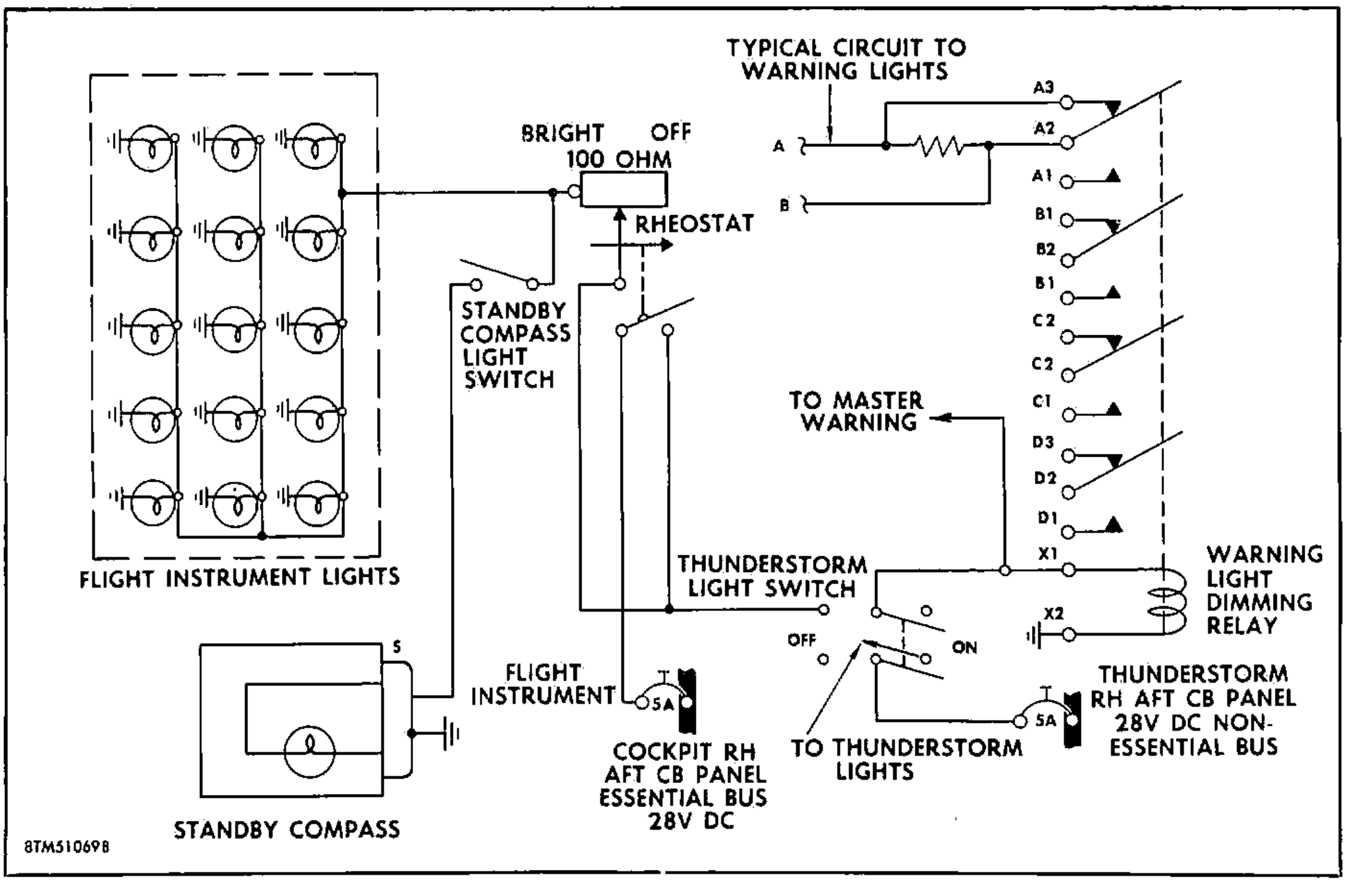

Figure 5-13. Flight Instrument Lighting Circuit Schematic

connected into the master warning light circuit to automatically switch the warning lights to bright, provided the thunderstorm light switch is ON while the flight instrument lights are on.

In the circuit schematic, figure 5-12, note that when the thunderstorm light switch is placed in the ON position, the circuit from the flight instrument lights to the warning light dimming relay is *opened*. With the circuit open, power is removed from the relay, thus de-energizing it. Now note that current to the various warning light circuits which are not controlled by the master warning box enters at point (A), flows through the top contacts, bypasses the resistors, and then flows out through point (B). Since the resistor in each circuit resists current flow, the current will always take the easiest path provided. With the resistors out of the warning light circuits, their lights will illuminate brightly when the thunderstorm lights are ON. The circuit to the master warning box connects to the solenoids in the three dimming relays in that box and provides the same brightness condition for the warning lights controlled by the master warning box as described above.

When the thunderstorm light switch is moved to the OFF position, the circuit from the flight instrument lights is completed through this switch to the dimming relays. This completed circuit energizes the warning light dimming relays when the flight instrument light rheostat is ON. Now note that when the relays are energized, power entering at point (A) must pass through the resistors to flow out point (B). This causes the warning lights which are independent of the master warning box and those lights controlled by the box to illuminate dimly.

FLIGHT INSTRUMENT LIGHTING SYSTEM.

As you can see in the schematic drawing, figure 5-13, the flight instrument lighting system contains 15 lights, a 100-ohm rheostat to control it, and a 5-amp flight instrument circuit breaker to protect it. Another portion of this circuit contains the standby compass, which is lighted internally, and a standby compass light switch. The output from the controlling rheostat is controlled indirectly by the warning light dimming relay through the thunderstorm light switch. The lights and their assemblies are imbedded in the pilot's instrument panel, giving red-edged lighting to the flight instruments. The rheostat is located on the light switch control panel. The flight instrument circuit breaker is in the right-hand cockpit aft circuit breaker panel.

The magnetic standby compass is located in the peak of the pilot's canopy. It is used to give a magnetic heading to the airplane in case the radio magnetic course indicator becomes inoperative. A complete explanation of this compass—why it is used, how it is used, and when it is used—is given in the supplement dealing with instruments of this series. The light is internal, and the standby compass light switch is located above the right-hand console next to the check list lights.

Although the standby compass light receives its power through the flight instrument light circuit, it has its own light switch. The standby compass light can be turned on when the flight instrument lights are on, but cannot be turned on when they are off.

The 100-ohm internal on-off switch is opened by rotating the rheostat knob counterclockwise to the OFF position. The switch is closed by rotating the rheostat clockwise from the OFF position.

The flight instrument lights are connected in parallel to the d-c essential bus. The internal on-off switch in the rheostat controls the brightness of the warning lights when the thunderstorm switch is OFF. When the thunderstorm lights are ON, the master warning lights are turned on bright by the thunderstorm light circuit, regardless of the position of the flight instrument light rheostat. And when the thunderstorm light switch is off, the slightest clockwise movement of the rheostat will dim the warning lights. But any additional movement clockwise will increase the brightness of the flight instrument lights. And when the rheostat is turned to full counterclockwise position, the flight instrument lights will be off, and the warning lights will be on bright.

COCKPIT FLOODLIGHT SYSTEM.

On the light control switch panel in figure 5-9, note the two powerstat variable voltage transformers labeled INST FLOOD and CONSOLE FLOOD. These powerstats control the intensity of the cockpit floodlight system. The system is divided into two parallel circuits, one to supplement the instrument panel edge lights, and the other to supplement the edge lights on the consoles.

The console floodlight system uses five lights above the right-hand console and seven above the left-hand console. They are mounted on the under sides of the right- and left-hand cockpit longerons. They are miniature 4-amp, 28-volt lamps.

The instrument floodlight system uses four lights: two on the underside of the left-hand instrument panel glare shield, and two on the underside of the right-hand glare shield.

Both systems receive power, through a single 5-amp circuit breaker, from the 115/200-volt a-c non-essential bus. The circuit breaker is located with the rest of the lighting system circuit breakers on the right-hand aft circuit breaker panel. See figure 5-14.

In both systems, the lights are connected in parallel with the low voltage terminal of the controlling powerstats. Rotating the powerstats clockwise from the OFF position illuminates the lights with increasing intensity until the BRIGHT position is reached. At this point the maximum output of the powerstats is reached—28 volts.

This brings to a close the lighting systems in the F-102A with the exception of a word regarding the maintenance and inspection of the systems. In the interior systems, just covered, maintenance of these lights normally consists of checking the systems for burned-out lights and replacing the bulbs as necessary. A spare bulb stowage area is provided in a container aft of the G-suit valve on the left-hand console. This container provides bulbs for use by the pilot during flight and should be kept filled.

MAINTENANCE AND INSPECTION OF LIGHTING SYSTEMS.

When inspecting the airplane lighting system, as in any electrical system, check the condition and the security of all visible wiring, connections, terminals, and switches; see that all switches and lamps are operating properly. You can locate the cause of many troubles by systematically testing each lamp circuit for continuity. Use a continuity checker for making these checks.

Keep the lamp lenses and reflectors clean and highly polished. When you find a cloudy reflector, remove the lacquer surface with acetone, polish the reflector surface with a mixture of lamp black and alcohol, then relacquer the reflector surface. If this does not remove the cloudiness, remove the reflector and turn it in for replating. Cloudy reflectors are usually caused by an air leak around the lens. Therefore, be sure to install a new gasket when you reassemble the lamp assembly. Likewise, pay particular attention to getting the lamps into proper focus and alignment.

Inspect the condition of the sealing compound around the navigation and formation light frames. If you find leaks or cracks, fill them with new compound.

If it becomes necessary to install a new bulb in a lamp assembly, you must be especially careful that the bulb is fitted into the socket and in the correct position. And—do not use force!

SUMMARY.

This training supplement has been written with multiple objectives in mind. Basically, it is written to give you a fuller understanding of the fundamental concepts of electricity and to supplement the technical orders issued and already in the field which deal with the F-102A interceptor. There is no substitute for Technical Orders—they give you the hard cold facts of procedures and

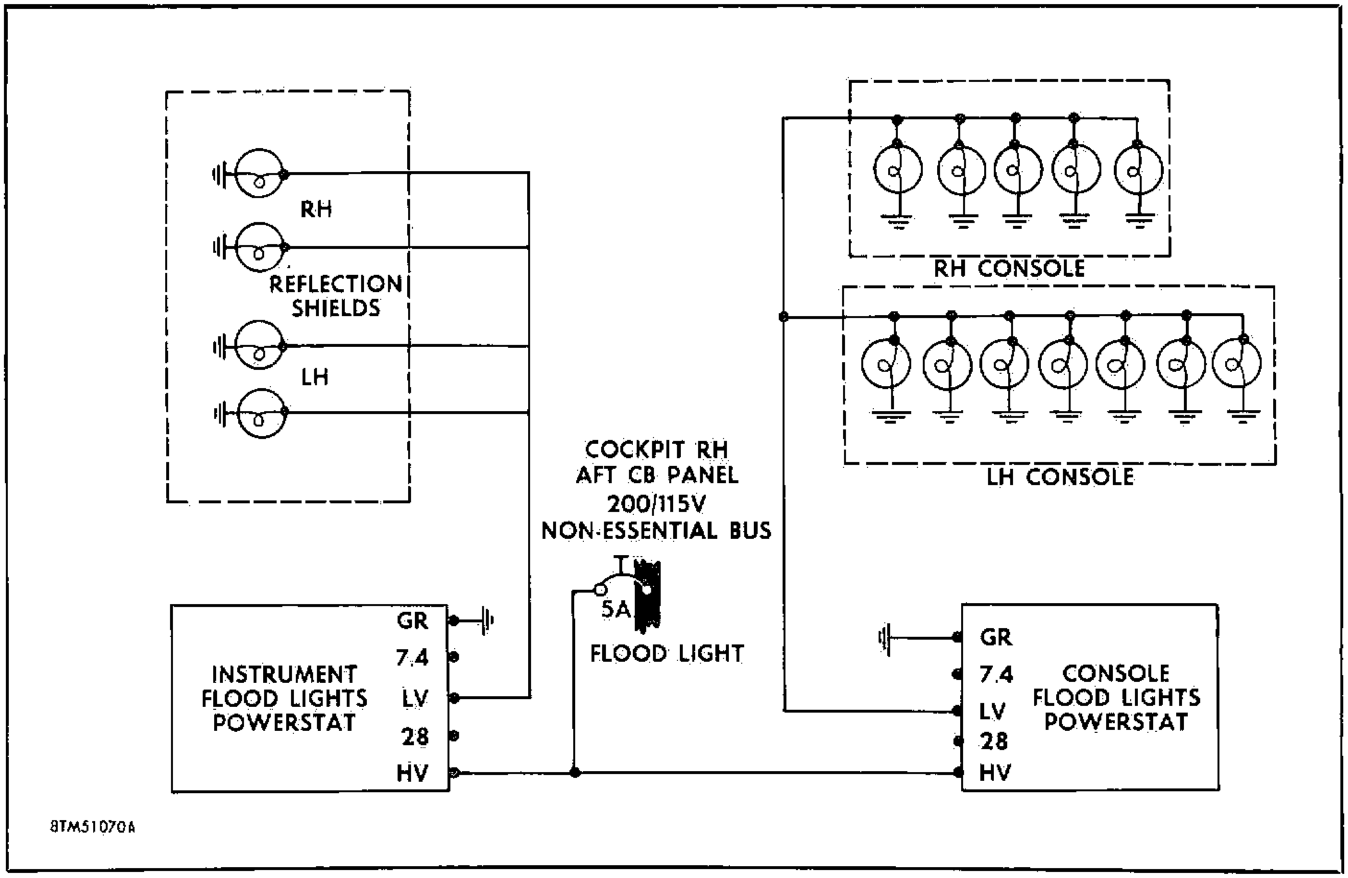

Figure 5-14. Cockpit Floodlight Circuit Schematic

maintenance musts. They should be studied, consulted, and used as prescribed. They are the official technical orders of your Air Force.

The training supplement is presented as an important addition to your other training—as a ready reference, a continuous refresher course in electricity, and a daily reminder of sound maintenance practices.

Chapter I is devoted to a general discussion of fundamentals—a more-or-less knitting together of the basic concepts of electrical power as it relates to the F-102A, or to any other airplane, for that matter. Chapter II discusses power distribution in general, the F-102A d-c generator, the controlling and protective devices in the F-102A d-c electrical power distribution system, and the d-c essential and non-essential bus system.

Chapter III deals historically with d-c power systems, and with those induced and natural environments encountered by any electrical component or any other electrical system. A good deal of space is devoted to wiring, to the potted plug, and finally to the d-c electrical power systems as installed in the F-102A. Chapter IV explains the a-c electrical power system and its peculiarities. Chapter IV is a combination actually of Chapters II and III, but from the alternating-current standpoint. Those portions of Chapters II and III, such as protective and controlling devices found in most electrical systems, environment, the potted plug, trouble shooting, and other subjects of a general nature are presented where the content seems to merit their discussion. Chapter V deals exclusively with the lighting systems as found in the F-102A.

Summarily, every single word and phrase in the five chapters comprising this training supplement pertains to the F-102A Electrical Systems in one way or another. If, on occasion, the writer resorts to levity to put a point across, he is not laughing when he does so; if you laugh on occasion, so much the better. Maintenance, however, on a complex, modern, high-altitude, all-weather interceptor is a deadly serious business and should be treated as such. It is hoped that this supplement helps to emphasize just how serious your job has become and will continue to become in the future. As aircraft fly higher, faster, and farther, the job at hand will demand even more exacting knowledge. It is further hoped that a thorough knowledge of what is contained in this supplement will help you to achieve that all-important goal—efficiency and confidence as a maintenance man, and, above all, reliability. And, if it whets your curiosity to know more, so much the better.

As a final word, throughout this supplement reference is made to the fact that all airplanes of the F-102A series do not have the same type components. Where this is true, we point out these exceptions to the best of our ability at this time. But the basic concepts are all there and it should not be difficult to grasp new modifications as they develop and are passed on to you. The newest and latest development information should be obtained from your F-102A series technical orders. Always keep this in mind: this training supplement does not replace the technical order brought to your attention by your commanding officer and issued to you directly—it simply points up and correlates the knowledge you have already accumulated and helps to direct your on-the-job training in maintenance techniques of this new airplane. More than once, from the very first page of this supplement, you are told that you can achieve the fine points of maintenance techniques only by actual experience—by the "doing." If you have had previous experience, don't let familiarity lead you up that dead-end street with a false sense of "know-how." Be sure. Be exacting. And, as we said when discussing cold weather operations, you may feel like a brass monkey out there in that bitter cold—but don't *think* like one.

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ALPHABETICAL INDEX

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F-102A

MAINTENANCE TRAINING SUPPLEMENT

(EXPERIMENTAL)

To Be Used in Conjunction With

T.O. 1F-102A-2-4

POWER PLANT INSTALLATION

CONVAIR
F-102A

MAINTENANCE TRAINING
SUPPLEMENT

POWER PLANT
INSTALLATION

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE

AIR FORCE—AFPS—SAN ANTONIO, TEXAS—5-57-2600

1 JULY 1956

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Foreword

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The F-102A Training Supplements have been prepared by Convair, A Division of General Dynamics Corporation, to provide you, the system mechanic or maintenance technician, with basic information concerning the F-102A airplane and its operating systems. There are ten of these supplements, each covering an airplane operating system or major component. The text is direct and informal and is supplemented with illustrations and diagrams to aid you in understanding how the systems operate. The following is a list of the Training Supplements on the F-102A airplane:

Title

Flight Control System
Hydraulic System
High-Pressure Pneumatic System
Low-Pressure Pneumatic System
Airplane General
Airframe Fuel System
Power Plant Installation
Airframe Armament System
Electrical System
Instruments

Each supplement describes an operating system or major component, how and why it operates, and maintenance problems that you may encounter. The entire major system is further simplified by describing each of its subsystems separately. Thus, when you understand how all of the subsystems operate, you will be familiar with the functions of the major system. This knowledge of the airplane systems will enable you to use other operational, service, and maintenance instructions.

Since the purpose of these publications is to familiarize and acquaint you with the airplane systems and components, specific values, measurements, and tolerances are not given. Any values that appear in these supplements are approximate only and are used to emphasize more clearly a certain operation or condition. You must refer to your 1F-102A-2-4. Technical Order and other pertinent handbooks for specific values when adjusting or checking a system and its components. These Technical Orders are revised periodically to include the latest maintenance procedures and data on the equipment installed in the F-102A.

The F-102A Maintenance Technical Orders consist of the following handbooks:

T.O. 1F-102A-2-1	**General Airplane**
T.O. 1F-102A-2-2	**Ground Handling, Servicing, and Airframe Group Maintenance**
T.O. 1F-102A-2-3	**Hydraulic and Pneumatic Power Systems**
T.O. 1F-102A-2-4	**Power Plant**
T.O. 1F-102A-2-5	**Fuel Supply System**
T.O. 1F-102A-2-6	**Air Conditioning, Pressurization, and Anti-Icing Systems**
T.O. 1F-102A-2-7	**Flight Control Systems**
T.O. 1F-102A-2-8	**Landing Gear**
T.O. 1F-102A-2-9	**Instruments**
T.O. 1F-102A-2-10	**Electrical Systems**
T.O. 1F-102A-2-11	**Radio-Communication and Navigation Systems**
T.O. 1F-102A-2-12	**Armament and Armament Electronics**
T.O. 1F-102A-2-13	**Wiring Data (F-102A)**
T.O. 1F-102A-2-13A	**Wiring Data (TF-102A)**

The Air Force is vitally interested in seeing that its equipment functions properly and is maintained as efficiently as possible. The Air Force, like a manufacturer of an automobile or commercial appliance, provides printed instructions for use with its equipment. The printed instructions you will use most frequently in maintaining the F-102A are the dash-2 Technical Orders listed above. They are available for your reference at the Maintenance Office on your base. You must refer to them frequently so that you will always have the latest maintenance information. The Training Supplements will help you to use these Technical Orders by answering many of the "hows" and "whys" that may puzzle you from time to time.

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Chapter I

J57 POWER PLANT

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Many of us, watching a present day high speed jet aircraft streak across the sky, find it hard to appreciate the tremendous strides taken in the past 50-odd years of development. Improvements in military aircraft ability to fly faster and higher and carry more armament have demanded improvements in powerplant engineering and fuel/oil chemistry to the point that the turbo-jet engine has become a precision instrument. It definitely was not always this way.

Inspection and maintenance procedures have been radically changed and have become more and more complicated. Therefore, the knowledge of the mechanic must be expanded. To properly inspect and maintain the high speed aircraft of today with all their advanced systems, YOU HAVE TO KNOW WHAT YOU ARE DOING. When you view the F-102A airplane, it is certain to impress you as an intricate and highly advanced mechanism. The airplane loses much of its complexity, though, when you break it down into parts and study the operation of each part.

The purpose of this training supplement is to acquaint you with the J57 engine as installed in the F-102A airplane. Much time has been spent in anticipating your questions about the engine and its related systems. Every attempt has been made to make this an interesting training supplement which can be used, not only for familiarization by the student mechanic, but also as a reference guide for the top mechanic in your squadron. The amount of information you digest from this supplement will determine your effectiveness as a member of the ground support team of the F-102A airplane.

DEVELOPMENT OF THE GAS TURBINE.

Many of us may think of the theory of the gas turbine as being relatively new. Actually, the gas turbine prin-

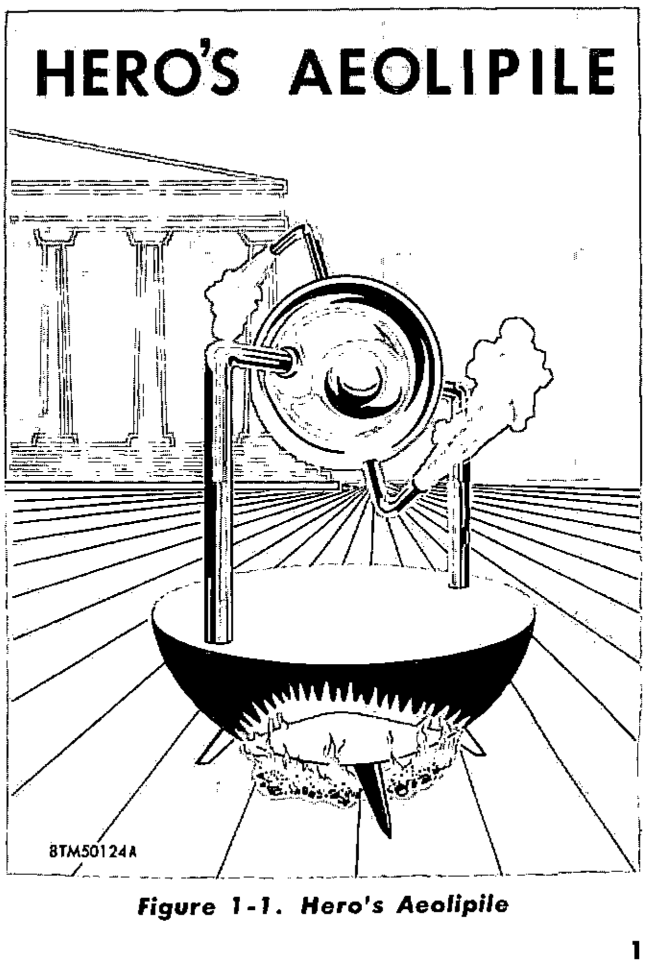


Figure 1-1. Hero's Aeolipile

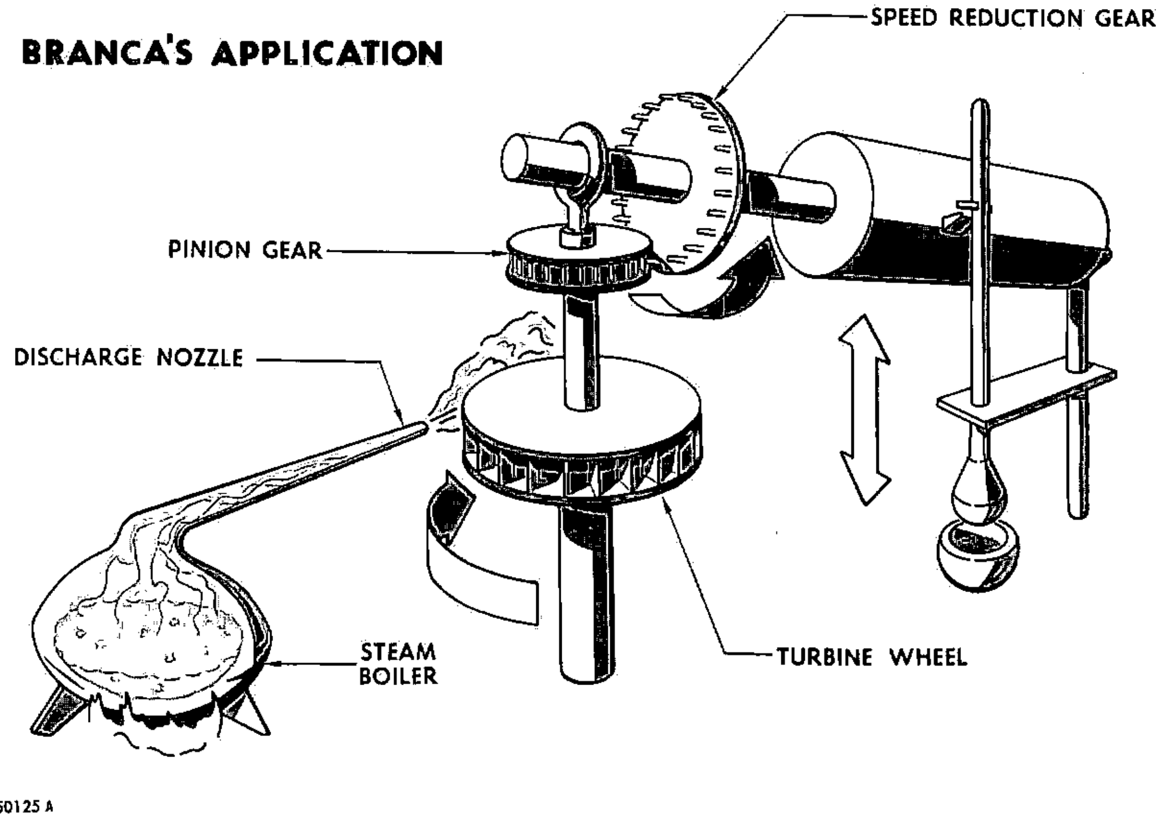


Figure 1-2. Branca's Turbine Application

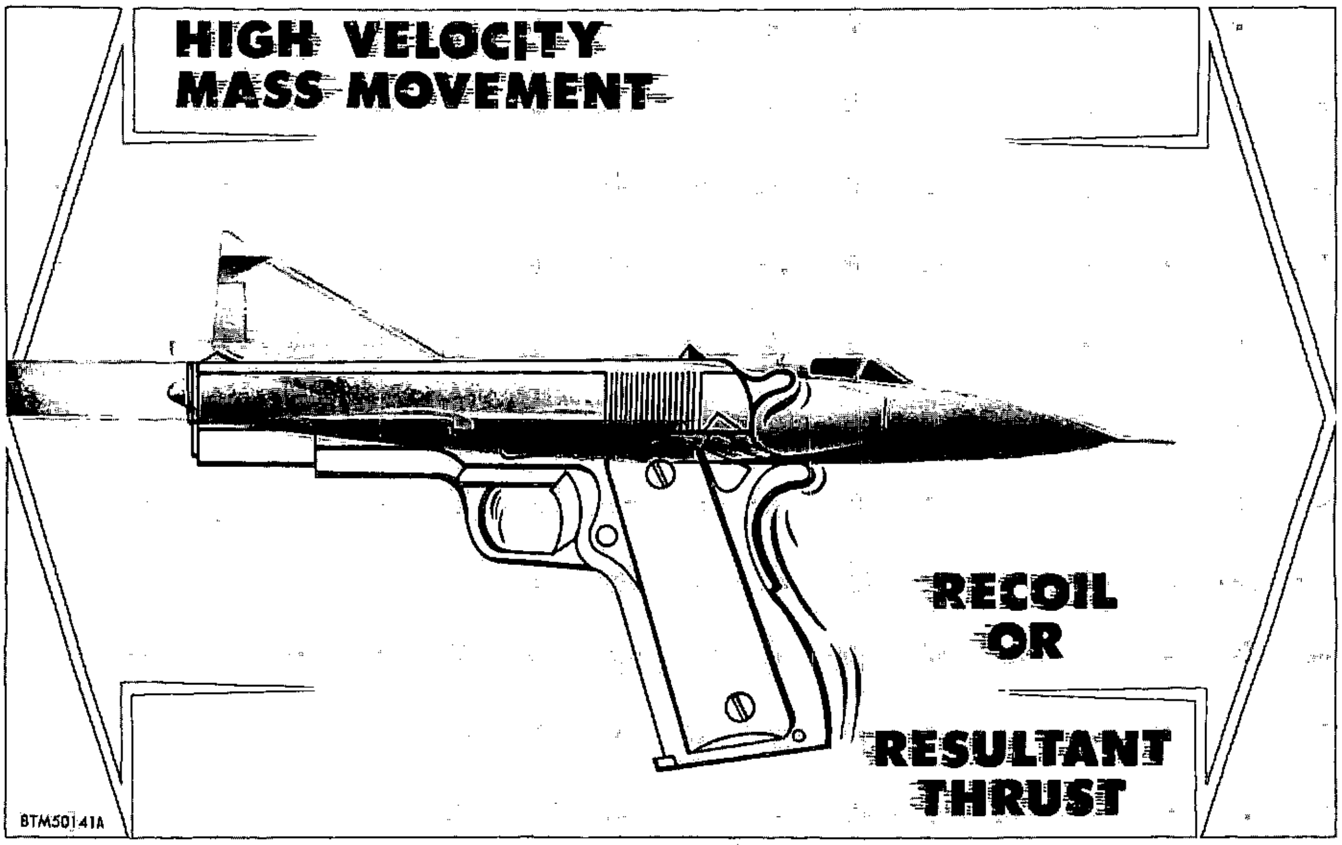


Figure 1-3. Recoil and Thrust Reaction Forces

ciple was used as far back as about 130 B.C. At that time, Hero designed and constructed what was known as "Hero's Aeolipile". You will notice in figure 1-1 that the Aeolipile was nothing more than a revolving boiler. Two nozzles discharged steam at right angles to the axis of rotation, thus imparting a spinning action to the boiler. The Aeolipile was nothing but a toy; however, it laid the ground work for later experiments which accomplished useful work.

Possibly the earliest date at which a machine accomplished useful work was in 1629, when an Italian engineer, Giovanni Branca, designed and built the first actual turbine. This machine, called "Branca's Turbine Application," is illustrated in figure 1-2. As you can see, it was an open-air combustion chamber with a spherical boiler and a discharge nozzle directed against a turbine wheel. Note also how the turbine wheel drives the pinions which in turn, through a speed reducer, drive a stamp mill. This was the first reduction gear. It is interesting to note that Hero's Aeolipile, and other machines since, were of the pure reaction type; while in Branca's Turbine Application we have the first concept of the impulse wheel.

The reaction of a mass when its velocity is changed may be considered jet propulsion. Newton's Laws of Motion state: (1) "every action produces a reaction which is equal in force and opposite in direction," (2) "the force exerted equals the mass times the rate of change in velocity." As an example of Newton's Laws of Motion, let us consider a pistol when fired, as illustrated in figure 1-3. To some, it may seem that a pistol recoils because of the expelled gases pushing on the air. This is not true—a pistol recoils in accordance with Newton's Laws of Motion. Whether a pistol is fired through water, ordinary atmosphere, or a vacuum, the recoil in all cases is exactly the same. This, of course, disproves the commonly accepted idea that the expelled gases push on the surrounding air. The high velocity jet blast of an airplane engine may be considered a continuous recoil, imparting an opposite reaction (thrust) to the airplane.

The automobile is commonly accepted as the first horseless carriage, but in 1680 Isaac Newton designed a model steam carriage of the pure reaction type. As you can see in figure 1-4, it housed a jet reaction unit on a four-wheel chassis. In this model, combustion took place in a closed combustion chamber—steam being generated in a spherical boiler and ejected rearward through a discharge nozzle. This action propelled the carriage forward. The driver controlled a steam cock in the discharge nozzle to control the speed. This device aptly illustrated Newton's Law of Motion, "for every action there is a reaction which is equal and opposite in direction," as mentioned above.

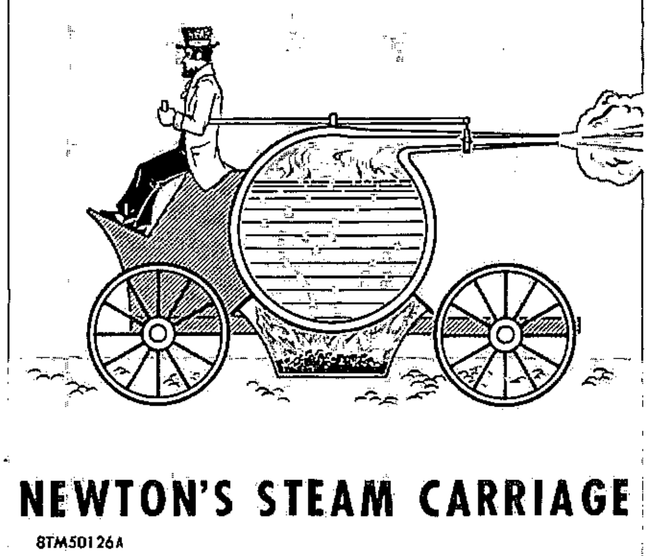


Figure 1-4. Newton's Carriage

RECENT DEVELOPMENTS IN JET ENGINES.

The patent which outlined the basic form for all modern gas turbines was applied for on January 16, 1930, in Great Britain, by Air Commodore Frank Whittle. Studying Whittle's patent sketch (see figure 1-5), you will notice its similarity to present day turbojet engines. Commodore Whittle first conceived his idea in 1928 while in his fourth term as a flight cadet at the R.A.F. College, Cranwell. However, it was not until May 14, 1941, that his dream became a reality and a turbine engine was installed in an airframe and flight tested.

While Whittle was busy conducting tests on his engine, the Germans brought out several interesting designs, among which was the Jumo-004. This engine was interesting in that it had an eight stage axial-flow compressor, six individual can-type combustion chambers, a single-stage turbine wheel, and a tail cone with an adjustable bullet. The compressor construction was interesting because of its disc design and method of assembly. The burners were also interesting because of the method of flame propagation.

During World War II, there was a constant interchange of ideas between England and the United States to hasten the development and production of jet engines. Under this agreement a Whittle engine was brought to this country for study. The General Electric Company was awarded a military contract to develop an engine for flight-test purposes. Bell Aircraft Company was selected to build the airframe. The Westinghouse Electric Company and Allison Engine

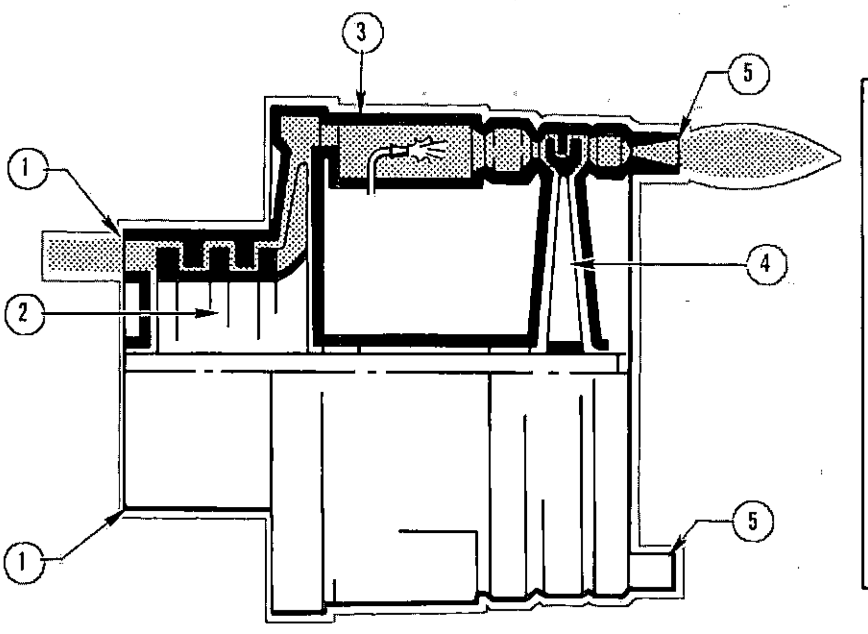


Figure 1-5. Whittle's Engine

Company were both awarded military contracts for research and development in the turbojet engine field. At the same time, the Rolls-Royce Company of England was given the original Whittle Engine to develop. After a series of modifications and improvements, the Rolls-Royce "Nenè" engine was produced. This engine was tested in the United States and the rights to produce it were purchased by the Pratt & Whitney Aircraft Engine Company. Many improvements and conformances to American standards have subsequently been made on the engine. Many organizations in the United States are currently working on numerous experimental engines. In view of this and the vast amount of engineering hours devoted to research and development, it is safe to conclude that the jet engine is here to stay.

TYPES OF JET ENGINES.

At this point, it becomes obvious that the term "jet engine" in itself means very little. Many different types of jet engines are being developed and produced. As a result of all this research and development we now have aviation gas turbines, athodyds, and rockets —all of which will be explained in the following paragraphs.

AVIATION GAS TURBINES.

Turbojet engines fall into two categories. They are either of the axial-flow or the centrifugal-flow type. The turbo-prop engine also falls into either of these categories; however, it is usually an axial-flow design incorporating a propeller. From figure 1-6, you can see the primary design differences between these types. Each has its advantages and disadvantages. For instance, the axial-flow design is more efficient because of its high pressure rise; but its manufacturing costs are greater, it is less stable, and its over-all length is greater. The centrifugal-flow design is more stable and is cheaper to manufacture; but it has a limited pressure rise. Also, its diameter is greater, and, as a result, the airframe can not be as streamlined as with the other design.

ATHODYDS.

Athodyds fall into two separate and distinct types— the pulse-jet and the ram-jet. The pulse-jet, like other jet engines, creates its propulsive force by accelerating the airstream passing through the engine. The outstanding characteristic of the pulse-jet is that the air is admitted into the combustion chamber through shutters or flapper valves in gusts, thereby causing the jet to be intermittent (pulse). The ability of the ram-jet

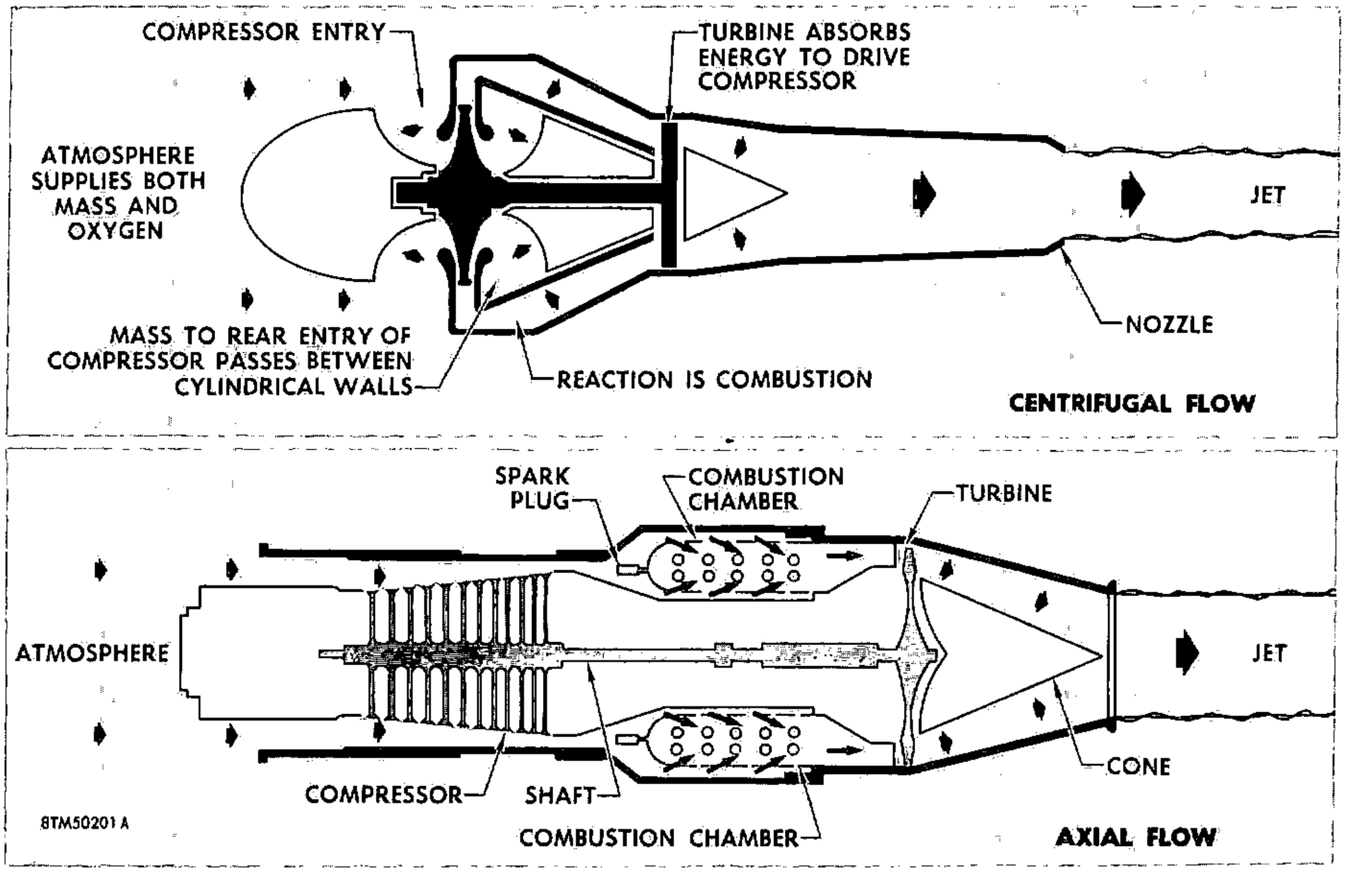


Figure 1-6. Aviation Gas Turbines

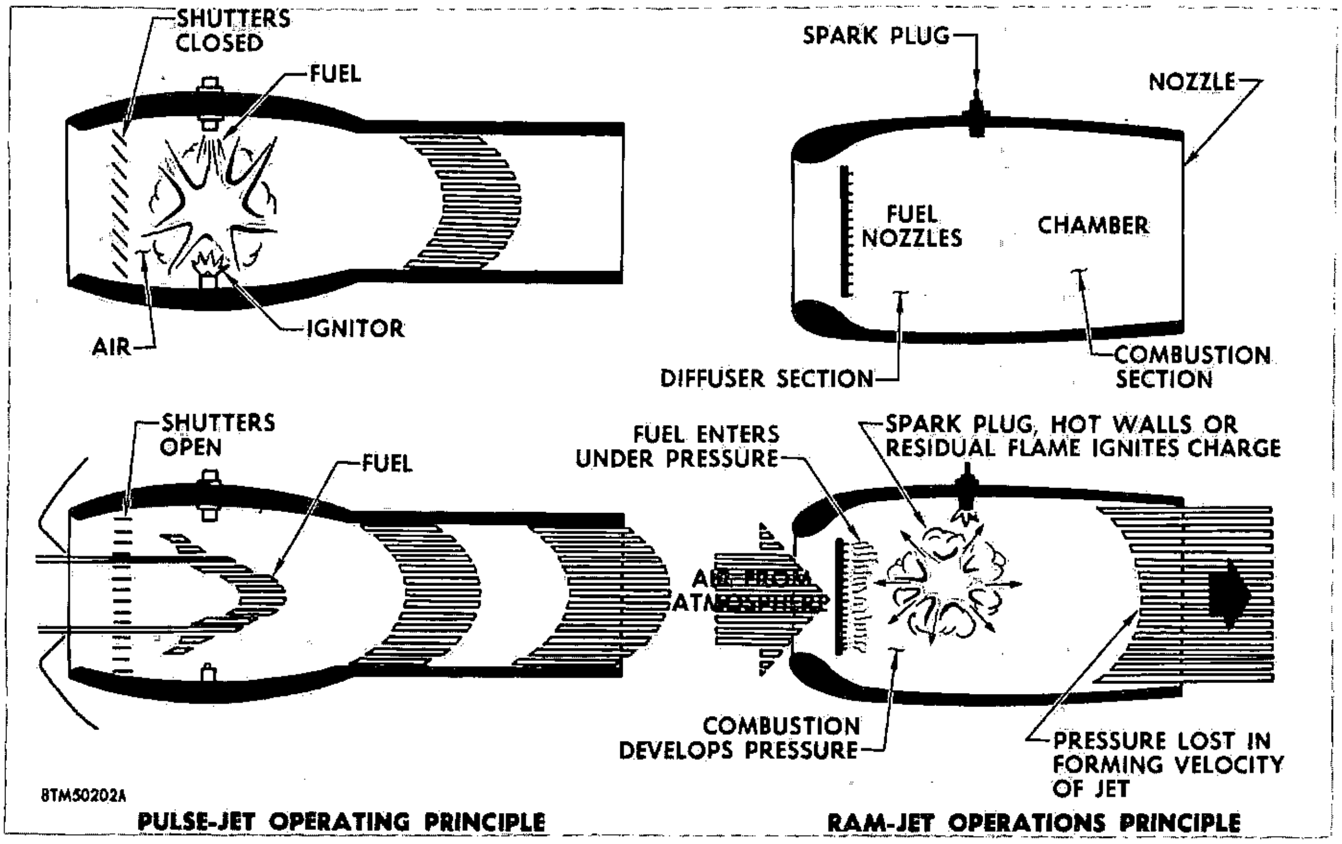


Figure 1-7. Athodyds

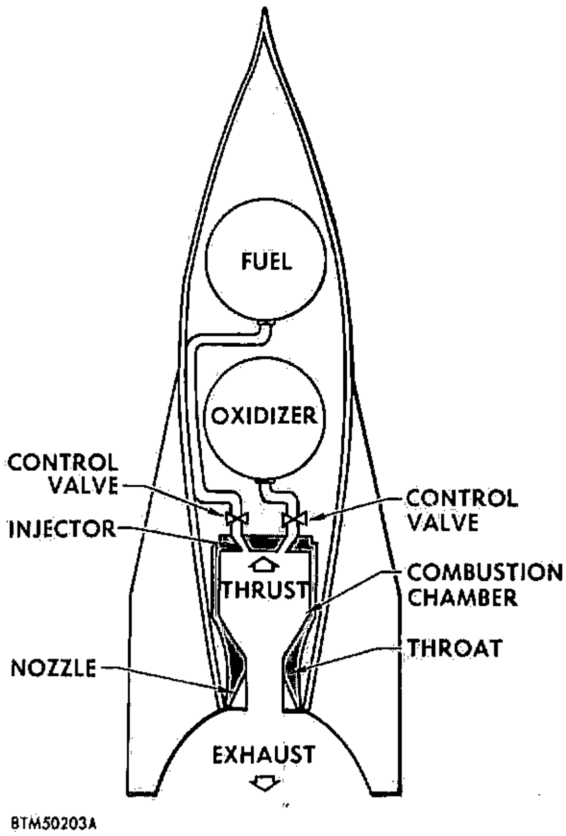


Figure 1-8. Rockets

to produce thrust depends on the velocity change in the same manner as any other reaction engine. It can be defined as a compressorless engine which depends upon the conversion of energy of the incoming air for its compression pressure. Figure 1-7 shows a cross-sectional schematic of both types of athodyds.

ROCKETS.

The rocket develops thrust by accelerating large quantities of gases generated by the chemical reaction of self-contained propellants. The chemical reaction is produced by mixing two propellants together inside the rocket. Figure 1-8 shows the principal elements of a bi-propellant liquid-fueled rocket. Notice, it has nothing more than a combustion chamber and a converging exhaust nozzle. The propellant gases are produced in the combustion chamber at pressures governed by the chemical characteristics of the propellants, their rate of consumption, and the cross-sectional area of the nozzle throat. The gases are exhausted into the atmosphere through the nozzle at supersonic speed. The nozzle converts the pressure of the propellant gases into active energy. The reaction to the discharged propellant gases is the thrust developed by the rocket.

The propellants employed in a rocket may be a solid, two liquids (fuel plus oxidizer), or materials containing an adequate supply of available oxygen in their chemical composition.

THE J57 ENGINE AND ITS COMPONENTS.

Whether you are an experienced or inexperienced mechanic, or whether you have had a wide variety of jet engine maintenance experience or have only recently graduated from Air Force Technical School, this engine should prove a challenge.

There are two engine models installed in the F-102A airplane. The earlier airplanes have a Pratt & Whitney J57-P-41 series engine which develops 14,800 pounds static thrust at sea level with afterburning. On later airplanes, a J57-P-23 series engine rated at 16,000 pounds static thrust at sea level with afterburner is installed. Figure 1-9 shows the main physical differences in the two engines.

Both engine models are of a unique design in that each incorporates a dual-spool compressor section (two separate compressors) working in conjunction with a three-stage turbine section (three turbine wheels). The compressors have an inter-compressor bleed system for matching compressor outputs throughout the engine operating range. Burner "can" construction is interesting because of the method of flame propagation and control of the flame within the "can." These two factors will be discussed later in Chapter II when you will learn about the engine fuel system. On earlier engines, the afterburner has a two-position iris-type (opens like a flower) discharge nozzle with 24 actuating cylinders. The later engine has a two-position flap-type (eight segments) afterburner discharge nozzle with only eight actuating cylinders. In figure 1-10 the two types of afterburner discharge nozzles are shown.

The engine is installed in the aft section of the fuselage. Because of the engine location, a split-type (dual) air intake duct, converging into a single duct just forward of the inlet guide vanes, is necessary. Removing and installing the engine is accomplished from the aft end of the airplane. This practice is necessary because of the "delta wing" design which prevents the more conventional method of "breaking" the fuselage and handling the engine from the fuselage "break point." The lubrication system is an engine-contained system with the exception of the air/oil cooler which is supplied by the airframe manufacturer. Fuel is supplied to the engine from six wing fuel tanks with a total usable fuel capacity of approximately 1070 US. gallons. There are also provisions for the installation of jettisonable external wing fuel tanks. As you can

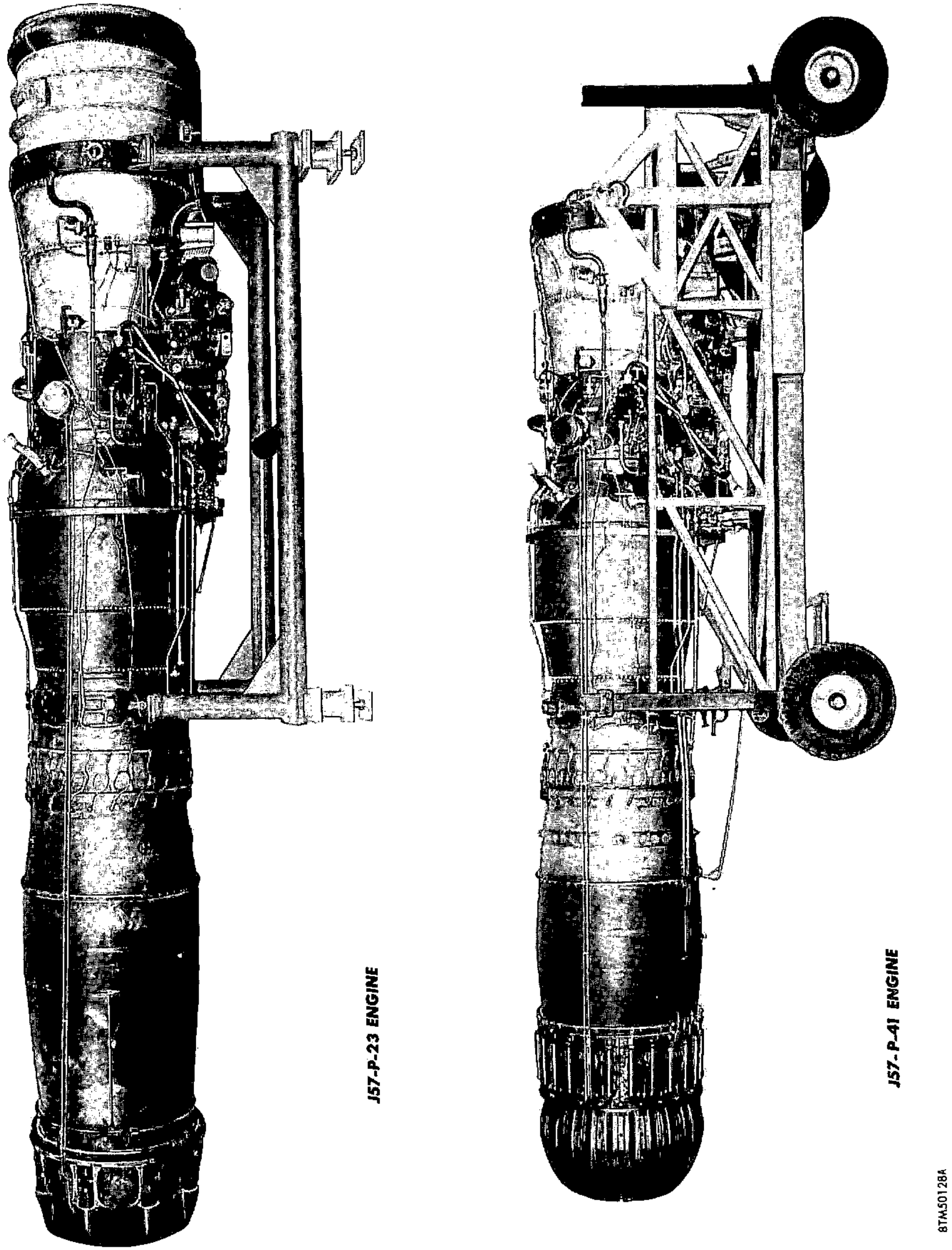
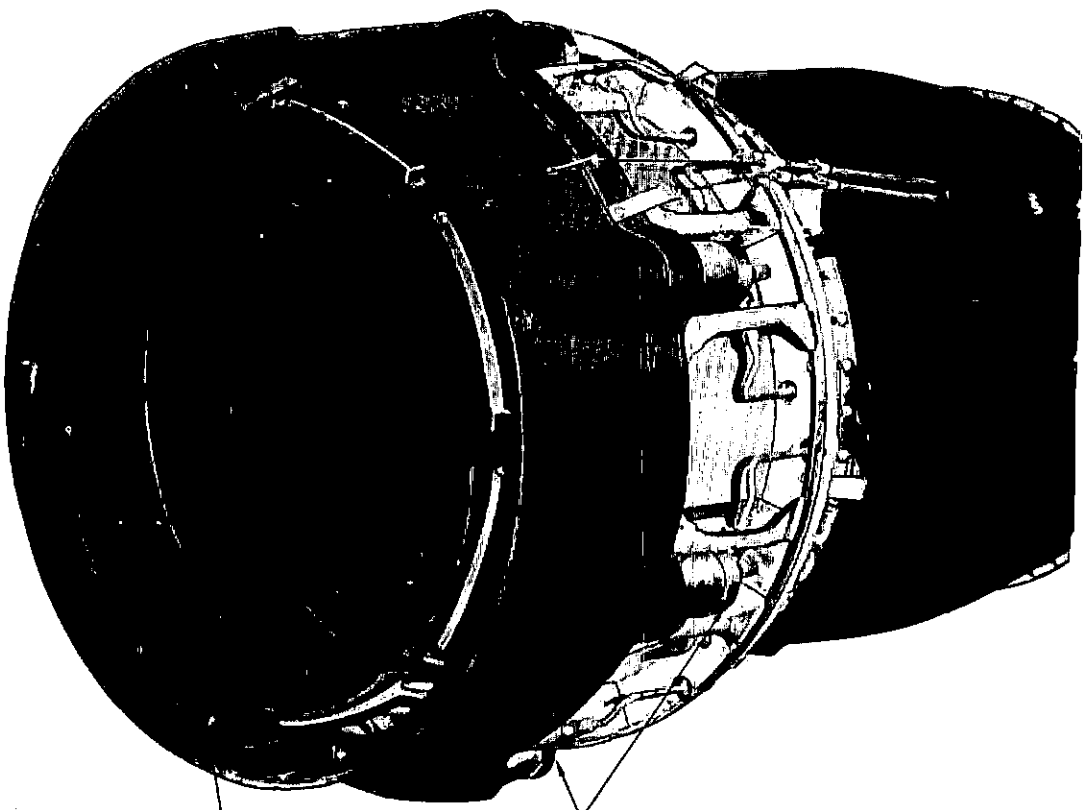


Figure 1-9. J57 Engine



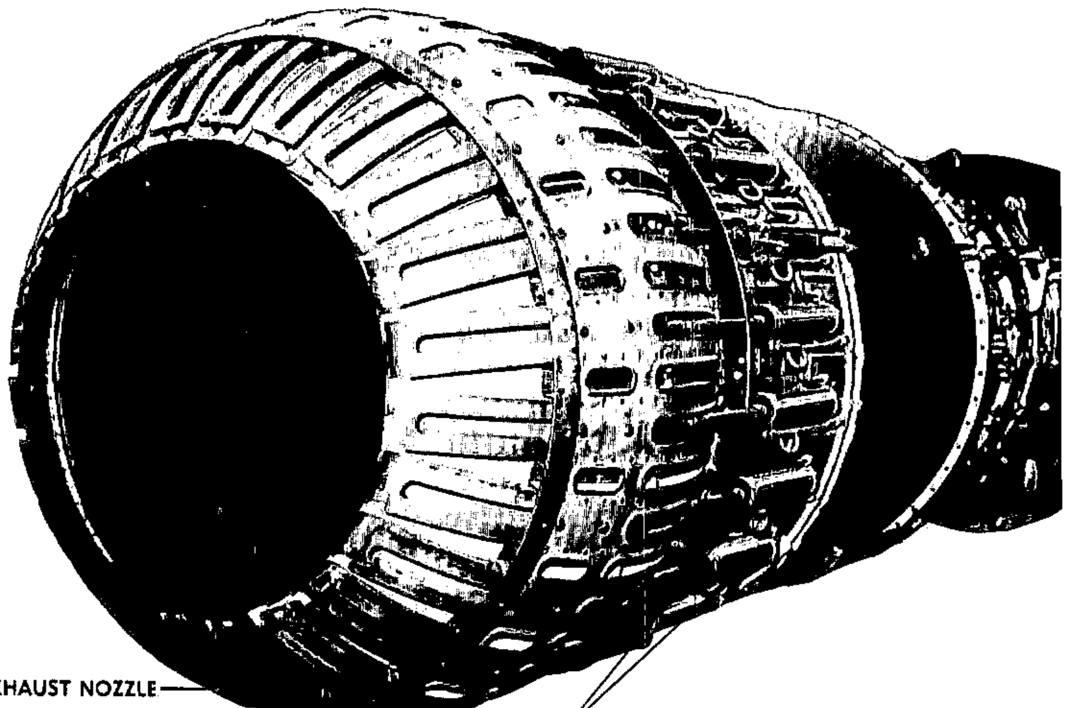


Figure 1-10. Afterburners

see, we have come a long way since the 16-horsepower machine the Wright Brothers flew at Kittyhawk in 1903.

COMPRESSOR SECTION.

From the sectional view of the illustration in figure 1-11, you can see that the forward N_1 (low pressure) compressor is made up of nine rotor stages, eight disk spacers, and a front and rear hub. The stator, or fixed, blades, are located radially between the rotor blades and are attached to the compressor case. The aft N_2 (high pressure) compressor is made up of seven rotor stages and six fixed blade stages. All rotating blades are relieved or have "thinned" tips to allow for closer tip tolerances. They are fitted to their rotor disks by contoured slots and are secured with clips. The blades are not rigidly attached, rather, each blade actually has a small amount of "play" or freedom of movement.

COMBUSTION SECTION (BURNER SECTION).

Compressor sixteenth-stage air pressure passes through the diffuser section that directs the airflow into the combustion area. The fuel manifold and nozzles are mounted on the burner cans and discharge fuel into the cans. The burner section housing is attached to the diffuser rear flange and contains the eight horizontal, radially-mounted and interconnected combustion chambers. As you can see in figure 1-12, the combustion chamber consists of perforated outer and inner liners. It has six openings on the forward face that align with the fuel nozzles. The cans can be removed individually and are basically interchangeable. The odd-numbered cans have female interconnect flame tubes, and the even-numbered ones have male interconnects. Because of this the odd-numbered cans are only interchangeable with other odd-numbered cans and even-numbered cans with other even-numbered cans. No. 4 and No. 5 burner cans have openings for igniter plugs (spark igniter guide). There is no pressure rise in the burner "can" despite the fact that combustion takes place within the can, producing both a temperature and a velocity increase. Actually, pressures at the turbine inlet are lower than those in the diffuser section. The turbine inlet collects the combustion section gases and adapts the gas flow to the turbine. The expansion of gases takes place across the turbines, furnishing the power required to drive the turbines. It is also interesting to note that the combustion is contained within the can; there is no flame passing through the turbine stages except during afterburner ignition.

TURBINE SECTION.

As noted previously, the turbine section is made up of three stages. As you can see from figure 1-11, the first stage turbine drives the aft or high-pressure compressor rotor. Note also how the final two interconnected stages drive the forward or low-pressure compressor rotor. Fixed vanes are incorporated in the turbine inlet, with stator blades located between the turbine stages. The shrouded turbine blade tips provide a good seal and permit thinner blade sections. The rotating blades are attached to the turbine disk by "fir tree" serrations and are secured by rivets lying fore and aft at the top of each "fir tree." The shrouded blade tips are scarf cut (cut on an angle) and interlock with adjacent tip shrouds; however, a small gap exists between adjacent tip shroud edges on the first and second turbine stages, permitting a small amount of "play." The third stage turbine blades are attached to the disk in the same manner, but are without the shroud edge gap in the static condition. During engine operation, however, the forces on the blades result in a small edge gap and blade freedom—similar to that found in stages 1 and 2. Seals, shaped like a knife-edge, are used to prevent the gases from entering the rotor base areas, and fixed turbine exit vanes straighten this air flow as it leaves the turbine. The turbine rear bearing support is secured by eight rods projecting radially through the turbine exhaust cast struts.

AFTERBURNER SECTION.

Normal engine thrust is augmented by afterburning such as needed for take-off, climb, or for any other flight condition requiring additional thrust. Afterburner operation usually increases normal engine thrust by approximately 50 per cent. This thrust augmentation results from the ignition of fuel introduced into the afterburner for this purpose. The oxygen that is necessary for combustion is furnished by the surplus air that is not required during the normal engine combustion process. Although afterburning is far from efficient, fuel-wise, it provides a relatively simple means of augmenting normal engine thrust.

The afterburner is merely a ram-jet type engine attached to the engine turbine section. It is composed of a diffuser section, burner section, and a variable nozzle. Looking at figure 1-13, you will see that fuel is introduced through 24 spraybars in the diffuser section. This fuel is ignited by the "hot streak" ignition method. In this "hot streak" type of ignition, the igniter valve introduces a small amount of fuel into No. 3 burner can. This momentary enrichment results in a flame—extending through the turbine blades—which ignites the fuel at the afterburner spraybars. Three circular flame holders, located in the diffuser section, retain the flame within the afterburner. A two-position pneumatically operated exhaust nozzle is automatically opened (during afterburning) and closed (for normal engine operation). During afterburner operation, the opening of the exhaust nozzle insures normal tailpipe temperatures and pressures.

The afterburner is a cantilever structure that features a double-wall construction in the aft portion for rigidity and cooling purposes. Afterburner fuel pressure

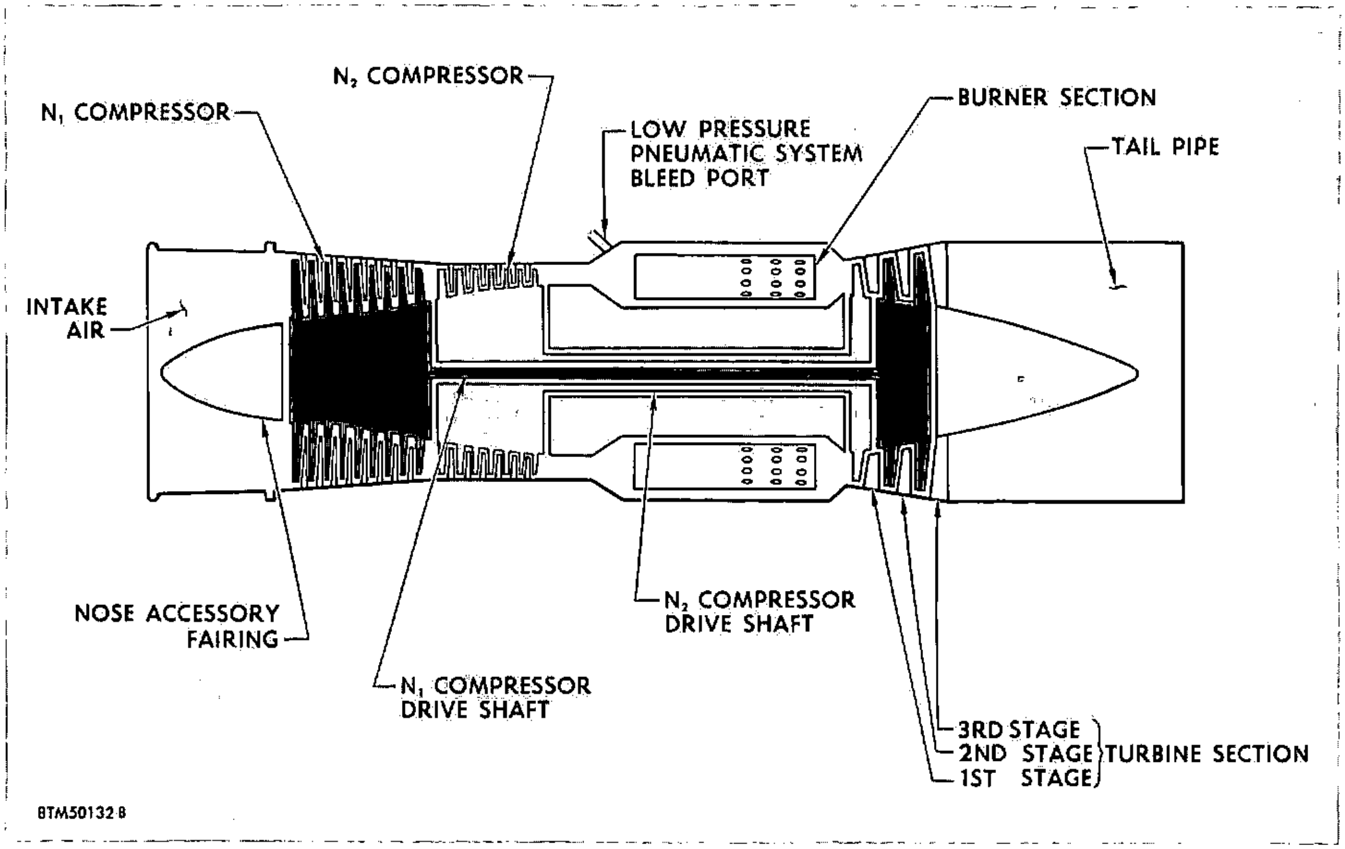


Figure 1-11. Compressor and Turbine Sections

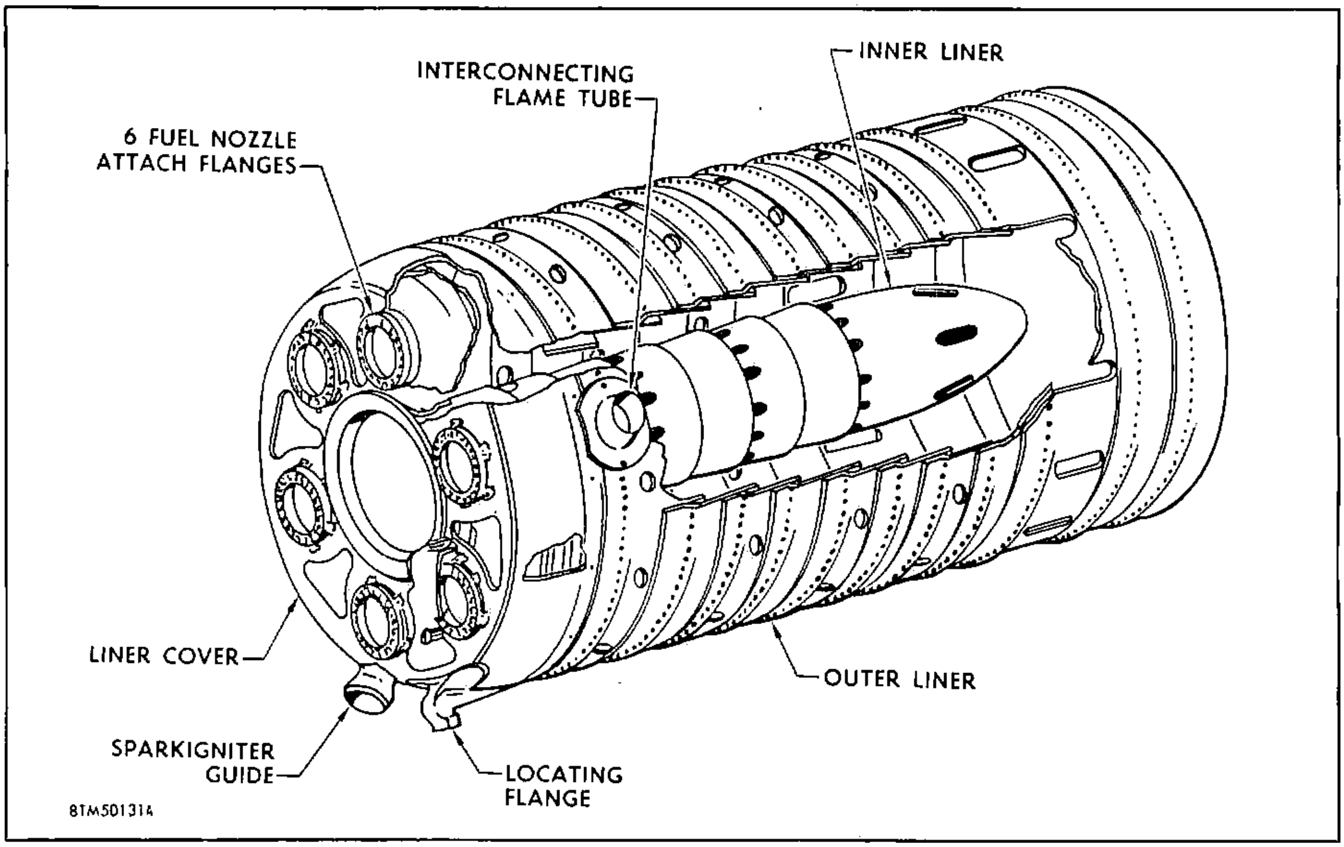


Figure 1-12. Combustion Chamber

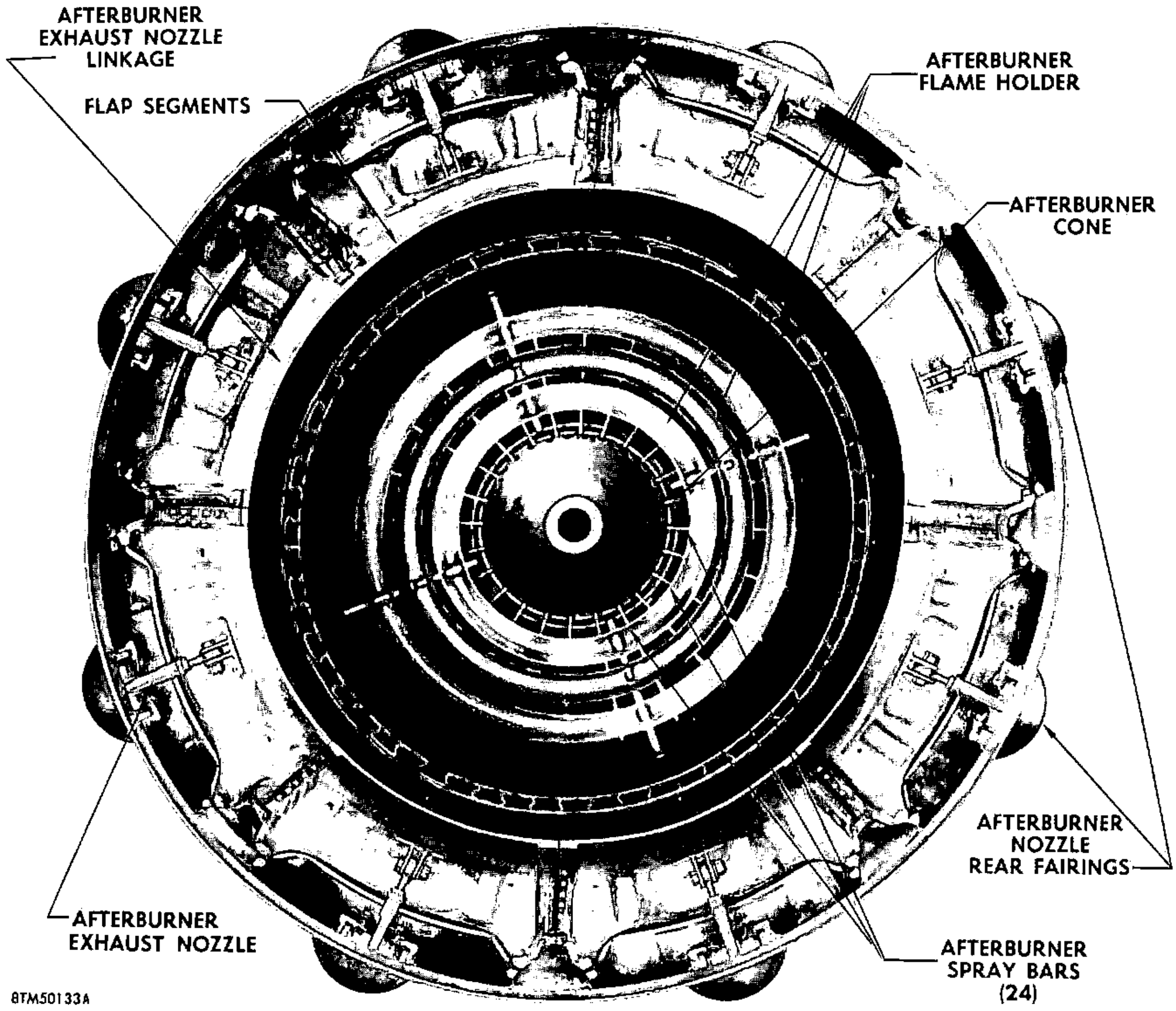


Figure 1-13. Afterburner Components

actuates the exhaust nozzle control valve, which ports sixteenth-stage air pressure to the OPEN side of the nozzle actuating cylinders. When afterburner fuel pressure falls off, spring pressure in the exhaust nozzle control valve ports sixteenth-stage air pressure to the CLOSE side of the nozzle actuating cylinders.

ACCESSORY SECTION.

The accessory section is located at the bottom of the engine in the "wasp waist" or smallest cross sectional area, as shown in figure 1-14. Power to drive this accessory section is taken from the high-pressure compressor rotor's rear hub—transmitted through matching gears on a canted shaft and forward through the horizontal shaft to the accessory drive case—at a ratio of 7 to 1. Driven units are the two hydraulic pumps (17) and (40), the pneumatic starter (16), the tachometer-generator (mounted on the forward face of the fuel pump/transfer valve (21), and the engine fuel control (35), located on the right and left aft face of the accessory drive housing, respectively). Other components located in this area are the fuel pressurizing and dump valve (28) and the two ignition transformers (26) and (33). The afterburner (A/B) igniter valve (2), A/B fuel regulator (23) and the exhaust nozzle control valve (27) are located on the right side of the engine in the "wasp waist" area. The Sundstrand constant-speed drive unit (39) is mounted between the starter and the mounting pad. Notice that some fuel and oil system components are located in this area too —these will be discussed later in this supplement.

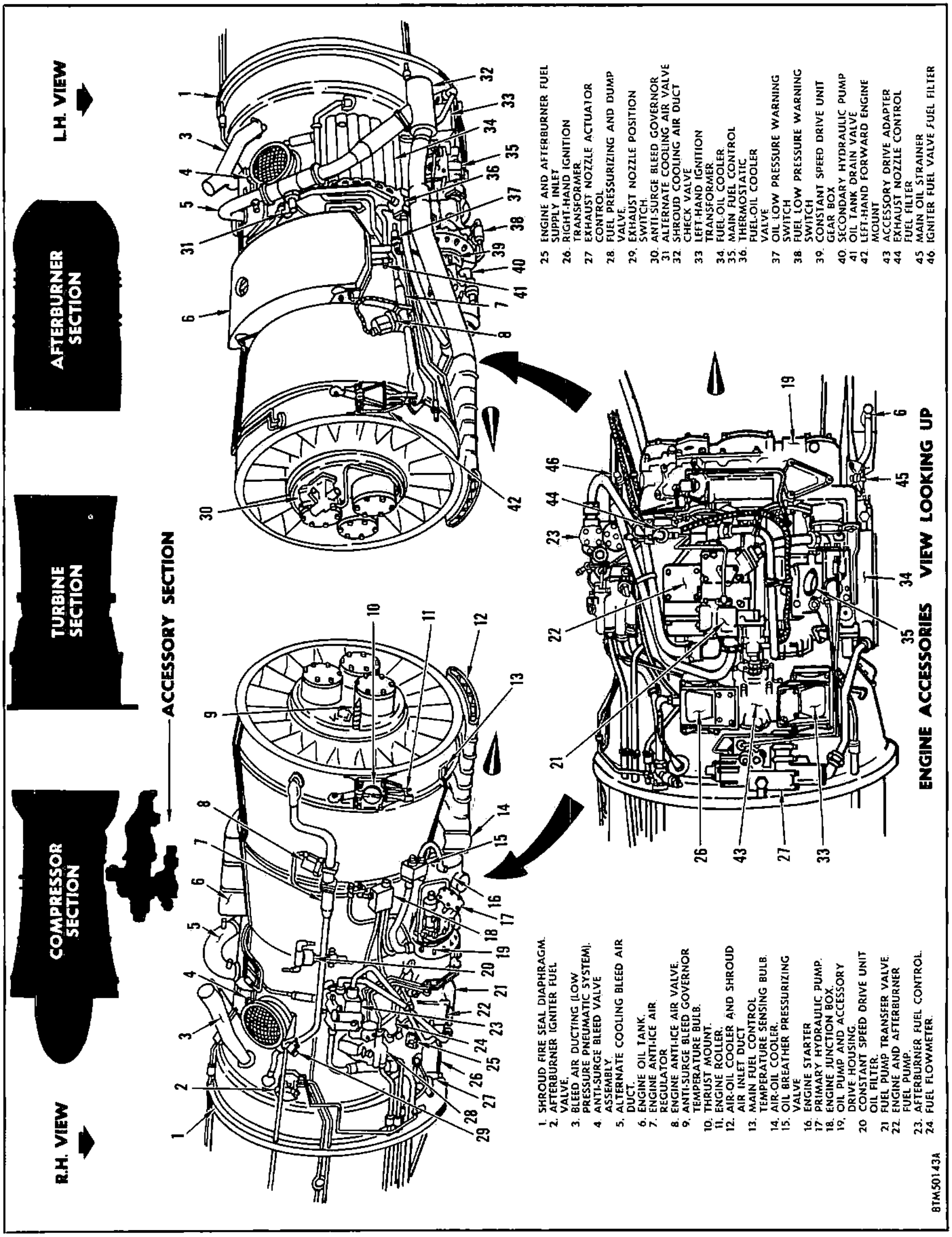


Figure 1-14. Accessory Section

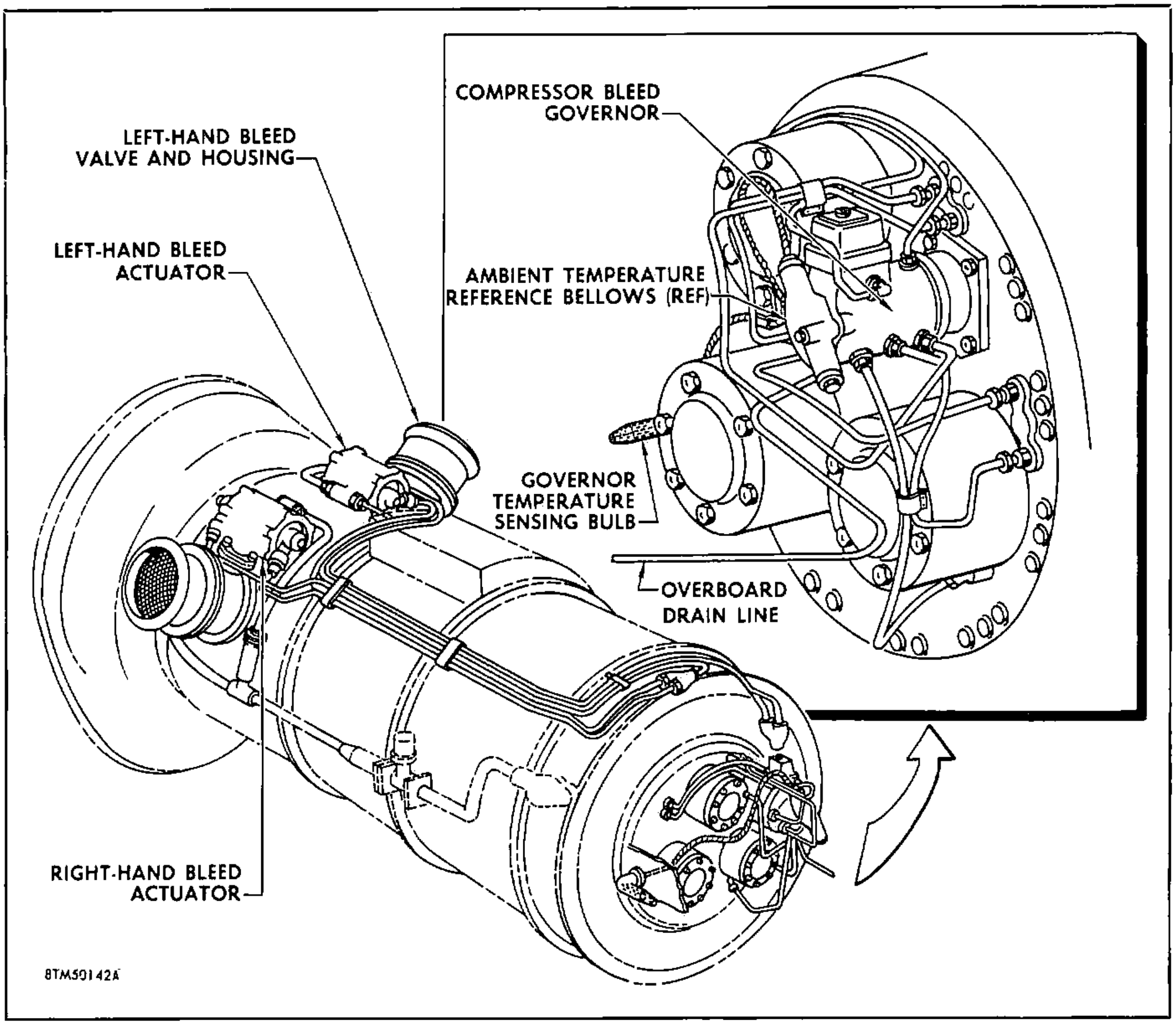


Figure 1-15. Compressor Bleed System

COMPRESSOR BLEED SYSTEM.

The compressor bleed system is used to maintain the best rpm and pressure relationship between the N_1 and N_2 compressors throughout the engine operating range. Its primary function is to minimize the possibility of compressor stall during acceleration and deceleration. The J57-P-41 engine incorporates two bleed valves, while the J57-P-23 engine features a single bleed valve. The function of both engine bleed systems is identical, but for clarity we shall study the dual bleed valve system. The two bleed valves and their actuators are located on either side of the top center line of the "wasp waist" section. The compressor bleed governor is mounted on the N_1 accessory section to control the bleed valve operation. At low power settings both bleed valves are open and a portion of N_1 compressor outlet is vented overboard. These two valves do not operate simultaneously; instead, during engine acceleration the right valve closes first and the left valve closes shortly thereafter. During engine deceleration the left valve opens first and the right valve opens later as engine speed is reduced. This operation is entirely automatic. The bleed valves are controlled by the compressor bleed governor, which senses N_1 rpm, along with temperature and pressure, at the inlet guide vanes. Varying low pressure compressor speeds, biased by inlet air temperature and pressure, position the pilot valves which direct sixteenth-stage (air pressure bled from the last stage of N_2 compressor) air pressure to either the OPEN or CLOSE side of the two bleed valve actuators.

ENGINE ANTI-ICING SYSTEM.

To prevent ice accumulation on the inlet guide vanes, heated sixteenth-stage air pressure is ducted through

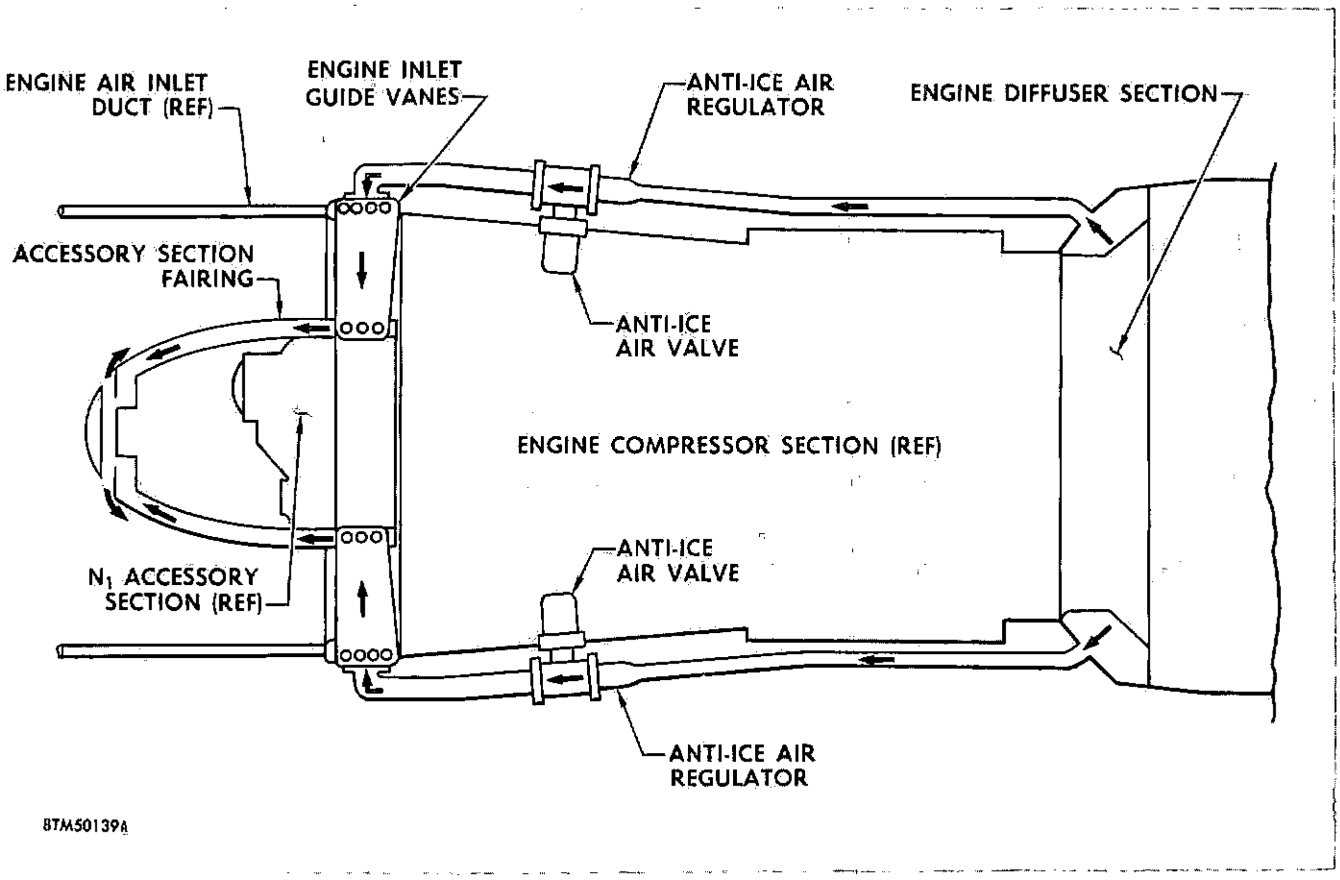


Figure 1-16. Anti-Icing System

hollow vanes. Figure 1-16 shows the two separate air lines leading forward from the diffuser section to the chamber surrounding the inlet guide vanes and through the vanes. Airflow is controlled directly by the two electrically-operated anti-ice air valves and indirectly by the two anti-ice air regulators.

The two air valves open when the anti-icing system is initiated by the engine anti-ice detector system. When the valves are open, air from the compressor diffuser section is routed to the inlet guide vanes. The anti-ice air regulators—located directly upstream from the shutoff valves—automatically control the air flow. Each regulator contains a bi-metallic coil spring that moves its valve toward the CLOSED position as the air temperature increases. Accordingly, airflow to the inlet guide vanes proportionately reduces as the anti-icing air temperature rises.

Notice that anti-icing air flows from the inlet guide vane outer chamber—between the outer case of the inlet guide vane and the outer shroud — inward through the hollow vanes to the inner chamber—between the inlet guide vane inner shroud and the inlet guide vane air passage cone. You can see that air flows forward from the guide vane roots through the double wall accessory section fairing and exits aft of the nose cap.

ENGINE MAIN BEARINGS.

Seven main bearings support and maintain alignment of the compressors, drive shafts, and turbines. You can find the exact location of the seven main bearings in figure 1-17. Bearing No. 1—a roller bearing located in the center of the forward compressor guide vane housing—supports the low pressure rotor front hub. A matched pair of ball bearings—attached to the compressor intermediate bearing support housing—make up the No. 2 thrust bearing that supports the low-pressure rotor rear hub. The high-pressure rotor front hub is supported by No. 3 roller bearing whose inner race is secured to the N_1 compressor rear hub. The second pair of matched ball bearings make up the No. 4 thrust bearing that supports the aft compressor rear hub. The bearing housing is secured to the diffuser case center section, and the inner race bears on the high pressure compressor drive shaft. Bearing No. 4.5 is the intershaft bearing and is located about halfway between bearing No. 4 and No. 5. The inner race of this roller bearing is secured to the forward rotor drive shaft, and the outer race bears against the inner surface of the aft rotor drive shaft. The first stage turbine is supported by another roller bearing, No. 5, whose outer race is attached to the aft section of the turbine front bearing support structure. Bearing No. 6 supports the two rear turbine stages. The

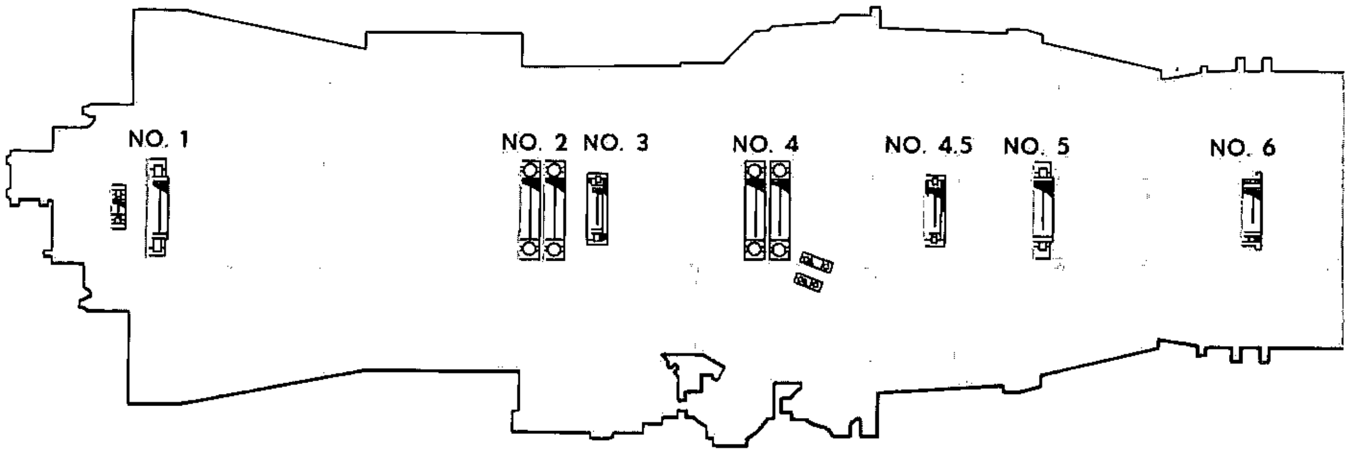


Figure 1-17. Engine Main Bearings

inner race of this roller bearing bears on the turbine rear hub; the outer race is supported by the turbine rear support. This support is secured by eight support rods projecting radially through the struts and ending in support journals on the outside of the case.

These bearings incorporate spring-loaded hard carbon seals to control oil leakage. Two of these seal rings are located aft of No. 1 bearing. No. 2 bearing has two seals located ahead of the bearing installation. Two smaller ring seals for No. 3 bearing are located aft of the bearing in the seal housing. Two carbon rings, aided by knife-edged seals, are located ahead of No. 4 bearing assembly. Oil escaping from No. 4.5 bearing is prevented from flowing aft between the two shafts by a pair of ring seals. Two more ring seals are located aft of No. 5 bearing to prevent oil leakage into the turbine area. Bearing No. 6 is sealed by two carbon ring seals located ahead of the bearing.

SUNDSTRAND CONSTANT-SPEED DRIVE UNIT.

The constant-speed drive unit is incorporated in the F-102A airplane engine installation for the purpose of driving the a-c and d-c generators, and also as a mounting pad for the engine starter. You can see that the drive consists of an engine mounted gear box, an interconnecting drive shaft, and an airframe mounted transmission and gear box assembly. The engine mounted gear box has a mounting pad for the engine starter. This gear box is directly connected to the engine compressor shaft by bevel gears.

The airframe mounted transmission and gear box assembly has two mounting pads, one for the a-c generator and one for the d-c generator. When the engine is operating, its rotation is transmitted from the engine mounted gear box, to the interconnecting drive shaft, to the airframe mounted transmission and gear box assembly, and finally drives both the a-c and the d-c generators.

As has been previously mentioned, the airframe-mounted transmission and gear box assembly has a mounting pad for the a-c and d-c generators. The a-c generator mounting pad provides rotation for the a-c generator in a counterclockwise direction (facing the pad) at a constant speed of 6000 ($\pm$60) rpm, when the input (engine rotation speed) to the engine mounted gear box varies between 3585 to 7020 rpm. The d-c generator rotational speed is not as critical as that of the a-c generator—as a result, it is driven at a direct ratio to engine rpm.

OPERATION OF THE J57 ENGINE.

Air entering the intake ducts of the airplane is directed into the forward N_1 low pressure compressor by the intake section inlet guide vanes. The action of the rotor blades against the stator blades increases the pressure of the incoming air. From the N_1 compressor, the air enters the aft N_2 high pressure compressor and is further compressed through seven stages of rotor and stator blades. At this time, there is a relatively high pressure increase accompanied by a slight velocity decrease. This velocity decrease is necessary to prevent the flame from blowing out in the combustion chambers.

As the air pressure leaves the N_2 compressor, it enters the burner section combustion chambers where it is mixed with injected fuel and ignited by two igniter plugs. All burner "cans" are interconnected, which permits simultaneous combustion in all eight "cans." Six spray nozzles in each "can" control the fuel spray

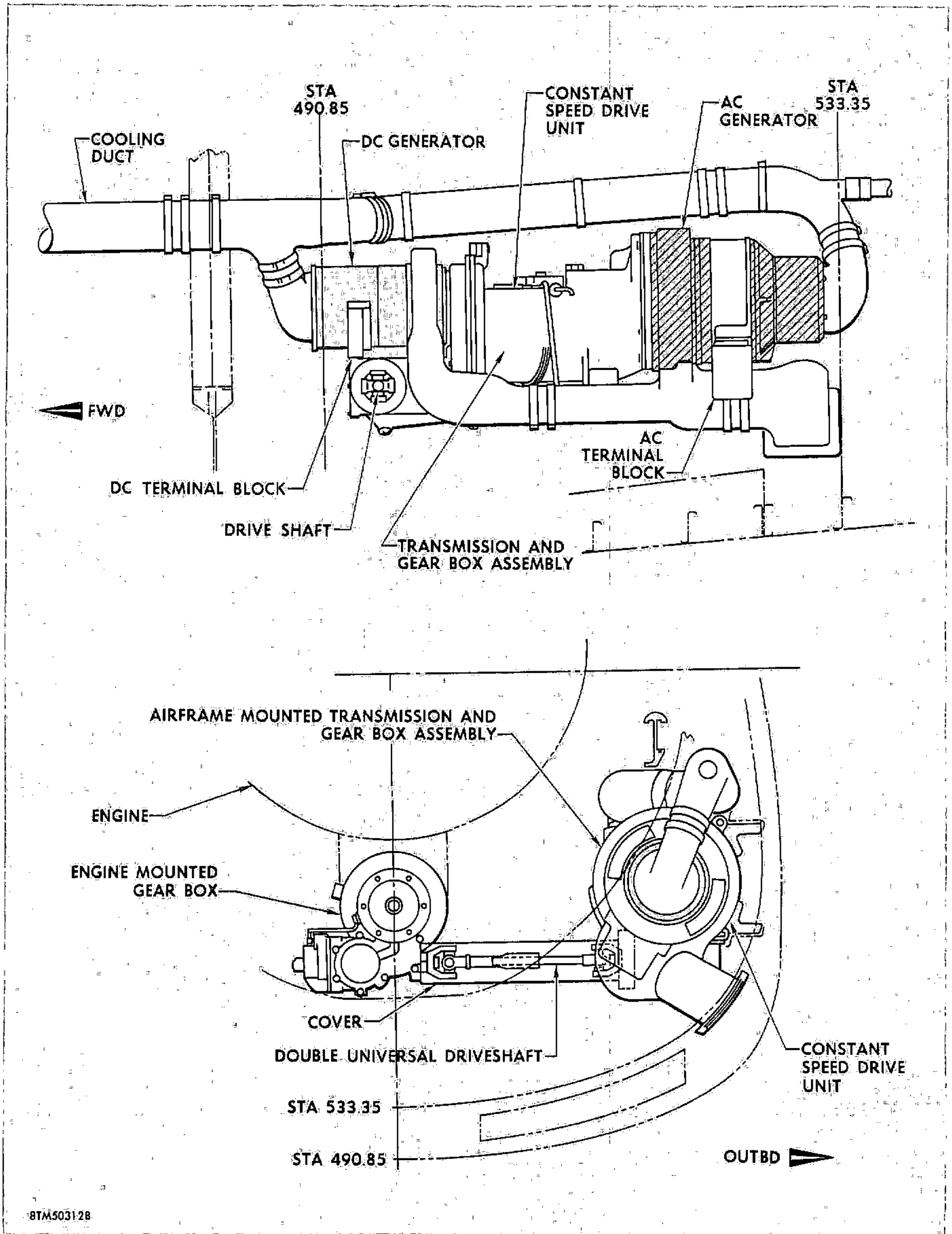


Figure 1-18. Sundstrand Constant-Drive Units

pattern, which in turn controls the flame pattern within the "cans." Rapidly expanding gases move aft through a three-stage turbine to drive the turbine wheels. The second and third stages of the turbine are directly connected to the N_1 compressor by a common shaft, and the first stage turbine is directly connected to the N_2 compressor shaft. Both drive shafts operate independently and rotate freely at their own speeds. The expanding gases force or drive the turbine wheels to turn at a high rpm. They in turn rotate the compressors at the same rmp. This causes air to enter the engine air intakes to provide the mass airflow to produce thrust. There is an intercompressor bleed system which will balance the two compressor outputs automatically should they become unbalanced. This increases engine efficiency and stability. Each of the earlier J57 engines has a dual intercompressor bleed valve, while later engines have single valves.

As these rapidly expanding gases speed rearward out of the tailpipe, they create a reaction in the opposite direction. This reaction produces the forward thrust necessary to move the airplane.

When the power lever in the cockpit is positioned into the afterburner detent, fuel is injected into the tailpipe by 24 spraybars and ignited by a flame from the No. 3 burner "can." There are three "V" shaped circular flame holders in the tailpipe which control the flame pattern within the afterburner. The two-position discharge nozzle opens, afterburner ignition takes place, and steady afterburning begins. As the throttle is retarded, fuel is shut off to the afterburner, the discharge nozzle is closed, and normal engine operation continues.

SUMMARY.

In this chapter you have learned about the J57 engine, its operation, its design features, and the physical location of components. By this time you should feel well acquainted with the engine and its closely related systems. If not, you should review this chapter before starting the next.

In the next chapter you will learn about the J57 engine systems—this will aid you in understanding the complete engine and its operation.

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Chapter II

THE POWER PLANT SYSTEMS

Contents *Page*

In a turbojet engine, such as the J57, the compressor and turbine wheels are mounted on a single shaft at opposite ends of the combustion area. When combustion takes place, the rapidly expanding gases flow through the turbine wheel vanes and cause the wheel to rotate. When the turbine wheel rotates, it drives the compressor, which packs air into the combustion areas. The increased combustion pressure imparts additional force to the expanding gases. This is the combustion cycle in a turbojet engine; and if there is some means of starting the rotation and combustion, the engine will produce a continuous flow of power.

In addition to the basic part, which accomplishes the combustion cycle, every turbojet engine must also have several associate systems. These associate systems function to assure that the engine will produce its continuous flow of power in the best operating conditions possible. In this chapter you will learn about the major associate systems in the J57 engine. These include the fuel, oil, starting and ignition, induction and cooling, and anti-icing systems.

THE J57 ENGINE AND AFTERBURNER FUEL SYSTEM.

The acceleration and deceleration control of a piston-type engine is usually accomplished by the throttle through the mechanical linkage between the throttle and the carburetor, and a similar arrangement for controlling the pitch of the propeller. With this type of installation, the rate of engine acceleration and deceleration is directly proportional to the *rate of movement* of these two controls in the cockpit. This condition, however, does not exist on jet engines.

During the very early stages of jet engine development, it was learned that rapid movement of the throttle in either direction usually resulted in a rich blow-out or a lean die-out of the flame in the engine combustion chambers. Blow-out merely means that if the engine operator moves the throttle forward too rapidly, he injects too much fuel into the combustion chambers and the flame is blown out. Conversely, if he retards the throttle too rapidly, the engine flame dies out from lack of fuel. Since most pilots were accustomed to the piston-engine type of fuel controls, blow-out and die-out in early jet operation were quite common. Power plant design engineers decided that the best solution was to relieve the pilot of all direct control over the engine acceleration and deceleration rate. This decision led to the development of the automatic fuel control unit, which is more commonly called the main fuel control.

The main fuel control allows the jet engine to accelerate and decelerate at a predetermined safe rate, thereby reducing the possibility of a flameout. This rate of acceleration and deceleration is kept constant by the main fuel control even though the power lever in the cockpit is moved rapidly. As you learn about the J57 engine fuel system in this chapter, you will

discover how the main fuel control unit governs both of these functions and relieves the pilot of almost all of the responsibilities of monitoring the fuel to the engine.

The J57 engine and afterburner fuel system is designed to schedule fuel in the proper amounts to the engine under all conditions of flight. This fuel system will supply the correct amount of fuel for fixed-power settings at varying temperatures and altitudes. The system will also permit proper engine acceleration and deceleration—without exceeding engine limitations, experiencing lean or rich blow-out, or experiencing compressor surge.

Although the engine and afterburner fuel system is designed to function as one individual fuel system, it will be treated in this manual as if it were composed of two subsystems—the engine main fuel system and the afterburner fuel system. The engine main fuel system meters fuel according to the requirements of the engine for any given power lever setting. The afterburner fuel system schedules the fuel for afterburner operation, actuates the exhaust nozzle control valve, and initiates afterburner ignition.

The engine-driven fuel pump/transfer valve is the only component that is common to both the engine and afterburner fuel systems. It is a combination unit with one part of the unit functioning as the fuel pump for both fuel systems and the other part of the unit scheduling fuel to the afterburner fuel system whenever required. The transfer valve part of this combination component also includes a safety device which automatically switches the fuel flow from the afterburner fuel system to the engine main fuel system in the event that the fuel pump for the engine main fuel system fails.

The components in the engine main fuel system include the fuel pump/transfer valve, the main fuel control unit, the fuel flowmeter, the fuel/oil cooler, the pressurization and dump valve, and the burner nozzles and manifolds.

The afterburner fuel system components are the fuel pump/transfer valve, the afterburner fuel regulator, the igniter valve, the exhaust nozzle control valve, the manual afterburner shutoff valve, and the fuel spraybars and manifolds.

THE ENGINE POWER CONTROL SYSTEM.

The pilot's power lever control quadrant is located on the left side of the F-102A cockpit. The movement of the power lever is transmitted mechanically to the engine main fuel control unit and to the mechanically-operated afterburner shutoff valve. As you will note in figure 2-1, a Teleflex cable installation connects the power lever quadrant bellcrank to the control mechanism in the left side of the engine compartment. This aft control mechanism is attached to a bellcrank on the engine control cross-over shaft. Solid linkage connects the bellcrank to the fuel control unit and to the afterburner shutoff valve.

The slot in the power lever control quadrant permits fore and aft movement of the power lever. It also permits the power lever to be moved outboard and inboard to a limited degree. The fore and aft movement of the power lever is transmitted mechanically to the fuel control unit by means of the Teleflex cable installation. The inboard and outboard motion of the power lever actuates microswitches in the quadrant which control the engine starting and afterburner initiation.

In figure 2-2, note that the fore and aft motion of the power lever is affected by both internal and external detents and a trigger-operated mechanical stop (power lever trigger). The detents serve a two-fold function; they prevent the pilot from inadvertently moving the power lever to some undesirable position, such as retarding the lever to the OFF position, and the like; they also assure that the power lever must be moved in such a manner that it will actuate the microswitches inside the quadrant.

A flat pattern of the quadrant slot is shown in the right hand portion of figure 2-2. Note the START detent which permits the lower lever to be moved outboard from the OFF position. This detent also limits the forward movement of the power lever when the lever is positioned to the extreme outboard side of the lever slot. If the power lever is to be moved forward beyond the START detent, the lever must be returned to the inboard side of the lever slot so that it will clear the IDLE detent. From the IDLE position, the power lever can be advanced forward still further until it reaches the FULL MIL POWER position. At this point, a mechanical stop prevents advancing the power lever any farther until the trigger on the power lever is depressed. When the operator pushes this trigger down, he can move the power lever from the FULL MIL POWER position to the TAKE OFF position. Movement of the power lever from the FULL MIL POWER position to TAKE OFF does not increase engine power; that is, the movement does not increase the amount of fuel that is delivered to the engine—the power lever movement merely positions the mechanical takeoff locks in the fuel control unit. These fail-safe takeoff locks prevent loss of engine power in the event of a malfunction in the burner pressure or inlet pressure sensing systems. The mechanical takeoff locks in the fuel control unit will be explained in greater detail later in this chapter.

When the power lever has been advanced to about 80% power, the AFTERBURNER detent in the quadrant cover allows the power lever to be positioned

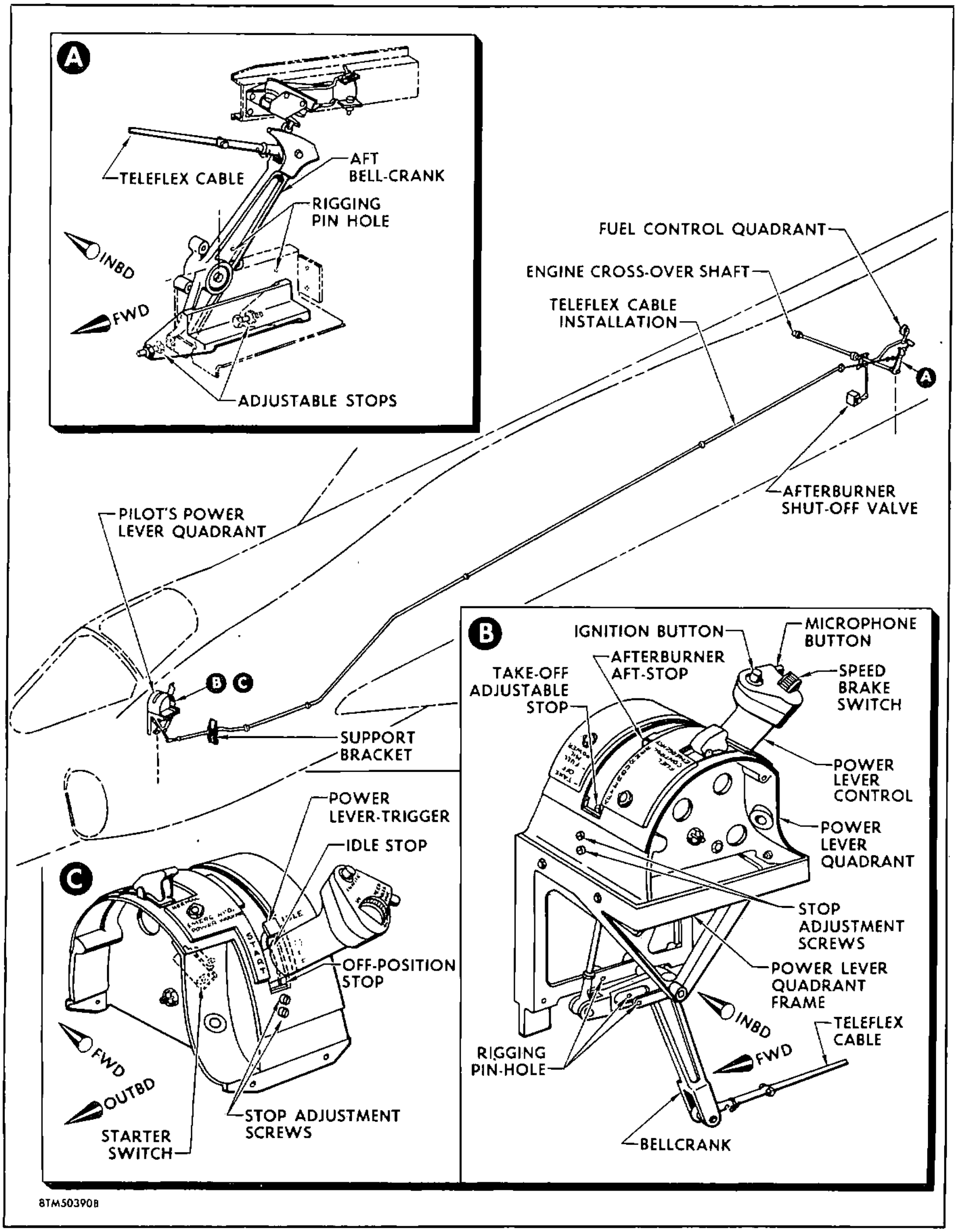


Figure 2-1. Power Lever Control System

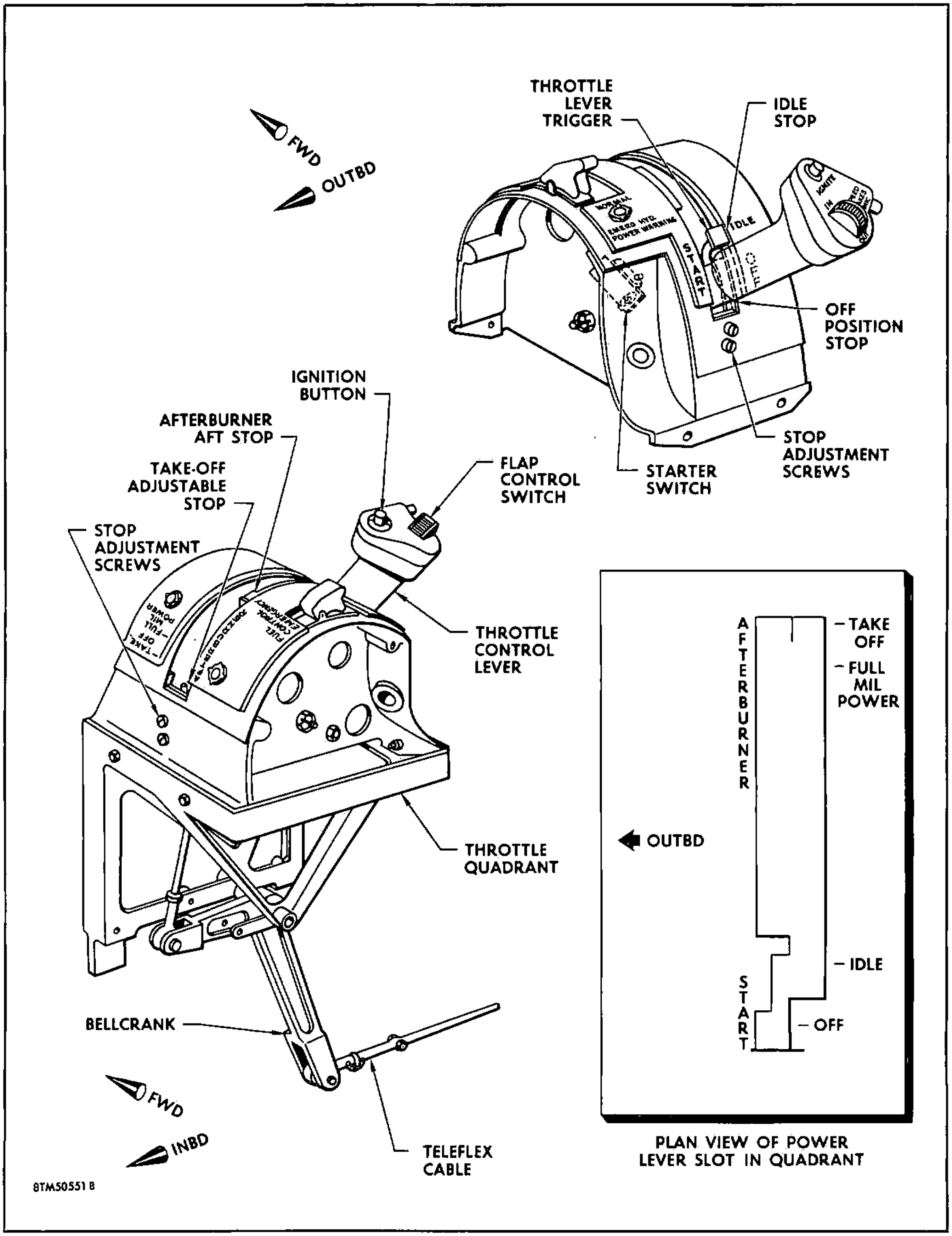


Figure 2-2. Pilot's Power Lever Positions

outboard. As the power lever is pushed outboard in this detent, it actuates microswitches inside the quadrant to initiate afterburning. A spring-loaded ball detent holds the power lever in the AFTERBURNER detent. The afterburning operation is normally stopped by bringing the power lever back to the inboard side of the quadrant, thus de-activating the microswitches in the quadrant. The power lever has to be retarded about five degrees from the full forward position before it can be brought out of the AFTERBURNER detent. This feature insures against the possibility of inadvertently stopping afterburning during take off. A mechanically-operated afterburner shutoff valve (controlled by the power lever movement) stops afterburning when the power lever is retarded below 80% power.

During power reduction the power lever contacts the IDLE detent in the quadrant cover. A slight outboard movement of the lever allows the lever to clear the IDLE detent and be retarded still farther back into the OFF position.

Referring back to figure 2-1, note that both the power lever quadrant and the engine compartment aft bellcrank have adjustable mechanical stops which limit power lever movement. These mechanical stops are preset on all airplanes. Normally you will not have to adjust these stops unless you are installing a new Teleflex cable assembly. Even when the engine is changed, the power control rigging is not disturbed, and adjustment of the mechanical stops is not required.

Rigging the power lever linkage after bellcrank or Teleflex cable replacement, however, should not present any complex problems. Before attempting to secure any part of the linkage, you must insert rig pins in both the power lever and engine aft bellcranks. The rig pins will keep the two bellcranks in their OFF positions while you install the Teleflex cable assembly. Both of the adjustable mechanical stops on the bellcranks should be backed off so that they will not interfere with the rigging process. When installing a new Teleflex cable assembly, you must adjust the rod end at each end of the cable until the rod end linkage bolts can be inserted. Do not remove the rigging pins until all bolts have been tightened and all safetying completed.

THE ENGINE FUEL SYSTEM.

The engine fuel system is usually considered to be the heart of the engine. This fuel system performs two essential functions: first, it provides a means for controlling the engine power output; and, secondly, it schedules the fuel to the engine as required for varing engine operation conditions. Power regulation could possibly be accomplished by controlling the air flow at the engine air intake. However, the mechanical problems imposed by such a control system make this type of installation very impractical. Accordingly, the fuel flow rather than the air flow is regulated in the J57 engine.

Modern jet engine design permits the pilot to select a power setting by merely positioning the power lever. The fuel control unit will then schedule sufficient fuel to prevent rich mixture blow-out during acceleration, lean mixture die-out during deceleration, and provide correct fuel flows for varying engine operating conditions during fixed power lever settings.

As mentioned earlier, the engine fuel system consists of an engine-driven fuel pump/transfer valve, the fuel control, fuel flowmeter, fuel/oil cooler, pressurizing and dump valve, and burner nozzles and manifolds. Figure 2-3, shows the entire engine fuel system. In these first few paragraphs you will be given a general description of the fuel system. Then, each component in the system will be discussed in detail.

Note on figure 2-3 that the fuel flows from the fuel pump/transfer valve to the main fuel control unit. This main fuel control unit has both a primary (normal) and an emergency metering system. The primary fuel system meters fuel automatically to the engine during normal operation. This automatic fuel metering is accomplished by the fuel control governor flyweight reaction against the power lever position. The amount of flyweight reaction is dependent upon the N_2 compressor pressure and the pressure in the burner "cans." As a safety feature in the primary fuel system, mechanical takeoff locks are incorporated in the fuel control unit to assure an adequate fuel flow during takeoff in the event that either of the pressure sensing systems fail. The emergency fuel system of the fuel control unit is entirely separate from the primary fuel system. Fuel metering during emergency operations is directly dependent upon the position of the power lever. There is, however, a device for limited altitude compensation in the emergency fuel system.

Referring to figure 2-3 note that the fuel flows from the fuel control unit through the fuel flowmeter, then passes through the fuel/oil cooler into the pressurization and dump valve. Fuel from this valve flows to the fuel nozzle manifolds located in the engine burner section. Each of the eight burner cans incorporate six dual fuel nozzles with primary and secondary openings.

How the Fuel Control Primary Fuel System Operates.

The main fuel control unit is the only component in the engine fuel system which the pilot can directly control. As a result, the fuel control unit is the most

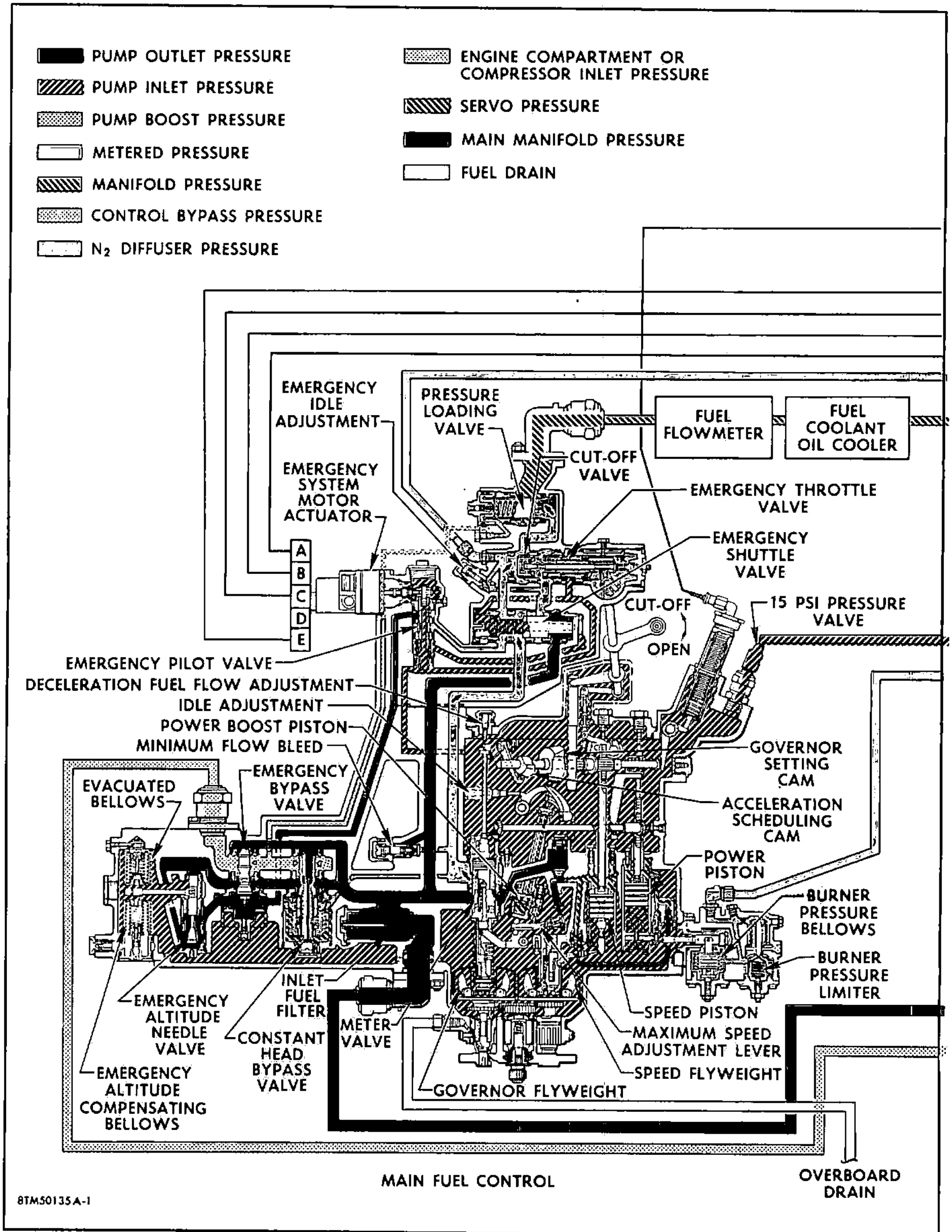


Figure 2-3. J57 Engine Fuel System Schematic (Sheet 1 of 2)

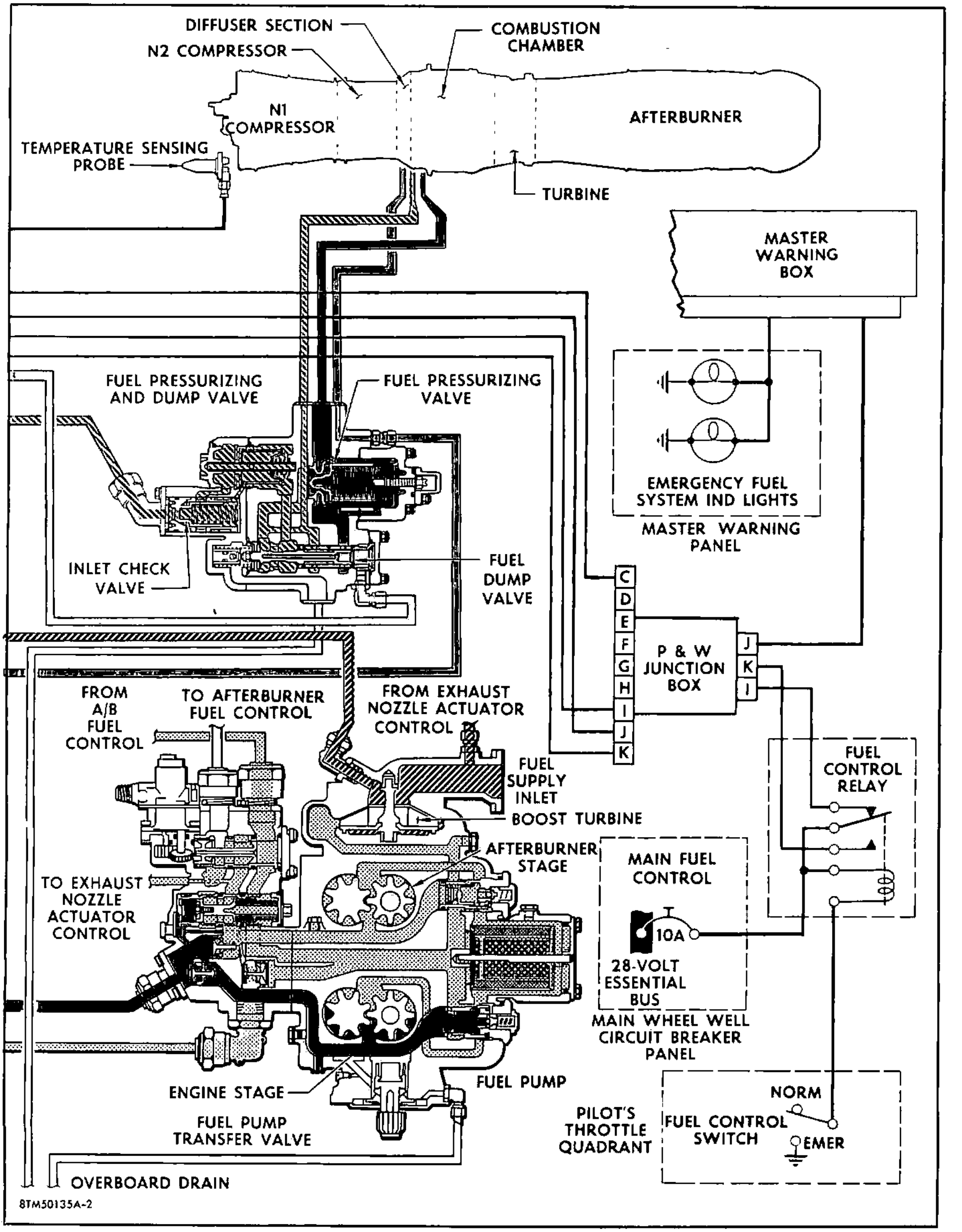


Figure 2-3. J57 Engine Fuel System Schematic (Sheet 2 of 2)

important single unit in the entire engine fuel system. This component provides fuel to the engine burner section at the proper volûme and pressure to maintain the engine performance as demanded by the power lever setting.

To perform these functions properly, the fuel control must be able to: provide acceleration control so that turbine temperatures are not exceeded and a flame-out condition or compressor stall is not experienced; provide deceleration control without lean mixture die-out; maintain a constant power setting regardless of altitude, air temperature, or airspeed; and provide emergency fuel without engine failure in the event the normal engine fuel system fails.

Fuel control units vary with different engine installations; however, by studying one particular type, the basic function of all fuel control units may be understood. The type discussed in this manual is the hydromechanical fuel control unit used on the J57 engine, which is shown in figure 2-4.

The main fuel control unit is mounted on the lower left side of the engine accessory drive housing. This unit is driven at a ratio of 0.344 to 1. Its basic function, as a speed-density fuel control, is to schedule fuel to the burner cans for normal engine operation without exceeding the engine limits. Fuel requirements are determined by the engine operating conditions and the position of the power lever. Thus, the fuel control unit is basically a speed governor that is biased by burner pressure and inlet air temperature.

The following paragraphs explain how the fuel control unit operates. By frequently referring to the schematic of the engine fuel system, (figure 2-3), as you read these paragraphs, you will obtain a better understanding of the main fuel control unit and its functions.

The primary fuel system supplies fuel for normal engine operation. Following the fuel flow from the pressure outlet of the fuel pump/transfer valve, see figure 2-3, you will note that fuel enters the fuel control unit and immediately passes through the spring-loaded fuel inlet filter. If the filter should become clogged, the incoming fuel will force the filter from its seat and bypass unfiltered fuel into the fuel control unit. Downstream from the filter, the fuel flow is divided—one fuel passage leads to the fuel metering valve, the second passage leads to the anti-spring side of the emergency shuttle valve in the upper part of the fuel control unit, and the third passage leads to the constant-head bypass valve. The constant-head bypass valve gets its name from the description of its job—it maintains a constant fuel head (pressure) across the fuel metering valve. The fuel in excess of that required by the metering valve is routed by the bypass valve back to the fuel pump/transfer valve. In the schematic of the engine fuel system, figure 2-3, this return flow is shown in yellow.

Metered fuel enters the hydraulically-balanced emergency shuttle valve (which is held open by spring pressure) and then flows to a cutoff valve. As shown in figure 2-3 this cutoff valve is mechanically operated by the power lever and the valve is always open when the engine is operating. As the fuel flows from the cutoff valve, it passes through a pressure loading valve and then out of the fuel control unit discharge port.

THE MAIN METERING VALVE AND BYPASS VALVE. The metering valve and the constant-head bypass valve combination can be considered the heart of the primary fuel system in the fuel control unit. These two valves determine how much fuel is metered to the engine combustion chambers. The fuel from the engine-driven fuel pump is routed to the metering valve where it flows through a single orifice, or hole, in the metering valve. The size of the metering valve orifice is variable and it is automatically controlled. A constant pressure drop across this orifice is maintained by the constant-head bypass valve.

The metering valve is a sleeve-type valve with two stepped slots (180° apart) and two deceleration slots (also 180° apart) which align with fixed slots in the valve sleeve. The valve sleeve is attached to the fuel control unit body and the valve moves inside the sleeve. The metering valve moves both radially and axially to establish the required metering orifice. In other words, the metering valve turns as well as moving fore and aft. The valve axial movement is obtained by the fore and aft movement of the power lever in the cockpit. This axial movement of the valve tends to open or close the metering valve orifice, depending upon whether the power lever is being advanced or retarded. This *axial* (fore and aft) movement is accomplished through the governor setting cam and attendant linkage by servo action to the governor spring. The power lever motion either increases or decreases the governor flyweight's spring tension and the resultant metering valve position depends on governor flyweight reaction and the governor spring pressure. The metering valve movement is also biased by two other factors: burner pressure acting through the burner pressure servo mechanism causes a *rotary* motion of the metering valve; a restraining action to axial movement is imposed during acceleration by the acceleration governor, cam, and linkage. In addition to these biasing factors, compressor inlet temperatures also control the metering valve position. The varying inlet temperatures result in movement of both the governor setting cam and the acceleration cam. Both of these cams, by means of cam followers and linkage, *axially* position the metering valve.

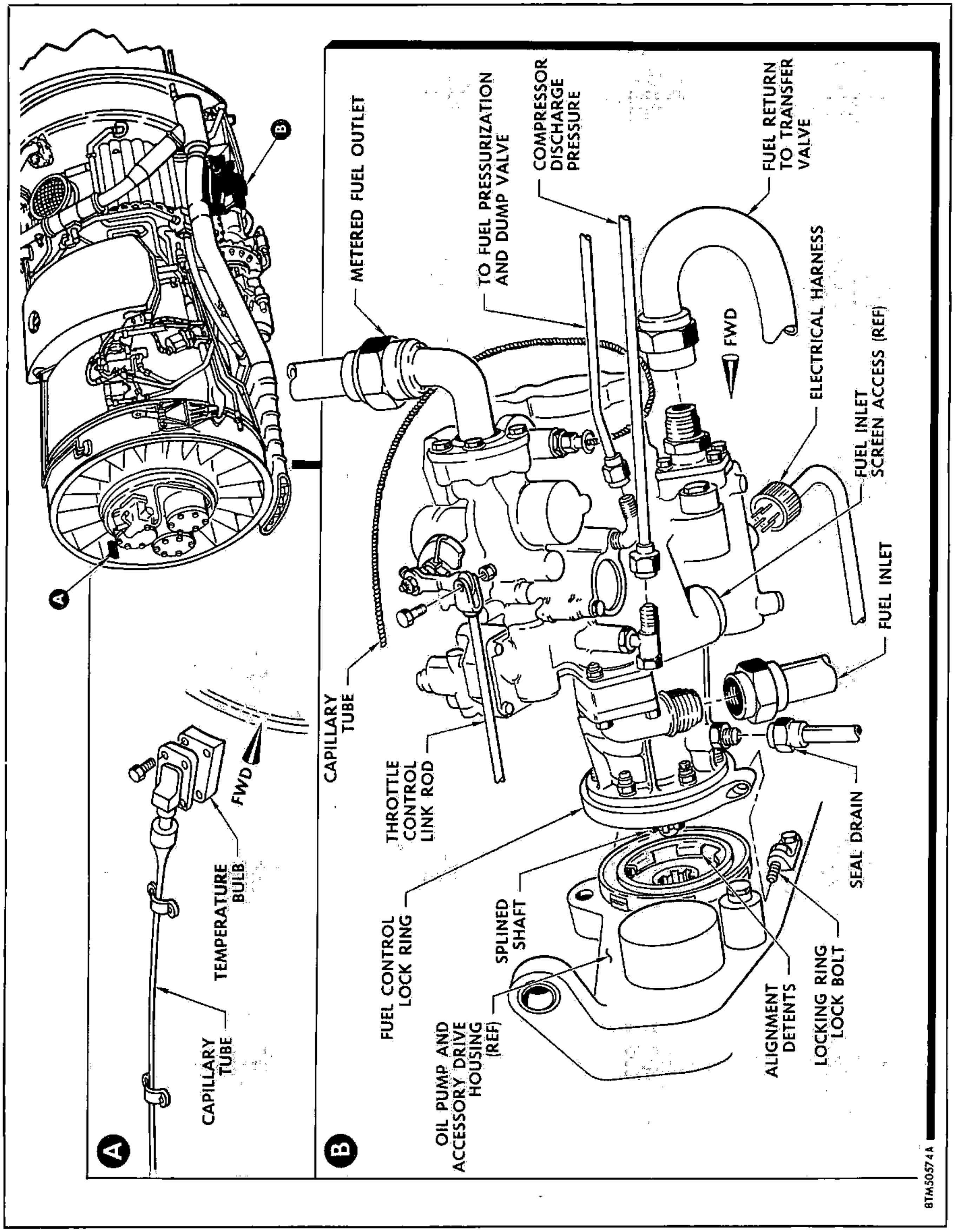


Figure 2-4. Main Fuel Control Unit

As mentioned before, the constant pressure drop across the metering valve is maintained by the constant-head bypass valve assembly. This keeps a constant pressure differential between the unmetered and the metered fuel. If the pressure drop across the metering valve starts to vary from its proper value, the bypass valve either opens or closes to regulate the amount of unmetered fuel that is to bypass the metering valve. The bypass valve accomplishes its job by means of spring pressure. A bellows and hollow-stemmed servo valve maintain the spring at a constant length by varying the bellows length to follow the movement of the bypass valve. This type of arrangement always keeps the same spring pressure on the bypass valve regardless of where the valve is positioned. Referring to figure 2-3, you will note that metered fuel is also routed to the bypass valve. The pressure of the metered fuel combines with the spring pressure to oppose the pressure of the incoming unmetered fuel. The resulting balancing action of the pressures position the bypass valve.

TEMPERATURE COMPENSATION IN THE FUEL CONTROL UNIT. The temperature of the air at the compressor inlet also affects fuel scheduling in the primary fuel system in the fuel control unit. This air temperature compensation is accomplished by mechanical linkage inside the fuel control unit which axially positions the governor setting and acceleration cams. Note in the engine fuel system schematic, figure 2-3, that a temperature sensing bulb is installed in the compressor inlet case. This sensing bulb is connected by a capillary tube to a bellows and spring-loaded piston in the fuel control case. The temperature sensing bulb is actually a liquid-filled bulb. A better view of the sensing bulb and its connecting capillary tube is shown in figure 2-4. The temperature variations of the engine compressor inlet air cause the liquid to expand and contract. This increasing and decreasing pressure of the liquid is transmitted to the fuel control unit by means of the capillary tube. The resulting motion of the bellows and piston caused by the liquid pressure variations moves the cam shaft axially. This axial cam shaft motion causes the cam followers to reposition in respect to the axial contour of the cams. The cam followers are mechanically connected to the metering valve, and their repositioning movement caused by the temperature variations affects the meter valve position.

As mentioned earlier in the description of the fuel control unit, you should never disconnect the capillary tube from the fuel control unit. Each fuel control unit has a temperature bulb and capillary tube attached to it at the factory and the units are matched for proper operation; consequently, the units should never be changed individually.

BURNER PRESSURE COMPENSATION IN THE FUEL CONTROL UNIT. The efficiency of the combustion of the fuel-air mixture in the jet engine combustion chambers is directly related to the pressure inside the combustion chambers (burner cans). The combustion efficiency is highest when the chamber pressure is high, and the efficiency steadily decreases as the chamber pressure drops. The pressure in the turbojet burner cans depends on engine speed, aircraft speed, and altitude. Since the fuel-air mixture depends on the combustion chamber pressure, the fuel control unit uses a sensing system so that the combustion chambers receive the correct amount of fuel for their existing pressures. This sensing system is known as the burner pressure compensation system.

A pressure probe in the No. 4 burner can acts as the sensing element for the burner pressure compensation system. The pressure line from the No. 4 can passes through the pressurization and dump valve housing and enters the burner pressure limiter inside the fuel control unit. In the engine fuel system schematic, figure 2-3, the burner pressure is labeled in the index as N_2 Diffuser Pressure, and this pressure is shown in a blue-green color.

As the pressure in the burner can changes, because of altitude and engine operating conditions, the pressure on the burner pressure bellows also changes. These bellows contract or expand (depending upon whether the pressure is increasing or decreasing) and activate the burner pressure servo. This servo action is transmitted through a rack-and-pinion, a shaft, and bevel gears to produce a *rotary* motion of the fuel metering valve. The result of this action is to either narrow or widen the fuel-metering slots in the fuel meter valve and to affect the primary fuel flow accordingly.

THE INTERNAL CUTOFF VALVE AND FUEL CONTROL BODY PRESSURIZATION. The mechanically-operated cutoff valve stops the fuel flow when the engine is shut down. In figure 2-3, the cutoff valve is shown in the upper portion of the fuel control unit. This valve is connected to the power lever linkage and the valve is open whenever the engine is operating. A return line, connecting the fuel control body to the fuel pump inlet, prevents fuel pressure buildup within the fuel control unit body when the engine is shut down. This line is shown in yellow on figure 2-3. A resultant pressure drop within the fuel control unit body is transmitted to the pressurizing and dump valve by a sensing line, and this pressure causes the dump valve to drain the fuel nozzle manifolds. A 15-pound, spring-loaded valve is incorporated to control the body/pump inlet return line. The resultant fuel control unit body pressure insures proper operation with warm fuel by preventing over-speeding of the internal flyweight speed governor.

INTERNAL PRESSURE LOADING VALVE. Metered fuel from the emergency shuttle valve enters the pressure loading valve chamber directly above the cutoff valve. Passage of the fuel to the control body discharge outlet is restricted by a spring-loaded piston that is backed up with fuel bypass pressure. The pressure exerted by the metered fuel to overcome the pressure on the loading valve insures sufficient pressure for normal servo action.

FUEL CONTROL INTERNAL SERVO SYSTEMS. Servo assist is utilized in the power lever actuating linkage, acceleration cam/speed piston linkage, and burner pressure metering valve linkage. Force reduction of these three linkage systems is essential for proper fuel scheduling. Translation of temperature variation from the governor setting cam to the governor flyweight spring is made more effective through servo action. Servo action also reduces the cockpit power lever load. Translation of burner pressure to the metering valve, and the speed flyweight reaction to the acceleration cam is made possible by the burner pressure servo and the speed servo assemblies, respectively.

The source of pressure for these servo systems is unmetered fuel taken from the metering valve area. The servo pressures are shown in yellow-green color in the schematic of the engine fuel system, figure 2-3. After passing through a filter, the unmetered fuel is ported to the three servo systems and, unless the servo system exit is restricted, the fuel passes out into the regulator case without causing servo action. If, however, a servo system exit is restricted, pressure immediately builds up within the respective system, and the servo piston moves until the exit restriction is removed. Removal of the restriction decreases internal pressure and stops servo piston motion. Actually, the restrictions of the servo exits are caused by half-balls being positioned over the exit. This positioning of the half-balls in each of the three servo systems determines the degree of servo action. When a half-ball restricts a servo pressure exit, the resultant servo action will unseat the half-ball (through mechanical linkage) after traveling a predetermined distance. This action may be traced by careful examination of the servo systems in the fuel system schematic, figure 2-3.

TAKEOFF LOCKS. A pair of mechanical locks is incorporated in the fuel control to insure adequate fuel flow in the event of either burner pressure or inlet temperature sensing system failure. During takeoff these ratchet-type locks are engaged mechanically when the power lever is advanced from FULL MIL POWER position to the TAKE OFF position. Theoretically, no power increase results during this power lever advancement, because the governor setting cam follower is following the cam "flat" during this operation. A rise in power does, however, occur from 54° power lever setting to the takeoff locks, or approximately 1½% military power increase. One locking lever is positioned adjacent to a collar on the cam shaft to prevent excessive shaft travel in the event of temperature compensating system failure. The second lock is positioned to prevent the burner pressure servo rack from closing the metering valve orifice below the predetermined position. Excessive engine overspeeding is prevented by governor flyweight action, which limits metering valve axial motion.

After the takeoff has been accomplished, it is mandatory that the power lever be returned to the FULL MIL POWER position to remove the takeoff locks. If the locks are permitted to remain in the takeoff position, they will severely limit both the inlet temperature and the burner pressure compensation that is necessary for proper fuel scheduling at high altitudes. Should either or both of the "locked" systems fail during takeoff, a slight power reduction may be noted. The amount of power fluctuation noted will depend on ambient air and altitude. If a component had failed, a more noticeable and possibly severe power reduction would be noted when the power lever releases the locks as it is brought back to the FULL MIL POWER position. Under this condition it will be impossible to re-engage the locks by repositioning the power lever back to the TAKE OFF position. Under normal operating conditions the power lever should be moved back from the TAKE OFF position before reaching 6000 feet.

How the Emergency Fuel Control System Operates.

The emergency fuel control system operates independently of the primary (normal) fuel control system. It is important to note that there is, for the present at least, no provision for automatically switching from the primary operating system to the emergency system. Instead, the emergency system can be activated only by actuating the emergency switch in the cockpit. A motor-operated rack-and-pinion drive positions the emergency pilot valve. In the normal position, the emergency pilot valve insures that the primary fuel control system output passes the emergency shuttle valve and is discharged in the conventional manner to the fuel pressurizing and dump valve. The emergency shuttle valve is hydraulically balanced during primary (normal) fuel scheduling. Spring pressure moves the shuttle valve to shut off the fuel passage to the emergency throttling valve, and the metered fuel is ported to the cutoff valve. When the cockpit emergency switch is actuated, the emergency system motor repositions the emergency pilot valve so that fuel pressure is ported to the anti-spring side of the emergency shuttle valve. The emergency shuttle valve then moves against the spring pressure, opens the passage for unmetered fuel to enter the emergency throttling valve, and also shuts off the metered fuel passage to the cutoff valve.

Since emergency fuel operation completely isolates the primary or normal fuel regulating components, an emergency throttle valve, emergency bypass valve, emergency altitude needle valve, and an emergency altitude compensating bellows (working in conjunction with an evacuated bellows) take over the fuel scheduling duties. Emergency fuel metering is achieved by varying the emergency throttle valve. The linear travel and flow through this valve are in turn varied by moving the valve's fixed orifice in relation to a fixed, contoured needle and an idle bleed, which operates to maintain minimum flow. A constant pressure drop is maintained by the emergency bypass valve. This valve is similar to that used in the primary or normal system, except that there is no compensation for varying spring length. Instead, the emergency bypass valve operates in conjunction with the emergency altitude needle valve to establish the desired metering load across the emergency throttle valve. The emergency altitude needle valve is operated by linkage from the emergency altitude compensating bellows. Accordingly, altitude variations will either increase or decrease the fuel flow to the emergency throttle valve.

During emergency operation, fuel scheduling is a *direct function* of power lever movement and no provisions are made to alter the fuel flow during acceleration, deceleration, or for variations of temperature and burner pressure. Because of this, the pilot must observe rpm, tailpipe temperature, and pressures very closely during emergency operation. Limited altitude (approximately 30,000 feet) compensation, however, is provided by action of the emergency altitude compensating bellows and the emergency altitude needle valve.

Flight Line Adjustments on the Fuel Control Unit.

Almost all of the adjustments that can be made on the fuel control unit must be done on a flow bench by fuel control specialists. The only adjustments which you will be allowed to make on the flight line are the IDLE and MAXIMUM flow adjustments. These two adjustment points are shown in the upper portion of figure 2-5. Each of the adjustment screws are designed with spring-loaded detent balls to provide click-lock adjustment. Fourteen clicks equal one full turn of the adjustment screws.

Turning the IDLE adjustment screw in a clockwise direction increases the preload on the governor spring through the cam follower, increasing IDLE fuel flow. Turning the screw in the counterclockwise direction, of course, decreases IDLE fuel flow.

The MAXIMUM adjustment screw repostions the maximum rpm lever inside the fuel control unit. This varies the effective length of the governor cam follower-rod. Turning the MAXIMUM adjustment screw clockwise increases the fuel flow. Because of the rather complex mechanical linkage between the adjustment screw and the maximum rpm adjustment lever, it is advisable to make four turns beyond the desired setting point when increasing the trim; then back off to the desired point. This method of first turning past the point and then back will eliminate the built-in backlash in the complex mechanical linkage. When reducing the MAXIMUM trim, however, it is necessary only to turn directly to the desired setting. The later model fuel control units will have a revised, spring-loaded, mechanical linkage that will eliminate the need for turning beyond the desired setting point and then backing off.

Fuel Pump/Transfer Valve.

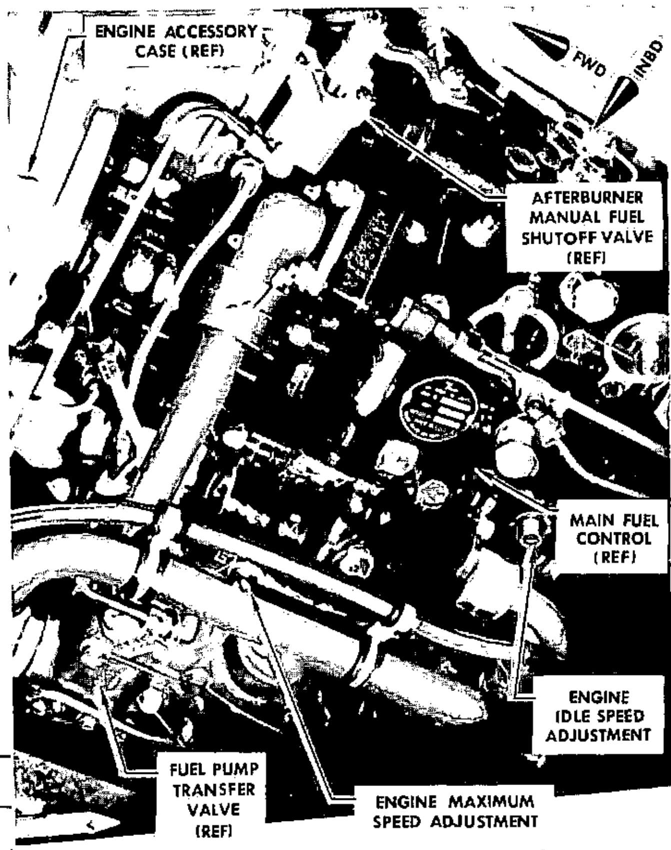
A three-stage engine-driven fuel pump, located on the right side of the engine wasp-waist section, provides fuel to both engine and afterburner (A/B). The fuel pump is shown schematically in figure 2-3. The first stage in the pump is a centrifugal boost turbine at the fuel inlet. Fuel from this turbine flows through a spring-loaded screen to the inlet side of the two gear stages. Both gear stages and the turbine stage are driven from a common shaft at a ratio of 0.344 to 1.0 and incorporate individual shear sections. Basically, one gear stage supplies fuel to the engine fuel control, the other supplies fuel to the A/B fuel regulator and associated units. Each gear stage has a pressure relief valve which limits the pressure rise to approximately 750 to 775 psi. A low-pressure warning light (in the cockpit) is connected to the engine gear stage outlet passage. This warning light system will be discussed later in Chapter IV.

As you will note in figure 2-6, a transfer valve is attached to the engine-driven fuel pump body. The primary purpose of the transfer valve is to provide fuel passage for engine operation and to supply fuel to A/B fuel components when afterburning is desired. During non-afterburning operations the normal A/B outlet is blocked by a motor-operated fuel shuttle valve. The fuel output from the A/B gear stage is then bypassed back to the gear inlet. When A/B operation is initiated, the motor-driven fuel shuttle valve opens both the fuel supply line to, and the return line from, the A/B fuel regulator.

An automatic fuel-regulating transfer valve, located between the A/B gear stage and the motor-driven shuttle valve, insures fuel flow to the engine fuel control in the event of engine gear stage failure. This two-position regulating transfer valve has engine gear stage fuel pressure bled to one side of the valve; this pressure opposes A/B fuel regulator bypass pressure and spring pressure on the other side. During normal operation the engine gear stage fuel pressure raises the valve against spring pressure and A/B fuel regulator bypass pressure, this permits the A/B gear stage

ENGINE TRIM PREPARATION

A. NO INLET DUCT SCREENS INSTALLED.
B. REFRIGERATION AIR "ON" (N_2) BLEED AIR).
C. CONSTANT SPEED DRIVE UNIT LOAD MAY BE EITHER ON OR OFF.



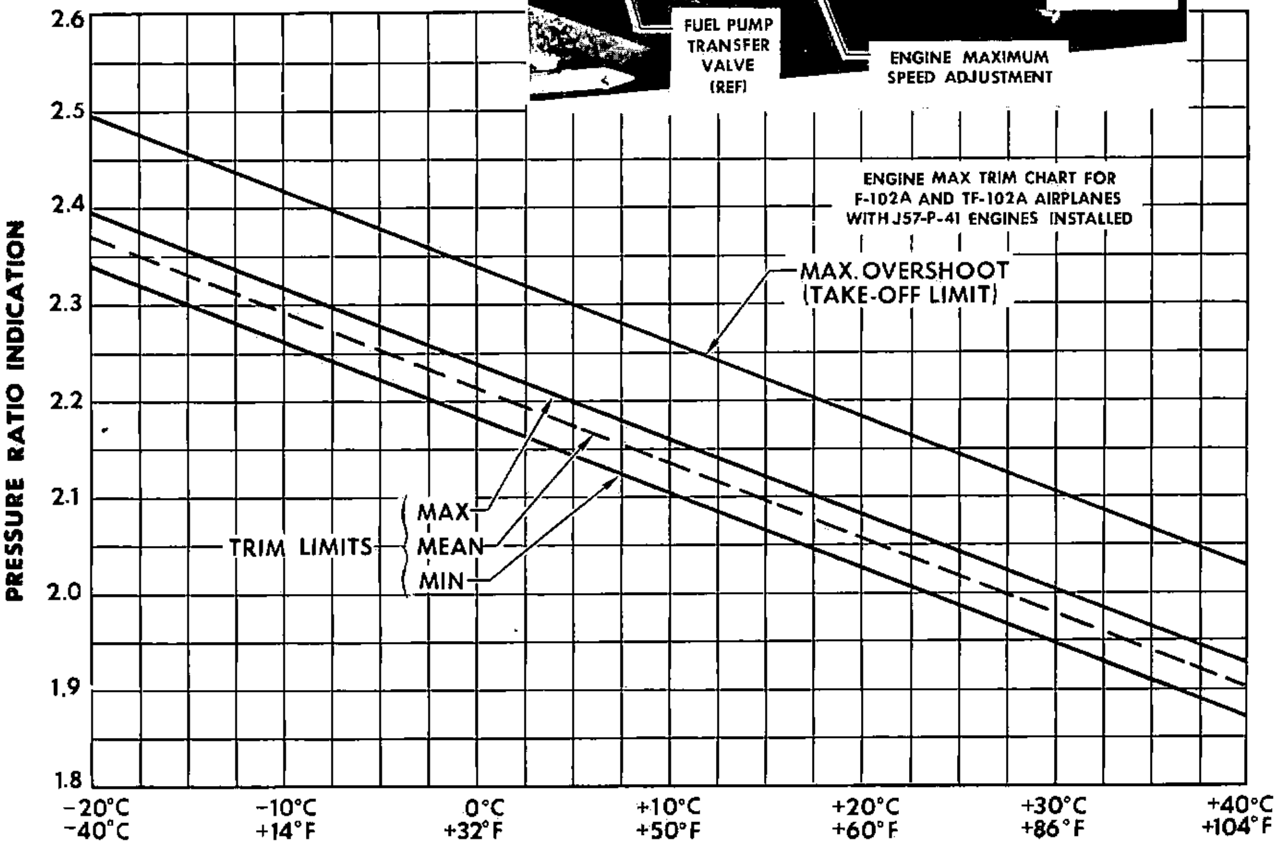


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Figure 2-5. Adjustment Points on the Main Fuel Control

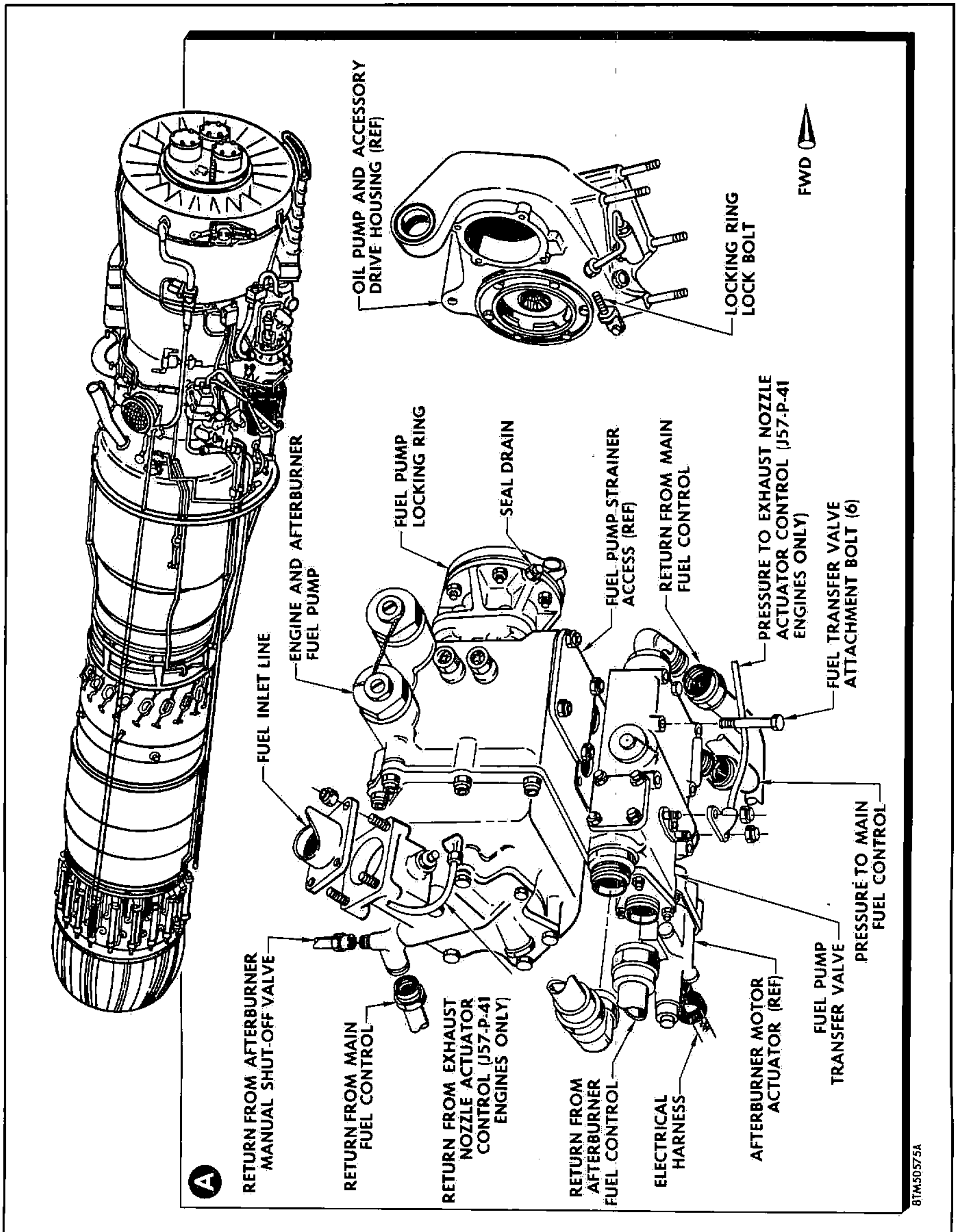

Figure 2-6. Engine-Driven Fuel Pump/Transfer Valve

fuel to flow to the motor-driven fuel shuttle valve. Should the engine gear stage fail, however, the resultant fuel pressure drop will cause the spring-loaded fuel regulating valve to move to block off the A/B gear stage outlet. Pressure building up in this line will unseat the spring-loaded check valve and permit fuel flow into the engine fuel regulator supply line, this prevents engine fuel starvation.

If the afterburner is operating and the engine gear stage fails, the A/B fuel supply will be limited or shut off entirely. Earlier airplanes did not incorporate provisions to supply fuel to the A/B during the emergency action noted. Later airplanes, however, will pass sufficient fuel through drilled passages in the transfer valve to permit limited afterburning.

The power lever quadrant in the cockpit has an A/B detent which parallels the regulator power setting slot through a range of 80% to 100% thrust for engines with A/B shutoff valves. Placing the power lever in this A/B detent actuates a microswitch in the A/B actuator motor circuit, and the afterburning cycle is initiated. When the power lever is taken out of the A/B detent, the A/B actuator motor is reversed and fuel flow to the A/B fuel regulator is stopped. Fuel scheduling for engine operation and for afterburning utilizes two separate sets of components downstream from the pump/transfer valve. The discussion immediately following will concern only those units used during engine (non-afterburning) operations.

Fuel Pressurizing and Dump Valve.

The metered fuel from the fuel control unit passes through the flowmeter and the fuel/oil cooler to the pressurizing and dump valve. As you will note in figure 2-7, this pressurizing and dump valve is mounted on the engine fuel manifold just aft of the fuel control unit.

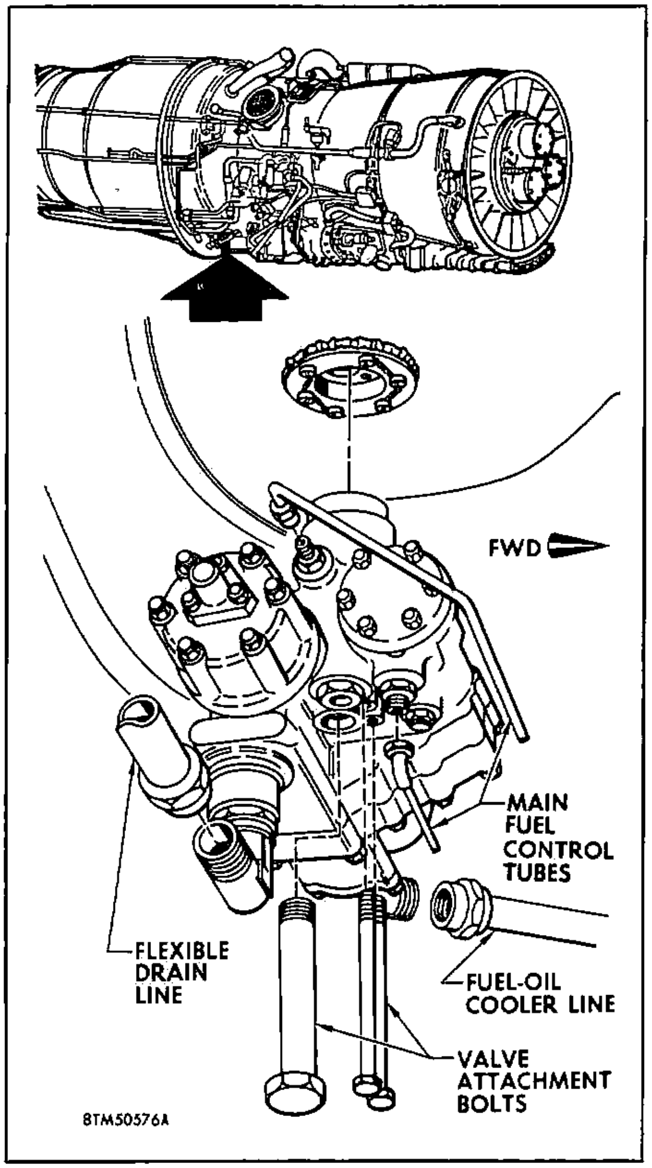

Figure 2-7. Fuel Pressurizing and Dump Valve

The pressurizing and dump valve accomplishes two functions—it controls the fuel flow to the primary and secondary injector nozzles in the engine burners, and it also dumps the nozzle manifold fuel when the engine has been shut down. When the engine is operating, fuel enters the unit through the inlet spring-loaded check valve and passes through a 200-mesh screen. Tracing the fuel flow on the schematic, figure 2-3, note that the fuel flows through the screen to the pressurizing valve. Fuel pressures from the fuel control unit body are routed by means of a sensing line (shown in brown) to the anti-spring side of the dump valve—this sensing pressure keeps the dump valve closed while the fuel passes on to the pressurizing valve in the unit.

During engine starting and low-power operation, all of the fuel (shown in green) flows directly to the pilot (primary) fuel manifold. But as the fuel pressure increases, the pressurizing (flow dividing) valve is unseated. This causes a part of the fuel (shown in blue) to flow to the main (secondary) fuel manifold. As the engine power increases, more fuel flows to the main fuel manifold.

When the power lever is brought back to the OFF position, the flow of fuel to the pressurization and dump valve is stopped. As the fuel pressure inside the fuel control unit body drops, the pressure in the sensing line also drops. The spring pressure on the dump valve overrides the decreased fuel control unit body pressure and forces the dump valve open. This drains the residual fuel from the manifolds and prevents any more fuel from entering. The spring-loaded inlet check valve in the pressurizing and dump valve prevents the fuel in the fuel/oil cooler from "boiling off" and dumping when the engine is shut down.

Engine Fuel Manifold.

The shrouded dual fuel manifold inter-connects the 48 dual fuel nozzles. This manifold is built up as one complete assembly. Fuel metering for the engine fuel system terminates at the burner cans—each of the eight cans has six dual nozzles. The pattern of fuel discharge in the burner cans is accomplished by the nozzle design, by air flow within the nozzle heads, and by swirl vane openings in the nozzles.

Maintenance on the Engine Main Fuel System.

The maintenance and trouble shooting requirements for the J57 engine fuel system are somewhat more complex than the requirements for piston-type engine fuel systems. As you have already learned in the preceding discussion of the engine main fuel system, each component in the fuel system does several jobs. As a member of the F-102A ground maintenance team, it will be your responsibility to assure that each component adequately performs its jobs, and that all components function together to supply fuel to the engine as demanded.

In all probability, your most difficult task in maintaining the engine main fuel system will consist in diagnosing engine malfunctions and than trouble shooting the fuel system to determine which component is causing the trouble. Although a trouble shooting guide is included in the engine maintenance technical order, some of the more common difficulties and their most probable causes are discussed in the following paragraphs.

About 90% of the engine fuel system troubles will be traceable to the fuel control unit. This should be understandable since the fuel control is a very complex system component. As you will recall, the fuel control unit should provide acceleration and deceleration control; maintain a constant power setting regardless of altitude, temperature, or air speed; and provide emergency fuel without engine failure in the event the regular engine fuel system should fail. Whenever one of these functions is not performed, you can almost be assured that the fuel control unit is malfunctioning. Since you are not allowed to perform any maintenance other than the IDLE and MAXIMUM flow adjustments on the fuel control, you must replace it with a serviceable control unit and send the defective control unit to the overhaul area for a bench check.

Referring to figure 2-4, you will note that the installation and removal of this engine fuel system component will be relatively easy. Before attempting to remove the unit from the engine accessory drive housing, you should disconnect all of the fuel, sensing, and electrical lines, and then provide some type of support for the unit. The capillary tube which connects the fuel control unit to the temperature bulb in the engine inlet must not be disconnected at the fuel control unit. The bulb and the connecting capillary tube must be removed with the fuel control unit. The reasons for this will be discussed later in this section. Note the locking ring and lock bolt directly under the attach point on the accessory drive housing. You will normally find it best to loosen the lock bolt until it begins to snug up, and then tap the bolt head until the bolt becomes loose. Keep doing this until the fuel control lock ring is free of the accessory drive housing flange. If the fuel control unit is to be left off for any time longer than a few minutes, be sure to cap all of the disconnected lines.

The installation procedure for the fuel control unit is essentially the reverse of the removal procedure; however, you should always coat all of the seals and gaskets with engine lubricating oil prior to installing the unit.

Other fuel system malfunctions which you might encounter will possibly occur in the emergency fuel system. The fuel control unit, as you will recall, also has a complete emergency metering system. If there is no emergency fuel flow at IDLE, replace the fuel control unit. Although there is no specifiially recommended procedure for checking the emergency fuel flow, some helpful suggestions for determining whether the emergency fuel system is functioning properly are as follows: (1) carefully watch the tachometer for rpm change as you actuate the emergency switch, (2) watch the tailpipe temperature indicator for any change in temperature, and (3) watch how soon the red light that indicates emergency fuel comes on after you actuate the switch. In those cases where the engine will not change over to the emergency fuel system or the fuel control system selects the emergency system with the control switch in the NORM position, you will probably find the difficulty in the electrical circuit to the emergency system actuator. By performing a continuity check on the circuit from the 28-vole d-c bus to the actuator, you will be able to determine whether the malfunction is in the actuator itself. The most logical sources of difficulties of this type are the fuel control relay on the forward side of the right main wheel well or the emergency system actuator on the fuel control unit. When a malfunction is traced to either of these units, the defective unit must be replaced with a serviceable item.

FILTERS IN THE ENGINE MAIN FUEL SYSTEM. In the discussion of the engine main fuel system, it was mentioned that several of the components incorporated filters to assure clean fuel to the burner chambers. As you will recall, each of these filters is spring-loaded to bypass fuel in the event the filter becomes clogged. The filters in the system are of the

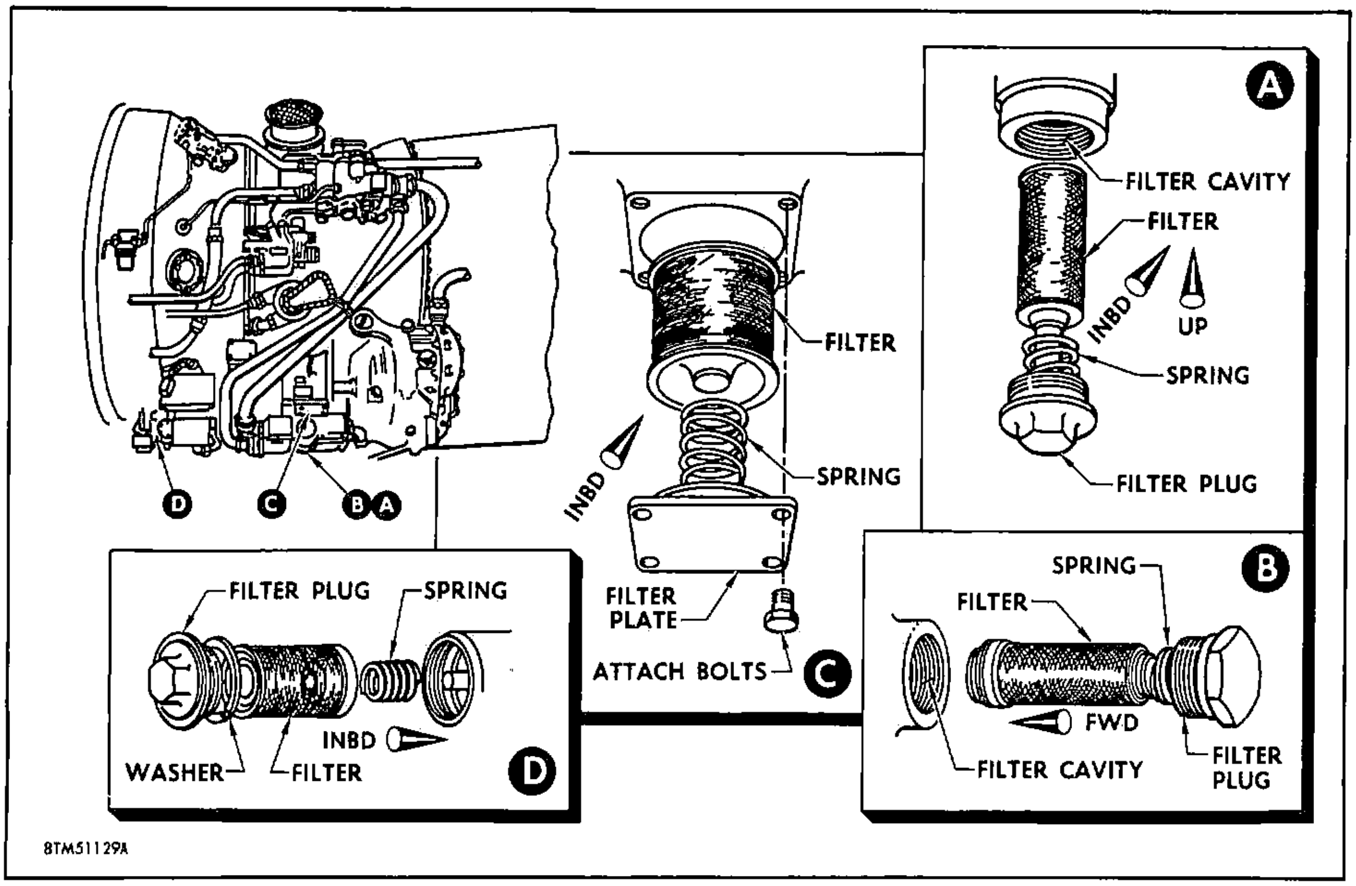

Figure 2-8. Fuel Filter in the Engine Fuel System—Schematic

200-mesh variety; that is, they are designed to restrict the passage of those particles which are larger than normally found in fuel. As you may already know, military specifications allow for some contaminants in jet fuels. The components in the engine fuel system are designed to function properly while handling fuels which do have contaminants in them; however, the components will not function properly if the size of the contaminants exceeds the size of the openings in the filters.

Cleaning the filters will be a part of the normal engine inspection activities. There are four filters in the engine main fuel system. As you will note in figure 2-8, they are located in the fuel control, fuel pump, and the pressurizing and dump valve. Although the filters do not all look alike, they perform the same function; and they are removed, cleaned, and replaced in the same manner. Note that each filter is held in place by a retaining plug.

To remove the filter, cut the safety wire and then unscrew the plug. All of the plugs have conventional right-hand threads. If the filters are not damaged in any manner, they can be cleaned by using either JP-4 fuel (MIL-F-5624), or naphtha, and then blowing the filter dry with clean, dry air. Installation of the filters for the engine fuel system is essentially the reverse of removal. Small holes in the heads of the plugs allow them to be safetied after they have been installed and tightened. You should always assure that the filter retaining plugs are tightened and safetied to eliminate the danger of fuel leaks and subsequent fire hazards.

THE AFTERBURNER FUEL SYSTEM.

The design of gas turbine engines limits the acceleration of jet aircraft during ground run, takeoff, and climb. This is due to the gas turbine engine's nearly constant thrust at all airspeeds at a given altitude and engine speed. In a turbojet engine, for example, the takeoff thrust may not be more than 15 or 20% above the normal rated thrust. This inherent design deficiency led to the development of thrust augmentation; thrust augmentation is merely a means of increasing the engine thrust to provide more power for takeoffs, climbs, or any other time when a rapid rate of acceleration is needed.

The J57 engine utilizes an afterburner as its means of thrust augmentation. As you learn about the afterburner fuel system in this section, you will see that the afterburner can be described as a ram-jet engine that is attached to the turbine section of the regular engine. The afterburner fuel system and the system

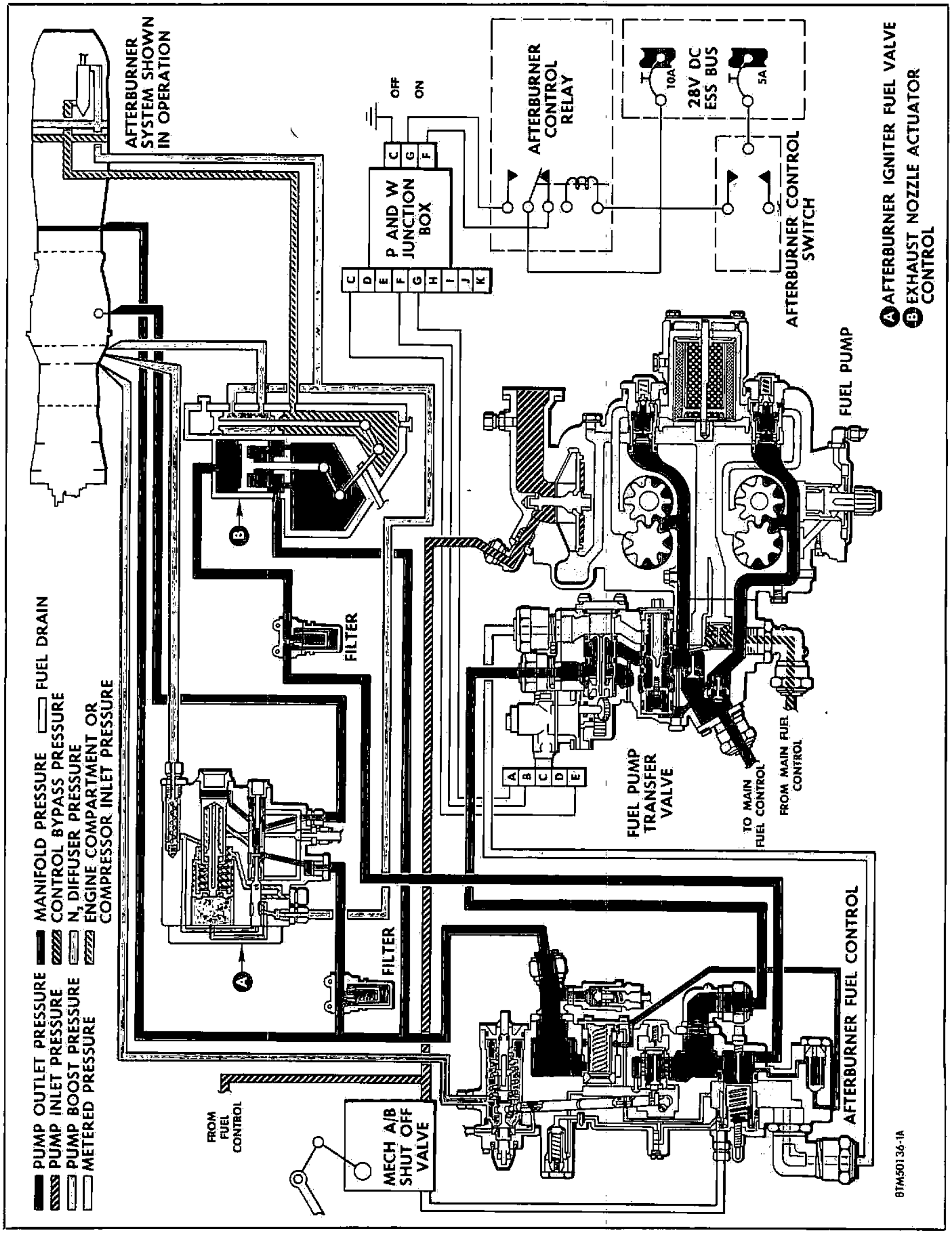

Figure 2-9. J57 Engine Afterburner Fuel System—Schematic (Sheet 1 of 2)

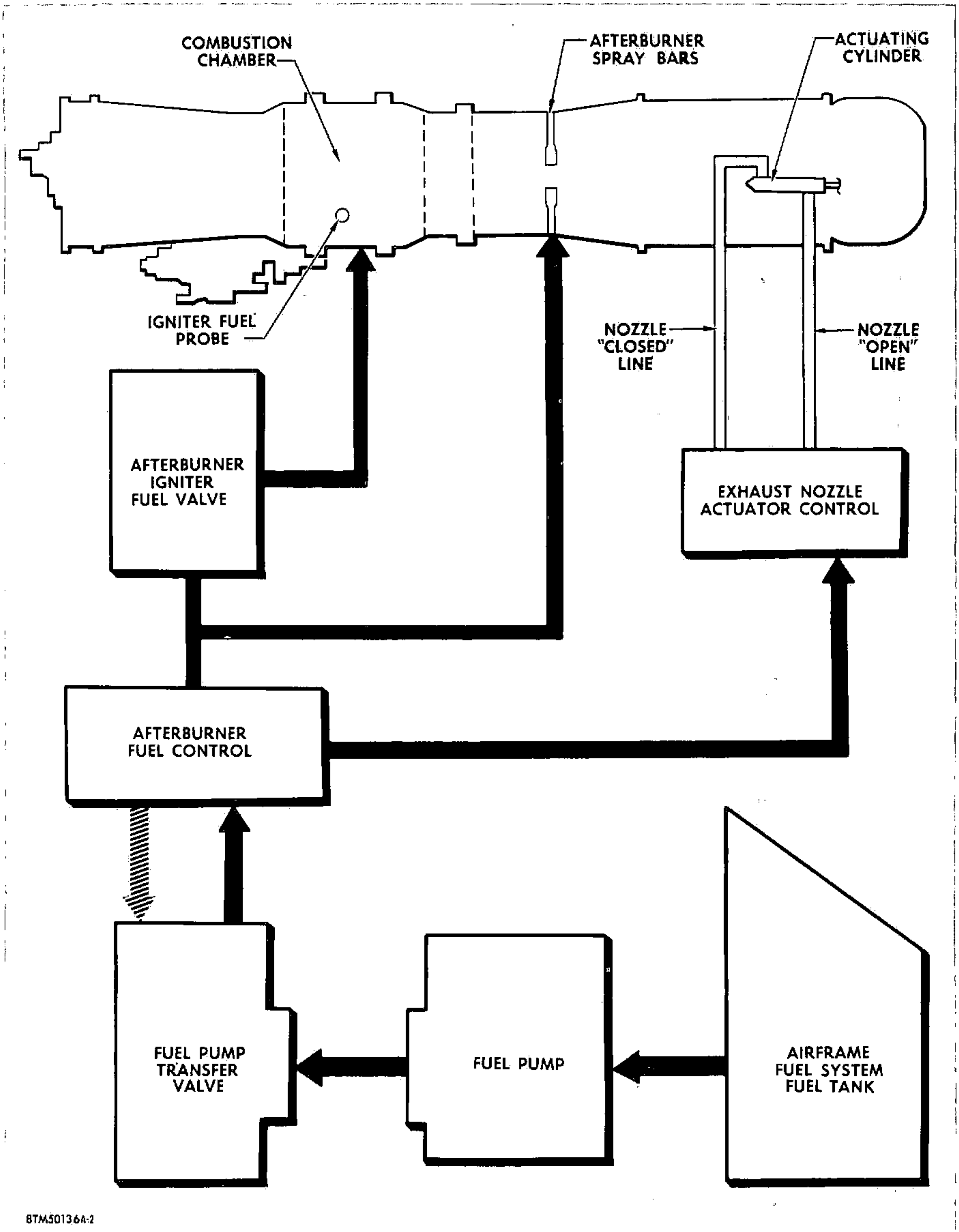

Figure 2-9. J57 Engine Afterburner Fuel System—Schematic (Sheet 2 of 2)

components are covered in the following paragraphs. By referring to the schematic of the system in figure 2-9, while reading the text, you will acquire a more comprehensive understanding of the afterburner fuel system.

Operation of the Afterburner Fuel System.

As mentioned earlier, in this section, the afterburner fuel system consists of the fuel pump/transfer valve, an afterburner fuel regulator, afterburner igniter valve, exhaust nozzle control valve, a manual shutoff valve, and the fuel spray bars and manifolds.

Afterburner fuel scheduling utilizes one gear stage of the engine-driven fuel pump. The pump outfit is either routed back to the gear stage input or flows to the afterburner fuel control. Note in the schematic that afterburner fuel scheduling depends on the position of the fuel regulating transfer valve and/or the motor operated fuel shuttle valve. During afterburning, fuel flows to the afterburner fuel control which then schedules fuel to the A/B spray bars according to the engine burner pressure. Fuel in excess of that required by the A/B system returns to the pump/transfer valve through the control pressure regulator return line.

The output of the afterburner fuel regulator flows to the 24 afterburner spray bars. A separate line from the afterburner igniter valve also connects into the spraybar supply line to supply fuel for "hot streak" ignition and to "trigger" the igniter valve. Fuel from the igniter valve is routed to the No. 3 burner can and the resulting momentary enrichment causes the flame to extend through the turbine wheel and back to the A/B spray bars. This flame ignites the fuel at the afterburner spray bars.

A small pressure sensing line connects the unmetered fuel side of the A/B fuel regulator to the anti-spring side of the exhaust nozzle control valve (this sensing line is shown in red). When A/B is selected, fuel pressure actuates the exhaust nozzle control valve which then directs sixteenth-stage compressor air to the *open* side of the nozzle actuating cylinders. When afterburner operation is stopped, the sensing line pressure is reduced and spring tension in the exhaust nozzle control valve repositions the shuttle valve. Then sixteenth-stage air pressure is ported to the nozzle *close* side of the actuating cylinders.

Normally, afterburner operation is initiated and terminated electrically. However, a mechanically-operated valve—attached to the power lever linkage—stops afterburner operation whenever the power lever setting is reduced below 80%. This provides a positive means of stopping afterburning in the event of an electrical failure in the afterburner circuit.

AFTERBURNER FUEL REGULATOR. The A/B fuel regulator, shown in figure 2-10, is located on the right side of the engine in the "wasp waist" section. Its purpose is to schedule fuel to the A/B spraybar manifold. This A/B fuel regulator control unit incorporates: a variable area-metering valve; a pressure regulator; an automatic shutoff valve; a burner can pressure sensing bellows; a pair of filters; and a drain valve.

Figure 2-9 shows the fuel flow through the regulator. Note that fuel entering the A/B fuel regulator control unit is routed to the metering valve—the excessive fuel is bypassed through the pressure regulator and routed back to the transfer valve. The metering valve orifice area varies in relation to the engine burner pressure. The bypass valve (pressure regulator) maintains a constant pressure drop across the metering valve by using spring and metered fuel pressure to resist unmetered fuel pressure. This arrangement is similar to the bypass valve in the engine fuel control unit. A check valve, having spring and bypass pressure resisting metered fuel, is located downstream from the metering valve. This valve serves a dual purpose: it keeps the A/B fuel regulator full; and prevents burner gases from entering the fuel regulator when the A/B is shut down. The burner can pressure-sensing bellows assembly in the A/B fuel control unit consists of a spring-loaded evacuated bellows linked to the metering poppet valve. Accordingly, burner pressure variation is reflected in the metering valve, and the metering valve then varies the A/B fuel with respect to engine operation.

Other components incorporated in the A/B fuel regulator are two filters and a manifold drain. The two filters provide filtered fuel to the sliding surfaces of the pressure regulator and shutoff valves. This filtered fuel provides a "washing action" which insures free movement. The A/B manifold drain valve is located in the regulator fuel outlet fitting and is slightly spring-loaded to the open position. During A/B operation, fuel pressure closes the valve; but when the A/B is shut down, the valve opens and drains the A/B fuel manifold. Normal engine operating pressures will also close the valve to prevent escape of hot gases.

AFTERBURNER FUEL IGNITER VALVE. As mentioned earlier, the ignition of A/B fuel is accomplished by the "hot streak" ignition method. The igniter valve is located high on the right side of the wasp waist section, as shown in figure 2-11. When A/B is initiated, the igniter valve directs a small charge of fuel into the No. 3 burner can. This enrichment results in a flame pattern that extends into the A/B area and ignites the fuel that is being discharged from the A/B spray bars. As shown in the schematic of figure 2-9, the igniter unit is made up of: an air pilot valve, a fuel pilot valve, and a spring-loaded air piston working in conjunction with a fuel piston.

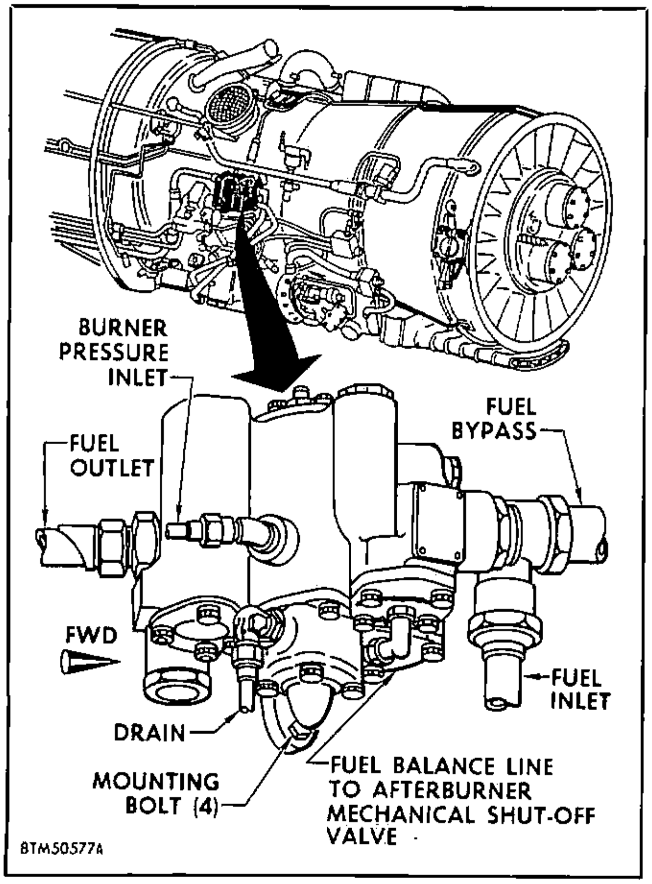

Figure 2-10. Afterburner Fuel Regulator

A line leading to the A/B igniter valve is connected into the A/B fuel manifold line downstream from the A/B fuel regulator. Fuel from this source passes through a filter and enters the igniter. During this charging action, fuel fills the fuel chamber, and at the same time, fuel is bleeding through a restricted passage into the fuel pilot valve. When the pressure becomes great enough, the fuel pilot valve moves the air pilot valve against burner pressure. When the fuel pilot valve moves, it shuts off the supply of incoming fuel and opens the fuel chamber outlet. The same movement causes the air pilot to open the burner pressure passage to the air piston. Burner pressure acting on the air piston compresses the air piston spring and forces the fuel behind the fuel piston into No. 3 burner can. The total afterburner ignition action takes approximately one-half second to complete and is a non-repeating action. Each component remains in the discharged position until the A/B operation is stopped. When A/B is discontinued, the fuel pressure—which was resisting engine burner pressure—drops off, and the burner pressure then moves the air pilot valve and fuel pilot valve back to the charge postion. This action shuts off burner pressure to the air piston, and the piston spring pressure returns the air piston back to its charged position. A check valve is installed in the afterburner igniter discharge line to prevent any hot gases from No. 3 burner can from entering the igniter valve during engine operation.

EXHAUST NOZZLE CONTROL VALVE. The A/B exhaust nozzle control valve, located just below and aft of the A/B fuel regulator, is shown in figure 2-12. Its function is to divert sixteenth-stage air pressure to either the OPEN or CLOSE side of the nozzle actuating cylinders. The afterburner nozzle is opened for afterburner operation and closed during non-afterburning operations. One of the key devices in the A/B nozzle actuating system is the air relay valve inside the exhaust nozzle control. A fuel pressure-sensing line connects the A/B fuel outlet passage in the pump/transfer valve to the anti-spring side of the air relay valve in the exhaust nozzle control valve. This pressure-sensing line is shown in red on the schematic of figure 2-9. On later airplanes, the sensing line source is from the A/B fuel regulator. This later configuration is not shown in figure 2-9. The air relay valve is positioned either by A/B fuel pressure or by spring pressure. During A/B operation the A/B fuel pressure moves the air relay valve against the spring pressure, and sixteenth-stage air pressure is then ported to the nozzle actuating piston open side of the nozzle actuating cylinders. Two spring-loaded flapper-type vent valves exhaust the discharge air from the relay valve. Normal fuel leakage

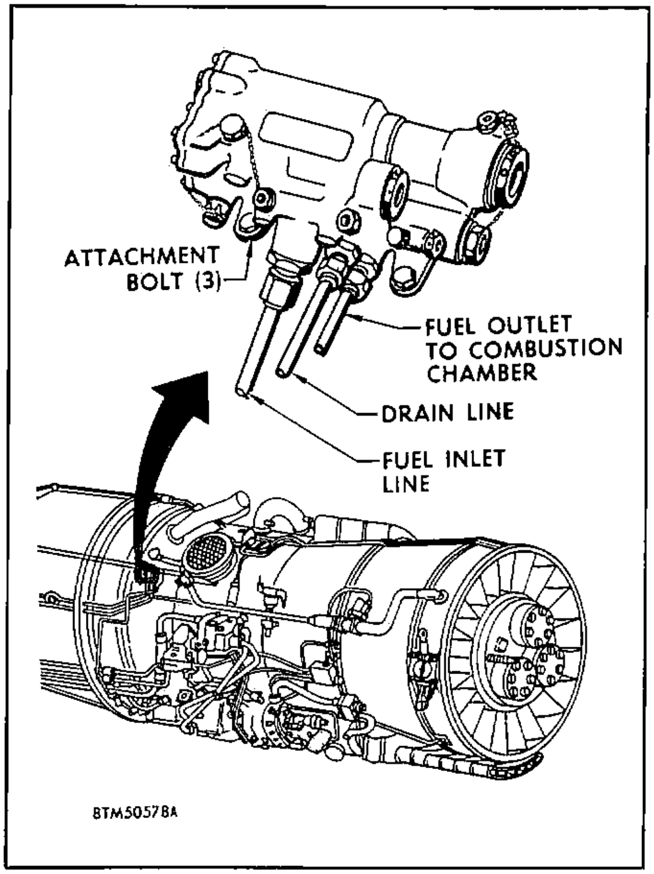

Figure 2-11. Afterburner Fuel Igniter Valve

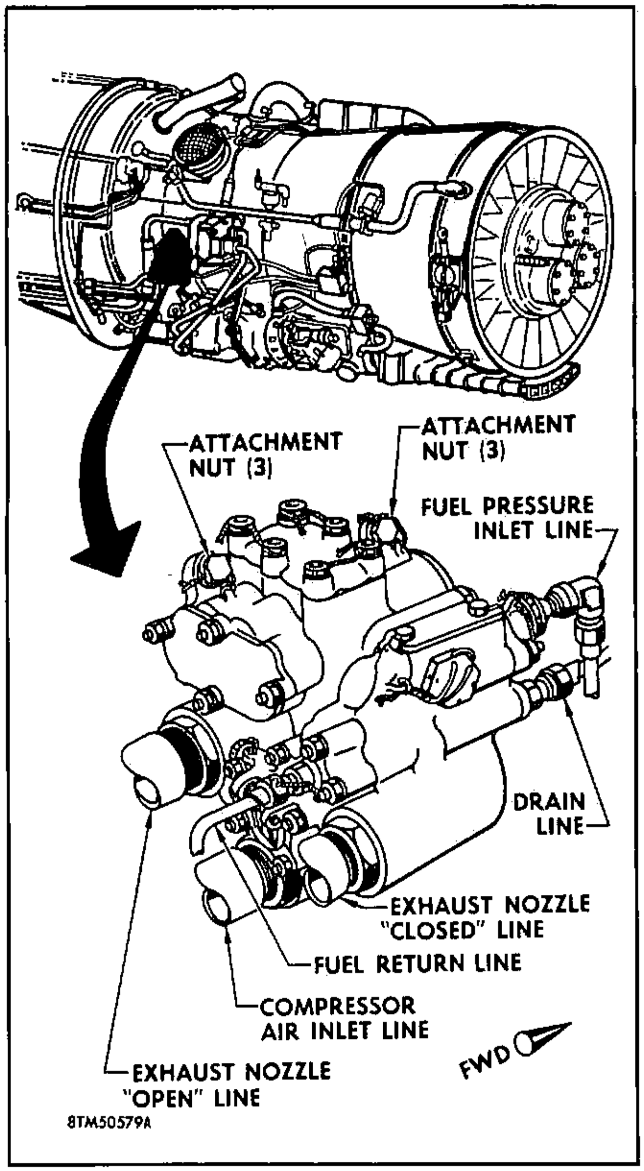

Figure 2-12. Exhaust Nozzle Control Valve

is drained overboard. The later engines are equipped with a differential-type exhaust nozzle control valve—the fuel section is separate from the air section. Refer to the maintenance technical order for information on these later-type control valves.

AFTERBURNER MECHANICAL SHUTOFF VALVE. Normally the A/B operation is stopped when the power lever in the cockpit is taken out of the A/B detent. This action energizes the A/B motor actuator which in turn closes the A/B fuel shuttle valve.

A fuel line that incorporates a mechanically-operated shutoff valve is installed in the A/B fuel system. This line (shown in yellow) at the left of the schematic, connects the spring side of the pressure regulator in the A/B fuel regulator to the fuel pump inlet. The shutoff valve is opened and closed mechanically by power lever movement. When the power lever is positioned for engine power outputs of 80% or more, the shutoff valve is closed and the A/B system functions normally. However, when the power lever setting is below 80%, the mechanical inter-connect between the shutoff valve and the power lever linkage opens the shutoff valve and A/B operation is then stopped. When the mechanically-operated shutoff valve is opened, the metered fuel that normally assists the pressure regulator spring to resist unmetered fuel pressure is routed back to the fuel pump inlet. This reduces the total pressure on the spring side of the pressure regulator and routes the fuel back to the pump/transfer valve. This stops A/B operation.

Maintenance on the Afterburner Fuel System.

Performing the necessary maintenance and trouble shooting on the afterburner fuel system will be quite similar to what you must do on the engine main fuel system. With the exception of the system filters, no regular maintenance is required. Trouble shooting will consist of determining why the exhaust nozzles fail to open or close or why afterburning does not occur.

Malfunctions in the afterburner nozzle system should be traced from the nozzle actuator control unit back to the nozzle actuators. If replacing the actuator control unit does not remedy the situation, the difficulty will lie in the associate air lines, in the relay valve, or in the nozzle actuators and their linkage. If the trouble is in the actuators and their linkage, you will find it best to lubricate all of the components using a commercially available graphite in a volatile liquid. This type of lubrication will reduce the drag and allow the actuators to move more freely. In case the air lines or the air relay valve are at fault, you will have to replace the components with serviceable items.

Cleaning or replacing the filters in the afterburner fuel system is just about the same as for the engine fuel system. Note in figure 2-13 that there are five filters in the afterburner fuel system. With the exception of the filter in the fuel control sensing line, all of the filter cavities must be cleaned with naphtha after the filters have been removed—P-S-661 is quite satisfactory for this purpose. The filters for the igniter control, the nozzle actuator control, and the afterburner fuel control units are not reusable—*new filters must be installed.*

THE J57 ENGINE OIL SYSTEM.

The J57 engine oil system is a dry-sump, self-contained, high-pressure system which supplies lubrication to the engine main bearings and accessory drives. This system is designed to use a synthetic lubricating oil — Specification MIL-L-7808.

Figure 2-14 shows the engine oil system in schematic form. Note how the oil from the oil tank is supplied

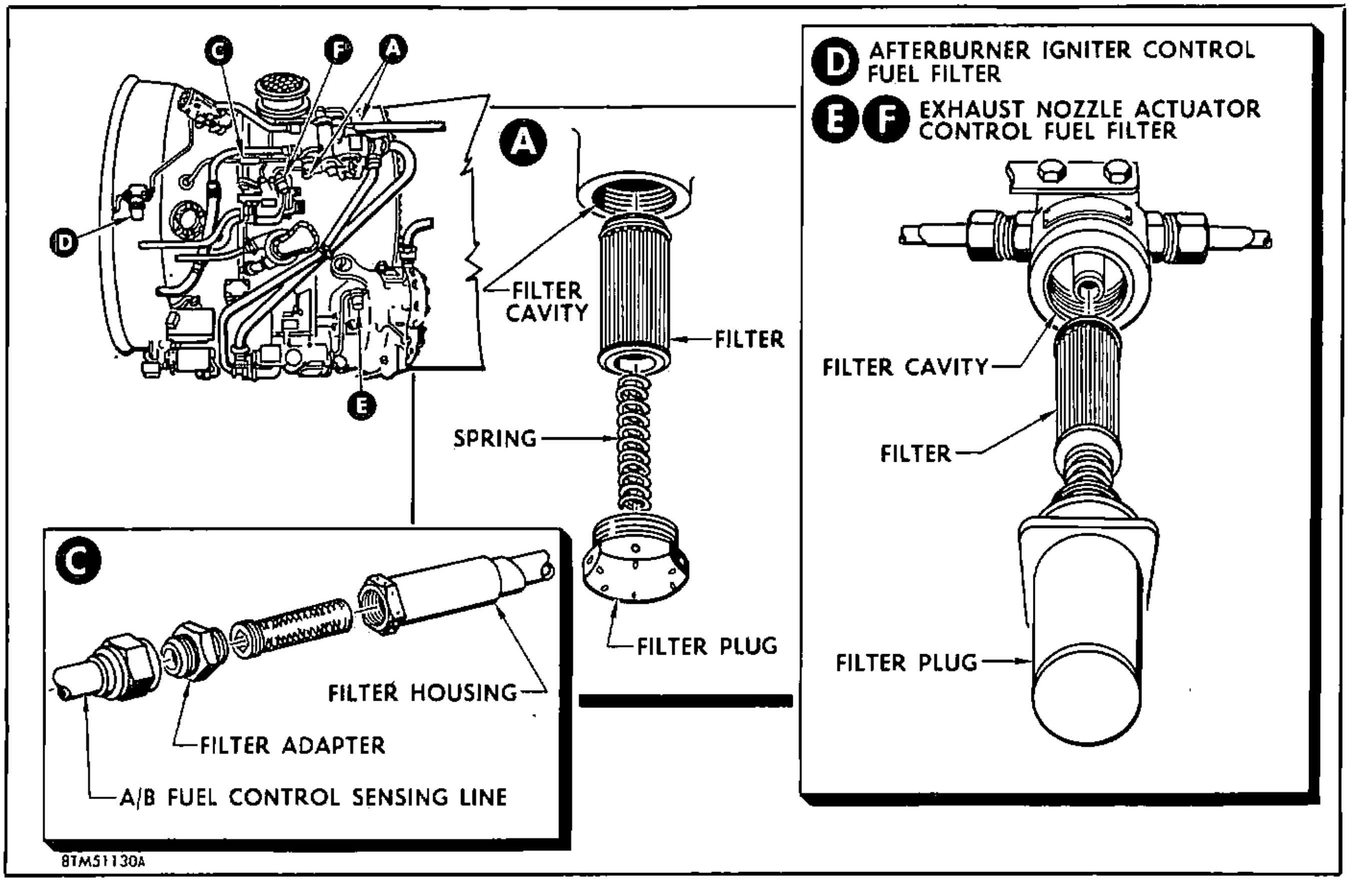

Figure 2-13. Fuel Filters in the Afterburner Fuel System

to the engine oil pressure pump by gravity. The oil forced from the pump is routed through the engine main pressure oil filter. This oil filter is equipped with a bypass valve which permits the oil to bypass in the event of filter clogging. From the filter, the oil is routed to the combination oil pressure regulator and relief valve. This component regulates the oil pressure at about 45 psi, which is the proper pressure differential for the oil metering jets at the engine bearings. From the regulator-relief valve the oil is routed to the bearings by means of external tubing and internal oil passages.

The oil from the bearings is picked up from the engine oil sumps by six scavenge pumps. Following the oil flow in the schematic, figure 2-14, note that the scavenge pumps force the oil through the air/oil cooler, then through the fuel/oil cooler. If the temperature of the scavenged oil is below 160°F, a thermostatic valve mounted on the fuel/oil cooler opens and routes the oil around the fuel/oil cooler and back to the engine oil tank. If the temperature of the oil is above 160°F, however, the thermostatic valve begins to close. At 186°F the valve is in the fully-closed position and all scavenged oil is routed through the fuel/oil cooler.

In the top of the oil tank, the returning oil passes through a de-aerator which removes the trapped air from the engine oil. This air is removed from the tank by means of the oil breather system. The engine oil system also provides lubrication for the engine-mounted Sunstrand constant-speed drive unit. This drive unit uses its own pressure pump, but the oil is taken from the engine oil tank and returned to the oil tank through a separate filter.

The oil tank is contoured to the engine diameter, has a 5.5 US gallon capacity, and is mounted on the upper left "wasp waist" area. The tank has a conventional filler cap for servicing and a cable-type dip stick for measuring oil quantity. Although the tank holds 5.5 gallons, only about 3 gallons are usable. One gallon is for reserve and about 1.5 gallons are contained within the flow lines in the oil system. The tank also has a 1.6-gallon area to allow for expansion during high engine temperatures.

HOW THE ENGINE OIL BREATHER SYSTEM IS PRESSURIZED.

The oil breather pressurization system is provided to insure proper oil flow away from the engine main bearings when the engine is operating at high altitudes. Note in the schematic of the engine oil pressure system, figure 2-14, that breather pressure is shown in dark color. This pressurizing air is supplied

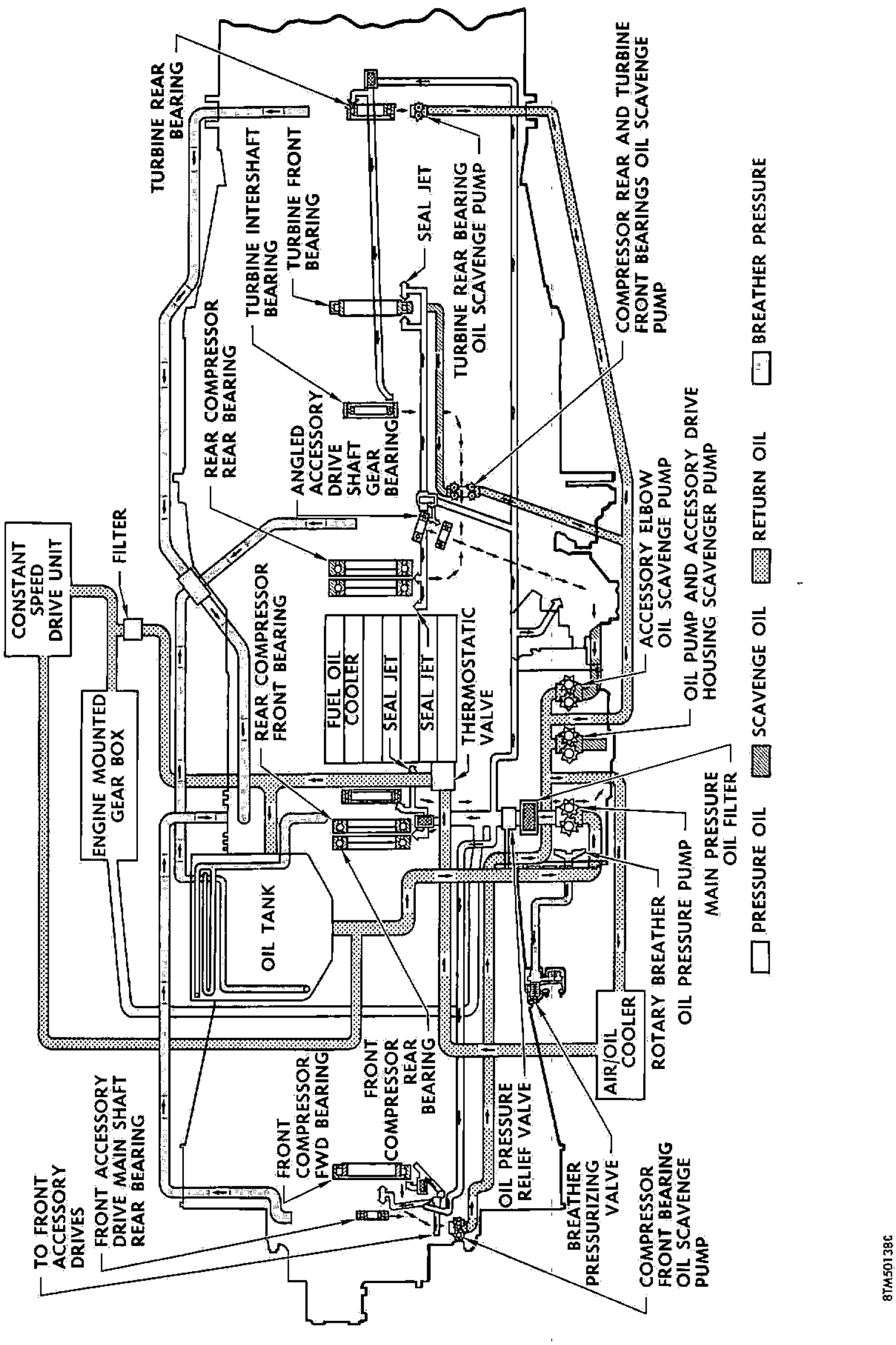

Figure 2-14. The J57 Engine Oil System Schematic

by engine compressor leakage, the compressor air leaks past the compressor seal into the inner case of the engine. Internal passages and tubing on top of the engine connect all of the bearing compartments, the oil tank, and the annular passage around the compressor into the breather system. This assures that all of the oil system will be supplied with the same breather pressure.

The rotary breather, shown in schematic adjacent to the oil pressure pump, removes oil from the pressurizing air by means of centrifugal action. As the pressurizing air passes over the impeller of the rotary breather, the oil is thrown out radially and the relatively oil-free air passes on to the hub of the breather. As the air reaches the hub, it is routed to the breather pressurizing valve which determines how much pressure will be maintained in the breather system. You should understand that the engine compressor air is leaking into the engine case at all times. The breather pressurization valve vents this compressor air to the atmosphere. At sea level operation the pressurization valve is open; however, with increasing altitude the valve gradually closes and regulates the escape of the engine compressor air so that the breather pressure is similar to that at sea level.

HOW THE OIL SCAVENGING SYSTEM OPERATES.

Six gear-type scavenge pumps are used to scavenge the main bearing compartments and the accessory drives. The pumps then force the oil to the oil coolers before it is returned to the tank. One scavenge pump is located in the lower part of the front accessory section; two are located in the right side of the main accessory section; two are located in the lower part of the No. 4 bearing area (rear compressor-rear bearing); and the sixth scavenge pump is located in the No. 6 bearing area (turbine rear bearing). Referring again to the schematic of the engine oil system, figure 2-14, you will note that all of these pumps discharge into one outlet line that runs to the air/oil cooler.

THE OIL LOW-PRESSURE WARNING SYSTEM.

The oil low-pressure warning system notifies the pilot whenever the oil pressure drops below a safe operating level. The warning system consists of a pressure switch, warning lights, and the associate electrical wiring. The pressure switch is mounted on the left side of the engine just aft of the oil tank. This switch is set to actuate, or close, whenever the oil pressure drops to 36 psi or below; if the fuel pressure rises to about 40 psi or more, the switch will deactivate, or open. The pressure switch measures the discharge pressure of the oil pump. The inside of the pressure switch is vented to the accessory drive housing. By comparing the oil pump discharge pressure with the oil system breather pressure, the pressure switch senses differential pressure regardless of altitude.

The warning light for this system is located on the pilot's right hand auxiliary instrument panel. The warning light and the electrical circuit for the oil low-pressure warning system will be discussed in detail in Chapter IV of this supplement.

SUNDSTRAND CONSTANT-SPEED DRIVE OIL SYSTEM.

All oil used for operation and lubrication of the Sundstrand constant-speed drive unit is supplied by the engine oil system. As you will recall from the discussion in Chapter I, the complete drive consists of an engine-mounted gear box and an airframe-mounted transmission and gear box assembly. The engine-mounted gear box receives its oil supply from a line connected into the engine oil pressure switch line. The fuselage-mounted gear box/transmission incorporates a pressure pump and receives its oil directly from the engine oil tank. Scavenging oil lines from both units come togther and return to the engine oil tank through a filter. There are breather lines which connect the constant-speed drive gear box/transmission with the engine breather system. By following the oil flow, as shown in figure 2-15, you will better understand the following description of the system. Oil from the engine oil tank flows to the two charge pumps. This flow is shown by the red diagonals. The smaller of the two pumps is directly driven by the airframe-mounted transmission and gear box assembly input, pumping oil at a rate varying with this input to about 2.75 gallons per minute. Oil from these charge pumps moves to the transmission cylinders by way of a filter which has a bypass valve that opens at a pressure differential of approximately 50 psi, should the filter become clogged. The oil moves into the cylinder block by way of a drilled passage in the manifold and eccentric shaft (6). Inside the transmission itself, oil is pumped from the pump cylinders (1) to the motor cylinders (2) (or from motor to pump depending upon the phase of transmission operation). The oil then moves out of the cylinder block by way of the manifold (6). This oil is maintained at a pressure of approximately 250 to 350 psi by the charge relief valve whi ports surplus oil to the lubrication lines. Another valve—the lubrication relief valve—maintains this surplus oil pressure at approximately 15 to 30 psi. The excess oil from the lubrication relief valve is ported to the transmission.

Appropriately-located jets direct this oil pressure on moving parts. Leakage or blow-by oil from the pistons also drains into the sump, where it is scavenged by another gear-type pump which returns the oil to the aircraft engine oil system.

The oil system for the engine-mounted gear box is quite simple in respect to that of the other component of the Sundstrand constant-speed drive. As shown in the upper right portion of figure 2-15, the oil for

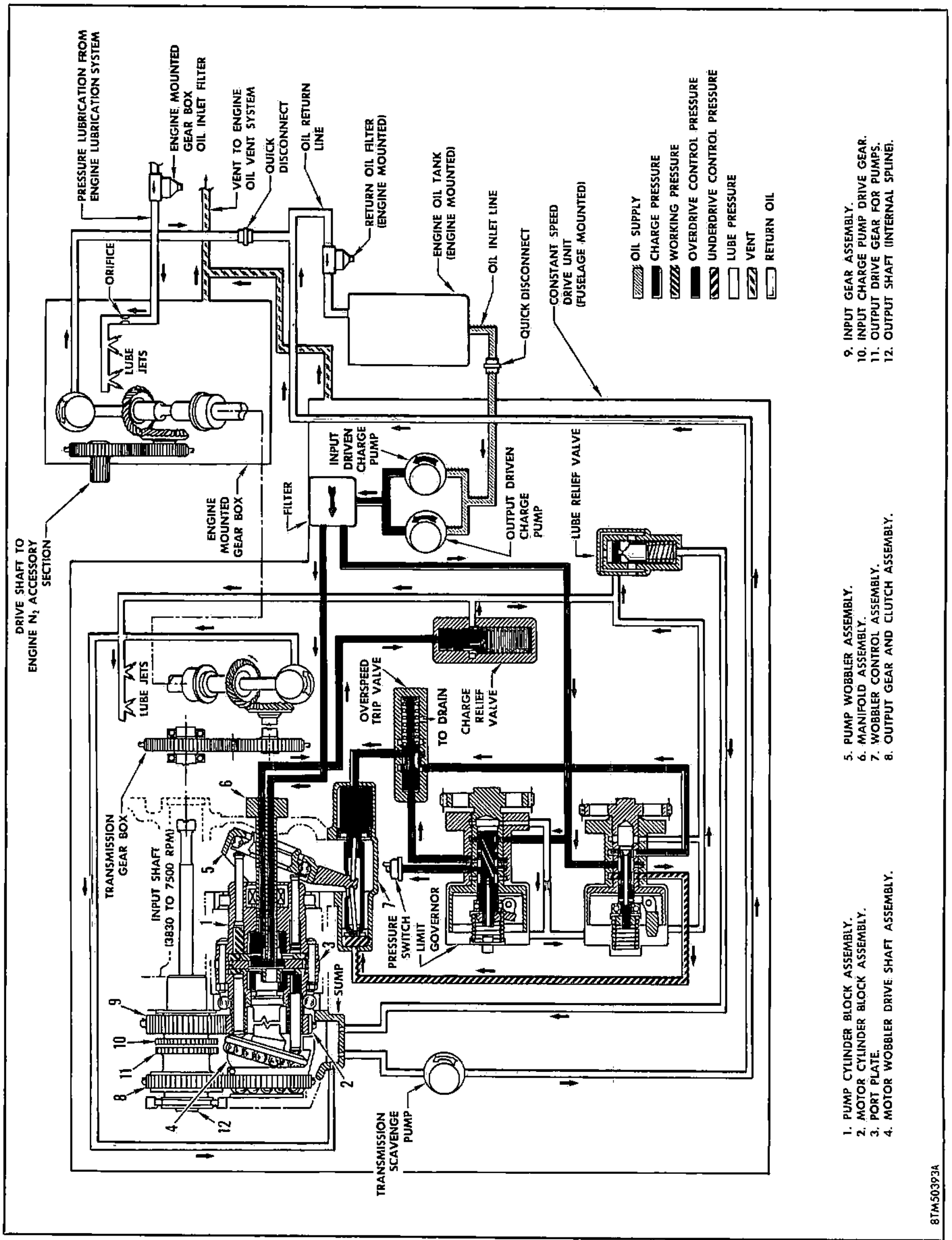

Figure 2-15. Constant-Speed Drive Unit Oil System

pressure lubrication of the engine-mounted unit is obtained from the engine lubrication system. The oil then passes out the outlet line where it joins the return oil from the transmission and gear box assembly oil to flow back to the engine oil system.

THE ENGINE STARTING AND IGNITION SYSTEM.

The engine starting and ignition systems are closely related to each other. Each system is dependent upon the other for completion of electrical circuits and for operation of the system. For this reason, the two systems will be described together in this section.

THE ENGINE STARTER.

The J57 engine uses a pneumatic starter. This starter, which weighs about 35 pounds, is mounted on the forward face of the engine N_2 accessory section. The starter is similar to an axial flow turbine and is actuated by compressed air from a gas turbine compressor (GTC). The flexible hose from the GTC unit is connected to the starter at the quick-disconnect unit in the right wheel well. The air from the GTC unit does not enter the starter until the pilot positions the power lever in the START position. Movement of the power lever actuates a microswitch in the quadrant, and the microswitch completes the circuit to the starter air regulator valve. As this starter valve opens, the air from the GTC unit strikes the turbine blades and exhausts through the starter air exit duct. The force of the air against the turbine blades causes the starter turbine wheel to turn, and the turbine, by means of a gear reduction train and output shaft, drives the engine. When engine starting speed is attained, the engine-mounted *starter* gear reduction train is disengaged from the engine by means of an integral clutch. A centrifugal action switch in the starter causes the air regulator valve to close when engine speed reaches 3400 to 3700 rpm. This action shuts off the air supply to the starter.

The lubrication of the engine starter is self-contained —the lubricating capacity is 12 fluid ounces. Normally, the fluid level should not have to be checked except when there is indication of oil leakage; however, the system has to be drained and refilled at intervals of 25-hours engine operating time.

ENGINE IGNITION.

As mentioned earlier, the ignition and starter circuits are dependent upon each other for successful engine ignition and starting. This can be seen more clearly by referring to the schematic of the engine starting and ignition system, (see figure 2-16). Let us trace the sequence of actions and current flow through the schematic for a normal engine start.

First, locate each of the eleven components in the schematic. The starting action is controlled through the power lever movement and the ignition button. The starter switch (5) (microswitch) is shown in the de-activated postion such as would be the case when the power lever is in the OFF position. As you will recall, the first step in starting the engine is to move the power lever outboard to the START position—this closes the starter switch. Current then flows from the 28-volt d-c essential bus, numbered 4 in the illustration, to the ignition switch and also to the engine starter, number 7. As current reaches the engine starter, the starter air regulator valve opens and compressed air from the ground GTC drives the starter.

As the engine speed reaches 12 to 16%, the operator depresses the ignition button. This action closes the ignition switch and energizes the starter relay. As already mentioned, depressing the ignition button does not cause ignition—it just arms the starter relay. This type of arrangement allows the operator to move the power lever forward without stopping the starter. When the power lever is moved forward towards the IDLE position, the starter switch is deactuated and fuel starts metering into the engine combustion chambers. Note in figure 2-16 that as the starter switch is deactuated, current is fed to the ignition power relay, numbered 1. With the engine turning at 12 to 16%, and with fuel being metered to the chambers, the engine starting operation is ready for ignition. As the ignition power relay is energized, current is fed through the engine junction box (11) to the ignition transformer (8). There are two of these transformers, however, only the No. 2 transformer is shown in the schematic. The other transformer, No. 1, is exactly like the one shown except that it supplies ignition to the No. 5 combustion chamber instead of the No. 4 chamber.

When the current first enters the ignition transformer, it passes through several input filters (choke coils and capacitors). Then it is led to a 6000 rpm motor. Note that the same current that operates the motor is tapped off to a cam-type switch. The cam, geared to the motor, turns at about 1200 rpm. This cam has the effect of "chopping up" the direct current input and turning it into pulsating d-c. This pulsating d-c then flows to the step-up transformer where the 24 volts are stepped-up to about 2000 volts. You will note that this high voltage then flows through the selenium rectifier which converts the pulsating current back to d-c. This rectification is necessary so that the ignition capacitor can be charged. In the figure 2-16 this ignition capacitor is shown directly below the second motor-driven cam. At the same time the ignition capacitor is being charged, the second cam is turning at 240 rpm. Each time this cam closes its ignition point, a pulse of direct current will flow through the "triggering" transformer to the ignition capacitor. This pulse of direct current is sufficient to spark across the igniter plug gap. As it sparks across the gap, it provides a "path" for the charge in the ignition

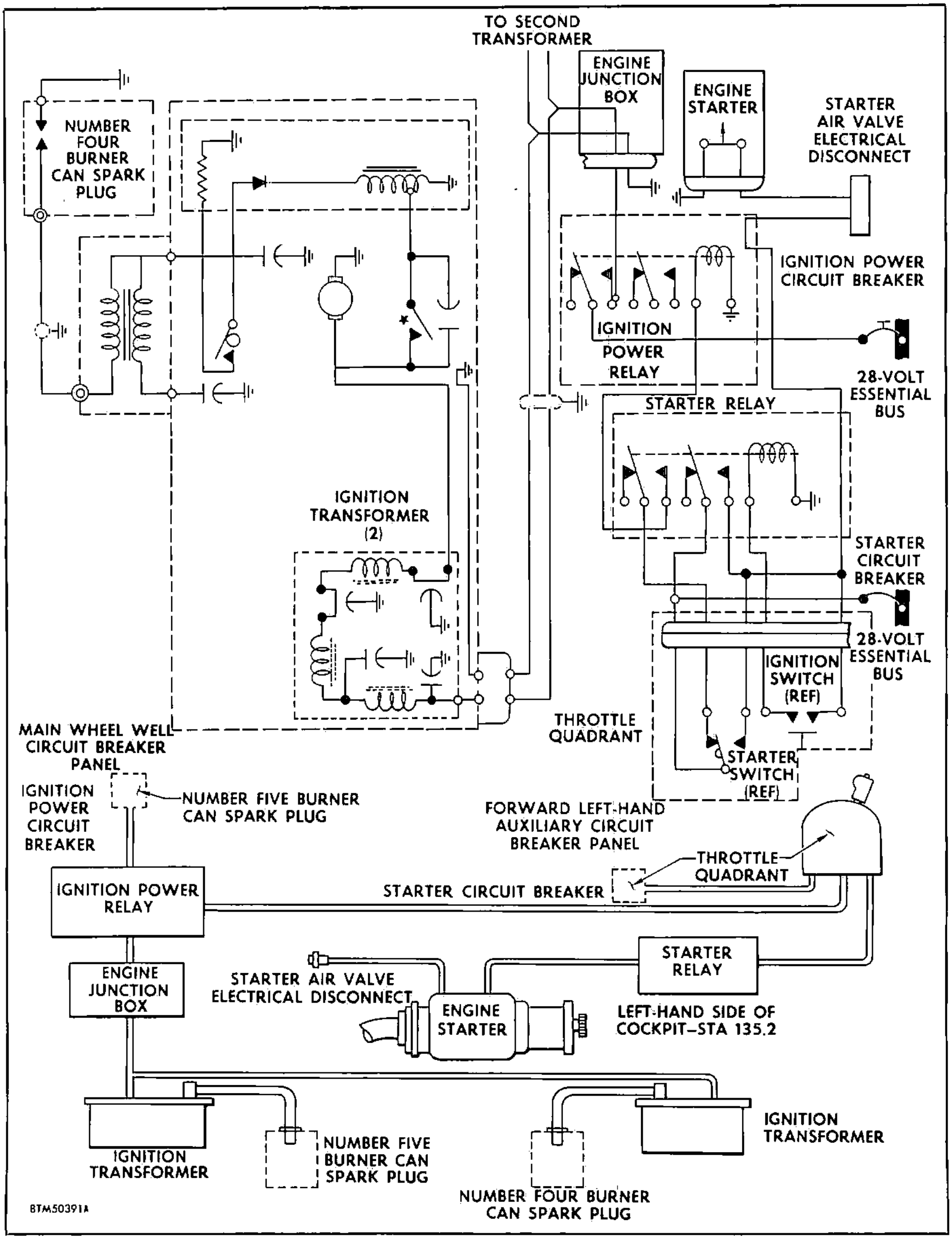

Figure 2-16. Engine Starter and Ignition Electrical System Schematic

capacitor. With an open "path" of small resistance, the charge on the capacitor flashes through the "triggering" transformer and across the igniter plug gap.

Summarizing the engine ignition circuit, you should retain several important facts. The power lever movement and the ignition button provide the only pilot control for the engine ignition system. By means of two relays—the starter relay and ignition power relay—and the two ignition transformers, this ignition system provides "hot" spark ignition for the No. 4 and 5 combustion chambers. You should also keep in mind that there is no timing device in the ignition circuit. Because of this fact, you should never keep the ignition button depressed longer than 30 seconds after the power lever has been advanced to IDLE. This 30-second time limit is necessary to prevent over-heating of the ignition transformers.

This electrical system, like the rest of the engine electrical systems, is covered completely in the F-102A maintenance technical order for the airplane electrical systems, T.O. 1F-102A-2-10.

HOW THE ENGINE IS STARTED AND OPERATED.

During the engine starting procedure, care must be exercised to correctly operate the starter, ignition, and power lever controls to successfully complete the engine start. Actuation of the engine starter is initiated by the outboard movement of the power lever to the START position. The power lever is spring-loaded to the OFF position and must be held over in the START position. The power lever movement operates a microswitch in the quadrant which opens the starter air valve. Opening the air valve permits compressed air from the gas turbine compressor to actuate the starter. The power lever is held in the START position until the starter is turning the engine from 12 to 16% rpm. Depressing and holding the ignition button energizes the starter holding relay. Ignition does not occur at this time, but it will permit the starter to continue cranking the engine while you move the power lever—with the ignition button still depressed—forward from the START position. This movement of the lever initiates ignition through the actuation of the microswitch in the quadrant. Operation of the engine starter will continue as long as the ignition button is depressed, or until the starter centrifugal switches actuate due to starter overspeed. Moving the power lever forward towards the IDLE position actuates the main fuel control. This permits fuel to be injected into the engine burner section. Ignition is supplied to the igniter plugs in the No. 4 and 5 burner cans. As the fuel is ignited, the engine rpm increases. The ignition button should be held down until the tachometer and temperature gage indicate a positive start, then the button should be released. If you should momentarily release the ignition button when the power lever is between OFF and IDLE and the engine does not start, return the power lever to the OFF position. This action will prevent additional fuel from being injected into the engine burner section. You must then repeat the complete starting cycle from the beginning. After returning the power lever to the OFF position, do not attempt another start until fuel drainage from the engine combustion chambers drain has stopped. Never operate the starter more than 90 seconds during any two minute period. After a normal engine start has been completed, the following procedure is recommended for checking the engine operation: set the pressure ratio indicator for the correct ambient air temperature, and then advance the power lever to the MIL POWER postion and allow all instrument readings to stabilize.

CAUTION

Do not operate the engine at MIL POWER any longer than five minutes.

While the power lever is in the MIL POWER range, check the tailpipe temperature for a maximum of 610°C. The indicator needle on the pressure ratio gage should fall within the maximum power range marked on the gage. None of the engine warning lights should be burning. Then move the power lever to AFTERBURNER.

CAUTION

Do not operate the afterburner longer than one minute.

A rapid increase in tailpipe temperature and an rpm reduction of about 4% usually mean that the exhaust nozzle has failed to open. When this occurs, terminate the afterburning operation immediately. Then, opening the afterburner power circuit breaker, move the power lever to FULL MIL POWER and then retard it—without hesitation—to some point below 80% power. Afterburning should cease at this 80% power. There should be no indication of engine roughness as the power lever is being retarded before afterburning is terminated. Return the power lever to IDLE, and allow an afterburner drainage period of two minutes.

When the preceding operations have been accomplished, actuate the emergency fuel control switch from NORMAL to EMERGENCY. The emergency fuel warning light should illuminate; then advance the power lever to MIL POWER. Fuel flow should be 6350 to 6950 pounds per hour at sea level. Returning the power lever to IDLE, note that normal engine operation continues throughout power lever range. Then, actuate the emergency fuel switch to NORMAL—the warning light should go out.

HOW TO STOP THE ENGINE.

Prior to stopping the engine, you should allow it to idle for approximately 5 minutes. In an emergency, however, the engine may be shut down immediately. Have a ground crew assistant connect a ground compressor, which is operating at maximum output, to the airplane ground receptacle. Connect a-c and d-c power to the airplane ground receptacles. Turn all fuel boost pump switches OFF, and retard the power lever to the OFF position. In the event that exhaust temperature rises above 225°C after engine shut down, actuate the power lever to the START position and crank the engine for about 20 seconds. Then, return the power lever to the OFF position.

ENGINE AIR INDUCTION AND COOLING.

Although the cooling system for the gas turbine engine is not as complex as that required for piston-type engines, considerably more effort must be spent in protecting the surrounding aircraft structure from the heat. This is usually accomplished by a shroud installed around the outside of the engine with cooling air directed between the shroud and the outside of the engine. Some engines require cooling for accessory drives, turbine wheels, bearing housings, and fuel pumps. Other engine installations require a method of cooling the oil. Gas turbine engines usually use any of three cooling mediums: air, oil, or fuel. Air is by far the most commonly used method of the three.

Air cooling must be accomplished by creating high air flow and pressure with a minimum loss from drag, turbulence, and other detrimental factors. There are four methods by which air cooling can be accomplished: induced air, ram air, auxiliary air (fan), and engine compressor bleed. The induced air method is the most commonly used, while the compressor bleed system is rarely used. The turbine wheels, bearing housing, oil coolers, and tailpipe usually use one of the above cooling methods. The tailpipe shroud is sometimes kept cool by an insulating blanket installed between the engine and the shroud. The burner cans are usually kept cool by internal air flow which is actually a part of the combustion system. Fuel pumps are almost always cooled by the fuel being pumped through them. Fuel as a cooling agent has been used in fuel/oil heat exchangers, but not to any great extent. Oil is used for cooling such components as accessory drives, bearings, and reduction gears—the oil itself being cooled by an oil cooler similar to that used on reciprocating engine installations. The cooling system, as a rule, requires no pilot attention or operation. The oil pressure gage in the cockpit is the only connection between the pilot and the cooling system. The oil pressure gage could indicate any serious over-temperature condition in a part being cooled or lubricated by oil.

Some of the earlier gas turbine engine main rotor bearing temperatures were very critical. As a result, bearing temperature gages were required. Since that time, engineering developments have solved the bearing problem. In all probability, bearing temperature gages will not be encountered in late model and future gas turbine engines.

IN-FLIGHT COOLING OF THE ENGINE COMPARTMENT.

In the F-102A, the engine compartment cooling airflow pattern varies between engine ground operation and flight operation conditions. Air flow during flight is based on a positive pressure condition existing in the engine intake ducts; while engine cooling during ground operation is dependent upon air bled from the N_1 rotor and reverse air flow through the air/oil cooler and generator cooling ducts.

Note in figure 2-17 that there are three sources of air and that each source is located in the engine intake duct. One source is bled from a scroll, numbered 2, located on the forward end of the engine stub duct. Part of the air taken from this scroll is passed through the air/oil cooler (13) and the remainder is ducted aft under the shroud (10) that surrounds the engine hot section. That part of the air which passes through the air/oil cooler is exhausted into the accessory section and flows aft between the shroud and the fuselage. The cooling air, which is ducted aft, flows upstream from the air/oil cooler and runs aft along the left side of the engine to the forward section of the engine shroud. Cooling air entering the shroud is diffused circumferentially by the distribution section and flows aft between the shroud and the engine exhaust nozzle.

GROUND COOLING OF THE ENGINE COMPARTMENT.

First of all, in-flight cooling below 150 knots IAS with the landing gear extended and ground cooling are the same thing; in other words, the air flow about the engine is identical in both cases. This type of cooling will be referred to as ground cooling in the text.

Cooling during ground operation varies substantially from that described in the preceding paragraphs. J57 engines installed in the F-102A airplanes incorporate a top center-line bleed which furnishes ninth stage N_1 air to cool the hot section of the engine. An electrically operated shutoff valve, a check valve, and two series-mounted flapper valves permit rescheduling of air flow for ground cooling and cooling at airspeeds below 150 knots with the gear extended.

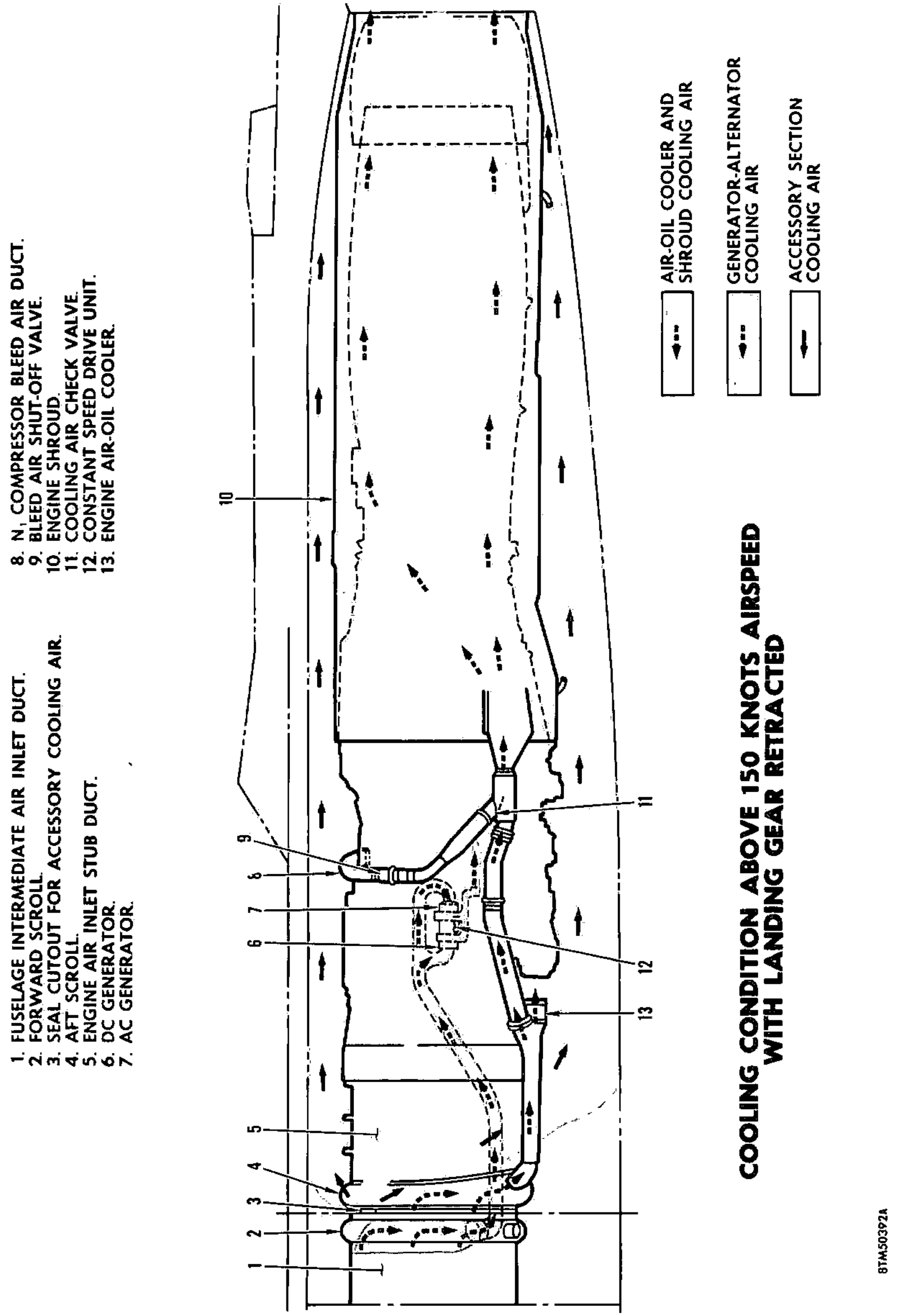

Figure 2-17. Engine In-Flight Cooling System Schematic

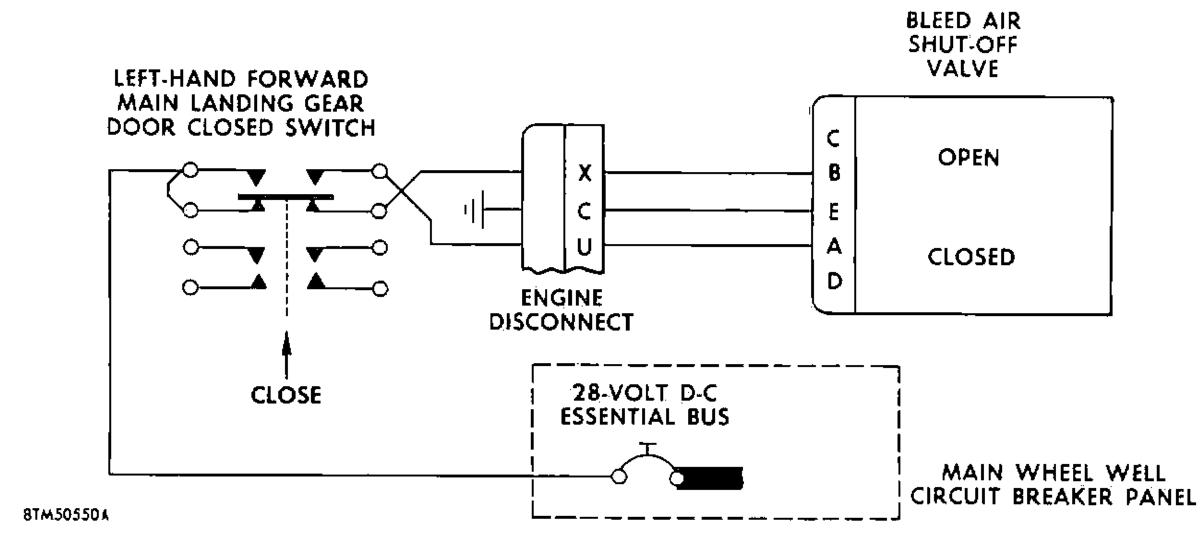

Figure 2-18. Engine Cooling Electrical Schematic

Notice in figure 2-18 that the electrical circuit for the shutoff valve is wired in series with the left landing gear door—thereby opening the valve when the gear is retracted. This is understandable, since higher airspeeds are associated with a clean airplane (gear up and locked); at these high airspeeds no help in cooling is needed from the top center N_1 compressor bleed system. The check valve (11) located in the cooling duct and connecting the stub duct scroll to the shroud, prevents N_1 compressor air from flowing forward into the scroll.

During ground run a partial vacuum exists in the engine air intake ducts, this vacuum causes a flow reversal in tthe cooling ducts. Cooling for the engine oil and the generators is also provided by this reverse flow during ground operation. The reverse air flow through the air/oil cooler is easily accomplished without check or flapper valves; however, the generator cooling operation is more complex. Generator cooling, both in the air and on the ground, is covered in the next section of this chapter.

When an airplane is on the ground and the engine is running, the cooling air bleed shutoff valve (9) is open. Note in figure 2-19 that this allows ninth-stage air pressure to flow down the duct on the left side of the engine and force the check valve (11) to open. This compressor air then flows under the shroud surrounding the engine. The air is exhausted around the exhaust nozzle between the engine and the shroud.

As mentioned, air flow reversal also takes place when the airplane is on the ground. This air flow reversal is caused by the low pressure condition in the intake ducts. Air is actually drawn in the aft end of the airplane between the engine shroud and the aircraft structure. This air is drawn forward through the air/oil cooler, and through the two previously mentioned slots between the scrolls. The air is then pulled back through the engine for the combustion process.

IN-FLIGHT COOLING OF THE A-C AND D-C GENERATORS.

The intermediate duct scroll takeoff is located in approximately the 4 o'clock postion. Note in figure 2-20 that the cooling duct from this takeoff runs aft down the right side of the engine to the a-c and d-c generators. Cooling air simultaneously enters the aft end of the a-c generator and the forward end of the d-c generator. Both ducts come together downstream of the generators and at this point there are two opposite-acting flapper valves. Exit air closes the overboard flapper valve and opens the other flapper valve to the accessory section. Under these conditions exit air from

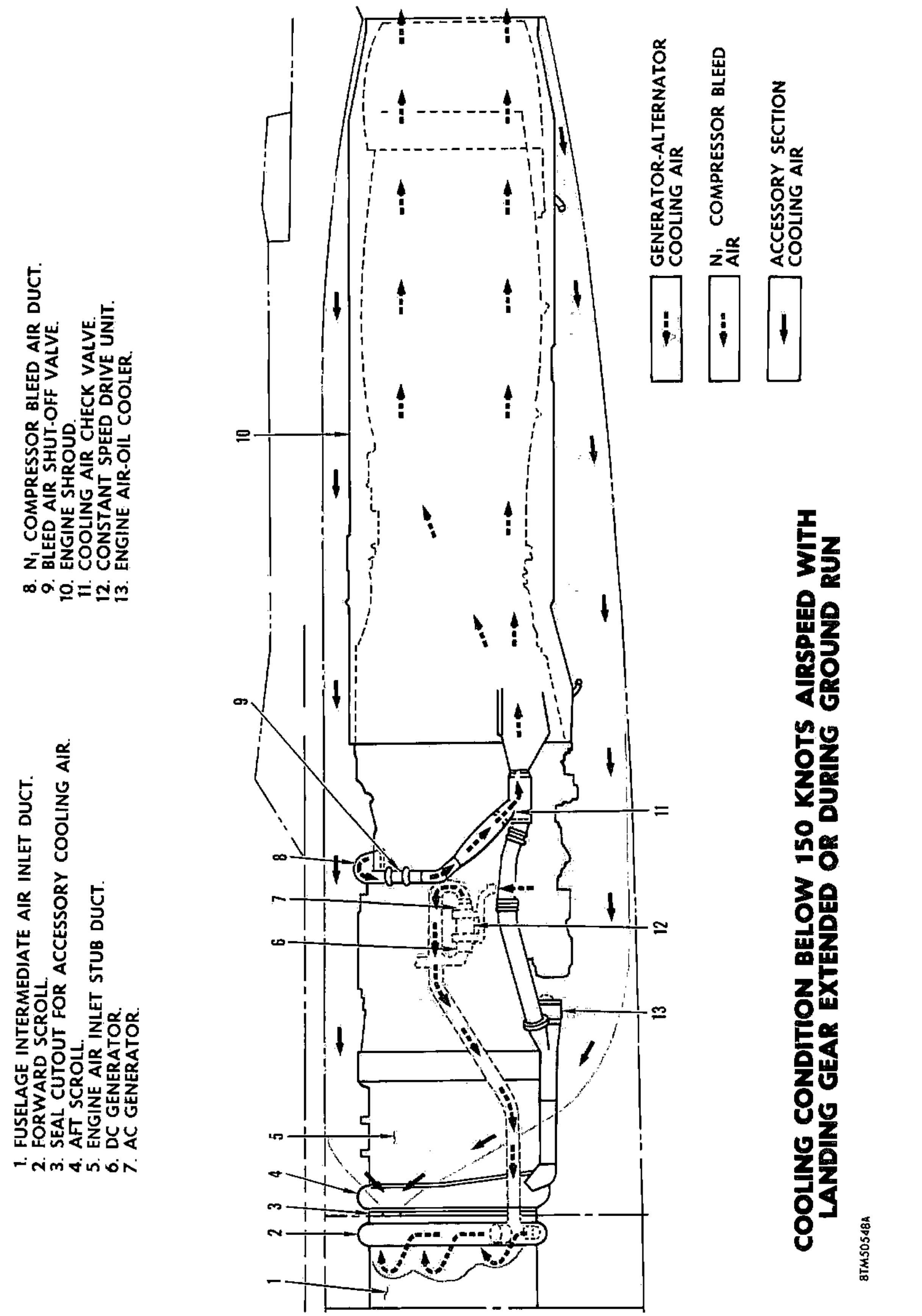

Figure 2-19. Engine Ground Cooling System Schematic

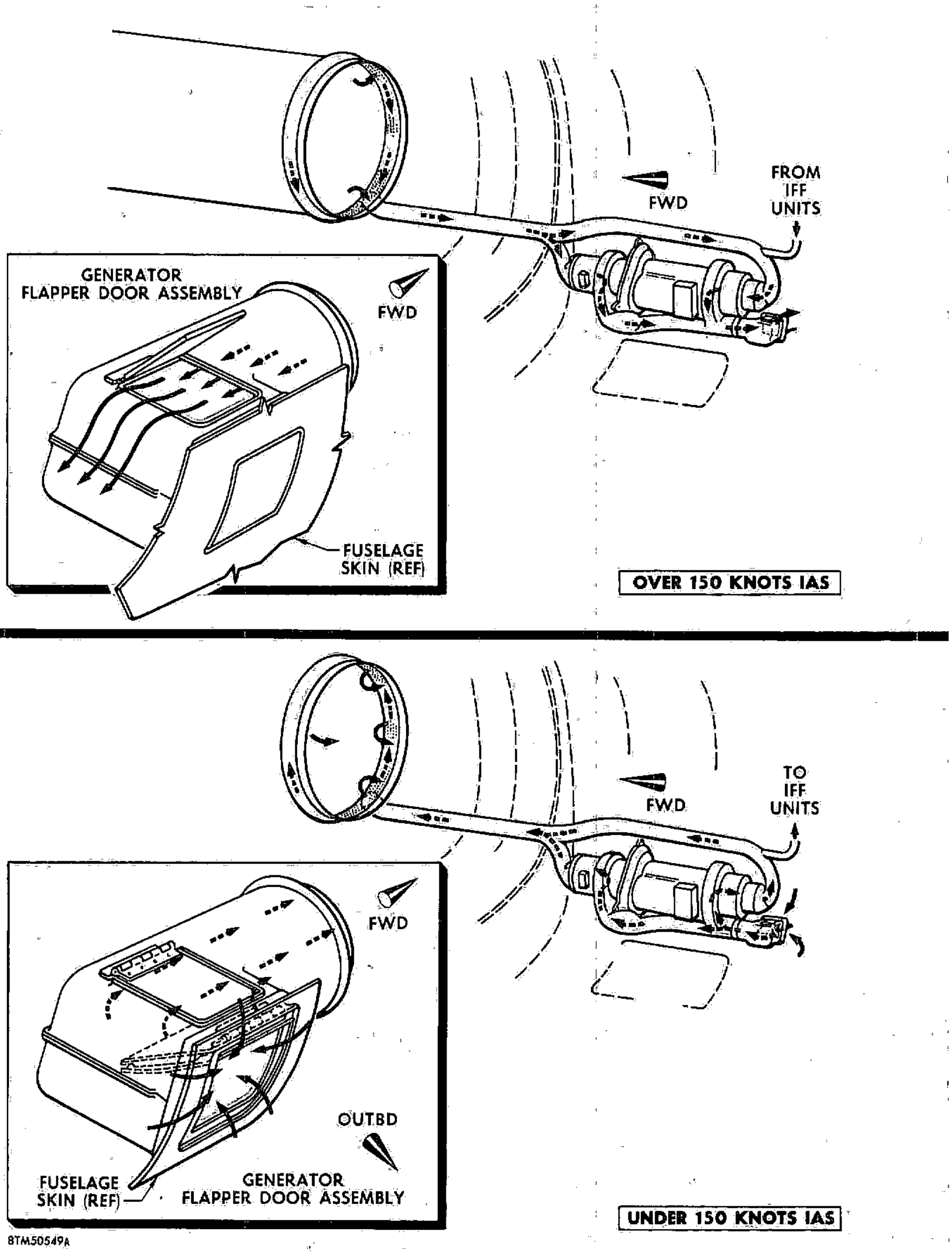

Figure 2-20. Generator Cooling System Schematic

both of the generators is further utilized for cooling the accessory section. The air continues aft and exits between the afterburner shroud and the tailpipe structure.

GROUND COOLING OF THE GENERATORS.

Generator cooling under low airspeed conditions (which is the same as ground operation) is accomplished on the reverse flow principle, as shown below. This reverse air flow opens the ground run door on the airplane skin and closes the accessory section flapper valve. In this manner, cooling air is drawn in from the outside and flows forward through the duct scroll. At this point the air is drawn into the compressor section and used for combustion purposes.

ENGINE ANTI-ICING SYSTEM.

Whenever icing conditions exist, ice will form not only on the airframe structure, but also in the engine inlet area. If ice formation is allowed to continue, it will seriously reduce the engine power output. In the F-102A airplane, provisions are made to prevent the ice accumulation on the inlet guide vanes and surrounding area by directing sixteenth-stage heated air through the hollow vanes and forward around the nose accessory fairing. This hot air is exhausted behind the nose accessory fairing cap and allowed to enter the engine intake. Airflow is controlled directly by two electrically-operated shutoff valves, and indirectly by two flow-control valves.

In figure 1-16 of Chapter I, the complete airflow pattern is shown. The electrical controls and the components for the engine anti-icing system will be discussed more fully in Chapter IV.

SUMMARY.

In Chapter II you have learned how the engine associate systems function to assure that the basic engine combustion cycle will produce its continual flow of power. These systems have included the fuel, oil, starting and ignition, induction and cooling, and the anti-icing systems. The knowledge of how these systems function and how each component plays its part in the overall function of each system will enable you to intelligently perform the required maintenace and trouble shooting on the J57 engine. Although, in some cases, specific values for pressures or temperatures were given, you should keep in mind that these were for explanatory purposes only. For the exact values, you should always refer to the maintenance technical order which covers the power plant installation in the F-102A, T.O. 1F-102A-2-4.

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Chapter III

ENGINE BUILD-UP AND HANDLING

Contents *Page*

The first two chapters in this supplement have acquainted you with the J57 engine, its components and major operating systems, and how the engine operates. These two chapters have explained *what* the engine does while it is installed in the airplane. As a power plant maintenance man, you should understand that you may have to accomplish the handling and maintenance requirements of the engine when it is *not* installed in the airplane. These requirements will consist of: preserving and depreserving the engine; building up the engine when it is first received from the manufacturer; and installing the engine in (and removing the engine from) the F-102A.

All of these requirements are covered in this chapter. After first learning some of the basic principles involved in the prevention of corrosion on engines, you will be familiarized with the preservation and depreservation requirements of the J57 engine. The major portion of the chapter is devoted to the engine build-up operation. In this discussion you will learn just what you must do to a new or overhauled engine before it can be installed in the F-102A. The last part of this chapter covers the engine installation and removal operations and the special tools which you will need to accomplish them.

Before proceeding with this chapter, there is one point which you should thoroughly understand. As mentioned earlier, both the J57-23 and the -41 engines are used in the F-102A. In some instances the maintenance requirements and techniques are not the same. In most of these cases the differences are mentioned in the text or shown in the illustrations. Before attempting any maintenance on either model engine, however, always determine the type of engine first and then check the maintenance technical order (T.O. 1F-102A-2-4), for the exact procedures and instructions for that particular model.

ENGINE PRESERVATION AND DEPRESERVATION.

The combat against corrosion of aircraft engines is primarily a fight against moisture. In the light of present-day technical knowledge, the chemical principles involved in metal corrosion are fairly well understood. It is generally accepted that there are two types of surface corrosion—the direct chemical attack on metals by corrosive liquids, and the electro-chemical attack in which the metal being corroded becomes a part of an electrolytic cell in the presence of moisture. Both types of corrosive action are effectively retarded by the *absence* of moisture. This fact has led to the use of sealed, metal containers to protect engines during shipment and extended storage periods.

THE USE OF JET ENGINE SHIPPING AND STORAGE CONTAINERS.

The metal shipping and storage container for a jet engine serves four important functions. First, it protects the engine from vibrations encountered in normal transit. Second, it protects the engine from shock encountered in abnormal or rough handling. Third, it protects the engine from the various atmospheric conditions encountered in world-wide distribution and storage. Fourth, the container permits the storage of an engine for greater lengths of time, without frequent inspection and represervation operations.

Because metal containers are sealed from the atmosphere, they require different sealing and dehydrating techniques from those used with wooden containers. For example, the use of a protective envelope over the engine inside the shipping container is unnecessary. It is also unnecessary to seal some of the engine openings, to prevent air from entering, or to install dehydrating plugs in other engine openings, to reduce the moisture content.

By using the sealed metal containers, the engine openings need only be covered with ventilation plugs which are designed to keep foreign matter out of the engine interior. Dehydration of the engine and the metal container is accomplished by placing a predetermined amount of bagged dehydrating agent in baskets inside the container and then pressurizing the engine container wth dehydrated air.

One of the most beneficial results of the use of metal containers is the limited attention which the stored engines demand. Adequate protection from the elements is afforded the engine by the container itself.. The only maintenance requirements are the maintaining of the air pressure within the container, and the observing of the prescribed inspection and representation periods.

EFFECTIVENESS OF ENGINE PRESERVATION.

The effectiveness of engine corrosion preventive procedures depends upon two factors: prompt and complete application of preservation procedures, and faithful compliance with the instructions for maintaining the preserved state of the engine during storage. As you can well understand, each of these factors is dependent upon the other. If you have not satisfactorily completed engine preservation in the first place, maintaining the preserved state of the engine will not do very much good. Conversely, if engine preservation was accomplished satisfactorily, improper maintenance of the stored engine, will also make the engine preservation ineffective.

You should understand that engine preservation maintenance is *not* representation; but it is the regular inspection, and if required, replacement of the dehydrating agent. If the integrity of the moisture vapor barrier inside the container has not been maintained, or if the dehydrating agent has become saturated, an internal inspection of the engine is required. This inspection should reveal whether corrosion exists on the interior of the engine; whether overhaul or complete representation is necessary; or whether renewal of the dehydrating agent will be sufficient.

Reports show that most of the corrosion of engines in storage is directly traceable to delay (permitting the onset of corrosion prior to preserving the engine); poor application of the required preservation procedures; or a lack of proper maintenance of preservation during extended periods of storage.

When the entire preservation process is properly carried out and adequately maintained, however, it has been proven that corrosion is not likely to occur while the engine is in storage.

Although the engine preservation process is completely covered in military specifications and technical orders, some of the more important points are mentioned in the following paragraphs.

Cleaning the engines and engine parts with chlorinated solvents, such as trichlorethylene and carbon tetrachloride, should be avoided whenever possible. Although chlorinated solvents are excellent cleaners, they tend to encourage corrosion on metal surfaces. Cleaning solvent, Federal Specification P-S-661, is preferred for hand cleaning the exterior engine surfaces to which corrosion preventative compounds are to be applied.

The proper installation of the dehydrating agent requires that this material be handled with considerable care and speed to allow a minimum of exposure time between removal of the agent from its package and installing the agent in the engine metal container. Partial saturation of the dehydrating agent (which may occur rapidly with exposure to outside air), reduces the effective life of the agent. Installation of the dehydrating agent requires reasonable care in timing, but is not so difficult that satisfactory preservation cannot be accomplished.

HOW TO PRESERVE THE J57 ENGINE.

Preservation of the J57 engine is quite simple if it is done while the engine is installed in the airplane. Basically, the preservation process consists of replacing the engine oil and fuel with preservative mixture and then spraying the engine compressor inlet with preservative mixture. The preservation operation is accomplished by running the engine and substituting preservative mixture for the fuel and oil. Although the process listed below is fairly detailed, you should never start to preserve a J57 engine until you have first read the controlling Military specifications and the engine maintenance technical order, (T.O. 1F-102A-2-4).

Preserving the Engine Oil System.

To preserve the engine oil system: first, place oil receptacles under the oil tank drain, the N_2 gear box drain, the N_2 rear gear box (elbow gear) drain, the drain on the left side of the N_2 gear box near the oil temperature bulb connection, and the fuel/oil drain. Remove the drain plugs and allow the oil to drain until the oil flow slows to a slow drip at all of the openings, then re-install and safety the drain plugs. The engine oil filter on the top of the main gear box has to be removed, completely disassembled, and cleaned with a suitable cleaner (JP-4 fuel is satisfactory for this purpose); after being cleaned the filter should be reinstalled. When you have accomplished the preceding operations, service the engine oil tank with $5\frac{1}{2}$ gallons of preservative mixture. For the exact type of preservative mixture, you should consult the controlling specifications and the engine maintenance technical order, (T.O. 1F-102A-2-4).

Run the engine up before you start on the fuel system preservation operation. To assure that the preservative mixture thoroughly lubricates the interior of the engine, operate the engine at 75% power for a period of ten minutes.

Preserving the Engine Fuel System.

After shutting the engine down, following the oil preservation run-up, you are ready to begin preserving the engine fuel system. Disconnect the pressurizing and dump valve sensing line at the fuel control unit, cap the fitting on the fuel control unit but leave the sensing line open to atmosphere. Place a 5-gallon drainage receptacle under the dump valve drain discharge point (located at the left of the engine accessory compartment access door); then disconnect the afterburner fuel line at the forward side of the afterburner manifold drain valve (located forward of the firewall). Connect an overboard discharge hose to the end of the disconnected afterburner fuel line and route it to another 5-gallon receptacle under the airplane. Disconnect the flexible fuel line to the engine fuel pump inlet port and connect a 2-inch flexible hose to the pump inlet port. Place the other end of the hose in a 10-gallon container of preservative mixture. As a special note, be sure to keep this 10-gallon container at least four feet above the pump inlet port. This will assure sufficient inlet pressure to the pump. If the container should become empty at any time during the following operation, shut off the engine starter and refill the container.

Next, pull out the ignition circuit breaker on the breaker on the main wheel well circuit breaker panel, connect the external source of a-c and d-c power, and post a fire guard. Now connect the GTC unit air duct to the starter air duct receptacle in the main wheel well. Eight circuit breakers must be closed—FUEL SEL LH, FUEL SEL RH, FUEL CONT, AB CONT, AB POWER, EXT AC PWR, OIL PRESS, and MASTER. These circuit breakers are located on the forward left hand auxiliary and main wheel well circuit breaker panels.

The rest of the fuel system preservation operation has to be timed (either a timer or a stop watch is satisfactory). Let us assume that you use a stop watch. Advance the power lever to the TAKE OFF position; have the ground operator start the GTC unit and adjust the unit speed to 100% run; start the stop watch at the same time the GTC unit reaches 100%. After 15 seconds have elapsed, move the power lever outboard to the AFTERBURNER detent. When the watch reads 25 seconds, re-position the power lever to FULL MIL POWER. After 30 seconds total time, actuate the fuel control switch to EMERGENCY. Five seconds later, place the fuel control switch in the NORM position and move the power lever to the OFF position and then forward to the TAKE OFF position at a slow, steady rate. After 60-seconds total elapsed time, simultaneously shut off the starter and bring the power lever back to the OFF position.

After the engine has stopped turning, disconnect the 2-inch flexible hose from the engine fuel pump inlet Then start the GTC again, adjust its speed to 100% and allow it to turn the engine over. With the e gine turning, spray about one-half pint of preservati mixture over the compressor inlet section. Be sure keep the spray gun about 18 inches from the c pressor inlet, and keep moving the gun constantl assure that the entire compressor entrance are sprayed evenly. When you have completed the s ing, disengage the starter valve, shut down the unit, and disconnect the starter from the engi

This completes the preservation operation gine while it is installed in the airplane. Y ready to proceed with engine removal.

HOW TO DEPRESERVE THE J57 EN

All jet engines shipped from overh manufacturers are preserved, to rosion. As a result, you will ha fuel system and drain the pr system. No special fluid is n flushing operation—regula isfactory. The basic idea operation is to drain t engine over (without forced through the e the preservative. P operation, fill the

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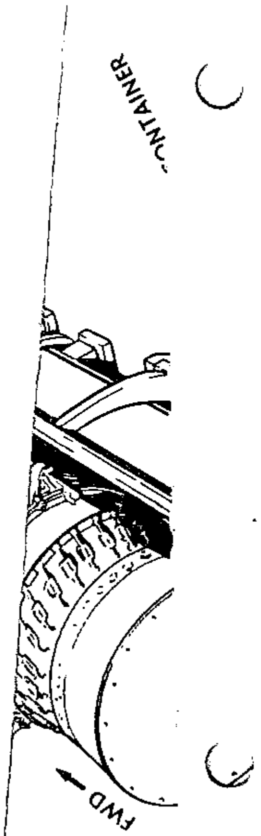

Disconnect the afterburner sensing line from the afterburner igniter and provide a receptacle for the oil from this line. Leave the drain plugs out until the oil discharge slows to a slow drip. Then, remove and completely disassemble the engine oil filter at the top of the main gear box. Clean this filter with a suitable cleaning solvent or JP-4 fuel. Then, re-assemble the filter and install it on the engine. When the preservative has drained out, replace and safety the drain plugs, and connect the sensing line. Then fill the engine oil tank with 5.5 gallons of oil, specification MIL-L-7808.

Flushing the Fuel System.

As mentioned earlier, some J57 engines are equipped with pneumatic starters and other engines are equipped with combustion-type starters. The procedure given in this training manual for flushing the engine f… system is for the pneumatic starter-equipped e… only. Information on flushing the engine f… n the combustion-type starter-equipped … given in the engine maintenance … en the information is available…

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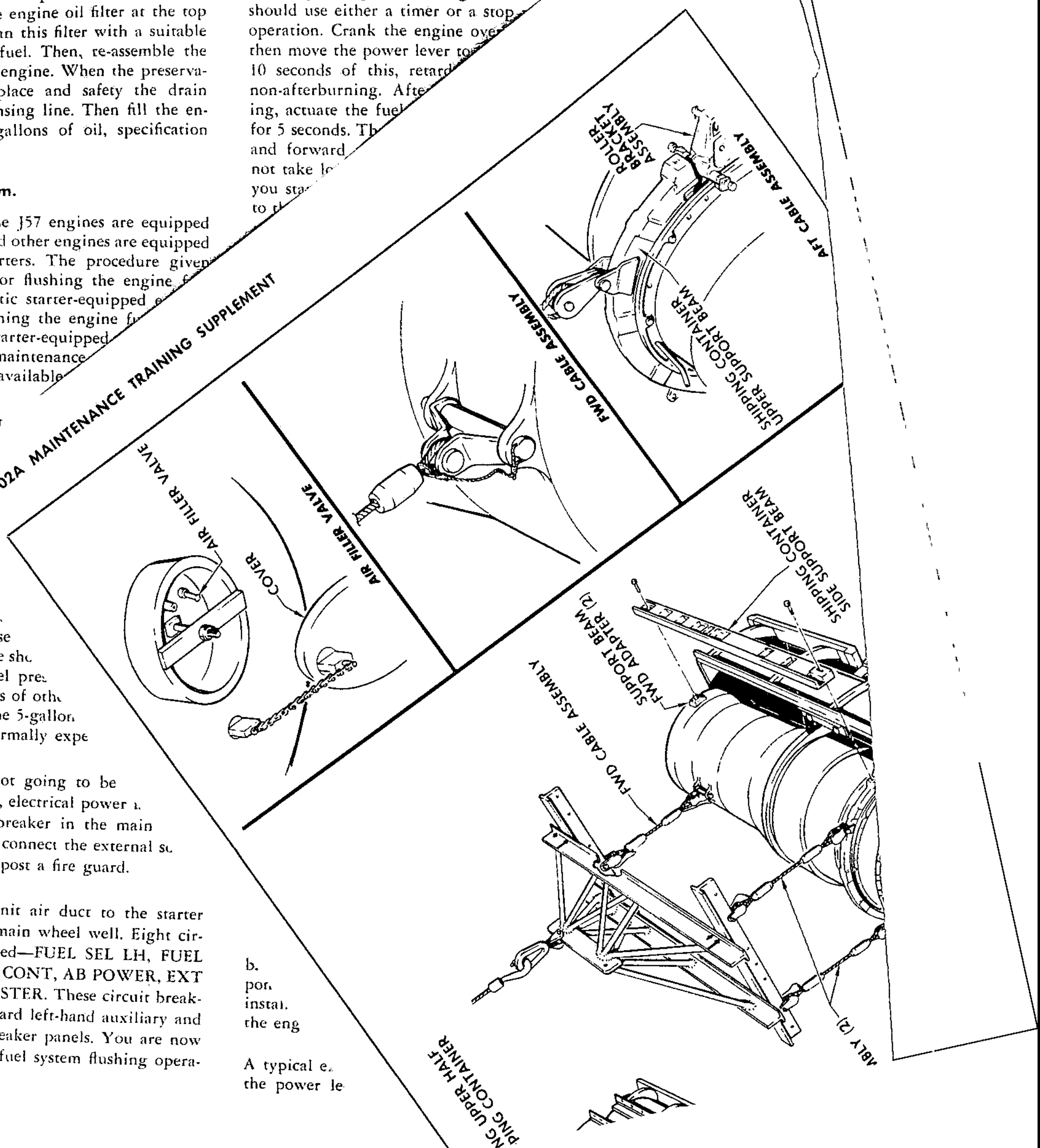

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until the oil flow slows to a slow drip at all of the openings, then re-install and safety the drain plugs. The engine oil filter on the top of the main gear box has to be removed, completely disassembled, and cleaned with a suitable cleaner (JP-4 fuel is satisfactory for this purpose); after being cleaned the filter should be reinstalled. When you have accomplished the preceding operations, service the engine oil tank with 5½ gallons of preservative mixture. For the exact type of preservative mixture, you should consult the controlling specifications and the engine maintenance technical order, (T.O. 1F-102A-2-4).

Run the engine up before you start on the fuel system preservation operation. To assure that the preservative mixture thoroughly lubricates the interior of the engine, operate the engine at 75% power for a period of ten minutes.

Preserving the Engine Fuel System.

After shutting the engine down, following the oil preservation run-up, you are ready to begin preserving the engine fuel system. Disconnect the pressurizing and dump valve sensing line at the fuel control unit, cap the fitting on the fuel control unit but leave the sensing line open to atmosphere. Place a 5-gallon drainage receptacle under the dump valve drain discharge point (located at the left of the engine accessory compartment access door); then disconnect the afterburner fuel line at the forward side of the afterburner manifold drain valve (located forward of the firewall). Connect an overboard discharge hose to the end of the disconnected afterburner fuel line and route it to another 5-gallon receptacle under the airplane. Disconnect the flexible fuel line to the engine fuel pump inlet port and connect a 2-inch flexible hose to the pump inlet port. Place the other end of the hose in a 10-gallon container of preservative mixture. As a special note, be sure to keep this 10-gallon container at least four feet above the pump inlet port. This will assure sufficient inlet pressure to the pump. If the container should become empty at any time during the following operation, shut off the engine starter and refill the container.

Next, pull out the ignition circuit breaker on the breaker on the main wheel well circuit breaker panel, connect the external source of a-c and d-c power, and post a fire guard. Now connect the GTC unit air duct to the starter air duct receptacle in the main wheel well. Eight circuit breakers must be closed—FUEL SEL LH, FUEL SEL RH, FUEL CONT, AB CONT, AB POWER, EXT AC PWR, OIL PRESS, and MASTER. These circuit breakers are located on the forward left hand auxiliary and main wheel well circuit breaker panels.

The rest of the fuel system preservation operation has to be timed (either a timer or a stop watch is satisfactory). Let us assume that you use a stop watch. Advance the power lever to the TAKE OFF position; have the ground operator start the GTC unit and adjust the unit speed to 100% run; start the stop watch at the same time the GTC unit reaches 100%. After 15 seconds have elapsed, move the power lever outboard to the AFTERBURNER detent. When the watch reads 25 seconds, re-position the power lever to FULL MIL POWER. After 30 seconds total time, actuate the fuel control switch to EMERGENCY. Five seconds later, place the fuel control switch in the NORM position and move the power lever to the OFF position and then forward to the TAKE OFF position at a slow, steady rate. After 60-seconds total elapsed time, simultaneously shut off the starter and bring the power lever back to the OFF position.

After the engine has stopped turning, disconnect the 2-inch flexible hose from the engine fuel pump inlet. Then start the GTC again, adjust its speed to 100%, and allow it to turn the engine over. With the engine turning, spray about one-half pint of preservative mixture over the compressor inlet section. Be sure to keep the spray gun about 18 inches from the compressor inlet, and keep moving the gun constantly to assure that the entire compressor entrance area is sprayed evenly. When you have completed the spraying, disengage the starter valve, shut down the GTC unit, and disconnect the starter from the engine.

This completes the preservation operation of the engine while it is installed in the airplane. You are now ready to proceed with engine removal.

HOW TO DEPRESERVE THE J57 ENGINE.

All jet engines shipped from overhaul depots or engine manufacturers are preserved, to prevent engine corrosion. As a result, you will have to flush the engine fuel system and drain the preservative from the oil system. No special fluid is needed for the fuel system flushing operation—regular JP-4 fuel is entirely satisfactory. The basic idea of the engine depreservation operation is to drain the oil and then to crank the engine over (without starting it), so that fuel will be forced through the engine fuel system and wash away the preservative. Before starting the depreservation operation, fill the airplane fuel tanks.

Depreservation of the Engine Oil System.

In comparison to the depreservation operation for the fuel system, the engine oil system depreservation operation is relatively simple. Place oil drain receptacles under the oil tank drain, the N_2 gear box near the oil temperature bulb connection, and the drain on the fuel/oil cooler. These are the same drains that were used to drain the oil in the engine preservation operation which was discussed earlier in this section. Remove each of the drain plugs and allow the preservative to run from the drains.

Disconnect the afterburner sensing line from the afterburner igniter and provide a receptacle for the oil from this line. Leave the drain plugs out until the oil discharge slows to a slow drip. Then, remove and completely disassemble the engine oil filter at the top of the main gear box. Clean this filter with a suitable cleaning solvent or JP-4 fuel. Then, re-assemble the filter and install it on the engine. When the preservative has drained out, replace and safety the drain plugs, and connect the sensing line. Then fill the engine oil tank with 5.5 gallons of oil, specification MIL-L-7808.

Flushing the Fuel System.

As mentioned earlier, some J57 engines are equipped with pneumatic starters and other engines are equipped with combustion-type starters. The procedure given in this training manual for flushing the engine fuel system is for the pneumatic starter-equipped engines only. Information on flushing the engine fuel system on the combustion-type starter-equipped engines will be given in the engine maintenance technical order when the information is available.

Several preparations must be made before you are ready to flush the engine fuel system. Some of the preparations are safety precautions and others are maintenance duties. Never omit *any* of these preparations. First, disconnect the pressurizing and dump valve sensing line at the fuel control unit and cap the fitting on the fuel control. The sensing line can be left open to atmosphere during the flushing operation. Next, disconnect the afterburner fuel line at the forward side of the afterburner manifold drain valve and attach a drain hose to the disconnected line. Place the free end of the drain hose in a 5-gallon receptacle. Another 5-gallon receptacle should be placed under the drain discharge of the fuel pressurization and dump valve. Although receptacles of other sizes can be used, it is best to have at least the 5-gallon size in case more fuel is discharged than normally expected.

Although the engine is not going to be started for the fuel flushing operation, electrical power is needed. Pull the ignition circuit breaker in the main wheel well circuit breaker panel, connect the external source of a-c and d-c power, and post a fire guard.

Now, connect the GTC unit air duct to the starter air duct receptacle in the main wheel well. Eight circuit breakers must be closed—FUEL SEL LH, FUEL SEL RH, FUEL CONT, AB CONT, AB POWER, EXT AC PWR, OIL PRESS, MASTER. These circuit breakers are located on the forward left-hand auxiliary and main wheel well circuit breaker panels. You are now ready to proceed with the fuel system flushing operation.

With the GTC unit operating at 100% speed, advance the power lever to the TAKE OFF position. Notify the operator of the GTC unit to open the starter air valve and begin engine cranking. From this point, you should use either a timer or a stop watch to time the operation. Crank the engine over for 15 seconds, and then move the power lever to AFTERBURNER. After 10 seconds of this, retard the power lever back to non-afterburning. After 5 seconds of non-afterburning, actuate the fuel control switch to EMERGENCY for 5 seconds. Then move the power lever back to OFF and forward to TAKE OFF. This movement should not take longer than 5 seconds. One full minute after you start the flushing process, retard the power lever to the OFF position and release the starter switch.

At this time the ground-assisting crew man should check the amount of fuel in the 5-gallon receptacles. A minimum of three gallons of fuel-oil mixture should have drained from the dump valve drain and about two gallons from the afterburner fuel line. If less than this has drained, you should repeat the flushing procedure. If the minimum amounts are in the receptacles, remove the hose from the afterburner fuel line, then reconnect the pressurizing and dump valve line and the afterburner fuel line.

The fuel-filled fuel control unit must be allowed to soak for a minimum of eight hours. This soaking period is required to assure satisfactory sealing of the internal packings,and to condition the internal synthetic rubber diaphragms. After the eight hour soaking period has elapsed, the engine is ready for run-up.

ENGINE BUILD-UP.

Each jet engine received from the manufacturer for installation in an airframe must have certain components and accessories installed, removed, and, in some cases, replaced. This operation of preparing an engine for airframe installation is commonly referred to as *engine build-up*.

Almost every type of jet engine is designed and manufactured for use in more than one particular model and type of aircraft. Since all types of aircraft are not designed the same, as far as size and available space are concerned, aircraft engines must have their accessories and components located in such a manner as to compensate for the design peculiarities of each particular type of aircraft. Because of this fact, it has become an accepted practice for the engine manufacturer to supply the airframe manufacturer with the basic engine that has just a few of the necessary components mounted. The airframe manufacturer then installs those other accessories and components which the engine will need in his type of aircraft.

A typical example of the need for engine build-up is the power lever cross-over shaft that is installed on the J57 engine by the engine manufacturer. This cross-over shaft must be replaced with another cross-over shaft by the airframe manufacturer. The original cross-over shaft is not compatible with the F-102A power lever mechanical operating system. Another example of this nature is the outer exhaust nozzle leaves on the earlier J57 engines with the "iris" type of exhaust nozzles. These leaves, like the cross-over shaft, were designed for engine installation on types of aircraft other than the F-102A. The engine shroud configuration on the F-102A makes these exhaust nozzle leaves unnecessary and they must be removed before the engine is installed.

Now that you have a general idea of the need for engine build-up, let's go through a sample engine build-up operation and prepare a J57 engine for installation in the F-102A.

UNPACKING THE J57 ENGINE.

The J57 turbojet engine is shipped from the factory in a pressurized metal container. In figure 3-1, the engine is shown in the shipping container as the cover is being removed. The skids on which the metal container is mounted provide a means for keeping the engine and container upright while the engine is being shipped.

Before you start to remove the engine container cover, check the dehydrating agent inside the container. A small humidity window in the end of the engine container allows you to visually check the small bag of dehydrating agent and determine whether the agent is blue or pink. A dehydrating agent with a solid blue color indicates that moisture has not entered the container, while a pink-colored agent indicates that moisture has entered the container and engine corrosion has probably already started. If the dehydrating agent is pink, the engine should be handled in accordance with the existing technical orders. Normally, though, you will find that the agent is colored blue, and after determining this, you can proceed with the container cover removal operation.

Always release the air pressure from the container before attempting to loosen the hold-down bolts and remove the cover. The relief valve for this purpose is situated on the end of the container adjacent to the humidity inspection window. In most cases, engine containers are pressurized to only about 5 psi so no great length of time is required to relieve the pressure.

After removing all of the cover bolts, use a chain hoist or other suitable hoist or sling assembly to lift the container cover off. Note in figure 3-1 how the sling assembly is attached to the engine to lift the engine from the container. The sling attaches to one front and two rear attach fittings. A special sling (SE-0945) will be furnished as special equipment with the F-102A to provide a more efficient means for removing the engine from the shipping container and for lowering it on the engine build-up stand. The SE-0945 sling has provisions for lifting the engine, the engine and afterburner, or just the afterburner itself.

The J57 engine is mounted on shipping rails inside the metal container. These shipping rails are secured to the container. In figure 3-1, note that there are four bolts in each rail. Prior to lifting the engine from the container, the nuts from these eight bolts must be removed. Figure 3-1 shows how the shipping rails are secured to the engine mounts.

After you have lifted the engine from the shipping container, keep the engine on the sling and remove the shipping rails from the engine. As shown in figure 3-2, removal of the rails allows the right front thrust mount and its roller, and the left front mount bracket and its roller, to be installed on the front of the engine. These rollers are used only for engine build-up, engine removal, and engine installation. As you will see later in this section, the rollers assist in moving the engine into the F-102A. All of the rollers must be removed after the engine has been installed in the F-102A. The engine can now be lifted onto the rails of the engine removal and installation trailer SE-0635. The engine is then ready for engine build-up.

UNPACKING THE J57 AFTERBURNER.

The afterburner is shipped from the factory in a wooden container. In figure 3-3, the container is shown with the top and one of the sides removed. Note that this container also has a center of balance marker on the container. The bolts shown in the end of the box in the right hand portion of figure 3-3 are inserted through the afterburner mating flange and the box. These bolts hold the afterburner firmly in place during shipment. Although not shown in the illustration, SE-0945 sling assembly is used to lift the afterburner from the box bottom and hoist it onto an SE-0730-803 afterburner adapter stand. This stand is also furnished as special equipment with the F-102A. It was designed for use with the Lockheed truck 205226 which is widely used by the Air Force. By using the Lockheed truck with the afterburner adapter stand, you can easily move the afterburner around. As you will find out from experience, this movability aids greatly in the assembly of the afterburner to the engine.

AFTERBURNER TO ENGINE BUILD-UP.

The adapter stand, the Lockheed truck, and the afterburner are shown in figure 3-4. Note that the afterburner rests in the SE-0730-803 adapter stand which in turn rests on the rails of the Lockheed 205226 truck. The engine, shown in the right portion of figure 3-4, is suspended in the SE-0635 engine removal and installation trailer.

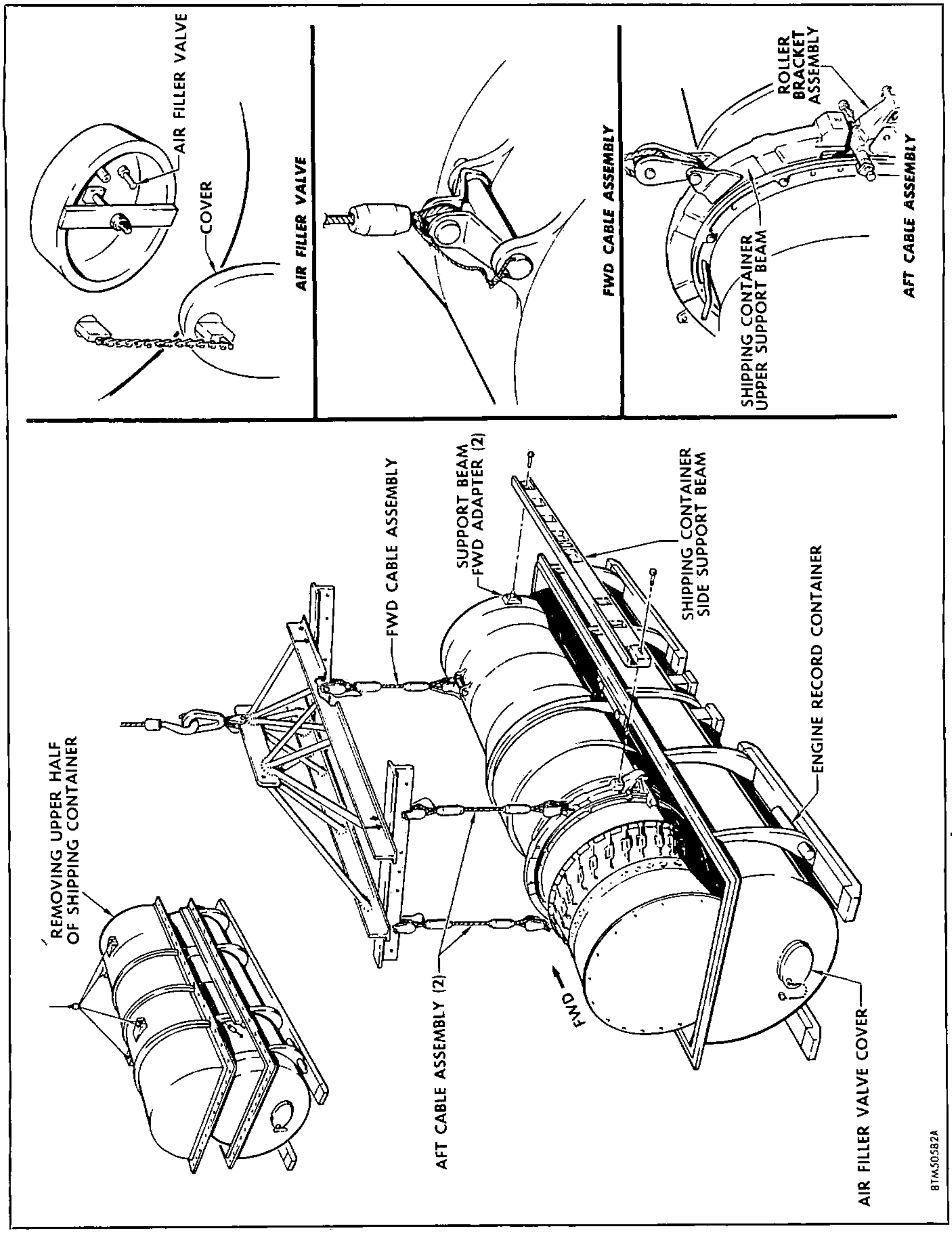


Figure 3-1. Unpacking the J57 Engine

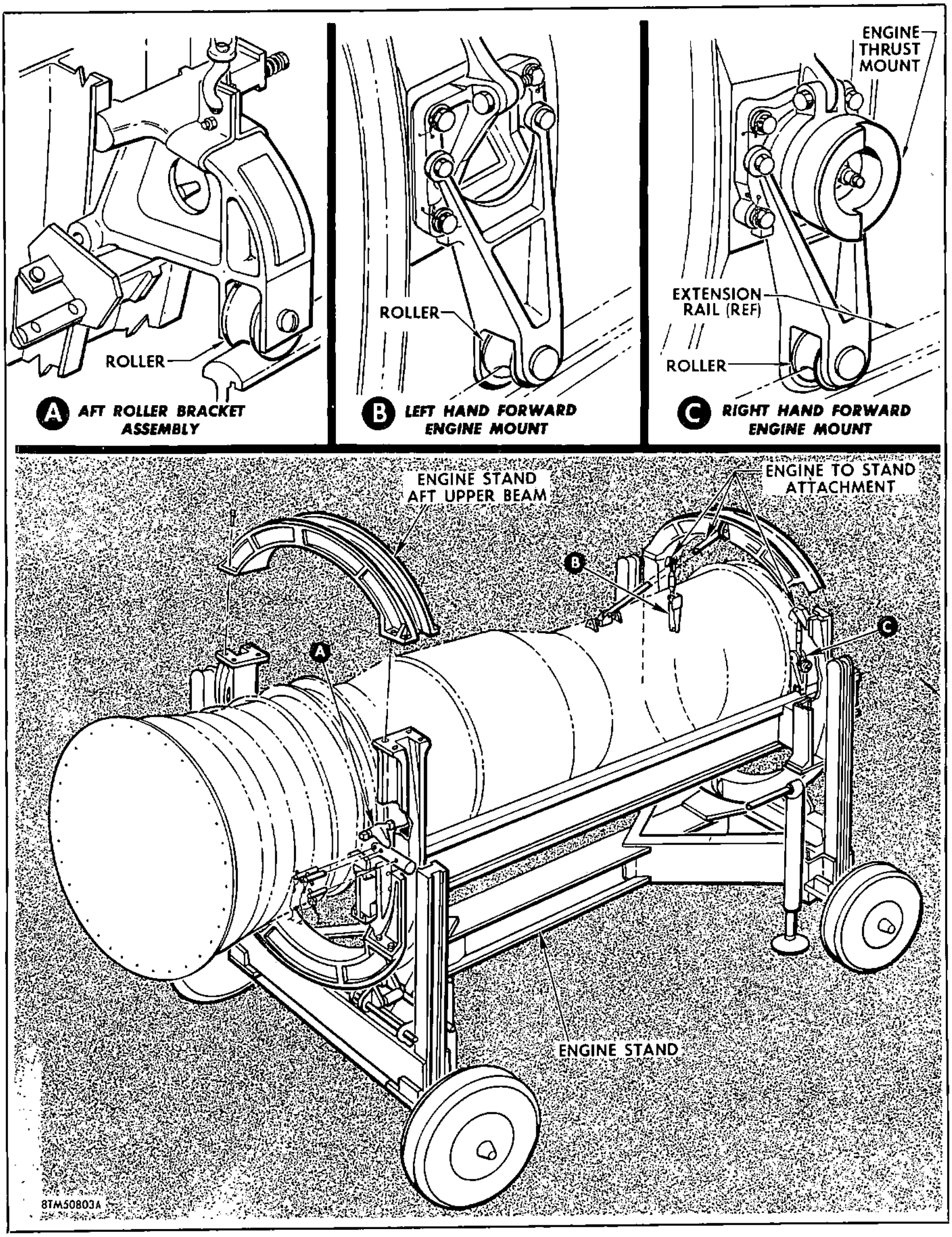


Figure 3-2. Preparing the J57 Engine for Engine Build-Up

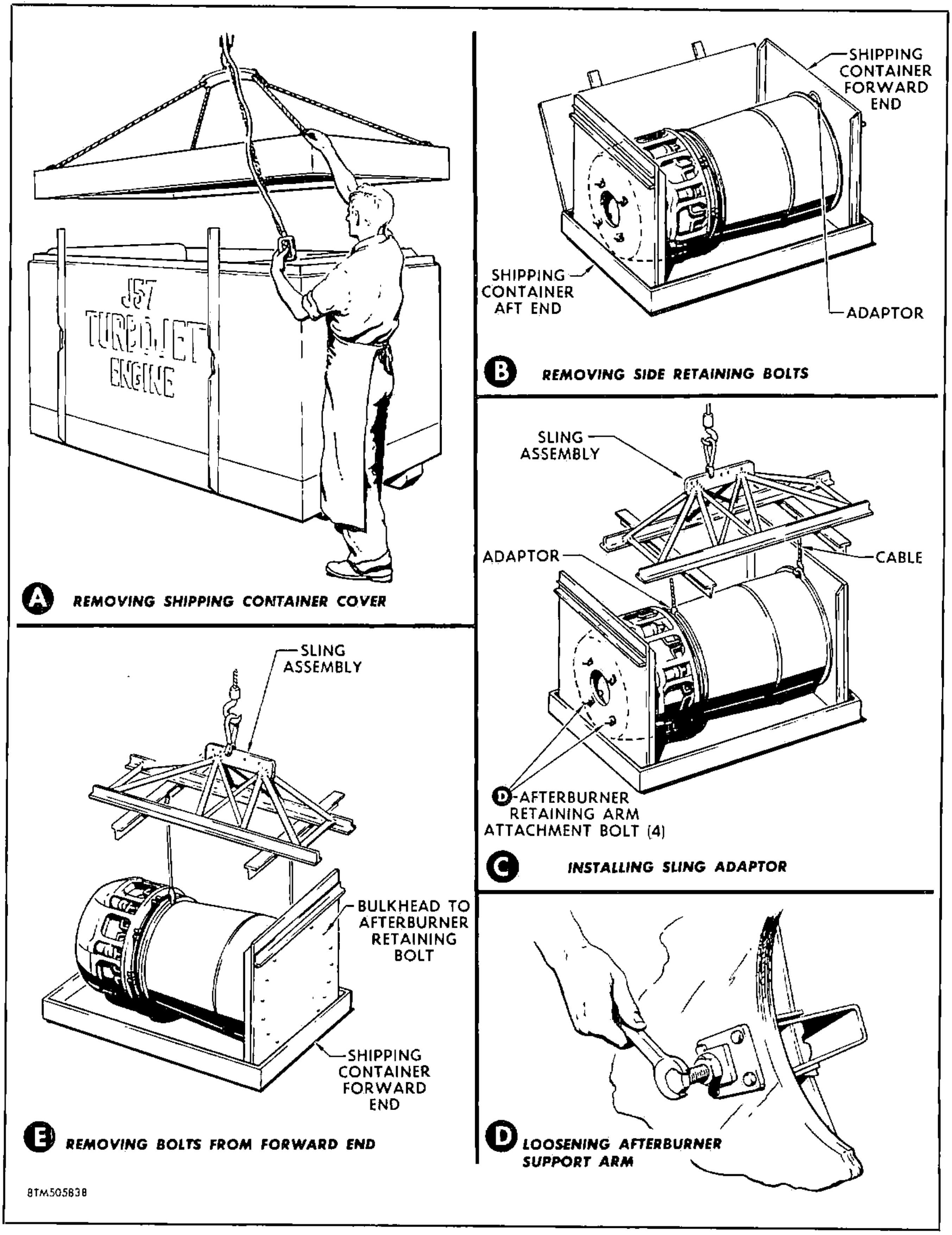


Figure 3-3. Unpacking the J57 Engine Afterburner

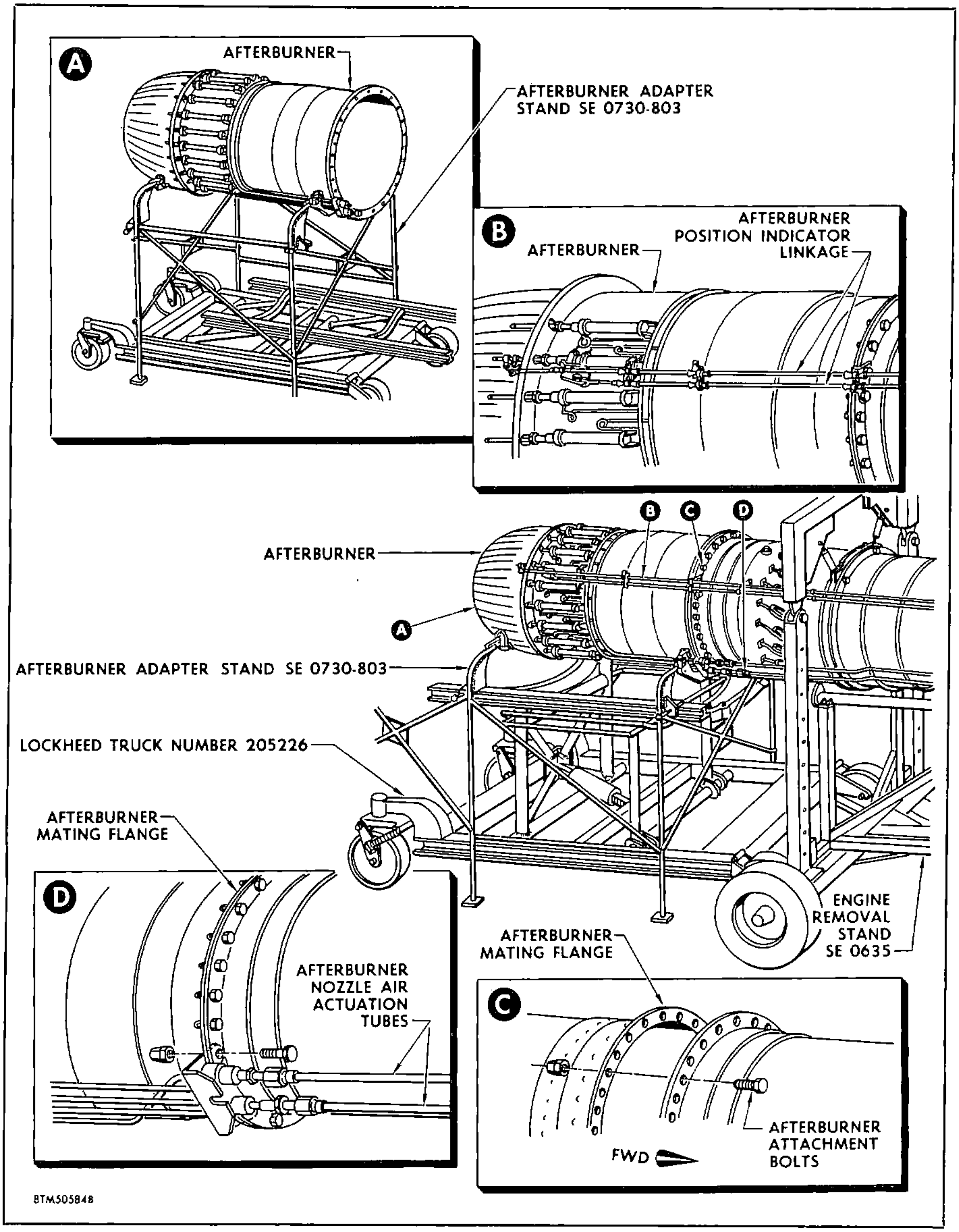


Figure 3-4. Installing the Afterburner

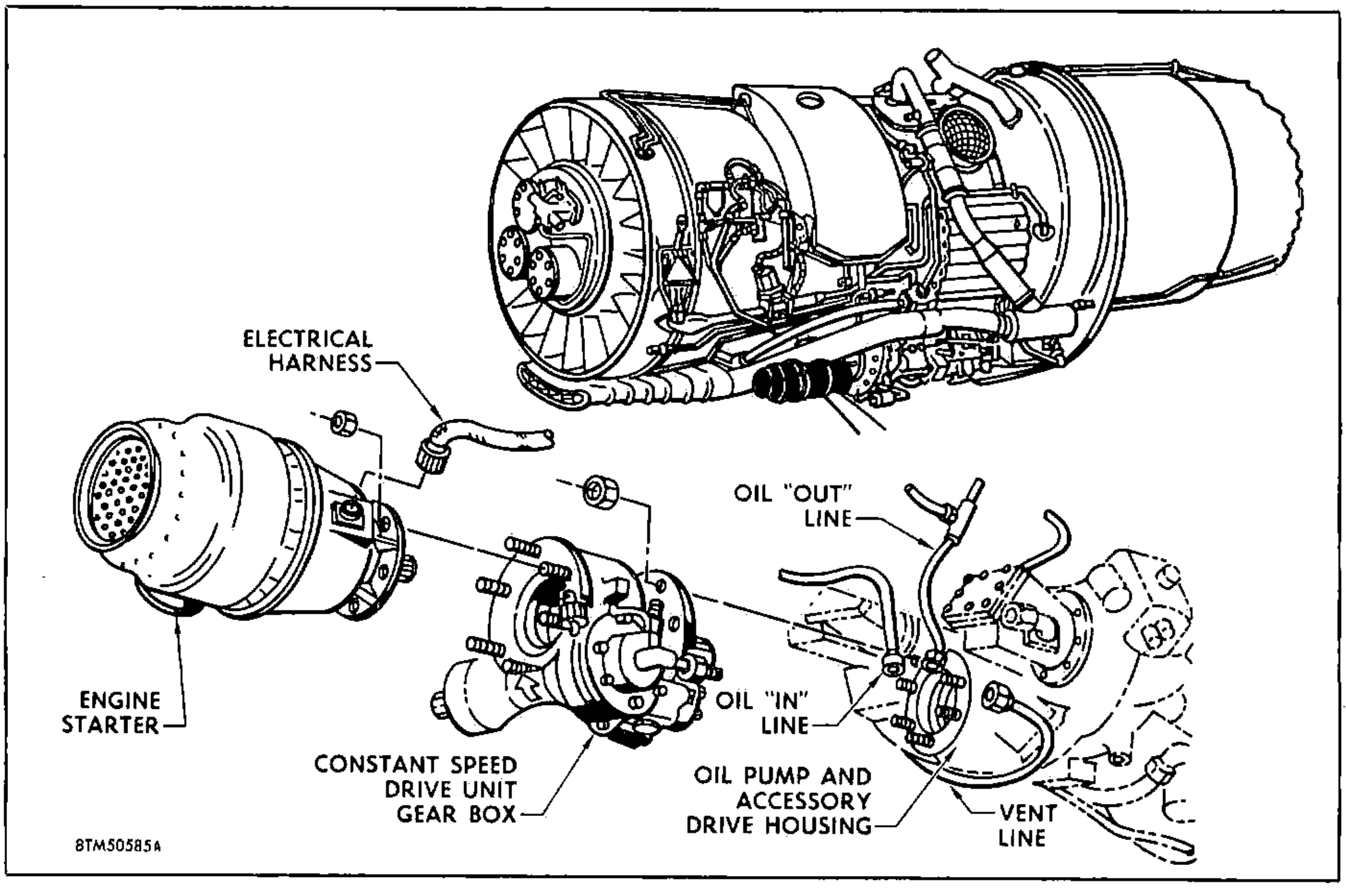


Figure 3-5. Sundstrand Gear Box and Engine Starter Installation

The afterburner installation is relatively simple. The adapter stand should be pushed up to the engine so that the afterburner mating flange aligns with the engine attach flange. Then install the attach bolts so that the bolt heads are towards the front of the engine as shown in view C of figure 3-4. When installing the attach bolts, you may find that a few bolts cannot be easily inserted in their proper holes. In this event, insert a drift pin in an adjacent hole and force the afterburner mating flange into the correct position—never hammer an attach bolt into its hole. You will probably find that there will always be a certain amount of "jockeying" involved in mating the afterburner with the engine attach flange; since there are 60 bolts to be installed, this is to be expected. Remember—never force the bolts into the holes.

In view B of figure 3-4, note how two of the attach bolts also hold the bracket for the afterburner nozzle air-actuation tubes. After these two bolts have been installed, the nozzle air-actuation tubes can be joined at their connection point just forward of the afterburner mating flange.

The afterburner attach bolts are torqued to approximately 70 inch-pounds. For the exact torque value, refer to the maintenance technical order, (T.O. 1F-102A-2-4). The attach bolts should be pattern-torqued; that is, you should start with the top bolt and work around in a counter-clockwise direction (looking from the front of the engine).

HOW TO INSTALL THE SUNDSTRAND ENGINE-MOUNTED GEAR BOX.

As you will recall from Chapter I, the Sundstrand constant-speed drive unit consists of two major components: an engine-mounted gear box; and an air-frame-mounted transmission and gear box. For the purpose of engine build-up, we are interested in only the engine-mounted gear box. Figure 3-5 shows the location of the gear box and how it is attached to the engine accessory drive case. Note that the gear box shaft is splined to the accessory drive. The gear box is secured on its mounting pad by six mounting studs.

The installation of this unit should be relatively simple—place the oil seal gasket over the mounting studs on the gear case, align the gear box shaft with the gear case spline, and then slip the gear box over the six mounting studs. Tightening the six nuts and connecting the oil IN and OUT lines completes the gear box installation.

HOW TO INSTALL THE ENGINE PNEUMATIC STARTER.

The engine starter is mounted on the forward face of the Sundstrand gear box as shown in figure 3-5.

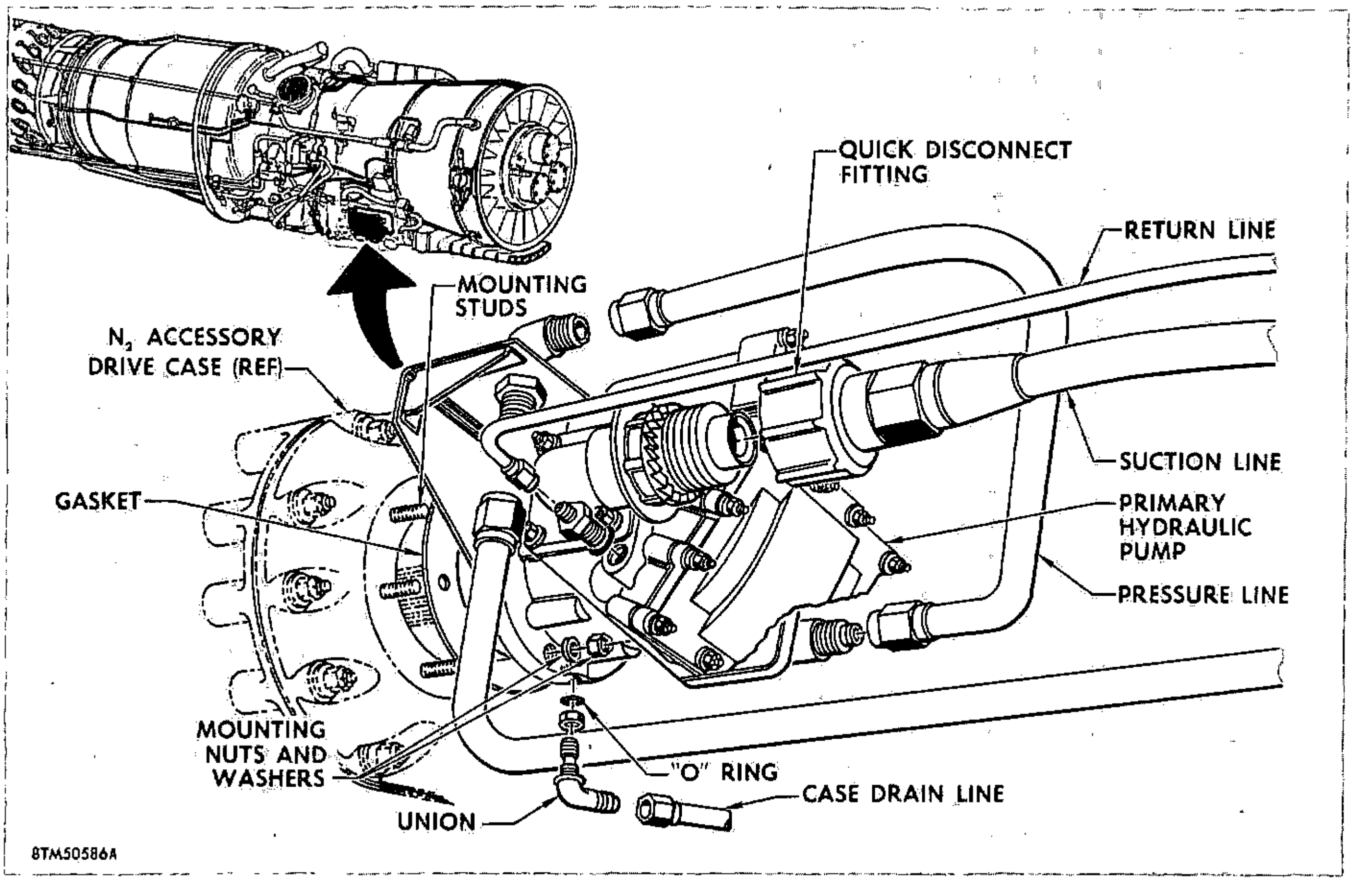


Figure 3-6. Primary Hydraulic Pump Installation

Note that the starter installation is just about the same as the gear box installation. After placing the gasket over the studs on the gear box, align the starter shaft spline with the gear box spline and slip the starter over the mounting studs. Then tighten the hold-down nuts and connect the electrical plug to the starter.

On some of the later model F-102A airplanes, a combustion-type starter is used. The installation of the combustion-type starter is quite similar to that of the pneumatic starter shown in figure 3-5. A starter mounting head (adapter) attaches to the constant-speed gear box and the combustion-type starter then attaches to the mounting head. In addition to the electrical connection, the combustion-type starter also has a starter air line and fuel line which must be attached. Complete information on the combustion-type starters will be furnished in the J57 engine maintenance technical order (T.O. 1F-102A-2-4), when the starters are furnished with the engine.

HOW TO INSTALL THE PRIMARY AND SECONDARY HYDRAULIC PUMPS.

Two hydraulic pumps must be installed on the J57 engine during engine build-up—the primary and secondary pumps. Each hydraulic pump supplies approximately 3000-psi hydraulic pressure to its respective hydraulic system. The two pumps are identical in appearance and function, and they are interchangeable. Note in figures 3-6 and 3-7, that the hydraulic pumps are mounted on the right and left side of the engine on the forward face of the engine accessory drive case. Each pump is splined individually to an accessory drive in the case.

A nebular spline must be used to install the pump. The nebular spline is a female spline adapter which fits over both the hydraulic pump spline and the engine spline. Usually, you will find it best to slip the spline adapter over the pump spline and then align the adapter with the engine spline. This will allow you to slip the pump over the six mounting studs. Tightening the bolts and connecting the hydraulic lines finishes the pump installation.

As a special note, always remember to fill the pump housing with hydraulic fluid, Specification MIL-0-5606, after you have installed the pump on the engine. The pump should be filled through the case drain line connection point.

Although the installation for only one of the pumps has been described, the installation procedure is just the same for the other. As mentioned, the primary and secondary hydraulic pumps are interchangeable.

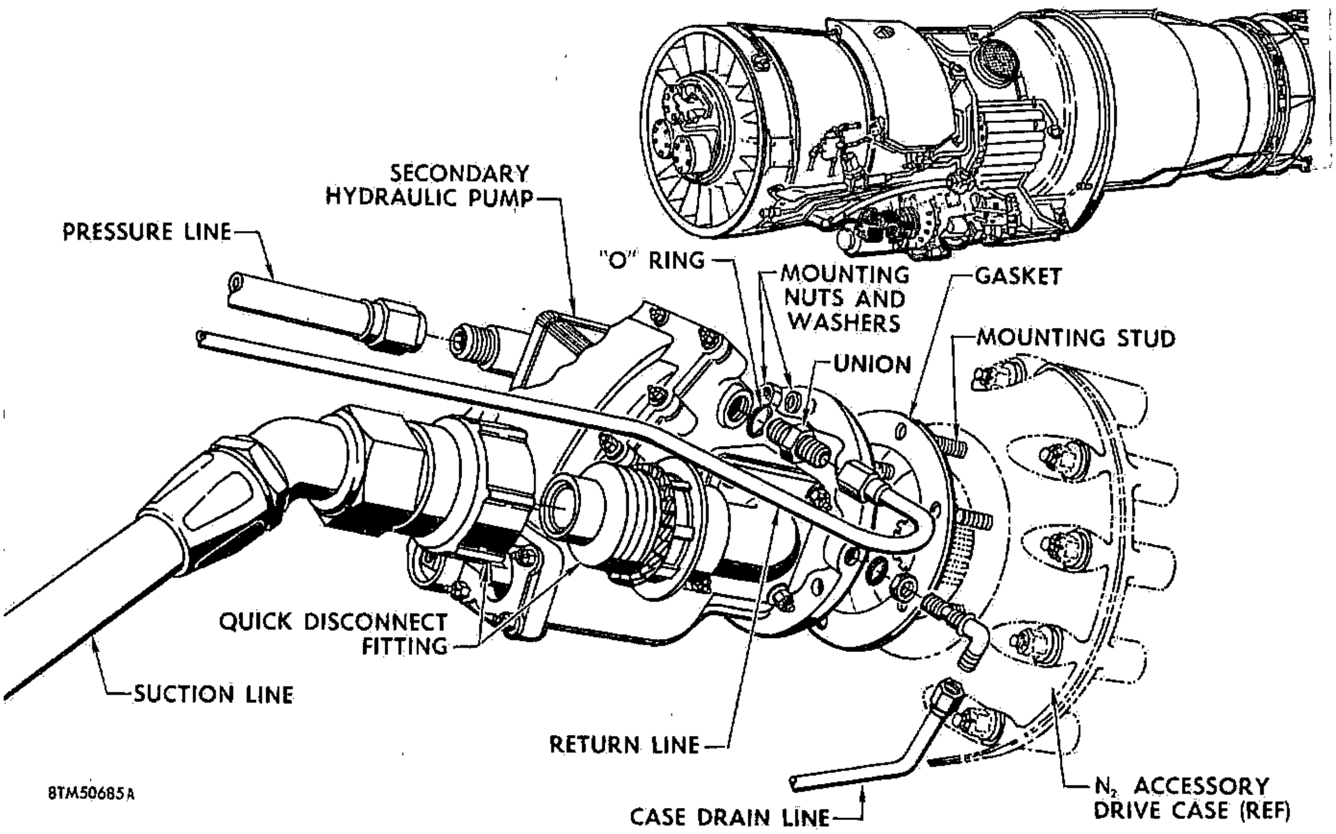


Figure 3-7. Secondary Hydraulic Pump Installation

The only difference is that either pump must be turned over before it can be installed on the other pump mounting pad. The primary pump, for example, must be rotated 180° when it is used as a secondary pump, to insure that the pump fittings align with the attaching hydraulic lines.

Comparing the installation of the two pumps, note that the case drain line for the primary pump connects to the bottom of the primary pump housing. The case drain line for the secondary pump connects to the side of the pump housing. The mounting face on each pump, primary and secondary, has four drain holes, but only one is used. The other three drains are plugged. This type of design permits the pumps to be installed on any type of engine and the most convenient drain opening can then be used.

HOW TO INSTALL THE TACHOMETER GENERATOR.

The tachometer generator is installed on the engine to furnish an indication of total engine rpm to the pilot on the tachometer instrument in the cockpit. The tachometer instrument is calibrated in percent of engine N_2 compressor rpm. The tachometer generator is installed on the accessory drive case immediately below the right hydraulic pump and the engine starter, as shown in figure 3-8.

The installation of the tachometer generator is a typical engine component installation and should not create any problem. Note that the generator is splined to the accessory drive in the case and is mounted on four studs. When installing the generator, place the gasket over the studs on the gear case and align the generator splined shaft with the spline in the accessory gear case. This will allow you to slip the generator down over the four mounting studs and tighten the hold-down nuts. Connecting the electrical connection on the generator and safetying the connector completes the installation.

HOW TO INSTALL THE POWER LEVER CROSS-OVER SHAFT.

The engine manufacturer's power lever cross-over shaft and bellcrank must be replaced with an assembly designed by the airframe manufacturer. This is necessary because the original cross-over shaft and bellcrank will not fit the F-102A's airframe-mounted power lever mechanical linkage system.

Figure 3-9 shows how the cross-over shaft and bellcrank ride in two bearings. Note how the shaft and bellcrank are secured to the bearings with collars and clevis pins. The bellcrank assembly slips over the end of the cross-over shaft on the left side and is clevis-pinned to the shaft. This makes the shaft and bellcrank one solid unit.

HOW TO INSTALL THE OIL LOW-PRESSURE WARNING SWITCH.

The oil low pressure warning system on the J57 engine notifies the pilot of dangerously low oil pressure conditions in the engine oil system. The oil low-pressure warning switch is actuated by a drop in the oil system pressure. When oil pressure drops occur, the switch closes and completes the electrical circuit to the oil low-pressure warning lights in the cockpit. When the pressure again rises, the switch opens and breaks the circuit.

The warning switch is installed on two mounting brackets between the N_1 and N_2 compressors on the left side of the engine as shown in figure 3-10. Two of the compressor flange bolts must be used to secure each pressure switch warning bracket. Tightening the switch bolts, and then connecting the electrical lead and the two one-fourth inch tubes, completes the installation of the switch. The operation of the warning systems and their components will be discussed in detail in Chapter IV.

HOW TO INSTALL THE FUEL LOW-PRESSURE WARNING SWITCH.

The fuel pressure warning switch actuates the warning light which notifies the pilot of a dangerously low

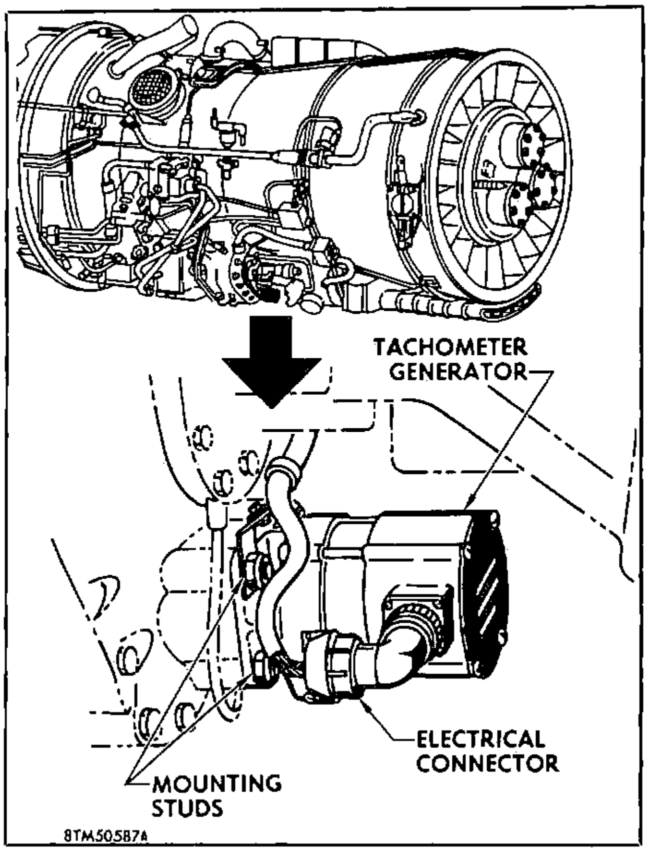


Figure 3-8. Tachometer Generator Installation

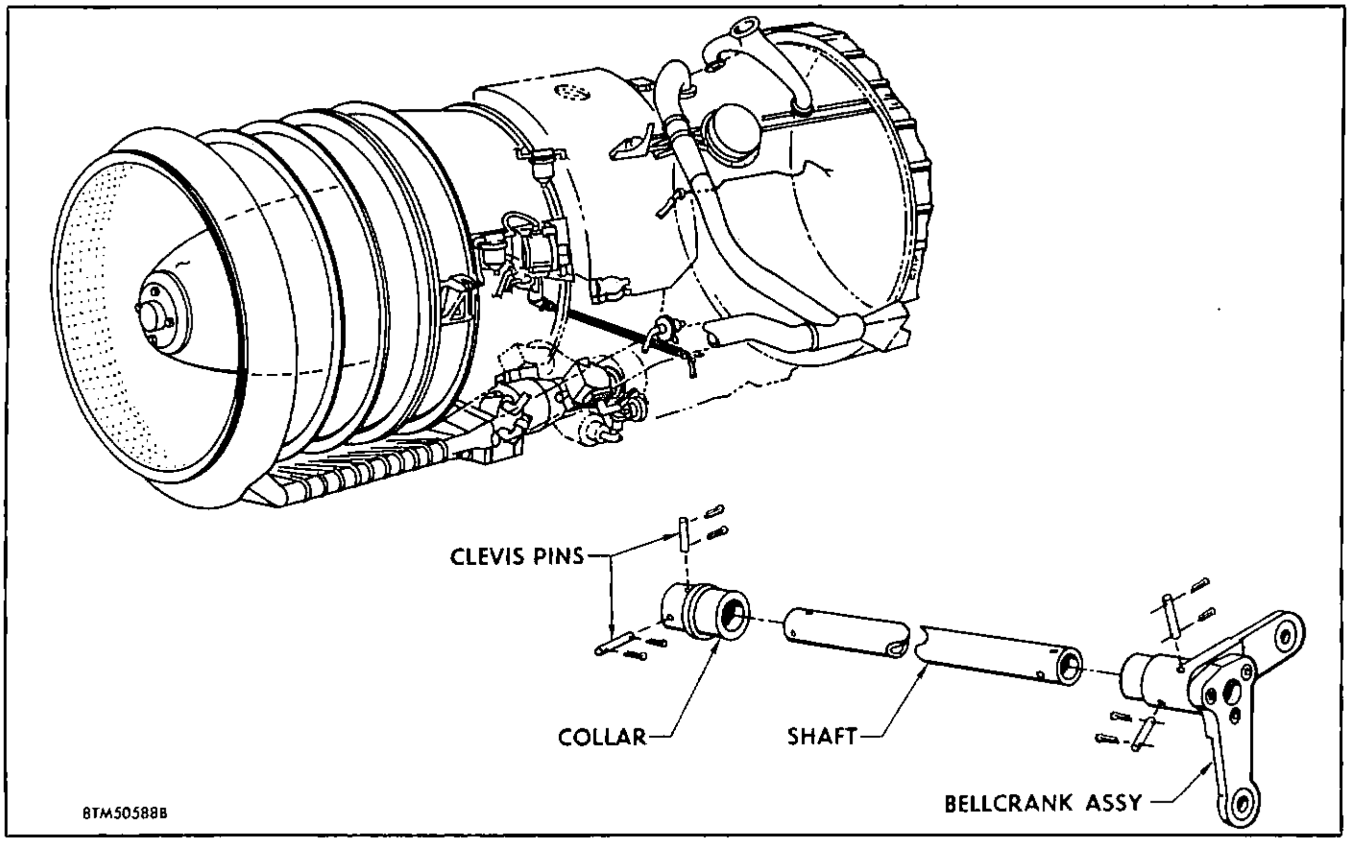


Figure 3-9. Engine Cross-Over

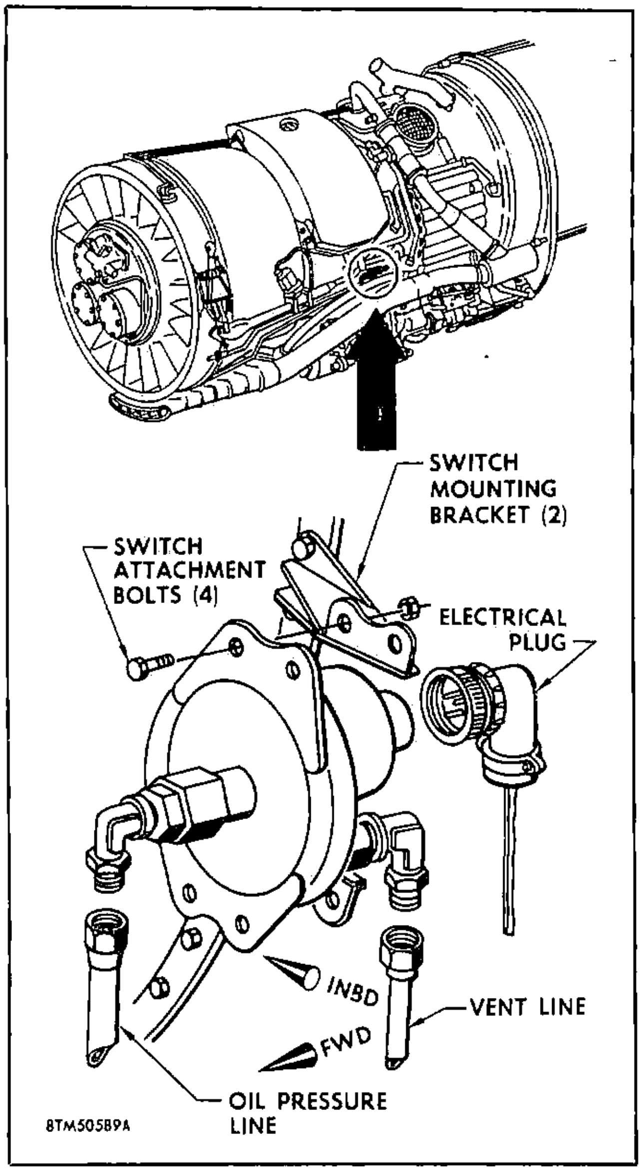


Figure 3-10. Oil Low-Pressure Warning Switch Installation

fuel pressure condition in the engine fuel system. When a drop in fuel pressure occurs, the switch closes and completes the electrical circluit to the warning light in the cockpit. When the fuel pressure rises the switch opens and the warning light goes out.

Note in figure 3-11 that the switch and mounting bracket are attached by bolts at three points on the lower left side of the accessory drive case. The fuel low-pressure switch has two tubing connections, the fuel pressure sensing line and the fuel drain line. Connecting these two lines and the electrical connection completes the low-pressure switch installation.

HOW TO INSTALL THE FUEL FLOWMETER.

The fuel flowmeter measures how much fuel the engine is consuming per hour. Note in figure 3-12 that the flowmeter is installed on the right side of the engine between the exhaust nozzle control valve and the main fuel control unit. The flowmeter is secured to the N_2 compressor case by three attach bolts. The arrows on the attach points on the flowmeter indicate the correct attachment of the fuel OUT and IN lines. The electrical lead that connects to the flowmeter runs to the fuel flow indicator in the cockpit. The internal details of this fuel flowmeter and the indicating instrument will be covered in Chapter IV.

HOW TO INSTALL THE AFTERBURNER NOZZLE POSITION SWITCHES.

The afterburner nozzle position switches detect the position of the afterburner nozzle. The switches then complete an electrical circuit to the indicating instrument in the cockpit. Note in figure 3-13 (detail B)

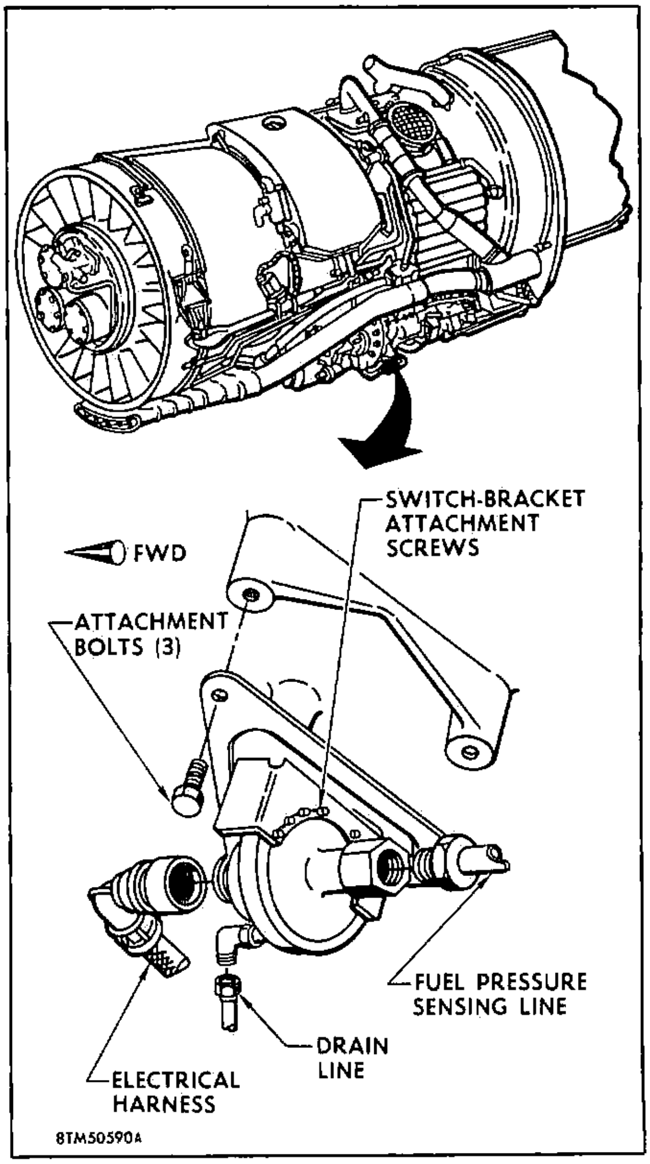


Figure 3-11. Fuel Low-Pressure Warning Switch Installation

that the switches are secured on the engine adjacent to the anti-surge bleed valve on the right side of the engine. In detail B of figure 3-13, you will not that a nozzle position cable connects to the position switch. The movement of the afterburner nozzle causes the switch cam to depress one of the switches. The indicator in the cockpit will then show that the afterburner nozzle is open or closed, depending upon whether the OPEN or CLOSED switch is actuated by the cam.

Although the switch installation is the same on both the -41 and the -23 engines, the position cable attachment at the exhaust nozzle is slightly different on the two engine models. This is shown in detail A of figure 3-13. On the -23 engine the cable bracket attaches directly to the nozzle door, while on the -41 engine the cable bracket attaches to the actuating cylinder. The operational and functional details of the afterburner nozzle position indicator switches and the indicating instrument will be covered in Chapter IV.

HOW TO INSTALL THE SUNDSTRAND CONSTANT-SPEED DRIVE OIL FILTER.

The oil filter for the constant-speed drive unit insures that no foreign material enters the engine main oil system through the drive unit oil system. This oil filter is mounted on the upper right side of the N_1 and N_2 compressor attach flange. Three bolts on the compressor flange must be "picked up" to secure the filter bracket. Note in figure 3-14 how the marman clamp holds the oil filter in place. The oil OUTLET and INLET lines attach to the filters by standard AN fittings.

HOW TO INSTALL THE ENGINE AIR INTAKE STUB DUCT.

The engine air intake stub duct is designed to direct air into the engine and also to bleed off cooling air for certain parts of the engine. Installation of the intake duct is shown in figure 3-15. Note that the duct attaches to the front frame of the engine. It is common practice to insert the attach bolts so that their heads are towards the front of the engine. The attach bolts must be torqued (as mentioned earlier in this chapter); all torque values for engine build-up should be obtained from the engine maintenance technical order, (T.O. 1F-102A-2-4).

HOW TO INSTALL THE AIR/OIL COOLER AND DUCTING.

The air/oil cooler cools the engine oil. As shown in figure 3-16, the cooler is mounted on the engine just forward of the engine starter. The cooler is secured to the engine with 12 bolts, six of which are engine flange attach bolts. The oil INLET and OUTLET lines connect to the left side of the cooler as shown. The other lines, shown in figure 3-16, routed through the

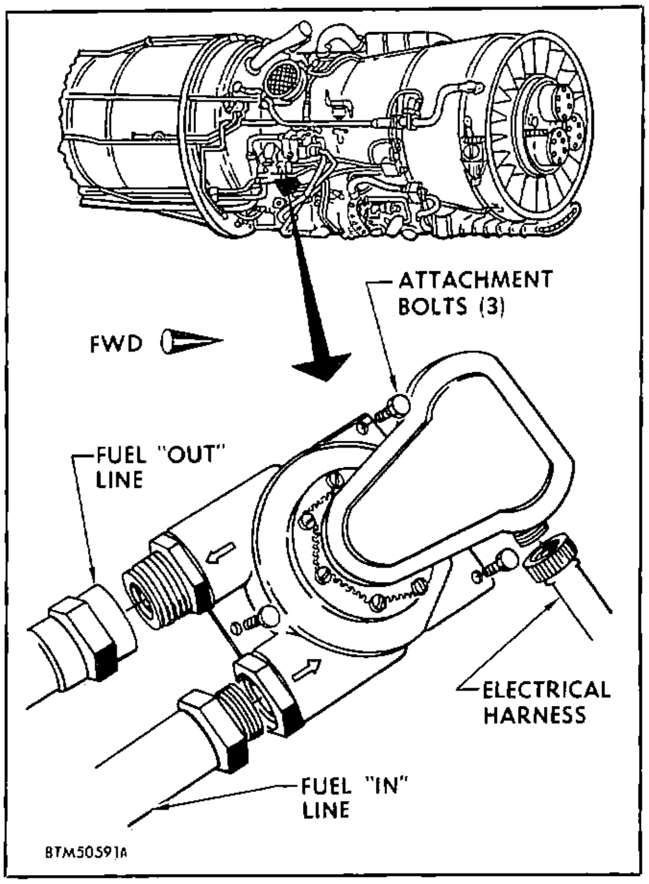


Figure 3-12. Fuel Flowmeter Installation

cooler bracket on the right side, belong in other systems. These lines use the cooler bracket as a stabilizer to prevent excessive vibration or chaffing of the lines Note in figure 3-16 how the air inlet duct attachs to the air/oil cooler. The duct flange must go inside the air/oil cooler flange.

HOW TO INSTALL THE ALTERNATE COOLING AIR VALVE.

The alternate cooling air valve assures that the engine receives cooling air at low air speeds or when the engine is operating on the ground. The air valve and its accompanying ducting are installed near the top of the engine on the N_2 compressor case. Note in figure 3-17 that the ducting for the air valve is made of flexible braided tubing. One end of the air valve ducting attaches to the engine shroud cooling duct; the other end of the ducting attaches to the air duct from the N_1 compressor case. Two flexible valve clamps hold the air valve securely in place between the two sections of braided ducting. Connecting the electrical lead and safetying it completes the installation of the alternate cooling air valve.

HOW TO INSTALL THE GROUND COOLING AIR DUCT CHECK VALVE.

As you will note in figure 3-18, this check valve has three openings—a ground cooling intake (from the

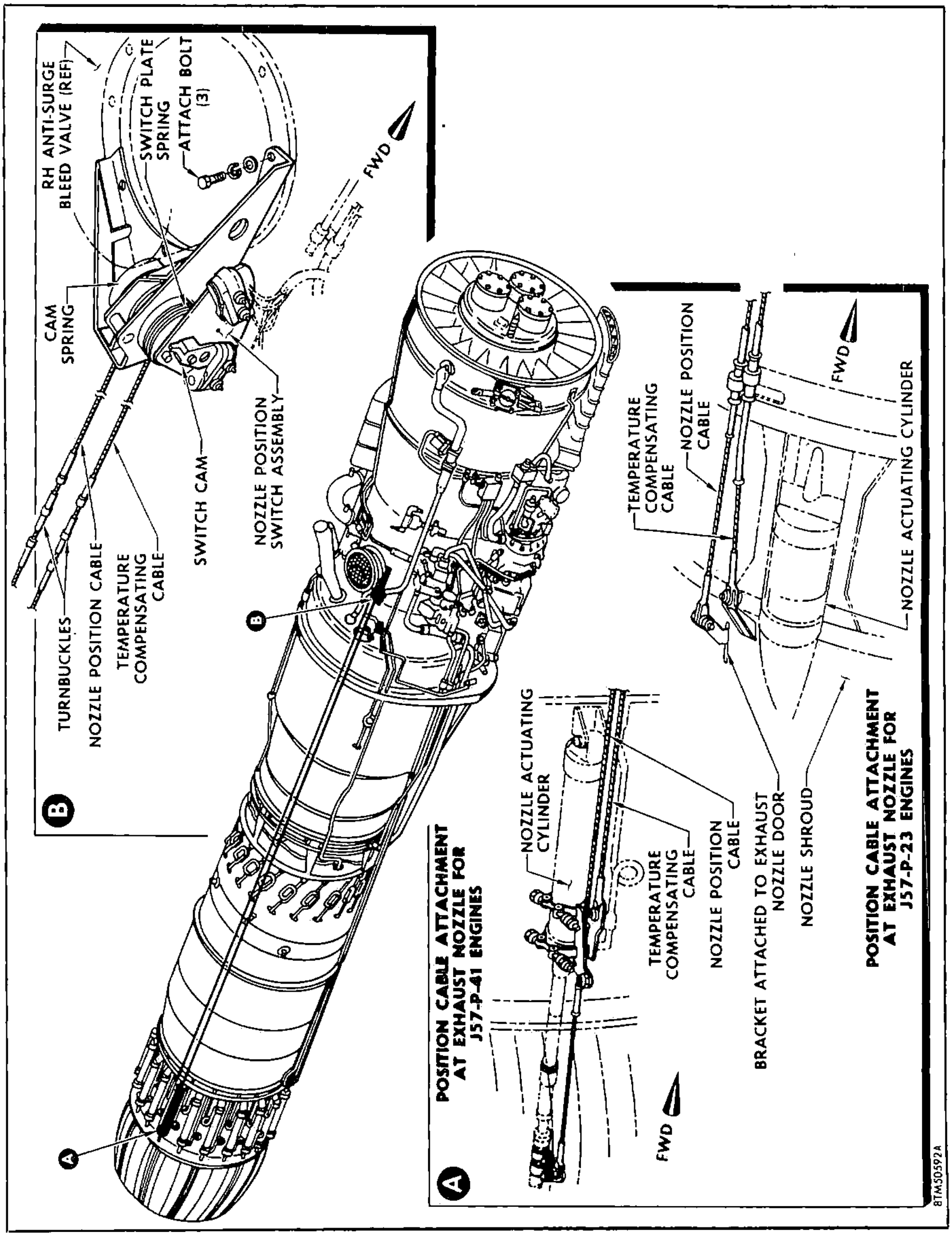


Figure 3-13. Afterburner Position Indicator Switch Installation

air valve just discussed), an in-flight cooling intake, and an outlet to the transition duct on the engine shroud. The check valve is mounted on the left side of the engine just below the fuel/oil cooler. The aft marman clamp secures the check valve to the engine. Note in figure 3-18, that the ground cooling air supply duct attaches to the check valves by means of two bolts. The smaller duct from the air supply ducting attaches to the check valve by means of a small marman clamp, while the outlet of the check valve attaches to the transition duct by means of a slip joint connection. This type of joint allows for duct expansion due to engine heat.

HOW TO INSTALL THE LOW-PRESSURE PNEUMATIC BLEED DUCT.

This low pressure pneumatic bleed duct is installed as a source of low-pressure pneumatic system pressure for operating various components in the airplane. The bleed duct is mounted on the top of the engine at the aft end of the N_2 compressor case as shown in figure 3-19. It is secured to the engine by three attach bolts at each connection point.

HOW TO INSTALL THE COMPRESSOR BLEED VALVE ADAPTER.

The compressor bleed valve adapters are installed directly under the bleed valves that are furnished with the engine. The bleed valves themselves are installed on the upper right and left sides of the engine. Although the J57-P-41 engine has two bleed valves—one on each side of the engine; the -23 model engine has only the left hand bleed valve installed.

Figure 3-20 shows a typical -23 engine. Note the adapter installed dircetly under the bleed valve. The airframe manufacturer has provided this adapter so that the bleed valve will reach the opening in the airframe. When you first receive an engine for engine build-up, you will have to remove the bleed valve from the engine and then secure the bleed valve

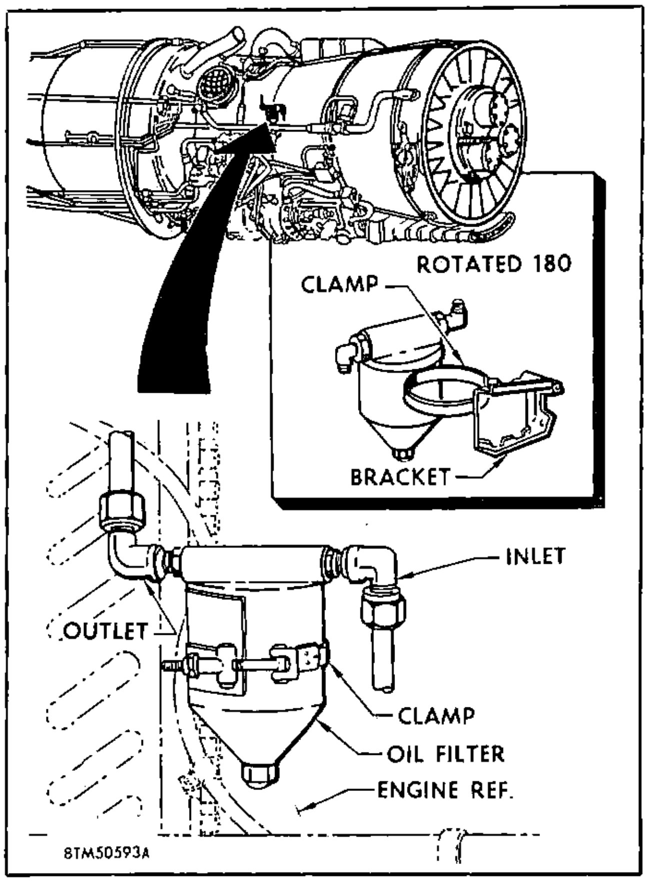


Figure 3-14. Sundstrand Drive Oil Filter Installation

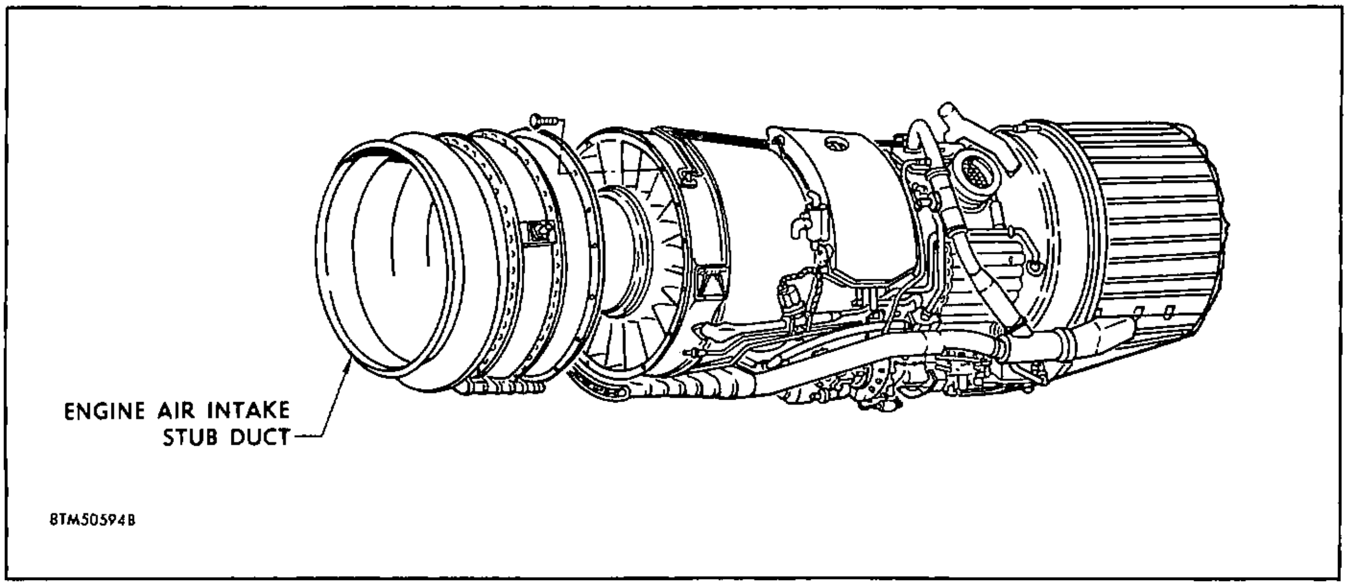


Figure 3-15. Engine Air Intake Stub Duct Installation

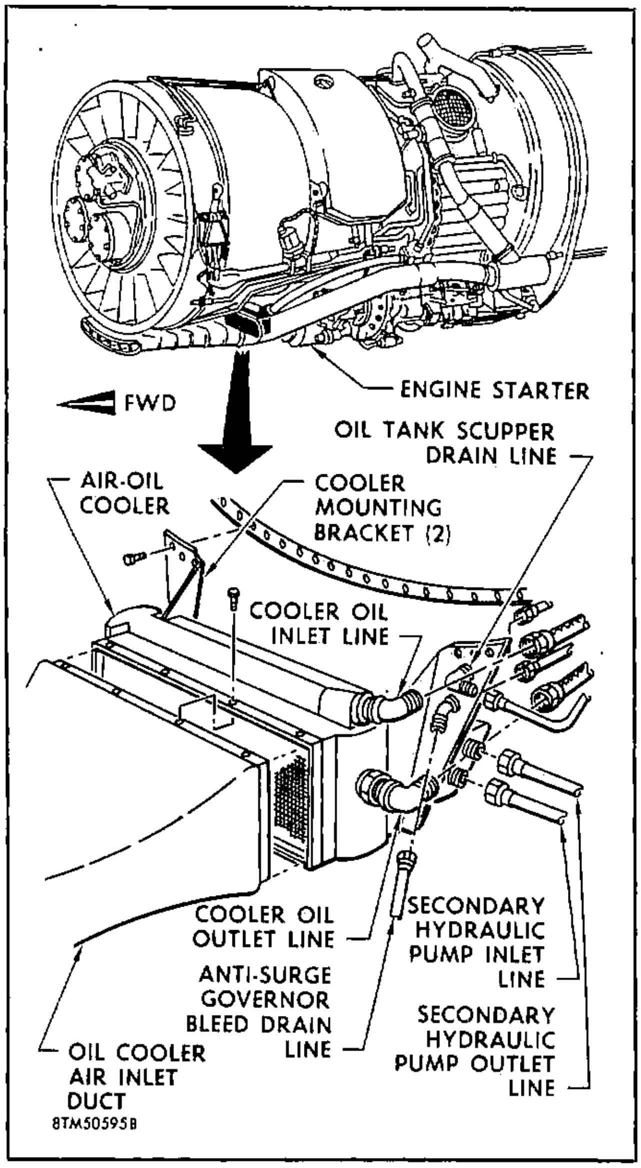


Figure 3-16. Air/Oil Cooler Installation

adapter with the same bolts that you took out when you removed the valve. The adapter then attaches to the bleed valve by means of the marman clamp as shown in figure 3-20. Three ¼-inch tubes connect to the fittings on the side of the bleed valve actuator.

HOW TO INSTALL THE EXHAUST NOZZLE ACTUATOR INSULATING BLANKET.

The exhaust nozzle insulating blanket is used only on J57 engines with the "iris" type afterburner exhaust nozzles. The blanket insulates the afterburner actuating components from the engine heat. The insulating blanket is fitted around the inside of the tailcone under the exhaust nozzle actuators, and it is laced in place with common safety wire. You will find that it is not necessary to remove the afterburner actuating mechanism to install the insulating blanket. Just loosen the forward attach bolts on the actuating cylinder so that the blanket can be slipped around the tailcone. For further information on the installation of the insulating blanket, refer to thhe engine maintenance technical order, T.O. 1F-102A-2-4.

HOW TO INSTALL THE ENGINE SHROUD.

The engine shroud provides efficient engine cooling and protects the airframe structure from excessive engine heat. As you can see from figure 3-21, the shroud is divided into two sections. The forward section of the shroud, with its transition duct, is positioned around the engine at the forward fire seal and is secured in eight places by eye bolts and nuts. The aft section is secured to the forward section by a large marman clamp. Note that the aft section is also supported in two places by two engine shroud support

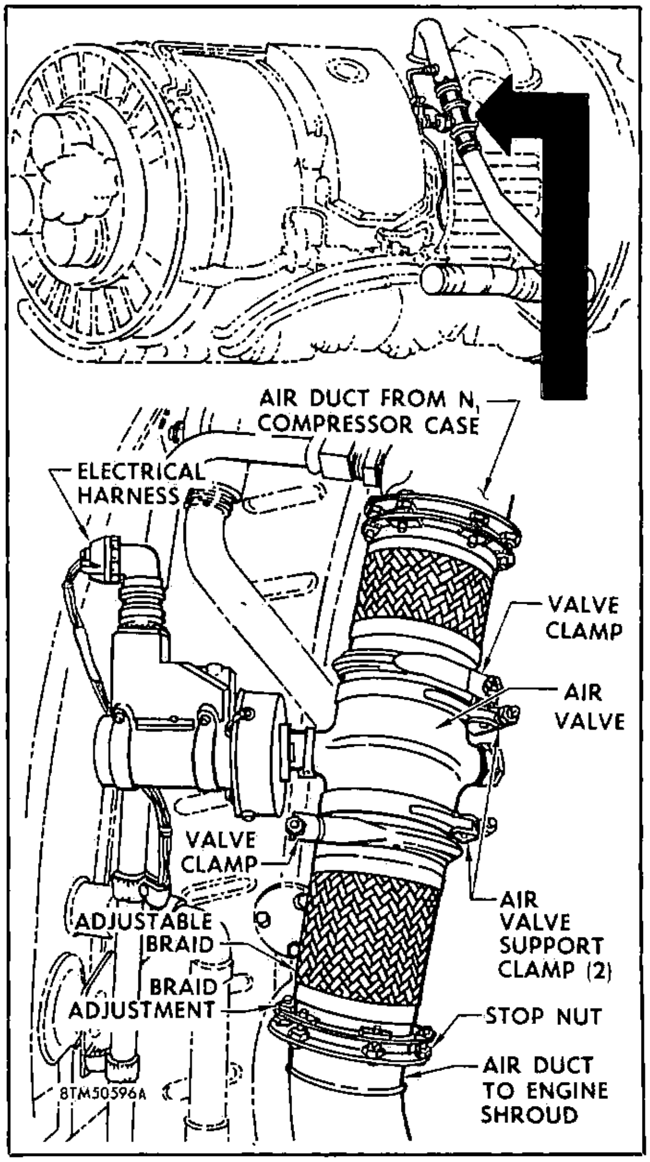


Figure 3-17. Alternate Cooling Air Valve Installation

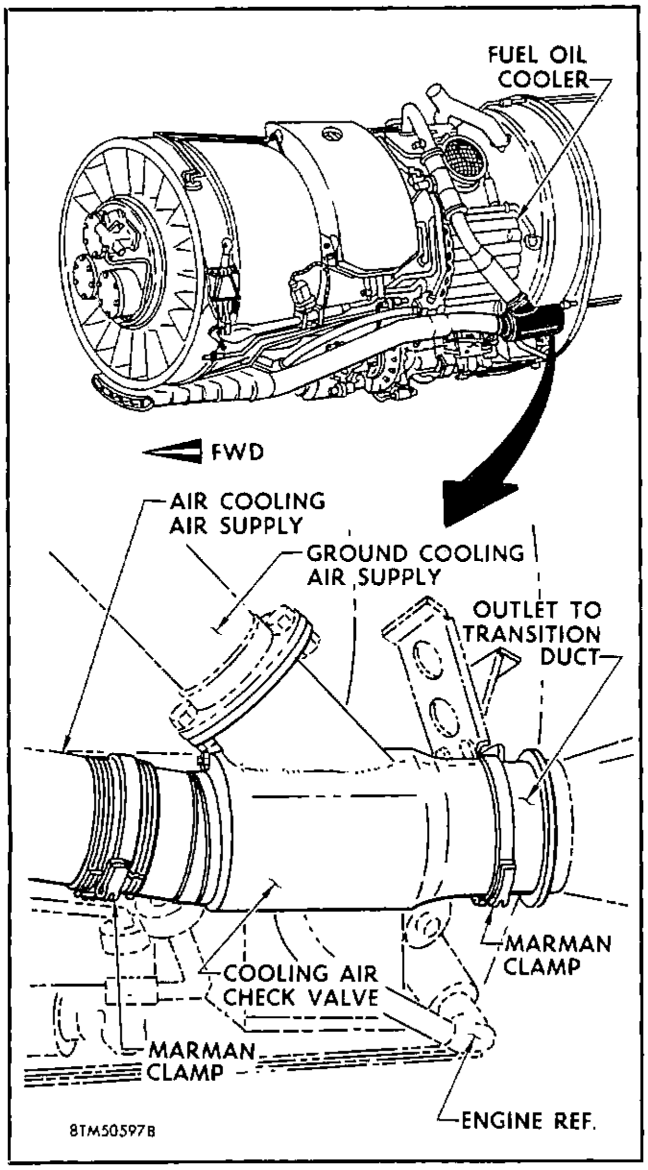


Figure 3-18. Cooling Air Duct Ground Check Valve Installation

brackets and turnbuckles. These two supports are accessible through sliding access doors in the shroud. The aft end of the engine shroud must be held away from the engine by support blocks during engine biuld-up. The four regular support turnbuckles hold the shroud away from the afterburner when the engine is installed in the aircraft.

HOW TO INSTALL THE N_1 ACCESSORY FAIRING.

The N_1 accessory fairing is mounted on the nose of the J57 engine at the air intake. The accessory fairing houses the bleed valve governor and the N_1 accessory case. Note in figure 3-22 that the fairing is secured in position by 12 studs in the accessory case. The hold-down nuts for the studs can be tightened only by reaching through the hole in the end of the fairing. The nose cap is then installed with four bolts to complete the fairing installation.

ENGINE INSTALLATION IN THE F-102A.

Installing the J57 engine in the F-102A airplane can be a compartively easy operation—providing you have and use the necessary ground support equipment. Following the procedures and instructions in the maintenance technical order (T.O. 1F-102A-2-4) will also make the operation as simple as possible. To acquaint you with the general procedure and requirements of the engine installation operation, let's go through a sample operation.

PREPARATION FOR ENGINE INSTALLATION.

The airplane must first be readied for the engine by stabilizing th airplane in a relatively level position. This leveling process can be accomplished quite easily by use of standard aircraft jacks. Then remove the tail cone (if it is not already removed) and either open or remove all of the fuselage access doors in the engine area. The tail cone removal is facilitated by the use of the Lockheed truck 205226 and the adapter stand SE-0731, which are shown in the lower right portion of figure 3-23.

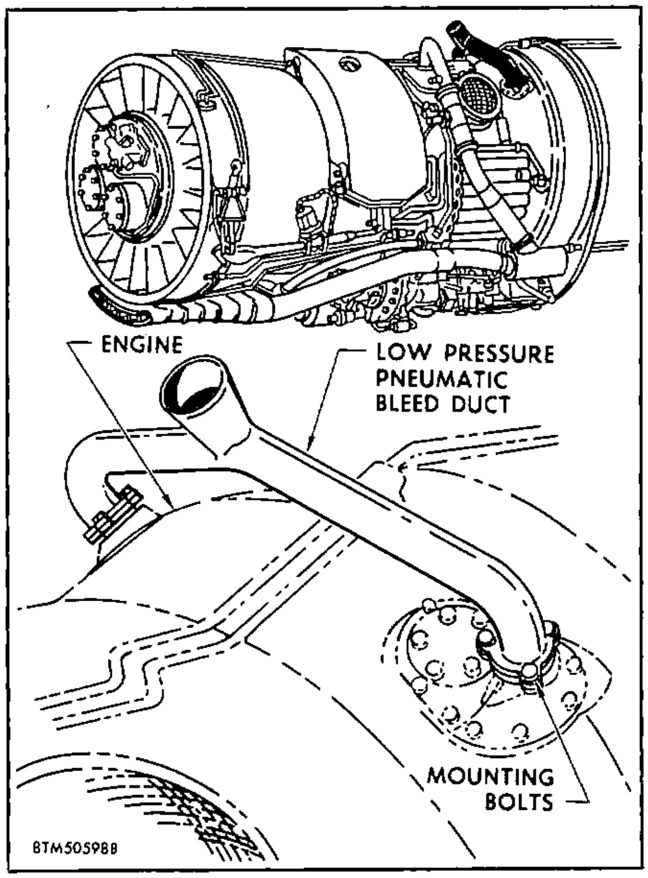


Figure 3-19. Low-Pressure Pneumatic Bleed Duct Installation

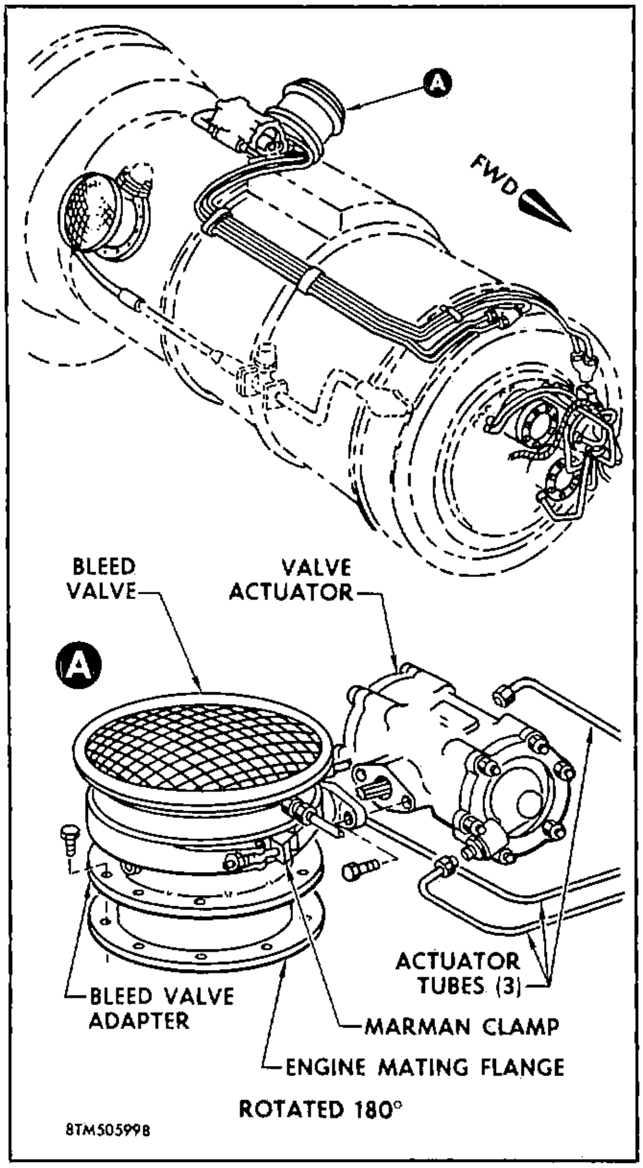


Figure 3-20. Compressor Bleed Valve Adapter Installation

Special installation rails and brackets must be installed inside the airplane to assist in rolling the engine into the airframe. These extension rails (SE-0857 -801 and -802) are furnished as special equipment with the F-102A. The extension rails and their brackets are shown in their installed position in the upper view of figure 3-23. Detail C in figure 3-23 shows the typical rail support bracket installation—note that the support bracket fits inside the slot in the side of the installation rail. You will have to install three of these rail support brackets on each side of the fuselage. The forward two support brackets, one on each side of the fuselage, stay in the aircraft permanently. These eight brackets support the total weight of the rails and the engine while the engine is rolled into the airframe.

After you have the engine installation rails firmly secured to the fuselage structure, you are ready to move the engine up to the airplane. It is a good practice to position the engine mount stand SE-0635 just aft of the fuselage so you can visually determine when the engine has been raised to about the right height. Then move the engine mount stand up close to the airplane. Using the jacks on the engine mount stand, raise the mount stand high enough that the rails on the stand align with the rails that you have installed in the airframe. When you have the two sets of rails aligned, insert the special rail splice bolts (provided with the stand) through both sets of rails so that the rails are locked together. Then attach the engine mount stand draw bar to the attachment bracket on the lower flange of the engine afterburner section. Keep in mind that this is just a general description of the operation. Detailed operating instructions for the engine mount are included on an instruction plate attached to the stand.

In detail C of figure 3-24 note the mount pad, the alignment ring, and the fuselage portion of the thrust mount. Place the alignment ring over the fuselage portion of the thrust mount. Then proceed with the next step—attaching the forward mount turnbuckles to the fuselage. It is a good practice to center the turnbuckle barrels between the turnbuckle eyes before you attach them to the fuselage. This will give you greater adjusting range with the turnbuckles after the engine is installed. At the same time you are attaching them to the fuselage, lubricate the turnbuckle threads wth grease—MIL-G-7187 or equivalent.

INSTALLING THE ENGINE.

Now you are ready to roll the engine into the airframe. Make sure that all lines and equipment will clear the airplane, then roll the engine forward on the engine installation rails until the right front engine thrust mount contacts the fuselage fitting. This is the fitting on which you have already placed the alignment ring. The assembled view of the forward mount fitting is shown in detail A of figure 3-23. In figure 3-24, note the two engine support link assemblies at the aft end of the engine. These assemblies indicated by the number 7 are more commonly called "dog bones." By adjusting the "dog bones" and the forward mount turnbuckle assemblies, you can lift the engine up from the engine installation rails. Raise the engine just far enough so that the engine support rollers turn freely. It is good practice to keep the number of turns equal for both of the forward mount turnbuckles.

Now, remove the engine installation rails, the rail support brackets, and the forward and aft engine rollers. As mentioned earlier, the two forward rail support brackets remain installed on the airframe. With the rails removed, adjust the forward right support

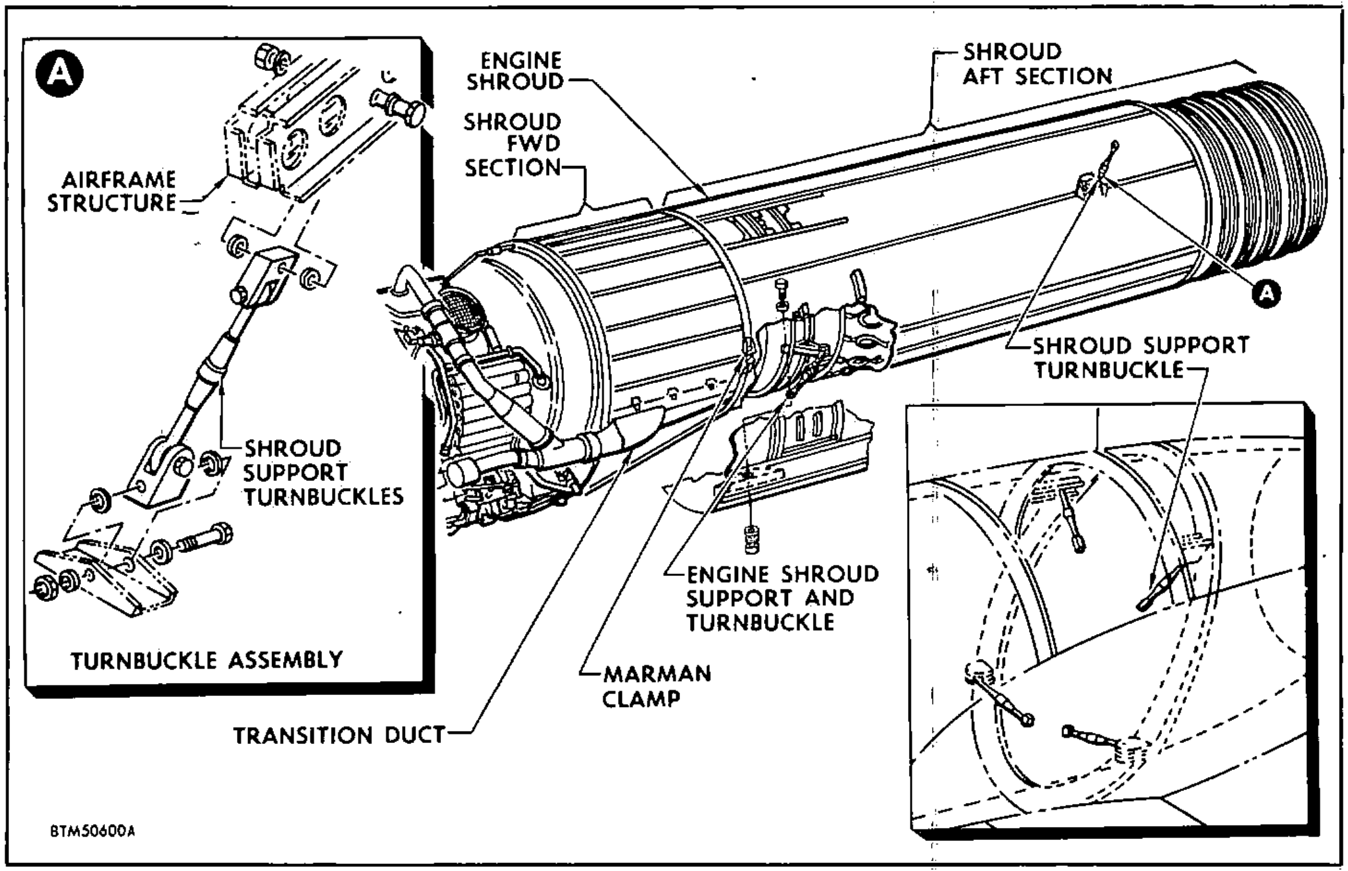


Figure 3-21. Engine Shroud Installation

linkage until the alignment ring can be positioned over the engine thrust mount. Keep track of the number of turns that the turnbuckle is adjusted. Then, install the clamp assembly over the alignment ring and the mounting connection, and secure the clamp. Adjust the forward left mount turnbuckle the same number of turns that you did on the right mount turnbuckle. This will suspend the engine in a level position.

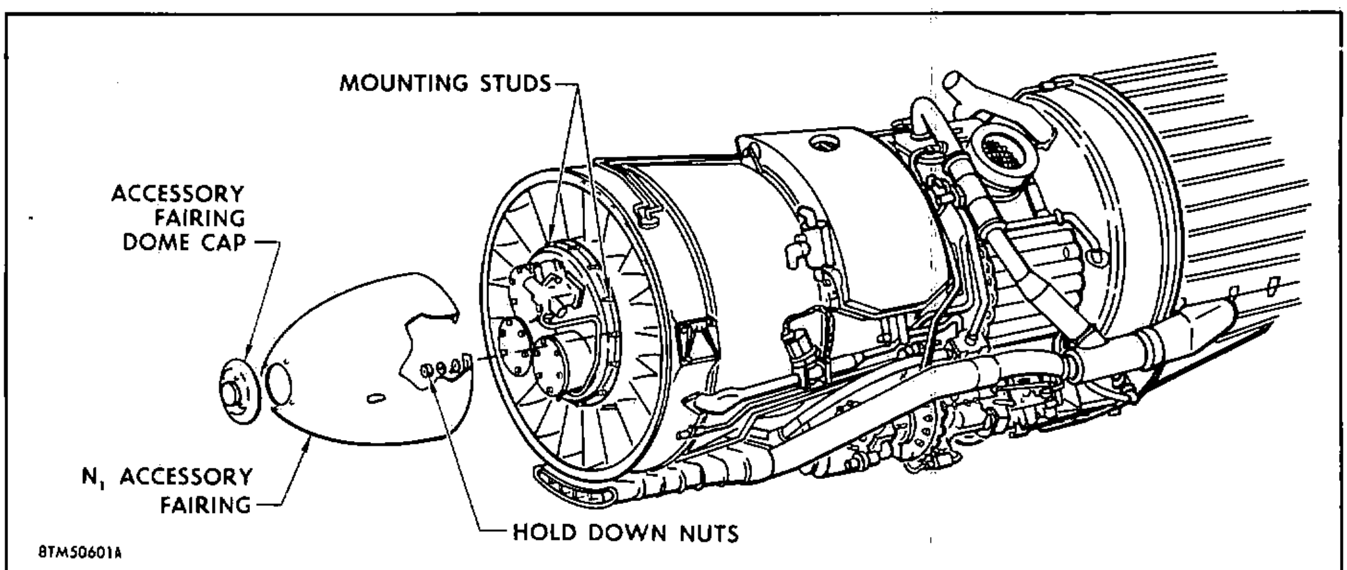


Figure 3-22. N_1 Accessory Fairing Installation

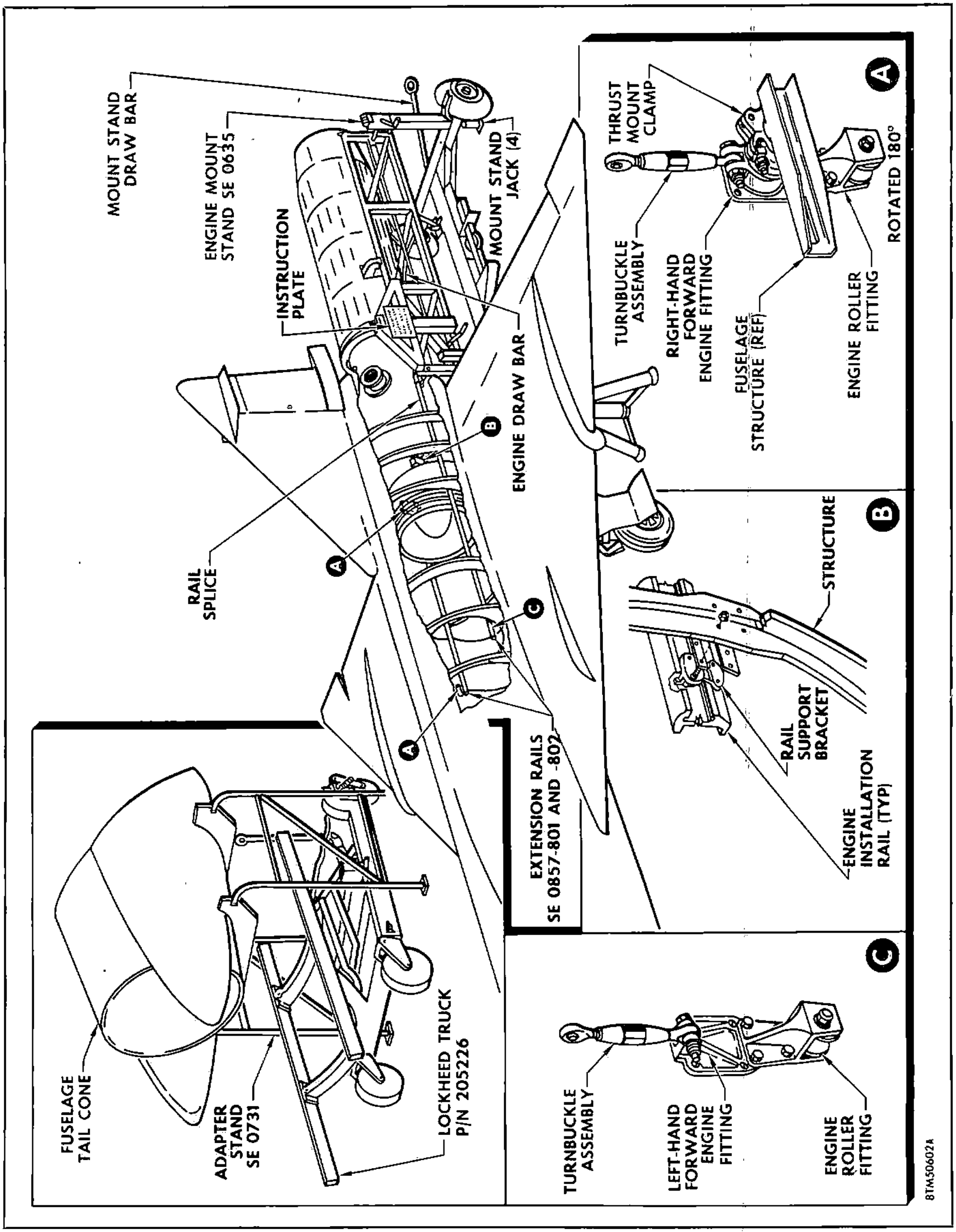


Figure 3-23. Engine Removal and Installation Equipment

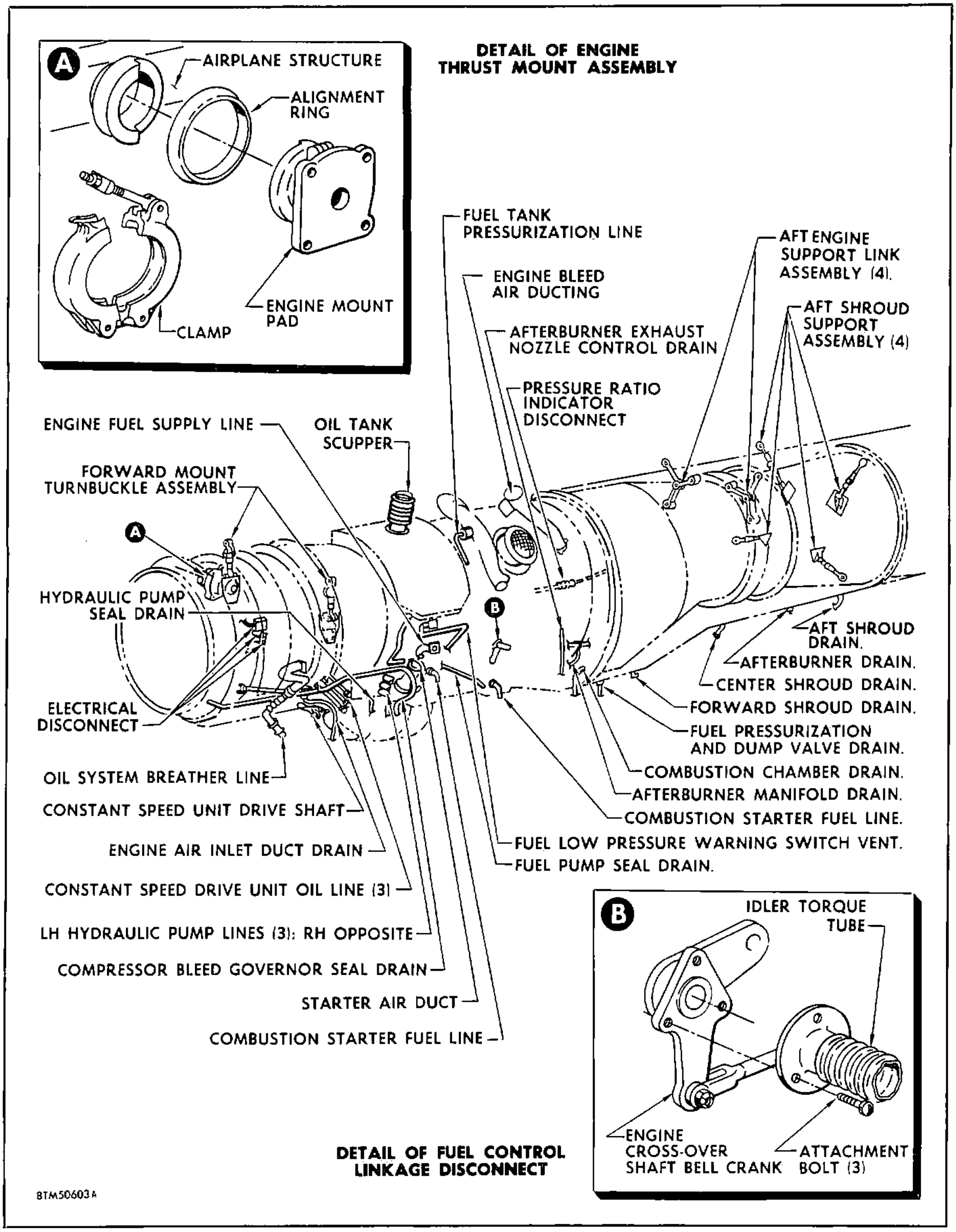


Figure 3-24. Engine Disconnect Points

To position the aft portion of the engine correctly, you must first install the tail cone assembly. All of the tail cone installation bolts need not be used since the cone installation is only temporary at this time. Having the tail cone assembly on the airplane will allow you to adjust the aft engine mounts so that the engine exhaust nozzle will center in the tail cone opening. Then remove the tail cone.

Earlier in this chapter it was mentioned that support blocks were necessary to support the engine shroud on the engine when the engine was not installed in the airplane. Your next step involves attaching the four shroud supports (8 in illustration) to the fuselage and then adjusting them until the shroud is centered around the engine. The support blocks can then be removed. Then safety wire the shroud supports, the forward mount support turnbuckles, and the aft engine mount adjustment bolts.

The J57 engine installation operation, from this point on, consists of making the connections at all of the engine disconnect points and installing the access doors. All connection points are accessible through the engine bay access doors. Once the engine has been "hung" in the airframe, there are about 28 separate steps to be accomplished to complete all engine connections. After completing all connections, the tail cone can be permanently installed. As the last step, install or close all of the access doors.

You will find that a wise and safe practice to follow during engine installation is the checking off of each item as you accomplish it. This practice will preclude any inadvertent ommissions. The preceding description of the engine installation process should give a general, overall view of the engine installation operation. For detailed information, refer to the power plant maintenance technical order, T.O. 1F-102A-2-4.

ENGINE REMOVAL.

Removing the J57 engine from the F-102A is just about the reverse of the engine installation operation discussed thus far. The preparation procedure, however, is exactly the same. As you will recall, this preparation procedure involved: stabilizing the airplane, removing the tail cone, installing the engine installation rails, and positioning the SE-0635 engine mount stand so that the rails could be spliced together. The rest of the engine removal operation is just the reverse of the installation procedure.

In some cases, the engine will be out of service for a very short time, and, in this event, it will not need to be preserved. Where the engine is going to be out of service for an extended period, however, it must be preserved to prevent corrosion within the engine. The engine preservation operation has already been outlined in this chapter.

SUMMARY.

In this chapter you have learned about the handling and maintenance requirements of the J57 engine when it is not installed in the F-102A. General practices and procedures for preserving and depreserving the engine were given to acquaint you with the need for engine preservation as well as with the operation itself. This chapter also outlined the engine build-up operation that is necessary to prepare a new or overhauled engine for installation in the F-102A. The last portion of the chapter was devoted to the engine installation and removal operations as they apply to the F-102A. In many cases, your maintenance duties will not include performing any of the engine maintenance, except for that required for installed engines on the flight line. Regardless of this, however, the knowledge of all engine requirements will prepare you to become a better J57 engine maintenance man.

Chapter IV

POWER PLANT INSTRUMENTS AND WARNING SYSTEMS

Contents *Page*

The engine instrument group, as the term implies, includes those instruments that make the pilot aware of the power plant operating conditions. Present-day jet engines, such as the J57, require some instruments that are not required by reciprocating engines. On the other hand, some of the instruments needed on reciprocating engines are not needed by jet engines. You will find that the engine in the F-102A (the J57) has only five instruments, but each of these instruments is vitally important to the successful operation of the engine.

The F-102A engine instruments and their positions on the instrument panel are shown in figure 4-1. These instruments include the pressure ratio indicator, tachometer, exhaust temperature indicator, exhaust nozzle position indicator, and the fuel flowmeter. The alternate pressure ratio indicator which you will find on some F-102A airplanes is also shown at the bottom of figure 4-1. All of these instruments and their associate systems will be discussed in this chapter.

THE TACHOMETER.

One of the instruments which tells the pilot how the engine is performing is the tachometer. Its function is to determine the revolutions per minute at which the engine N_2 compressor is turning and to provide an indication of that speed on the instrument dial.

Several types of tachometers are used in present-day aircraft. In a single-engine airplane, which has the engine mounted directly ahead of the cockpit, a mechanical tachometer is frequently used. This kind of tachometer is driven directly from the engine by a flexible shaft and operates much like an automobile speedometer. However, most modern aircraft use electrical tachometers, which eliminate the need of the direct drive shaft from the engine. The F-102A tachometer consists of an engine-mounted generator which is electrically connected to an indicator in the cockpit, as shown in figure 4-2. The tachometer generator is mounted on the J57 engine accessory drive case and is splined to the accessory drive gearing. When the engine is running, the generator also rotates. Note that the generator is secured to the engine accessory drive case by four mounting studs. As the generator turns, it produces an electrical signal which is transmitted to the indicator. The indicator, in turn, converts this electrical signal into the correct amount of N_2 compressor rpm and then shows this rpm on the instrument dial.

THE TACHOMETER GENERATOR.

The tachometer generator used on the F-102A is attached to the front of the accessory drive housing on the right side of the engine. As you learned in Chapter III, it is pad mounted; that is, the generator is

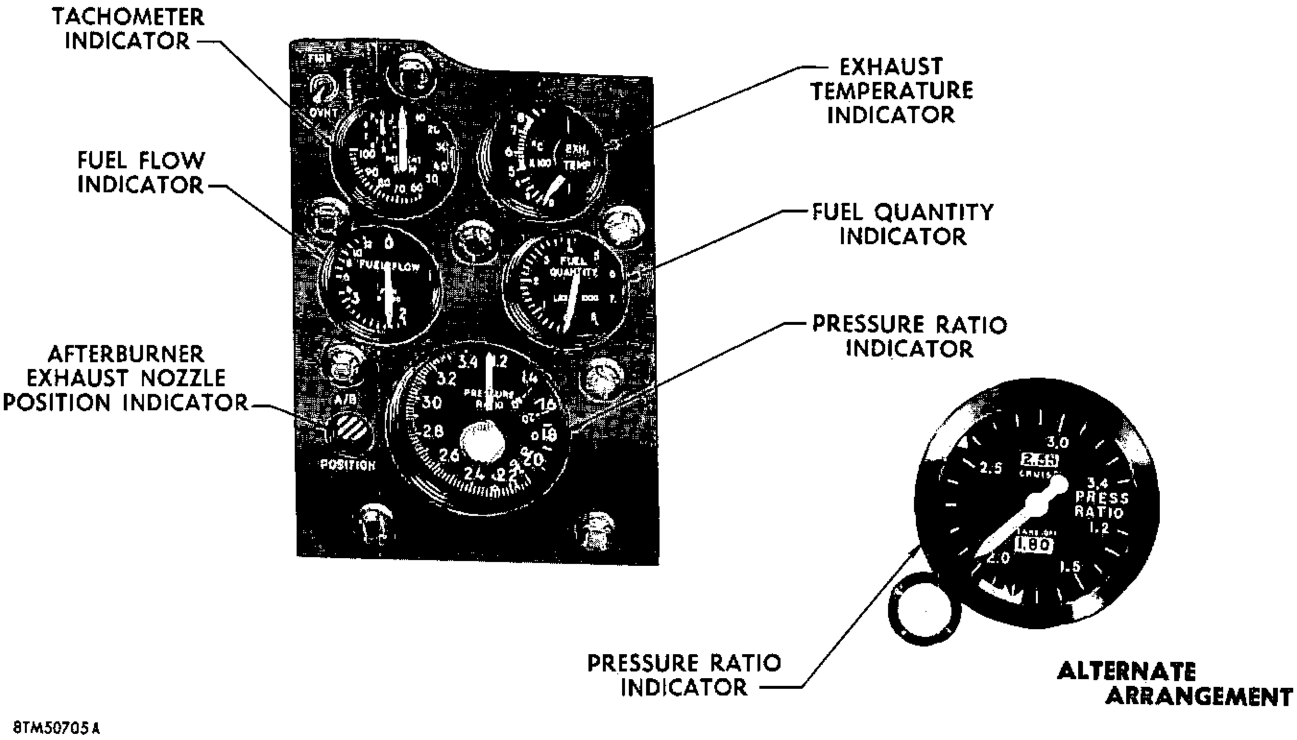


Figure 4-1. Engine Instruments

bolted to a flat, machined surface on the engine accessory housing. The square-end shaft shown on the end of the generator in figure 4-4 projects from the generator into the engine accessory drive section where it is turned by the accessory drive gears. When the engine is operating at maximum rated rpm, the tachometer generator turns at 4200 rpm.

The tachometer generator shaft runs through the center of the rotor (armature) shaft. A pin connects the two shafts at the end opposite the mounting pad. This type of construction permits the generator drive shaft to absorb any vibrations that might result from slight wear and misalignment at the shaft connecting points. Thus, the rotor shaft—and consequently the rotor—will spin at the same rpm as the generator drive shaft. A bearing at each end of the rotor shaft supports the shaft within the generator.

The rotor of the generator is a two-pole permanent magnet, made of very hard steel. Surrounding the rotor is the stator, made of a laminated, soft iron core around a three-phase winding. When the rotor is turned by the drive shaft, current is induced in the windings of the stator. Since the rotor has two poles (one north and one south), one complete revolution of the rotor will result in one cycle of alternating current in each of the stator windings. Three wires come from the stator and carry this induced current to the generator receptacle, and from there to the tachometer indicator in the cockpit.

THE TACHOMETER INDICATOR.

There is considerable similarity between the tachometer generator and the basic mechanism of the tachometer indicator. This should be understandable since the indicator incorporates a synchronous motor which has many of the same characteristics of the tachometer generator. The N_2 compressor rpm in the F-102A is indicated by the tachometer indicator in the percent of the engine's maximum rated speed. The dial of the indicator is calibrated from 0% to 100% in 2% increments, beginning at the top of the dial and extending around for 270°. The large pointer on the indicator rotates around the large dial face. In figure 4-2, you will note that the indicator also has a small pointer and dial in the upper left portion of the large dial. This small pointer rotates around its dial once for every 10% change in engine rpm. Using both the large and the small scales, the indicator can actually show up to 110% engine rpm.

As you might already know, the J57 engine rotates at approximately 10,000 rpm when it is developing maximum thrust. Some J57 engines, however, will develop

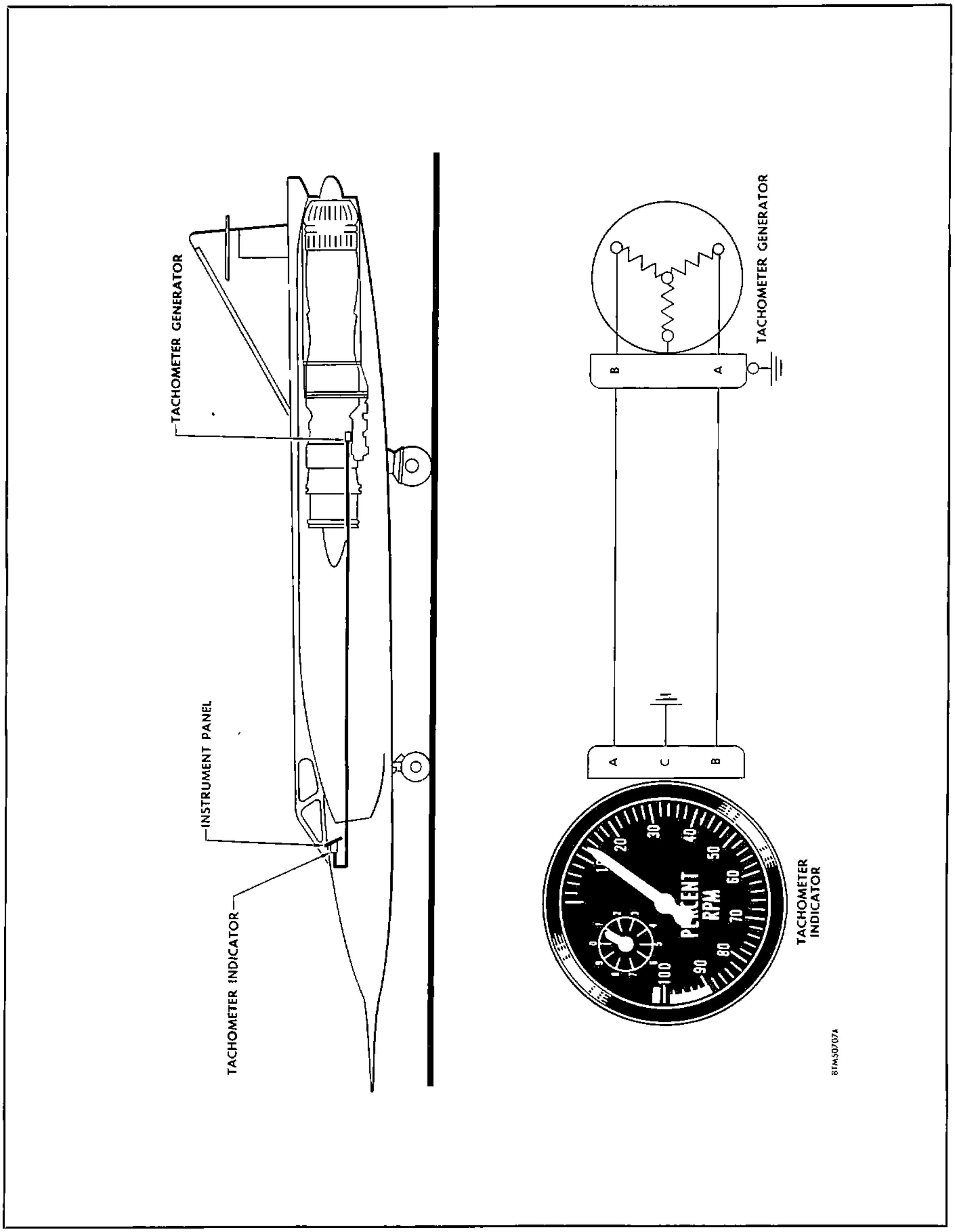


Figure 4-2. The Tachometer Indicating System

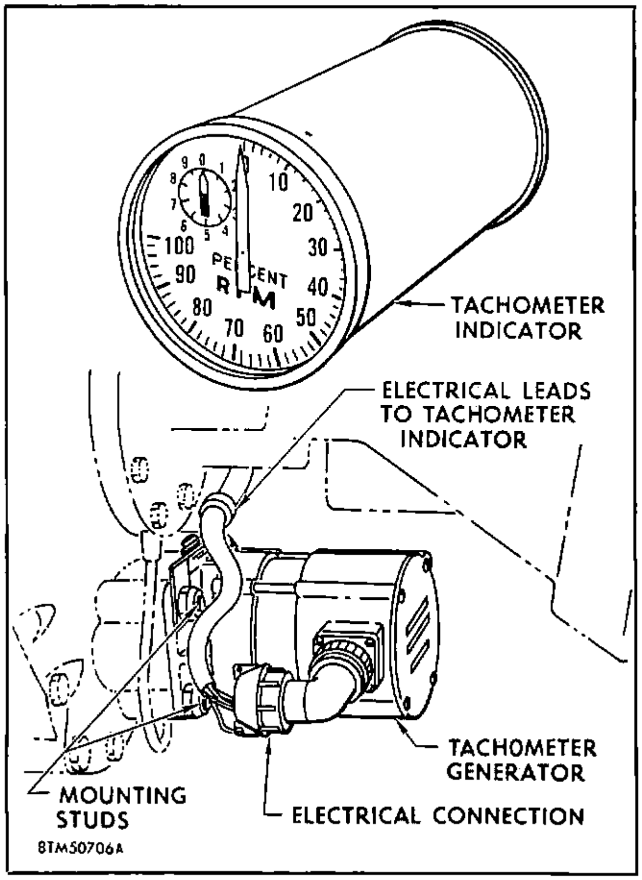


Figure 4-3. The Tachometer and Tachometer Generator

their maximum thrust at an rpm percentage figure that is less than 100%. This variation in rpm percentages in different engines is a natural outgrowth of modern jet engine design and cannot be eliminated. The rpm at which a jet engine develops its maximum thrust is always stamped on the engine data plate. If you have a J57 engine, for example, that has 9600 rpm stamped on its data plate, that engine will develop its maximum thrust when the tachometer indicator is registering only 96%.

You should realize then, from the above paragraph, that a slight variation in engine rpm is not sufficient cause to trouble shoot the tachometer indicating system. The important thing to remember is that the tachometer indicator should register about the same rpm that is stamped on the engine data plate when the engine is developing its maximum thrust.

As just mentioned, the tachometer indicator incorporates a synchronous motor which is quite similar to the one in the tachometer generator. The electrical power that operates the indicator motor comes from the tachometer generator. Since the indicator stator is wound in the same manner as the indicator stator in the generator, a rotating magnetic field is set up in the stator as it receives the induced current from the generator. The rotor in the indicator motor is a permanent magnet. As you probably know, a permanent magnet will line up in the magnetic field around an electrical conductor. Because of this fact, the indicator rotor is dragged around by the rotating magnetic field set up by the generator current. Since the tachometer is geared directly to the engine accessory drive section, the rpm of the indicator rotor is an accurate indication of the rpm of the N_2 compressor.

The exploded view of the tachometer indicator (figure 4-5) will give you a good idea of how the indicator rotor action is turned into a dial indication of engine rpm. Note that the shaft which turns the indicator rotor also turns another permanent magnet called the drag magnet. Around the drag magnet is the drum, which—because of proximity to the drag magnet—is dragged around with the drag magnet. The hairspring, however, lets the drum rotate just so far. The amount that the drum can rotate depends upon the amount of current from the tachometer generator, and the strength of the holding hairspring inside the indicator. Since the hairspring strength is always constant, the drum movement will be directly proportional to the current from the tachometer generator and the engine rpm. From the preceding description you should remember that the rpm of the engine is transmitted to the tachometer generator by gears, then to the tachometer indicator electrically, and finally to the indicator pointer by the magnetic force of the drag magnet.

MAINTENANCE OF THE TACHOMETER INDICATOR SYSTEM.

One of the most frequent troubles with tachometer systems—and especially electrical tachometers—is that

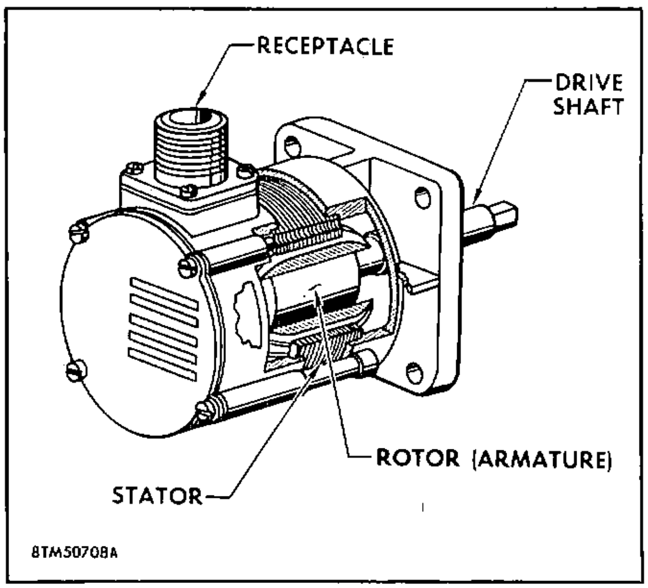


Figure 4-4. Cutaway View of the Tachometer Generator

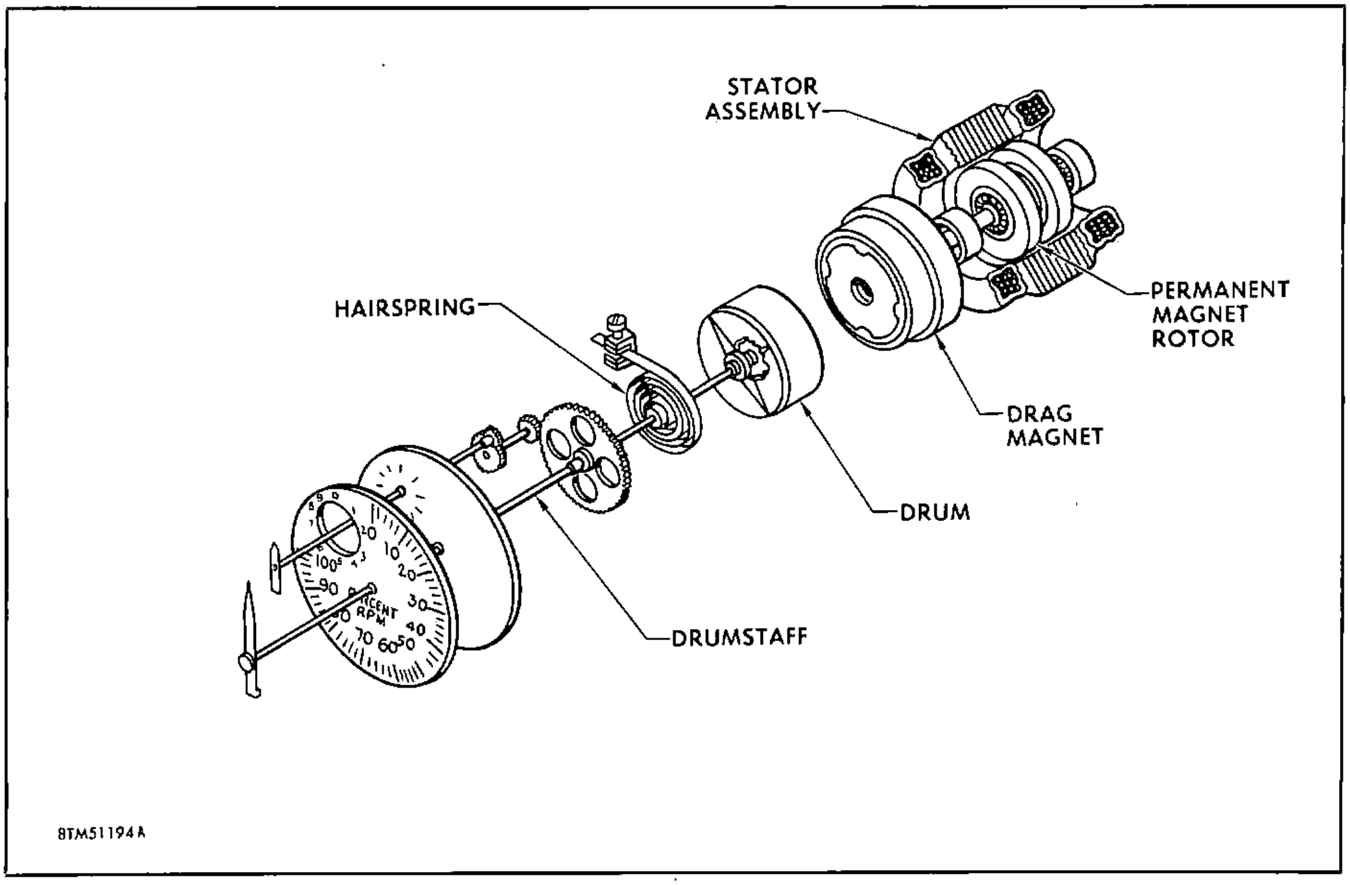


Figure 4-5. Exploded View of the Tachometer Indicator

the indicator pointers have a tendency to stick or "hang up" during engine starting. This type of trouble, however, is not a malfunction of any part of the system, and it can be remedied by lightly tapping the instrument panel in the immediate vicinity of the tachometer indicator. If the indicator needle still sticks after the engine is started and you have advanced the power lever into the IDLE range, then something is definitely wrong and you will have to shut down the engine and trouble shoot the tachometer indicating system.

An inoperative tachometer system can be traced to the indicator, generator, or the connecting electrical harness. Since you are not allowed to perform any internal maintenance or repairs on either the generator or the indicator, the easiest way to trouble shoot the system is to replace the components. An accepted method for line maintenance personnel is to first replace the indicator with an instrument that is known to be functioning satisfactorily. This will tell you whether the trouble is in the indicator or in the generator and wiring. If the condition persists after the indicator has been replaced, replace the tachometer generator. If the condition still has not been cured, then you will have to either replace the electrical harness or run a continuity check to determine whether there are any breaks or shorts.

THE PRESSURE RATIO INDICATOR SYSTEM.

On reciprocating engines, the pilot needs two basic indications to determine how much power the engine is developing. These two indications are engine rpm and engine manifold pressure. On jet engines the pilot must also have two basic indications to know how much power the jet engine is developing. One of these indications, the N_2 compressor rpm, was discussed in the preceding section. The other indication is the ratio of the engine exhaust pressure to the intake air pressure.

This pressure ratio is very important on all jet engines; especially on dual spool (two-compressor) type turbojet engines, such as the J57. In the dual-spool type of engine, it is possible for one of the compressors to operate at near-stall conditions, without the tachometer showing any appreciable change in rpm and without the pilot being aware of it. To really know how the engine is performing, the pilot needs an accurate measurement of the thrust that the engine is producing. The pressure ratio indicator provides this information.

Two different types of pressure ratio systems are used in the F-102A airplane. Early airplanes use the *direct*

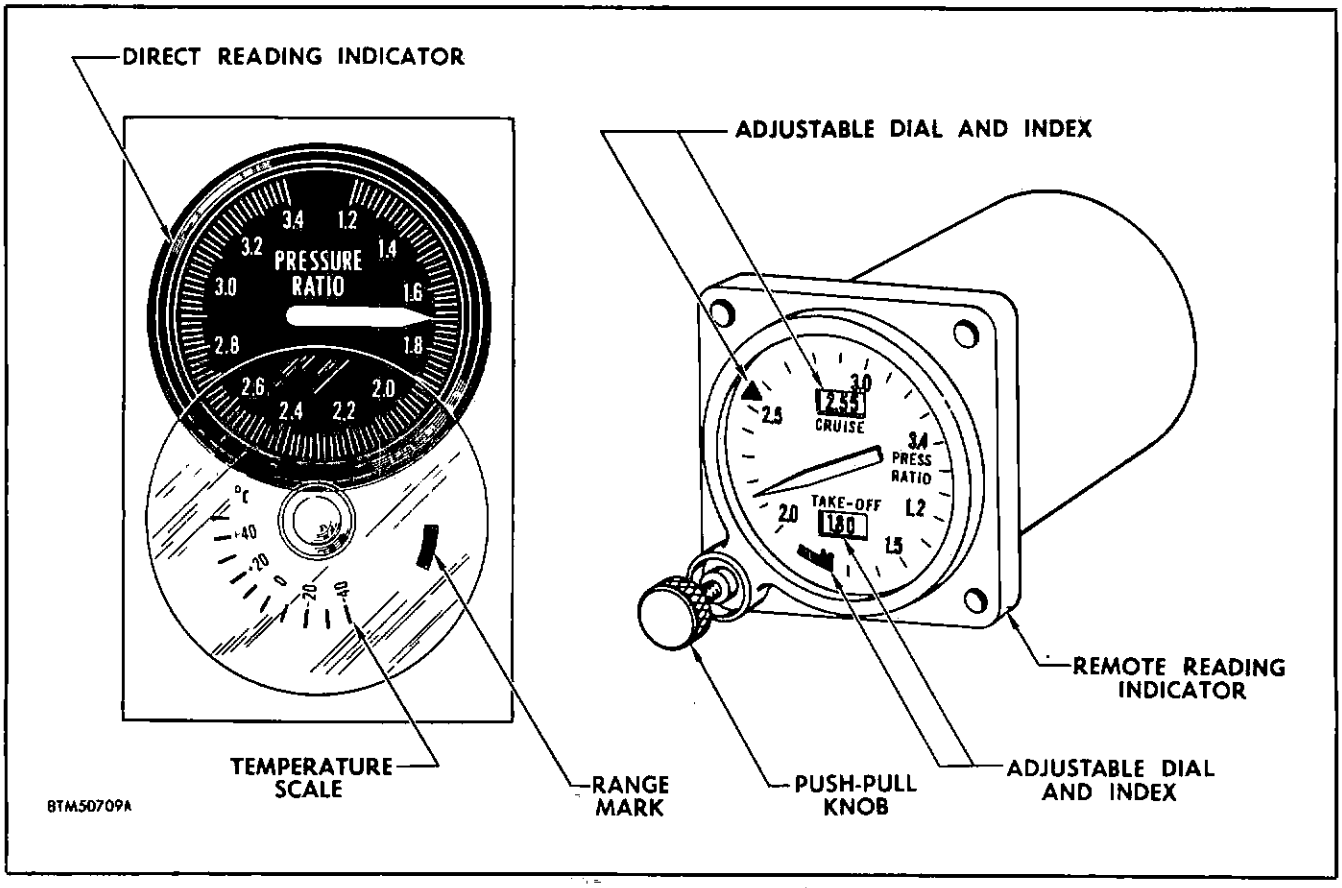


Figure 4-6. The Direct and Remote Pressure Ratio Indicators

reading system, while the later models use the *remote reading system.* In the direct reading system the pressures which are to be compared (intake and exhaust), are piped from the sensing areas and introduced directly into the indicator instrument case. In the remote reading systems, however, a remote transmitter is used to measure the two pressures and then send electrical signals to the indicator in the cockpit. Both the direct and the remote reading indicators are shown in figure 4-6. Each type of indicator is discussed individually in the following paragraphs.

THE DIRECT READING INDICATOR.

The direct reading indicator has a fixed dial that is calibrated from 1.2 to 3.4 to show the ratio of the tailpipe exhaust pressure to the engine intake pressure. The indicator also has a transparent dial which, in figure 4-7, has been detached so that you might see its dial markings better. This transparent dial can be rotated by manually turning the knob in the center of the dial.

To use the pressure ratio indicator correctly, the pilot must first know the temperature of the outside air. This information is provided by the outside air temperature indicator. By turning the center knob on the pressure ratio indicator, the pilot can set the transparent dial so that the outside temperature will coincide with the temperature index mark on the fixed dial.

If the outside temperature were 0°C, for example, the pilot would turn the transparent dial until the 0° mark was lined up with the temperature index mark. The range mark on the transparent dial would then indicate the most efficient power lever settings. The pilot would move the power lever until the pressure ratio indicator pointer was in the dial area covered by the range mark.

How the Direct Reading System Indicates Pressure Ratio.

Figure 4-8 shows an operational schematic view of the direct reading type of pressure ratio indicator. The diaphragm within the frame assembly is the tailpipe pressure diaphragm. When exhaust pressures enter this diaphragm, the diaphragm expands and raises the entire frame assembly. The amount of expansion depends upon the difference between the exhaust pressure inside the diaphragm and the pressure on the outer diaphragm surface.

The frame assembly can also rotate, or pivot, on its two axis points. This frame assembly movement is

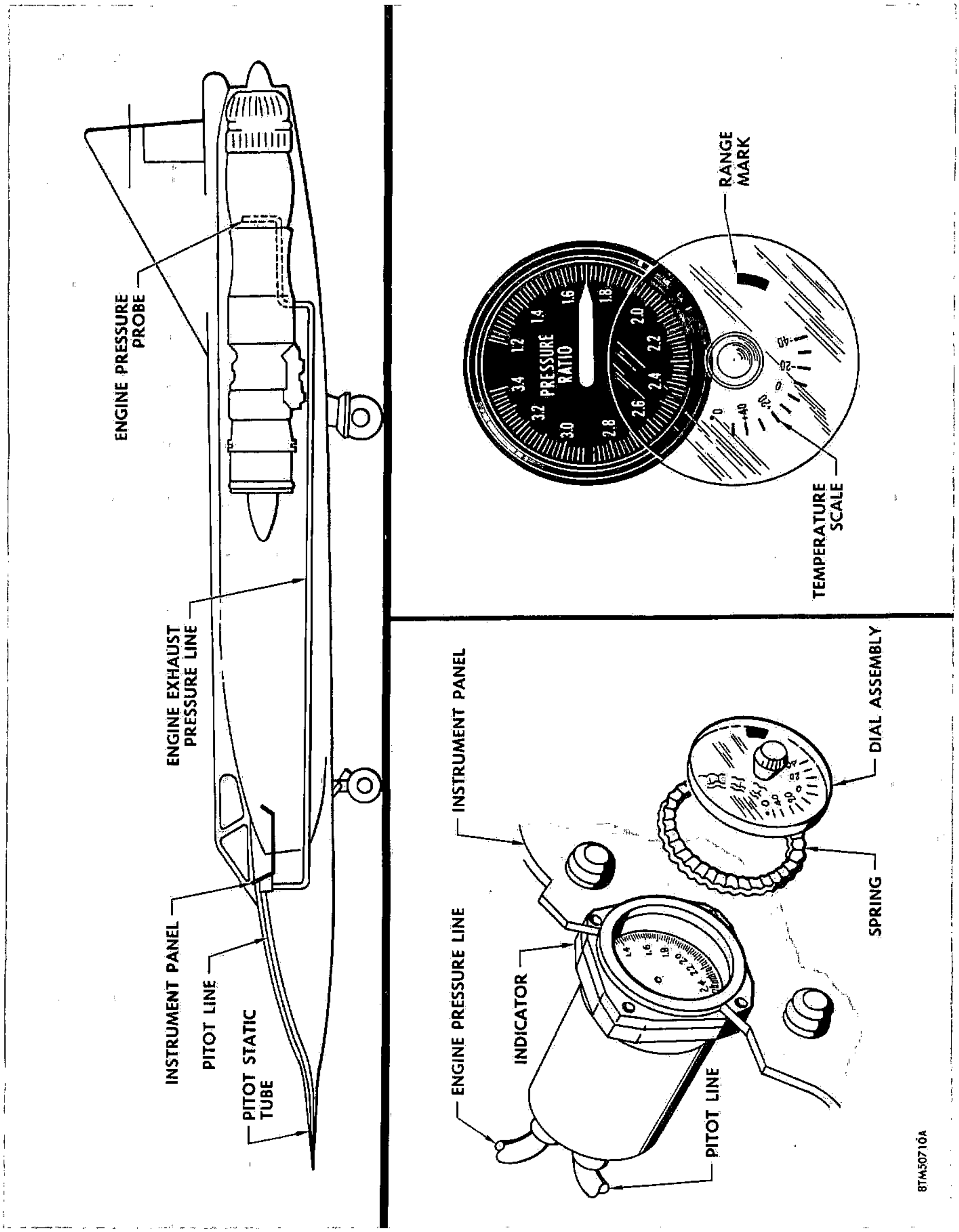


Figure 4-7. The Direct Reading Pressure Ratio Indicator System

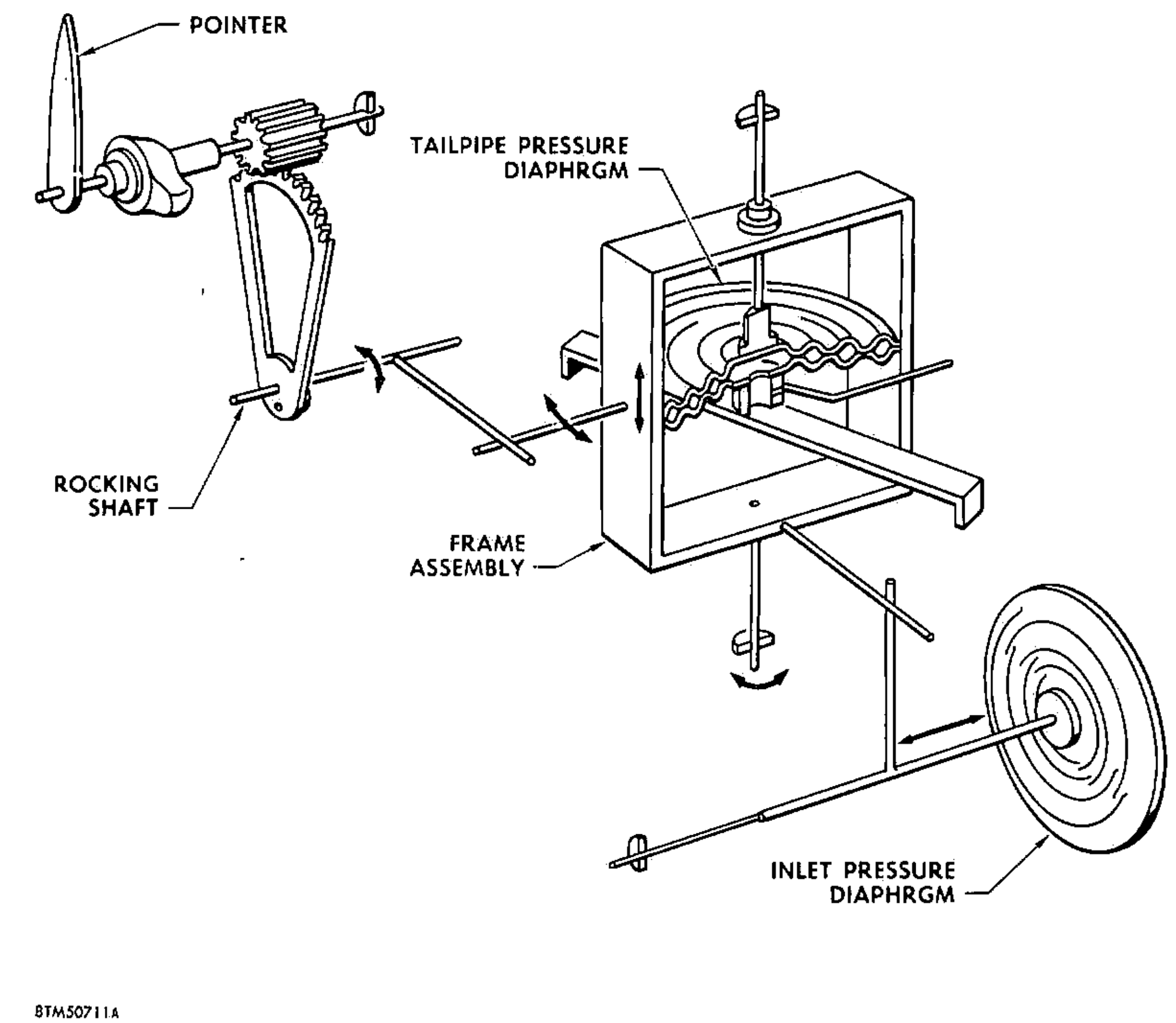


Figure 4-8. Direct Reading Pressure Ratio Indicator Operational Schematic

controlled by the inlet pressure diaphragm. This diaphragm is a sealed, evacuated unit (aneroid type). Since the internal pressure of this diaphragm is always constant, the diaphragm will expand and contract according to pressure changes on its outer surface. The pressure on the outer diaphragm surface is the engine inlet pressure. As the engine inlet pressure increases, the diaphragm contracts and allows the frame assembly to turn away from the rocking shaft. When the inlet pressure decreases, however, the diaphragm expands and pushes the frame assembly around towards the rocking shaft.

Note that the rocking shaft is geared to the indicator pointer. The rocking shaft is turned by the up-and-down movement of the frame assembly. As just noted, the up-and-down movement of the frame assembly is controlled by the tailpipe pressure diaphragm. However, the amount of pointer movement that can result from any specific tailpipe pressure depends upon how close the frame assembly is to the rocking shaft; and, as we have also noted, this depends upon the inlet pressure diaphragm. Note that the closer the frame assembly moves to the rocking shaft, the more the rocking shaft will be turned by a given amount of vertical displacement of the frame assembly.

From the operational schematic, (figure 4-8), you can see that the indicator pointer movement is controlled by the tailpipe pressure and biased by the engine inlet pressure. By using constant mechanical advantages at each of the mechanical links in the indicator, the direct reading indicator can show the ratio, of the engine exhaust (tailpipe) pressure to the engine inlet pressure.

There is one disadvantage in using a direct reading type of pressure ratio indicator; exhaust gases are piped directly into the cockpit of the airplane. A dangerous condition could result if the exhaust gas line were to leak in the cockpit area. To eliminate this possibility, a remote reading type of indicating system is installed in the later models of the F-102A.

THE REMOTE READING PRESSURE RATIO INDICATOR SYSTEM.

If you understand the operation of the direct reading type of indicator, you should have no trouble understanding the remote reading type of indicator. It furnishes the same information to the pilot, and in a similar manner. The measuring mechanism, however, is mounted in the fuselage area between the two engine intake ducts. This measuring mechanism is called a transmitter, and it connects electrically to the remote reading indicator in the cockpit. As you will remember from figure 4-1 in this chapter, the remote reading pressure ratio indicator uses the same mounting hole on the instrument panel that the direct reading indicator does.

The remove transmitter has two diaphragms just as the direct reading indicator does. Any pressure differences in either the engine inlet pressure or the exhaust pressure are sensed by these two diaphragms. But, instead of transmitting the movement of the diaphragm to the rocking arm and then to a pointer, the remote transmitter has a small synchronous motor which the sensing diaphragms control.

According to the amount of diaphragm movement, the remote transmitter will send an electrical signal to the remote reading indicator in the cockpit. The indicator also has a synchro which is controlled by the signal from the transmitter, and it moves the indicator pointer an amount that is proportional to the amount of diaphragm movement in the transmitter.

The Remote Pressure Ratio Transmitter.

Figure 4-9 shows a side and end view of the remote pressure ratio transmitter assembly. Note on the end view that there are two tube fittings—one marked LOW and the other marked HIGH. The low-pressure connection receives the inlet air pressure while the high-pressure connection receives the exhaust pressure. In the center, you can see the electrical receptacle that provides power to the synchro transmitter inside the transmitter assembly, and carries the output signal from the transmitter to the synchro receiver in the indicator. The pressure and electrical connections are all on the base of the unit, rather than on the transmitter itself, because the base is attached firmly to aircraft structure and the transmitter is shock-mounted to the base. The curved tubes from the base to the transmitter carry the pressures to the transmitter case without interfering with the shock absorbing mountings.

HOW THE REMOTE TRANSMITTER PRODUCES A SIGNAL. As previously mentioned, the remote transmitter has two diaphragms inside an airtight case. One diaphragm is an evacuated bellows acted upon by inlet air pressure alone; the other diaphragm has exhaust pressure inside and inlet air pressure outside. An increase or decrease in either of these pressures will tend to move the diaphragms. The movement of the diaphragms is passed through mechanical linkage and gears and results in the rotation of a rocking arm. This rocking arm is connected to the rotor of a synchro-transmitter inside the case. As the rocking arm is rotated, the rotor of the synchro-transmitter is rotated proportionally.

The rotor is connected to an electrical source of power; and, as it is positioned within the stator windings, it will induce a specific current within each of the three windings. The rotor of the synchro-transmitter and the rotor of the synchro-receiver (in the indicator) are connected in series; and, as a result, both rotors will seek identical positions within their respective stators. This means that the same voltage is being induced into the stator of each of the transmitters. This, in effect, transmits an accurate indication of the ratio of pressures to the calibrated scale on the indicator dial.

The Remote Reading Indicator.

There are several differences between the remote indicator and the direct reading instrument. Note in figure 4-10 that the dial of the remote indicator is calibrated for the same limits as the direct reading indicator, but it is positioned differently. There are two index markers and two windows in the face of the dial. The readings in the small windows, and the position of the index markers, can be changed by the push-pull knob on the front of the indicator.

When the knob is pushed in, it will turn the numbers in the lower window and reposition the lower index marker. When the knob is pulled out, it will turn the numbers in the upper window and reposition the upper index marker. The upper window and the marker indicate the cruise setting desired by the pilot, while the lower window and marker indicate the takeoff reading which should show during takeoff. These dials and markers are mechanically connected to the knob and manually set by it. They are in no way affected by the indicating mechanism inside the indicator that turns the pointer.

Inside the indicator case is a synchro-receiver, a magnetic amplifier, and a servo motor. The position of the synchro in the transmitter unit provides the signal to the indicator through an electrical connection in the rear of the indicator case. The signal positions the synchro-receiver which in turn sends a signal via the amplifier to the servo motor. The servo motor then turns the point shaft to deflect the pointer and indicate the pressure ratio.

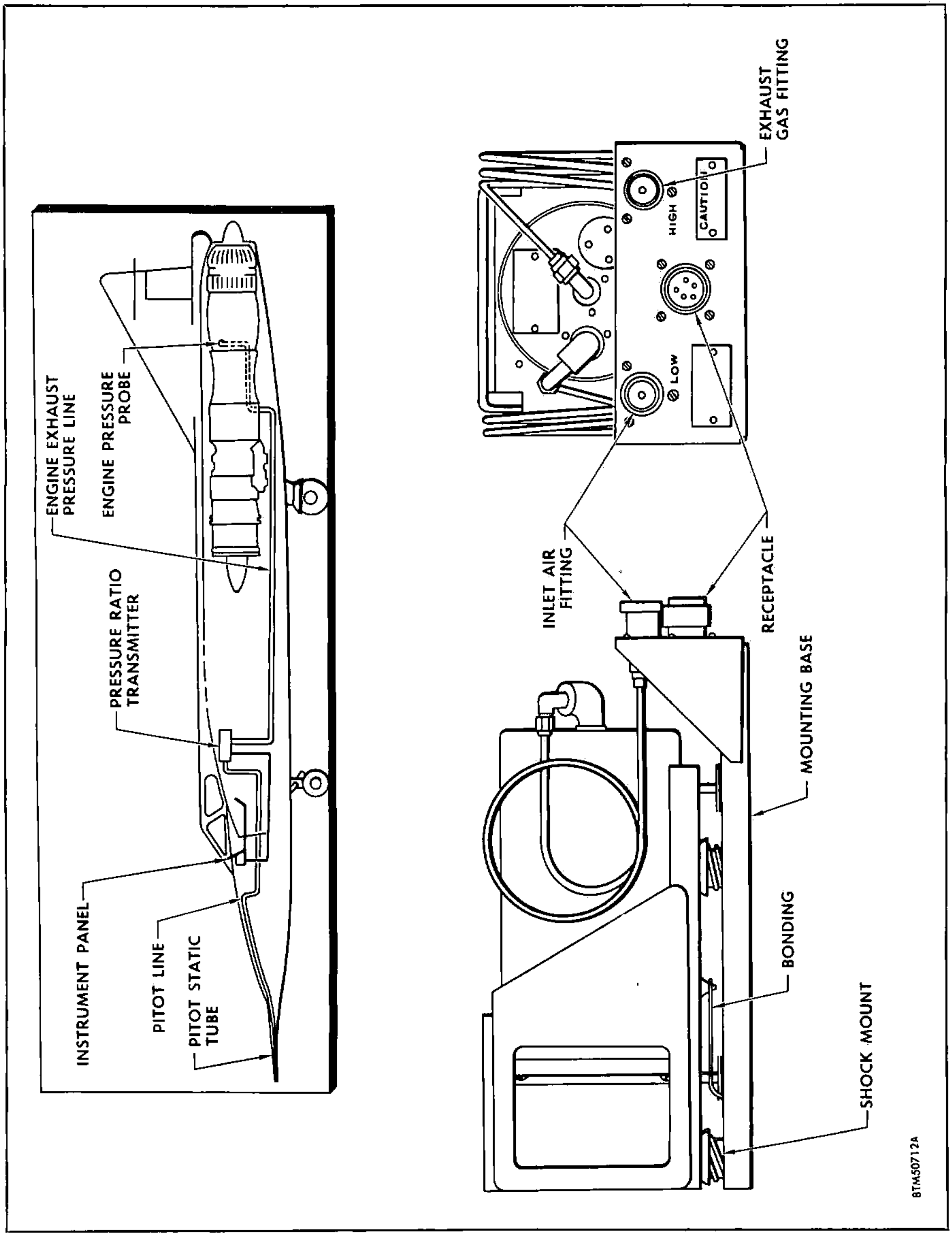


Figure 4-9. Remote Pressure Ratio Transmitter

To set the remote indicator properly, the pilot of the F-102A must do two things. First, he contacts the control tower to get the existing temperature at the field and the estimated temperature at the altitude at which he intends to cruise. He then consults his performance charts to determine what the settings on the indicators should be for the two temperatures. When he turns the push-pull knob, the index marker and the numbers in the windows will show the same reading.

You can see an example of these readings in figure 4-10. Note that the upper index marker is between 2.5 and 2.6 while the indication in its corresponding window is the same, or 2.55. Note also that the lower window shows a reading of 1.80 and that its index marker is at the corresponding position on the dial.

The windows serve only to give the pilot an accurate and simple method of telling what his index setting is. Note that the lower index (for takeoff) has a raised portion on one end. This is the most desirable position for the pointer during takeoff. However, it is safe to proceed with takeoff even if the needle goes beyond that point, or as long as it remains within the zone covered by the range marker. When the aircraft reaches the selected cruise altitude, the pointer should match the upper cruise index marker. The pilot then knows that his engine is performing efficiently.

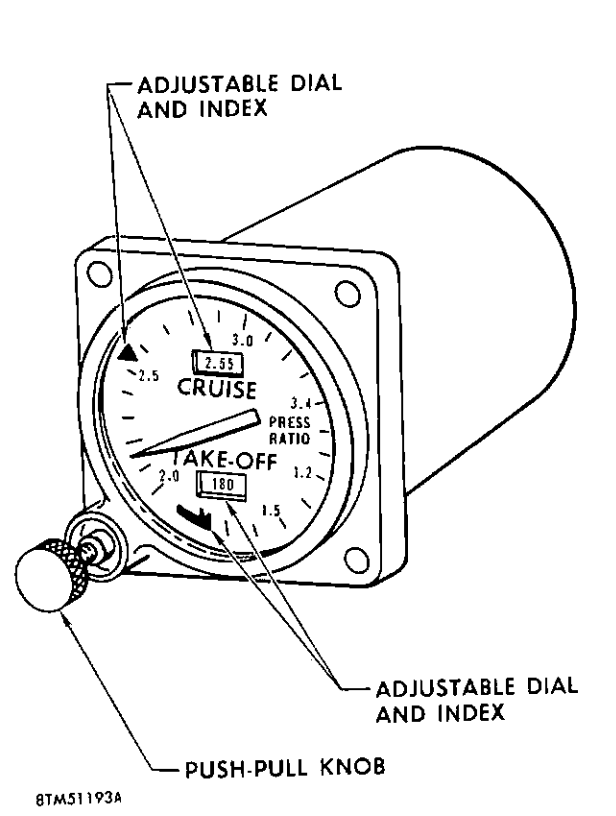


Figure 4-10. Remote Pressure Ratio Indicator

MAINTENANCE OF THE PRESSURE RATIO INDICATOR SYSTEM.

Any troubles in the pressure ratio indicating system can usually be traced to a faulty transmitter, faulty indicator, or clogged sensing lines. When trouble shooting a malfunctioning indicating system, it is best to first replace the indicator with a unit that is known to be serviceable. If the replaced indicator does not cure the trouble, the same thing should be done with the transmitter. When these two steps do not cure the trouble, you should then check the sensing lines to see whether they are clogged. This involves connecting a low-pressure air source to the sensing lines. It is possible to damage some of the other components in the pitot-static system if the exact pressures and procedures are not used during the line check. Always refer to the F-102A Instruments Maintenance Handbook (T.O. 1F-102A-2-9) whenever you troubleshoot the sensing lines. The last thing to check in the pressure ratio indicating system is the electrical wiring between the transmitter and the indicator. This can be done easiest by disconnecting the wiring from the transmitter and indicator, and then using a continuity light to check each wire for an open circuit or short to ground.

AFTERBURNER EXHAUST NOZZLE POSITION INDICATOR SYSTEM.

All jet engines that have afterburners must have some means of controlling the opening at the engine exhaust nozzle. This control of the afterburner nozzle is necessary, because pressures and temperatures produced by the engine vary between normal burning and afterburning. As previously mentioned, the nozzle is closed during normal burning—that is to say, the opening is restricted. In this position, the exhaust opening is the best size to provide maximum exhaust velocity for the existing temperature and pressure conditions at the exhaust nozzle. If this opening is too large, the maximum thrust cannot be developed by the engine.

When afterburning is initiated, the temperature at the exhaust nozzle increases greatly. This temperature increase causes additional expansion of the exhaust gases, and if the nozzle opening is not enlarged the pressure and temperature in the afterburner will become excessive. The J57 engine has an adjustable exhaust nozzle with an indicator to tell the pilot whether the nozzle is in the open or closed position. This indicator is located on the right side of the instrument panel next to the pressure ratio indicator.

THE EXHAUST NOZZLE POSITION INDICATOR.

The exhaust nozzle position indicator is a simple instrument. It consists of two solenoids which rotate a spring-centered, three-position card within a small

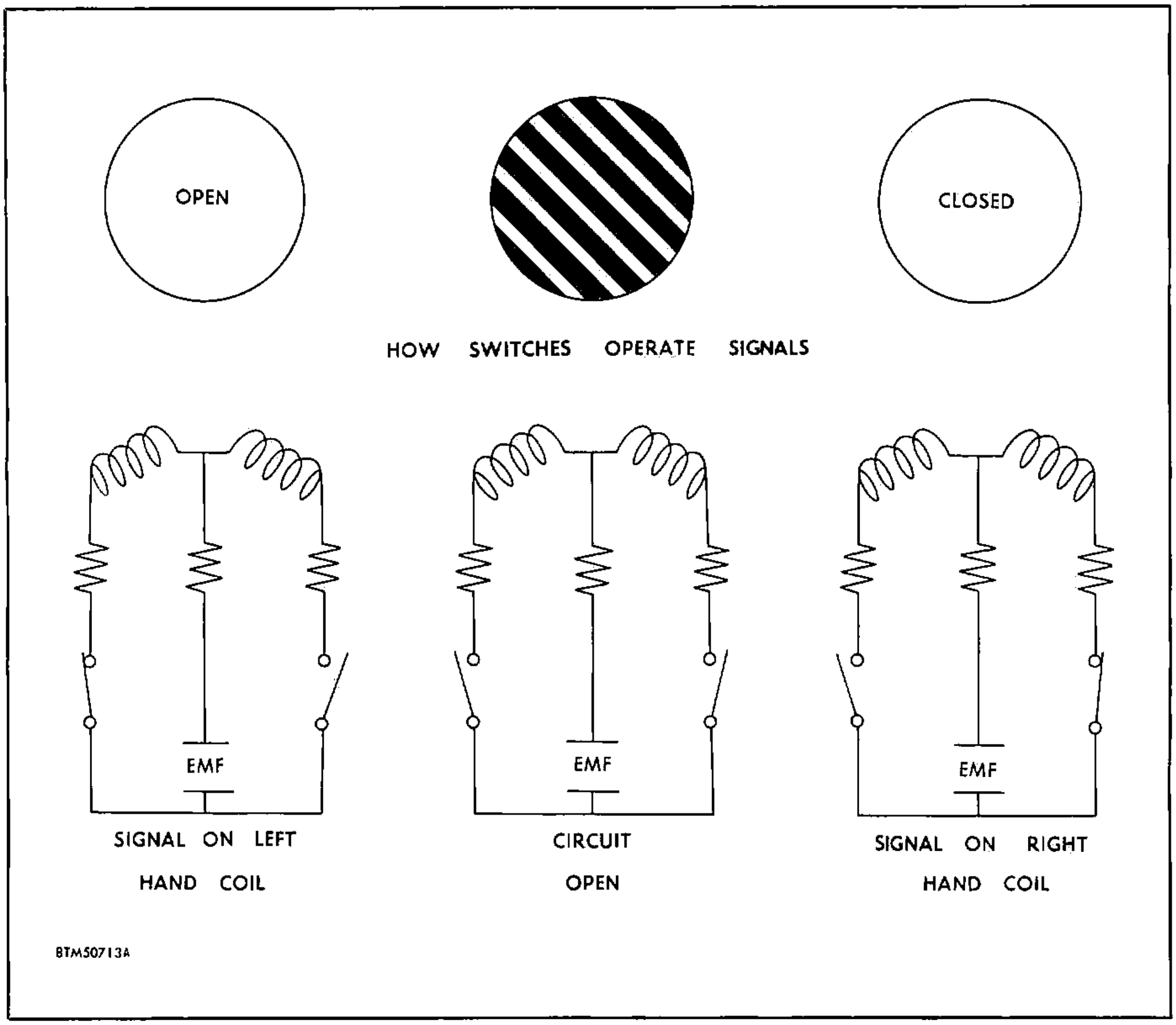


Figure 4-11. Afterburner Exhaust Nozzle Position Indication Circuit

sealed case. A window in the front of the case shows one of the three positions of the card.

There are three indications which the exhaust nozzle position indicator can give; the one which appears in the window at any particular time depends upon whether one of the two solenoids is energized or neither solenoid is energized. In figure 4-11 you can see the three types of indications and the corresponding electrical schematic that causes each of them to appear. Note that there are two switches in the system, only one of which can be actuated at any particular time. In the left diagram, you can see how the *open* solenoid is energized when the left switch is closed. The circuit is completed to ground from the airplane's 28-volt, d-c system, represented by emf (electromotive force). The middle diagram shows both switches open, so neither solenoid is energized. The diagonal lines (barber-pole) appear on the dial because a spring holds the rotating card in the center position. This indication appears when both switches are open or when power failure occurs. In the third diagram, the circuit is completed through the other switch energizing the *closed* solenoid. The CLOSED indication will appear as long as that switch is closed and the power supply is not interrupted.

HOW THE SYSTEM OPERATES.

The two switches are attached to the right side of the engine just below the compressor bleed valve. As you can see in figure 4-12, two cables connect the switch to the nozzle positioning cylinders. The upper cable is the nozzle position cable; the lower cable is a temperature compensating cable. Both are attached to spring plates in the switch assembly and serve to

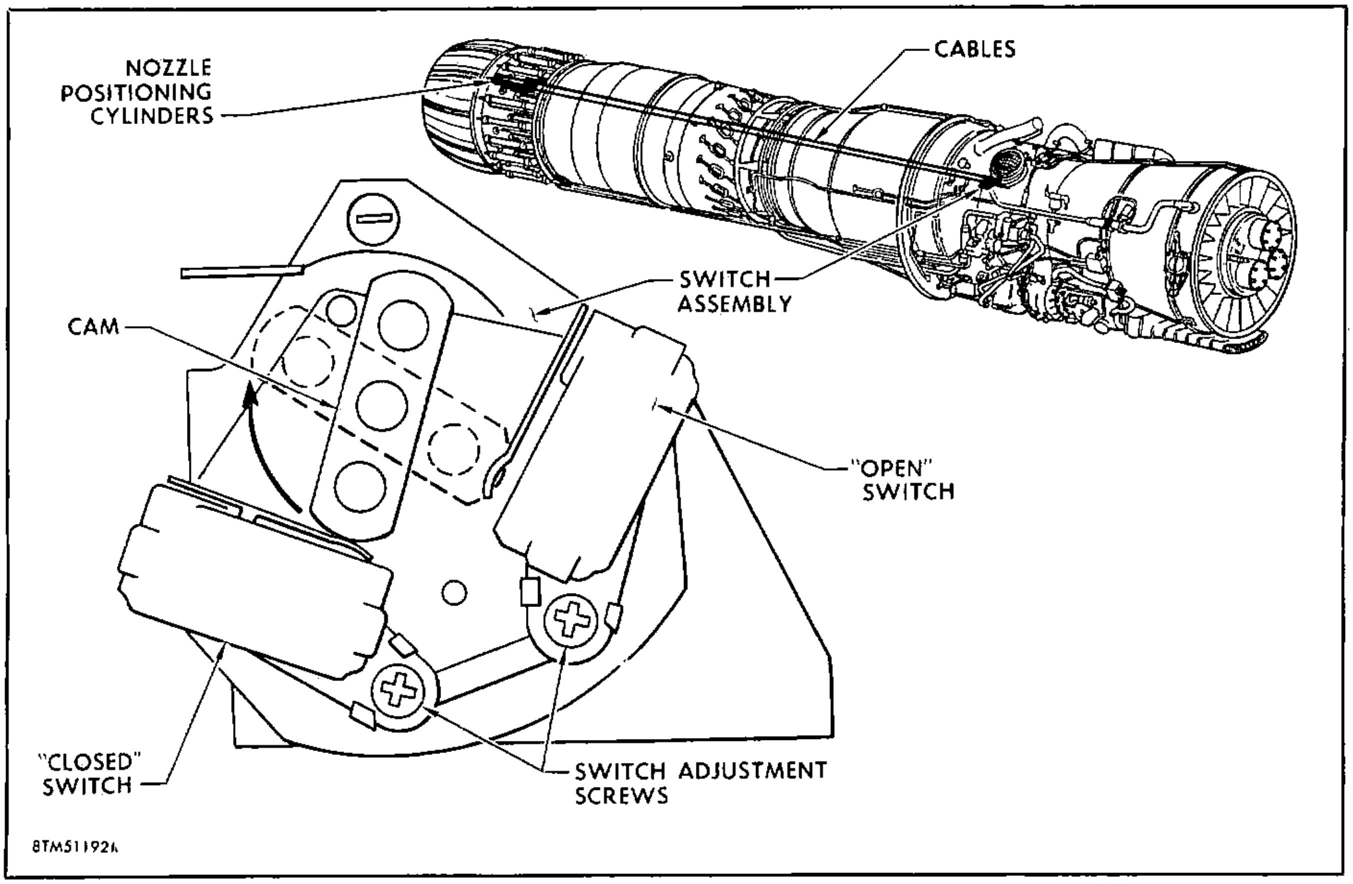


Figure 4-12. Afterburner Exhaust Nozzle Position Switches

rotate a cam. The enlarged view of the switch assembly shows how the cam actuates the switches.

Note that figure 4-12 shows the actuating arm of the *closed* switch depressed by the cam. With the switch in this position, the circuit of the indicator is energized as shown on the right view of the electrical schematic which was just discussed and the CLOSED reading shows in the window. The dotted line showing the outline of the cam shows where the cam stops when the afterburner nozzle is open. This position results in an OPEN indication because the *open* solenoid is energized. It should be obvious from the illustration that both switches are open momentarily as the cam rotates from one switch to the other. Thus, the "barber-pole" shows on the indicator very briefly each time the afterburner nozzle is opened or closed.

You should keep in mind that the switching mechanism does not open or close the afterburner; that operation is accomplished automatically when fuel pressure in the afterburner fuel control reaches certain limits. The sole purpose of the switch is to energize the correct indicator solenoids at the correct time.

HOW THE SYSTEM IS RIGGED.

From time to time, it becomes necessary to rig a cable-operated system. The afterburner position indicator system is no exception. Changing of a component in the system, a faulty indication in the cockpit, or a broken cable is sufficient reason to re-rig the system. First of all—what is rigging? Rigging is the act of making a series of adjustments to a mechanical or electro-mechanical system so that it will perform its function properly. In the afterburner position indicator system we adjust two turnbuckles (and, if necessary, the two switches) to achieve this goal.

To rig the system properly, we must have an SE-0947 rigging gage. Two models of the rigging gage are used: the J57-P-41 engine uses an SE-0947 gage, while the J57-P-23 engine uses an SE-0947-801 gage. They are quite similar and perform exactly the same function. The only difference between the two tools is that their methods of attachment to the engine differs. The gage should be attached to the position indicator switch mounting bracket as shown in view B of figure 4-13. As you will note, the gage has a movable pointer, a scale, and two rig pin holes. The rig pin holes (with rig pins installed) are used to keep the correct preload on both the cam and switch plate return springs. The dial indicator tells you whether you have changed your rigging while tightening the cables. If the rigging has been changed, the pointer will be pulled away from its index (zero) point. The switches are adjusted

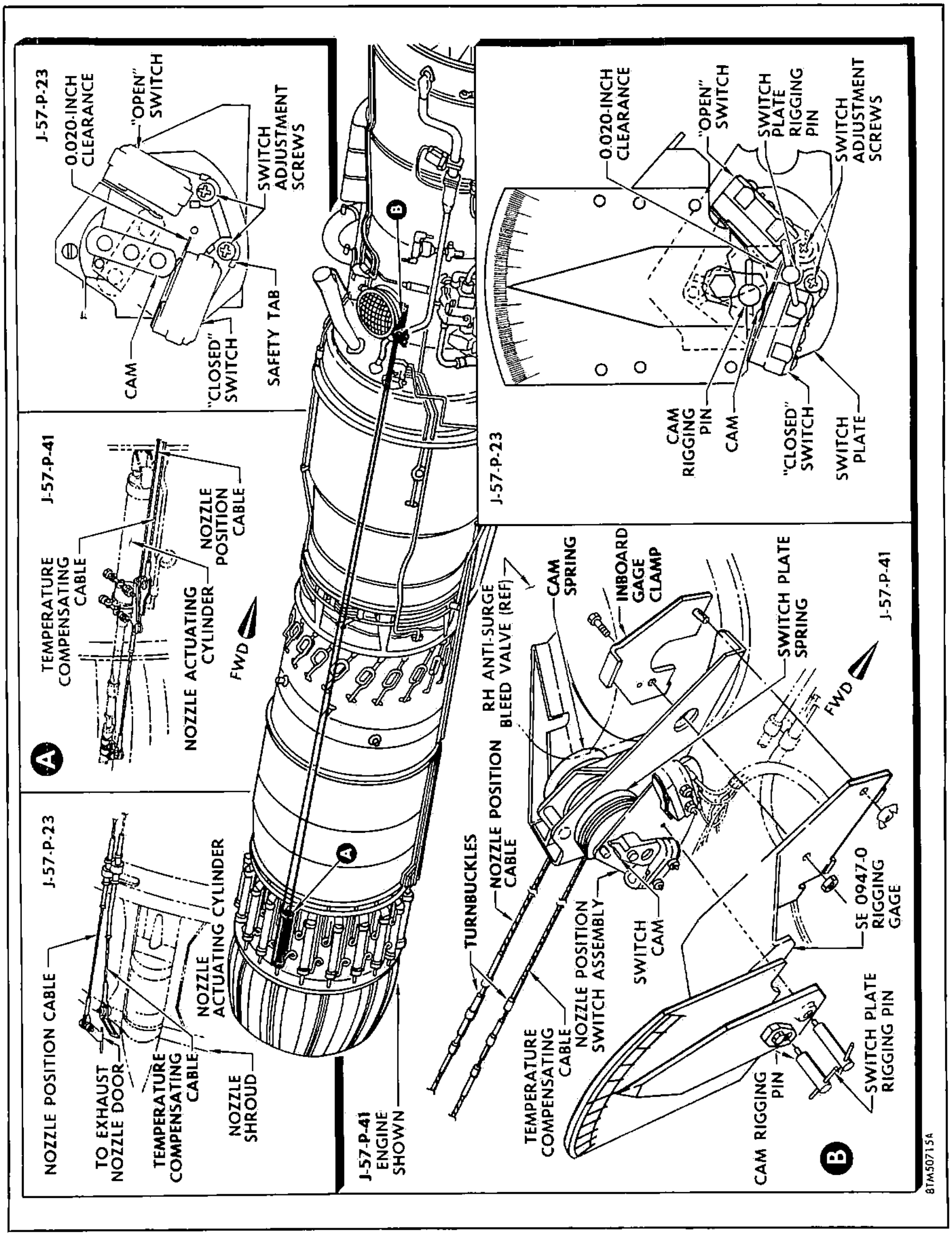


Figure 4-13. Rigging the Afterburner Exhaust Position Indicator System

by moving them physically—that is, the entire switch is moved either closer to or farther from the operating cam.

These instructions are for familiarization purposes only and are not intended as instructions for rigging the system. For specific rigging instruction, refer to your Power Plant Maintenance Handbook, T.O. 1F-102A-2-4.

MAINTENANCE OF THE AFTERBURNER POSITION INDICATOR SYSTEM.

Usually, any trouble encountered in the afterburner position indicator system can be immediately corrected by the rigging process just mentioned. However, if the rigging of the system is satisfactory, the indicator itself should be checked against a known satisfactory unit. In case the trouble still exists, a continuity check of the wiring should isolate the trouble.

EXHAUST TEMPERATURE INDICATOR.

The exhaust temperature of a jet engine, like the cylinder head temperature of a reciprocating engine, is a good indication of the over-all operating temperature of the engine. By way of further comparison, the jet engine exhaust temperature indicator is very similar to the instrument used to measure the cylinder head temperatures of piston engines. Both are thermocouple thermometers which measure the difference between electrical potentials of two metals in contact with each other. Figure 4-14 shows the exhaust temperature indicator and one of the four thermocouples used in the F-102A exhaust temperature indication system.

THERMOCOUPLES.

A thermocouple is a combination of wires, each wire made of a different metal with each having a different electrical potential. If two such wires are connected together at one end and that junction point is heated above the ambient temperature of the opposite end of each wire, the joined wires become a source of electricity, the potential of which varies with the temperature. This physical phenomenon, known as thermo-electric effect, is the principle behind every thermocouple.

The thermocouples used in the exhaust temperature indicating system are made of chromel-alumel material. They are mounted in probes which project into the tailpipe section of the engine just aft of the turbines, and are approximately 90° apart. The thermocouples are connected in parallel with each other by leads made of chromel and alumel material.

THE EXHAUST TEMPERATURE INDICATOR.

The temperature of the thermocouples is shown on the dial of the indicator in degrees centigrade—the scale is calibrated from 0° to 1000°. As you can see in

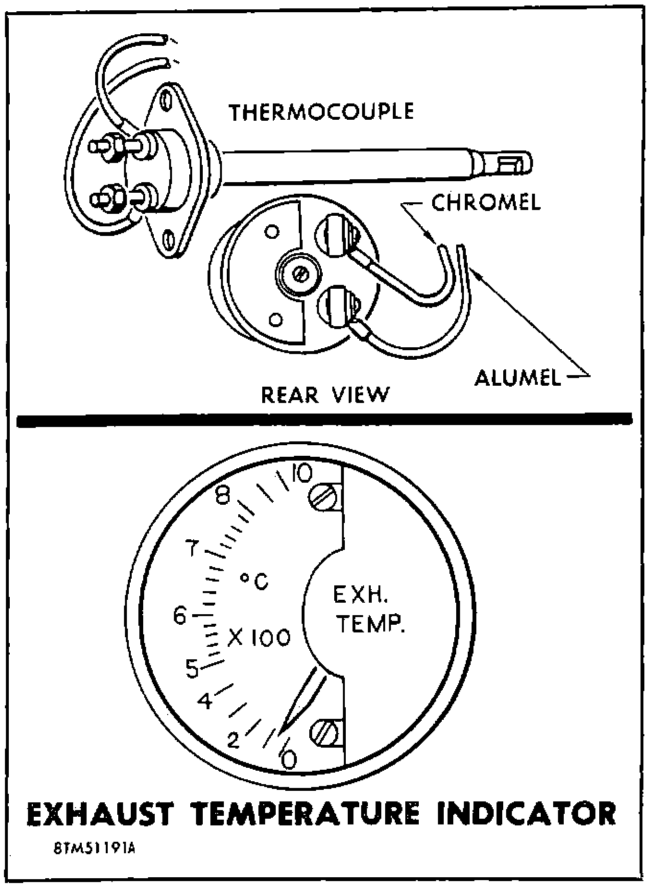


Figure 4-14. Exhaust Temperature Indicator and Thermocouple

figure 4-14, there are no external controls or adjustments on the indicator. Internally, the indicator consists primarily of a moving coil mounted on pivots within a curved permanent magnet. Rotation of the coil is limited by two springs, one on each end of the coil. A pointer moves with the coil to show the temperature indication. The entire indicator is sealed and filled with helium.

Now take a look at the schematic illustration (figure 4-15). Note that the leads from the thermocouples connect to the coil through the springs, making a complete circuit. As you know, when current flows through a conductor, such as this coil, a magnetic field is set up. This magnetic field around the coil has both strength and direction, just as the field around a permanent magnet.

Note also that the coil is situated directly between the ends of the curved permanent magnet. The magnetic flux around such a magnet is concentrated between the two ends or poles. Thus, the coil tends to take a definite position between the ends of the permanent magnet. As you already know, like poles repel and unlike poles attract, so the coil tends to turn until each of its poles is close to the opposite pole of the permanent magnet.

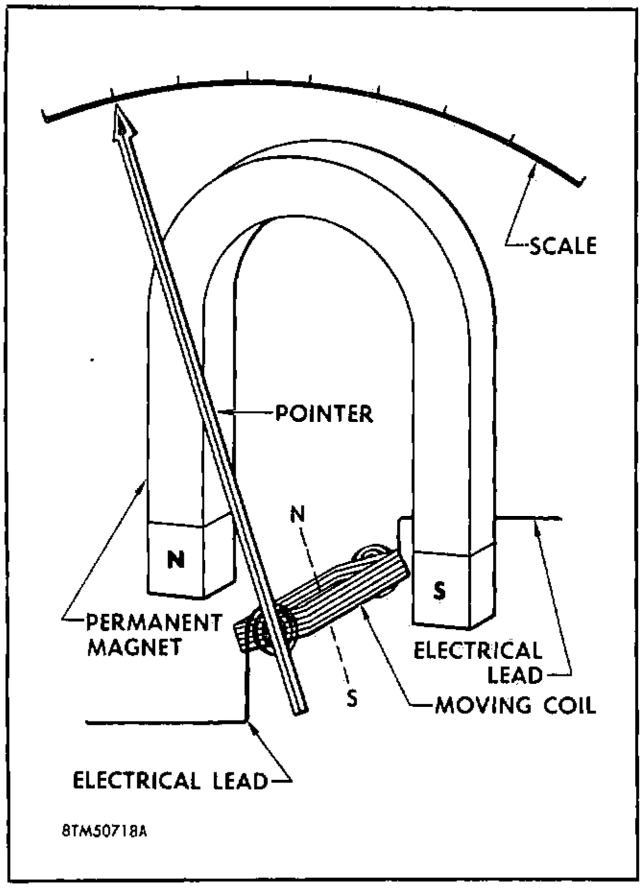


Figure 4-15. Exhaust Temperature Indicator Operational Schematic

If there were no restraining springs, the coil would always line up perfectly (with the poles in the normal relative positions) whenever there was a temperature difference between the "hot" end (thermocouple) and the "cold" end (indicator) of the system. Obviously, such an indicator would be of no value since it would always read the same. By the use of springs of the correct tension, the coil is permitted to move only an amount proportional to the strength of the magnetic field around it, which, as mentioned earlier, varies with the temperature differential.

Temperature Compensation.

The coil springs just discussed also serve another purpose; they are temperature compensators. The amount of rotation of the coil depends on the strength of its magnetic flux, which in turn is proportional to the difference in temperature between the thermocouple and the indicator. But that isn't exactly what we want to know. If the exhaust temperature is 600°C, we want the indicator to say 600°C, regardless of the temperature in the cockpit.

You can see then that the indicator must be set to read the cockpit temperature first so that the additional rotation of the coil will cause it to read actual exhaust temperature. The springs accomplish this for us because they are made from laminations of different metals which react differently to temperature changes.

Figure 4-16 shows how these bimetallic springs work. Note that the strip of brass in the laminated metal expands more than the strip of iron when heat is applied, causing the laminated strips to bend. In the same manner, the springs in the exhaust temperature indicator tighten up or straighten out with changes in cockpit temperature. In this way the coil, and therefore the pointer, are rotated to indicate the cockpit temperature.

The additional rotation, caused by the difference in temperature between the indicator and thermocouple, brings the pointer to a position proportional to the total exhaust temperature. Thus, the indicator pointer reflects the total of the temperature at the indicator, plus the difference between the temperatures at the indicator and the thermocouple. If you disconnect the thermocouple leads the indicator will show the approximate cockpit temperature.

Another temperature compensation problem results from the variations in electrical resistance of most metals with temperature changes. At a given temperature difference, the voltage generated in the circuit will produce a current inversely proportional to the resistance (Ohm's law). For any particular temperature difference between the "hot" and "cold" ends of the system, the current must always be the same. Therefore, a "neutralizer" is included in the indicator. This neutralizer is a resistor in which the resistance becomes less as the temperature increases. In that way

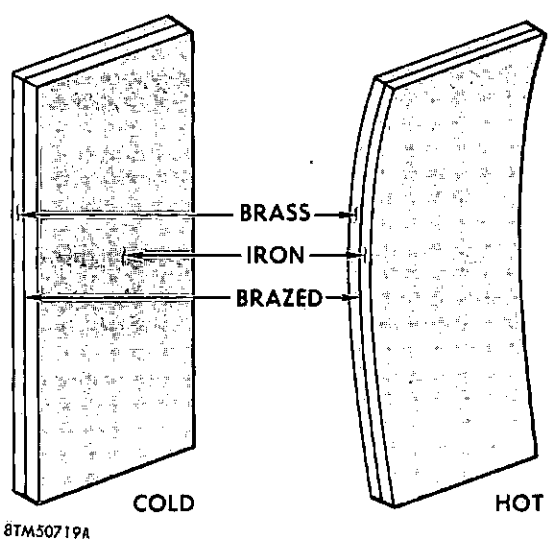


Figure 4-16. Effect of Temperature Changes on Bimetallic Strip

it keeps the total resistance of the system constant for any temperature difference.

SYSTEM CALIBRATION PROVISIONS.

We have discussed the importance of keeping the resistance of the system completely constant so the indicator will receive the right amount of current and give a correct indication. There are several things that might alter this resistance and consequently affect the accuracy of the indication. For example, if you replace any of the components of the system, as would happen when you change engines, there could be a slight change in the resistance. Even changing a terminal on the leads could alter the resistance. Therefore, a calibrating resistor is included in the system and is located on the right side of the cockpit above the master warning control box. You can use an ordinary Wheatstone bridge-type of tester to determine the correct adjustment of the resistor.

A special testing unit (SE-0783) is required to test the accuracy of the complete system. This test unit operates on a 115-volt, 60-cycle power source. You will note in figure 4-17, that the test unit has a lead incorporating a probe heater with an extension handle. The calibration and functional check of the system is made by heating each probe individually and checking the temperature indication in the cockpit against the reading of the test unit. When each probe has been checked individually, probes should all have heaters installed and the complete system checked against the test unit. If this indication is not within ±10°C of the indication on the test unit, the system must be calibrated. This is accomplished by adjustment of the calibrating resistor in the cockpit.

FUEL FLOW INDICATING SYSTEM.

Any large jet engine uses tremendous quantities of fuel. The rate at which the engine uses fuel varies greatly, depending upon the power lever setting. Since airplane space and weight restrictions limit the amount of fuel that can be carried, the jet pilot is always vitally concerned with how fast the fuel supply is being consumed. Without this information, he cannot estimate accurately how long he can stay away from his base.

The rate of fuel consumption is also an indication of the efficiency of the engine. For these reasons, the F-102A uses a fuel flow indicating system. The two main components in the indicating system are shown in figure 4-18. The transmitter is mounted on the engine, while the indicator is situated on the engine instrument portion of the instrument panel.

THE FUEL FLOW TRANSMITTER.

The transmitter for the fuel flow indicating system is attached to the right side of the engine, adjacent to the oil pump and accessory drive housing. It is located in the main fuel line between the fuel control unit and the fuel oil cooler. An a-c synchro-transmitter is contained within the transmitter unit to send signals to the fuel flow indicator in the cockpit.

From the discussion of the pressure ratio indicator, you should be familiar with how the synchro-transmitter position of this component operates, so just the mechanical portion will be discussed here. The operational schematic in figure 4-19 will help you to see how the flow transmitter operates. Fuel flowing through the transmitter enters the port on the right and leaves through the port on the left, as shown by the arrows. Notice that a hub, with a vane projecting above it, is mounted in the flow area. A spring within the hub resists the force of the fuel flow, and tends to keep the vane in the upstream position at all times. The actual position taken by the hub and vane assembly depends on the rate of flow of the fuel which surrounds it. Therefore, we can call the vane and hub assembly the gage unit of the fuel flow transmitter. Now, let's follow the train of action to see how the position of the vane and hub is carried to the synchro.

How the Transmitter Produces a Signal.

Notice in figure 4-19 that there is no direct mechanical connection between the gage unit, which does the measuring, and the transmitting unit, which signals the indicator in the cockpit. The shaft of the hub is not geared to the sector shaft. Instead, a permanent magnet on the hub shaft surrounds a permanent bar magnet which drives a pinion gear. The relative positions of these magnets will tend to stay the same; when the gage unit rotates the ring magnet, the bar magnet turns with it so that the opposite poles of the two magnets are always lined up. You can see then how the reaction of the bar magnet and pinion to the movement of the ring magnet will move the sector shaft and drive the synchro. This method of driving the synchro permits the electrical part of the fuel flow transmitter to be isolated from the fuel-carrying part.

THE FUEL FLOW INDICATOR.

The fuel flow indicator is calibrated to show the rate of flow from 0 to 12,000 pounds per hour. As you have seen in the illustration of the flowmeter (figure 4-18), the dial is graduated every 100 pounds up to 3000 pounds, and in 1000 pound increments from 3000 to 12,000 pounds.

Since the fuel flow transmitter positions a transmitting synchro, it should be obvious that the indicator is a synchro instrument containing a synchro-receiver. Power to operate this synchro system comes from the airplane's 26-volt, 400-cycle, a-c electrical source. You have already learned the fundamentals of the synchro-receiver, so further details on this receiver will not be discussed.

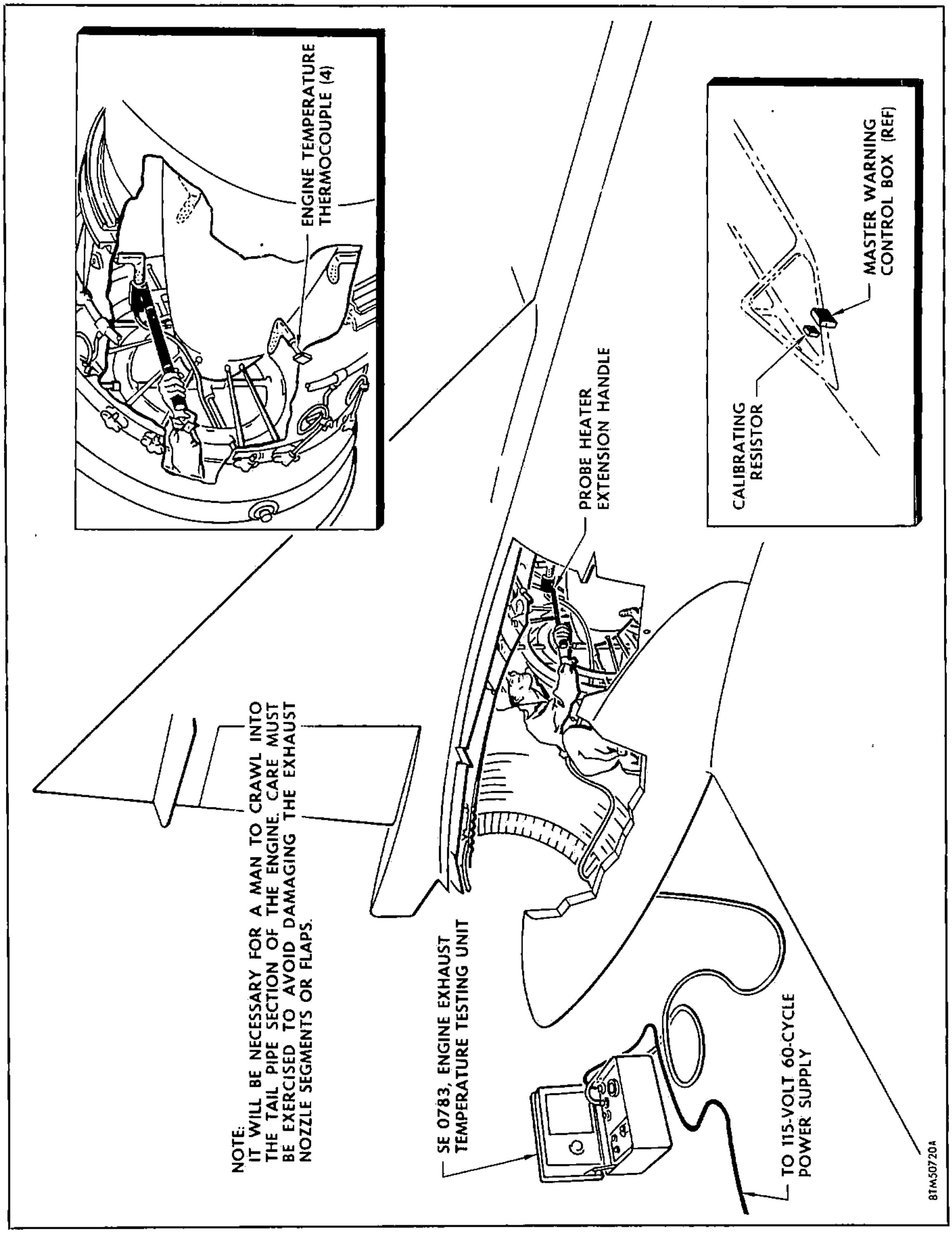
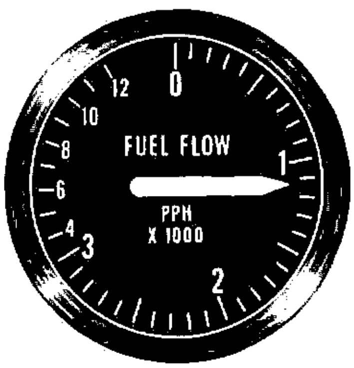
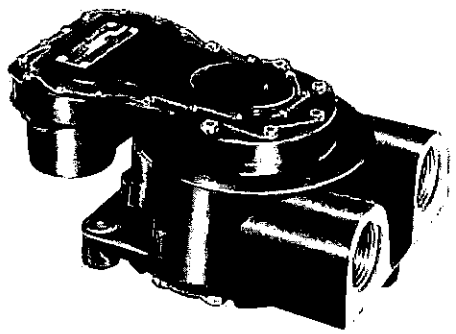


Figure 4-17. Exhaust Temperature Indicator System Test and Calibration



FUEL FLOW INDICATOR



FUEL FLOW TRANSMITTER

8TM50721A

Figure 4-18. Fuel Flow Indicator and Transmitter

Maintenance of the System.

If you should have any reason to doubt the accuracy of the indicator, disconnect the transmitter and plug the leads into a master synchro-transmitter. If the indicator pointer smoothly follows the movement of the test transmitter, the trouble lies in the airplane's fuel flow transmitter. There are no external provisions for adjusting either the indicator or the transmitter, so you must replace the faulty unit.

THE POWER PLANT WARNING SYSTEMS.

There are five individual warning systems for the J57 jet engine. These include the structural overheat, fire detector and overheat, fuel pump, oil pressure, and engine anti-ice warning systems. If you consider the meaning of the name "warning system," you will realize that these systems merely notify the pilot of the dangerous conditions in the power plant. The pilot himself must take the corrective action to eliminate the condition or to protect himself and the airplane from any possible danger.

The warning lights for all of the warning systems in the F-102A are centrally located in one master warning light panel. The panel, consisting of 16 different lights, is located on the right side of the pilot's instrument panel. The warning lights are numbered from 1 through 16. In addition to its number, each light also has the system or function name stencilled on the lens. These names are not visible, however, unless the bulbs in back of the warning light lenses are illuminated.

In addition to the 16 individual warning lights, the F-102A also has an amber-colored master warning light. This master light is located on the upper right side of the main instrument panel. When any condition in the airplane causes one of the 16 warning lights to light up, the master warning light also comes on. Since the master light is located directly in front of the pilot, it attracts his attention more quickly than the individual lights. It notifies him to check the individual lights on the warning light panel.

After the pilot has taken the necessary corrective action, he can press the RESET switch on the right control console. This switch extinguishes the master warning light. If another of the 16 warning lights should come on, the master warning light will light up again. The pilot then determines which warning light is on, takes the necessary corrective measures, and then presses the RESET switch again.

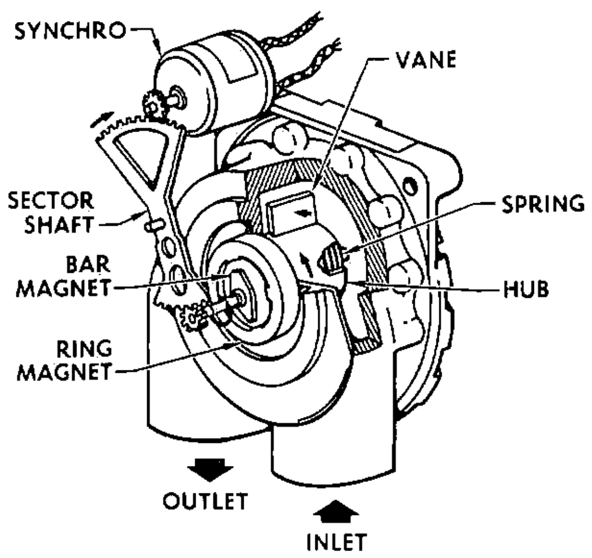


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Figure 4-19. Fuel Flow Transmitter Operational Schematic

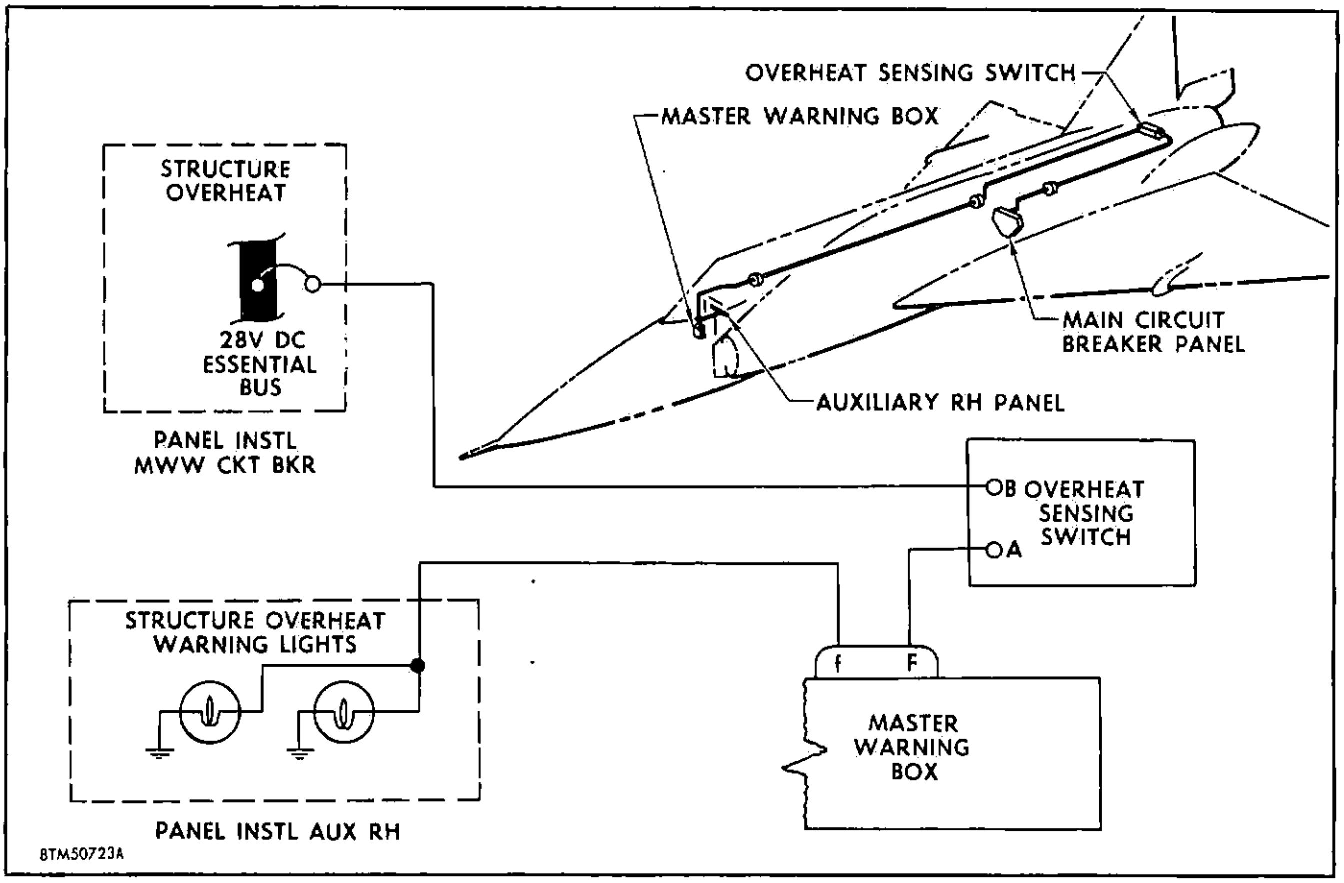


Figure 4-20. Structural Overheat Warning System Schematic

The power plant warning systems use the warning lights numbered 1, 4, 5, 7, and 8. Each of the power plant warning systems will be discussed individually in the following sections of this chapter.

THE STRUCTURAL OVERHEAT WARNING SYSTEM.

The structural overheat warning system uses the first (top) warning light on the warning light panel. This warning system notifies the pilot of excessive heat conditions in the fuselage structure at station 614.00 or the area just ahead of the tail cone fairings.

The detecting device for the structural overheat system is a mercury-operated temperature sensing switch which is attached directly to the fuselage structure. Only that portion of the switch which touches the fuselage structure is sensitive to temperature changes. This limits the switch to sensing structural temperatures only. The switch is preset to actuate whenever the structural temperatures rise above 245°F.

When the switch is actuated, both the master warning and structural overheat warning lights come on. These lights notify the pilot that a power reduction should be made and a reduced-speed flight condition established until the temperature decreases. If the system warning lights come on during ground run-up, you should shut the engine down immediately and investigate the condition.

Figure 4-20 shows the electrical schematic of the structural overheat warning system. Note that the current is taken from the 28-volt d-c essential bus. The five-amp, push-pull type circuit breaker, which protects the circuit from current overloads, is located on the main wheel well circuit breaker panel. Following the current from the essential bus and the circuit breaker, note that it travels to the switch. When this switch is actuated by an overheat condition, the current will flow to the master warning box where it causes the master warning light to illuminate, and then on the individual system warning light. The structural overheat warning system as described above is used on the very early models of the F-102A.

THE FIRE DETECTOR AND OVERHEAT WARNING SYSTEM.

The fire detector and overheat warning system, as its name implies, does two things—it notifies the pilot of an engine overheat condition and notifies him of a fire condition. Although each section of the warning system does just one job, both sections use the same warning light in the cockpit. By routing the electrical signal from the overheat circuit through a flasher, the warning light is made to flash on and off whenever an engine overheat condition exists. In the event of fire, the warning light will come on and burn steadily.

You should not confuse the engine overheat warning system with the structural overheat warning system. As just mentioned, the structural overheat warning system is used on just the very early models. The fire detector and overheat warning system, described in the following paragraphs *is* used on *all* models.

Both the overheat and the fire warning sections of the warning system use detector loops. These detector loops are the temperature sensing elements of the system. The loops are actually a type of coaxial cable with electrical connecting fittings on each end. The cables are constructed of an inner electrical conductor within an outer sheath of corrosion-resistant alloy.

The space in the cable between the inner conductor and the outer covering contains a thermistor-type heat-sensitive compound. The electrical resistance of this heat-sensitive compound varies inversely to the temperature. Under normal operating conditions the compound acts as a good insulator; but when a "hot" spot develops anywhere along the detector cable, the resistance of the compound drops and allows current to flow from the inner conductor to the outer portion of the coaxial cable. This completes the detector circuit to ground and the warning light comes on.

How the Fire Detector and Overheat Warning System Operates.

Figure 4-21 shows where the components of the warning system are located in the airplane and how the system operates. Note in the upper portion of the illustration that the overheat detector loop and the fire detector loop are situated in different sections of the engine compartment. The detector relays, the overheat flasher, and the detector control boxes are located in the upper electronics compartment.

To get a good idea of how the warning system operates, let us trace the current flow in the schematic in the lower portion of figure 4-21. The current for the warning system, like all of the other warning systems, is taken from the 28-volt d-c essential bus. After it passes through the push-pull 5-amp circuit breaker, the current flows to the two detector control boxes and to the junction point of the two relays. Note that the warning lights must receive their current from either one of the detector control boxes before the lights will illuminate. However, the control boxes cannot send current to the lights until the warning light circuit is grounded at some point.

In the preceding section, you learned that the detector loops ground out whenver they are subjected to a "hot" condition that lowers the internal resistance of the detector loops. Whenever one of the cables grounds out, its detector control box circuit will be completed, and current will flow to the lights. If the overheat loop is subjected to the "hot" spot, the current from the overheat detector control box has to flow to the overheat detector flasher before it passes on to the warning lights. This causes the warning light to flash on and off during the overheat condition. Since the fire detector circuit does not have a flasher unit, a fire condition causes the warning light to burn steadily.

The test switch, shown in the right hand portion of the electrical schematic, will allow you to ground check the warning system without heating any part of the coaxial cable detector loops. This switch is located to the left and above the tachometer. A small test light is mounted on the right side of the test switch. The test switch does the same job that the detector loops do during an overheat or fire condition —it grounds the circuit and allows the respective control box to send current to the warning lights. By positioning the switch to either the FIRE or OVHT position, the pilot can determine whether the warning systems are functioning properly.

Maintenance of the Fire Warning and Overheat Warning System.

Malfunctions developing in this warning system may stem from the control boxes or the detector loops. Both the boxes and the loops are delicate units and can be damaged during installation by rough handling or improper installation. Because of the nature of this warning system, the only indications of any malfunction will be the warning light coming on when neither an overheat or fire condition exists, or the lights not coming on when the undesirable conditions do exist.

From the discussion of how the warning system operates, you should know that a premature warning light indication is caused by some part of the detecting circuits shorting out to ground. The easiest way to determine whether the control boxes or the detector loops are defective is to replace the control box with a unit that is known to be good. If the substitute control box does not cure the difficulty, you will have to check all of the detector loops for possible breaks or loose electrical connectors. No rework is allowed on any of the loop segments, so you will have to replace any defective detector loop with a serviceable item.

THE ENGINE FUEL LOW-PRESSURE WARNING SYSTEM.

The engine fuel low-pressure warning system notifies the pilot when sufficient fuel pressure is not being supplied by the main engine stage of the fuel pump.

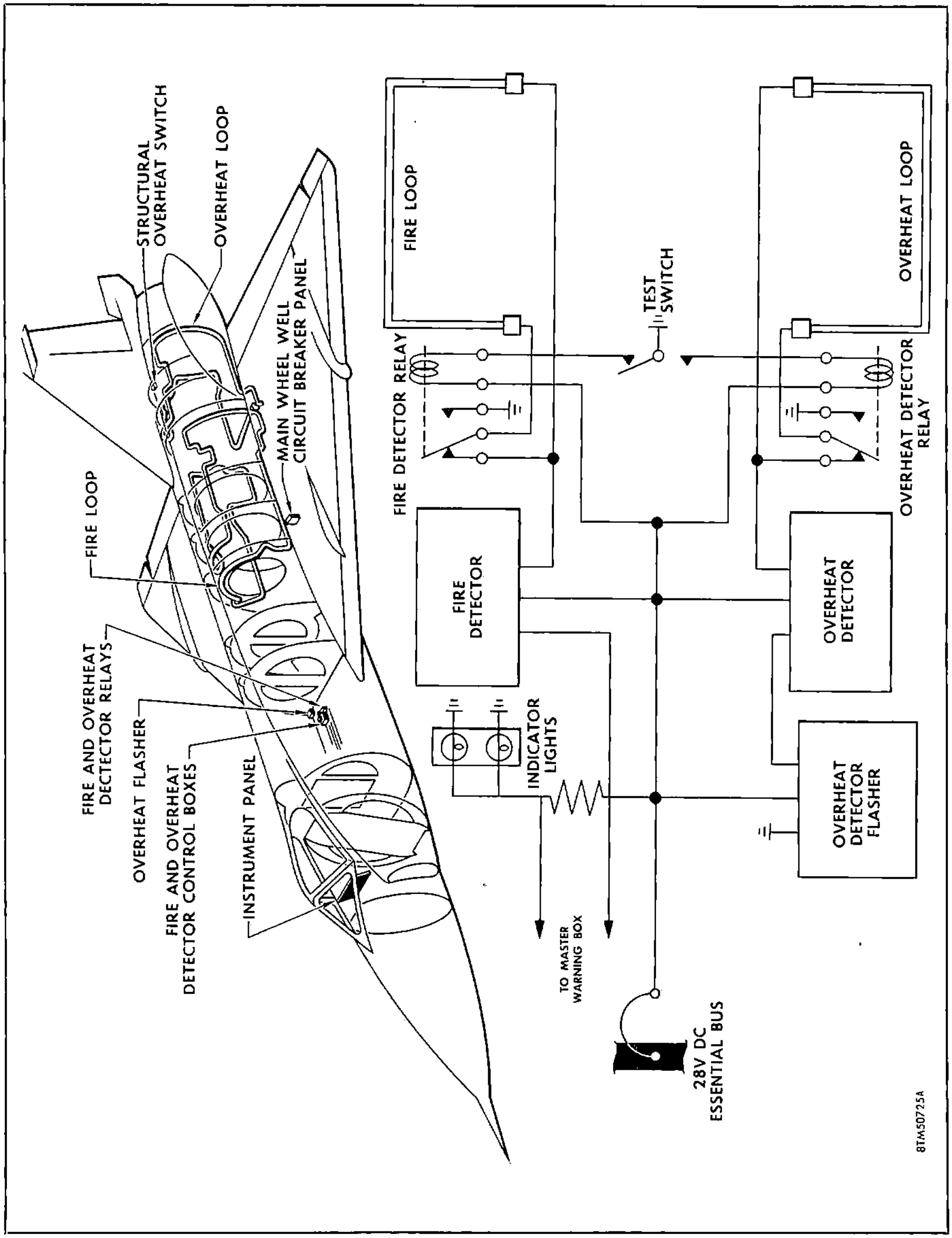


Figure 4-21. Fire Detector and Overheat Warning System Schematic

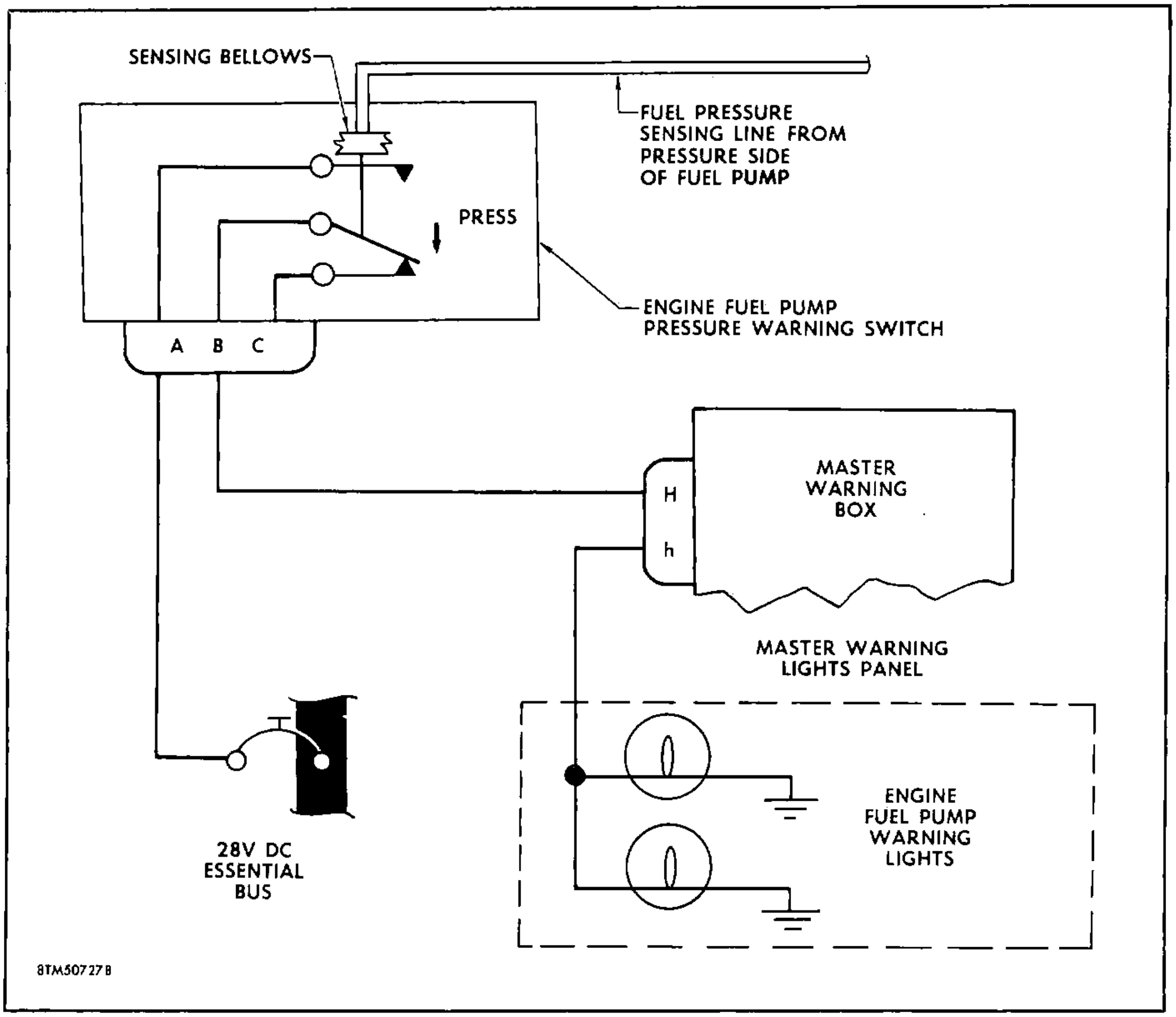


Figure 4-22. Engine Fuel Pump Low-Pressure Warning System

The system consists of a warning light on the warning indication panel, and a pressure switch on the lower left side of the oil pump and accessory drive housing of the engine. A pressure sensing line is installed between the switch and the pressure side of the fuel pump engine stage. Note in figure 4-22, that when fuel pressure drops in the fuel pump outlet line, the pressure warning switch closes and allows current from the 28-volt d-c essential bus to illuminate the warning light in the cockpit. As you can see in the illustration, the pressure warning switch is operated by the sensing bellows. Fuel pressure is routed to the outside of the bellows and forces the switch to open. The switch is shown in the open position in the schematic (figure 4-22). When the fuel pressure is low or when the engine is shut down, the sensing bellows expand and close the switch. This allows current to flow from terminal B to the master warning box and then to the engine fuel pump warning lights.

The warning lights will always come on when the engine is first being started. After the engine has attained starting speed, however, the fuel pump will put out sufficient pressure to force the bellows to open the switch and the lights will go off. The sensing bellows of the pressure switch are set to open the switch before the fuel pressure reaches 125 psi and to close the switch when the pressure drops below 105 psi.

Maintenance of the Fuel Low-Pressure Warning System.

You will find that the maintenance requirements on the fuel low-pressure warning system will be about the same as the requirements on the other warning systems. You should keep in mind, however, that the warning lights will light temporarily when the engine is being started. After the engine has attained idling speed, the lights should go off if the warning system is functioning properly.

The malfunctions of the warning system will include premature lighting of the warning lights or the lights will not illuminate at all. As you have noted in figure 4-22, the warning lights are supplied current whenever the pressure switch closes. When you trouble shoot a malfunctioning fuel low-pressure warning system, always check the light bulbs first. If they are not burned out, you will probably find it best to replace the pressure switch with a serviceable unit. Although this is not actually trouble shooting the warning system, the ease with which the switch can be replaced makes this a simpler remedy than checking the entire circuit with a continuity light or voltmeter. If replacing the switch does not correct the trouble, however, you will have to check out the entire circuit for an open or broken lead, or a short to ground. If the warning system still malfunctions after completing the circuit trouble shooting, the trouble lies in the fuel pump, and this requires replacing the pump.

THE ENGINE OIL LOW-PRESSURE WARNING SYSTEM.

The engine oil low-pressure warning system notifies the pilot whenever the engine oil pressure drops below 36 psi. The warning lights will go off when the oil pressure raises back to 40 psi. The pressure switch, which detects the oil pressure, is mounted on the left side of the engine just aft of the oil tank.

You will note in figure 4-23 that the oil low-pressure warning system is almost identical to the fuel pump low-pressure warning system. As you might well imagine, this warning system operates in the same manner.

Whenever the oil pressure drops below the 36 psi level, current from the 28-volt d-c essential bus will flow through the 5-amp push-pull type circuit breaker, pass through the pressure switch, and then illuminate the warning lights in the cockpit. When the oil pressure increases to 40 psi, the reverse situation takes place. The pressure switch is forced open by the oil pressure, and this breaks the warning light circuit. When this occurs, the warning lights go out.

If the oil low-pressure warning system should malfunction, you will find it best to follow the same procedure given for trouble shooting the fuel pump low-pressure circuit. After checking the warning light bulbs and the circuit breaker, replace the pressure switch. The last thing to do is trouble shoot the entire circuit with a continuity light or voltmeter. If the malfunction still persists, however, you will have to remove and replace the engine oil pump with a serviceable unit.

THE ENGINE INTAKE ANTI-ICE WARNING SYSTEM.

The engine intake anti-ice warning system notifies the pilot whenever the engine anti-ice system malfunctions. Icing is a very serious matter on any part of the airplane, especially in the engine air intake area. Therefore, it is imperative that the engine intake anti-ice warning system operates properly.

The F-102A uses hot air taken from the engine compressor section to heat the inlet duct guide vanes and the accessory fairing area. The anti-icing air flow is controlled by electrically-actuated valves and regulators installed in each of the lines. The valves operate in conjunction with the anti-icing control system. When the valves are open, heated air flows into the engine inlet guide vane manifold. The heated air then passes inward through the double-walled accessory fairing and vents out the cap of the fairing.

How The Anti-Ice Control System Operates.

Before you can understand the operation of the engine intake anti-ice warning system, you must know how the anti-ice system operates when it is functioning normally. Therefore, we will analyze the circuits shown in figure 4-24 and follow the operation of the control system. This schematic shows the system as it would be during flight under "no-ice" conditions.

First, locate the ice detector assembly and the interpreter assembly. These two assemblies are the heart of the engine anti-ice control system. The detector assembly detects the presence of ice in the engine intake duct and then signals the interpreter assembly. The interpreter assembly energizes the anti-ice control relay which in turn actuates the anti-ice system.

The anti-ice system is controlled electrically by a three-position switch in the cockpit. This is shown in the lower part of figure 4-24. Note the three positions —AUTO, MAN, and OFF. First, let's see how the system works when the anti-ice switch is in AUTO. Note the 10-amp anti-ice power circuit breaker. All power for the detector and interpreter comes from the 28-volt d-c essential bus through the 10-amp anti-ice power circuit breaker. The current flows from the circuit breaker, through the ignition power relay, and then to the connection A1 on the interpreter assembly. The ignition power relay is energized only when the engine is being started. Consequently, the connection at A1 is always connected directly to power (across 2 and 3 as shown in the schematic) except during engine starting.

Following current flow A1, note that it travels through the switch in the ice detector assembly. Since a "no-ice" condition exists, the current flows back to A6 in the interpreter assembly. Note that relay R1 is connected to A6 and also to ground. Therefore, relay R1 is energized when the detector switch is in the "no-ice"

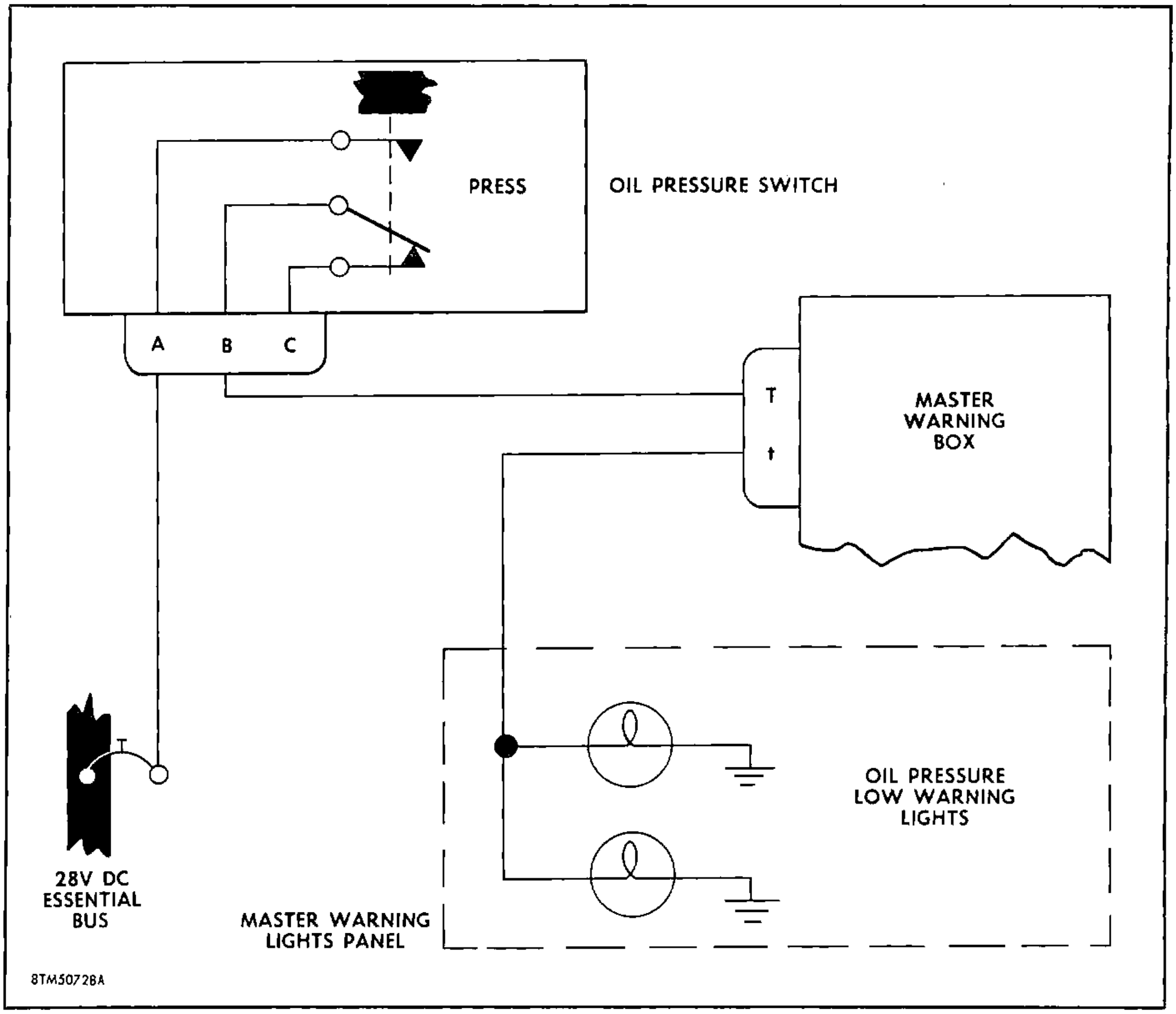


Figure 4-23. Engine Oil Low-Pressure Warning System Schematic.

position. Relays R3 and R4 are connected to B3 and contact C on the detector switch. These relays are deenergized in the "no-ice" condition as shown in the schematic. Since R2 is energized only when R4 is energized, R2 is also deenergized in the "no-ice" condition.

From the preceding discussion, you should have a general idea how the circuit functions when there is no ice in the detector tube.

Now, let's see what happens when the detector probe ices up. When the ice condition occurs, the detector switch closes. The operation of this switch is described later. Remember that A6 is connected to A1 through the switch in the R1 relay; therefore, B3 now has current since it is connected to A6 through C, B, and B2. Relays R3 and R4 now energize. This sequence results in the probe heater being connected to power through A1, the R3 relay switch and B4. The R2 relay and the heater at relay R1 are energized through R4. The R2 relay closes its switch to furnish a ground to the anti-ice control relay which then receives current from the 28-volt d-c bus through the 5-amp anti-ice control circuit breaker. The anti-ice control relay then closes to energize the anti-ice system.

The anti-ice system receives its power through the 5-amp anti-ice control circuit breaker. As the probe heater heats up, it melts the ice in the probe and the detector switch returns to the "no-ice" position. Relays R2, R3, and R4 deenergize, and the probe heater and the R1 relay heater are disconnected from power. Although the mechanical time delay at relay R2 is deenergized, it does not open at this time. This relay contains a clock mechanism that keeps the switch closed until 60 seconds after the relay has deenergized.

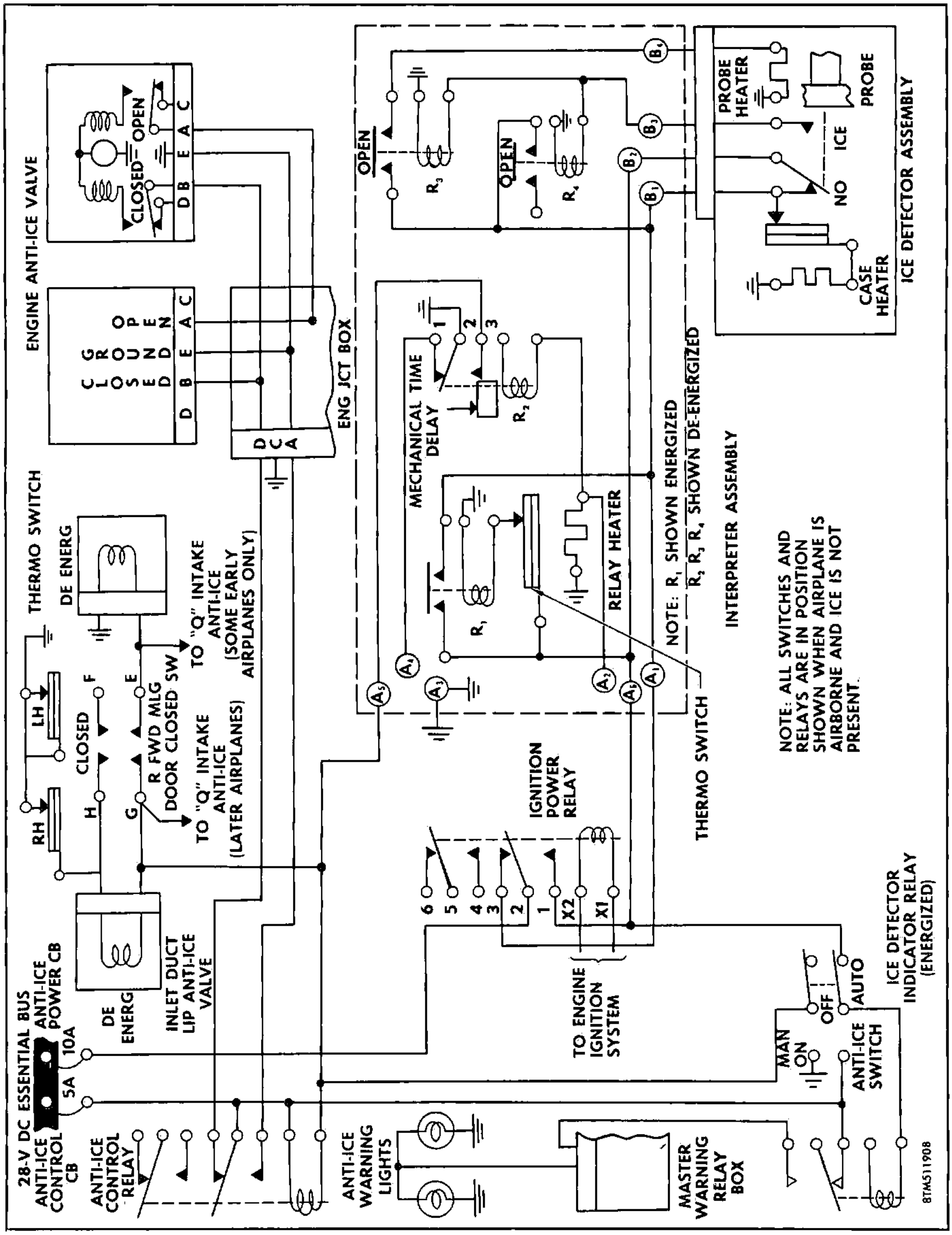


Figure 4-24. Engine Intake Anti-Icing Warning System Schematic

This delay feature insures that the anti-ice system will stay on long enough to do some good even though the detector switch returns to the "no-ice" position after just a few seconds.

How The Anti-Ice Warning System Operates.

Normally, the probe heater will melt the ice at the probe and return the probe switch to the "no-ice" position within 17 to 20 seconds. If it does not, it indicates that the detector is malfunctioning. To deenergize the relays during a malfunction and to keep the probe heater from overheating the probe, are the functions of the thermoswitch at relay R1. About 17 to 20 seconds after the detector switch moves to the ice position (across B and C) the R1 relay heater will cause the thermoswitch to heat up enough to break the circuit to R1 relay. The R1 relay switch will then open and A6 (also B) will no longer be connected to power. Since B3 is connected to B2 across the detector switch (at B and C) the R2, R3, and R4 relays will de-energize and the probe heater and the R1 relay heater will be disconnected. Sixty seconds later, the switch at the R2 relay will open the circuit to ground across 2 and 3; the anti-ice control relay will deenergize; and the anti-ice system will shut down.

Malfunction of the detector destroys the effectiveness of the automatic control system and the ani-ice system will not operate. If the pilot is informed of the malfunction, he can move the ANTI-ICE switch to MAN, thus grounding the anti-ice control relay. The anti-ice system will then operate until he moves the switch to OFF. Informing the pilot is the job of the anti-ice warning system. The warning lights on the right-hand auxiliary instrument panel illuminate and warn the pilot of the malfunction. These warning lights come on when the master warning box is energized. You can see in the schematic (figure 4-24) that the warning box is connected to contact 3 on the ice detector indicator relay. When the relay is energized, contact 3 will not be connected to power. Notice that the relay is connected to point A6 on the interpreter assembly through the anti-ice switch when it is at AUTO.

Now as we noted earlier, A1 is normally powered and A6 is normally connected to A1 (power); therefore, the detector indicator relay is normally energized except when the engine is being started and the ignition power relay is energized. The warning light will be on during a malfunction of the detector or interpreter assembly or when the anti-ice switch is at OFF. As you will learn below, the warning light will also be *on* when the switch is at AUTO *if* the air stream velocity in the engine intake duct is less than about 40 knots.

Ice Detector Assembly.

The ice detector assembly is mounted in the upper part of the engine air intake duct. The assembly consists of a probe and a main housing or case, with a pressure diaphragm inside the case. The chamber on one side of the diaphragm is connected to openings on the upstream side of the probe, while the chamber on the other side senses pressure from openings on the downstream side of the probe. When the airplane is moving, the upstream pressure will exceed downstream pressure. The diaphragm controls the ice detector switch. When the airstream is moving at less than 40 knots, the pressure differential across the diaphragm is not enough to move the switch to the "no-ice" position. If the ANTI-ICE switch is in AUTO at this time, the ANTI-ICE warning lights will illuminate. This does not necessarily indicate a malfunction. The warning lights should go out as soon as the airstream exceeds 40 knots.

When the airstream exceeds 40 knots, the pressure will force the switch to the "no-ice" position. However, if ice blocks the probe intakes, the pressure difference across the diaphragm will drop and the switch should move to the "no-ice" position. If the probe intakes should become clogged and will not open after 17 to 20 seconds, the interpreter will turn off the probe heater and the detector will be inoperative until the probe openings are no longer blocked. The heater in the detector case keeps the case warm to prevent condensation of moisture. The thermoswitch in the heater line interrupts the current to the heater when the temperature reaches a certain point. This heater will obviously cycle off and on continuously.

The detector assembly is a rather sensitive item, and it is better not to attempt maintenance on it without complete information. If the probe becomes clogged or otherwise becomes defective, replace the assembly with a new unit. If you must repair it, be sure to consult the technical order covering this component.

Maintenance Of The Engine Intake Anti-Ice Warning System.

From reading the preceding description of the engine anti-ice and anti-ice warning system, you should understand that a thorough comprehension of how the warning system operates cannot be had without first understanding the electrical wiring diagram of the anti-ice system itself. The maintenance requirements for the warning system will be somewhat more difficult than the requirements for the other engine warning systems. When the pilot reports that the anti-ice warning system is malfunctioning, however, troubleshoot the circuit with the same technique that you use on other circuits. Always start with the most logical trouble sources and end with the least likely. After assuring that the warning light bulbs, circuit breakers, and switch are functioning satisfactorily, check the electrical circuit with a voltmeter. Keep the electrical power ON. This type of trouble shooting will isolate the component that is malfunctioning. Any inoperative component must be replaced with a serviceable item.

SUMMARY.

This Training Supplement has presented you with a general review of jet engine principles and described the J57 turbojet engine as it is installed in the F-102A. After reviewing the development of jet engines in Chapter I, you acquired a general overall knowledge of the J57 engine and its associate systems. In Chapter II you learned how the fuel is metered to the engine and controlled in accordance with atmospheric and altitude conditions. Detailed descriptions and maintenance suggestions were also given for the oil, ignition and starting, in-flight and ground cooling, and the engine anti-ice systems. Chapter III described engine preservation and depreservation techniques and also covered the engine installation and removal operations. The last chapter explained how the engine instruments and warning systems function and outlined some of the common maintenance requirements.

Two points have been mentioned several times in the four chapters in this supplement, but they are important enough to be mentioned once more. These points concern the relation of F-102A Maintenance Technical Orders to this Training Supplement. In some cases, the description of the engine and its associate systems in this supplement contain specific values and pressures. These values and pressures are used for explanatory purposes only, and they should not be used on the flight line. The latest information of this nature can be found in the *Maintenance Technical Orders.*

In other instances, you have learned that all of the engines do not have the same type of components. Although specific engine model numbers have been given in some examples, this information has been omitted in others. This information, too, should always be obtained from the Technical Orders.

Keep in mind that this Power Plant Installation Training Supplement has not replaced the Technical Orders. It has only acquainted you with the J57 engine and outlined some of the maintenance needs. This knowledge has prepared you to use the Maintenance Technical Orders more intelligently so that you will be a more valuable member of the F-102A ground support team.

ALPHABETICAL INDEX

*The Symbol * Indicates An Illustration*

ALPHABETICAL INDEX (CONT)

*The Symbol * Indicates An Illustration*

ALPHABETICAL INDEX (CONT)

*The Symbol * Indicates An Illustration*

CONVAIR
F-102A

MAINTENANCE TRAINING
SUPPLEMENT

HIGH-PRESSURE
PNEUMATICS SYSTEM

F-102A

TRAINING SUPPLEMENT

(PERIMENTAL)

To Be Used in Conjunction With

T.O. 1F-102A-2-3

T.O. 1F-102A-2-12

HIGH-PRESSURE PNEUMATIC SYSTEM

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE

AIR FORCE—APPS—SAN ANTONIO, TEXAS—5-57-2600

1 JULY 1956

Foreword

Jaffith

The F-102A Training Supplements have been prepared by Convair, A Division of General Dynamics Corporation, to provide you, the system mechanic or maintenance technician, with basic information concerning the F-102A airplane and its operating systems. There are ten of these supplements, each covering an airplane operating system or major component. The text is direct and informal and is supplemented with illustrations and diagrams to aid you in understanding how the systems operate. The following is a list of the Training Supplements on the F-102A airplane:

Title

Flight Control System
Hydraulic System
High-Pressure Pneumatic System
Low-Pressure Pneumatic System
Airplane General
Airframe Fuel System
Power Plant Installation
Airframe Armament System
Electrical System
Instruments

Each supplement describes an operating system or major component, how and why it operates, and maintenance problems that you may encounter. The entire major system is further simplified by describing each of its subsystems separately. Thus, when you understand how all of the subsystems operate, you will be familiar with the functions of the major system. This knowledge of the airplane systems will enable you to use other operational, service, and maintenance instructions.

Since the purpose of these publications is to familiarize and acquaint you with the airplane systems and components, specific values, measurements, and tolerances are not given. Any values that appear in these supplements are approximate only and are used to emphasize more clearly a certain operation or condition. You must refer to your 1F-102A-2-3 Technical Order and other pertinent handbooks for specific values when adjusting or checking a system and its components. These Technical Orders are revised periodically to include the latest maintenance procedures and data on the equipment installed in the F-102A.

The F-102A Maintenance Technical Orders consist of the following handbooks:

T.O. 1F-102A-2-1	**General Airplane**
T.O. 1F-102A-2-2	**Ground Handling, Servicing, and Airframe Group Maintenance**
T.O. 1F-102A-2-3	**Hydraulic and Pneumatic Power Systems**
T.O. 1F-102A-2-4	**Power Plant**
T.O. 1F-102A-2-5	**Fuel Supply System**
T.O. 1F-102A-2-6	**Air Conditioning, Pressurization, and Anti-Icing Systems**
T.O. 1F-102A-2-7	**Flight Control Systems**
T.O. 1F-102A-2-8	**Landing Gear**
T.O. 1F-102A-2-9	**Instruments**
T.O. 1F-102A-2-10	**Electrical Systems**
T.O. 1F-102A-2-11	**Radio-Communication and Navigation Systems**
T.O. 1F-102A-2-12	**Armament and Armament Electronics**
T.O. 1F-102A-2-13	**Wiring Data (F-102A)**
T.O. 1F-102A-2-13A	**Wiring Data (TF-102A)**

The Air Force is vitally interested in seeing that its equipment functions properly and is maintained as efficiently as possible. The Air Force, like a manufacturer of an automobile or commercial appliance, provides printed instructions for use with its equipment. The printed instructions you will use most frequenrly in maintaining the F-102A are the dash-2 Technical Orders listed above. They are available for your reference at the Maintenance Office on your base. You must refer to them frequently so that you will always have the latest maintenance infromation. The Training Supplements will help you to use these Technical Orders by answering many of the "hows" and "whys" that may puzzle you from time to time.

TABLE OF CONTENTS

Introduction

This supplement consists of three chapters which cover the F-102A High-Pressure Pneumatic System. Chapter 1 presents a general introduction to pneumatic principles, shows the advantages of using air-powered systems, and compares a basic pneumatic system with an aircraft system, such as the one in the F-102A. In Chapter 2 you learn about the high-pressure pneumatic power supply in the F-102A and how the power supply components function. Chapter 3 concludes this supplement with a description of all the operating subsystems that use high-pressure air.

Chapter I

FUNDAMENTALS AND APPLICATIONS OF THE PNEUMATIC PRINCIPLES

Contents *Page*

The basic concept of the F-102A high-pressure pneumatic system is to provide a suitable power system for the operation of the armament launching components of this airplane. This type of power system is also used advantageously for other equipment where fast action, reliability, light weight, and compactness are necessary requirements. To emphasize the importance of the selection of the high-pressure pneumatic system as a power source for operating the armament launching components, you have only to examine the overall concept of the F-102A.

Previously we had aircraft armament consisting of guns and bombs. The guns were aimed and fired by remote control. Bombs were contained in bomb bays, the doors of which had to be opened to drop the bomb load. These processes were performed remotely by electric motors and hydraulic actuating systems.

With the development of aircraft reaching and exceeding sonic speeds, these systems proved to be slow, cumbersome, and heavy. They were not satisfactory for sonic-type aircraft. At high speeds, the drag factor of an airplane becomes of tremendous importance. The airplane must maintain a "clean" configuration to assure high speeds and stable flight. In other words, the airplane must have as few disruptions of the air flow as possible.

Although some armament systems on high-speed aircraft were designed to have hydraulic and electrical power to open and close the bomb bay doors and to extend and retract the armament launchers, the hydraulic and electrical systems could not provide the instant action needed without adding great weight to the airplane.

The solution to this problem has been the use of a high-pressure pneumatic system. In this Training Supplement, you will learn about the high-pressure pneumatic system which powers the armament system on the F-102A interceptor.

USE OF HIGH-PRESSURE PNEUMATICS IN THE F-102A.

The high-pressure pneumatic system in the F-102A is a ground charged source of air pressure power that performs a number of functions. A clean, dry supply of compressed air is provided by a ground compressor unit through the airplane ground filler connection. The compressed air from this ground charging unit is stored at a pressure of 3000 psi in air flasks located in the armament bays, and in the hollow portions of the main landing gear drag braces.

Now let's see how fast this system operates when making a pass at an enemy target. This fast action of the F-102A is shown in figure 1-1. Count off three seconds —one, two, three—and, in about this amount of time, the high-pressure pneumatic system opens the armament bay doors, flips out the armament for firing, retracts and locks the armament launchers after firing, and closes the armament bay doors.

From the above description you will agree that if the high-pressure pneumatic system had only this duty to perform, it would be reason enough for having it on the F-102A. However, in addition to accomplishing these duties the system provides air pressure for the rudder feel cylinder, extends the ram air turbine into its operating position for emergency hydraulic power, provides air pressure for emergency extension of the

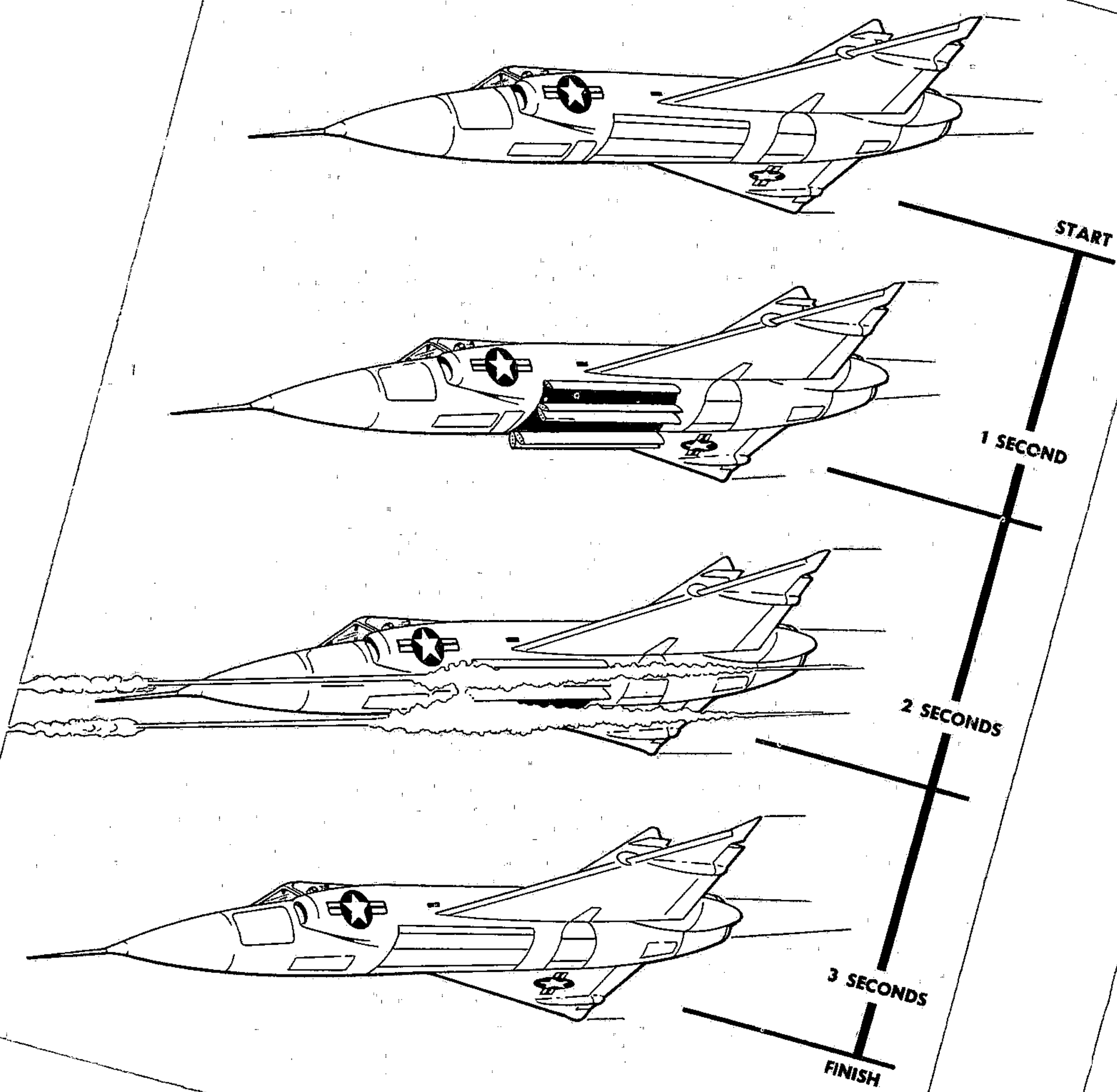


Figure 1-1. Fast Action

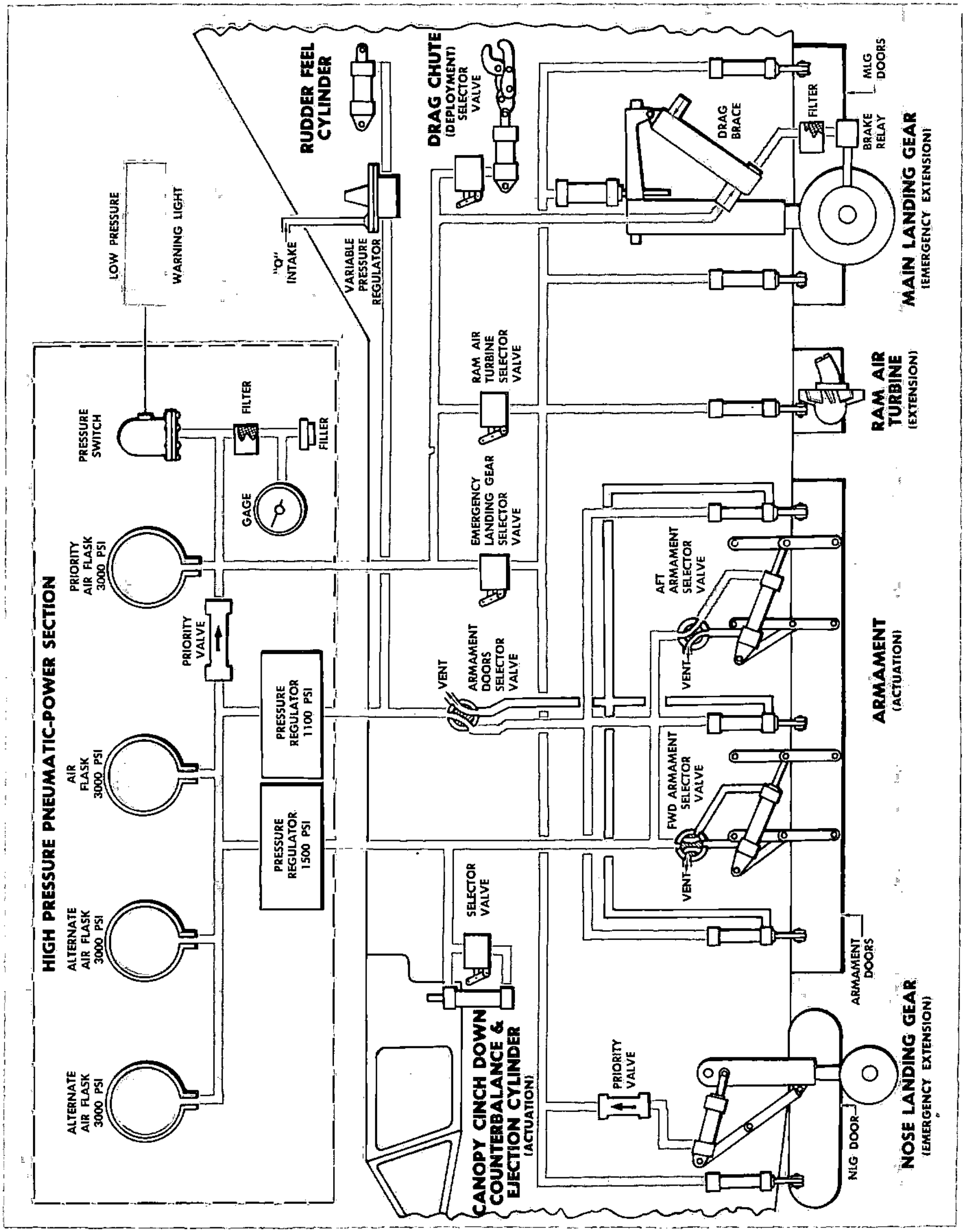


Figure 1-2. F-102A High-Pressure Pneumatic System

Figure 1-3. The Atmosphere

landing gear, deploys the drag chute on landing, and actuates the main wheel brakes. On later airplanes, this system also supplies air pressure to the canopy counterbalance, ejection, and cinchdown actuating cylinder.

Figure 1-2 illustrates the F-102A high-pressure system. Note that this system is divided into a power section (the dark portions) and the operating systems (the lighter portions). The system is charged from a ground air compressor through the filler valve in the power section, and air pressure is stored in the four air flasks shown in the power section and in the main landing gear drag braces. The priority valve, between the priority air flask and the other three flasks, maintains an emergency supply of air pressure for those systems—the landing gear, wheel brakes, ram air turbine, and drag chute—which are connected in the illustration by the dark lines. The lines connected to the 1100 and 1500 psi pressure regulators supply air pressure to the remaining systems—the armament system, canopy counterbalance cylinder, and the rudder feel cylinder. Chapter 2 of this supplement describes the power section in detail, and Chapter 3 covers the operating subsystems.

The responsibilities of this system are many and its value as a power system is great. These duties may prompt you to think that this system is very complicated. However, after you study the functions and operations of this system, you will discover that this impression is wrong.

It is true that the high-pressure pneumatic system is a relatively new system in aircraft, although compressed air is one of the oldest and most widely used sources of power. Your study of this system can be made simpler by comparing its functional characteristics to the many familiar usages of compressed air. But, before we get into the high-pressure system, let's review some of the principles of pneumatics and the use of air as a source of power.

PNEUMATIC PRINCIPLES.

Pneumatics is defined as the controlled use of air as a source of energy. In pneumatic systems, air is compressed and is used as a power source. Isn't this a simple definition for a system performing so many varied and responsible tasks? However, to understand the functional operations of a high-pressure pneumatic system better you should know something about air and the characteristics of compressed air.

The Atmosphere.

The atmosphere consists of a thick blanket of air surrounding the earth. Air is composed of tiny molecules of different gases—mostly nitrogen and oxygen. These molecules are very small. Actually, since the distance between the molecules is very great in comparison to their size, air is mostly empty space. These molecules are continually in motion, colliding with each other and with surrounding objects. It is the combined force of all these collisions that gives the effect of atmospheric pressure. Obviously, the fewer molecules of air in a given space, the fewer collisions there will be. The fewer collisions there are, the lower the pressure. Air pressure at sea level is relatively great—the weight of the air above presses on the air below, forcing more molecules into a given space. This increase of molecules causes an increase in the number of collisions and therefore an increase in pressure. At higher altitudes the molecules are farther apart; therefore the pressure is lower.

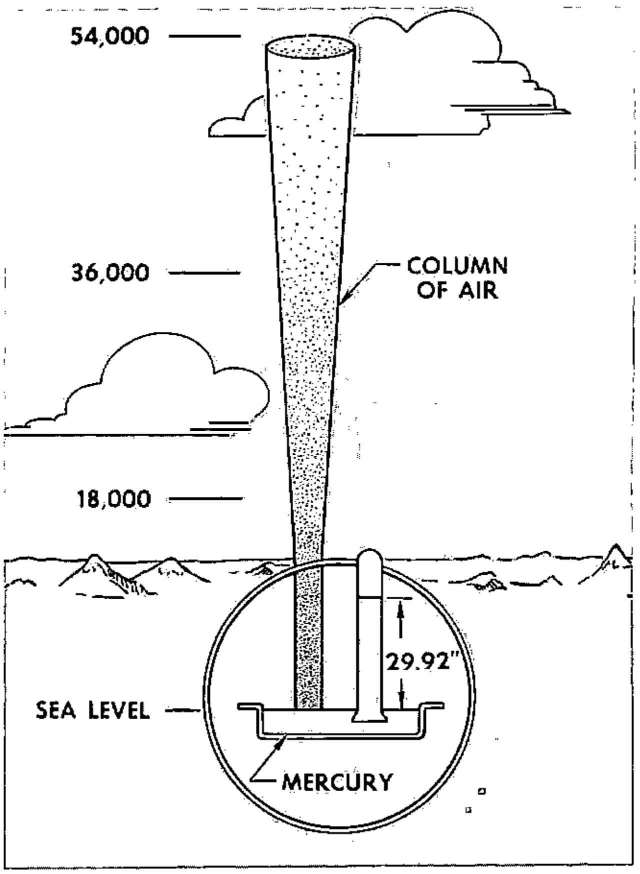


Figure 1-4. Atmospheric Pressure Measurement

You have probably seen a mercury barometer—it is a device that measures atmospheric pressure. As you know, when a unit of air is raised higher above the earth, the column of air above becomes shorter. Study the example in figure 1-4. The cubic foot of air, shown at sea level, is compressed by the weight of the column of air extending above it to the upper limit of the atmosphere. This atmospheric pressure supports the column of mercury, shown in the tube, at a level where the weight of the column of mercury is equal to the weight of the column of air—this is 29.92 inches Hg at sea level. A change in atmospheric pressure causes a corresponding change in the height of the column of mercury (Hg).

Effects of Compressing Air.

We have said that air consists of tiny molecules of various gases—mostly nitrogen and oxygen. Actually, under normal atmospheric pressures and temperatures, air is primarily empty space with molecules speeding around in random directions. The distance between the molecules is great in comparison to their size. When pressure is increased, the molecules are pressed closer together, but are still widely separated. Since these molecules are in continuous motion, they are constantly colliding. The collision of one molecule with another or with the sides of a container causes friction or heat. When air is compressed, the collision rate increases; therefore the temperature rises. The greater the compression, the higher the temperature.

You can see this relationship in figure 1-5. Notice the piston before compression. The molecules are far apart and the temperature bulb and the pressure gage have low readings. When the piston compresses, the molecules move together and the temperature and the pressure readings increase. This relationship works in the reverse direction, too. If the pressure drops, the temperature will drop. This *heat of compression* may result in a very great temperature rise, the exact temperature being determined by many factors. For instance, if air is compressed to one-tenth of its original volume, the temperature may increase from an original 15°C (59°F) to 343°C (650°F). The heat of compression in the engine of the F-102A causes the temperature of this compressed air to rise as high as 427°C (800°F).

Effects of Heating Air.

The speed of the molecules of compressed air is directly related to the temperature of the air. The higher the temperature, the faster the molecules move and the greater the pressure.

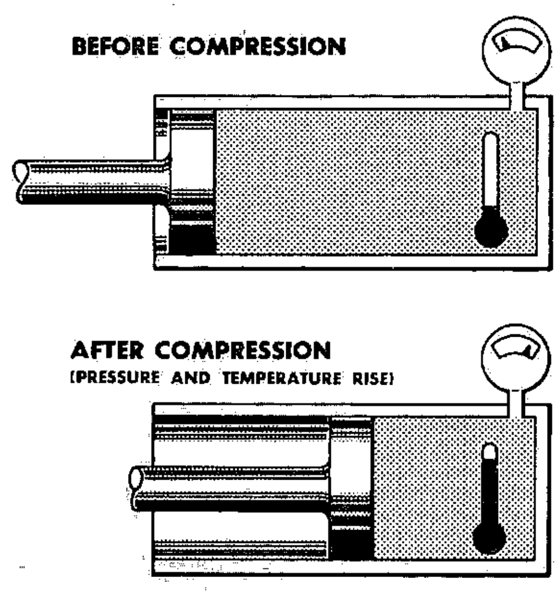


Figure 1-5. Effect of Pressure on Air

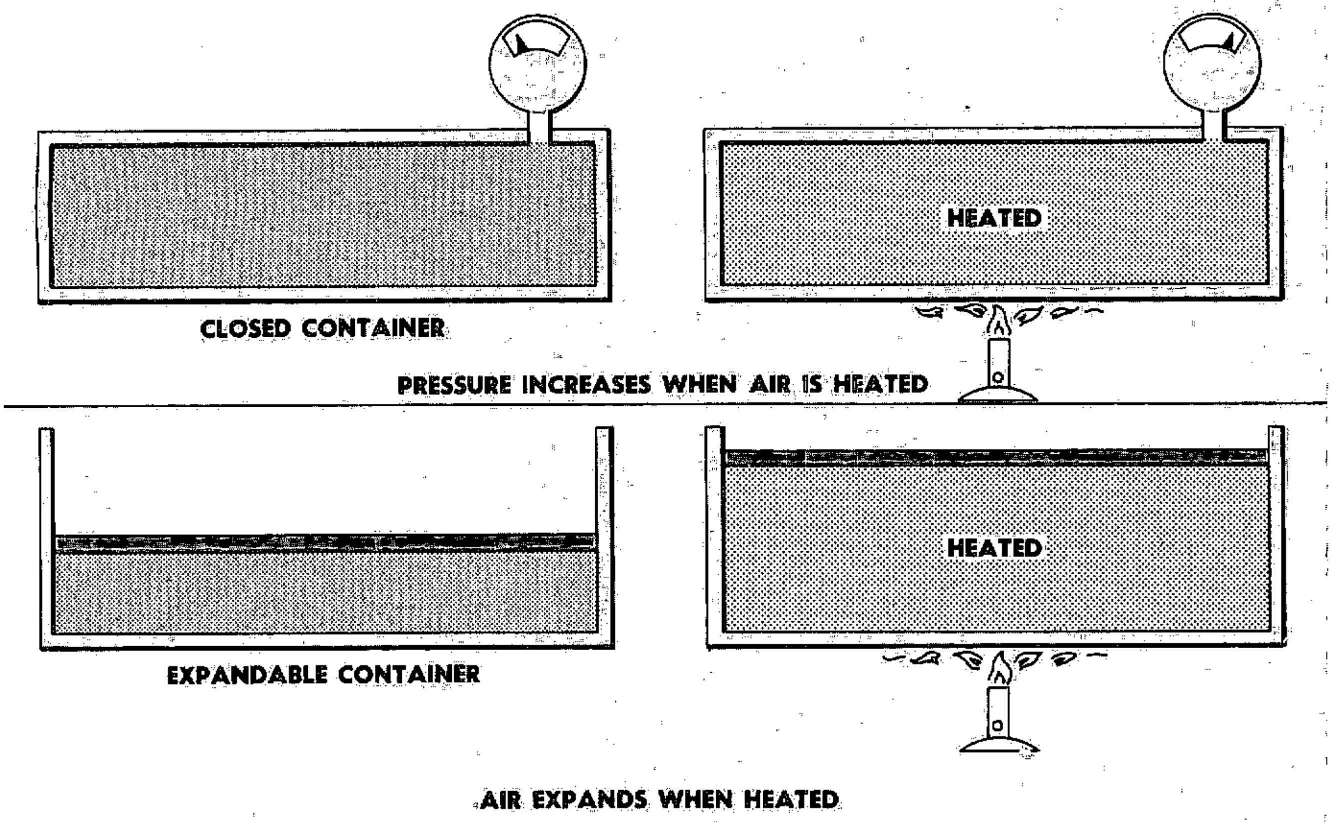


Figure 1-6. Effects of Heating Air

In figure 1-6 you can see that when free air is heated in an open, expandable container, the molecules will travel faster and the space between the molecules increases. In other words, the air will expand. Since there are fewer molecules within a definite space, there will be fewer collisions. The greater force of the collisions will balance the reduction in the number of collisions; therefore, the pressure will not change. However, if air is heated in a closed container the air cannot expand and the greater force of each collision will result in an increase in pressure. This relationship is also reversible. If the temperature of air in a closed container drops, the pressure will drop.

Compressed Air as a Source of Power.

If a moving piston compresses air—as in the action of the piston illustrated in figure 1-5 or in the action of the ground compressor unit used to charge the F-102A high-pressure pneumatic system — the compression squeezes a given volume of molecules closer together. With the volume made smaller and the distance the molecules can travel made less, heat is generated by the increased molecular collisions with the surrounding surfaces. This action increases pressure within the piston chamber. This air pressure is then transferred into a tank for storage. In the F-102A this air pressure is stored in air flasks where its controlled use operates system actuating devices.

You have learned that air pressure is increased by increasing temperature. We also know that the work required to compress air increases the internal energy of the air, as well as its temperature. Thus, the pressure of air is increased by reducing its volume and/or increasing its temperature.

When charging the F-102A high-pressure pneumatic system air flasks, you should have an awareness of these relationships. Temperature increases as the pressure in the system is increased—the volume remaining the same. The air pressure stabilizes when the air and pneumatic system components cool to the ambient (existing outside) temperature. Therefore, it is necessary for the pressure in the system to stabilize. This allows the air, system tubing, and system air storage flasks to cool—and the presure will decrease accordingly.

Common Uses of Compressed Air.

You are probably familiar with the everyday applications of the use of compressed air that are shown in figure 1-7. These units include the grease lift that raises your automobile and the grease gun that lubricates it, the aircraft rivet gun and drill motor, the air hammer for street repairs, the inflation of automobile tires, and the paint spray gun. It would be hard to find a reasonably modern industrial plant that does not use a number of pneumatic units daily. Even on the farm,

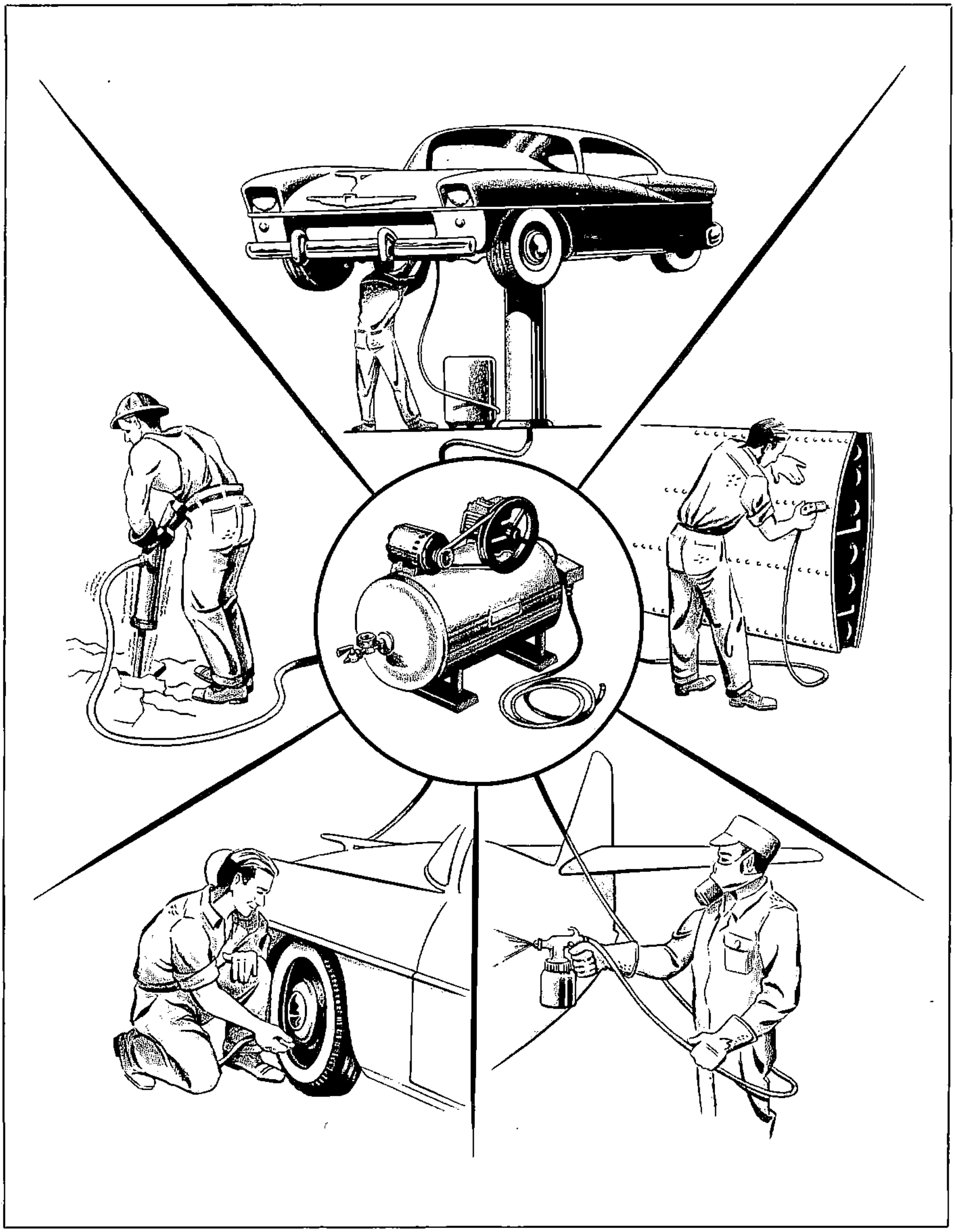


Figure 1-7. Common Usages of Compressed Air

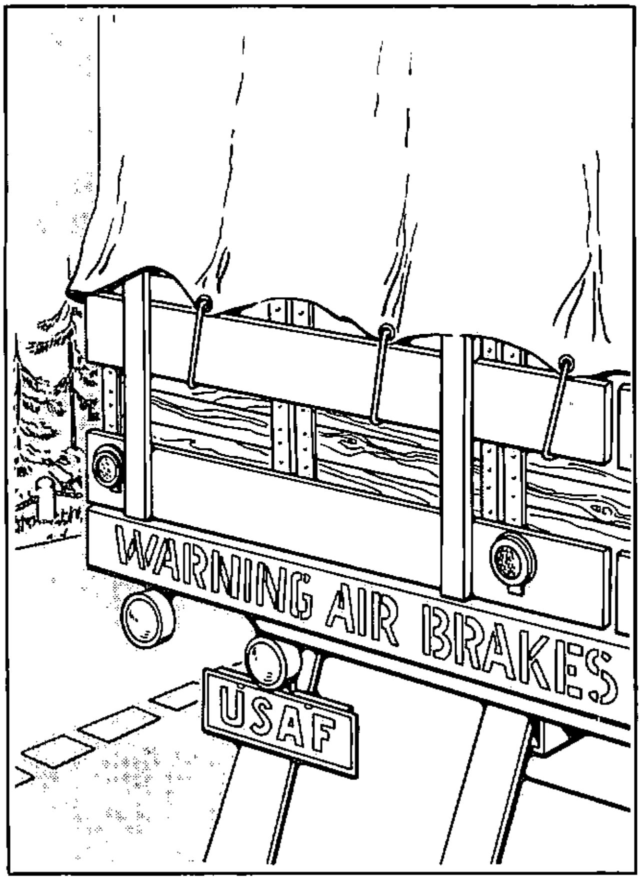


Figure 1-8. Compressed Air for Brake Operation

compressed air is used for such simple jobs as silo filling, agricultural spraying, and milking machines. Another example of an efficient application of pneumatics is the braking system of buses and trucks. Similarly, the wheel brakes on the F-102A are also operated by air to produce effective braking. The brake system of the F-102A is discussed in Chapter 3.

There is also a steadily growing usage of compressed air combined with hydraulic control units. In this type of equipment, the compressed air is used as a power source and the hydraulic units are used to control the equipment. Heavy-duty trucks and equipment in factories use such an arrangement, where air pressure and hydraulics are combined to create motive power. This dwell-retract, or governed action, type of equipment operates more economically and cycles more efficiently at high operating pressures than would be possible were air or hydraulic power sources used individually. In the F-102A you will find that the wheel brakes are hydraulically operated and pneumatically actuated.

A BASIC PNEUMATIC SYSTEM.

In the schematic of a basic pneumatic system (figure 1-9), note that the motor-driven compressor draws air in through a filter. This filter cleans the air before it enters the compressor cylinder. From the compressor, air is routed to the storage tank for use in the system. This storage tank offers a constant and steady supply of air upon demand, and eliminates the pulsating pressure that would occur if air were used directly from the compressor. The pressure gage on the storage tank indicates the air pressure in the tank, and the pressure switch on the tank automatically cuts off power to the compressor when the tank pressure reaches a predetermined level.

The check valve at the tank outlet permits free flow of air from the tanks, but prevents any reverse flow. The pressure regulator and relief valve downstream from the check valve regulates the pressure going to the actuator and relieves any excess pressure. In the system shown, the relief valve setting would probably be about the same as the setting of the pressure switch on the storage tank.

Note that the air leaving the pressure regulator enters the selector valve where it is directed to one side or the other of the pneumatic actuator cylinder, or it is stopped completely. In the system, this selector valve is shown in the CLOSE position so that air pressure will retract the actuator cylinder and close the scissor-type flaps. In the right corner you can see the two other positions in which the valve can be placed. By using these two alternate positions, check how air would flow, or be directed to the actuator. In each of the three positions, note how *exhaust* is connected to the sides of the actuator.

AIRCRAFT PNEUMATIC SYSTEMS.

Two distinct pneumatic power sources are used on the F-102A: a high-pressure system that receives its pressure from ground-charged air flasks, and a low-pressure system that is supplied by the bleed air from the engine compressor. There is very little similarity between the two systems. When you compare the power sources, you see that these two systems are entirely independent of each other.

In fact, the only reason for mentioning the low-pressure pneumatic system here is to emphasize the independence of the two systems; just because both are called pneumatic systems is no reason that you should expect them to be interdependent. In the high-pressure pneumatic system, mechanical work is performed, but air used by the low-pressure pneumatic system is expanded and temperature-conditioned for use in the utility systems. Its pressure is seldom greater than 16 psi, but in the high-pressure system, the pressure used is as high as 3000 psi in the flasks and 1500 psi in the lines. The low-pressure pneumatic system is discussed in the Low-Pressure Pneumatic System Supplement of this training series.

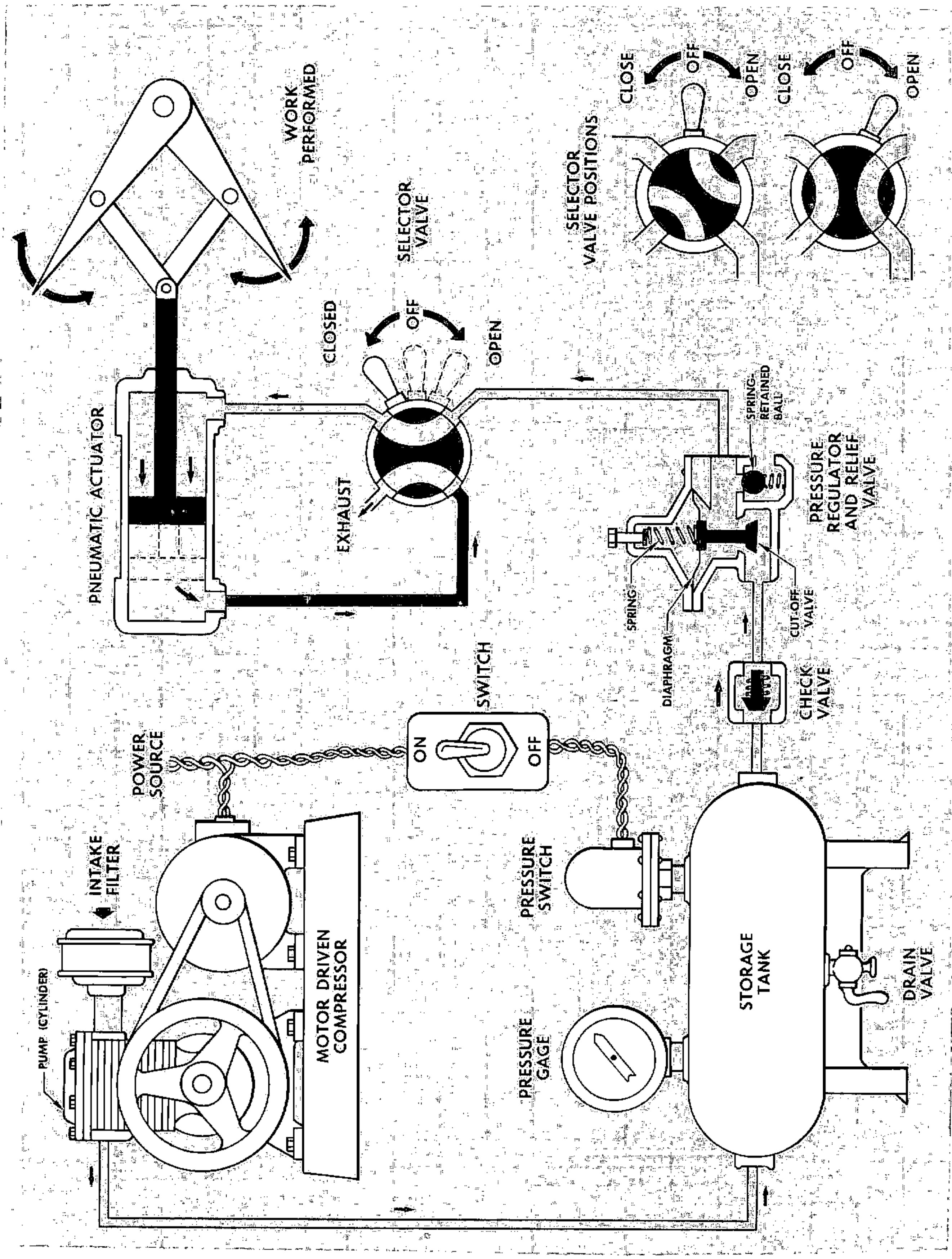


Figure 1-9. Basic Pneumatic System

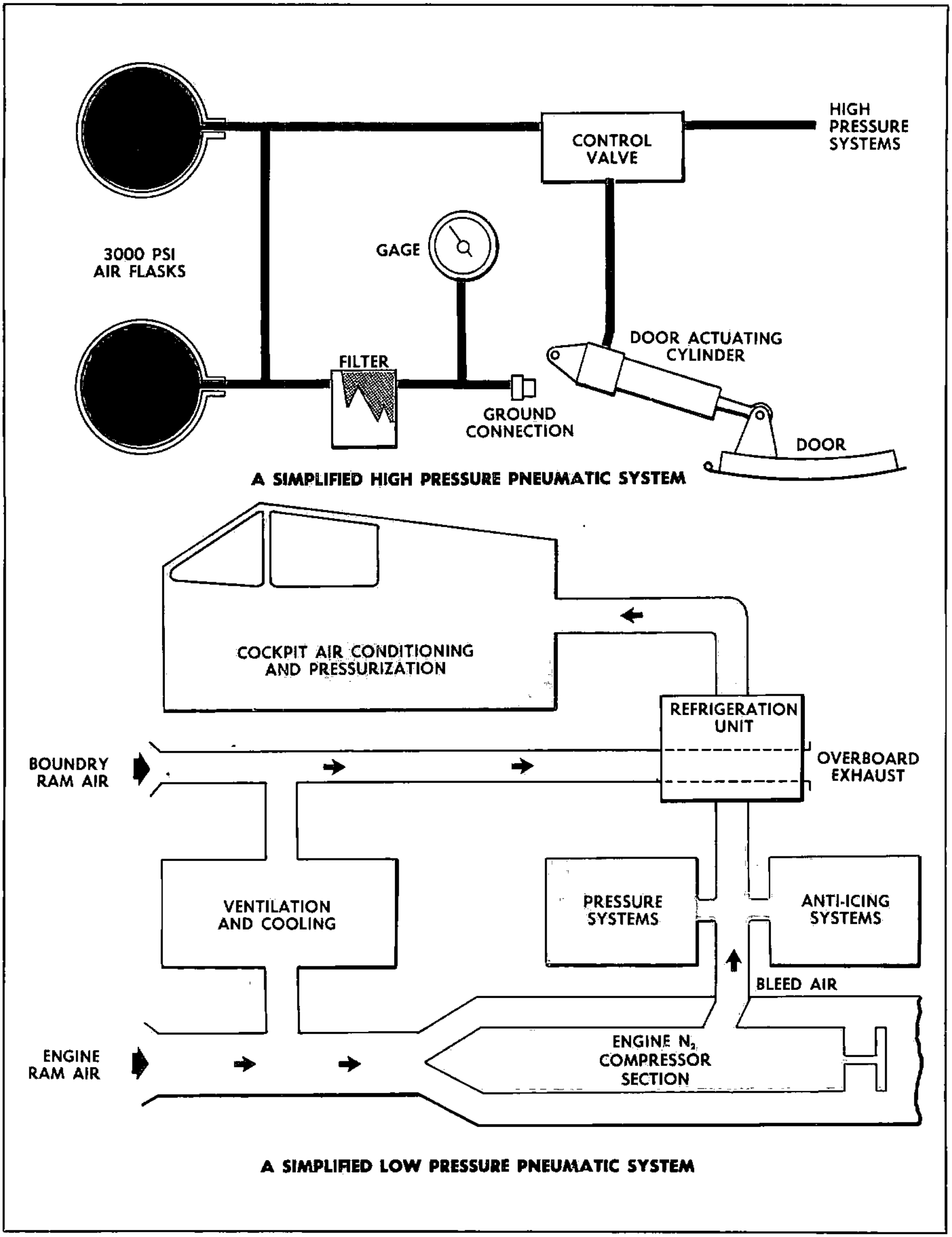


Figure 1-10. High Pressure Versus Low Pressure

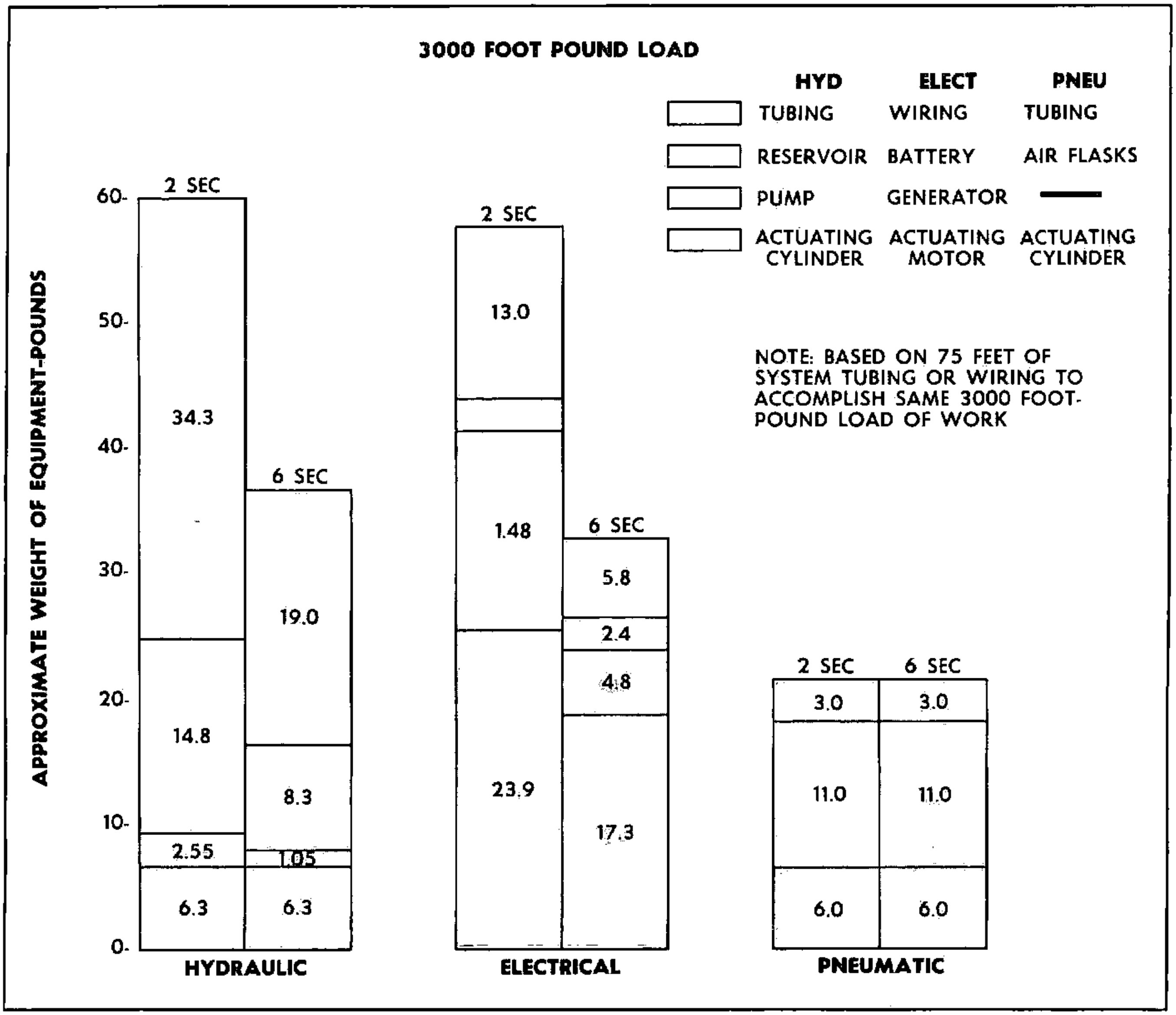


Figure 1-11. Power Systems Compared

WHY HIGH-PRESSURE PNEUMATIC SYSTEMS ARE USED.

The work that must be expended to actuate movable units on aircraft often exceeds the physical capabilities of the pilot. At the same time, the increase in size and speed of aircraft during recent years has all but ruled out mechanical transmission systems. To provide power amplification and to eliminate mechanical links, many aircraft use a multitude of remotely controlled actuating devices such as aircraft motors and actuators to perform this work. Until recently, all accessories were powered from either electrical or hydraulic power sources. It is obvious then that in providing these power sources, a strain or load is placed on the aircraft power plant in the form of more and larger engine accessories, additional engine power takeoff gear box pads, and the inevitable increase in airframe weight. We now see the necessity of aircraft actuating devices. We also see that by using the conventional power systems (electrical and hydraulic), a price has to be paid by the draining of power from the power plant. By the use of a self-supplied high-pressure pneumatic power system installation, the direct drain of power from the power plant is eliminated and the airframe weight is greatly reduced. Other features of this type of system are fast action, reliability, and the compactness of installations.

BASIC DIFFERENCES.

The most distinctive differences between high-pressure pneumatics as used in this airplane and those used by industry are the power supply sources and the working pressures. Most industrial equipment is operated by working pressures of 60 to 100 psi which are provided by air compressors delivering air at pressures of 80 to 125 psi. In the F-102A high-pressure pneumatic system, air flasks store the initial supply of air under a pressure of 3000 psi. The initial supply pressure is reduced and regulated to various working pressures as desired for the subsystem operating components. Most of the subsystems operate on pressures of 1100 and 1500 psi. Some compressed air is stored in accumulators located in the hollow portion of the main landing gear drag braces. All these units are charged from ground equipment before the flight.

Naturally, there is a marked difference in the size, weight, and efficiency of the pneumatic units and components used for industrial purposes, compared to those used in the F-102A high-pressure system. For industrial applications, the component weight and size is not an essential consideration. Industrial cylinders and valves, for example, are normally large and heavy, tolerances are wider, non-working surfaces are left with rough finishes, and the choice of materials is seldom critical. Through research and testing, the components used on the F-102A high-pressure pneumatic system have been made light and small, yet ample to provide all the power needed, and these components are only a fraction of the size and weight of their industrial counterparts.

ADVANTAGES OVER ELECTRICAL AND HYDRAULIC SYSTEMS.

You will recall that this high-pressure pneumatic system is used because it is fast-acting, reliable, light, and compact. To enable you to realize the importance of these factors better, figure 1-11 illustrates the outstanding features of the high-pressure pneumatic system compared to those of aircraft hydraulic and electrical systems. The chart is based on the weight of equipment required for each system to perform the same amount of work in the same amount of time.

In the bar on the right-hand side of the illustration, showing the amount of equipment required to perform an operation in 2 and 6 seconds, note that the weight of the required equipment is exactly the same in both instances. In other words, it requires no heavier equipment in a pneumatic system to do a job in 2 seconds than it would if the job were to be done in 6 seconds. In both cases, the equipment required weighs about 20 pounds.

In the bar graph for the hydraulic system, an entirely different situation is shown. Note that it takes only about 34½ pounds of hydraulic equipment to do the job in 6 seconds, but when the time element is reduced to 2 seconds, the weight of hydraulic equipment required increases to almost 60 pounds. Note that the same situation holds true for the electrical system. To do the job in 6 seconds, 30 pounds of electrical equipment is needed, but to decrease the time element to 2 seconds, about 30 additional pounds of electrical equipment is needed.

This pneumatic system has several other weight-saving and operating speed advantages that you must consider as you study this chart. Let's take examples of where the main weight factors of the hydraulic and electrical systems are located, and then compare these weight factors to those of the high-pressure pneumatic system for this airplane. Most of the weight of a hydraulic system is in the pressure and return tubing for the operating equipment, in the hydraulic fluid, and in the hydraulic pumps. Most of the weight of an electrical system (for the short periods of operation to accomplish the work specified on the chart) is in the motors, electrical conductors, and other units that must be provided. For the high-pressure pneumatic system, you have single line tubing to the actuating cylinders. No return lines are needed because the exhaust air can be dumped overboard.

You may be questioning at this time, "Since this system has so many advantages over the hydraulic and electrical power sources, why isn't it used for other actuating equipment?"

One of the main reasons that the high-pressure pneumatic system is not used for many other system installations is that the power source for this system is limited by the number and type of installations using the system. Remember, there is no return for the air pressure as a working force and the supply cannot be replenished in flight. Incidentally, two other features of this system not previously mentioned but of primary importance for consideration are: There is no fire hazard involved for this system, and it is less vulnerable to enemy fire—since return tubing or bulky installations are not required.

Chapter II

HIGH-PRESSURE PNEUMATIC POWER SUPPLY SYSTEM

Contents *Page*

Before we begin a detailed study of the high-pressure pneumatic system let's take a "bird's eye view" of its basic functions. These functions can best be described as belonging in three categories: There must be a power source; there must be a means of regulating or controlling the air pressure; and there must be actuating equipment to perform the mechanical work of the various subsystems.

Because of weight and space limitations, a self-contained air compressor is not used as a power source for this airplane. Instead, the pneumatic power source is provided by air pressure stored in air flasks. Compressed air is also stored in the hollow portion of the main landing gear drag braces. This drag brace air is reserved for the operation of the main wheel brakes. The air storage containers are charged with a ground compressor unit.

A means of regulating and controlling the air pressure power source is provided to start, direct, reduce, and stop the air pressure in its path to perform the assigned work. This is accomplished in the F-102A by such pneumatic equipment as priority, selector, check, shuttle, relief, and shutoff valves; in addition, pressure switches and pressure regulators are provided to assist in this task. Finally, actuating cylinders are provided for receiving the regulated air pressure and transforming the stored energy into the mechanical energy required for the operation of the various units.

SAFETY PRECAUTIONS.

In this chapter, we will discuss only the power supply system. However, before we begin the study of the power supply system—what it is, how it is obtained and stored, and how it is delivered to perform the work of the various subsystems—you are reminded of certain safety precautions that must be observed when operating equipment or performing system maintenance. Mainly, you are reminded of a very simple fact—just respect the potent power and fast action of this system. Personal contact with escaping air at high pressure is extremely dangerous. Remember, too, a flick of the pneumatic controls means snap-action results. And finally, all pressure must be relieved from the system before removing or disconnecting any component.

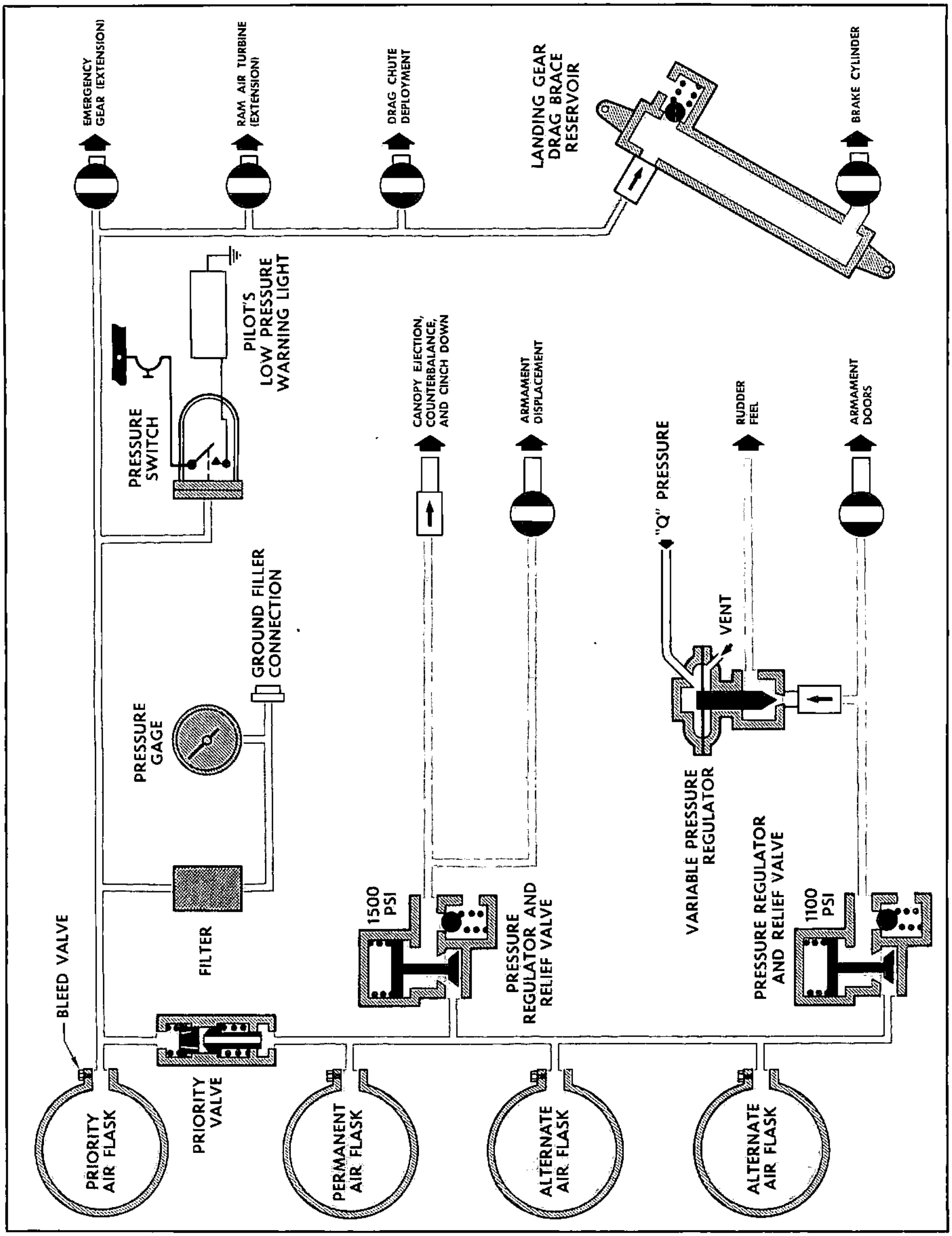


Figure 2-1. High-Pressure Pneumatic Power Supply System Schematic (Sheet 1 of 2)

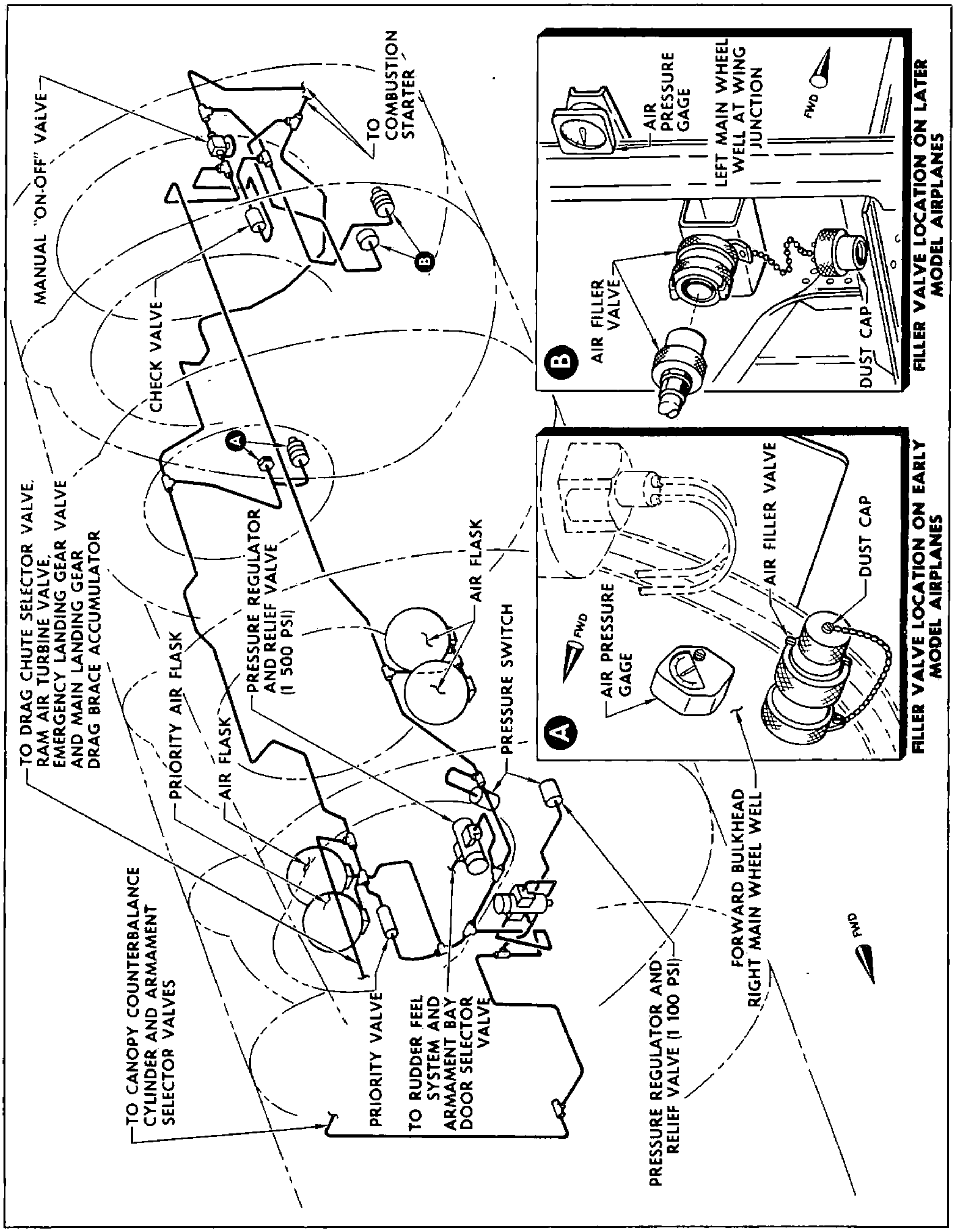


Figure 2-1. High-Pressure Pneumatic Power Supply System Schematic (Sheet 2 of 2)

The pneumatic power source is provided by air pressure stored in air flask installations and in the hollow portion of the main landing gear drag braces. These air storage reservoirs are charged with clean, dry air with a ground compressor unit to a stabilized pressure of 3000 psi. The system air supply must always be clean and dry. Air contamination is the avowed enemy of any high-pressure pneumatic system. Should there be moisture in the air supply, condensation and formation of ice particles will develop at low temperatures. The ice particles can cause severe damage, even system failure, by bombarding the equipment and tubing. Only a well maintained ground compressor unit can prevent such action in this system. A standpipe moisture trap installed in each air storage flask is a second precaution against moisture in the air supply. Thus, it is very important that you always check all moisture traps in both the ground compressor unit and the pneumatic power system so that moisture will not be introduced into the system.

Not only must the air pressure supply for this system be dry, but it must also be clean to prevent faulty equipment operation. To clean the air of foreign particles, filters are provided in both the ground compressor unit and the pneumatic power system. And, just as you must often check the moisture drain traps to prevent faulty equipment operation, so should you often check all the filter elements for cleanliness.

HIGH-PRESSURE PNEUMATIC POWER SYSTEM.

In the schematic illustration (figure 2-1), note the chief components of the high-pressure pneumatic system and the relationship of these units to each other. All components of the power system, except the filler connection and pressure gage, are in the armament bays; the filler connection and system gage are on the forward bulkhead in the main wheel well.

In the schematic in the lower right portion of the illustration, note that two of the four air flasks are alternate flasks as shown in the locational view. These alternate flasks are the rear flasks in each outboard armament bay and can be removed if not required on a flight. The two forward flasks in the outboard bays are permanent installations and must be in the airplane for flight. The permanent flask in the right armament bay is also a priority flask. It maintains a reserve supply of air pressure for emergency operations when the pressure in the three remaining flasks drops below the setting of the priority valve.

HOW THE POWER SYSTEM OPERATES.

The high-pressure pneumatic system is charged to 3000 psi with clean, dry air through the ground filler connection shown on the schematic. The system gage, adjacent to the filler connection, registers the air pressure in the system. Whenever you charge the system, allow the pressure to stabilize before making the final check on the system pressure. This stabilization period allows the air to cool to the surrounding air temperature. Usually this results in a drop in system pressure. The system is then charged further to bring it up to its full charge of 3000 psi.

As you can see in the schematic, air entering the system passes through a filter which further purifies the air. From the filter, the air is routed to the four storage flasks and to accumulators in each landing gear drag brace. The air pressure in the drag brace accumulators is used for the wheel brake system. On early airplanes each main gear has two drag brace accumulators while later model airplanes have only one drag brace accumulator on each main gear. The check valve at each main gear drag brace prevents air pressure from returning to the high-pressure system. However, this valve permits the flasks to further charge the drag brace accumulators, if the flask pressure is greater. The wheel brake system and the drag brace accumulators will be discussed later.

Note that a priority valve is placed between the two permanent flasks. When the pressure in the system falls to approximately 1200 psi, the priority valve closes and prevents pressure in the priority flask from balancing the pressure in the three remaining flasks. The priority flask is reserved for the emergency operation of the subsystems shown on the extreme right of the schematic. A pressure switch closes when this priority system pressure drop to about 1500 psi. The closing of the switch contacts completes a circuit which turns the pilot low-pressure warning light on (see figure 2-1).

The three storage flasks—and the priority flask when pressure is about 1200 psi—supply pressurized air through the pressure regulators and check valves to the systems shown in the center of the illustration. The regulators control the pressure going to the operating subsystems. Selector and control valves direct the air pressure in the subsystems. You will notice a variable pressure regulator in the center of the schematic. This regulator valve uses ram air entering the "Q" intake head on the fin leading edge to control the high-pressure air entering the rudder feel system.

Only the operation of the high-pressure pneumatic power system (shown on the schematic in the darker lines) has been given here. You will find the operation of the subsystems that are listed on this schematic in the next chapter.

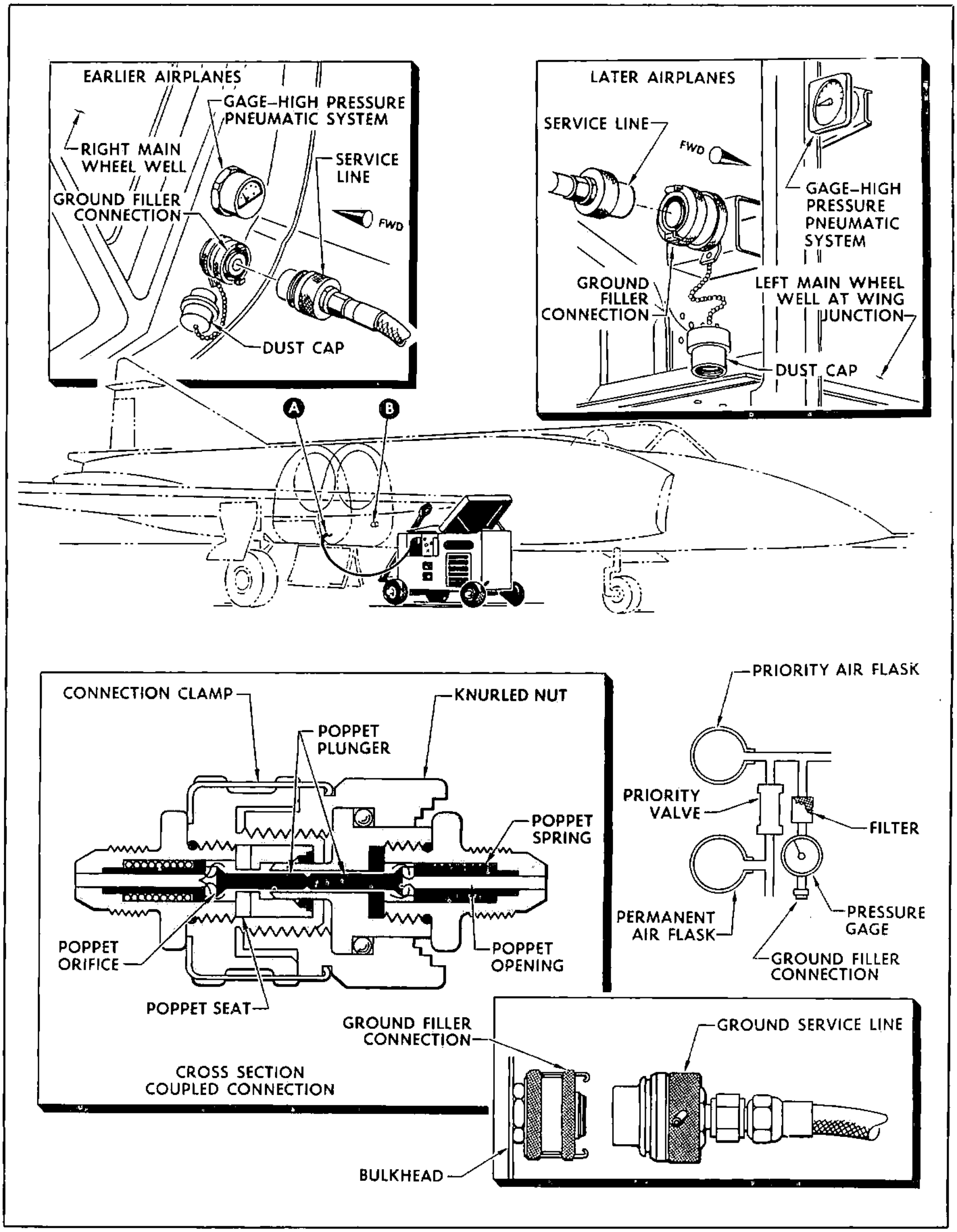


Figure 2-2. Ground Filler Connection

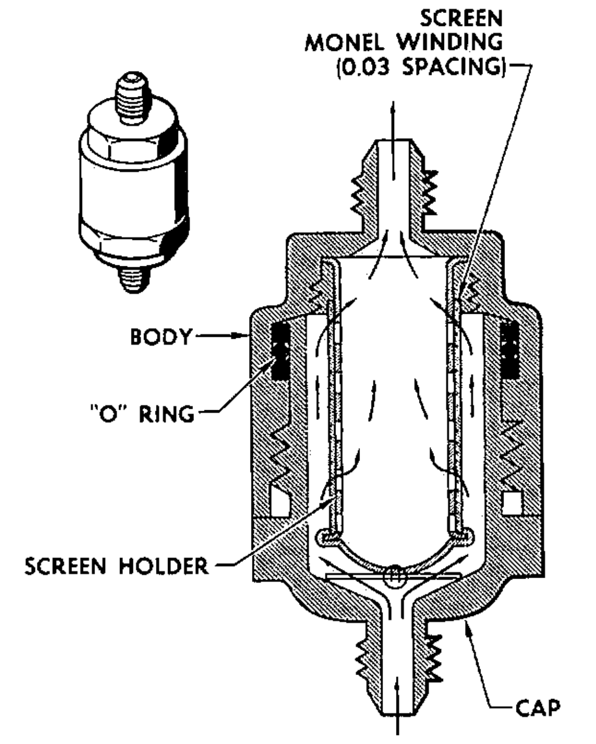


Figure 2-3. Air Filter

COMPONENTS OF THE POWER SYSTEM.

So far, you have learned what the components in this supply system do, but not how they were constructed or how they do their job. So let's start at the beginning of the system—where it is charged—and learn something about the operation of the supply system components.

Ground Filler Connection.

A self-sealing double-check type ground filler connection is provided to charge the pneumatic power supply system. See figure 2-2. This unit is composed of two separate plug-threaded type parts. The fixed half, or receptacle for the pneumatic power system, is located on the forward bulkhead in the right main wheel well, as shown in details A and B. The other half of the connection is attached to the end of the ground air compressor unit service line.

As shown in details A and B, each half of the coupling is provided with a dust cap. It is important both halves of this connection be capped when the ground compressor unit is disconnected. Capping of the connections prevents dust, moisture, or other foreign material from entering the system as it is charged. By the above description you can see that this connection provides a means of automatically sealing both the system air storage supply and the ground service line pressure when disconnected. However, there always remains a small amount of trapped air between the two halves of the connection.

When disconnecting the service hose, loosen the hose knurl nut one turn to allow the trapped air pressure to escape. Also notice that connection clamps are provided to prevent the ground filler connection from disconnecting during servicing. These clamps automatically lock and unlock when the ground service line knurl nut is turned for connecting or disconnecting. Another feature of this type coupling is that you can quickly and safely make the connection against 3000 psi air pressure by using only normal finger torque.

Now let's trace the flow of air pressure into the system when the ground air pressure hose is connected. First let's see how the air pressure is blocked when the coupling is disconnected. When connecting the ground unit, a check in each half is forced away from its seat and against the force of the spring; this provides a passageway for the air between the service line and the airplane system filler lines. When disconnected, the spring in each half forces the check against the seal, and this seals off the air pressure for both the airplane system filler line and for the ground unit service line. This air seal in each half of the coupling is completed before the coupling is completely disconnected.

Air Filter.

The air supply in the pneumatic system must be clean for proper system operation. A filter is installed in the system filler line between the filler connection and the air storage flasks, as shown in figure 2-1. A cutaway schematic of this air filter is shown in figure 2-3. It is a simple unit consisting of a two-piece O-ring seal case and monel wire wound filtering element. The spacing between the monel windings is 0.003 inch so that it acts as a screen for air entering the bottom. This air passes through the monel screen from the outside to the center, and then out to the port at the top.

This filter can be disassembled for cleaning. Since any foreign material will be trapped on the outside of the screen, you must use care in cleaning this element and not blow high-pressure air into the element. High-pressure air will rupture the element. Also be sure to blow in it from the downstream port so that the dirt or foreign material will be easily dislodged—blowing in the filter from the upstream port will wedge the foreign particles more tightly in the filter.

Air Storage Flasks.

In figure 2-4, you can see the air flasks as they are located in each of the two aft outboard armament bays. The two forward flasks must always be in the

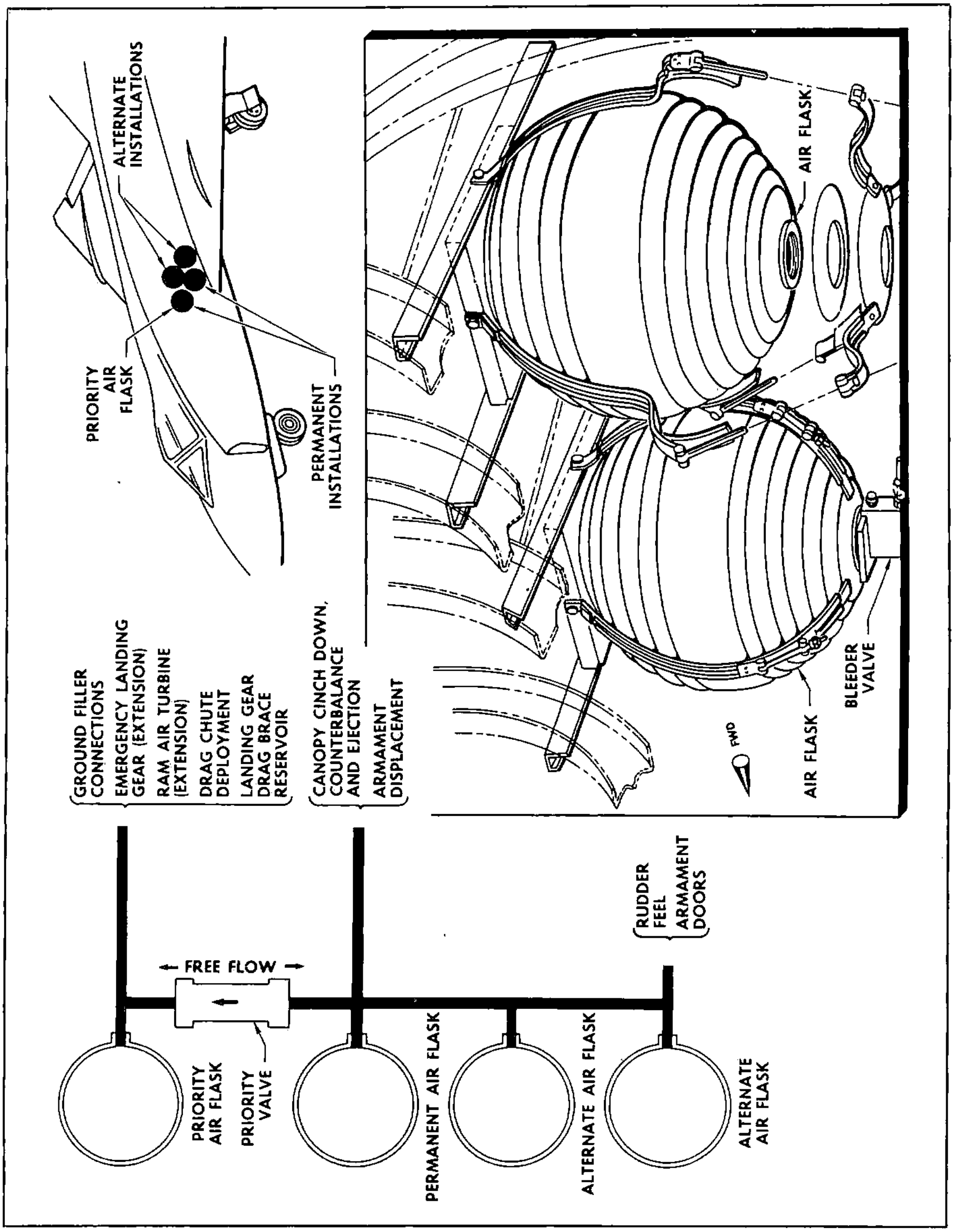


Figure 2-4. Air Flask Installations

aircraft for pneumatic system operations. The two aft flasks are alternate installations and may or may not be installed on a flight depending on the type of mission.

The priority (right forward) flask is directly connected to the manually operated valves which control the following emergency systems (see figures 2-1 and 2-4): the ram air turbine extension, emergency landing gear extension, drag chute deployment, landing gear drag brace storage accumulators, and the main landing gear brake cylinder. When the pneumatic power supply system pressure drops to 1200 psi, the priority valve automatically isolates the remaining air pressure in the priority air flask and prevents it from entering the other three flasks. This pressure in the priority flask is then reserved to operate the above mentioned emergency subsystems. The priority valve and its operation are discussed later in this section.

In figure 2-5, note the detailed installation of two of these air flasks and a cross-sectional view of a typical one. Note that the aft flask is the optional installation; and if it is removed, all that is necessary is to cap the connecting system lines. The flasks are installed by four straps with tension bolts, and the inlet-outlet line connects to the bleeder valve fitting. These flasks are of shatterproof construction with an outer surface, or pressure enclosure, composed of laminated-wound fibre glass which is impregnated with a resin. An inner rubber liner contains and stabilizes the compressed air pressure supply. Each air flask weighs 16 pounds and has an air storage volume of 870 cubic inches. This flask is 40 per cent lighter than the conventional metal air flask capable of storing the same volume of air at the same pressure.

Another important feature offered by this type of air flask is that it is not affected structurally by a wide variation of temperatures—from —65°F (—53.9°C) to 200°F (93.3°C). This safety and reliability feature of the air flasks is required in providing high-pressure air storage for operating the emergency systems under all conditions. Suppose an emergency should arise in flight whereby, after armament firing, the armament bay doors would fail to close. Imagine the extreme low temperature imposed on the air flasks, since they are located in the rear armament bays.

And then for an extreme opposite temperature condition, you have only to recall that heat is generated when the air flasks are charged with high-pressure air. By these two examples, you can see the importance of the features offered by this type of air flask, for the material used in its manufacture has a very low index of expansion; thus it is not likely to crack.

Note that an internal standpipe feeds air pressure to the system and also receives the flask air charge. In this way, any moisture content is prevented from entering the supply system by settling around the base of the standpipe. An air bleed valve is provided at the base of the standpipe to bleed air pressure from the system and to drain moisture condensation from the flasks. This feature is so simple that you are apt to overlook its importance. But again, remember that contamination in the air supply can cause severe damage to the pneumatic equipment.

Power System Priority Valve.

The power system priority valve is a poppet-type valve, located in the pneumatic pressure supply line between the priority flask and left forward air flask. See figure 2-6. When we discussed the air flask installation you learned that when the system air pressure drops to 1200 psi, the supply in the priority flask is reserved or isolated for operation of the emergency subsystems. This action is accomplished by the priority valve which automatically closes and prevents the priority valve from balancing the pressure supply in the other flasks. The priority valve will also permit higher pressures in the other flasks to bleed into the priority supply until a 1200-psi level occurs in the other flasks.

As you will note in the cutaway view of the illustration, pressure from the priority flask or ground charging valve enters at the top of the priority valve, while balancing pressure from the remaining three flasks enters at the bottom. Air pressure in excess of 1200 psi enters the top of the valve and unseats the poppet by pushing it down against its lower seat. Air pressure in excess of 1200 psi then flows past the poppet (note its triangular shape in the detail) and unseats the piston assembly, thus allowing air to flow into the other three flasks (which can now act as priority valves). The poppet and piston remain unseated (open) as long as pressure is in excess of 1200 psi.

When pressure in the system drops to 1200 psi due to the use of air by the operating subsystems, the piston assembly seats against the poppet and closes off pressure from the priority flask. The spring behind the piston assembly is preadjusted to close the piston at 1200 psi. In addition to the priority flask pressure, the poppet spring also holds the poppet against the piston assembly. If pressure in the non-priority flasks should exceed the priority flask pressure, the poppet will be unseated to allow a balancing of pressure between the other three flasks and the priority flask. This assures that the emergency operating systems shall always have pressure as long as any remains in the power supply system.

Never attempt an adjustment or repair of this unit while the valve is installed in the airplane. Should the valve perform improperly, replace it with a new or a serviceable unit. The pressure exerted by the piston spring is the only adjustment provided in this valve. This adjustment is preset by the factory or overhaul activity.

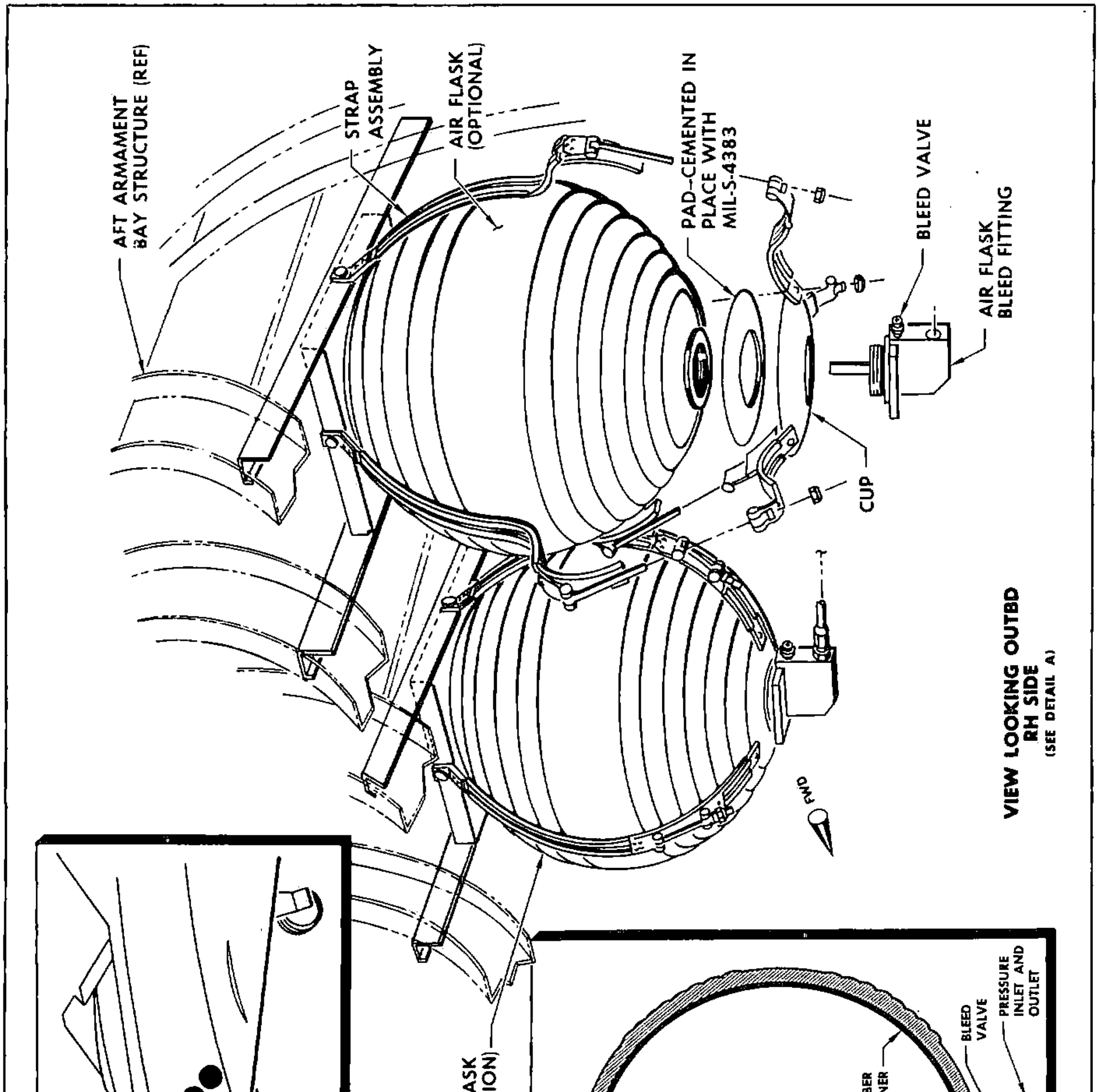


Figure 2-5. Typical Air Flasks

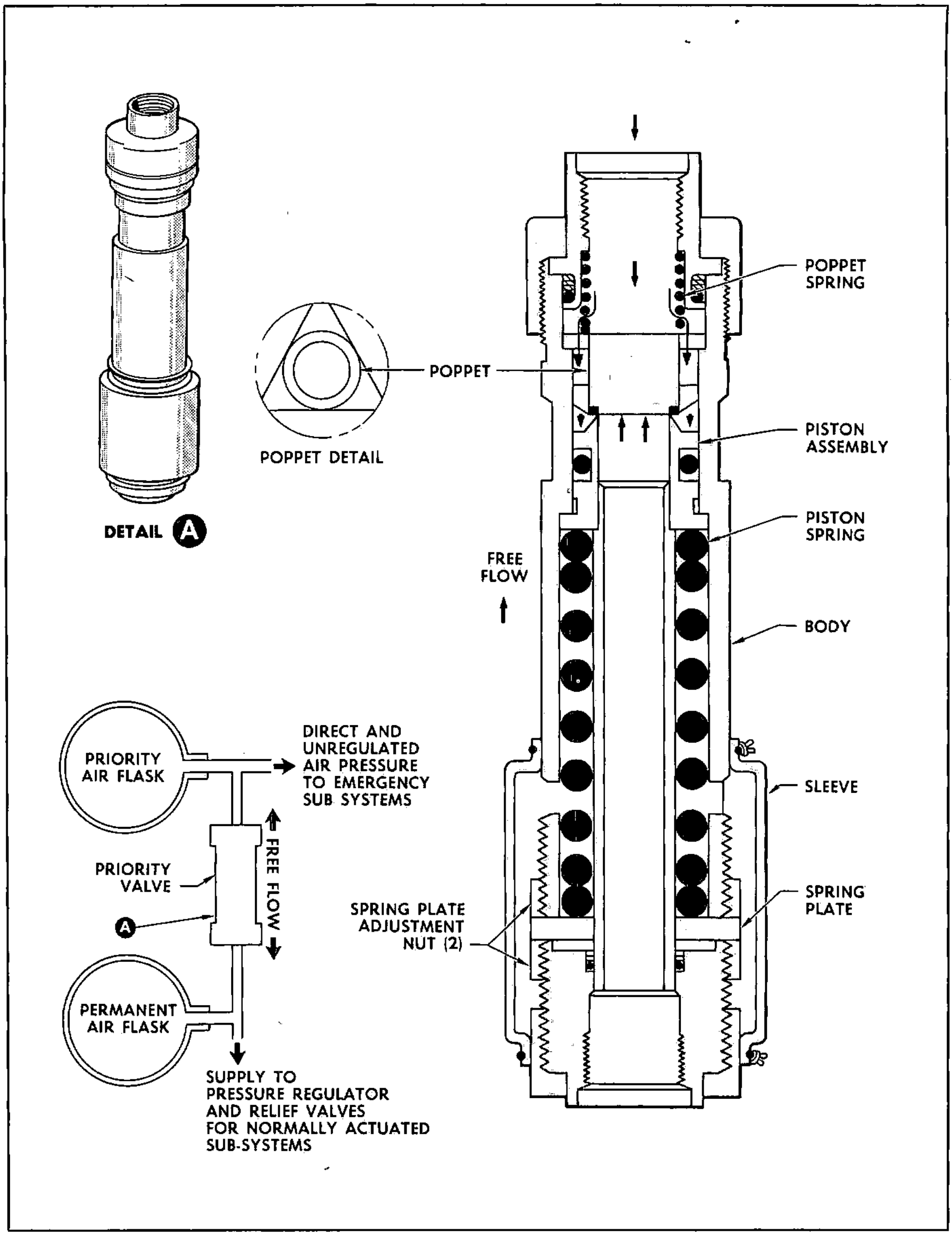


Figure 2-6. Power System Priority Valve

Pressure Regulator and Relief Valves.

As you know, the emergency subsystems are connected directly to the priority air flask. This means that these subsystems are provided with an unregulated supply of air pressure ranging from a fully serviced system pressure of 3000 psi to the lowered pressure of whatever the priority air flask pressure may provide. The remainder of the high-pressure pneumatic subsystems must be provided with a regulated flow of air pressure.

Two pressure regulator and relief valves are installed in the high-pressure pneumatic system to provide the regulated air pressure supply. See figure 2-7. One regulator reduces the pressure to 1100 psi for use by the armament bay doors and the rudder feel subsystems; the other reduces the air pressure to 1500 psi for use by the armament displacement and the canopy ejection and cinch-down subsystems. These two regulators are identical in construction but have different preset pressure adjustments. Therefore, they come tagged with the information as to what pressures they are adjusted to.

This combination pressure regulator and relief valve is of the poppet-type construction, consisting of three valves encased in two housings. Note that the pilot regulator is located in the small housing mounted externally on the unit. The main housing contains a slave regulator valve in the lower section and a relief valve in the upper section.

The pilot regulator section consists of a housing, a spring-loaded regulator valve piston with a flow head, and a cap end with an adjusting screw. Incidentally, here again you must not make any adjustments since it is preset on the bench and cannot be adjusted on the line. The function of the pilot regulator is to regulate the air pressure to the slave regulator.

HOW THE PILOT VALVE FUNCTIONS. In following the operation of the pilot regulator on the cutaway illustration, note that air from the air storage flasks enters the main housing and flows around the regulator valve piston and into the pilot regulator chamber between the sleeve and the outer wall. The flow head, attached to the bottom of the piston, seats on a ring at the bottom of the pilot regulator valve spring. When the pressure builds up on the piston in the inlet passage, it overcomes the piston spring tension and pushes the piston up to close the flow head port. The flow head port will remain open and in an intermediate position that regulates the air pressure to the slave regulator according to the flow demand of the actuating subsystem.

HOW THE REGULATOR VALVE OPERATES. Note that the slave regulator is located in the lower main housing. It consists of a hollow cylindrical slave piston that slides in an upper and lower sleeve and an air valve seat attached to the piston. The piston is retained in the lower sleeve by a snap ring and on the top by the air valve seat. The function of the slave piston is to regulate the air pressure to the air outlet port regardless of the volume required by the system.

When there are no subsystem flow requirements and the valve outlet pressure is approximately at specified subsystem pressure, the slave piston will be stabilized, causing the air valve to be seated on the face of the upper sleeve. When a demand for air flow is made by the actuating subsystem, a pressure drop occurs at the valve outlet chamber. Immediately, the inlet pressure acting inside the slave regulator piston raises the piston to open the air valve so that inlet pressure can flow past the air valve seat to provide the air pressure demanded by the subsystem. At the same time, the pilot regulator reacts to the pressure drop in the slave outlet passage, and the flow head port opens to continue the required air pressure.

RELIEF VALVE OPERATION. The upper main housing of the assembly contains the relief valve. This valve is incorporated in the assembly as a safety precaution should the regulator malfunction and tend to overpressurize units in the subsystems. The relief valve consists basically of a spring-loaded relief valve and seat, a spring-loaded pilot valve, and two vent ports. The relief valve is retained in the regulator valve housing by its seat at the lower end and by the return spring and boss at the bottom of the head which you see threaded into the valve.

The pilot valve is retained within the relief valve piston by the boss and inserted orifice fitting in the regulator valve head or seat. The pilot valve and its stem are hollow and slide in a sleeve at the top of the relief valve. The sleeve has a drilled orifice which connects to the chamber formed by the inside wall of the housing cap and the upper head end of the relief valve. The sleeve retains the pilot valve spring in the relief valve and is adjusted on the test bench to control the pilot valve spring tension; thus you won't make any adjustment on it.

The relief valve functions when the air pressure in the regulator valve outlet port chamber exceeds the preset tension of the pilot valve spring and forces the pilot valve open. This excess air pressure unseats the pilot valve and flows through the center of the pilot valve stem, through the drilled orifice in the spacer and into the chamber between the head end of the relief valve and the housing cap.

This air pressure, acting on the relief valve head, overcomes the return spring tension and forces the relief valve to open, thus venting the excess pressure in the regulator outlet chamber to atmosphere. The pressure also flows through a restricted orifice shown

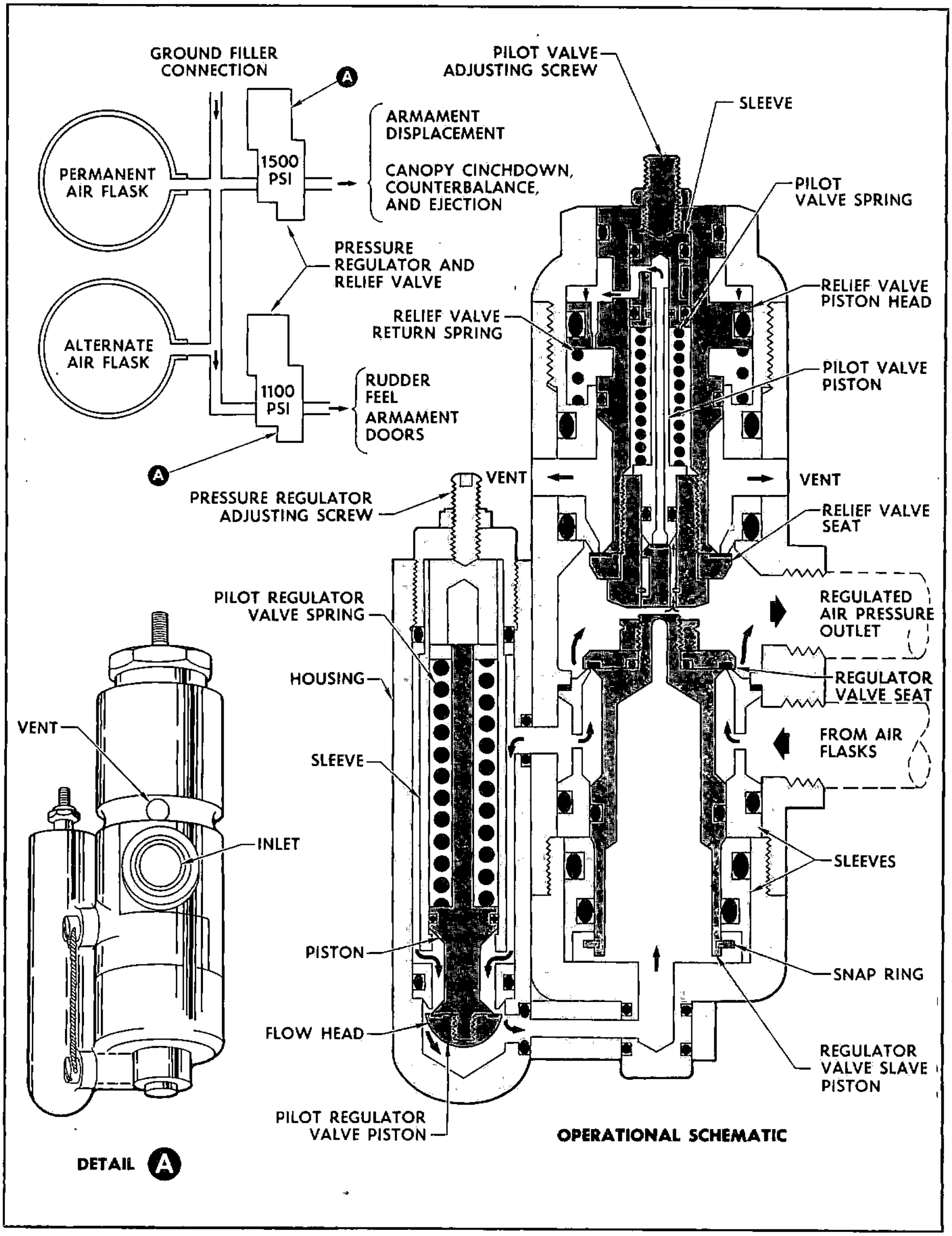


Figure 2-7. Pressure Regulator and Relief Valve

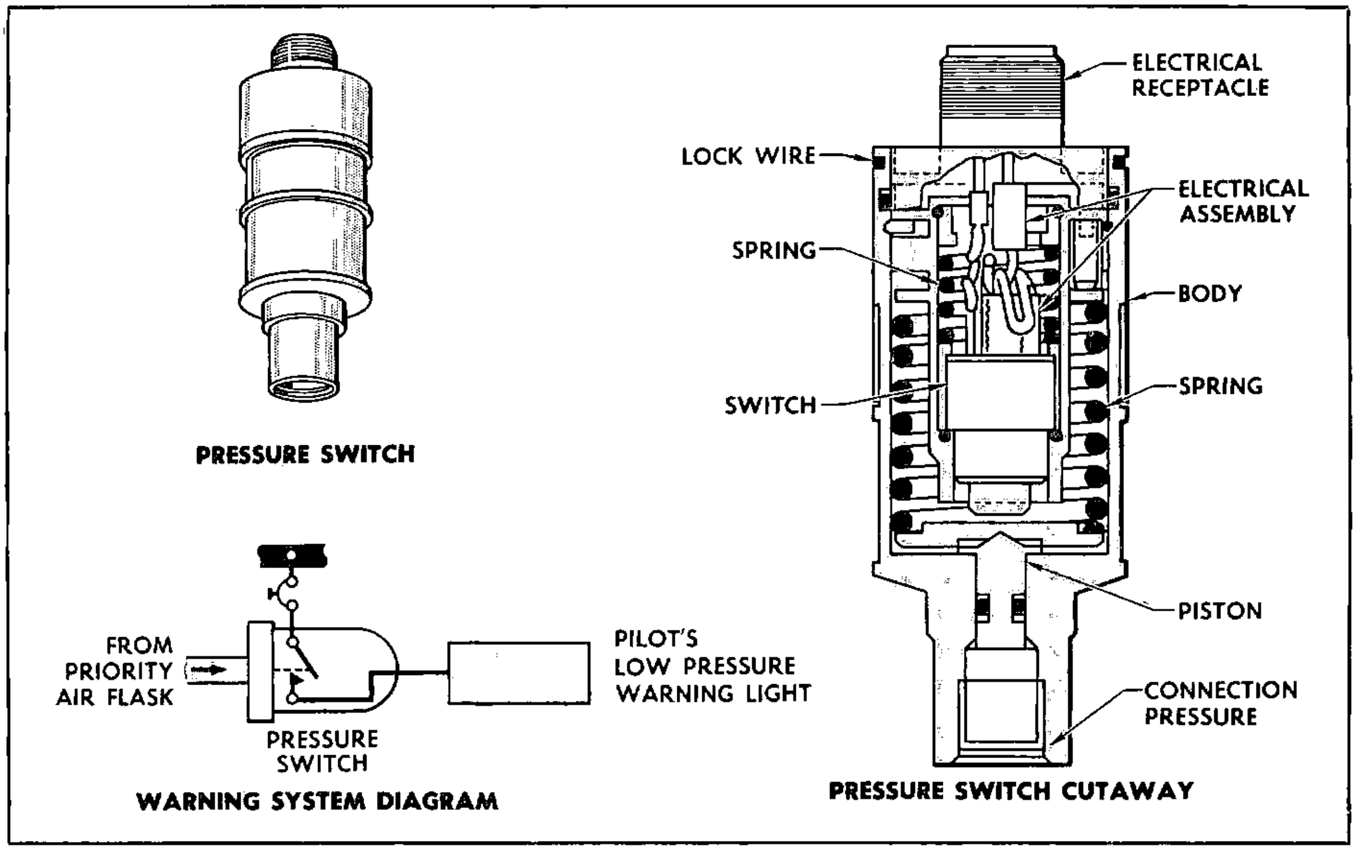


Figure 2-8. Pressure-Low Warning Switch

on the left side of the relief valve head and into the return spring chamber. When the pressure in the outlet port chamber is lowered to the designated pressure of the subsystem, the pilot valve spring closes the pilot valve, and the relief valve return spring closes the relief valve to maintain the subsystem designated pressure.

As we have discussed and followed the functions of the pressure regulator and relief valve on the illustration, you can readily see that this unit is the heart of its operating subsystems. It is, therefore, very important to see that a "heart attack" does not develop—air contamination by moisture or dust can cause this fatal heart attack. You are again reminded not to attempt an adjustment or repair on this unit; this must be accomplished on a test bench. Should you have trouble with the unit, replace it.

PNEUMATIC PRESSURE-LOW WARNING SYSTEM.

You have learned in this description of the pneumatic supply that it is ground-charged to 3000 psi before flight. Once this system pressure is depleted in flight, there is no way to replenish it. For this reason, a warning system is provided to caution the pilot if and when the pressure in the supply system becomes dangerously low. This warning system terminates at the warning indicator panel on the right side of the cockpit instrument panel. Its light is placarded PNEU PRESS; it is the sixth light down on the panel.

In addition to the warning light, the pneumatic pressure-low warning system consists of a pressure switch, figure 2-8, and the connecting circuitry. In the schematic illustration, figure 2-9, note how this switch is connected into the warning circuit.

The pneumatic pressure-low warning light illuminates when the system pressure becomes insufficient for a combat cycle. A combat cycle consists of opening the armament bay doors, extending and retracting the armament launchers, and closing the doors. This cycle requires a minimum pressure of about 1500 psi. Below that point, the pressure switch closes and permits current to flow from the power source at the circuit breaker through the master warning box to the warning lights where it is grounded.

The pressure switch in this warning system will open when the system pressure again builds up to approximately 1700 psi. Therefore, the warning light will remain on—while power is available at the 28-volt dc essential bus—until the air flasks are refilled. If the pilot reports that the light goes on and off in flight, you should suspect malfunctioning of the pressure switch or the line leading to it.

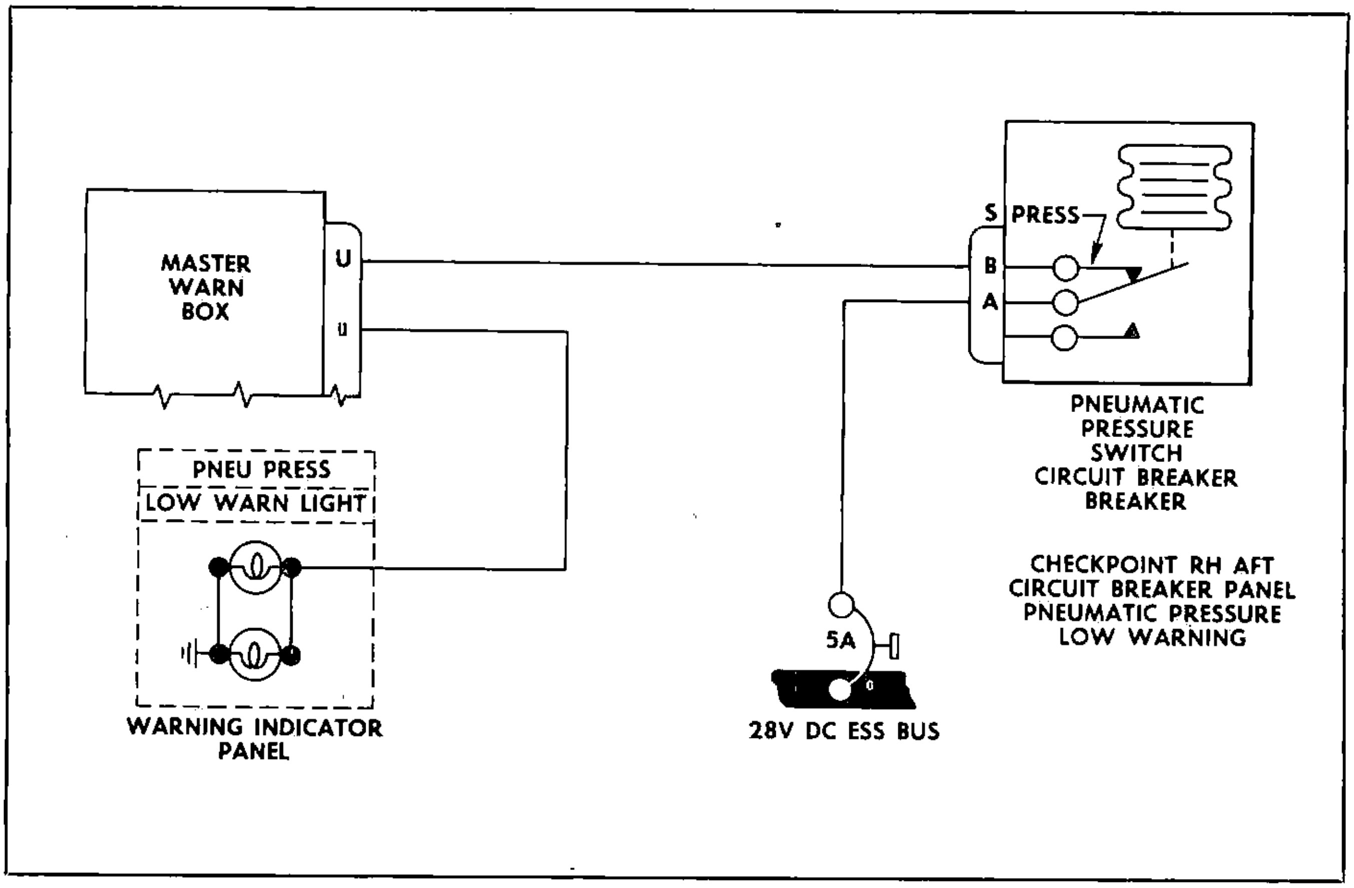


Figure 2-9. Pressure-Low Warning System Schematic

SERVICING AND TESTING THE PNEUMATIC SYSTEM.

As shown in figure 2-10, the high-pressure pneumatic system is charged from a ground air compressor unit through a ground filler connection. On earlier airplanes, the ground filler connection is located in the right main wheel well, and on later airplanes in the left main wheel well. The system is initially charged to a pressure of 3200 psi. The ground compressor unit is then turned off for some time to let the pneumatic system stabilize. About 30 minutes is required for the temperature of the compressed air in the system to cool and stabilize. The system is again charged if necessary to bring the stabilized pressure up to 3000 (±50) psi.

When a stabilized pressure of 3000 psi is obtained, the system is ready for flight, or if you are testing the system for leaks you are now ready to start the leak test. However, before we see how the leak tests are made, let's take a look at the ground air compressor unit.

GROUND AIR COMPRESSOR UNIT.

By now you should know some of the important demands of the ground air compressor unit. For instance, you should know that the compressor operation must be unaffected by extreme climatic conditions and that the unit must be capable of charging the high-pressure pneumatic system with a supply of clean, dry air at a pressure of 3000 psi. Unless this supply is both clean and dry, serious system equipment troubles will result.

The ground air compressor unit, shown connected to the airplane on figure 2-10, is designed for continuous duty operation under normal temperature and relative humidity conditions. It is a four-stage machine having a rated capacity of 15 cubic feet per minute (cfm) at 3500 psig (absolute pressure). A heavy-duty, four-cycle, air-cooled engine drives this compressor through a V-belt. The entire assembly is mounted on a four-wheel trailer, which is equipped with hand operated brakes on the rear wheels to hold the equipment steady during operation.

The temperature of the high-pressure air is reduced after it leaves the fourth stage of the ground compressor by passing the compressed air through an *aftercooler* and into a *receiver*. This compressed air is then piped from the receiver through two dehydrators which insure dry air for the airplane high-pressure pneumatic system. These dehydrators are called *dessicant chambers*, and are shown in the detail of the compressor control panel.

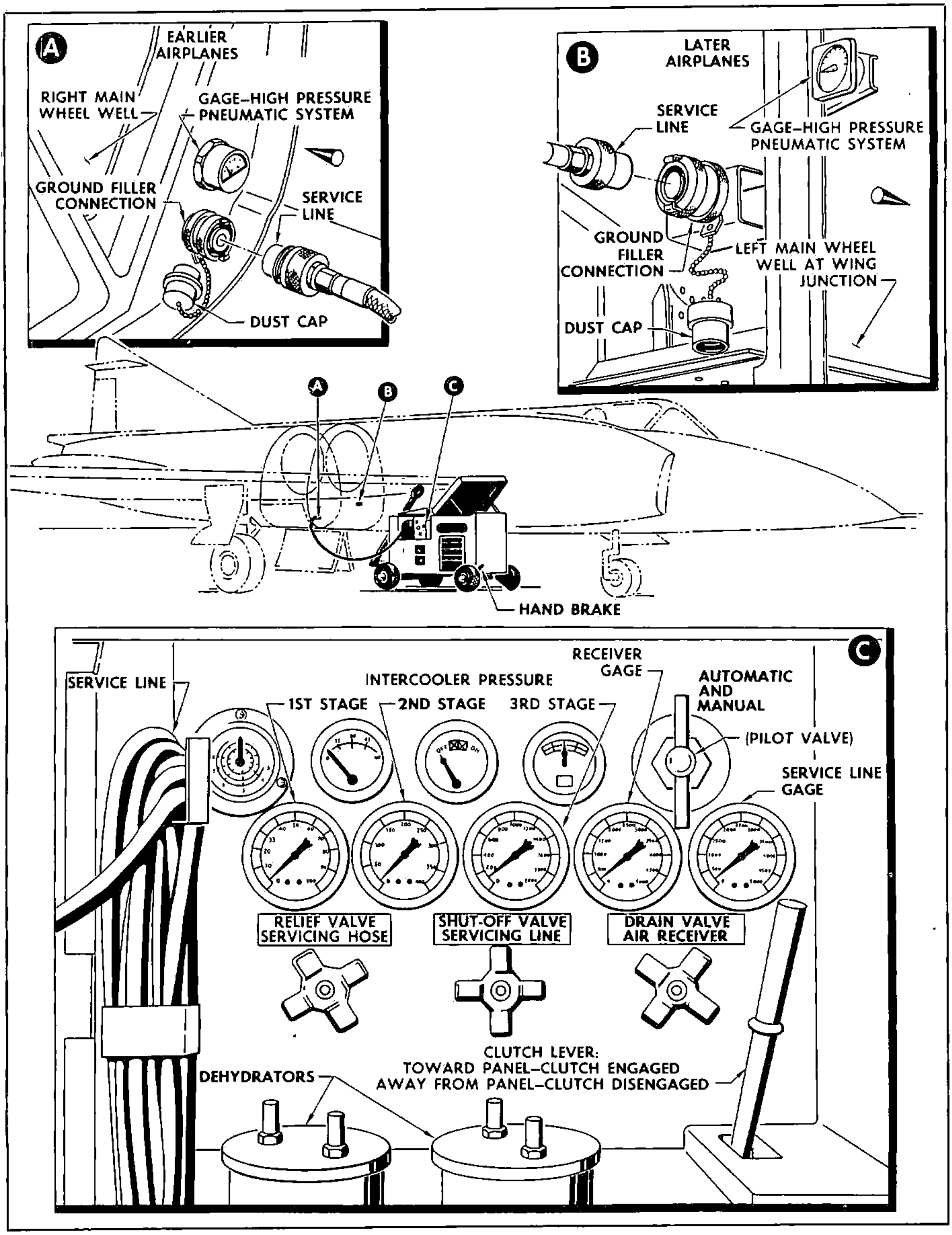


Figure 2-10. High-Pressure Pneumatic System Ground Air Compressor

High-pressure air to the service line is controlled by a priority valve. This valve will not permit air to flow into the service line until pressure has built up to 2400 psi. When the pressure in the system reaches 3500 psi, the compressor controls will automatically maintain this pressure by loading and unloading.

Complete instructions on starting, operating, and shutting down the ground air compressor are contained in the F-102A Maintenance Manual (T.O. 1F-102A-2-2).

Refer to this manual before operating the compressor. Even though there are several safeguards incorporated in the ground compressor unit which prevent air supply contamination, there are also several precautionary measures that you should take personally to prevent contamination of the air supply. You should keep the components of this unit as clean as possible, and should select an operating site as nearly dust-free as possible. Both the engine and compressor air cleaners must be cleaned daily—more frequently under severe dust conditions.

All moisture must be drained from the drain traps provided at the various components of this unit, but only when the compressor air pressure is unloaded. You must also follow very carefully the instructions given on the dehydrator instruction plate. Of course, remember at all times the dangers of performing maintenance on high-pressure pneumatic system equipment.

Charging the System with Air.

The four air storage flasks in the high-pressure pneumatic system are charged through a ground filler connection in the main wheel well. The filler and the system pressure gage are shown in figure 2-10.

To charge the system, remove the dust cap from the filler connection and attach the service hose from the air compressor to the filler valve. Be very careful that the connection is well locked; then start the compressor and charge the system to 3200 psi as indicated on the system gage. This pressure will normally drop to 3000 psi as the system cools after the charging operation.

NOTE

Only at initial filling must you wait at least 30 minutes for system pressure to stabilize. Recharge the system, as required, if system pressure drops below 3000 psi.

When the high-pressure pneumatic system has a stabilized pressure of 3000 psi, shut down the air compressor and loosen the air service hose at the filler connection approximately one turn. This will relieve service hose pressure. Then remove the service hose and replace the filler valve dust cap.

Relieving System Air Pressure.

Whenever you must remove any component in the high-pressure pneumatic system from the various subsystem control valves upstream to the storage air flasks, you must relieve the system of all air pressure. There are no shutoff valves at the air flasks. If you remove any components downstream from the subsystem check valves, it is also wise to relieve system pressure. If someone should inadvertently open a control valve in a disconnected subsystem, a blast of escaping air could cause bodily injury.

To relieve pressure in the entire system (all four air flasks, but not the wheel brake air reservoirs in the landing gear drag braces), open the bleed valve on the bottom of the forward flask in the right hand aft armament bay approximately 2 turns. This is the *priority* flask and bleeds all four flasks.

WARNING

Escaping high-pressure air is extremely dangerous. Do not allow it to strike your face or eyes, and do not hold your hands tightly over escaping air.

If at some time you wish to retain air pressure in the priority flask, but wish to relieve pressure in the other three flasks (this will be on rare occasions only), open the bleed valve on any flask other than the priority flask. This will relieve pressure in all three flasks and drop the priority flask pressure to about 1200 psi. The gage on the forward bulkhead in the main wheel well will indicate the system pressure at any time.

A check valve in each main landing gear drag brace accumulator prevents depletion of wheel brake air pressure when the pneumatic system pressure is relieved. We will cover the relieving of wheel brake pressure in the drag brace accumulators in the next chapter.

TESTING PNEUMATIC SYSTEM FOR LEAKS.

Since the power supply section of the pneumatic system is comparatively simple and contains only a few operating components, the most trouble you can have with this system will result either from contamination by dirt or from leaks somewhere in this portion of the system. Cleaning the items which may become restricted is not much of a problem. You will replace most of these units with serviceable components, or in some cases clean those items which line maintenance is permitted to disassemble. Leaks, however, are not always so easy to find.

To leak-test the system, first charge the system *to a stabilized 3000 psi.* Then allow the system pressure to stand for about one hour and recheck the system pressure. The leakage rate in this time should not exceed 100 psi. If the leakage rate exceeds this figure, then you must start looking for the source of trouble.

Air leaks may be located in two ways—either by sound or by soap testing connections. Should there be a fast leak in the system you will be able to hear the escaping air. Slow leaks, however, must be located by brushing a mild soap solution over the suspected faulty line fittings and equipment connections. Should you locate any air leaks, repair or replace the faulty fittings or the equipment; then retest the system until it is within the allowable leakage rate. At this time you are again cautioned that personal contact with escaping high-pressure air is extremely dangerous. Should repairs or replacements of pneumatic tubing or equipment be made, you must first relieve the system of all of its air pressure. You must also remember not to overtorque the fittings when repairing leaks.

The priority valve may also be tested during this procedure by opening any one of the air flask bleed valves—other than the one on the priority flask—approximately two turns and allowing all pressure to be relieved. This action bleeds all but the 1200 psi air pressure in the priority flask. The system pressure can be noted on the system pressure gage in the main wheel well. Should the pressure gage reading not be within a reasonable tolerance (about ±50 psi), then the valve is faulty and should be replaced.

After completing the leak test, wash all the soap solution from the lines and fittings that were tested, with clear water or a dampened cloth, and dry with a rag. The system may then be charged to perform the various subsystems tests and checks. These tests and checks for the subsystems are described in the next chapter.

A NOTE ABOUT MAINTENANCE.

You have been reminded several times of the troubles air contamination can cause in the pneumatic system. These troubles are chiefly connected with improper air pressures, restricted valve orifices, and the sticking or binding of system components. Since you cannot overhaul these units on the flight line, you will have to isolate the faulty units and replace them. But, as in caring for your own health, the best medicine for the troubles is to remove the causes. To remove the causes of malfunctioning pneumatic equipment, you must properly maintain the ground compressor unit, its air filter and dehydrators. You must also check the airplane pneumatic system air flask drain traps and the system air filter for moisture or dust collection. The ground compressor dehydrators and the airplane air flask standpipes are provided to trap moisture in the air supply. The air filters in both the compressor and the airplane system are provided to trap dust and foreign material. Preventive maintenance is the best and easiest maintenance.

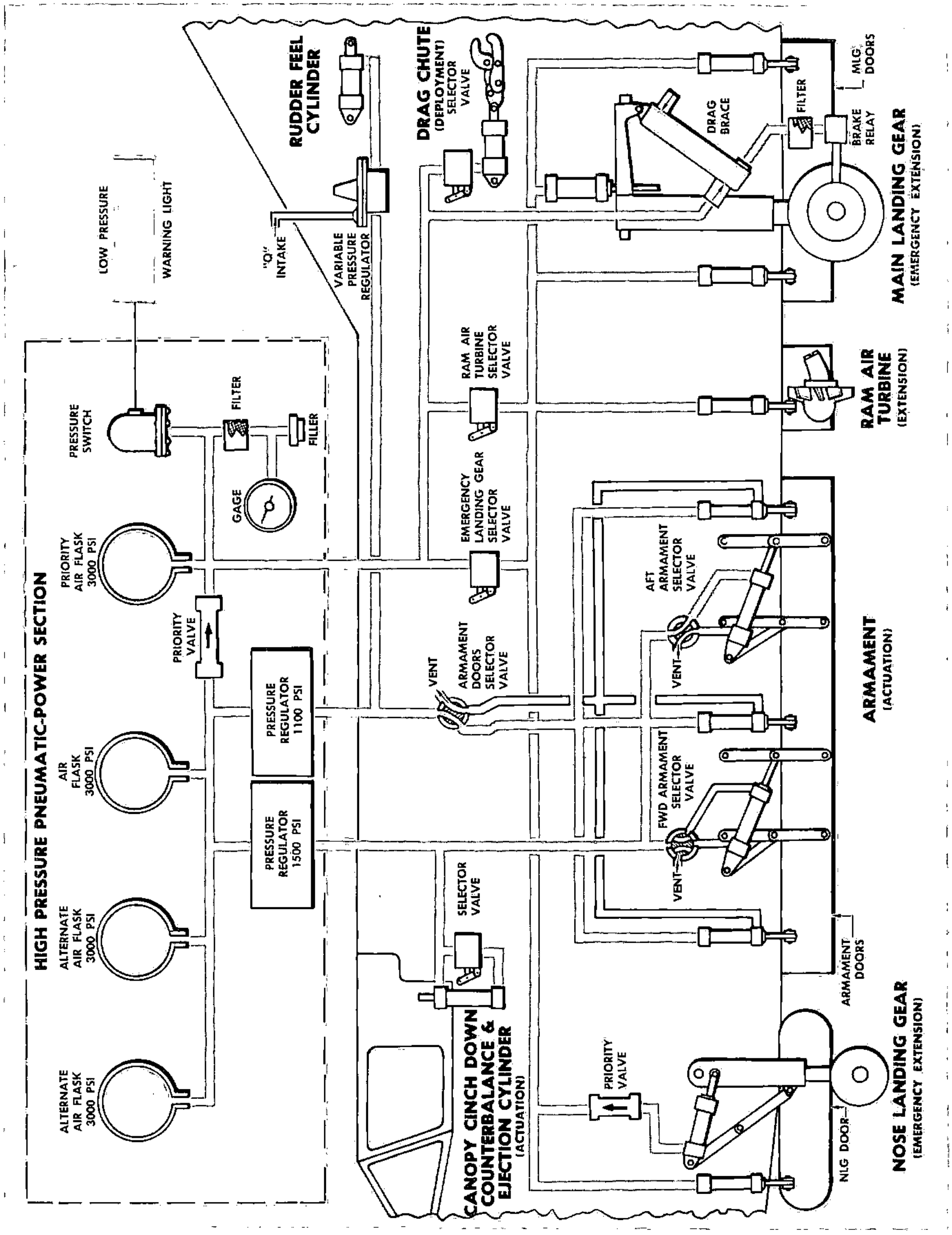


Figure 3-1. High-Pressure Pneumatic Subsystems Schematic

Chapter III

HIGH-PRESSURE PNEUMATIC SUBSYSTEMS

Contents *Page*

In Chapter I you learned that the basic concept of the high-pressure pneumatic system for this airplane is to provide an efficient power system for the operation of the armament launching system. Because of the prominent role of the armament pneumatic subsystems, it is only logical that most of our study in this chapter is devoted to the explanation of how this task is accomplished.

In figure 3-1, you can see that several other subsystems besides the armament system utilize the advantages offered by the high-pressure pneumatic system. Note also that the subsystems are divided into three categories, the 1100 and the 1500 psi regulated pressure subsystems which obtain air pressure supply through their respective pressure regulator and relief valves, and the unregulated pressure, direct line connected, emergency type subsystems.

You are reminded again of certain important facts to consider as you study or perform maintenance on this high-pressure pneumatic system and its equipment. First, keep in mind that your primary concern is to locate and replace faulty system components—not to repair the units. The reason for this is that most high-pressure pneumatic system component repairs require specialized equipment, facilities, and training which are not included in the category of organizational maintenance. And before you can understand the operation of the system, you must know how the individual components operate. Thus, in this supplement you will learn of the system and component functional characteristics, but the repair details will be omitted; those are described in the applicable technical order.

You are also again reminded of a very important safety factor that must be adhered to—respect the potent power and fast action of this system. Personal contact with escaping air at high pressure is extremely dangerous. Remember, too, a flick of the pneumatic controls means snap-action results.

ARMAMENT LAUNCHING SUBSYSTEMS.

Let's briefly examine the importance of the high-pressure pneumatic system in operating the armament launching equipment. At the beginning of this supplement you learned that the high-pressure pneumatic system provides the power source that opens the armament bay doors, extends the armament launchers, retracts and locks the armament launchers, and closes the armament doors. All of these actions are performed in about three seconds! But let's be a little more specific about these actions.

NOTE

> The electrical control of the armament bay doors and the launchers is classified data. Information on the control circuits should be obtained from the Armament System Training Supplement or the applicable Technical Order Maintenance Manuals.

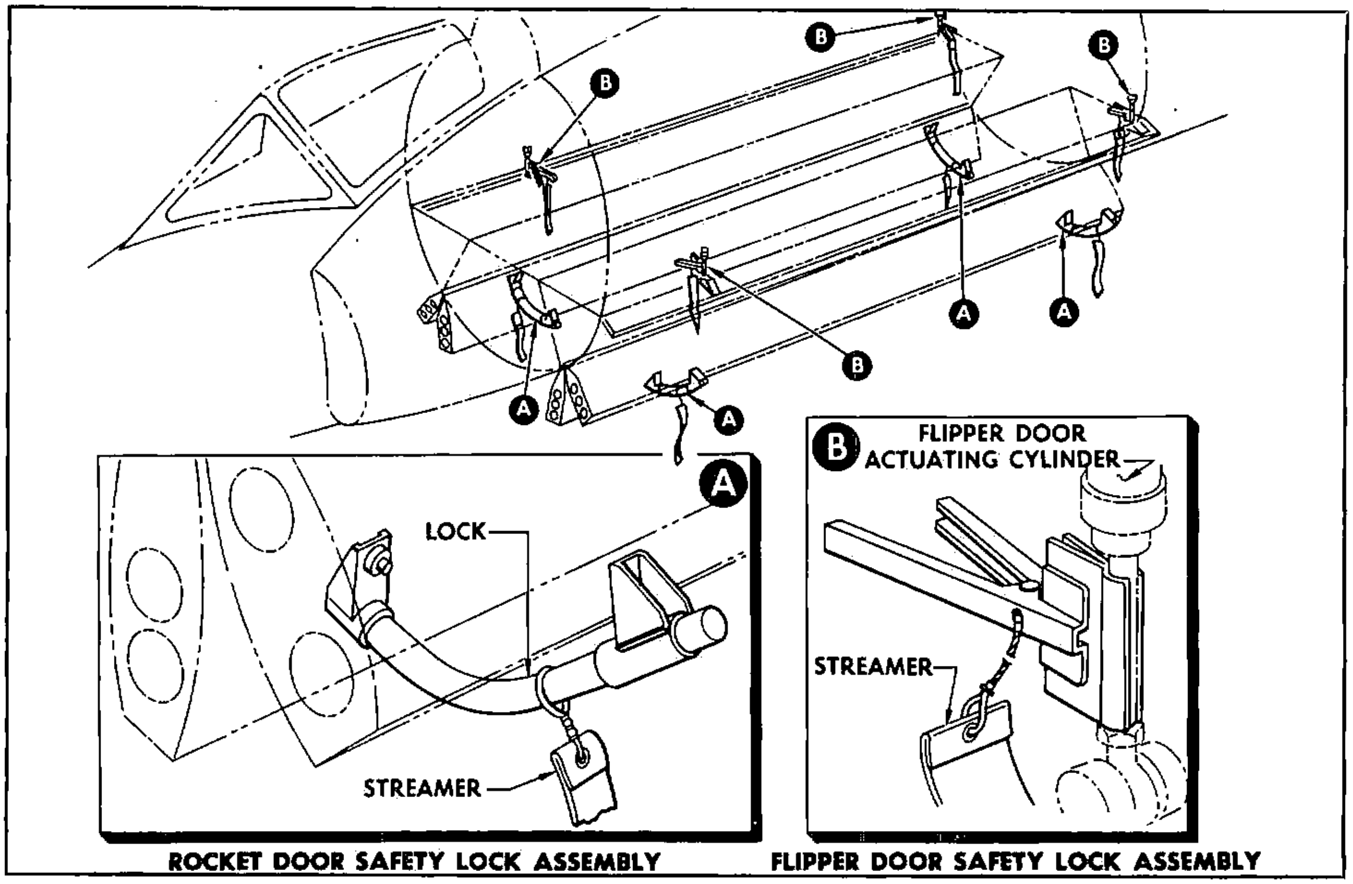


Figure 3-2. Door Ground Safety Locks

By referring to the schematic (figure 3-1) you can follow the operation of the armament bay doors and the launchers. The armament bay door selector valve receives an electrical signal from the fire control system to open the doors. Air pressure, routed by the armament door selector valve to the *down* ports of the 18 door actuating cylinders, opens the doors.

The limit switches on the armament bay doors actuate when the doors reach the fully open position and send an electrical signal to energize the armament selector valve. This valve then routes air pressure to the respective launcher unlatching cylinders and the launcher actuating cylinders. A small volume of air first operates the unlatching cylinders before the launcher actuating cylinders extend. The unlatching cylinders unlock the launcher mechanical uplock latches. Air pressure directed to the respective actuating cylinders then lowers and snubs the launchers at their firing positions.

After the armament is fired, an electrical signal is received by the armament selector valve causing it to route air pressure to the *up* ports of the launcher actuating cylinders. The launcher assemblies then retract and lock. When the assemblies are up and locked, their respective uplock switches actuate to send a signal to the armament door selector valve which routes air pressure to the *up* ports of the 18 armament bay door actuating cylinders. This action closes the armament bay doors. Mission accomplished. Operating time—about three seconds!

The armament pneumatic launching equipment performing the task just described is divided into two functional subsystems—one for the armament bay doors and one for the armament launcher assemblies.

GROUND SAFETY LOCKS.

When the airplane is on the ground, special ground safety locks, shown in figure 3-2, hold the armament bay doors open. These locks must be installed any time the doors are opened for access to the armament bays. They prevent the accidental closing of the doors on personnel working in the armament bays. Note that two types of safety locks are used. A clamp-type lock (B) is installed on the piston rods of the outboard (flipper) door actuating cylinders, while a retaining-type lock (A) holds the inboard door of each outboard bay to its adjacent center bay door. Remember—always check that the armament bay door safety locks are installed before entering the bay area. You can easily tell when the locks are installed by the long, red, REMOVE BEFORE FLIGHT streamers attached to them.

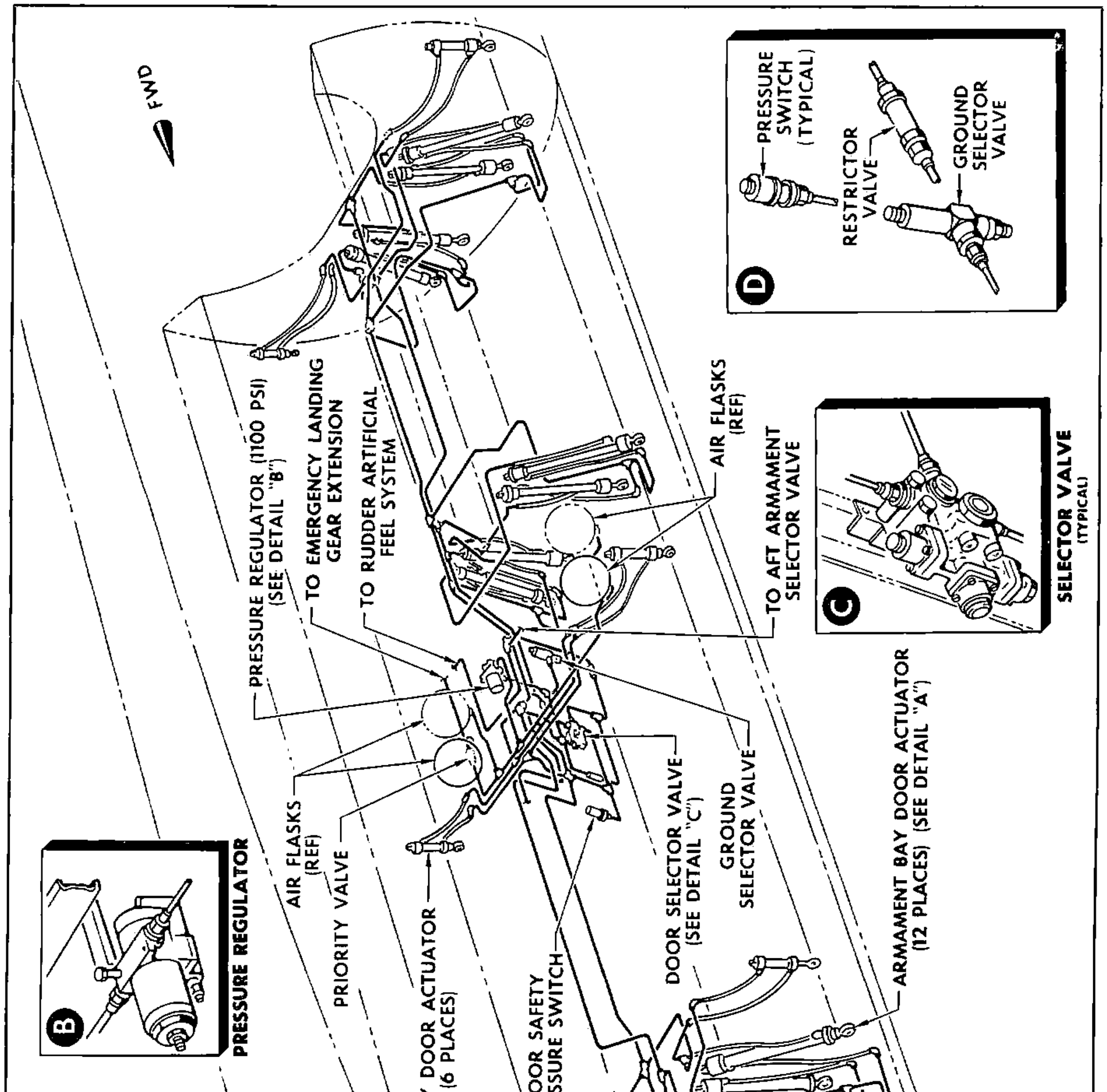


Figure 3-3. Armament Bay Door Pneumatic System

ARMAMENT BAY DOOR PNEUMATIC SYSTEM.

As you can see in figure 3-3, three bays are located in the lower midsection of the fuselage. They are designated as the left, right, and center armament bays. Note, too, that each bay has a forward and an aft section. All armament bay doors are actuated by pneumatically actuated cylinders which are supplied with air pressure from the high-pressure pneumatic system.

As you see in figure 3-1, the door actuating cylinder air pressure is routed through a pressure regulator and relief valve. The regulator is designated as detail B in figure 3-3, and is located in the aft portion of the center armament bay. In Chapter 2 you learned that both 1100 and 1500 psi regulated air pressures are provided by the two pressure regulator and relief valve assemblies in the pneumatic power system. The armament bay doors operate from the 1100 psi pressure regulator.

Door Selector Valve.

Note in the perspective and schematic illustrations of the armament bay door pneumatic system that after air is routed through the 1100 psi pressure regulator and relief valve, the system line then directs the air pressure to the door selector valve.

The door selector valve is a solenoid operated, double-cylinder, piston-type air valve. Figure 3-4 shows both a sectional perspective view and an operational schematic view of the valve. You can see that this valve directs air to either the *open* or *close* ports of the armament bay door actuating cylinders. This valve has one pressure extrance port, two pressure out ports, and a vent port. Note that the pressure line connects to the entrance port. The two pressure out ports are identified on the casing by letters A and B. You can also see that port A directs air to the *close* side of the door actuating cylinders, while port B directs air to the *open* side of the cylinders. The vent port of the selector valve exhausts air from the actuating cylinders.

An electrical signal from either the fire control system, the manual trigger on the pilot's control stick, or the ground operating switch on the nose wheel well switch panel, controls the solenoids of the door selector valve. When a *door close* signal originates from one of these controls, solenoid A is energized and solenoid B is deenergized. This action causes air pressure entering the valve to flow past the ball-seat of solenoid A and to displace the poppet valve which is shown in the schematic detail. This air pressure pushes the poppet to the left and permits inlet pressure to flow past the poppet valve ball seat and out port A to the cylinder *close* side. Solenoid A remains energized until a *door open* signal is received, or until electrical power is removed from the airplane.

This door selector valve poses two types of problems—malfunctioning of the electrical solenoids, or the sticking of valves in the *retract* position. If you isolate the selector valve as the source of trouble, remove the valve and replace it with a serviceable unit. Do not replace the valve if you can correct the malfunction by tightening loose pneumatic system fittings or by repairing the electrical connections. You should, however, be very careful when tightening the loose fittings so as to prevent stripping of the threads by overtorquing.

Door Actuating Cylinders.

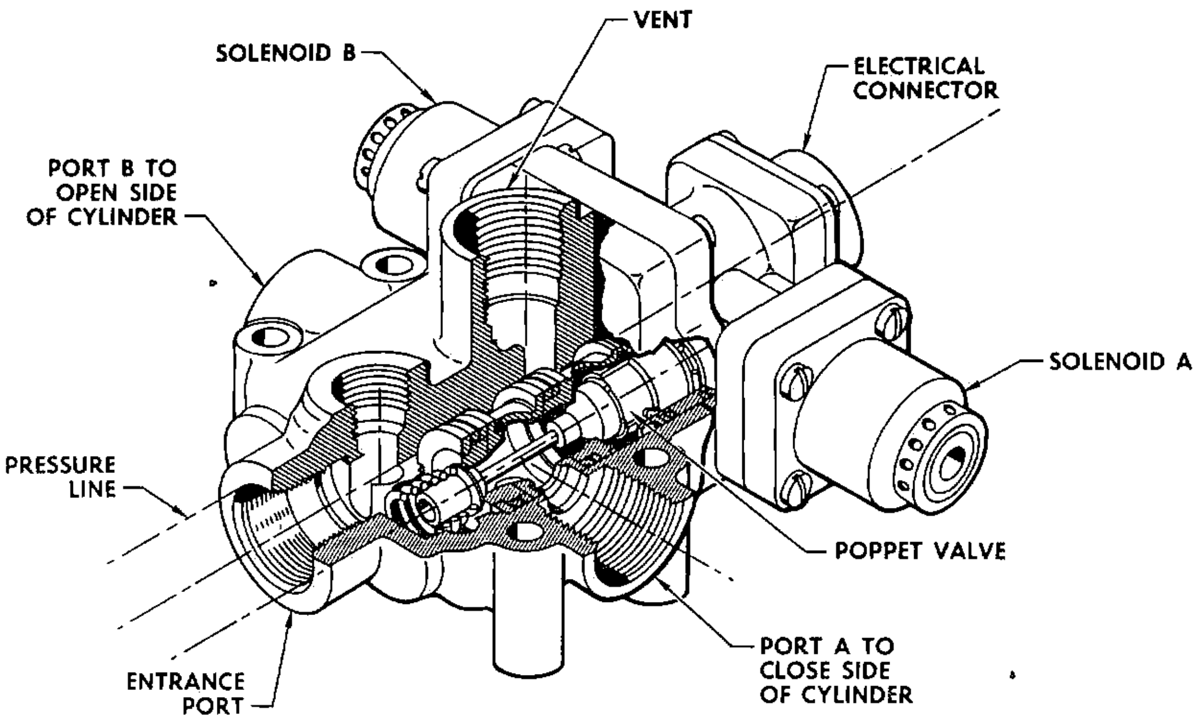
In figure 3-5, note that 18 double-acting pneumatic cylinders open and close the six armament bay doors. Note also that three cylinders operate each of the doors. The right and left outboard flipper door actuating cylinders are smaller in size than the actuating cylinders on the four rocket carrying doors.

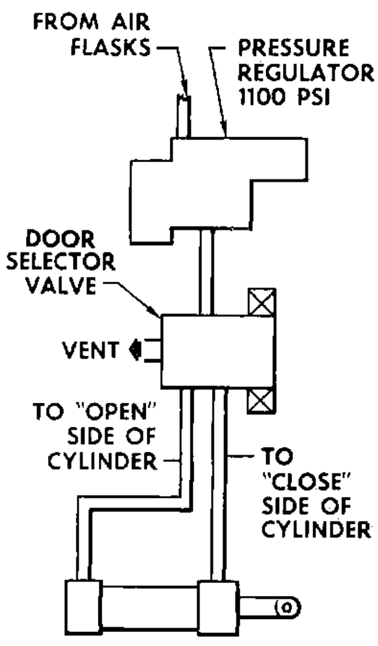
The head ends of the cylinders attach to the airframe structure with a hinge-type mounting. The rod ends of the cylinders attach to the doors in the same manner. Such an arrangement permits a pivoting movement of the cylinders during the door-operating cycle. Another feature shared by all door cylinders is an internal locking device that locks the piston in the *door close* position. A poppet valve arrangement within the cylinders (see detail A) unlocks these locking devices during the door opening cycle.

You can also see, in figure 3-5, that 12 cylinders operate the main armament doors. A typical main armament door cylinder is shown in detail B of this illustration. The 12 cylinders are similar in their operation, but vary in their stroke (length of piston travel) and their size. At corresponding positions on the right and left doors, these cylinders are identical in size and have the same stroke. The cylinders at corresponding positions on each center door are also identical in size and stroke. The various cylinder strokes and sizes make up for the door curvature that follows the contour of the fuselage.

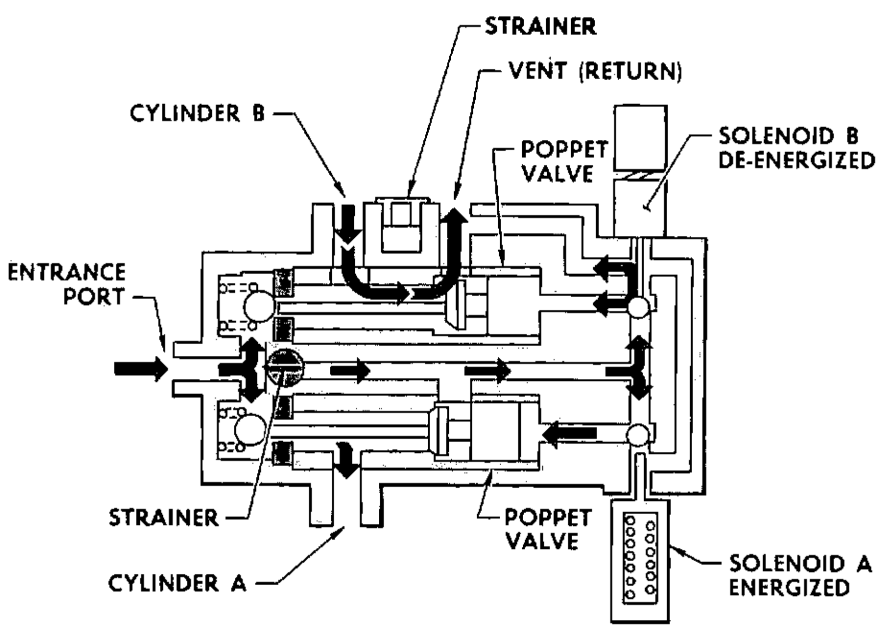
Detail A of this illustration shows a flipper door cylinder. Note the locking finger device which locks the piston in the *door close* position. All six cylinders on the right and left flipper doors are identical units—they are not paired, as are the main armament door cylinders. As you can see, the flipper door cylinders are similar in construction and operation to the larger cylinders of the main armament doors.

Notice that air enters the cylinders through two ports, one on each end of the cylinder. Air entering the head end of the cylinder extends the piston rod to open the doors. Air entering the rod end of the cylinder retracts the piston rod to close the doors. A piston locking





DOOR ACTUATOR CYLINDER
(TYPICAL 18 PLACES)



DOOR CLOSE VALVE OPERATION SCHEMATIC

Figure 3-4. Door Selector Valve

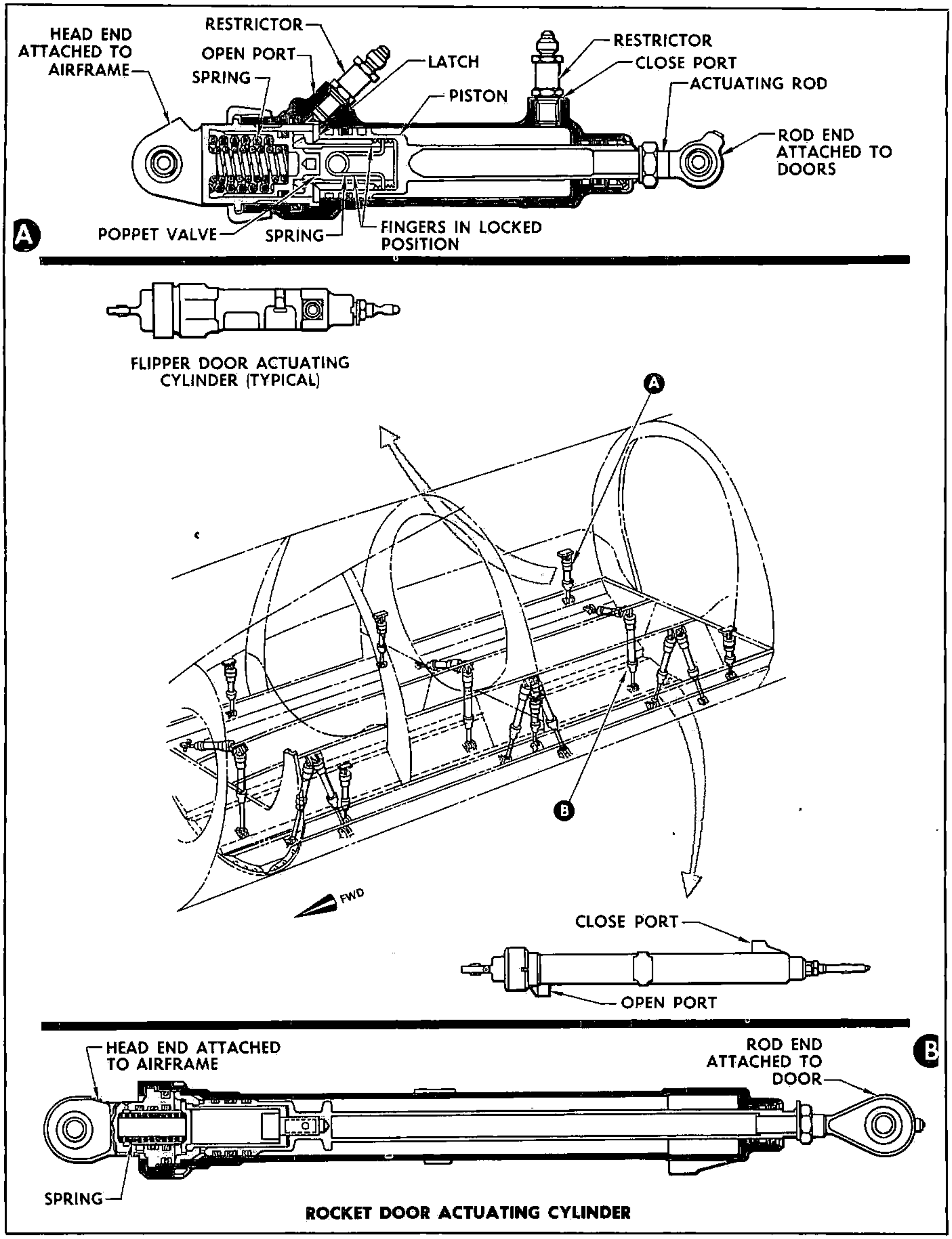


Figure 3-5. Typical Door Actuating Cylinders

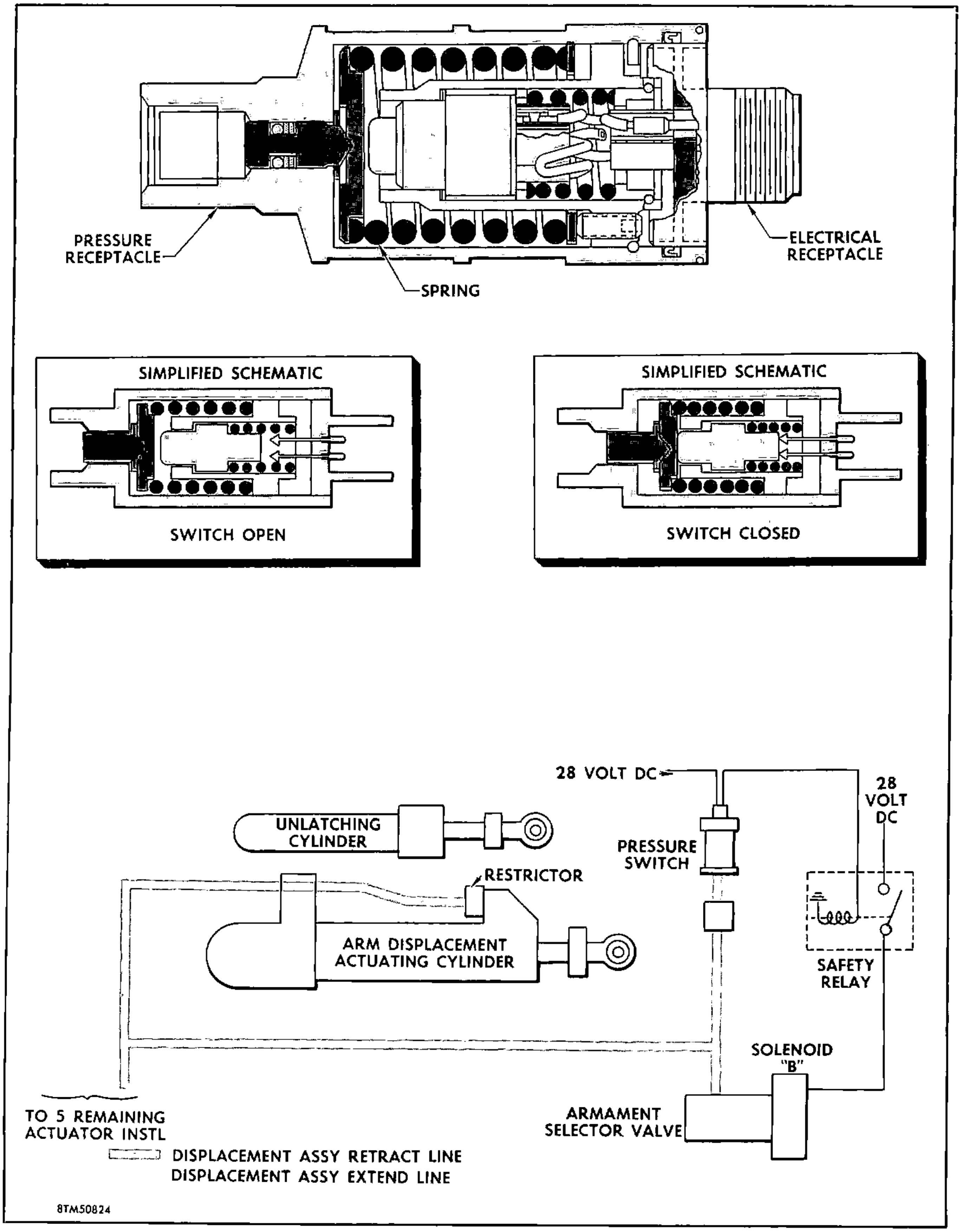


Figure 3-6. Pressure Operated Switches

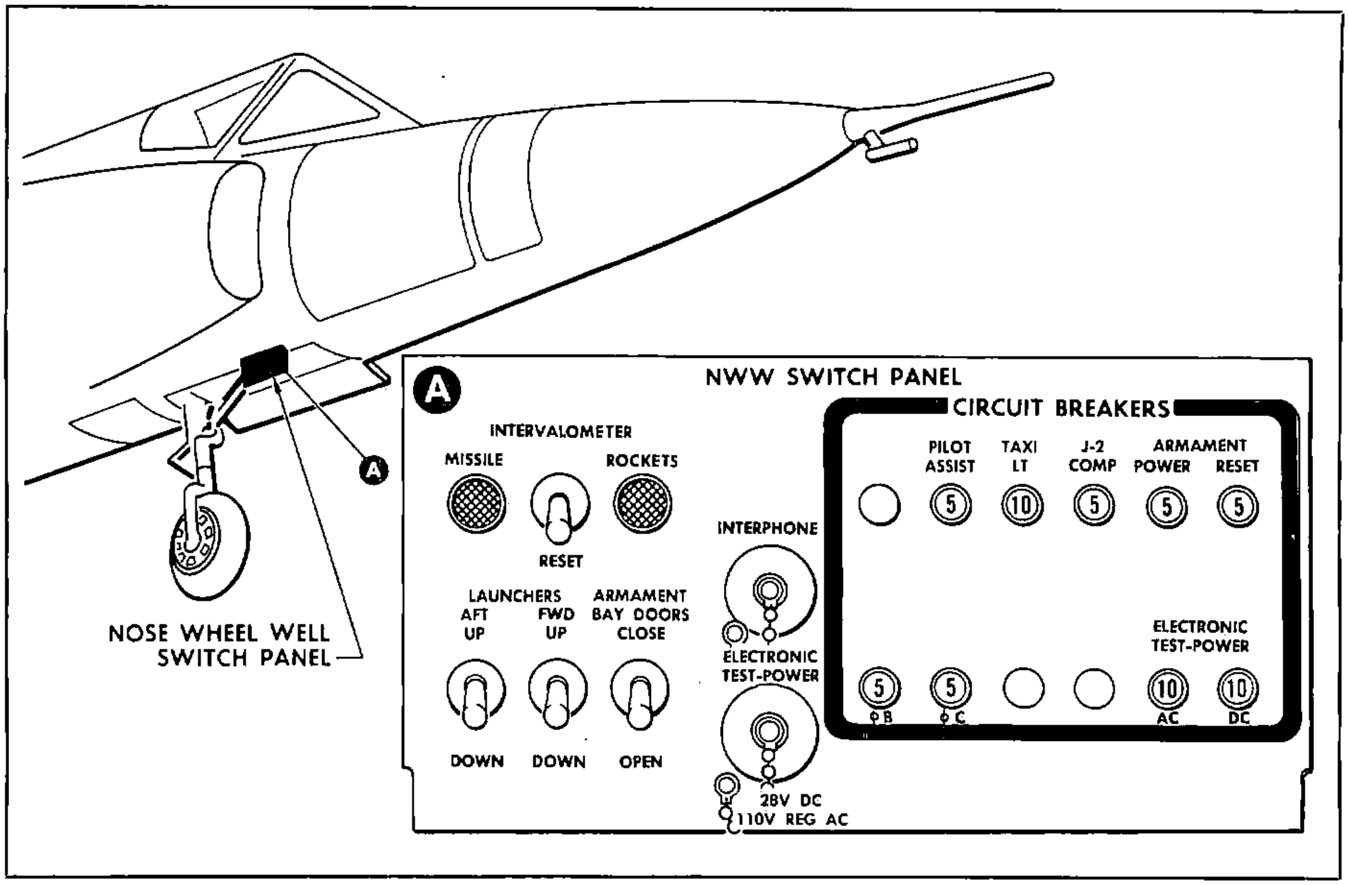


Figure 3-7. Nose Wheel Well Switch Panel

mechanism, controlled by the piston poppet valve, extends fingers to retain the piston in the *door closed* position. The holding or releasing action of this mechanism depends on the pressure acting on the piston poppet valve. The door actuating cylinder *open* and *close* ports contain restrictors. These restrictors allow the free flow of air into the cylinders and restrict the flow of air out of the cylinders. When pressure is applied to one side of the cylinder, a snubbing action results since the air in the vented side of the cylinder has a restricted flow. This snubbing action keeps the piston from violently hitting against the end of the cylinder case and damaging the doors.

Pressure Switches.

In figure 3-6, note the pressure switch, located in the door close line of the armament displacement actuating cylinder. This switch will not permit door operation if the air pressure in this line drops below 500 psi. It also assures sufficient air pressure for the snubbing action of the cylinders when the doors are opened.

The pressure switches prevent "unsnubbed" opening of the doors and extension of the launcher assemblies. A minimum pressure of 500 psi, on the close side of the door actuators or on the retract side of the launcher assembly actuators, will actuate the pressure switches. As you can see, without this snubbing pressure on the actuators, the door open or the launcher extend circuits would not be complete, and no action of these units would take place.

Ground Operation of the Doors.

A door position switch on the nose wheel well switch panel controls the ground operation of the doors. You will find that this switch provides a convenient means of opening and closing the doors during maintenance and operational checks. Figure 3-7 shows the location of the switch on the panel. It is a toggle-type switch and has three positions: OPEN, CLOSE, and a center OFF position. Spring tension automatically returns it to the center OFF position. Notice that the two momentary positions—OPEN AND CLOSE—identify the functions of the switch. A hinged guard (not shown in the illustration) covers this switch and prevents the accidental operation of the doors.

Alternate Ground Control Method.

The remote ground cord assembly (figure 3-8) provides an alternate method of controlling the door operation. This assembly consists of a 20-foot extension cord with a disconnect plug on one end and a junction box on the other. Note that the plug end attaches to a

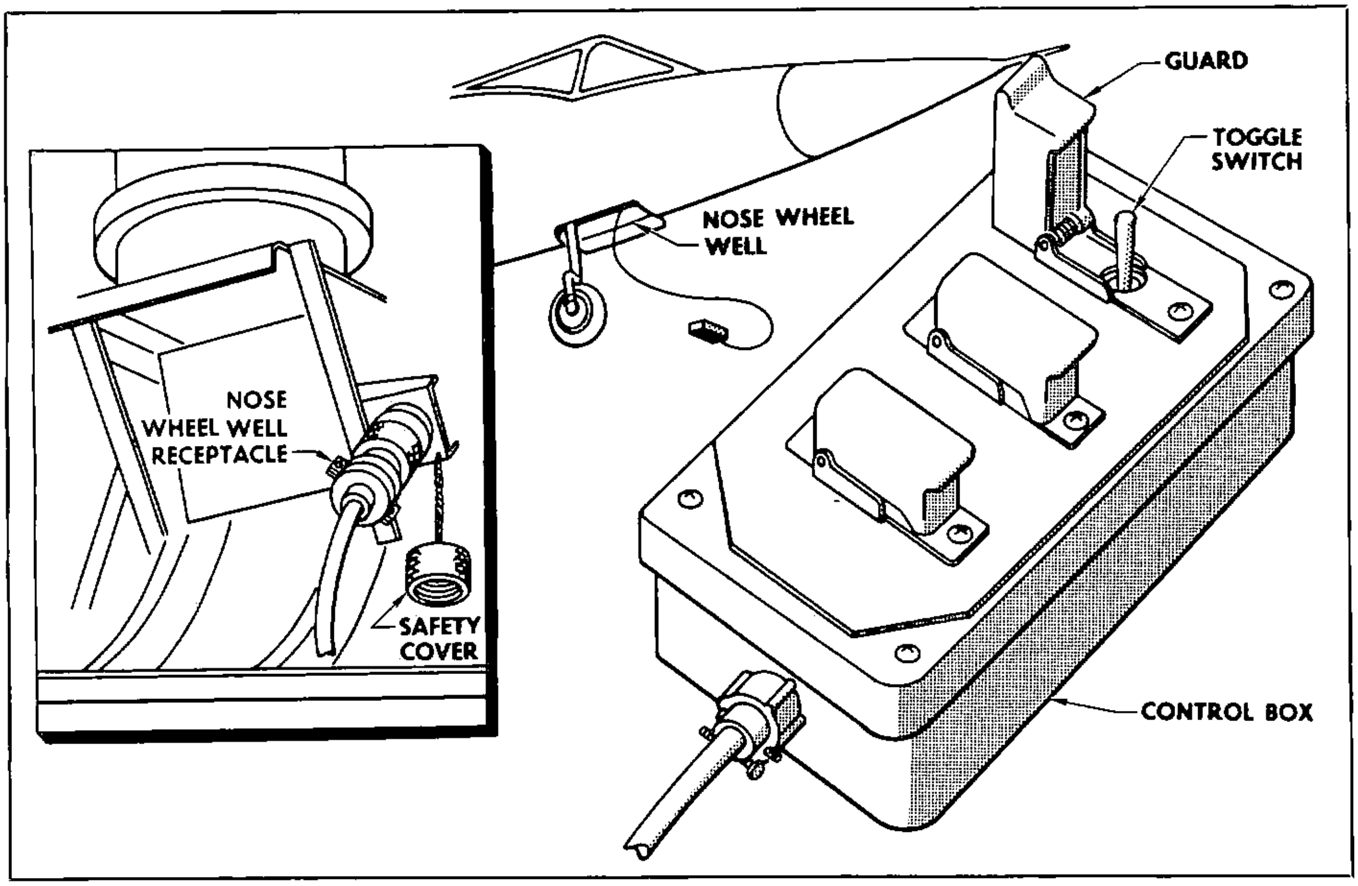


Figure 3-8. Remote Ground Cord Assembly

receptacle in the nose wheel well. (On later airplanes this receptacle is in the main wheel well.) This receptacle connects to the same circuits as the control switches on the nose wheel well switch panel. The remote junction box has three guarded control switches identical to those on the nose wheel well switch panel. Two of these switches control the forward and aft launcher assemblies, and the third switch controls the armament bay doors. The identification of the switch positions corresponds to those on the nose wheel well switch panel. This method of remote operation controls the doors when loading live armament in the aircraft. Thus, personnel can take a safe position, clear of the aircraft, during door operation.

Erratic Operation of the Doors.

The absence of snubbing pressure will cause the doors to open too fast. Various factors may cause this condition. Installing the restrictor valves backward in the ports of the actuating cylinders allows a free flow of air out of the cylinders. Thus, snubbing pressure escapes. A faulty system pressure regulator and relief valve will also cause erratic door operation.

When pressure is low on the *close* side of the door cylinders, the safety pressure switch will prevent pneumatic pressure from reaching the *open* side of the door actuating cylinders. This switch operates at a minimum pressure of 500 psi. Any defect which would cause it to operate incorrectly results in the "unsnubbed" opening of the doors. The electrical safety relay connects in series to this pressure switch. While using various pressures in the pneumatic system, check the operation of the pressure switch and the safety relay with a voltmeter. This will indicate the proper operation of the pressure switch at these various pressures.

Failure of the Doors to Open.

If the doors fail to open, you should first check all components outside of the bays before manually opening the armament bay doors. Since the pneumatic system operates the cylinders which open and close the doors, and electrical power operates the devices controlling the pneumatic system, you should first make a power distribution check. If the pneumatic system pressure gage reads above 2200 psi and voltage reaches the nose wheel well switch panel, you can then eliminate the external power units as a source of trouble and check the components of the door pneumatic system.

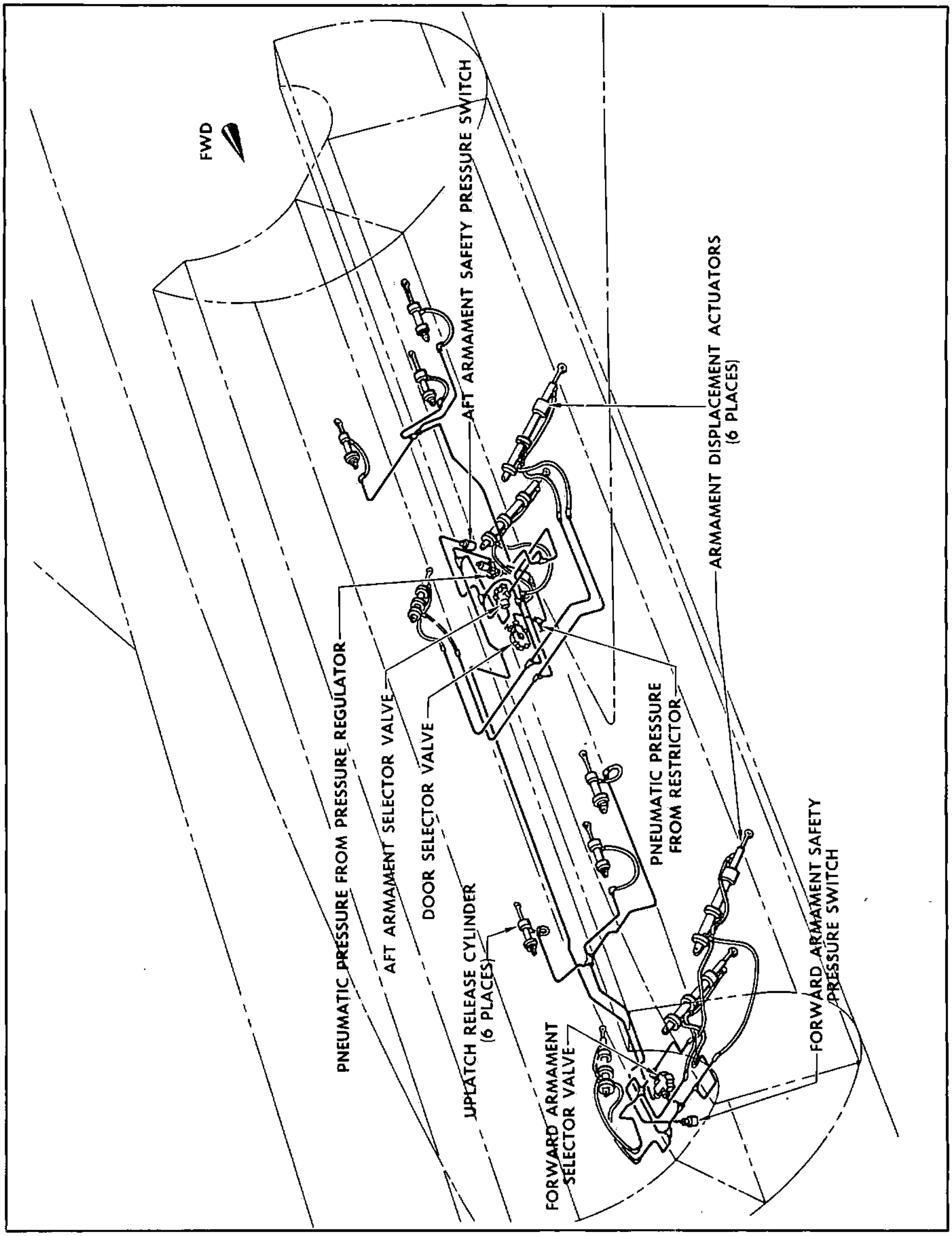


Figure 3-9. Armament Launcher Pneumatic System

ARMAMENT LAUNCHER PNEUMATIC SYSTEM.

There are two armament launcher assemblies in each of the three armament bays—one forward and one aft. A pneumatic powered actuating cylinder extends and retracts each of these six launcher assemblies. One end of each actuating cylinder attaches to the airframe while the rod end connects to the launcher unit. These double acting cylinders extend the assemblies for the armament firing and retract them after firing.

In figure 3-9 you can see the location of the various components of the launcher pneumatic system. The forward and aft armament selector valves control the operation of the launcher actuating cylinders and unlatching cylinders. The forward selector valve controls the air pressure to the three forward launcher actuating cylinders. When the selector valve directs pressure to the extend port of the three forward launcher actuating cylinders, the cylinders extend the forward assemblies into firing position. When the forward selector valve directs pressure to the retract ports of the forward actuating cylinders, the cylinders retract the forward assemblies into the bays. The aft selector valve controls the three aft actuating cylinders in the same manner.

Launcher Actuating Cylinders.

Detail "B" of figure 3-10 shows a typical launcher assembly actuating cylinder installation and a cutaway. Notice that a snubber spring and trapped air in the buffer cylinder cushion the piston during the retraction cycle. The buffer cylinder check valve permits metered air pressure from the *retract* line to enter the *extend* side of the cylinder. This trapped air forms a buffer in the head of the cylinder during the retract cycle. This cylinder does not contain the locking mechanism which you learned about in the door actuating cylinders. As you can see in the illustration, the extend pressure simply pushes the actuator rod, or piston, out to its fully extended position.

Armament Selector Valves.

When we discussed the launcher assembly operations, you learned that two armament selector valves control the flow of air pressure to their respective launcher actuating cylinders. These selector valves are identical to the door selector valves illustrated in figure 3-4. A faulty selector valve is fairly easy to locate—in the majority of cases, the launcher assembly just doesn't operate.

However, as in checking trouble for other pneumatic system components, you should first check that you have at least 2200 psi of system operating pressure. Unlike other pneumatic system components, the selector valve also has electrical solenoids that must operate properly. Therefore, a check must be made to see that the unit is receiving electrical power. If, after these checks are completed, the launcher assembly is still inoperative, bleed the system air pressure and replace the selector valve.

Extending Displacement Assemblies.

After you open the doors and install the ground safety locks in their correct locations, use the LAUNCHER UP and DOWN switches—on the nose wheel well switch panel or the remote ground cord—to control the launcher assembly operation. When you place one of these switches in the DOWN position, the respective group of actuating cylinders will extend the assemblies. If the assemblies do not extend, this indicates the absence of snubbing pressure on the retract side of the assembly actuating cylinders. Momentary actuation of the switch to the UP position, then back to the DOWN position, should overcome this situation.

Maintenance Problems in the Launcher System.

Maintenance problems of the launcher pneumatic system will be connected chiefly with improper air pressures, air leaks, restricted orifices, and the sticking or binding of certain components. Since you cannot overhaul these units on the flight line, you will have to isolate the faulty component and replace it.

When isolating a pneumatic system malfunction, first determine that the high-pressure pneumatic system has the required pressure by checking the pressure gage located at the ground filler connection. For a series of pneumatic system checks, you should recharge the system whenever the pressure is below 2200 psi. Leaks at the various connections in the system lines, the control valves, the pressure switches, and the actuating cylinders will affect the system air pressure. You can hear the hissing sound of a leak, or when you use a mild soap solution you can see the air bubbles formed from the escaping air.

You may encounter binding, misalignment, and end play of connections in the mechanical malfunctioning of the displacement and armament door assemblies. With high-pressure pneumatic power operating the system, you can not easily detect these conditions of operation. Therefore, under these conditions you should check the assemblies with the pneumatic actuators disconnected. Information concerning the mechanical and electrical operational checks and tolerances should be obtained from the Armament System Training Supplement of this series and from the Maintenance Manual T. O. 1F-120A-2-12.

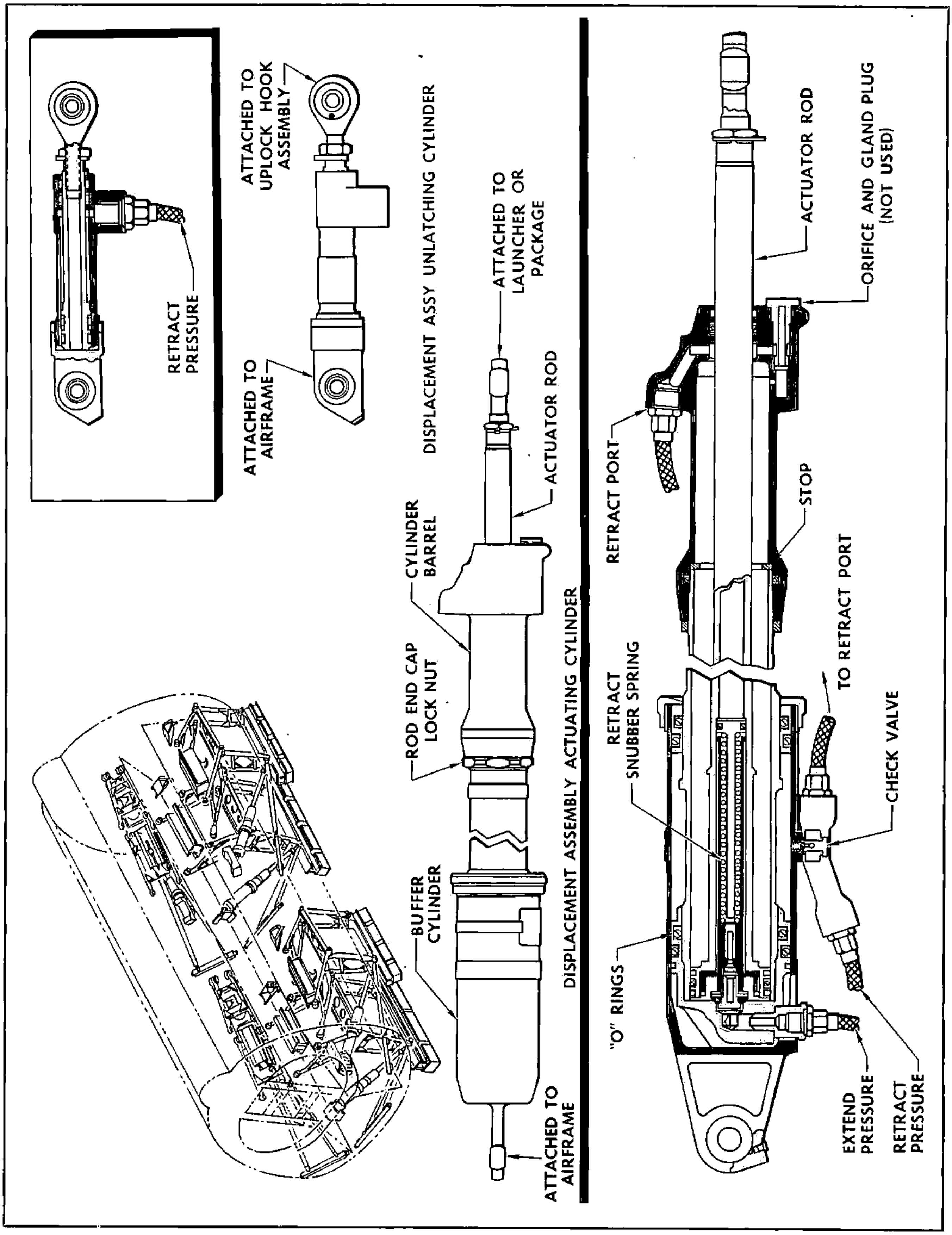


Figure 3-10. Launcher Assembly Unlock and Actuating Cylinder

LANDING GEAR EMERGENCY EXTENSION SYSTEM.

The main and nose landing gears and their respective wheel well doors are normally extended and retracted by hydraulic power. This operating hydraulic power is controlled electrically by the manual up-and-down movement of the normal landing gear control handle on the extreme left side of the cockpit instrument panel.

But what would happen if either the hydraulic system or the electrical system or both failed in flight? Then the pilot would need some emergency means of extending the gear for a landing. On the F-102A this emergency extension is accomplished by introducing high-pressure air into the gear and gear door cylinders to open the gear doors and lower the gears.

The high-pressure pneumatic system supplies this air pressure, and the emergency extension of the gears is controlled mechanically by pulling the emergency gear handle located under the left side of the instrument panel. Furthermore, this emergency landing gear extension can be accomplished even when the pneumatic system supply has dropped to about 900 psi.

If you wish to study the entire landing gear system, refer to the Airplane General Training Supplement of this series. However, in this High-Pressure Pneumatic System supplement we will cover only the emergency pneumatic system.

HOW THE SYSTEM OPERATES.

Figure 3-11 shows the emergency gear extension pneumatic system. Now let's see how the landing gear operates during an emergency extension. With the normal landing gear control handle in the UP position, the gears and doors are up and locked. For emergency extension, the pilot must pull the LG EMER EXTEND handle located near his left knee and under the instrument panel. Pulling this handle actuates the Teleflex cable which connects the cockpit control handle to the heart of the emergency landing gear extension subsystem—the control valve assembly.

Pulling out on the cockpit control handle raises the lever shown at the bottom of the control valve to open the valve and actuate the emergency down switch. Should there be electrical power to the landing gear control system, the switch actuation removes electrical power from the entire landing gear control circuit beyond this switch. When electrical control power is removed, the system hydraulic selector valves automatically neutralize. This blocks the flow of hydraulic pressure at the selector valves.

At the same time the LG EMER DOWN switch is actuated, the control valve is forced opened by the cam lever. Air pressure from the high-pressure pneumatic system supply, entering the top of the valve, then passes through the valve to the extend side of the landing gear and door hydraulic actuators. The pneumatic pressure in the lines repositions the shuttle valves at each of the gear and gear door actuating cylinders (see figure 3-13), thus blocking the hydraulic lines and permitting pneumatic pressure to enter the actuators. This shuttle valve repositioning now provides the means of the entire landing gear emergency extension operation being a pneumatic function.

There must, however, be a proper distribution of pneumatic pressure to open the landing gear doors before gear extension and to lower the gears in the proper sequence. This action is accomplished by the priority valve in the nose gear extension actuator line and two restrictor valves—one in the main gear line and one in the nose gear line.

Note that there is no restrictor in the main gear door line, permitting the actuating cylinders at the doors to open the doors before the main gear extends. The main gears will extend as their doors reach the full open position. The restrictor in the nose gear line slows down the operation of both the nose gear and its door until the main gear doors have started to open.

Now note the priority valve in the nose gear line. This valve is designed to remain closed until there is sufficient pressure to open it. Since the nose gear door cylinder has no restriction, it will open first; then all pressure can be concentrated on it. With sufficient pressure (about 700 psi), the valve opens to permit air pressure to enter the nose gear cylinder and lower the gear.

Since air will contaminate the hydraulic fluid area downstream of the shuttle valves, the system requires bleeding and recirculation with fresh hydraulic fluid after every emergency landing gear operation. You must jack the airplane to do this, and cycle the gear operation about five times.

CONTROL VALVE ASSEMBLY.

As previously mentioned, the heart of the emergency landing gear extension subsystem is the control valve assembly. This assembly consists of a mechanically actuated LG EMER DOWN switch and a control valve as shown in figure 3-11. The assembly is located on the forward left side of the nose wheel well under the cockpit floor.

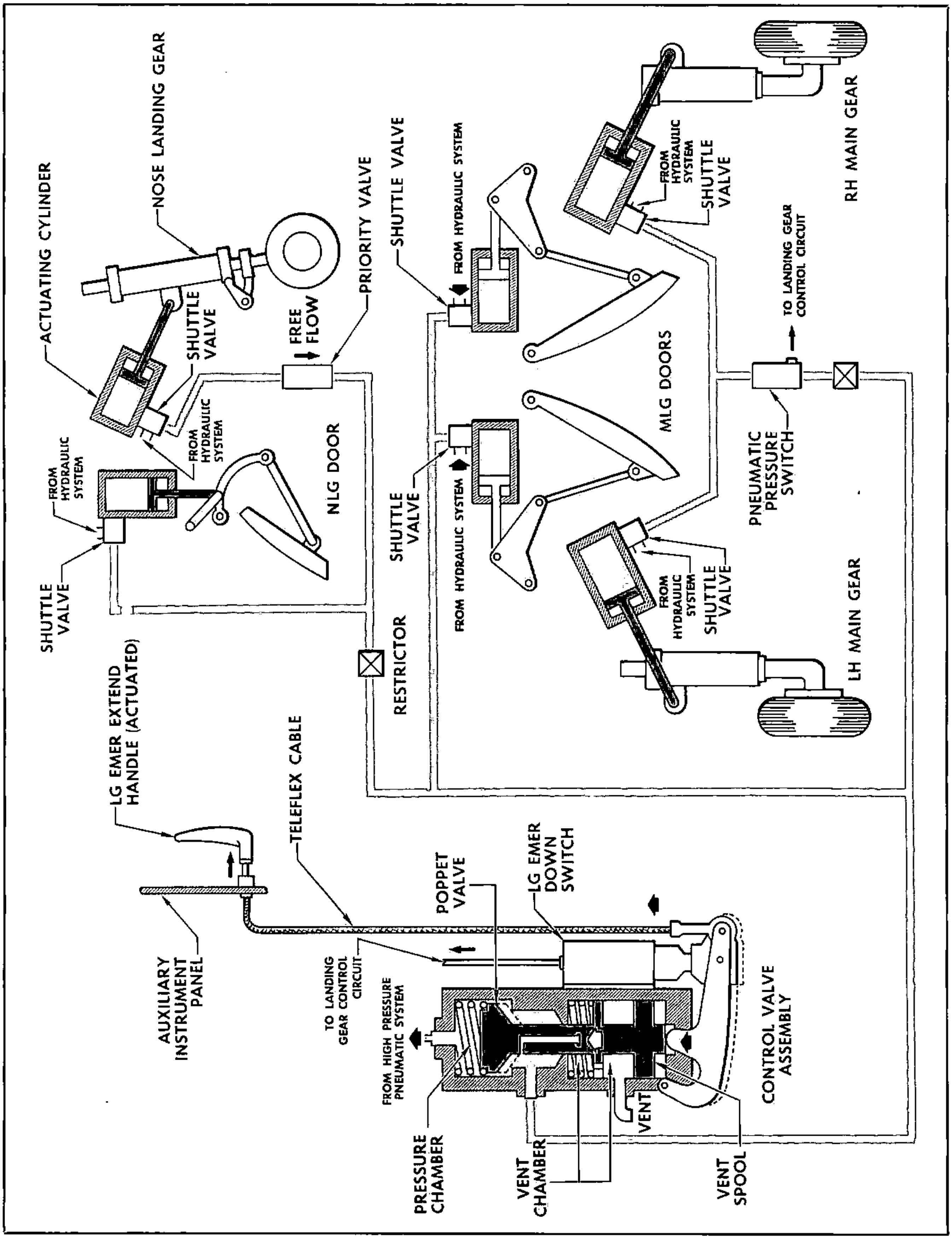


Figure 3-11. Emergency Landing Gear Extension Schematic

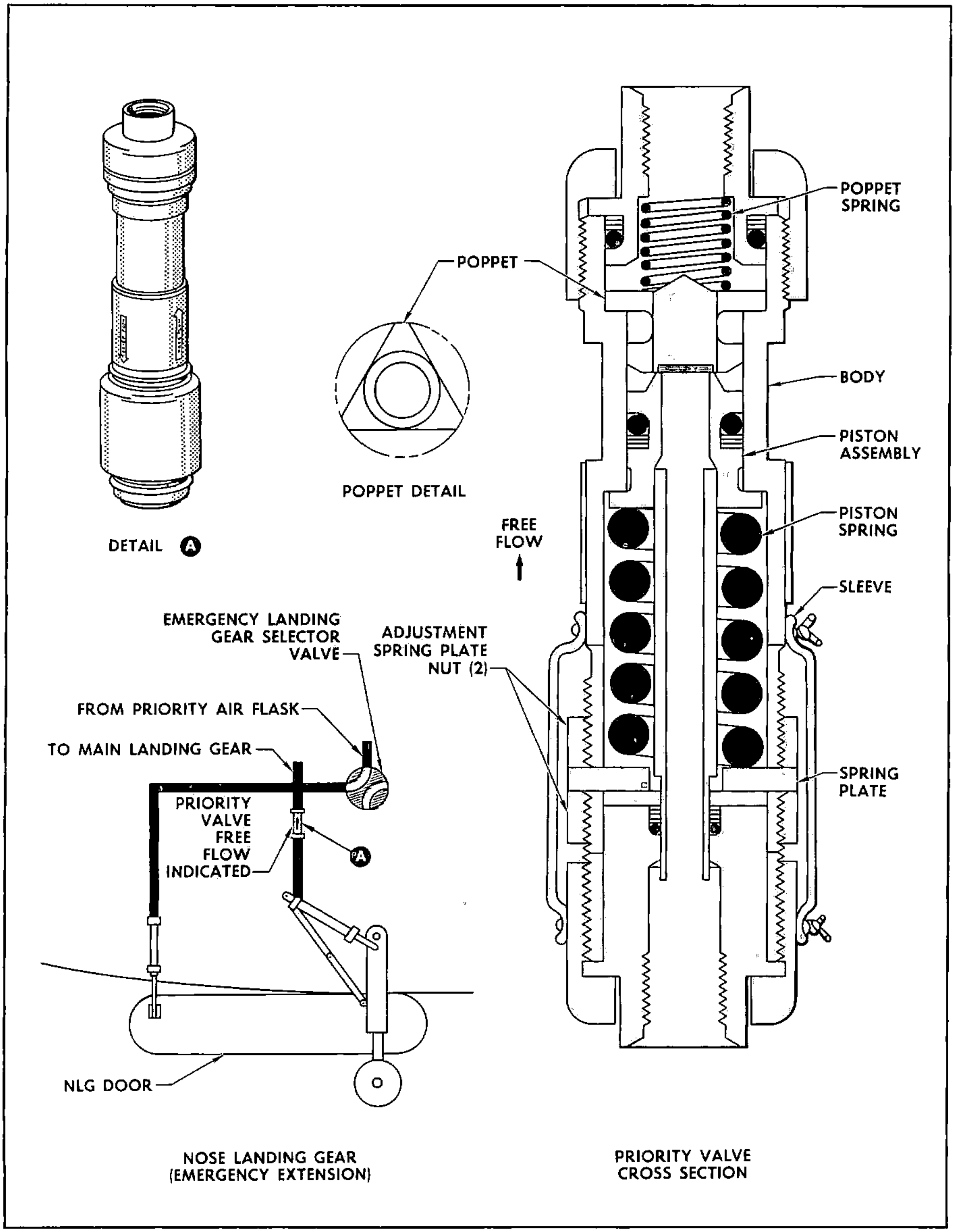


Figure 3-12. Nose Landing Gear Priority Valve

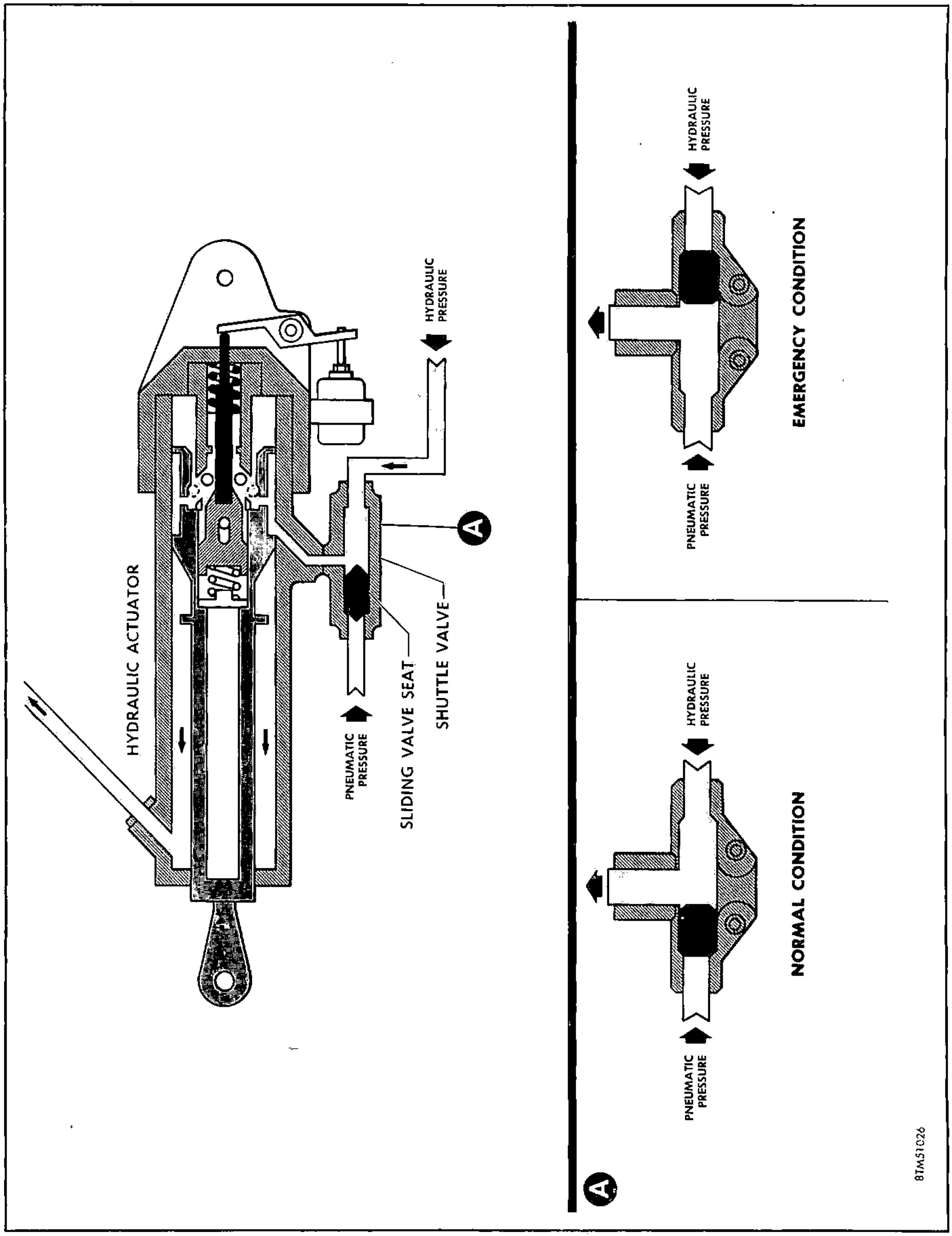


Figure 3-13. Landing Gear Shuttle Valves

The LG EMER DOWN switch on the valve assembly is a normally closed type (NC) switch that is wired in series with the landing gear control circuit. When the control switch is actuated by pulling the cockpit control handle for emergency landing gear extension, it opens the circuit to the electrically actuated hydraulic selector valves, thus blocking hydraulic pressure to the selector valves. When the handle is pushed in, the landing gear control circuit is again closed. This action completes the circuit so that the hydraulic selector valves can again control normal landing gear operation.

In the cross-section view of the control valve in figure 3-11, you will note that all pneumatic pressure lines of this subsystem are manifolded at this unit. Notice also that the valve is spring-loaded to the off or closed position. Pneumatic pressure for this subsystem is connected directly from the priority air flask to the valve pressure chamber. Pressure is held in this upper chamber by the spring-loaded poppet type valve seated on the valve outlet port.

When the valve is actuated by the lever at the bottom, the poppet is unseated and raised against the spring pressure to allow air pressure to enter the emergency gear-down line. Now note that the stem of the poppet valve has a drilled passage through its center that extends into the valve vent chamber. When the cockpit control handle is pushed in for normal hydraulic landing gear operation, this passage connects the valve outlet port to the vent chamber. This provides venting of the pneumatic lines to the gear actuating cylinders and prevents static air pressure from building up when the actuating cylinder shuttle valves are in position for normal (hydraulic) landing gear operation.

The control valve vent chamber incorporates a spring-loaded vent spool. When the cockpit control valve handle is pulled for emergency landing gear extension, the vent spool stem seats against the poppet valve stem. Thus, venting from the valve pressure chamber is sealed at the same time as the poppet valve is unseated at the valve outlet port. When the handle is pushed in and the valve lever is down, the vent spool drops away from the poppet valve stem, thus opening the gear-down line to the vent opening.

PNEUMATIC PRESSURE SWITCH.

Earlier we mentioned that as the LG EMER EXTEND handle is pulled, the emergency down switch opens to remove power from the landing gear circuit beyond this switch. But to do this, a landing gear emergency extend relay solenoid must be actuated. A pneumatic pressure switch, located in the main landing gear emergency extension line (figure 3-11), maintains power to this solenoid until the pneumatic pressure in this switch builds up to 30 to 200 psi. Somewhere in this pressure range, the pressure switch opens and breaks the landing gear circuit to neutralize the landing gear selector valves.

This switch also functions to prevent damage to the secondary hydraulic system reservoir by remaining open until the pneumatic pressure has again dropped to between 30 and 50 psi. It then closes the gear electrical circuit to provide power to open the selector valves for hydraulic operation of the landing gear.

THE PRIORITY VALVE.

The priority valve in the nose gear emergency down line is shown in figure 3-12. Note that it is the same type as the priority valve you learned about in the supply system in Chapter II. The function of this priority valve is to establish the proper sequence of operation between the nose gear and the nose gear door during emergency gear extension.

This valve is spring-loaded to remain seated until pneumatic pressure entering the top of the valve reaches 700 psi pressure. By the time this pressure is reached the nose landing gear door will have been opened, thus permitting a full flow of system pressure to unseat the valve piston and flow to the nose gear actuating cylinder. Note that when pneumatic pressure is relieved from the subsystem, vent air from the nose gear actuating cylinder can unseat the valve poppet and free flow back through the valve for venting at the control valve.

SHUTTLE VALVES.

Shuttle valves are located at each gear and gear door actuating cylinder to control the flow of either hydraulic fluid or air pressure into the cylinders for gear-down operation. No shuttle valves are placed at the gear-up port on the actuating cylinders.

In figure 3-13 you can see a shuttle valve connected to a hydraulic actuator, and two conditions of this shuttle valve. In the main view, the shuttle valve is in its normal condition. In this position, the sliding valve seat is against the pneumatic pressure line so that only hydraulic fluid can move the actuator. When hydraulic pressure is at zero psi and pneumatic pressure is introduced, the valve slides over to block the hydraulic pressure port and to allow pneumatic pressure to move the actuator. *This action is shown in the emergency condition.* In either position, the sliding valve seat seals one line so that pressure of the motivating agent will not contaminate the other line and its system. The sliding valve seat moves only when either air or hydraulic pressure is applied against it.

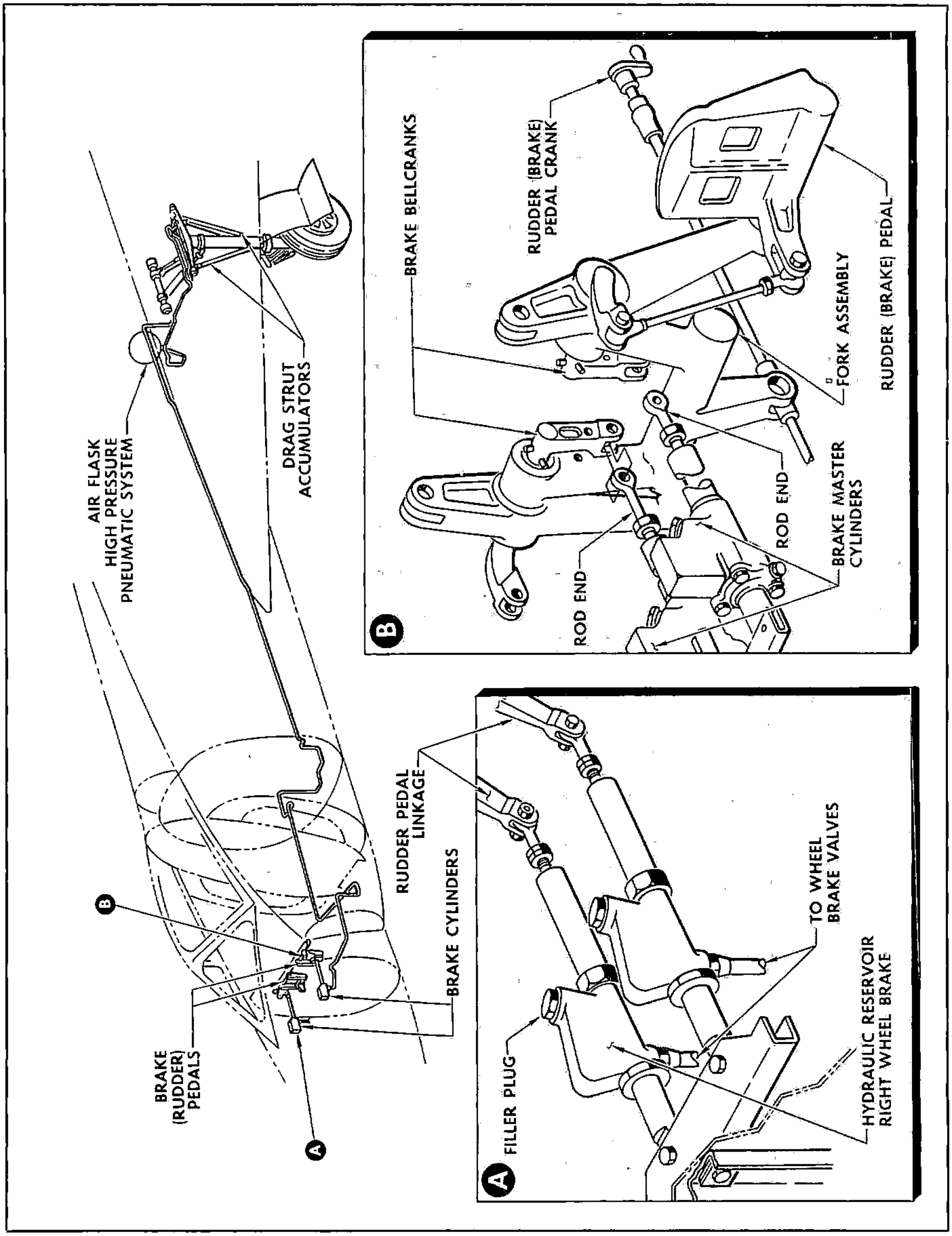


Figure 3-14. Wheel Brake Control System

OPERATIONAL CHECKOUT OF EMERGENCY GEAR EXTENSION.

A checkout of the emergency extension of the landing gear is very important and is necessary at regular intervals. This checkout is performed with the airplane jacked clear of the ground, and with the gears and doors initially in their retracted or closed positions. The high-pressure pneumatic system should be charged to a maximum pressure of about 1000 psi, since this emergency subsystem should operate satisfactorily at a pneumatic pressure of this value.

The crewman in the cockpit should start the test by pulling aft on the emergency gear extend handle. The man and nose gear wheel well doors must open first and then the gears must extend and lock. If this does not happen in a reasonable amount of time, push the emergency gear extend handle in, against the stop. This action bleeds the air pressure from the emergency landing gear system so that you can investigate the trouble. However, the emergency gear extend handle must always be pushed in, even after successful emergency landing gear extension operations, so as to vent all air from the system. Don't forget, the landing gear must be cycled about five times after an emergency gear test to be sure all air is expelled from the gear hydraulic system.

A malfunction of the emergency extend system is usually traced to a faulty pneumatic pressure switch, a faulty priority valve or a leaky shuttle valve in one of the actuators. Just as for most other high-pressure pneumatic system equipment, do not attempt a repair of the component; replace it instead.

WHEEL BRAKE SYSTEM.

The wheel brakes on the F-102A main landing gears are air operated through hydraulic control relay valves. The hydraulic control portion of the wheel brake system is completely independent of the two main hydraulic power systems—the primary and secondary hydraulic systems. Also, there are no connections between the two hydraulic systems and the two pneumatic subsystems which operate the right and left wheel brakes. Each brake system is completely independent of the other.

The wheel brakes on this airplane are operated by toe action of the rudder pedals. These brakes are the disc type in which three rotor discs are turned by the main gear wheels and two stator discs are linked to the brake assembly that is rigidly attached to the landing gear axle. The stator discs are equipped with integral metal-ceramic pressure blocks that produce the braking action, much the same as the lining on your automobile brakes. Only the action of the lining is the same, but not the type of brakes, as you will see later in this section.

HOW THE RUDDER PEDALS OPERATE THE BRAKES.

When the pilot wishes to slow or stop the airplane while it is moving on the ground, he depresses the top of the two rudder pedals with his toes. In figure 3-14 you can see how the pedals are hinged at two pivot points. The linear, or rudder, movements of these pedals pivot about the bell crank attachment to the fork assembly, while angular, or brake, movements pivot about the lower end of the bell crank. To actuate the brake master cylinders the pilot makes angular movements downward on the pedals.

It is possible to actuate the wheel brakes while the two pedals are in any rudder position. The detailed illustration of the brake pedal shows that one wheel brake system can be actuated without actuating the other. Only the angular motion of the pedals pivoting about the pedal hinge point causes the master brake cylinder piston rod to move. This actuation pulls the master cylinder piston rod outward and forces hydraulic fluid to flow through the system to a brake relay valve mounted on each main landing gear.

This hydraulic pressure actuates the brake relay valve to allow high-pressure air to enter the wheel brake for braking actuation. The master brake cylinders for both wheel brake systems are located under and forward of the pilot's instrument panel. In figure 3-15 (detail A) you can see that these master brake cylinders have removable filler plugs. The filler plugs provide a means of servicing the master cylinders with hydraulic fluid, Specification MIL-O-5606. Two small access doors, one on each side of the windshield, provide access to the master brake cylinders for servicing.

HOW THE WHEEL BRAKES ARE ACTUATED.

The wheel brakes are actuated by toe action of the rudder pedals applying hydraulic pressure on the relay valves as shown in figure 3-15. Hydraulic pressure entering the relay valve pushes the piston down against its spring. This piston in turn unseats the poppet-type valve and allows filtered air pressure from the drag brace accumulator to pass through the relay to the piston on the wheel brake. As you can see in the schematic, the brake stator and rotor discs are forced to contact each other when the pressurized air enters and actuates the wheel brake piston. When the brake pedals are released, hydraulic pressure which is holding the relay valves open is also released. Thus, the relay valves return to their spring-loaded, normally off position and vent the air pressure from the brake piston.

Pneumatic pressure for normal brake operation is about 1330 psi maximum. This air pressure, for the brake operation only, is initially stored at 3000 psi in the hollow section of each main landing gear drag brace accumulator. The source of this pressure is

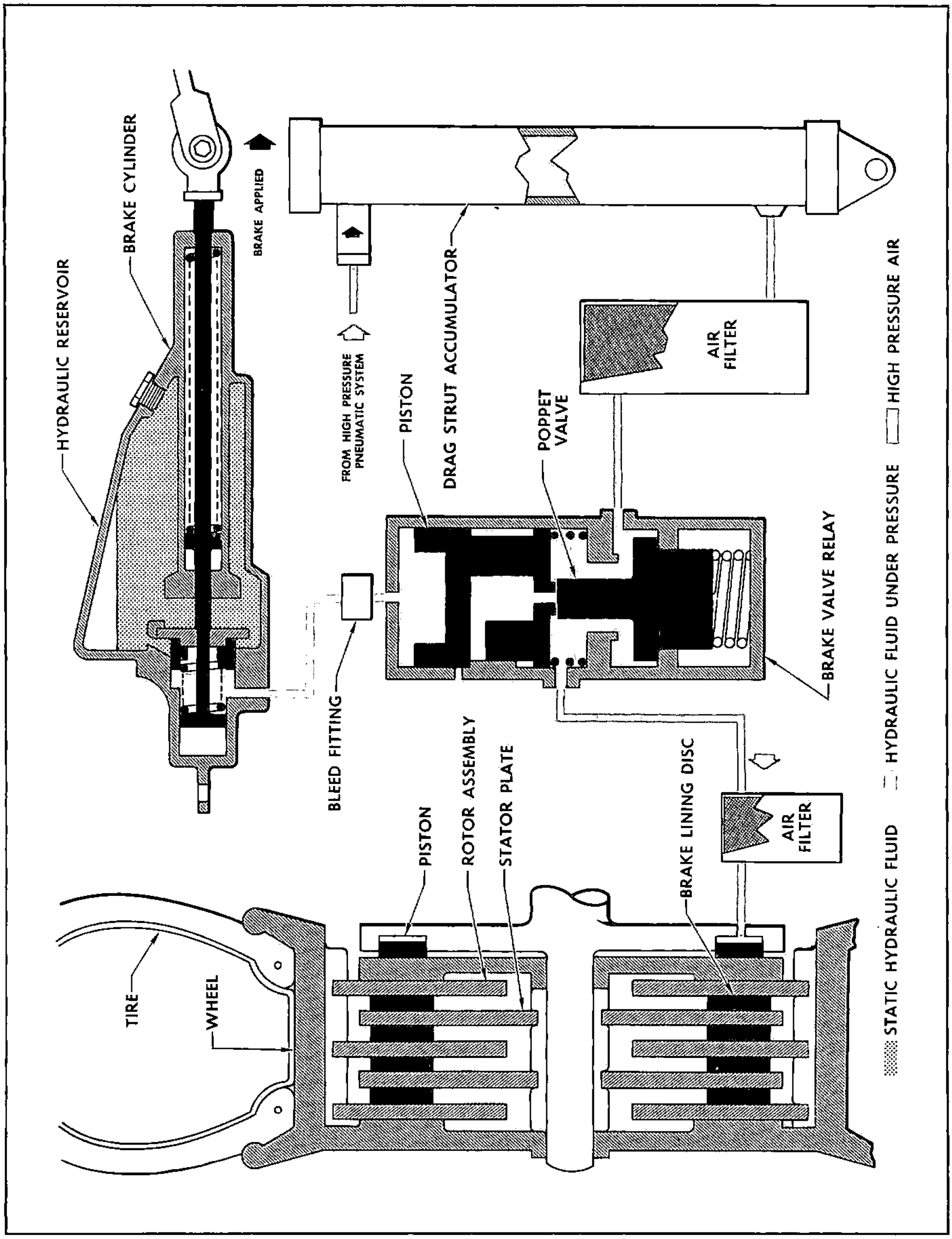


Figure 3-15. Wheel Brake System Schematic

through the high-pressure pneumatic supply system, and is ground charged at the same time as the pneumatic supply system. The drag brace accumulators are isolated from the main supply system by check valves which assure an adequate supply of air pressure, even if the pneumatic system supply is depleted.

WHEEL BRAKE SYSTEM COMPONENTS.

All components in one wheel brake system are duplicated in the other brake system. The wheel brakes operate independently of each other, so each system has its own operating components. The following paragraphs describe these components.

Hydraulic Relay Valves.

One brake relay valve is mounted on each main landing gear strut, and each valve is completely independent of the other. These relay valves are piston-type units that are hydraulically operated to admit high-pressure pneumatic system air to the wheel brake assembly. Each valve incorporates a vent, which is opened to vent air pressure from the wheel brakes after the pilot releases the brake pedal. When the brake pedals are released, the relay valve returns to its normal, static, brake-off position. Seals are incorporated in the relay valve to prevent hydraulic fluid and pneumatic pressure intermixing.

In figure 3-15, this relay valve is shown in its actuated position wherein it is directing air pressure to the wheel brake, as you learned earlier in this brake system discussion. Now when the pilot releases the pressure on the brake pedals, all hydraulic pressure is removed from the relay valve piston. This spring-loaded piston then moves up to its upper case stop. In this position the hollow passage in the piston is exposed to the vent port.

At the same time that the piston moves up, the spring under the poppet valve rushes the poppet up and closes the air inlet port to the relay valve. With both the poppet and piston in their up positions, the air line to the wheel brake is then connected to vent through the passage in the piston. This releases all pressure at the brakes and the wheels are free to turn.

During maintenance or replacement of the relay valves, care must be taken that the relays are correctly installed so as to prevent damage to the brake system. Be sure that both air lines and the hydraulic line are connected to their correct ports on the valve. The bleed fitting at the relay valve provides a means of bleeding air from the brake hydraulic system. This hydraulic system is bled by depressing the brake pedal, then opening the bleed fitting until all air is expelled. Don't forget to close the bleed valve before releasing brake pedal pressure. Also, don't forget to replenish the brake reservoirs after bleeding.

Brake System Air Filters.

Two air filters are installed in each wheel brake system to prevent foreign material from entering the brake relay valve and the brake disc mechanism and causing damage to the equipment. In figure 3-15, you can see that one of the filters is located between the drag brace accumulator and the brake relay valve. The other filter is located between the brake relay valve and the wheel brake piston.

The importance of clean pneumatic pressure is obvious by the installation of these filters; therefore you must quite often remove and clean these filter elements. Before removing for cleaning or replacement, however, you must relieve all air pressure in both the pneumatic system air flasks and in the drag brace accumulators. The filters are cleaned by using a suitable solvent as an air pressure spray. The elements are then dried with filtered dry air, installed in their housings, and replaced in the brake system air lines.

Wheel Brake Air Pressure Accumulators.

The hollow portion of each main landing gear drag brace is utilized as wheel brake air pressure accumulators. On earlier airplanes there are two drag braces at each main gear, as shown in figure 3-16. On later airplanes there is only one drag brace at each main landing gear.

On the earlier airplanes the two drag brace accumulators are manifolded together, thus serving as a single wheel brake air pressure accumulator. Each of these assemblies is provided with a check valve at the air inlet port, a relief valve, a bleed fitting, and a drain fitting. The check valves prevent the accumulator assembly air pressure from venting back into the main pneumatic system air flasks.

Naturally, it is very important to check these valves for correct installation. Be sure that the stenciled arrow on the valve points toward the accumulator assembly. A relief valve is installed in each accumulator assembly to prevent overcharging. This valve remains seated up to 3200 psi of pressure, but vents to full flow at 3800 psi of pressure.

Removable bleed fittings and condensation drain fittings are located on the lower end of each accumulator assembly as shown in detail A. These wheel brake air pressure accumulators must be bled and drained at specified intervals and you must be mindful when accomplishing this that you are working with high-pressure air.

To perform this bleeding and draining operation, first slowly relieve all air pressure from the main pneumatic system priority air flask; then loosen the wheel brake accumulator bleeder plug just enough to permit air to escape slowly. After all pressure is released,

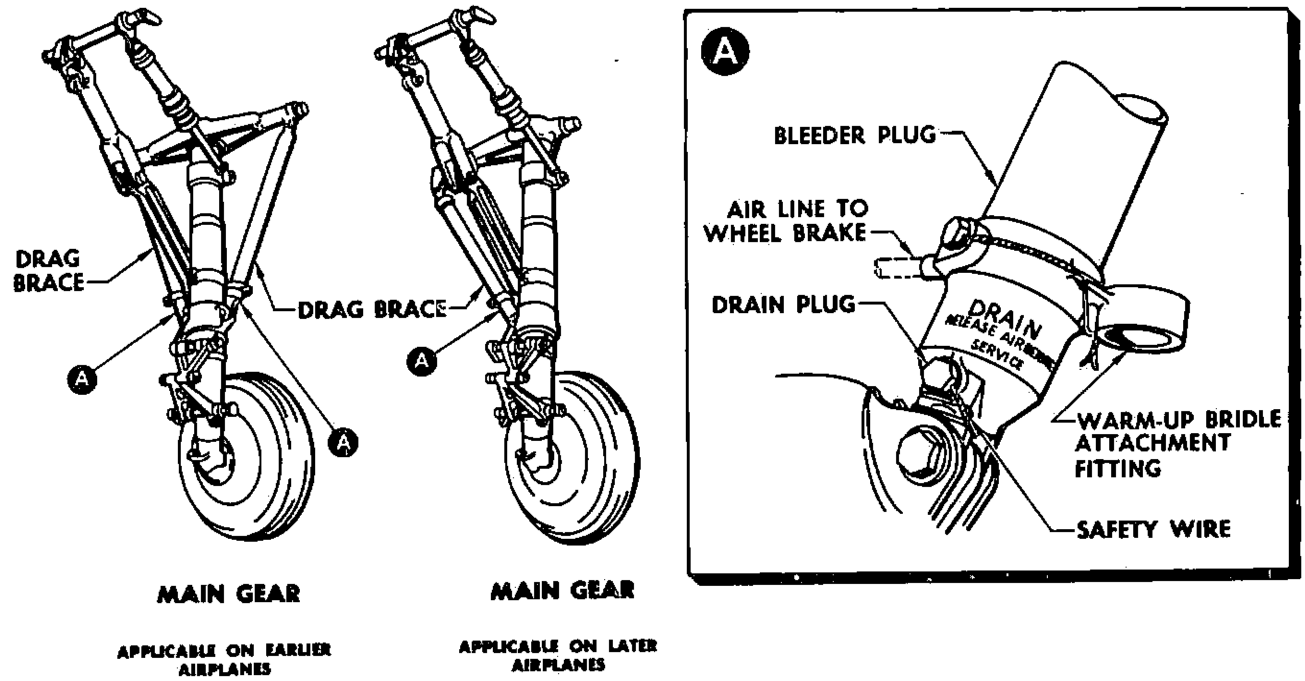


Figure 3-16. Brake System Accumulators

remove the drain plug and drain the accumulator condensation. After completing this maintenance function, reinstall, tighten, and safety the bleeder and drain plugs.

Charging Wheel Brake Accumulators.

The wheel brake air pressure accumulators are automatically charged with air whenever the high-pressure pneumatic system is ground charged. The check valves at the drag brace accumulator assemblies allow air pressure to enter the accumulators, and prevent depletion of this air pressure by the high-pressure pneumatic system.

The Wheel Brakes.

In figure 3-17, note that the two wheel brake assemblies are disc-type brakes in which the alternate discs are made of a cerametallic material. The wheel brake housing and disc assembly is bolted to its main landing gear axle. Six adjustment screws are provided around the perimeter of the housing for adjustment of the running clearance between the brake discs. Inside each brake housing is a ring-type piston which extends pneumatically to compress the discs and provide braking action. The six screws are adjusted to give a clearance for easy turning, and to provide even wear on the discs when the brakes are actuated.

The two stator discs are linked rigidly to the brake housing and do not turn with the wheel. On the other hand, the three rotor disc assemblies are keyed to the wheel and turn as the wheel rolls. The braking action takes place when pneumatic pressure at the brake piston forces the rotor discs against the stationary friction block of the stator discs. Detailed adjustment instructions and tolerances are given in the applicable Maintenance Manual, T.O. 1F-102A-2-8.

DRAG CHUTE PNEUMATIC SUBSYSTEM.

A drag chute is used on the F-102A to slow the airplane speed after landing, and thus shorten its landing roll. The chute is similar to the canopy-type parachutes used by airmen when jumping from a disabled airplane.

Referring to figure 3-18, you can see that the drag chute pneumatic system components are located in four general areas of the airplane: the aft armament bay, the cockpit, the nose wheel well, and the vertical fin island. Note that the main pneumatic components of the drag chute system consist of a power source at the high-pressure pneumatic system air flasks, a cockpit control handle, a selector valve assembly, and the pneumatic actuating cylinder which actuates the drag chute jaw and triggering mechanism.

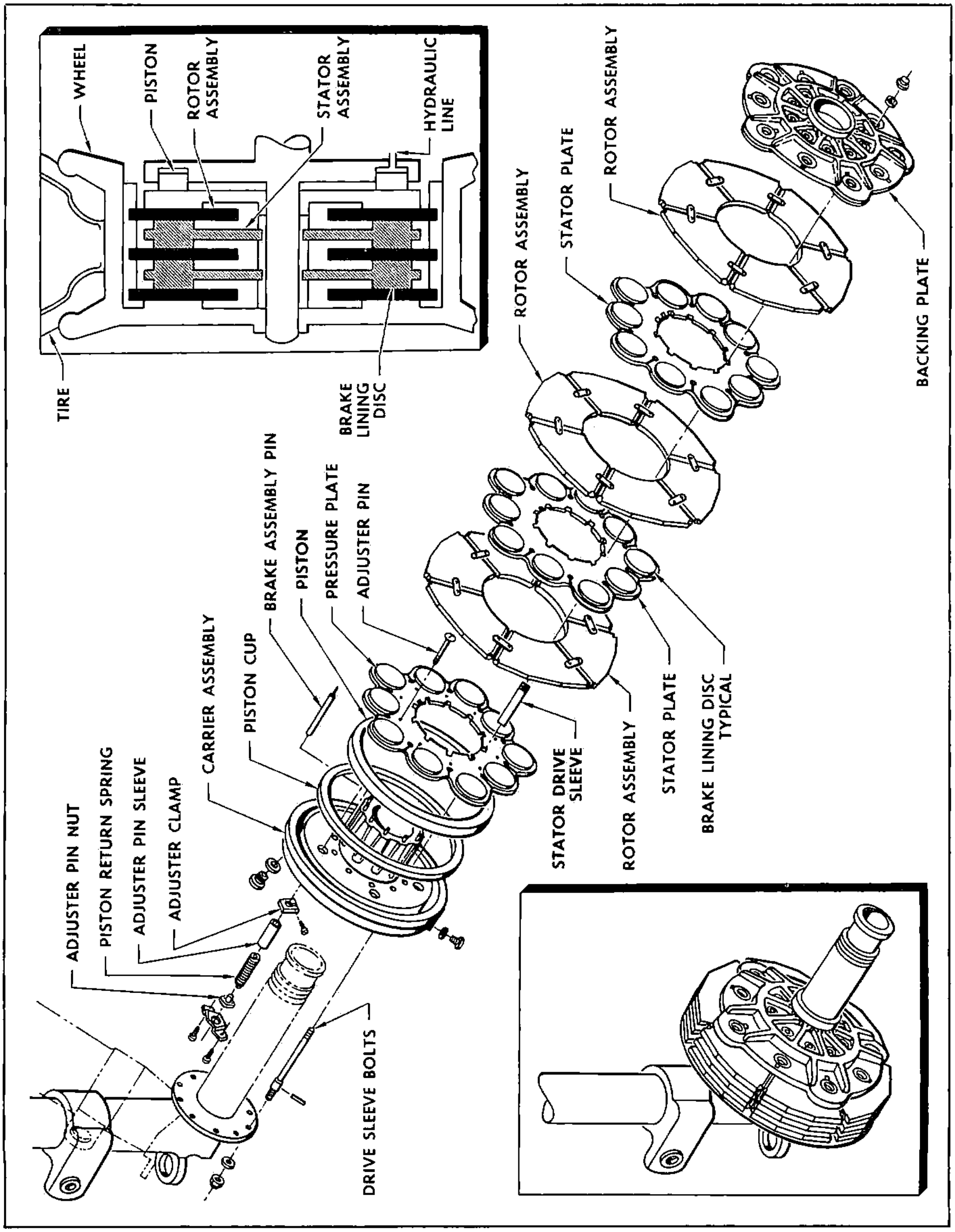


Figure 3-17. Wheel Brake Assembly

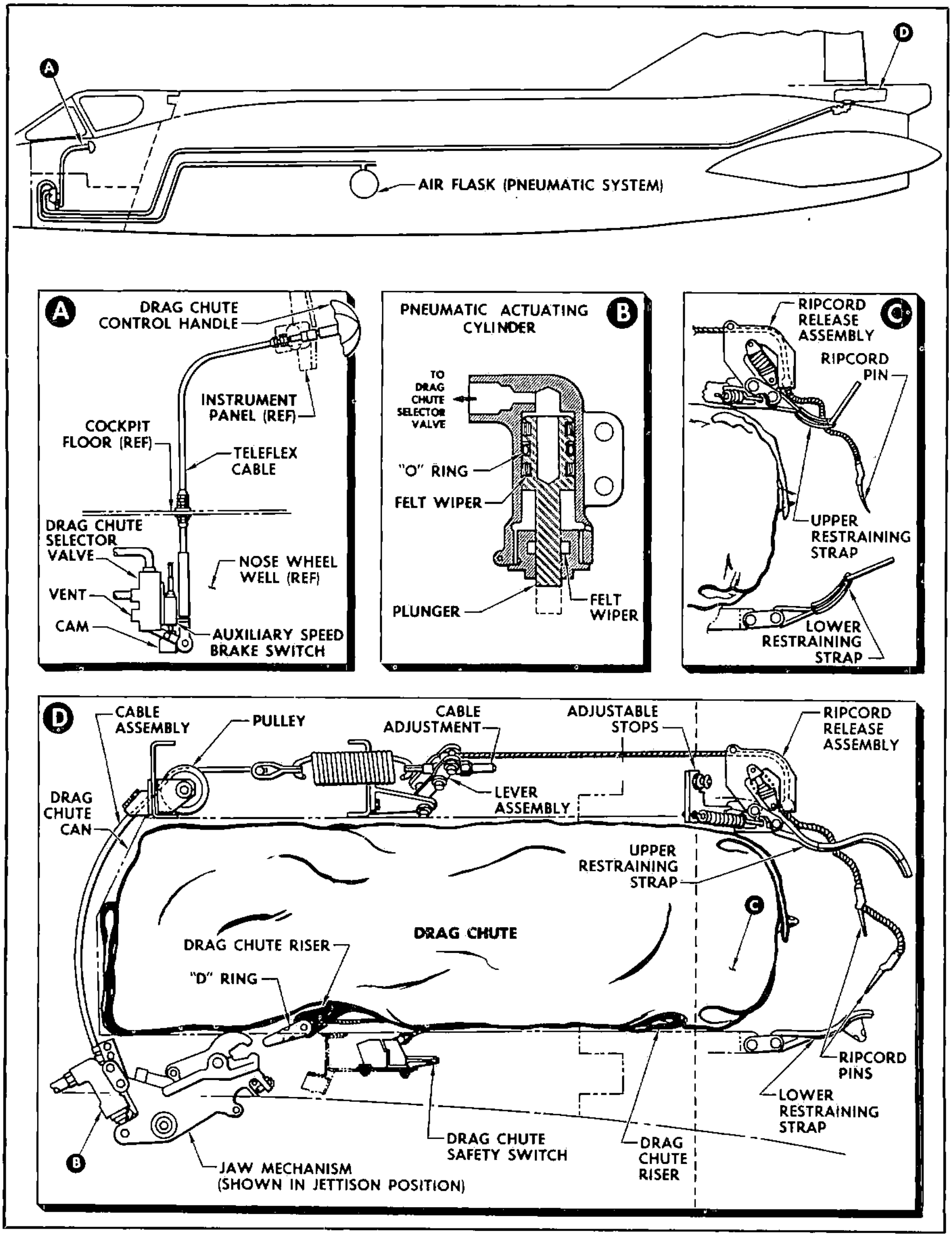


Figure 3-18. Drag Chute Pneumatic Subsystem

The source of pneumatic pressure is stored in the high-pressure air flasks located in the aft armament bays. The drag chute control handle, located in the cockpit, is pulled to deploy the chute and is pushed back in for chute jettisoning. The control handle is connected by a Teleflex cable to the pneumatic selector valve lever located in the forward left side of the nose wheel well. The lever in turn actuates the selector valve assembly which consists of an auxiliary speed brake switch mounted on the drag chute selector valve.

The lever has a mounted cam that moves upward to actuate the drag chute auxiliary switch. The switch controls the opening of the speed brake doors when the drag chute is to be deployed, and later operates in conjunction with the drag chute time-delay switch to control the closing sequence of the doors. The latter operation prevents the doors from closing too soon and interfering with the drag chute risers (see figure 3-18).

As you can see in detail C, the drag chute pneumatic actuating cylinder actuates downward when air pressure is admitted by the selector valve. The cylinder is attached to a jaw mechanism (shown in detail B) that holds the drag chute riser "D" ring until the cockpit control handle is pushed in for jettisoning of the chute. A cable assembly connects to the actuating cylinder at the point where the cylinder is attached to a lever on the jaw mechanism. The cable, when pulled by the cylinder operation, actuates the drag chute triggering mechanism and pulls the ripcord release assembly to deploy the drag chute.

DRAG CHUTE PNEUMATIC SELECTOR VALVE.

The drag chute pneumatic selector valve in this subsystem is identical to the emergency landing gear and ram air turbine extension control valves. Figure 3-19 shows a cutaway of the valve and indicates its operating principle. Basically, the valve components consist of a poppet, spring-loaded to the closed position. The poppet isolates pneumatic system pressure from the valve pressure chamber and outlet port.

The selector valve also incorporates a spring-loaded actuating plug, which extends from a manually operated lever outside the valve into the vent chamber. Note that the poppet stem in the outlet chamber has a vent hole drilled in its side which connects with a drilled passage up through the lower part of the poppet stem. Notice also that the lower part of the stem extends into the vent chamber.

You can see now that when the cockpit control handle is pulled, the direct-line connected priority air flask pneumatic pressure is relieved from the selector valve pressure chamber which in turn operates the drag chute actuating cylinder. When the control handle is pushed in, the subsystem air pressure is then vented back through the selector valve vent chamber to atmosphere.

Operation of the Selector Valve.

The operation of the selector valve accomplishes two things: it admits pneumatic system pressure from the priority air flask to the drag chute actuating cylinder, and it vents static pressure in the drag chute cylinder to the atmosphere. When the pilot pulls the drag chute control handle, the lever attached to the bottom of the selector valve pivots up, as shown in figure 3-19, to move the actuating plug upward and contact the end of the poppet stem. This action forces the poppet off its seat to admit pressure to the drag chute actuating cylinder. At the same time, it seals the passageway through the stem into the vent chamber.

In figure 3-18 you can see the result of this flow of pressurized air through the valve. The pressure actuates the drag chute actuating cylinder to pull the chute release mechanism, and at the same time causes the chute retaining jaws to close on the chute "D" ring. This action also cocks the deployment trigger mechanism. When the speed brake doors have opened and the chute has deployed and accomplished its purpose in slowing the airplane, the pilot then pushes the control handle into its stop to jettison the chute.

In doing this, the lever drops down, as shown in the left view (figure 3-18), and releases the actuating plug in the valve to its spring-loaded position away from the poppet stem. The poppet then reseats and shuts off pneumatic system pressure to the drag chute cylinder. The static pressure in the drag chute cylinder then vents through the hole in the poppet stem to atmosphere. With pressure released from the drag chute cylinder, the spring-loaded jaws which hold the "D" ring of the chute risers are free to open. The "D" ring is then jerked from the jaws by airflow drag on the chute and the chute is jettisoned.

DRAG CHUTE ACTUATING CYLINDER.

The drag chute pneumatic actuating cylinder (figure 3-18, detail C) is a simple cylinder and piston. Air entering the top of the cylinder forces the piston to extend. One unusual feature of this cylinder, however, is that the piston shaft is not attached (bolted or pinned) to the object it actuates. Rather, the smooth end of the piston shaft bears on a roller that is bolted to a lever on the jaw mechanism. When the selector valve admits air pressure to the cylinder, the piston plunger thrusts out to depress a lever in the jaw mechanism.

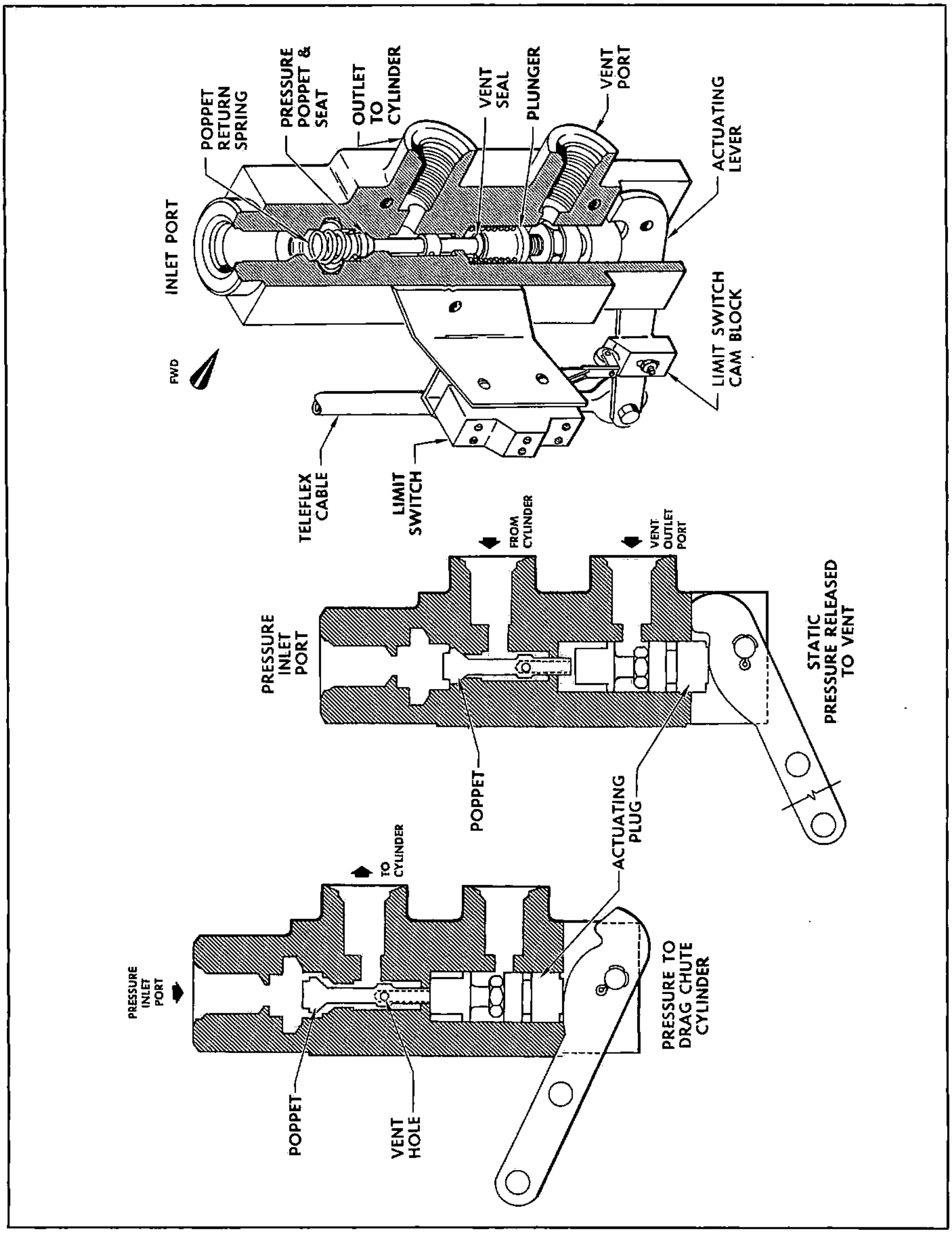


Figure 3-19. Drag Chute Pneumatic Selector Valve

This overcomes a spring in the jaw mechanism, and causes the jaws to close tightly on the "D" ring. One end of the cylinder is vented to atmosphere. This vent allows ambient air to escape when the cylinder is actuated, or allows the entrance of ambient air when the other end of the cylinder is vented through the selector valve, as just described. Action of the spring in the jaw mechanism returns the cylinder piston to its normal position when air pressure is removed by the closing of the drag chute selector valve.

DRAG CHUTE JAW MECHANISM.

The jaw mechanism controls the "D" ring in three phases of operation: in-flight, chute braking, and chute jettison positions. In figure 3-20 you can see the three positions the jaw mechanism assumes in its different phases. In the in-flight position the jaws are partially open. In this position, they retain the "D" ring in position for use in deployment, but permit the ring to be pulled out if the chute is accidentally deployed while the airplane is in flight.

The next view (the chute braking position) shows the jaws closed tightly over the "D" ring. Note that in this position the linkage bars attached to the spring capsule are in an almost straight line with each other (actually slightly overcenter). This locks the jaws in the closed position. Actuation of the air cylinder places the jaws in this position for chute deployment. Next, you can see the jaws in the wide open position for chute jettisoning.

How the Drag Chute Cylinder Operates the Jaw Mechanism.

Referring to figure 3-20, note that in the in-flight position the actuating cylinder is in the retracted position (cylinder actuating shaft or plunger completely retracted), and that the linkage bars and pin at the spring capsule slot are in neutral position. This is the position the jaw mechanism will always assume unless it is either actuated to the full open position or to the completely closed position.

When the actuating cylinder shaft (plunger) is in the fully extended position, as in the chute-braking view, the two lever-roller assemblies are forced down by the plunger extension, thus moving the linkage to the left end of the spring capsule slot. The jaws are then tightly locked over the "D" ring. This action also pulls a cable that "cocks" the ripcord release assembly on top of the drag chute housing "can" by stretching a spring which deploys the chute as soon as the speed brake doors open past a minimum of 30°.

When the pilot pushes the control lever in, the actuating cylinder shaft retracts. This action frees the spring capsule to return the linkage bars to the center position in the capsule slot. Naturally, the chute is deployed at this time and its risers are pulling on the "D" ring, so that the "D" ring is jerked out of the jaws because of the airload on the drag chute. In jerking out of the jaws, the ring must momentarily spring the jaws fully open so that the "D" ring can pull past them.
Springing the jaws to full open carries the linkage bars and pin to the right end of the spring capsule slot. The jaws are thereby spring-loaded again by a second spring in the capsule, and the instant the ring is free they are returned to the in-flight position. In that position, they are ready for the next "D" ring insertion when a repacked chute is again installed.

In event the jaws remain locked over the "D" ring, and should the chute be accidentally deployed, a shear pin breaks to release the chute. The shear pin is incorporated as the axle pin of the shackle which couples the "D" ring to the chute risers, and it will break and jettison the chute if the chute is deployed at speeds in excess of 180 knots indicated airspeed.

MAINTENANCE.

Very little maintenance is required for the drag chute pneumatic subsystem. However, there may be times when an air leak is discovered in the pneumatic selector valve or in the actuating cylinder. Should this be the case, the faulty equipment must be replaced, and no attempt must be made to repair the faulty unit in flight-line maintenance.

RAM AIR TURBINE EXTENSION SYSTEM.

In the Hydraulic System Training Supplement of this series, you will learn about the F-102A emergency hydraulic system. This system supplies emergency hydraulic pressure to the flight controls, should either the hydraulic pumps or the engine malfunction and stop providing normal hydraulic system pressure. If failure of normal hydraulic system pressure is due to engine stoppage, the pilot can either maintain a speed sufficient to windmill the engine and get normal hydraulic pressure, or he can extend the emergency hydraulic system ram air turbine into the airstream. If lack of hydraulic system pressure is due to pump malfunctioning, he must extend the ram air turbine.

The ram air turbine drives an attached pump to produce emergency hydraulic pressure for limited flight control operation. This system is mechanically controlled and pneumatically actuated by the high-pressure pneumatic system. It is a simple system and consists of a control cable, a control valve, and an actuating cylinder. This system is shown in a schematic form in figure 3-21.

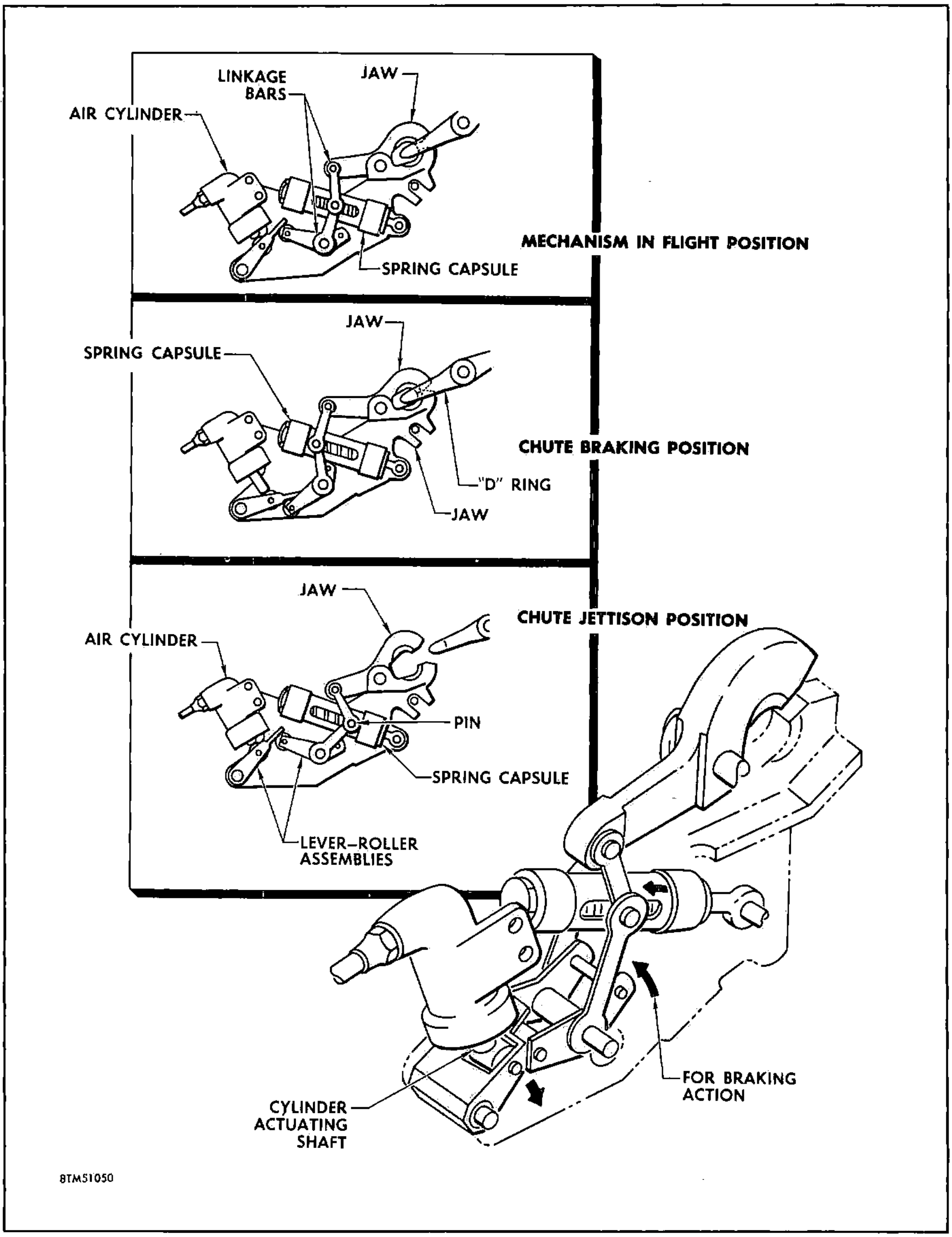


Figure 3-20. Drag Chute Jaw Mechanism

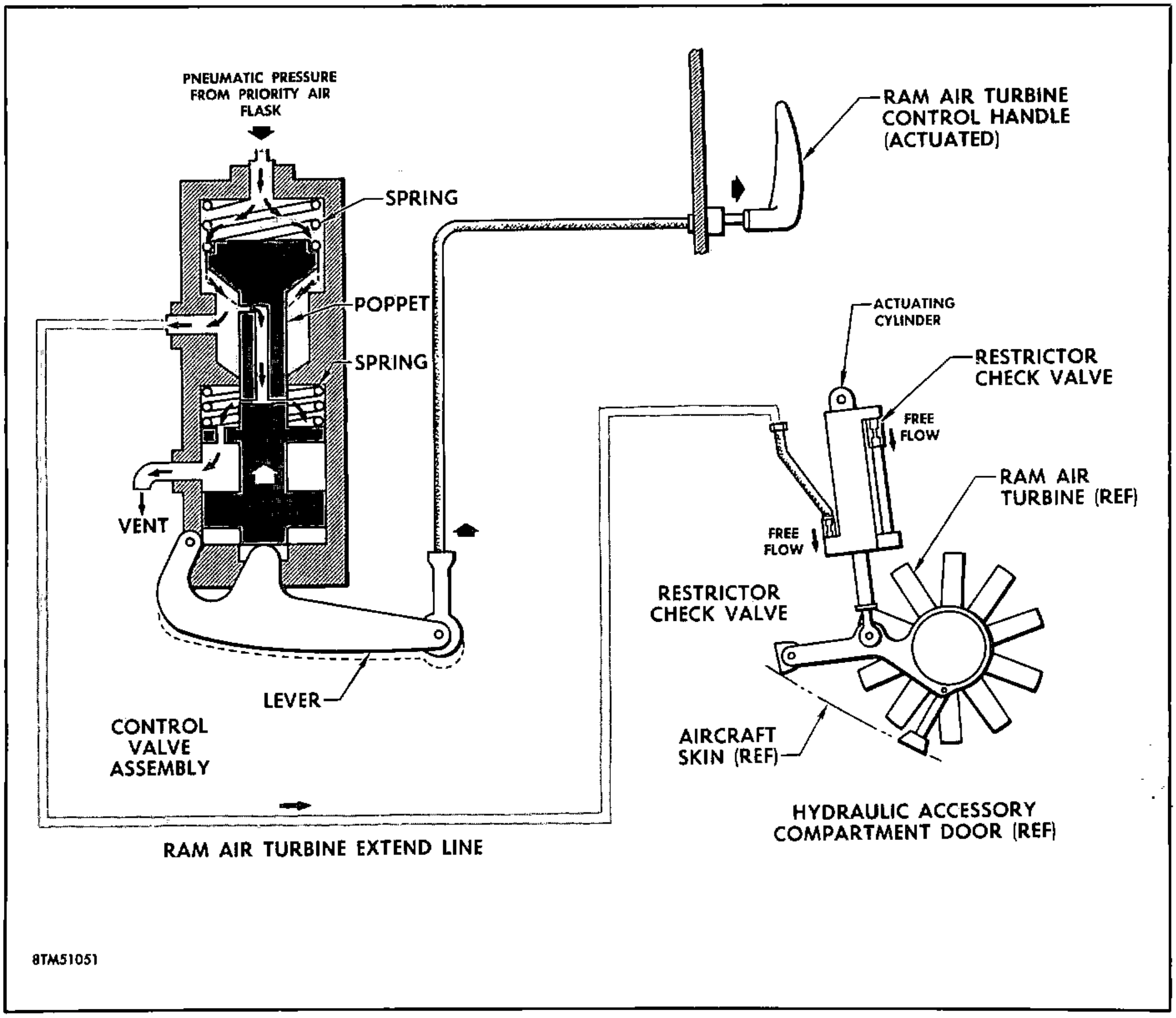


Figure 3-21. Ram Air Turbine Extension System Schematic

Extension of the ram air turbine is limited to certain speeds, listed in the pilot's Flight Handbook. If he exceeds the maximum speed for turbine extension, he will damage the entire assembly. If he is under the minimum speed the turbine will not turn fast enough to produce sufficient hydraulic pressure.

HOW THE EXTENSION SYSTEM OPERATES.

In figure 3-21 you see that the system is put into operation by pulling the control handle in the cockpit. This control handle is installed on the aft side of the power lever quadrant and is placarded EMER HYDRAULIC POWER PULL. The control handle is connected by a Teleflex cable to a pneumatic control valve located in the nose wheel well. (The valve and control handle are similar to those used in the landing gear emergency system, figure 3-11.) With the handle pulled, air pressure from the high-pressure pneumatic system priority air flask is routed by the control valve to the ram air turbine actuating cylinder.

Note that the restrictor check valve at the cylinder inlet provides a free flow of air pressure into the retracted cylinder. The snubbing (lower) side of the piston is connected to the extension (upper) side through another restrictor, or check valve. Air flow through this valve is restricted, thereby causing sufficient snubbing pressure to be built up in the lower end of the cylinder before enough pressure is built up on the upper side of the piston to unlock and extend the cylinder. Since air loads in flight tend to pull the ram air turbine out of the airplane, the combination of the two restrictor-check valves limits the speed of extending the assembly.

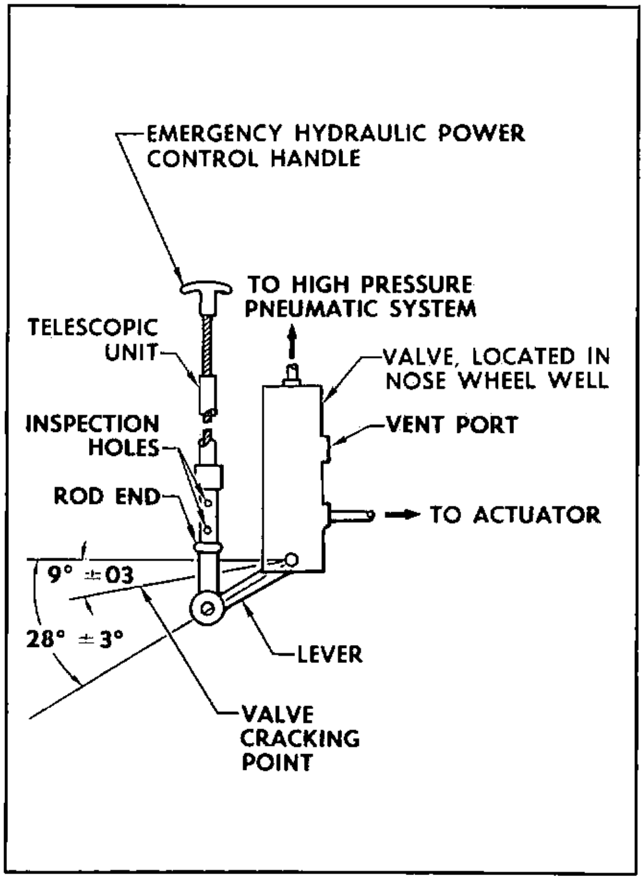


Figure 3-22. Control Valve Adjustment

Returning the control handle in the cockpit to its original position vents the air in the actuating cylinder back through the control valve to atmosphere. An external latch release on the cylinder may then be operated manually so that the unit and attached door can be pushed back into and flush with the fuselage. The turbine can be pushed back only on the ground, and when once deployed the turbine stays out until the aircraft has been landed.

Control Valve.

The ram air turbine extension control valve is almost identical to the emergency landing gear system control valve that you learned about earlier in this chapter. The only difference is that an externally mounted electrical control switch is not attached to this valve.

When the cockpit control handle is pulled and the valve lever is raised, air pressure from the high-pressure pneumatic system priority air flask is routed through the control valve pressure chamber and into the extend line to the turbine actuating cylinder. When the cockpit control handle is pushed back in for manual retraction of the ram air turbine, air pressure in the turbine extension subsystem is then vented back through the system line to the valve vent chamber and then to atmosphere.

Maintenance of the Actuating Mechanism.

Just as certain precautions and adjustments apply to other high-pressure pneumatic system equipment, you also have a few precautions and adjustments to consider as you perform maintenance on the ram air turbine extension system. First, you are again reminded to respect the snap-action results of pneumatically operated equipment. In this respect, make sure that all personnel is cleared from the ram air turbine door area before extending it. In addition, we repeat that personal contact with high-pressure air is extremely dangerous.

In figure 3-22, note the adjustment that must be made at the valve lever for correct operation of the control valve. The rod end fitting of the Teleflex cable must be adjusted so that the lever is 28° (±3°) below horizontal with the control handle at its full *in* position, and with Teleflex cable and rod end secure.

To properly fit the ram air turbine inside the fuselage and to insure the proper angle when the unit is extended, certain adjustments must be checked. As shown in figure 3-23, the fully extended actuating cylinder should measure 25.4 inches from the center of the cylinder attach bolt to the center of the bolt attachments at each end of the cylinder on the extended piston rod.

The small adjustable rod between the turbine and the door correctly positions the turbine at 3¾ inches to prevent it from striking other units in the hydraulic accessories compartment when the door is closed. It is also very important to check that the restrictor check valves at the actuating cylinder are installed with their stencilled arrows pointing in the direction shown at each end of the actuating cylinder. Should one of these valves be reversed, the door would either extend too fast or too slow, depending on which valve was reversed.

To close the ram air turbine door, position a crewman in the cockpit to push the control handle in, so as to vent the air pressure in the actuating cylinder line back through the control valve. Then release the actuating cylinder down-lock lever, as shown in detail "A," by pushing lever toward the cylinder. The door is then ready to close manually. After closing, check that the door fits flush with the airplane skin, and that the gap between the door and skin meets the tolerances outlined in the F-102A Technical Order Maintenance Manual.

RUDDER FEEL SYSTEM.

In this airplane, airloads on the control surfaces are not transmitted back to the pilot, since the hydraulic portion of the flight control system is irreversible. Consequently, an artificial feel system is required to

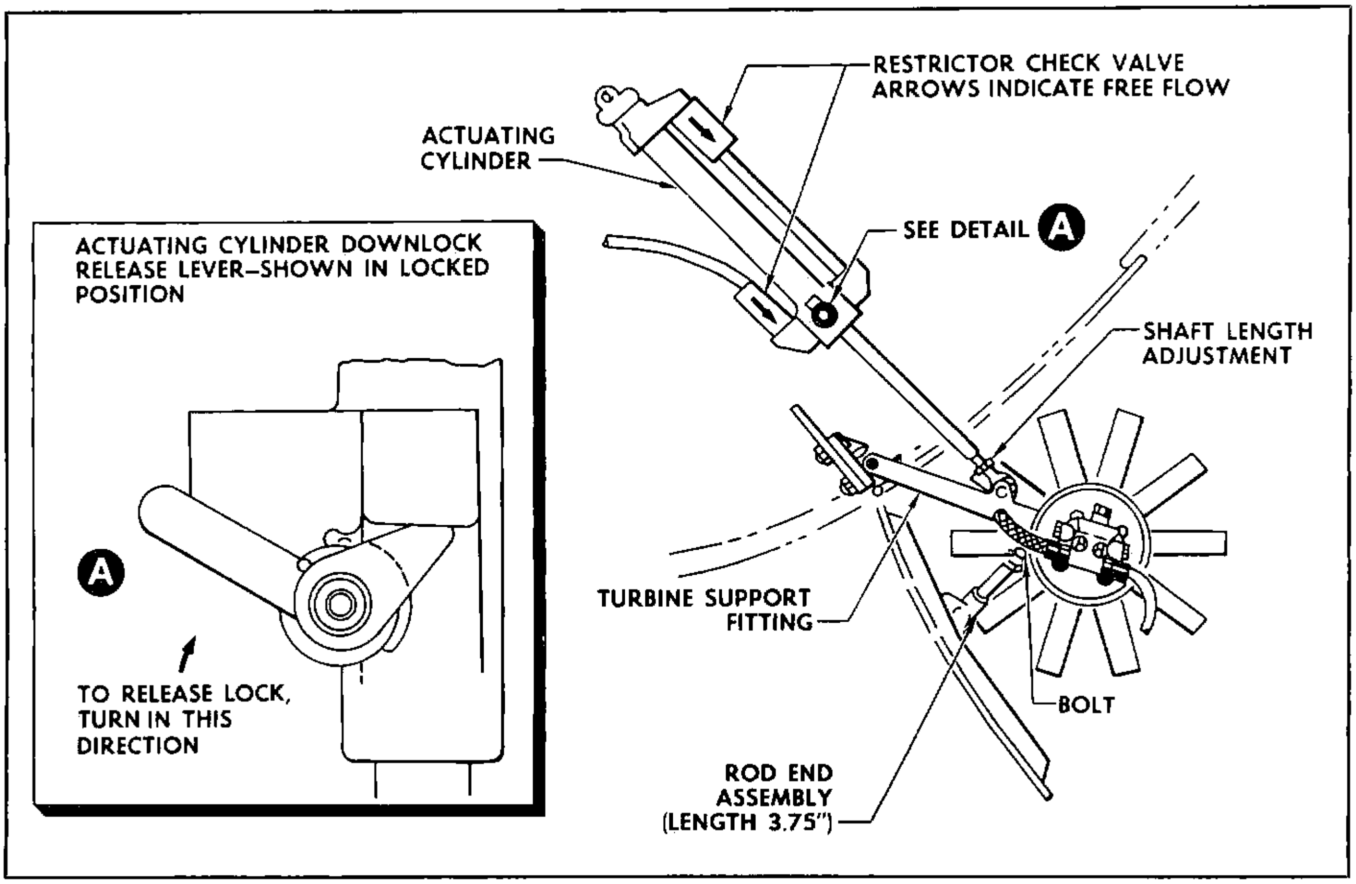


Figure 3-23. Ram Air Turbine Assembly Adjustment

simulate airload conditions to the pilot. The function of the rudder feel system is to provide the pilot with artificial feel forces at the rudder pedals. The feel force varies resistance in moving the rudder pedals proportional to the air speed of the airplane.

In figure 3-24 you can see that the power source for the rudder feel system is obtained from the regulated air pressure supply of the high-pressure pneumatic system 1100 psi pressure regulator and relief valve. This constantly regulated air pressure is routed to a variable air pressure regulator that compares this regulated air pressure to ram air pressure entering the "Q" intake.

This regulator governs the amount of pressure applied to the rudder feel cylinder. As the air pressure in the cylinder varies, the force on its piston varies. The force is then felt by the pilot as resistance to movement of the rudder pedals.

In this schematic, note that a check valve is installed at the pneumatic subsystem inlet line. The purpose of this check valve is to isolate this subsystem from the rest of the high-pressure pneumatic system. Thus, some rudder feel force is always maintained in the event pneumatic system pressure is low or depleted.

VARIABLE AIR PRESSURE REGULATOR.

The variable air pressure regulator is located in the lower forward portion of the vertical stabilizer. (See figure 3-1.) The regulator consists of a spring-loaded diaphragm and a pressure regulator. The function of the regulator is to compare "Q" (ram air) pressure air with high-pressure air and regulate the amount of high-pressure air to the rudder feel cylinder. This air pressure to the rudder feel cylinder is proportional to the speed of the airplane. When airplane speed is decreased, some of the high-pressure air is vented to the atmosphere through the regulator so that the pressure in the feel cylinder corresponds to the reduced speed.

RUDDER FEEL CYLINDER.

The rudder feel cylinder, actuated by the high-pressure pneumatic system regulated air pressure from the variable air pressure regulator, is a simple cylinder and piston assembly. This cylinder is situated in the vertical fin just above the fuselage. As shown in figure 3-24, the feel cylinder receives regulated air pressure from the high-pressure pneumatic system. The air pressure in the cylinder varies in proportion to the "Q" (ram) pressure, which is determined by airspeed and altitude.

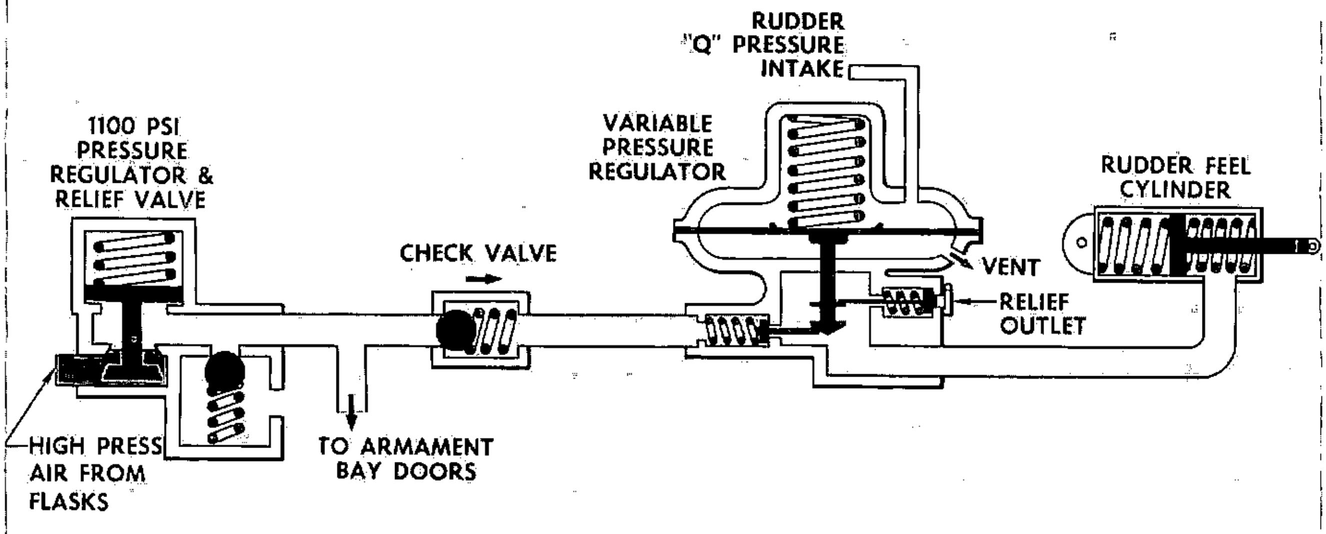


Figure 3-24. Rudder Feel System Schematic

Note that the force on the piston in the rudder feel cylinder will also be proportional to the amount of air pressure in the feel cylinder. The force on the piston is transmitted directly to the rudder control linkage and imposes a load on the rudder controls. This load is felt by the pilot as resistance movement of the rudder pedals. The piston inside the rudder feel cylinder is mounted so that movement of the rudder pedals in either direction from the neutral position works against the force inside the feel cylinder.

MAINTENANCE OF THE SYSTEM.

If low rudder pedal forces are experienced in the flight control system, the most likely cause is a leak in the "Q" pressure system. Another cause of low rudder pedal forces could be the malfunctioning of the variable air pressure regulator, or the check valve in the high-pressure pneumatic system line being reversed. A test for checking this system is outlined in the applicable F-102A Maintenance Handbook, T.O. 1F-102A-2-7. Checks must also be conducted whenever new equipment is installed or when the system has been disconnected and reconnected.

CANOPY COUNTERBALANCE AND CINCHDOWN SYSTEM.

Normal canopy operation is accomplished manually, with the assistance of the combination double-piston, pneumatic-ballistic type cylinder. The pneumatic function of the cylinder serves to counterbalance the canopy for normal raising and lowering, and then it applies a cinchdown force on the canopy during the last several inches at closing. This cinchdown force compresses the canopy seal to provide an effective weather and pressure seal.

The pneumatic action of the cylinder is controlled by a spool-type selector valve that is manually actuated by a crank assembly. An external button is installed on the left side of the airplane to open the canopy from the outside. When this button is depressed, it returns the selector valve to the counterbalance position and allows the canopy to be raised from the outside.

As you can see in figure 3-25, the high-pressure pneumatic system air pressure entering the canopy cylinder system is reduced from its normal system pressure of 3000 psi to 1500 psi by the combination pressure regulator and relief valve. (Also see figure 3-1.) Incidentally, this is the same regulator that provides air pressure to the armament displacement actuating cylinders. See figure 2-7.

After the air passes through this regulator, it then passes through a check valve and directly into the rod end of the canopy actuating cylinder. This end of the cylinder is pointing up in the airplane, and is always pressurized. Now notice that another line extends from the cinchdown line to the canopy selector valve, then through the valve to the other end of the actuating cylinder (counterbalance line). Also note the vent line at the right end of the selector valve. When the canopy is open, the selector valve is in a position that allows air to pass through it and into the counterbalance line.

Therefore, both ends of the cylinder are pressurized, and this action provides the counterbalancing force that holds the canopy open. When the canopy is closed, the selector valve is positioned so that it stops air from entering the counterbalance line to the bottom end of the cylinder. In this position the valve also vents the static air from the counterbalancing end.

The always pressurized section of the cylinder (left end in the illustration) then has no resistance from air pressure on the other end. Thus, the pressure in the upper (left) end pulls down on the canopy as a cinchdown force. A second piston in the lower (right) end of the cylinder operates only when its ballistics cartridge is fired for the jettisoning of the canopy.

On later airplanes, a canopy *lock-open* shutoff valve is installed upstream from the selector valve. This is a solenoid operated valve. When electrically energized by the canopy being open, this shutoff valve blocks the flow of air from one side of the cylinder piston to the other, thus preventing accidental closure of the canopy, which could injure personnel. When deenergized, flow is unrestricted in either direction.

For emergency operation or opening of the canopy, the ballistics part of the canopy system actuates the counterbalance cylinder to jettison the canopy. This system will fire the cartridge attached to the cylinder, which will in turn actuate the second piston of the cylinder. This second (ballistics) piston then thrusts an inner sleeve and the upper cylinder cap completely out the top of the cylinder with a sufficient force to shear the canopy hinge pins and jettison the canopy clear of the airplane. Detailed descriptions of this part of the canopy system are given in the Airplane General Training Supplement of this series.

HOW THE SELECTOR VALVE CONTROLS THE CANOPY CYLINDER.

The canopy cylinder selector valve has two positions—counterbalance and cinchdown. In the upper view of figure 3-26 you see the counterbalance position in which pneumatic pressure is routed to the lower end of the cylinder. This pneumatic pressure is provided to equalize and counteract the existing pressure in the upper end of the canopy cylinder, thus providing the counterbalancing feature.

In the lower view of the illustration, the selector valve is positioned to prevent the entrance of air pressure to the lower end of the cylinder and at the same time exhaust existing static air pressure from the lower end of the cylinder. When the valve is in this latter position, the cylinder is in its cinchdown configuration.

Note in the upper view of this illustration, that the spool shaft is spring-loaded to its extreme extended travel from the valve. As you can see, this places the spool in a position (toward the extreme right) to route air pressure through the valve to the lower part of the canopy cylinder, and at the same time close off the internal chamber from the vent chamber flap. When released by the canopy lock linkage, the spool moves to this position due to the action of the internal and external springs shown at each end of the valve spool.

In the lower view, note that the spool shaft is at its other extreme of travel (toward the left). Now you see that the spool is positioned to close off air pressure to the lower end of the counterbalance cylinder. As this happens the existing static pressure is vented from the lower end of the cylinder by way of the escape passage in the spool, the vent chamber, and past the vent flap. The spring-loaded vent flap is blown open by the escaping air pressure. It closes again when the pressure is exhausted from the cylinder. Some of the O-rings you see in the various positions of the valve spool control air pressure leakage from the valve proper, while others prevent seepage from one area to another within the valve.

SUMMARY.

In this supplement you have learned that pneumatics is defined as the controlled use of energy developed by a compressed air power source. Even though compressed air is one of the oldest and most widely used sources of power, pneumatics is relatively new for aircraft usage.

On the F-102A, the high-pressure pneumatic system is an entirely new concept, even for aircraft usage.

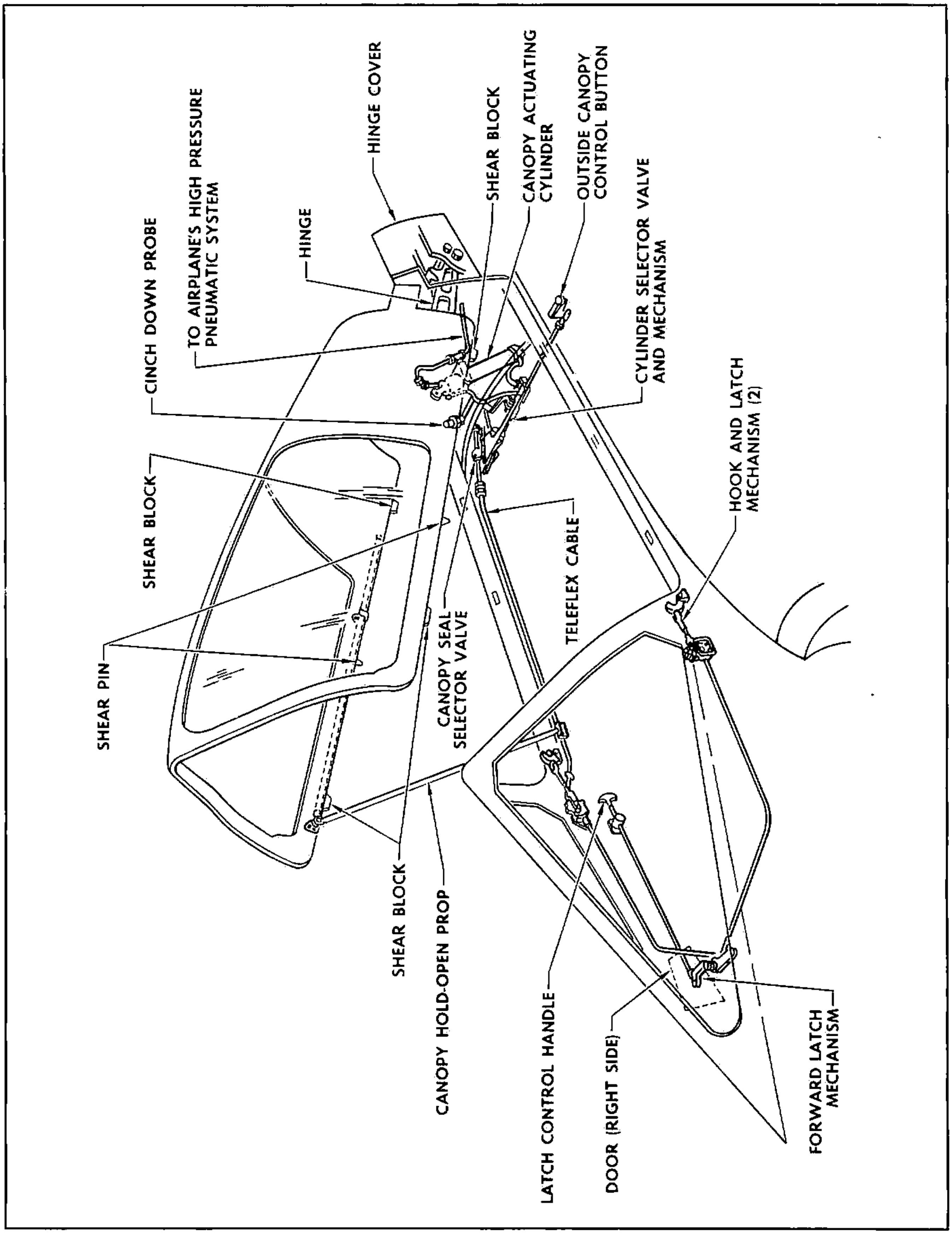


Figure 3-25. Canopy Actuation Pneumatic Schematic (Sheet 1 of 2)

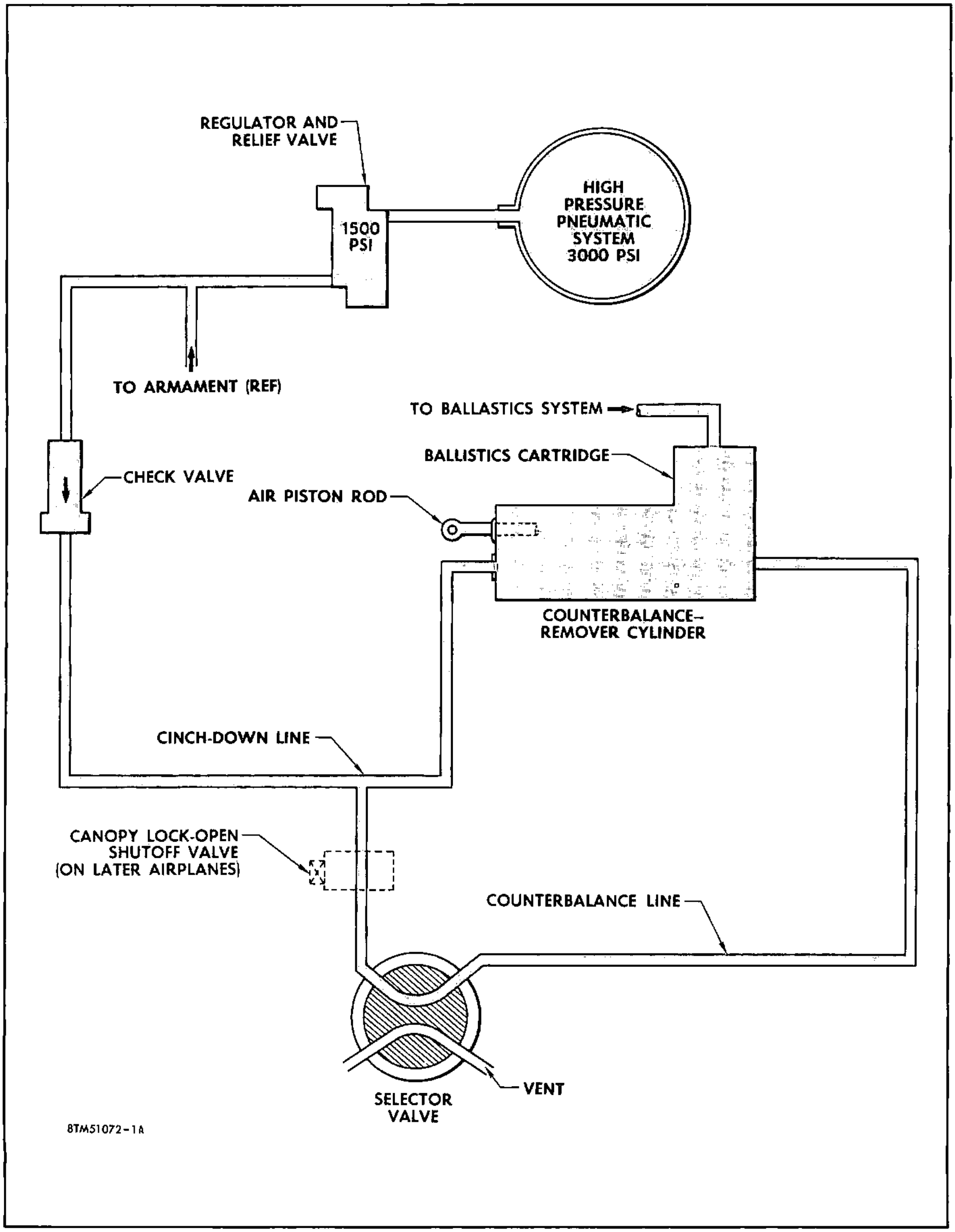


Figure 3-25. Canopy Actuation Pneumatic Schematic (Sheet 2 of 2)

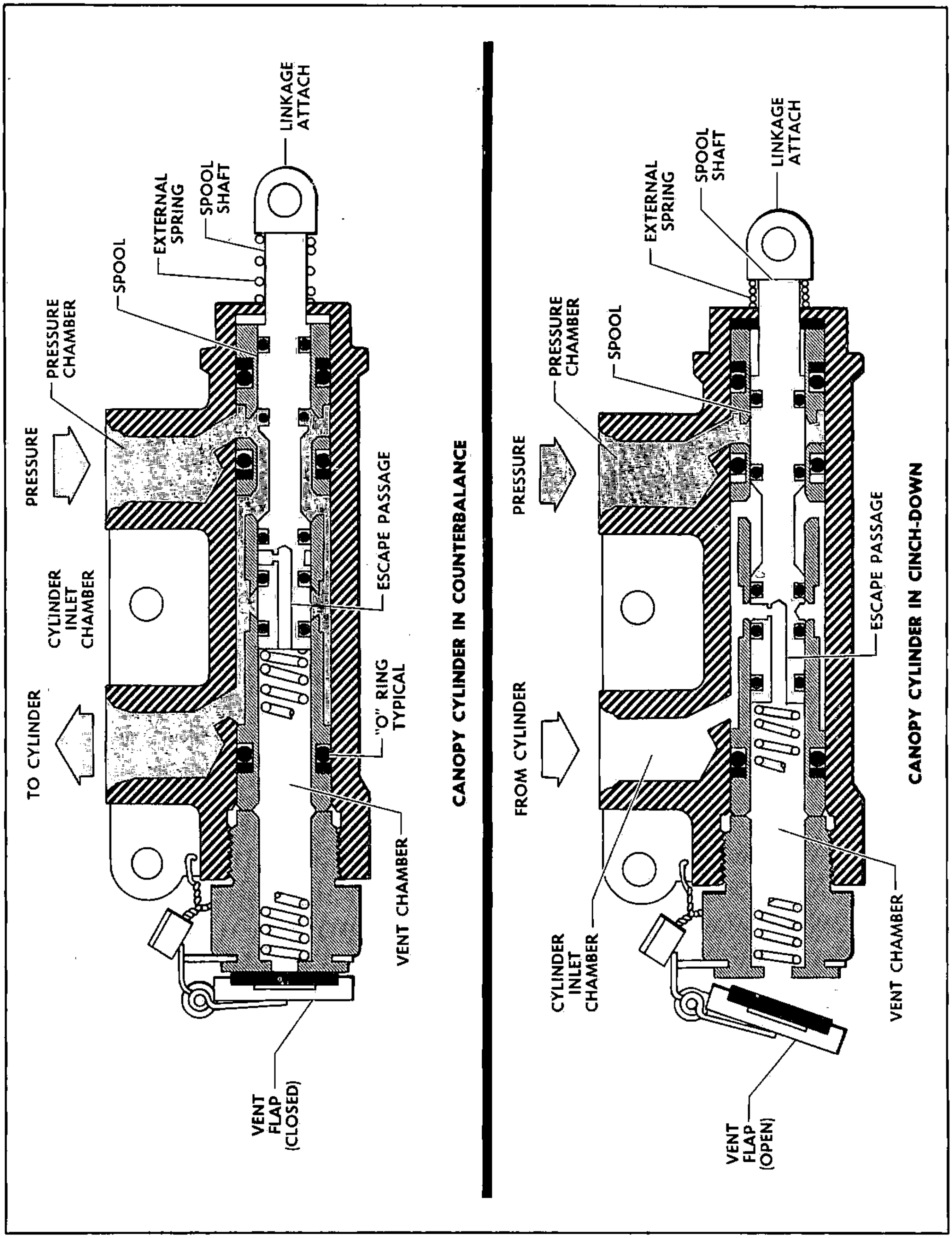


Figure 3-26. Canopy Counterbalance Remover Cylinder Selector Valve

Unique features offered by this system are fast action, reliability, compactness, and light-weight installations. However, as we have discussed the important functions of this power system and its operating subsystems, we have also discovered the simplicity of compressed air operated systems.

In addition to the operational and maintenance information which you will retain from this manual, you should also remember several other important facts. These facts pertain to the use of this manual. In many cases, the descriptions of the various pneumatic subsystems contained specific values and pressures. These values and pressures were used for explanatory purposes only, and they should not necessarily be used on the flight line.

The latest information of this nature is given in your F-102A Technical Order Maintenance Manual.

In some instances, references have been made to the fact that not all the F-102A airplanes have the same types of components in their pneumatic systems. No attempt has been made to point out the particular airplanes which have certain components or those airplanes that have other types. This information, too, must be obtained from your F-102A Maintenance Manual. You should keep in mind that this Training Supplement does not replace the Technical Order handbook—it is for your on-the-job training in the maintenance of this new airplane. From this point on you will learn the fine points of maintenance by actual experience.

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ALPHABETICAL INDEX

*The Symbol * Indicates An Illustration*

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ALPHAPETICAL INDEX (CONT)

*The Symbol * Indicates An Illustration*