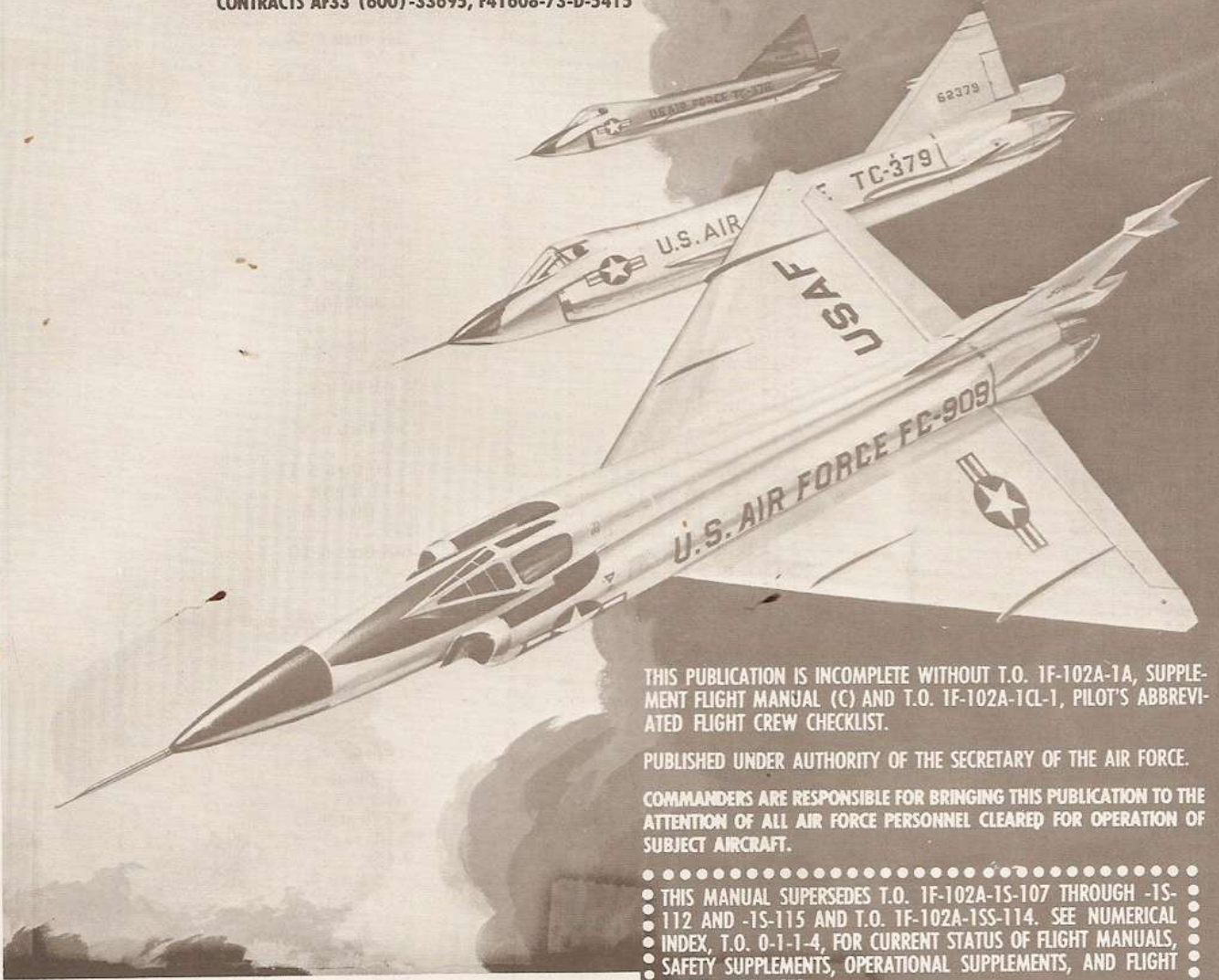
T.O. 1F-102A-1

FLIGHT MANUAL
F-102A & TF-102A
USAF SERIES AIRCRAFT

CONTRACTS AF33 (600)-33695, F41608-73-D-5415



THIS PUBLICATION IS INCOMPLETE WITHOUT T.O. 1F-102A-1A, SUPPLEMENT FLIGHT MANUAL (C) AND T.O. 1F-102A-1CL-1, PILOT'S ABBREVIATED FLIGHT CREW CHECKLIST.

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE.

COMMANDERS ARE RESPONSIBLE FOR BRINGING THIS PUBLICATION TO THE ATTENTION OF ALL AIR FORCE PERSONNEL CLEARED FOR OPERATION OF SUBJECT AIRCRAFT.

THIS MANUAL SUPERSEDES T.O. 1F-102A-1S-107 THROUGH -1S-112 AND -1S-115 AND T.O. 1F-102A-1SS-114. SEE NUMERICAL INDEX, T.O. 0-1-1-4, FOR CURRENT STATUS OF FLIGHT MANUALS, SAFETY SUPPLEMENTS, OPERATIONAL SUPPLEMENTS, AND FLIGHT CREW CHECKLISTS.

BASIC AND ALL CHANGES HAVE BEEN MERGED TO MAKE THIS A COMPLETE PUBLICATION.

CHANGE NOTICE

LATEST CHANGED PAGES SUPERSEDE THE SAME PAGES OF PREVIOUS DATE

Insert changed pages into basic publication. Destroy superseded pages.

21800

AIR FORCE 20 Nov 78-200-REPRINT

27 DECEMBER 1971
CHANGE 2—1 OCTOBER 1974

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Upon receipt of the second and subsequent changes to this manual, Flight Manual control officers shall ascertain that all previous changes have been received and incorporated. Prompt action should be taken if the manual is incomplete.

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ADDITIONAL COPIES OF THIS PUBLICATION MAY BE OBTAINED AS FOLLOWS:

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CURRENT FLIGHT CREW CHECKLIST
T.O. 1F-102A-1CL-1
27 DECEMBER 1971
Changed 15 April 1974

*The asterisk indicates pages changed, added, or deleted by the current change.

FLIGHT MANUAL, SAFETY SUPPLEMENT AND OPERATIONAL SUPPLEMENT STATUS

This page will be published with each Safety Supplement, Operational Supplement, Flight Manual Change, and Flight Manual Revision. It provides a comprehensive listing of the current Flight Manual, Flight Crew Checklist, Safety Supplements, and Operational Supplements. The supplements you receive should follow in sequence. If you are missing one listed on this page, see your Publications Distribution Officer and get your copy. Refer to Numerical Index, T.O. 0-1-1-4, and supplements thereto for current status of Flight Manuals, Safety Supplements, Operational Supplements, and Flight Crew Checklist.

CURRENT FLIGHT MANUALS	DATE	CHANGE NUMBER
T.O. 1F-102A-1	27 December 1971	2-1 October 1974
T.O. 1F-102A-1-1 (Conf)	16 September 1969	1-1 February 71

CURRENT FLIGHT CREW CHECKLIST	DATE	CHANGE NUMBER
T.O. 1F-102A-1CL-1	27 December 1971	2-15 April 1974

SUMMARY OF SAFETY SUPPLEMENTS INCORPORATED IN THIS CHANGE

Number	Date	Short Title	Flight Manual Pages Affected
SS-114	31 Jan 74	Emergency Gear Extension	3-35, 3-39, 7-2

CURRENT SAFETY SUPPLEMENTS

None

SUMMARY OF OPERATIONAL SUPPLEMENTS INCORPORATED IN THIS CHANGE

S-107	27 Sep 72	Ground Operation With The Fuel Control in Emergency	2-19
S-108	6 Feb 73	Correct Errors	1-8, 2-22, 2-23, 3-29, N-4
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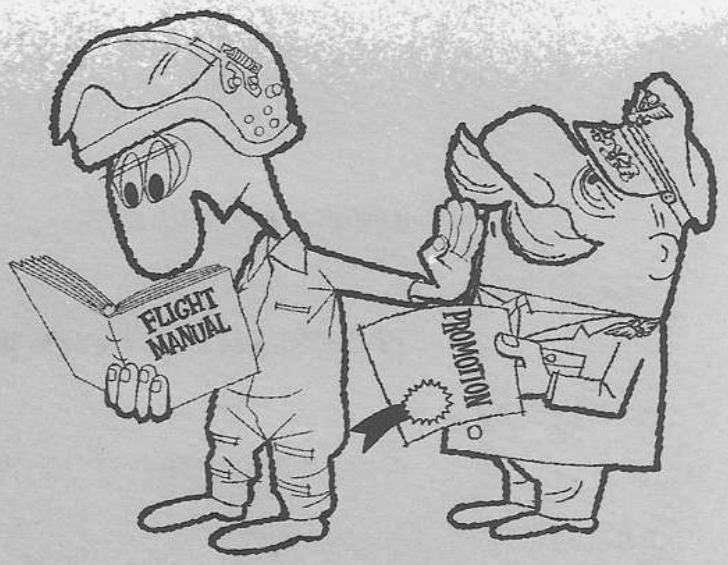
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WAIT...

THIS INFORMATION IS **IMPORTANT!!**



SCOPE

This manual contains the necessary information for safe and efficient operation of the F-102A and the TF-102A. These instructions provide a general knowledge of the airplane, its characteristics, and specific normal and emergency operating procedures. Your flying experience is recognized and basic flight principles are avoided.

PERMISSIBLE OPERATIONS

The Flight Manual takes a "positive approach" and normally states only what you can do. Unusual operations or configurations (such as asymmetrical loading) are prohibited unless specifically covered herein. Clearance must be obtained from SAAMA/MMEAF before any questionable operation is attempted which is not specifically permitted in this manual.

SOUND JUDGMENT

Instructions in this manual are for pilots inexperienced in the operation of these airplanes. **This manual 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.

STANDARDIZATION AND ARRANGEMENT

Standardization assures that the scope and arrangement of all Flight Manuals are identical. The manual is divided into ten fairly independent sections to simplify reading it straight through or using it as a reference manual. The first three sections must be read thoroughly and fully understood before attempting to fly the airplane. The remaining sections provide important information for safe and efficient mission accomplishment.

CHECKLISTS

The Flight Manual contains only amplified checklists. Abbreviated checklists have been issued as separate technical orders—see the back of the title page for T.O. number and date of your latest checklist. The checklist consists of three parts, arranged in the following order: Emergency Procedures, Normal Procedures, and Performance Data. The numbered items (line items) in the emergency and normal procedure sections correspond to identically numbered items in the amplified procedures of Sections II and III of the Flight Manual. The checklist contains airplane performance data in tabular and chart forms for both (F) and (TF) airplanes. Use of these data is explained in Part 8 of the Appendix in the Flight Manual.

CHECKLIST CONCURRENCY

As changes are made to the amplified checklists in the Flight Manual, concurrent changes will be made to the abbreviated checklist so that both will agree. However, a change to the Flight Manual may not affect the amplified procedures, and thus the Flight Manual date may not be the same as the checklist date. To determine the checklist applicable to a given Flight Manual issue, refer to the bottom of the Flight Manual "A" page under "Current

21,802-1

Flight Crew Checklist." For purposes of determining the concurrency between the Flight Manual and the checklist, the latest date of a Safety Supplement or an Operational Supplement affecting the checklist will be considered to represent the latest change date of the Flight Manual.

SAFETY SUPPLEMENTS

Information involving safety will be promptly forwarded to you by Safety Supplements. Supplements covering loss of life will get to you in 48 hours by TWX, and those concerning serious damage to equipment within ten days by mail. The current status of each Safety Supplement affecting your airplane can be determined by referring to the Numerical Index, T.O. 0-1-1-4, and supplements thereto. The title page of the Flight Manual and the title block of each Safety Supplement should also be checked to determine the status of existing supplements. You must remain constantly aware of the status of all supplements—current supplements must be complied with, but operations should not be restricted by complying with a replaced or rescinded supplement.

SAFETY OF FLIGHT CHECKLIST CHANGES

Whenever you receive an interim Safety Supplement affecting your checklist, write in the appropriate information. Printed replacement checklist pages will be made available to you when a formal Safety Supplement is issued. A notation on the bottom inside corner of these pages will indicate that they reflect certain Safety Supplements; however, discarding of a Safety Supplement is authorized only through the title page of the Flight Manual or T.O. 0-1-1-4).

HOW TO GET PERSONAL COPIES

Each pilot is entitled to personal copies of the Flight Manual, Safety Supplements, and checklists. The required quantities should be ordered before you need them to assure their prompt receipt. Check with your supply personnel—it is their job to fulfill your Technical Order requests. Basically, you must order the required quantities on the Publication Requirements Table (T.O. 0-3-1). Technical Orders 00-5-1 and 00-5-2 give detailed information for properly ordering these publications. Loose leaf binders and sectionalized tabs are available for use with your manual. These are obtained through local purchase procedures and are listed in the Federal Supply Schedule (FSC Groups 75, Office Supplies, Part 1).

WARNINGS, CAUTIONS, AND NOTES

The following definitions apply to "Warnings," "Cautions," and "Notes" found throughout the manual:

WARNING

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

CAUTION

Operating procedures, techniques, etc., which will result in damage to equipment if not carefully followed.

Note

An operating procedure, technique, etc., which is considered essential to emphasize.

YOUR RESPONSIBILITY— TO LET US KNOW

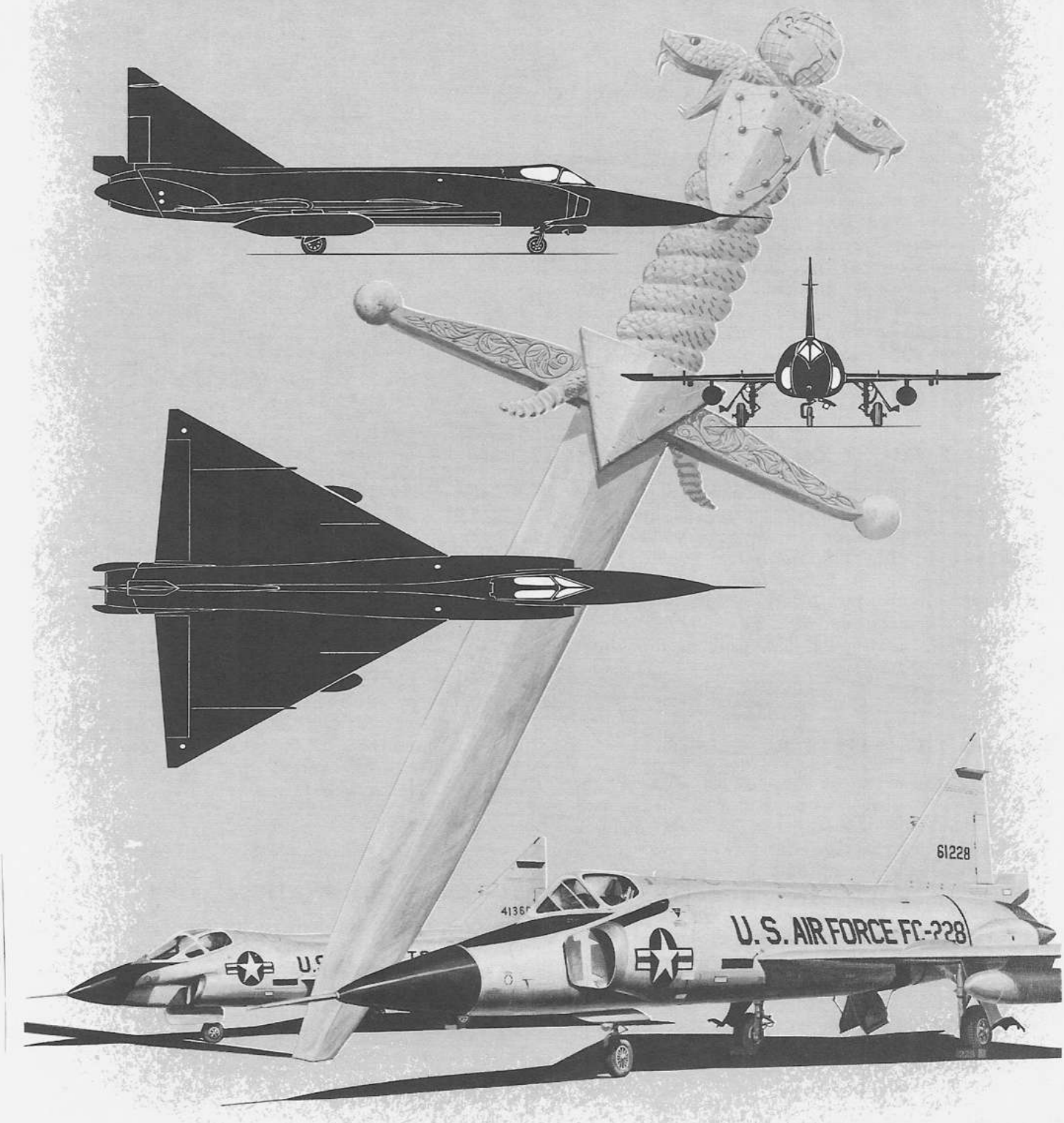
Every effort is made to keep the Flight Manual current. Review conferences with operating personnel and a constant review of accident and flight test reports assure inclusion of the latest data in the manual. **However, we cannot correct an error unless we know of its existence.** In this regard, it is essential that you do your part. Comments, corrections, and questions regarding this manual or any phase of the Flight Manual program are welcomed. These should be forwarded on AF Form 847 in accordance with AFR 60-9 to San Antonio ALC/MMEAF, Kelly AFB, Texas 78241, through your Command Headquarters.

AIRPLANE DESIGNATION CODE

Where text or illustrations are not specifically identified for particular models, it may be assumed that such items are common to both the F/TF-102A airplanes. Illustrations which do not show minor variations in the airplane models are labeled "Typical." Text and illustrations applicable to F-102A or TF-102A airplanes are identified by the following code symbols:

CODE	AIRPLANE
F	F-102A
TF	TF-102A

the F-102A & TF-102A delta dagger



21803



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THE AIRPLANE

The F-102A is a single-place, supersonic, all-weather interceptor built by General Dynamics/Convair. The TF-102A is a two-place, side-by-side trainer version of the F-102A and is designed for combat use, if conditions make such use necessary. The airplanes are equipped with a radar fire control system and are powered by a J57-P-23 axial-flow turbojet engine with afterburner. The airplanes are best characterized by the large 60-degree delta wing and the absence of a conventional empennage. Later (F)* and (TF)** 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. (TF) airplanes are equipped with two ejection seats. 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.

Note

Airplanes modified by TCTO 1F-102-1193 have an improved IFF/SIF radar transponder system (AIMS). This TCTO also incorporated an improved pitot static probe which essentially eliminated airspeed installation errors. Therefore, the readings on the airspeed-Mach indicator are now calibrated airspeed and true Mach number. Mach numbers above 0.5M have a small correction when extreme accuracy is required; however, under normal operation,

**56-1317 & on.*
***AF 56-2336 & on.*

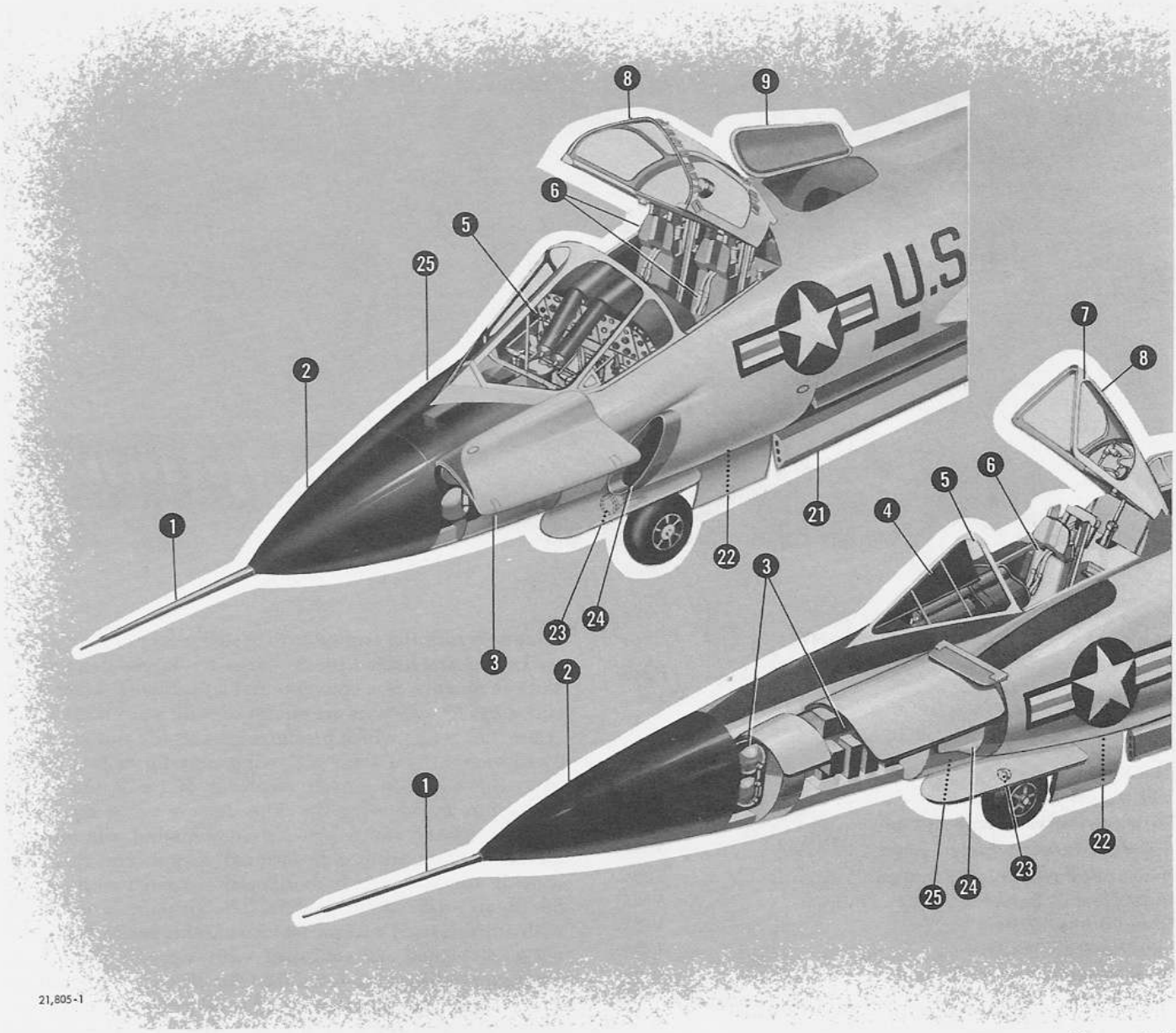


Figure 1-1

they can be considered the same as true Mach. During the transonic speed range, some lag is encountered in airspeed readings; but it is not considered significant. A correction is provided in the event extreme accuracy is required. For airplanes not modified by TCTO 1F-102-1193, speeds will remain indicated airspeed (KIAS) and indicate Mach number (IMN) with appropriate corrections required to attain calibrated airspeed and true Mach number. Airspeeds considered critical will be shown as XXX KCAS (XXX KIAS) to differentiate between configurations. Airspeeds not considered critical or airspeeds that are approximate will be listed as XXX KCAS (KIAS) to show applicability to both configurations.

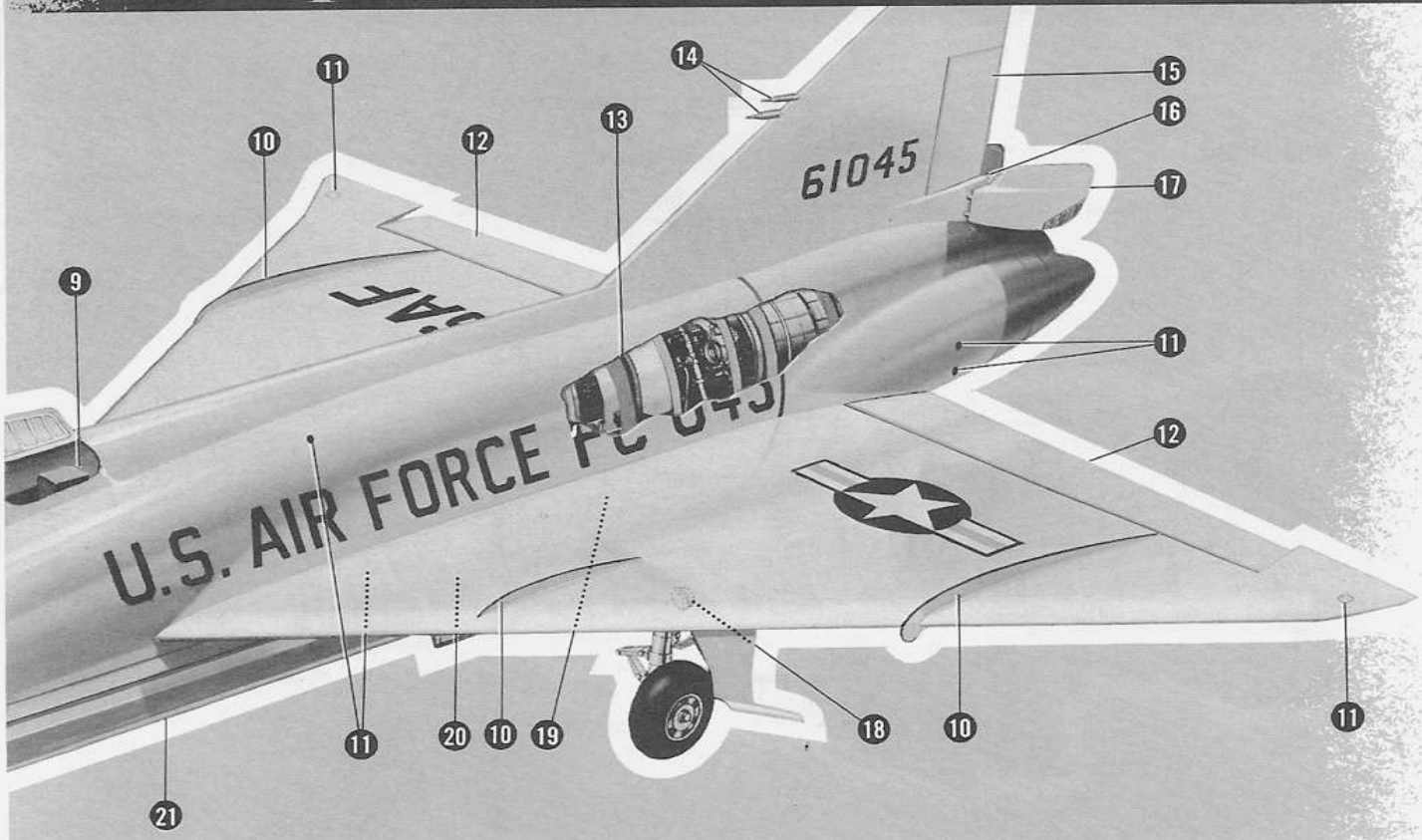
DIMENSIONS

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

(F)

Wing span	38 feet, 1.6 inches
Length (including pitot boom)	68 feet, 1.8 inches
Height (to tip of vertical stabilizer)	21 feet, 2.5 inches
Tread	14 feet, 2.25 inches

airplane general arrangement



1. Pitot Boom
2. Radome
3. Forward Electronics Compartments
4. Vision Splitter
5. Windshield
6. Ejection Seat
7. Accelerometer (Some Airplanes)
8. Canopy
9. Upper Electronics Compartment
10. Wing Fence
11. Position Lights
12. Elevon
13. Engine
14. Ram Air (q) Intakes
15. Rudder
16. Drag Chute Compartment
17. Speed Brakes
18. Landing Light
19. External Power Receptacle
20. Aft Electronics Compartment
21. Armament Bay Door
22. Intermediate Electronics Compartment
23. Taxi Light
24. Intake Duct
25. Battery

21,805-2

(TF)

Wing span	38 feet, 1.6 inches
Length (including pitot boom)	63 feet, 4.3 inches
Height (to tip of vertical stabilizer)	21 feet, 2.25 inches
Tread	14 feet, 2.25 inches

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

GROSS WEIGHTS

Gross weights vary according to various mission loading conditions. On (F) airplanes, gross weights range from approximately 28,150 pounds to approximately 31,276 pounds. On (TF) airplanes, gross weights range from approximately 28,978 pounds to approximately 32,104 pounds. The figures below are averages, and are provided to show average gross weights at different configurations:

(F)

	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

**Includes the weight of the pilot, survival equipment, trapped fuel and oil, etc.*

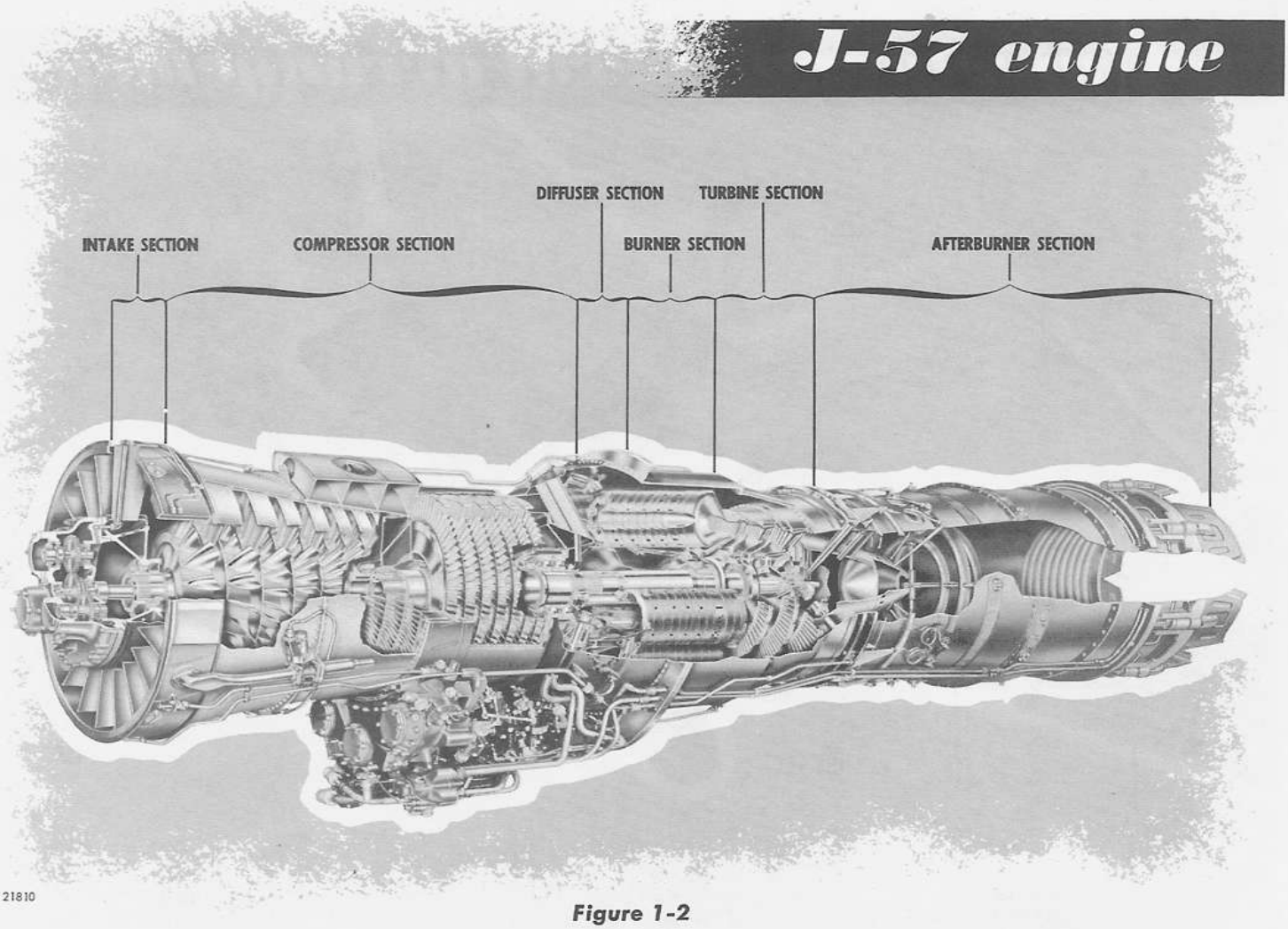


Figure 1-2

(TF)

	CLEAN	WITH EXTERNAL TANKS
Empty Weight**	20,731	21,062
Usable Fuel	7,053	9,848
Armament	1,194	1,194
TOTALS	28,978	32,104

***Includes the weight of the crew, survival equipment, trapped fuel and oil, etc.*

Note

- The figures above are averages and are not to be used in flight planning. For further information refer to the Basic Weight Check List and Loading Data, T.O. 1F-102A/(T)A-5, and to Weight Limitations, Section V of this manual.

- For (F) airplanes equipped with one full external tank and one full chaff pod, total gross weight is 30,939.

ENGINE

Thrust is supplied by a Pratt and Whitney J57-P-23A engine with afterburner (figure 1-2). Approximate standard sea level static thrust rating 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 characteristics 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.

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 1-3). 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 1-3) supplies the fuel pressure required by the engine and afterburner systems. The unit is driven through a single shaft and consists of three pumps. A centrifugal pump draws in fuel and forces it to two gear-type pumps. One gear-type

engine fuel control system (typical)

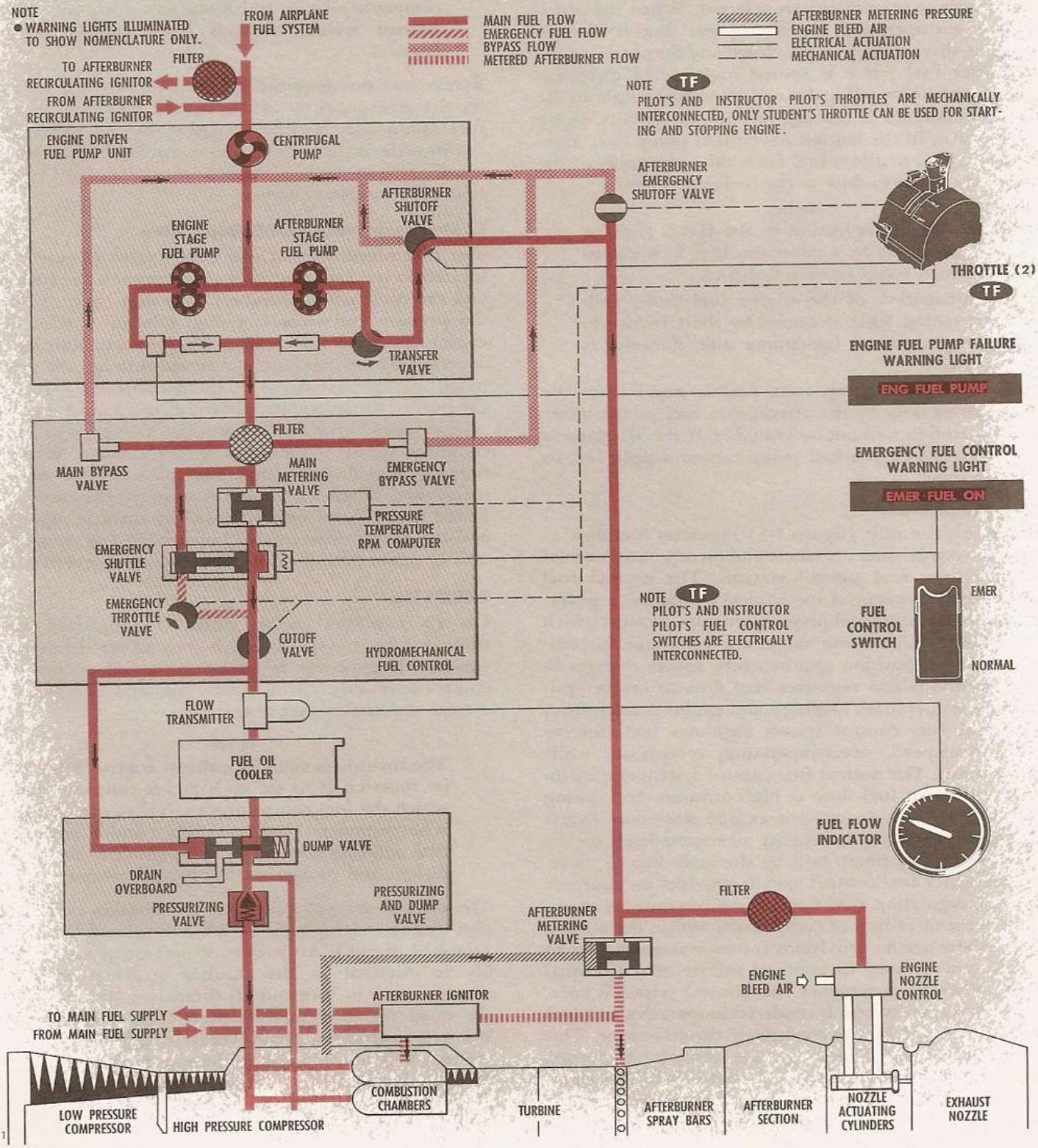


Figure 1-3

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.

Note

- Rapid throttle reduction to the IDLE position may contribute to a minimum fuel flow condition which could cause a flameout.
- If illumination of the engine fuel pump failure warning light is caused by shaft failure in the engine-driven fuel pump unit, flameout is imminent.

If the engine fuel pump fails, fuel is supplied to the engine system only by the afterburner fuel pump; afterburner operation will not be available. If the afterburner pump fails, the engine fuel pump cannot supply fuel to the afterburner.

Fuel Control Unit

The fuel control unit (figure 1-3) 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 Pressurizing and Dump Valve

The fuel pressurization and dump valve (figure 1-3) 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 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

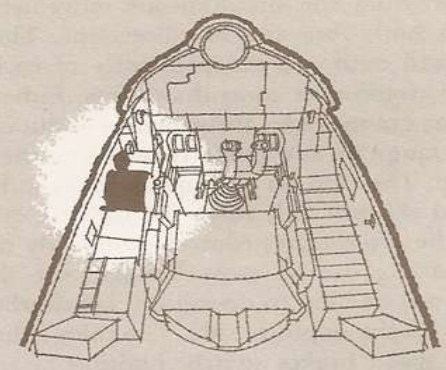
On (F) airplanes engine thrust is controlled by the throttle (figure 1-4) which is located on the left-hand console. The throttle is mechanically linked to the fuel control unit and controls the normal and emergency fuel system and afterburner operation.

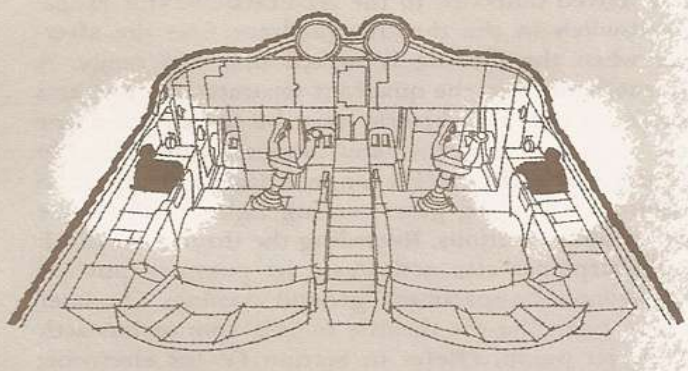
Note

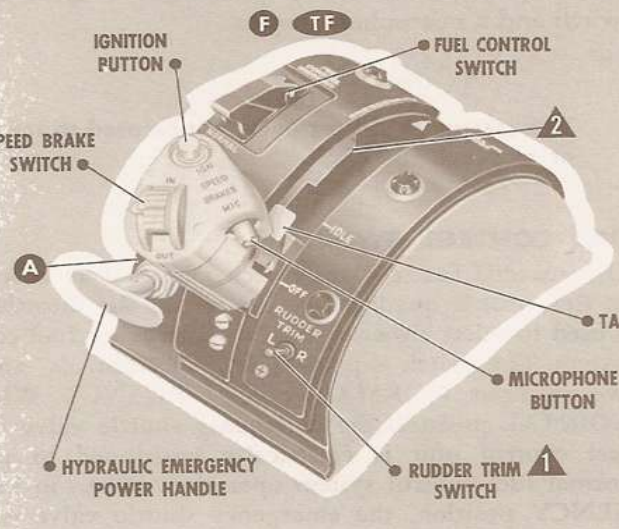
The throttle is actually a thrust control lever as its function is to set up a thrust condition for which the fuel control meters fuel to the engine with automatic compensation (under normal fuel control) for engine rpm, ambient air conditions and compressor discharge pressure.

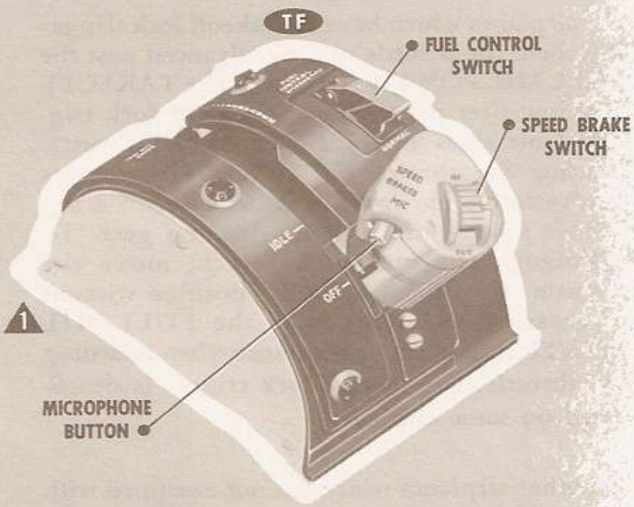
The throttle grip contains the speed brakes switch, ignition button, microphone button and takeoff lock trigger (some airplanes). An automatic anti-creep spring assembly is installed in the throttle quadrant to prevent throttle creep in fore and aft directions, thus eliminating the need for a manual 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 air-motors the engine. With the ignition button depressed, moving the throttle inboard to OFF energizes the starter and ignition circuit and supplies fuel to the starter. At the forward IDLE

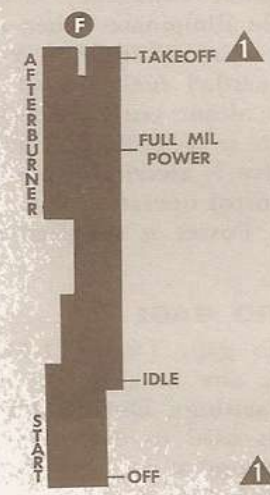
throttle



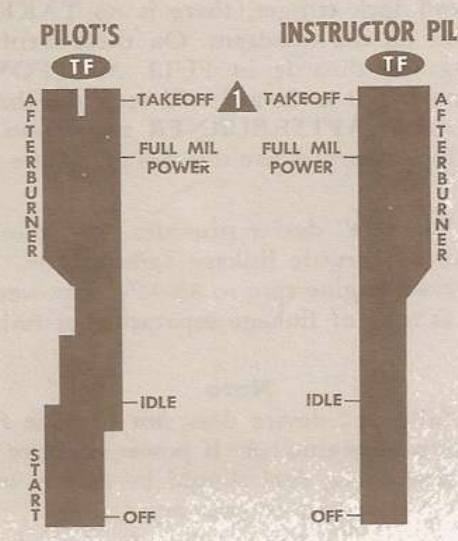












1 SOME AIRPLANES

2 TF SHOWN; SEE DETAIL FOR F AFTERBURNER REAR STOP CONFIGURATION

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Figure 1-4

stop, the throttle must be moved inboard to clear the stop. This places the throttle in the IDLE position of the normal range and supplies fuel to the combustion chambers. The throttle may then be advanced to FULL MIL POWER position. 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. Throttle position can also cause the landing gear warning light to illuminate under certain conditions. 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.)

Note

- On airplanes which have the takeoff lock trigger installed, the throttle may be advanced past the FULL MIL POWER position to the TAKEOFF position after depressing the takeoff lock trigger (this does not augment engine thrust). Placing the throttle in TAKEOFF prevents inadvertently moving the throttle out of afterburner when retracting the landing gear. To disengage the mechanical lock, move the throttle aft of the TAKEOFF position without depressing the trigger as the FULL MIL POWER stop offers no resistance when retarding the throttle. The takeoff lock trigger is deactivated on some airplanes.

- 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 five degrees aft of the stop.

A spring "fail-safe" device provides "get home" power in the event of throttle linkage failure. The "fail-safe" device increases engine rpm to 85-92% if power is below this range at time of linkage separation or failure.

Note

- The "fail-safe" device does not provide for a reduction in engine rpm if power is above 92% rpm. Remaining fuel should be closely monitored to ensure a safe recovery.
- An existing characteristic of the fuel control system results in an engine rpm of approximately 95% if the fuel control drive spline fails when operating in the normal fuel control system.

On TF airplanes, two throttles are provided and are located on the left and right-hand consoles. The throttles are mechanically linked to the fuel control unit and control the normal and emergency fuel system. By means of switches in each quadrant, the throttles also control afterburner operation. Mechanical linkage between the throttles provides for simultaneous movement of both throttles in both fore and aft directions. The throttles can be moved outboard independently of each other to actuate the afterburner control switches. Either the pilot or instructor can move his throttle into or out of AFTERBURNER range at any point forward of the aft afterburner stop. The stop on each quadrant is so biased that a continued aft pull will force the throttle inboard and deactuate the afterburner control switch. By this means, without making use of cross linkage, the throttle on either side can be used to disengage the afterburner control switch on the opposite side. The pilot's throttle grip contains a speed brakes switch, ignition button, microphone button, and takeoff lock trigger (some airplanes). The instructor's throttle grip contains a speed brakes switch and a microphone button.

Note

The instructor's throttle cannot be used to start or stopcock the engine.

FUEL CONTROL SWITCH

The guarded fuel control switch (figure 1-4) 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. On TF airplanes, a guarded fuel control switch is located on each throttle quadrant, outboard of the throttle. When either switch is in EMERGENCY position, the emergency shuttle valve is elecrically positioned to permit emergency fuel control operation and the warning light will illuminate. Power is supplied from the 28-volt dc essential bus.

ENGINE PRESSURE RATIO GAGE

The engine pressure ratio gage (figure 1-5) is used during preflight engine checks and for establishing inflight cruise thrust settings. During the preflight 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

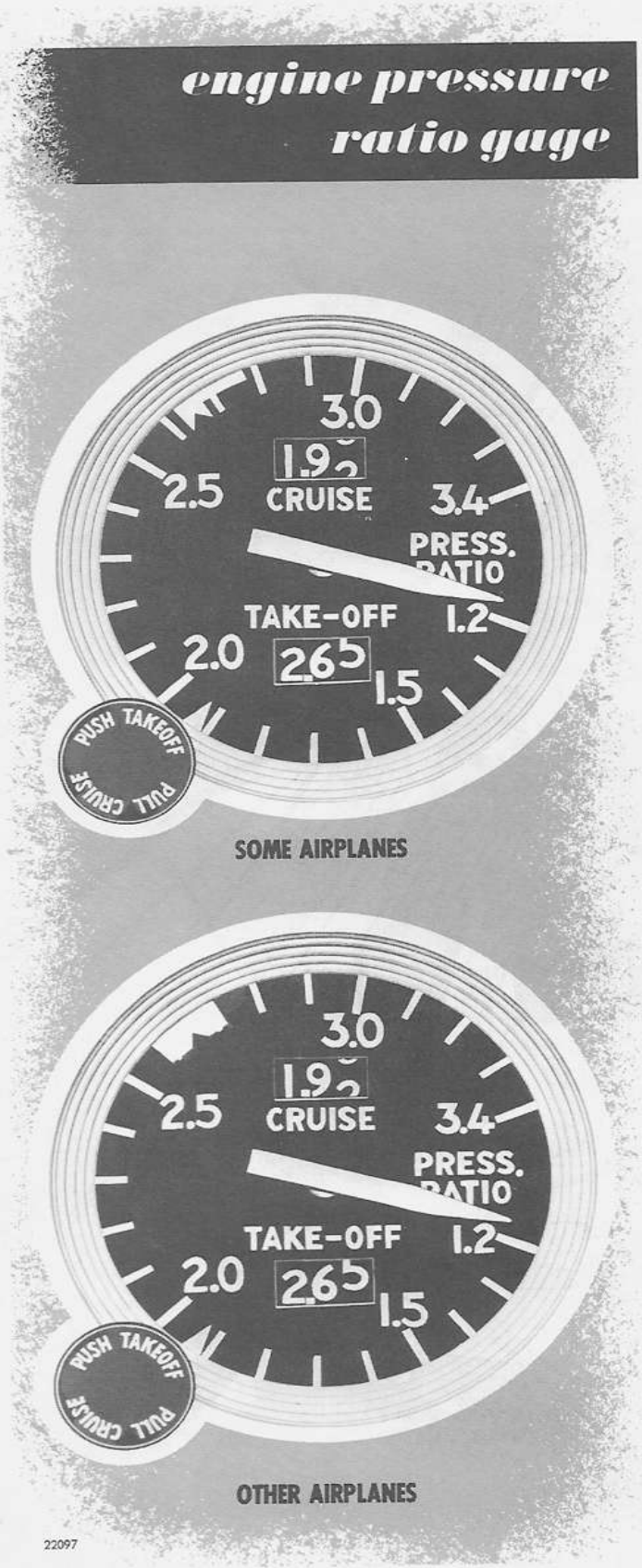


Figure 1-5

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 chart in the Appendix if outside air temperature is known.

Note

The fact that outside air temperature is not normally available precludes the use of the cruise feature of this gage.

The extreme ends of the elongated reference mark represent the minimum and maximum takeoff limits. If the engine pressure ratio gage does not have the elongated reference marker, use a measurement of ±0.03 inch unless minimum and maximum settings are available. The notched portion is utilized to set the target engine pressure ratio obtained from the takeoff check chart in the Appendix. 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 (figure 1-7) 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 rpm. The military thrust 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 (airplanes without engine hot-section analyzer installed; figure 1-7), 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 some 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.

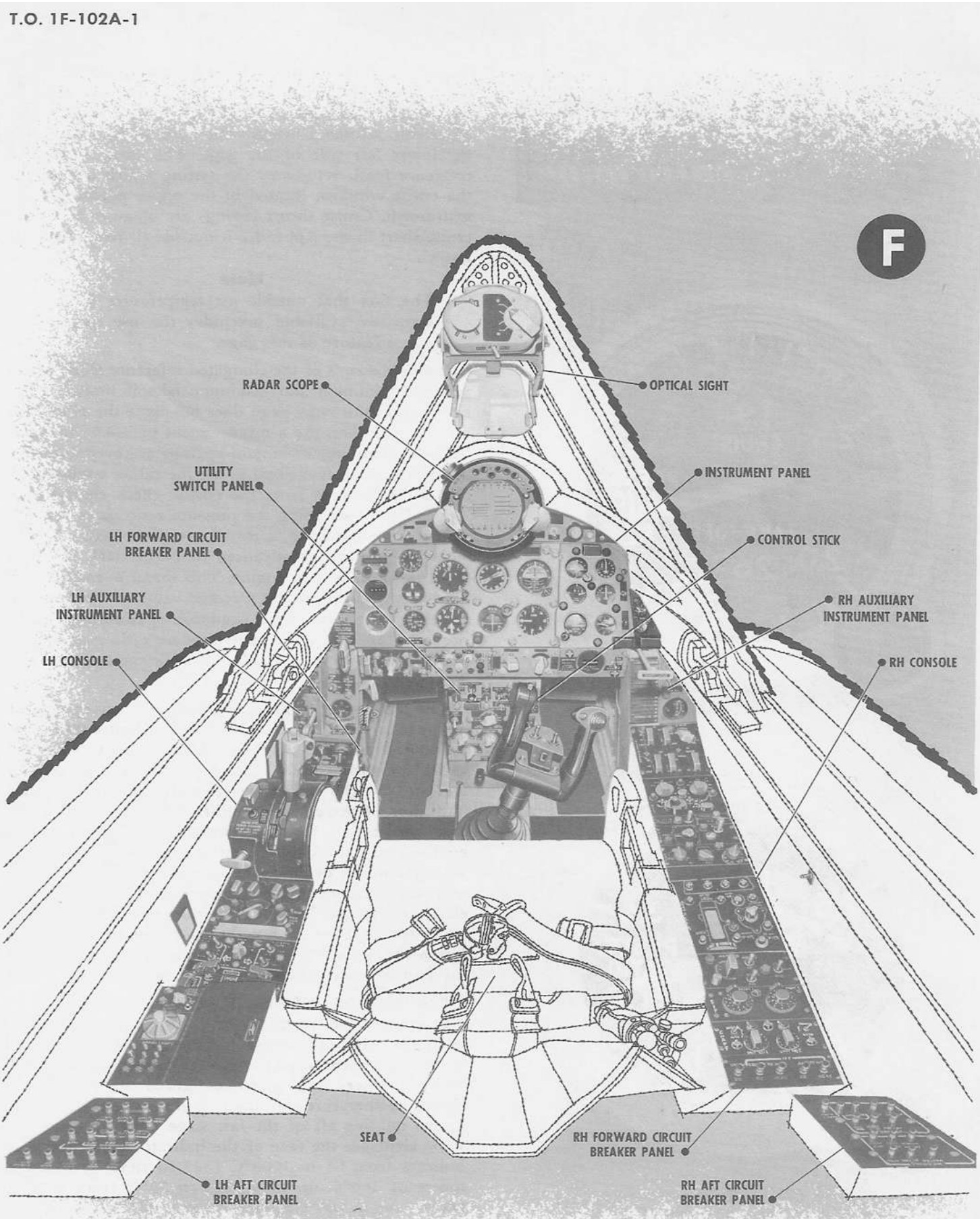
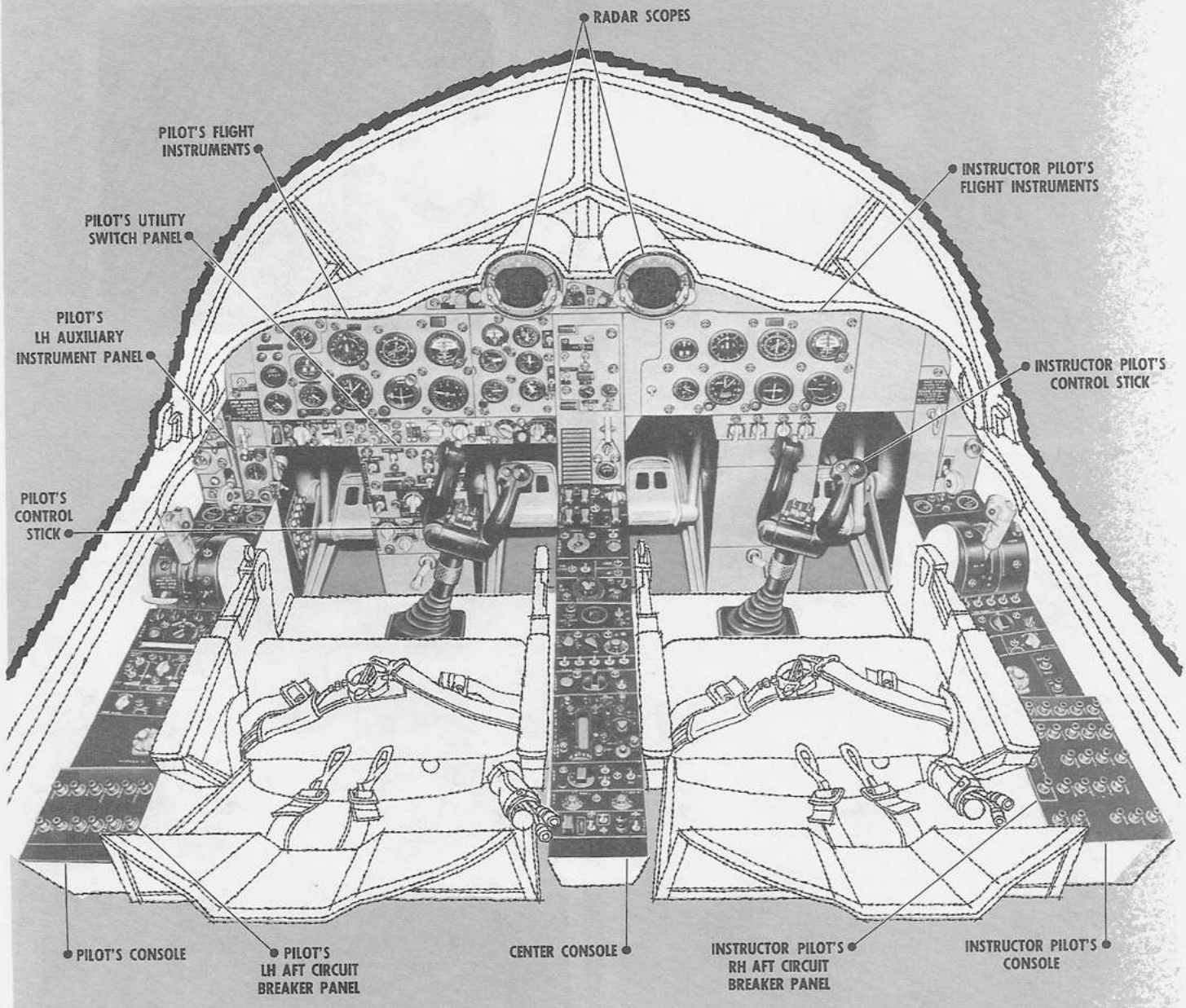

F
RADAR SCOPE
OPTICAL SIGHT
UTILITY SWITCH PANEL
INSTRUMENT PANEL
LH FORWARD CIRCUIT BREAKER PANEL
CONTROL STICK
LH AUXILIARY INSTRUMENT PANEL
RH AUXILIARY INSTRUMENT PANEL
LH CONSOLE
RH CONSOLE
SEAT
RH FORWARD CIRCUIT BREAKER PANEL
LH AFT CIRCUIT BREAKER PANEL
RH AFT CIRCUIT BREAKER PANEL
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Figure 1-6

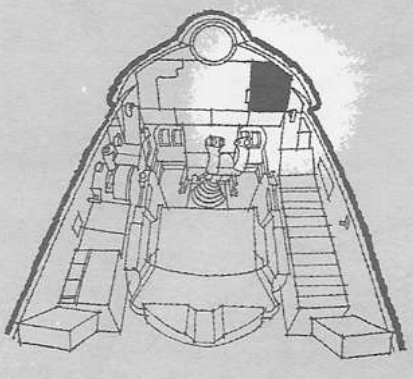
cockpit general arrangement (typical)

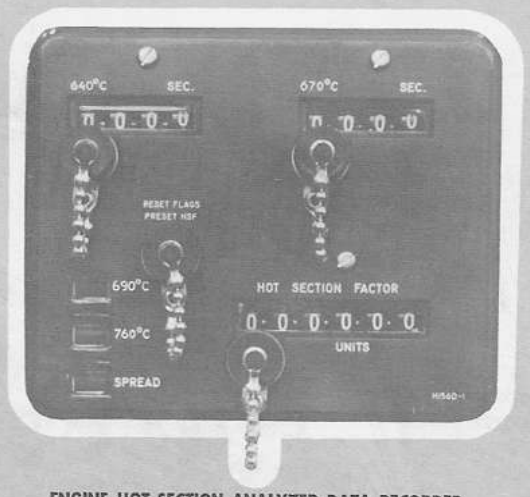
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ENGINE HOT-SECTION ANALYZER DATA RECORDER
(LEFT MAIN WHEEL WELL)



SOME AIRPLANES



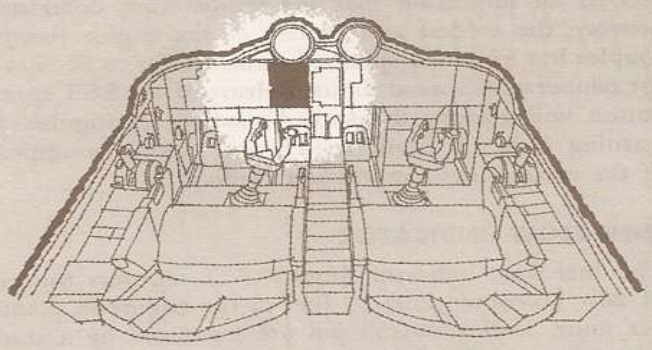
OTHER AIRPLANES

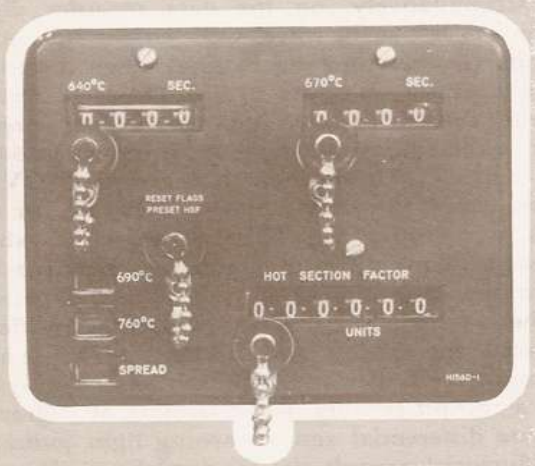
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Figure 1-7

engine instruments (typical)

TF

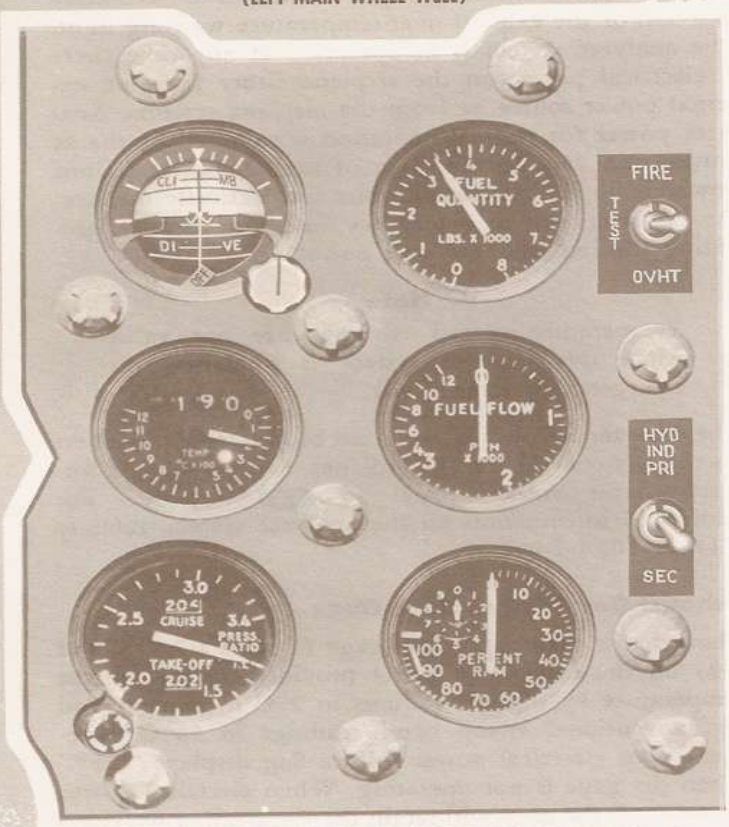




ENGINE HOT-SECTION ANALYZER DATA RECORDER
(LEFT MAIN WHEEL WELL)



SOME AIRPLANES



OTHER AIRPLANES

21835-2

ENGINE HOT SECTION ANALYZER

The engine hot-section analyzer system provides exhaust gas temperature readings, exhaust thermocouple temperature spread check readings, and a warning light indication for engine exhaust temperatures over 665°C. The analyzer system equipment consists of an exhaust gas temperature gage, a temperature spread check switch, a data recorder, a 115-volt ac, 400 cycle emergency inverter, and four thermocouples. The analyzer system records accumulatively, on the data recorder digital counter in the left main wheel well, engine hot section factor units (temperature versus time) when the engine exhaust temperature is above 575°C. Other features of the recorder consists of two digital counters that record seconds of engine operation at temperatures above 640°C and above 670°C. The recorder also registers "OVER" for engine temperatures over 690°C; "OVER" for temperatures over 760°C; and "OVER" when the exhaust thermocouple temperature differential exceeds specified limits (temperature spread check). The temperature spread check system measures the temperature differential between the coldest and hottest of the exhaust thermocouples. On demand, the spread check system provides a cockpit reading of the number of degrees of temperature differential and a warning light indication if the differential exceeds the specified limit. The temperature spread warning light feature is a secondary function of the exhaust over-temperature warning light. The analyzer system is in operation at any time there is electrical power on the airplane either from an external power source or from the airplane systems. Analyzer power for normal operation is supplied by the ac nonessential bus. In the event of normal power failure, power from the dc essential bus automatically operates the analyzer system emergency inverter, thereby providing the required 115 volt power.

Note

Temperature spread readings are not valid when the analyzer system is operating on emergency inverter power.

The analyzer circuitry is protected by two circuit breakers on the left console on F airplanes, and on the left forward circuit breaker panel on TF airplanes. For additional information on the analyzer system, refer to T.O. 1F-102A-2-9.

Exhaust Gas Temperature Gage

The exhaust gas temperature gage (figures 1-7, 1-8, and 1-9) on the instrument panel provides engine exhaust temperature by digital readings in 2°C increments and by conventional sweep hand readings in 50°C increments. An electrical power failure flag displays "OFF" when the gage is not operating. When electrical power is removed, the gage will retain the last reading provided by the analyzer system. A red overtemperature warning light is incorporated in the lower right side of the gage face. This light is illuminated when exhaust gas temperature exceeds 665°C. The light is also used in conjunction with the engine thermocouple temperature spread check. The gage receives normal power from the ac nonessential 200/115-volt, three-phase bus and emergency power from the 28-volt dc essential bus.

EGT Spread Button

The EGT spread button, placarded "EGT SPREAD," is installed on the left console (figures 1-10 and 1-11). When the button is depressed, the exhaust gas temperature gage reading will indicate the engine thermocouple temperature differential reading. Illumination of the exhaust gas temperature gage warning light at this time is an indication that the temperature differential between the coldest and hottest of the engine thermocouples has exceeded specified limits. Refer to Section V for temperature spread limits. Release of the EGT spread button will remove the spread reading, extinguish the warning light (if illuminated), and permit resumption of the exhaust gas temperature readings.

FUEL FLOW INDICATOR

The fuel flow indicator (figure 1-7) 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. 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.

ENGINE FUEL PUMP FAILURE WARNING LIGHT

The engine fuel pump failure warning light (figure 1-36), 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 (figure 1-36), located on the warning light panel, illuminates and displays "EMER FUEL ON" when the fuel control switch is moved from NORMAL to EMERGENCY position. On TF airplanes, the warning light will illumninate when either fuel control switch is placed in the 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 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 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. An all-points ignition system is incorporated which permits energizing the igniters from any throttle position when the airplane is airborne. When the all-points ignition system is used with the aircraft airborne, the starter circuits are bypassed.

Ignition Button

The ignition button (figure 1-4) is used to energize the igniters during starting and is located on the throttle. During ground starts and when performing an unsuccessful air start procedure, 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, it is not normally necessary to arm the ignition circuit by placing the throttle to START. The all-points ignition system arms the ignition circuit and the button is "hot" whenever the weight of the airplane is off the nose gear or the nose wheel is more than 50° from center. On (TF) airplanes, an ignition button is located on the pilot's throttle only. The ignition button receives power from the 28-volt dc essential bus.

Engine Ignition Disconnect Switch

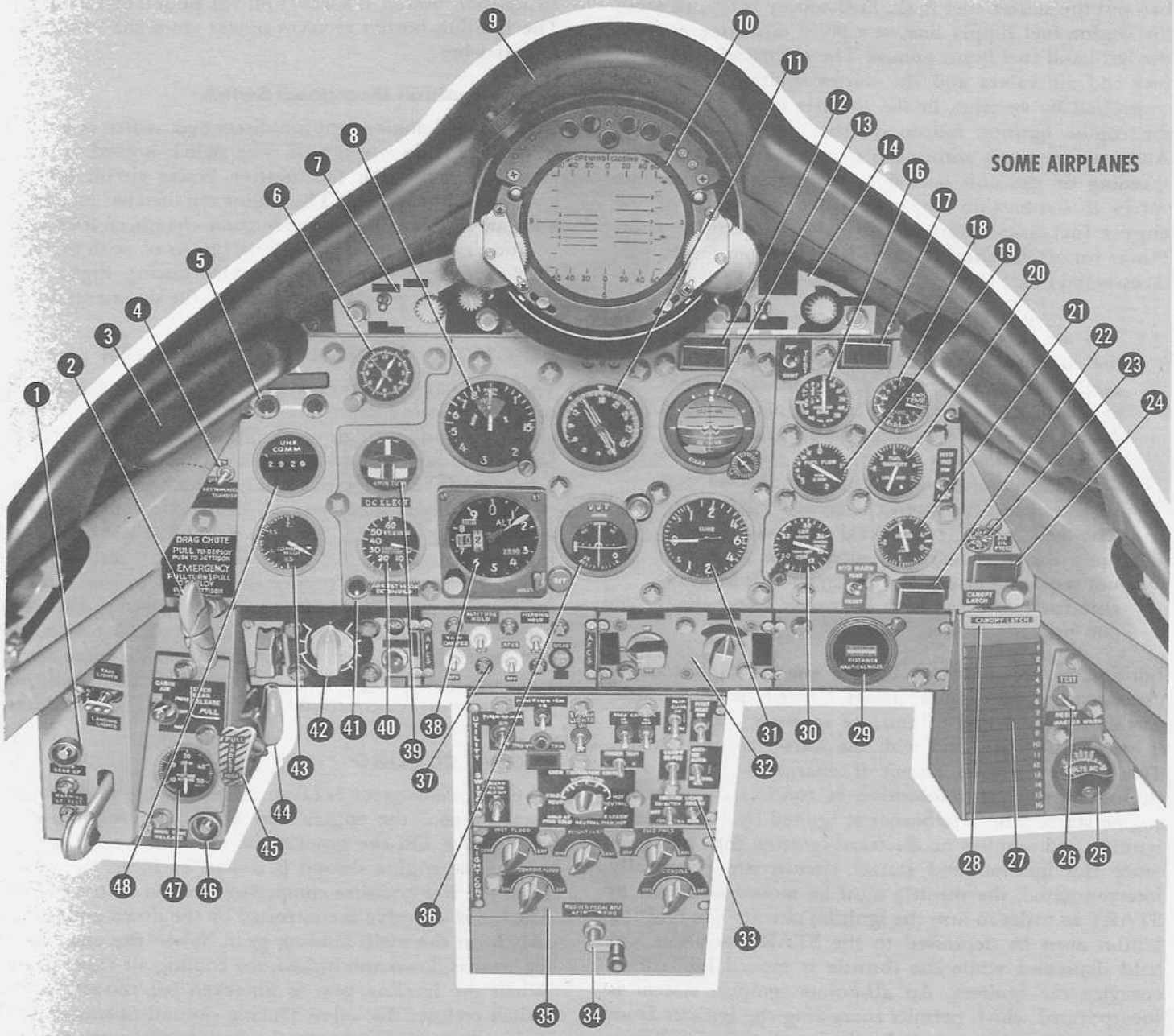
The guarded engine ignition disconnect switch is located in the left main wheel well. The switch is used to interrupt power between the ignition power circuit breaker and the engine igniters. The engine can then be "motored" without ignition in the combustion chambers. The two-position switch is placarded "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.

AUTOMATIC IGNITION (RELIGHT) SYSTEM

The automatic ignition system provides instantaneous relight following flameouts which are caused by momentary interruptions of air or fuel flow. The device responds within one-half second following flameout and initiates relight while the engine is still hot and before any appreciable rundown. The ignition deactivates automatically upon engine recovery. In case relight is not obtained, ignition remains on for approximately 15 seconds. The automatic ignition is connected parallel to the existing ignition circuit. The unit senses compressor discharge pressure and pressure decay in excess of 10 psi/second and causes a snap action diaphragm to close a microswitch and energize the ignition system. This system does not preclude pilot responsibility to initiate airstart procedures.

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



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. Airspeed-Mach Indicator
9. Glare Shield
10. Radar Scope
11. Radio Magnetic Indicator
12. Engine Fire and Overheat Warning Light
13. Attitude Indicator (MM-3)
14. Right-Hand Scope Control Panel
15. Standby Attitude Indicator
16. Tachometer
17. Master Warning Light
18. Exhaust Gas Temperature Gage
19. Fuel Flow Indicator
20. Fuel Quantity Gage
21. Hydraulic Pressure Gage
22. Hydraulic Pressure Low Warning Light
23. Oil Pressure Gage
24. Canopy Unlocked Warning Light

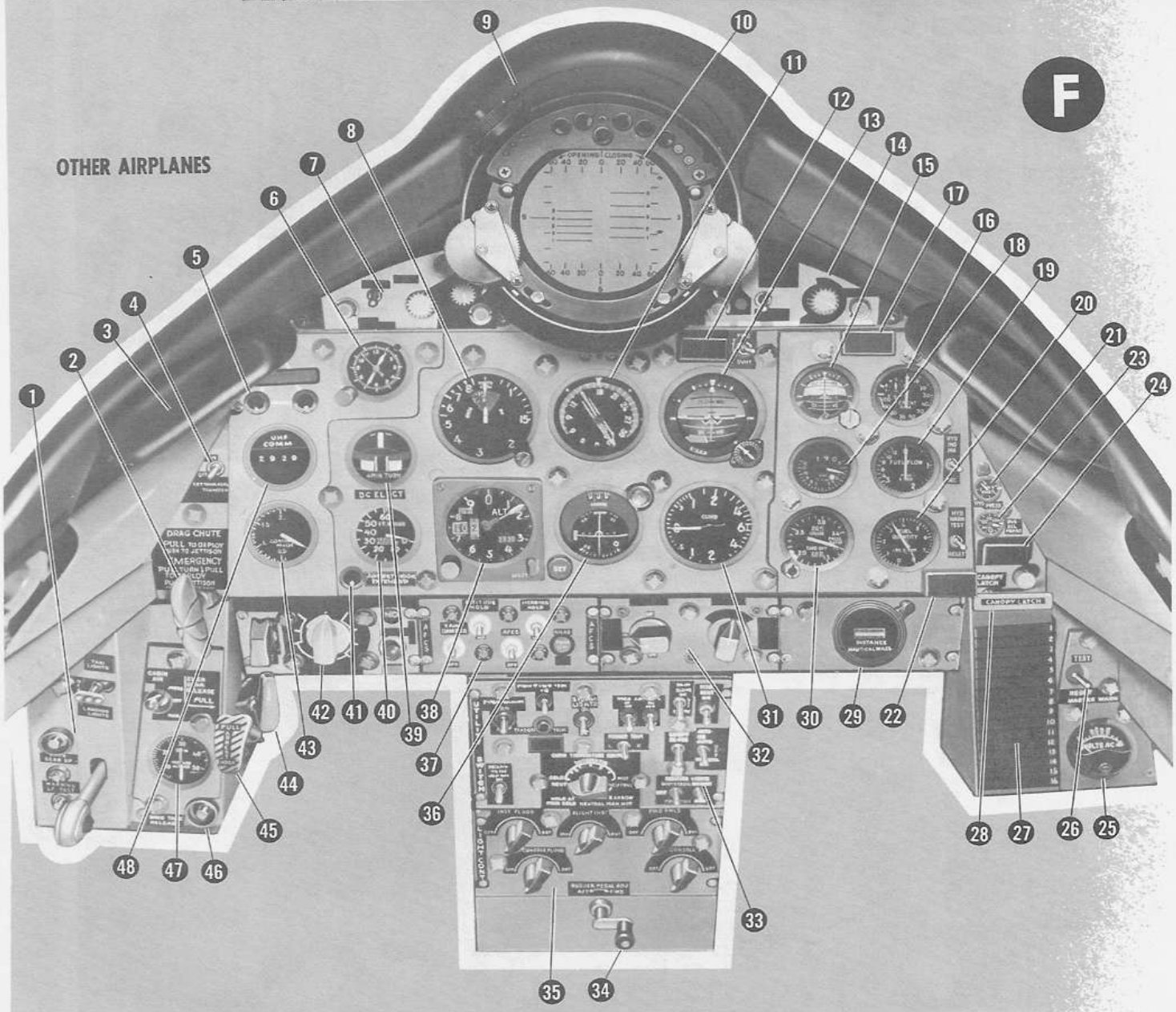
22,128-1 1 SOME AIRPLANES

Figure 1-8

instrument panel (typical)

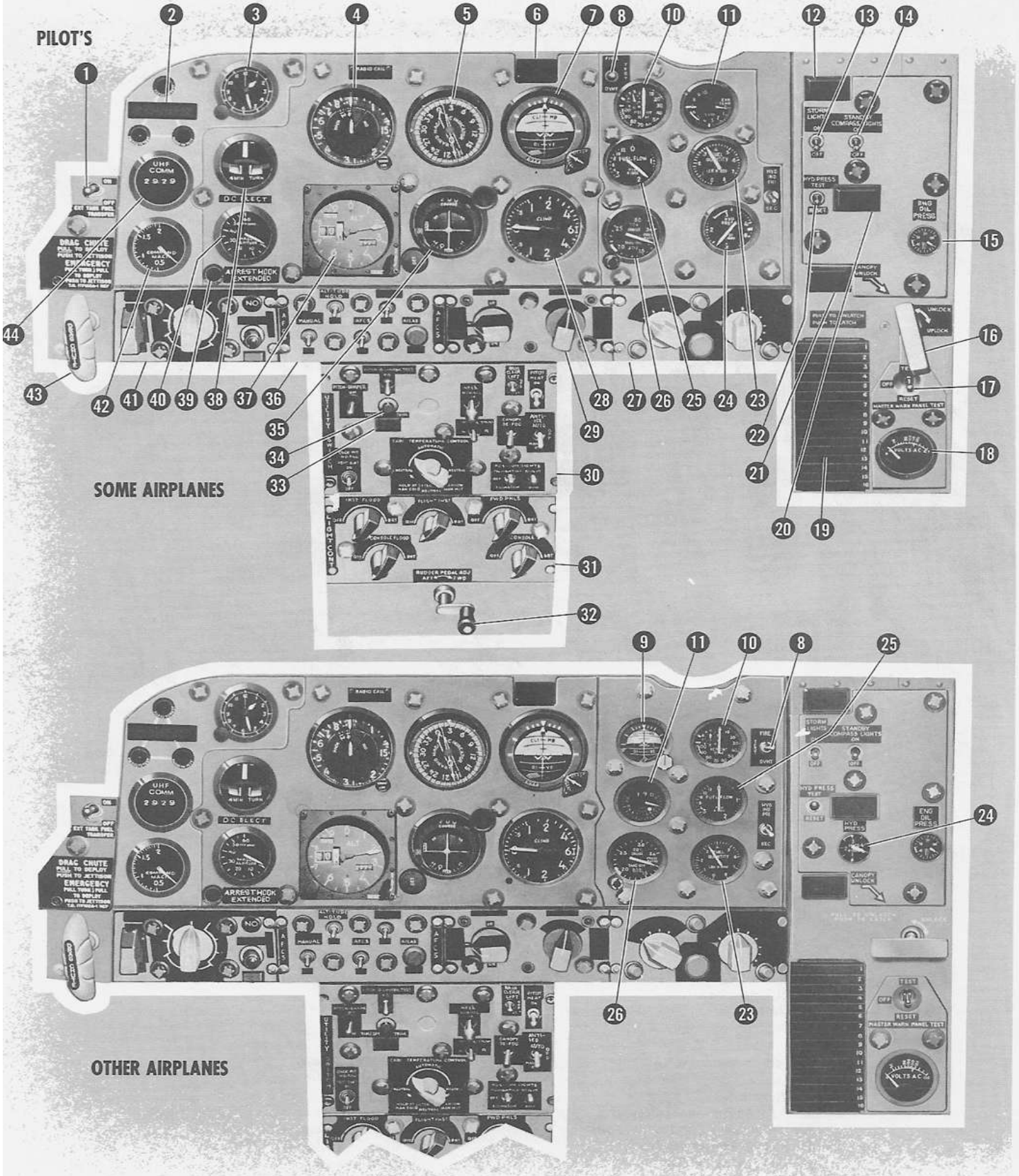
F

OTHER AIRPLANES



25. AC Voltmeter
26. Master Warning Test and Reset Switch
27. Warning Light Panel
28. Canopy Latch Handle
29. Tacan Range Indicator Panel
30. Engine Pressure Ratio Gage
31. Vertical Velocity Indicator
32. Antenna Scanning Control Panel
33. Utility Switch Panel
34. Rudder Adjustment Crank
35. Lighting Control Panel
36. Course Indicator
37. AFCS Control Panel
38. Altimeter
39. Turn-and-Slip Indicator
1 40. Target Altitude Indicator
41. Arrest Hook Light
42. Armament Control Panel
1 43. Command Mach Indicator
44. Landing Gear Emergency Extension Handle
45. Arrest Hook Release Handle
46. Wing Tank Release Button
47. Cabin Pressure Altitude Gage
48. Remote Indicator (UHF)

22, 128-2

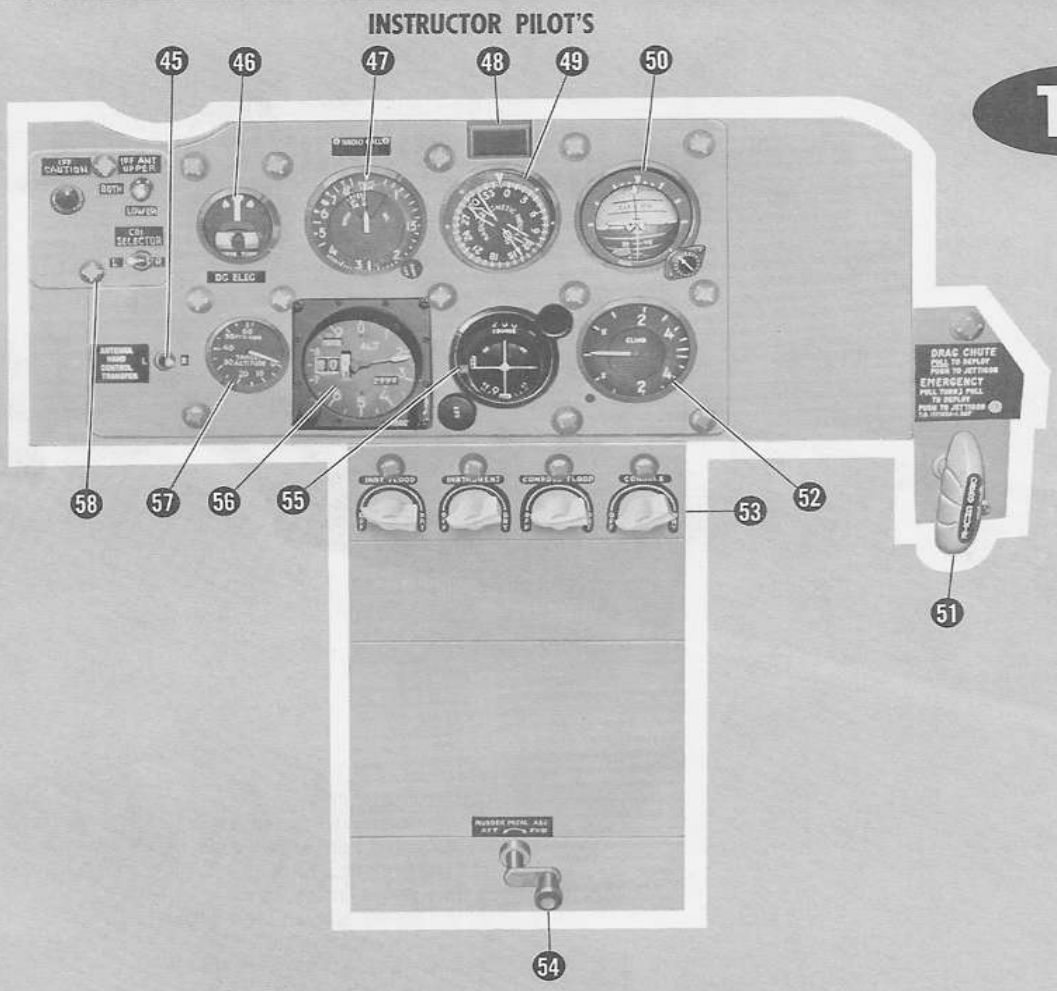


2, 128-3

Figure 1-9

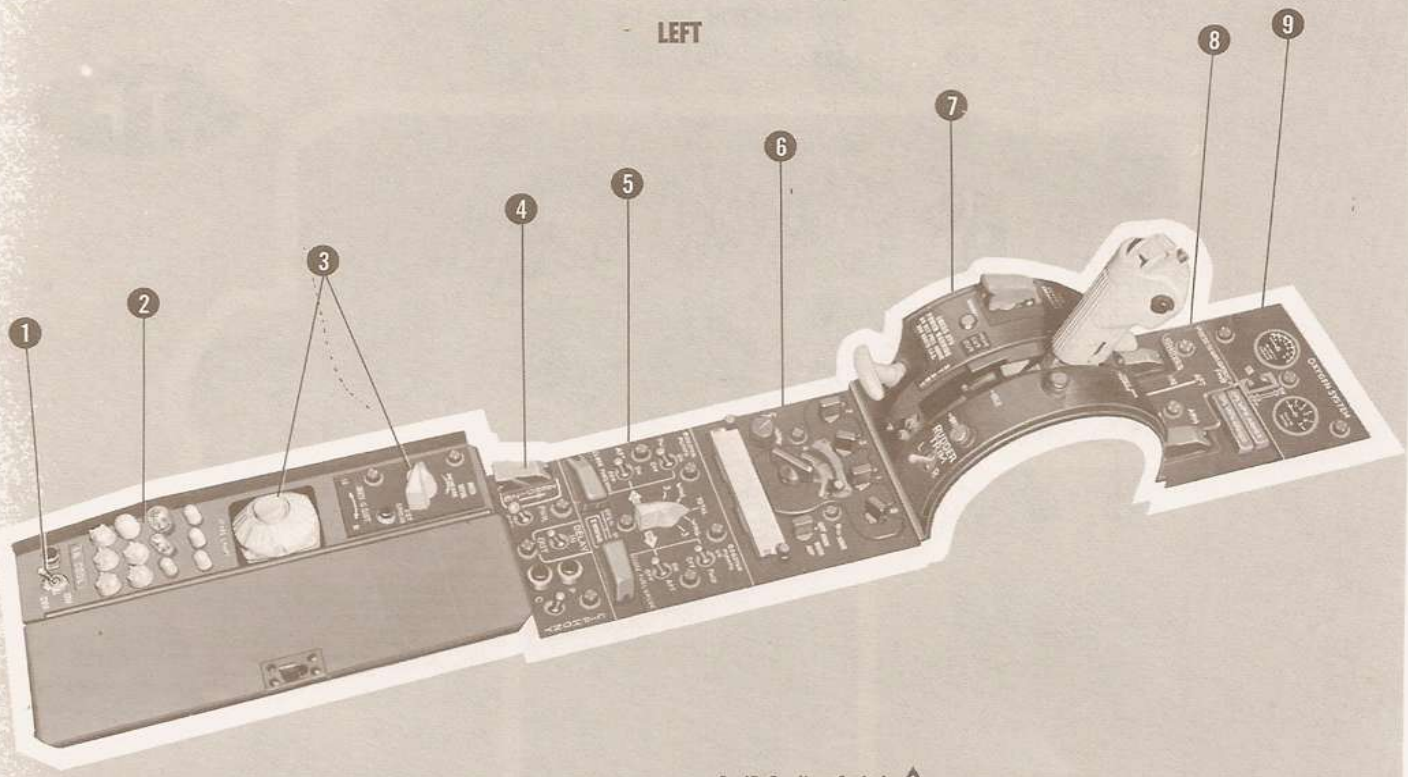
instrument panel (typical)

TF



1. External Tank Fuel Transfer Switch
2. Landing Gear Position Indicators
3. Clock
4. Airspeed-Mach Indicator
5. Radio Magnetic Indicator
6. Engine Fire and Overheat Warning Light
7. Attitude Indicator (MM-3)
8. Engine Fire Warning Test Switch
9. Standby Attitude Indicator
10. Tachometer
11. Exhaust Gas Temperature Gage
12. Master Warning Light
13. Thunderstorm Lights Switch
14. Magnetic Compass Light Switch
15. Oil Pressure Gage
16. Canopy Latch Handle
17. Master Warning Test and Reset Switch
18. AC Voltmeter
19. Warning Light Panel
20. Hydraulic Pressure Low Warning Light
21. Canopy Unlocked Warning Light
22. Hydraulic Pressure Low Warning Light Test and Reset Switch
23. Fuel Quantity Gage
24. Hydraulic Pressure Gage
25. Fuel Flow Indicator
26. Engine Pressure Ratio Gage
27. Radar Control Panel
28. Vertical Velocity Indicator
29. Antenna Scan Control Panel
30. Utility Switch Panel
31. Lighting Control Panel
32. Rudder Pedal Adjustment Crank
33. Takeoff Trim Indicator Light
34. Takeoff Trim Button
35. Course Indicator
36. AFCS Control Panel
37. Altimeter
38. Turn-and-Slip Indicator
39. Arrest Hook Light
40. Target Altitude Indicator
41. Armament Control Panel
42. Command Mach Indicator
43. Drag Chute Handle
44. Remote Indicator (UHF)
45. Antenna Hand Control Transfer Switch
46. Turn-and-Slip Indicator
47. Airspeed-Mach Indicator
48. Engine Fire and Overheat Warning Light
49. Radio Magnetic Indicator
50. Attitude Indicator (MM-3)
51. Drag Chute Handle
52. Vertical Velocity Indicator
53. Lighting Control Panel
54. Rudder Pedal Adjustment Crank
55. Course Indicator
56. Altimeter
57. Target Altitude Indicator
58. IFF Antenna Selector Panel

22, 128-4



1. IR Cooling Switch 1
2. Spare Lamps Panel
3. Mask Defog, EGT Spread 1, and Anti-G Suit Control Panels
4. Ciphony Control Panel 1
5. Fuel System Control Panel
6. UHF Command Radio Control Panel
7. Throttle Quadrant
8. Armament Auxiliary Control Panel
9. Oxygen Control Panel

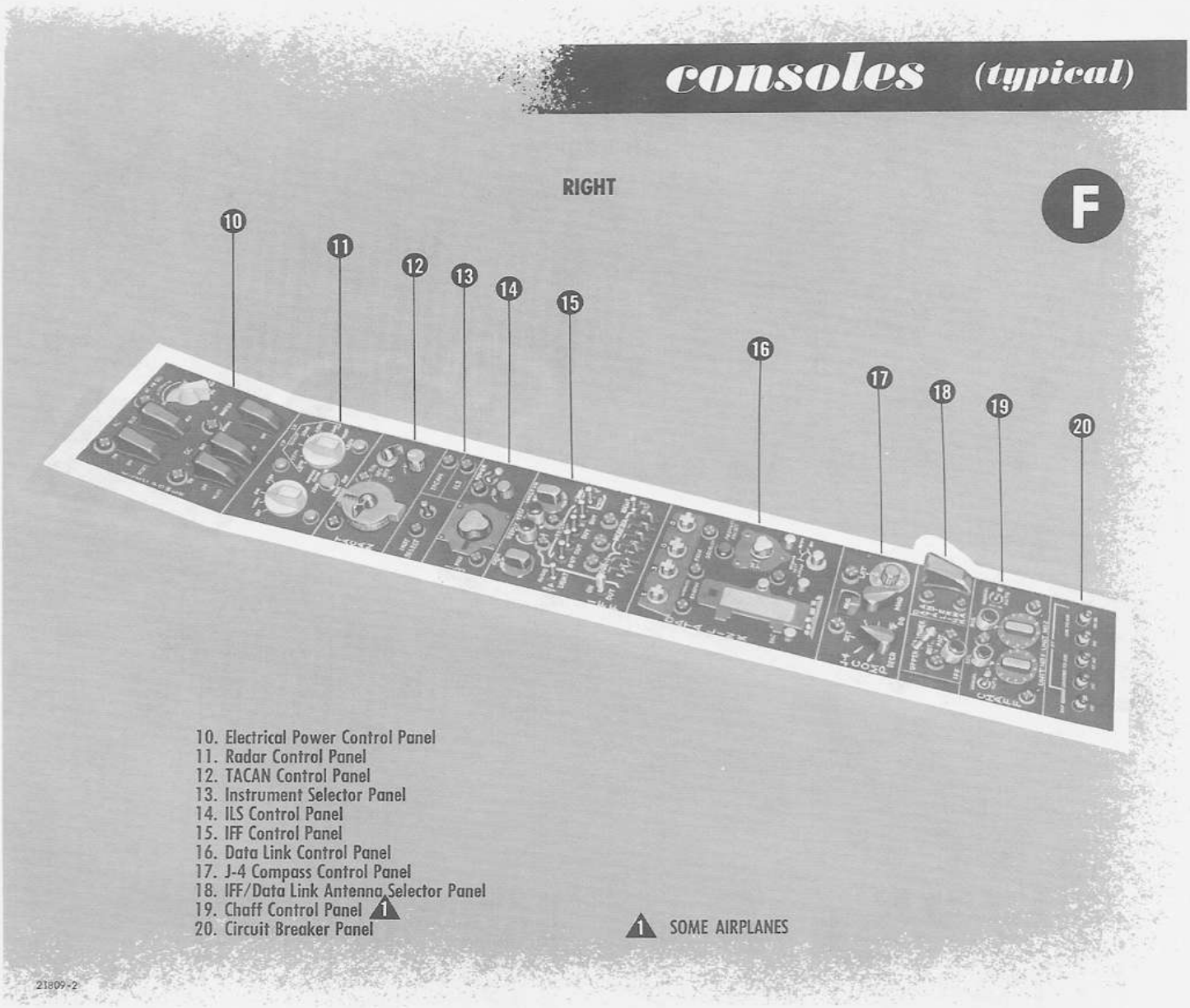
21809-1

Figure 1-10

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 (figure 1-4). 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

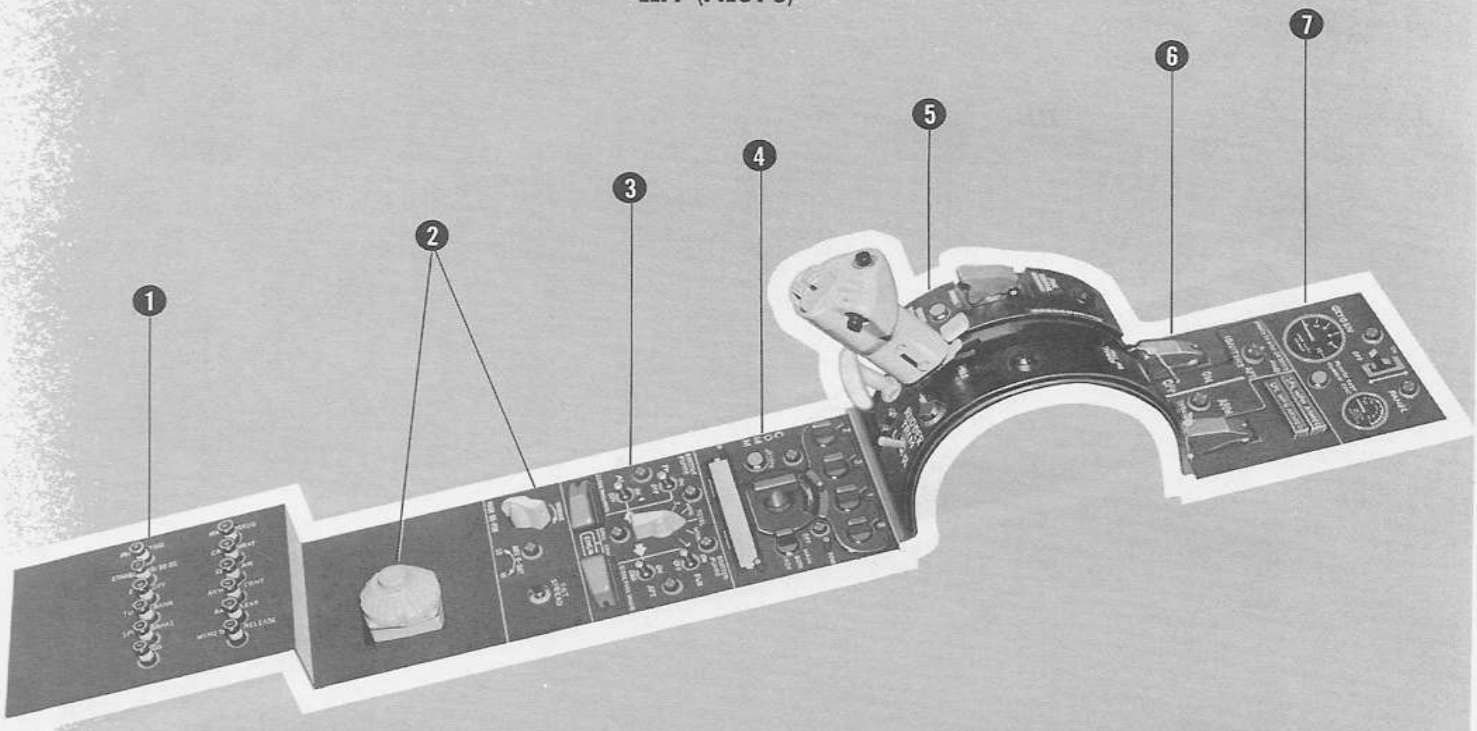
- If the engine fuel pump fails, fuel is supplied to the engine system only by the afterburner fuel pump. Afterburner operation will not be available and the exhaust nozzle will close or remain closed.
- The afterburner can be used while the engine is operating on the emergency fuel control system if 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 1-3). This metered fuel flow actuates the 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.



1. Circuit Breaker Panel
2. Mask Defog, SEGT Spread 1, and Anti-G Suit Control Panels
3. Fuel System Control Panel
4. UHF Command Radio Control Panel
5. Throttle Quadrant (P)
6. Armament Auxiliary Control Panel
7. Oxygen Control Panel (P)
8. Electrical Power Control Panel
9. Tacan Range Indicator Panel
10. Tacan Control Panel
11. Instrument Selector Panel
12. ILS Control Panel
13. IFF/SIF Control Panel
14. Data Link Control Panels
15. Interphone Control Panel (P)
16. J-4 Compass Control Panel
17. Ciphony Control Panel 1
18. IR Cooling Switch 1
19. Oxygen Control Panel (IP)
20. Throttle Quadrant (IP)
21. Interphone Control Panel (IP)
22. Mask Defog and Anti-G Suit Control Panels (IP)
23. Circuit Breaker Panel

1 SOME AIRPLANES

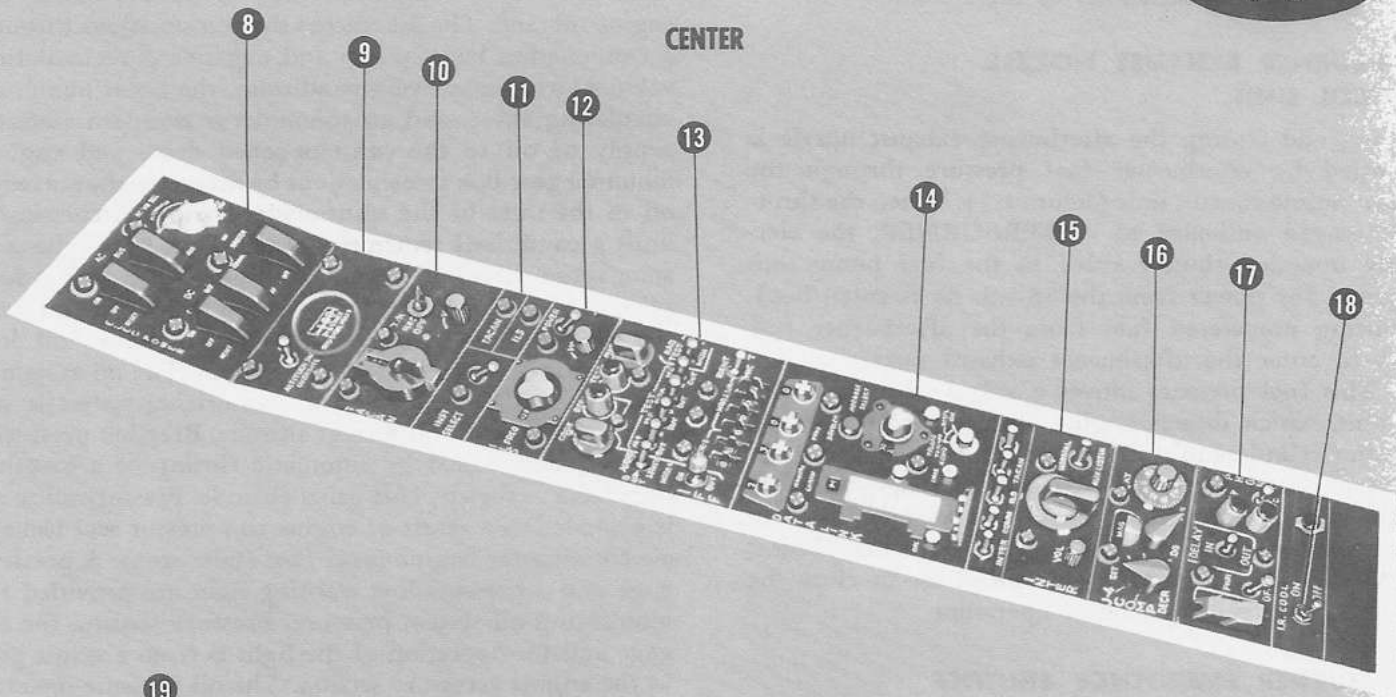
21808-1

Figure 1-11

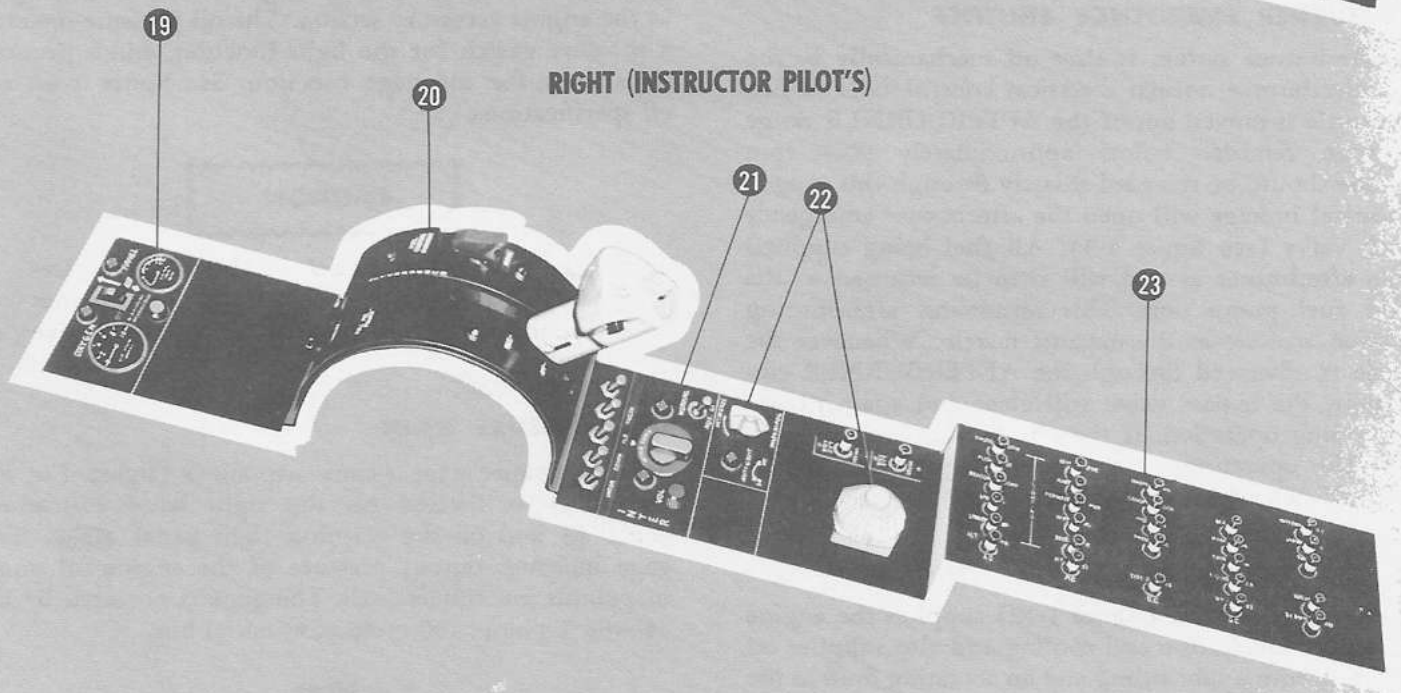
consoles (typical)

TF

CENTER



RIGHT (INSTRUCTOR PILOT'S)



21808-2

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. 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 1-3). 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.

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 shutoff valve (see figure 1-3). 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 cutoff 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 (figure 1-12) 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 two quarts 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 and engine-mounted gear box transmission oil supply is taken from a standpipe mounted in the bottom of the engine oil tank. The oil reaches the transmissions through a combination boost pump and negative g recirculating valve. During negative g conditions, the boost pump, recirculating valve, and an accumulator maintain a steady supply of oil to the constant-speed drive and engine-mounted gear box transmissions by diverting the scavenge oil to the inlet of the transmissions and recirculating it until g conditions return to normal. Since oil to the constant speed drive/engine-mounted gearbox system is supplied through the standpipe, seven quarts of oil will remain for engine operation in the event an oil loss occurs through the CSD-EMGB system. The oil system is equipped with an oil breather pressurizing system to ensure proper engine oil flow at altitude. Breather pressurizing is accomplished by automatic closing of a breather line valve as the airplane gains altitude. Pressurization air is provided as a result of engine compressor seal leakage air entering the engine inner case (lube area). A pressure gage and a pressure-low warning light are provided for monitoring oil system pressure. Pressure sensing for the gage and for operation of the light is from a single port in the engine accessory section. The oil pressure operates a pressure switch for the light function, and a pressure transmitter for the gage function. See figure 1-44, 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.

OIL PRESSURE GAGE

An oil pressure gage (some airplanes) (figure 1-8; figure 1-36) is located on the right hand instrument panel (F) and on the warning light panel (TF). This gage indicates output pressure of the engine oil pump in pounds per square inch. The gage is powered by the 26-volt, 1 phase, 400 cycle ac essential bus.

Note

Minor fluctuations of the oil pressure gage needle will normally occur and are permissible.

OIL PRESSURE-LOW WARNING LIGHT

The oil pressure-low warning light (figures 1-35 and 1-36), located on the warning light panel, illuminates

engine and constant speed drive oil supply system (typical)

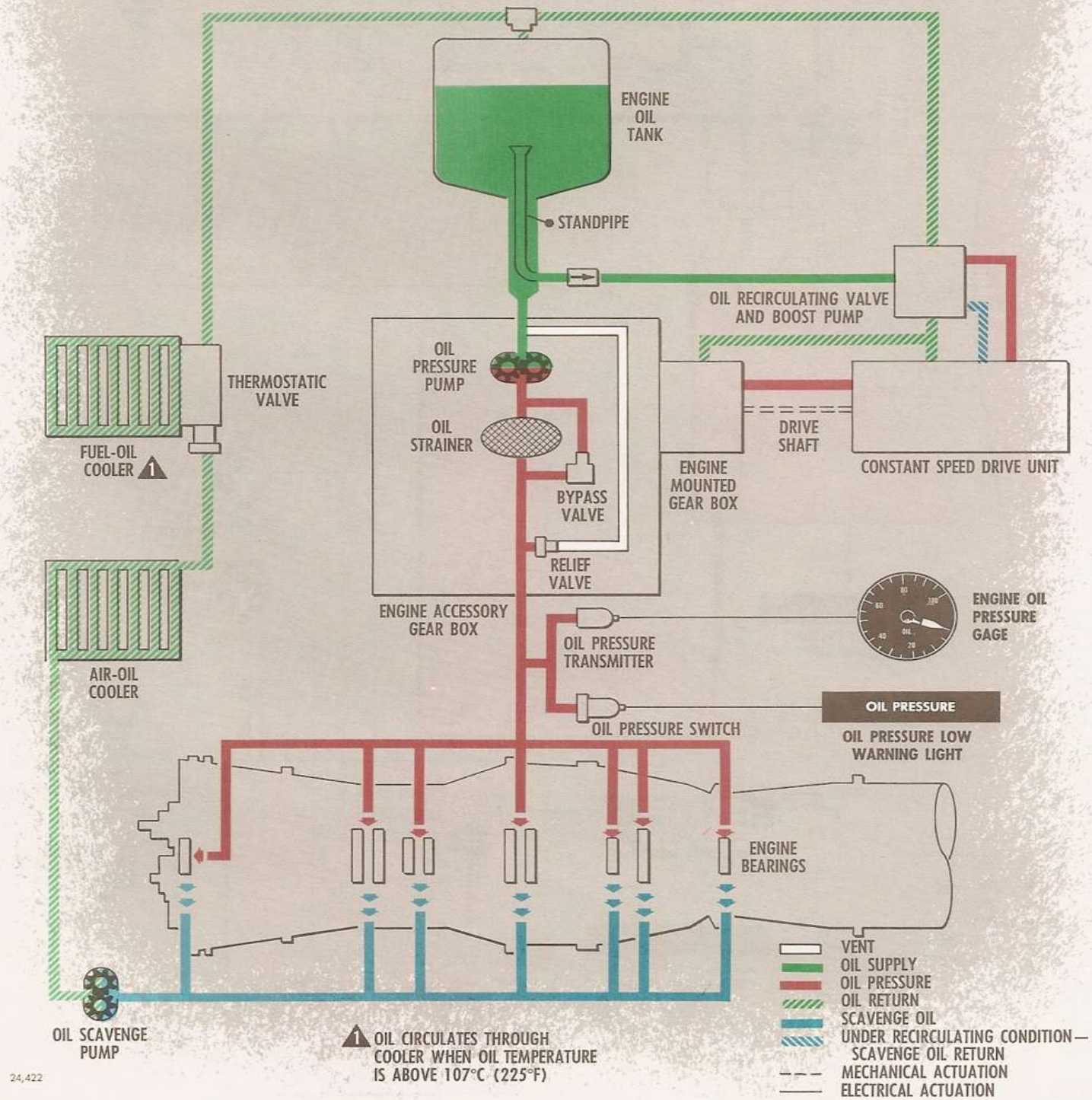


Figure 1-12

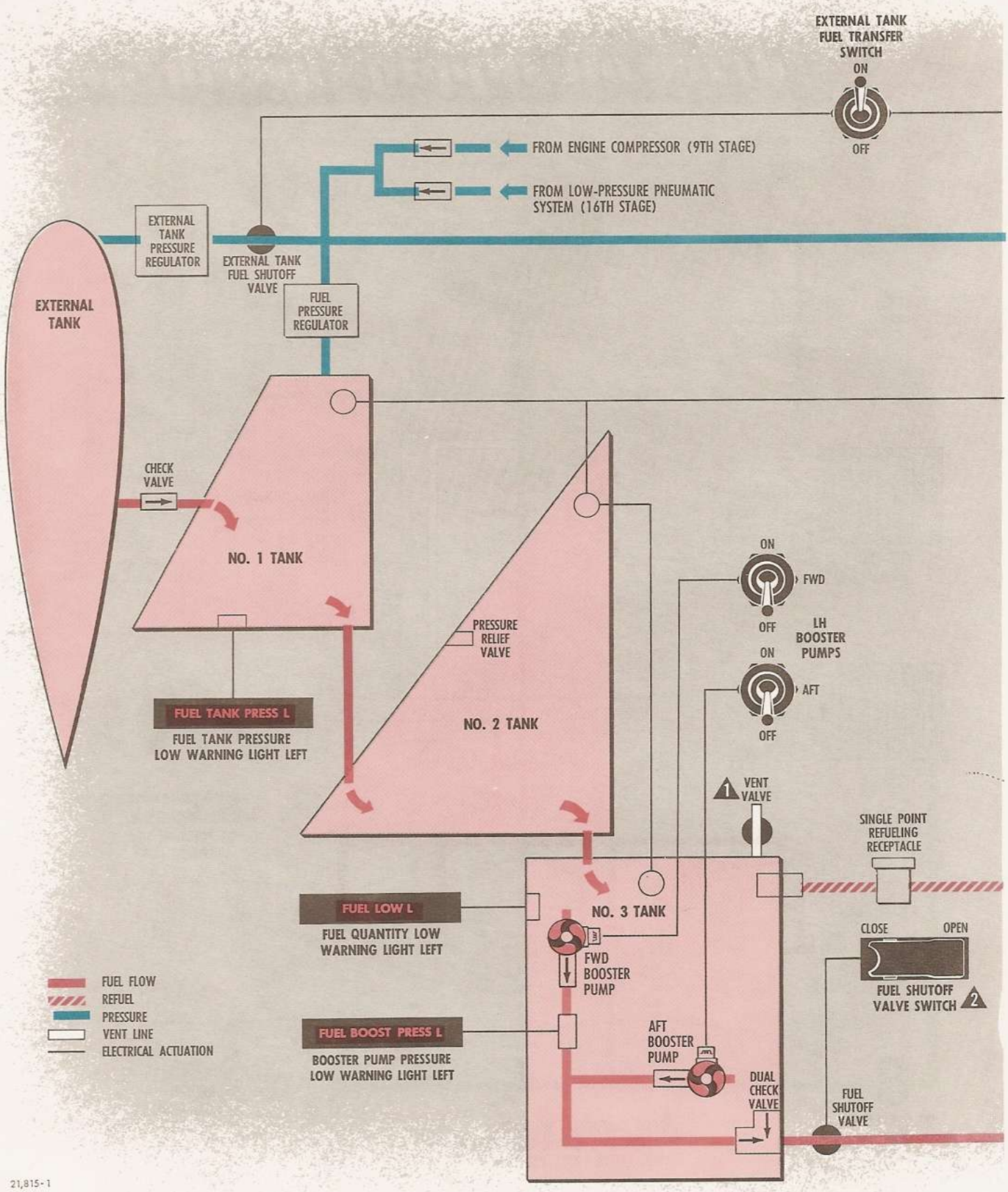
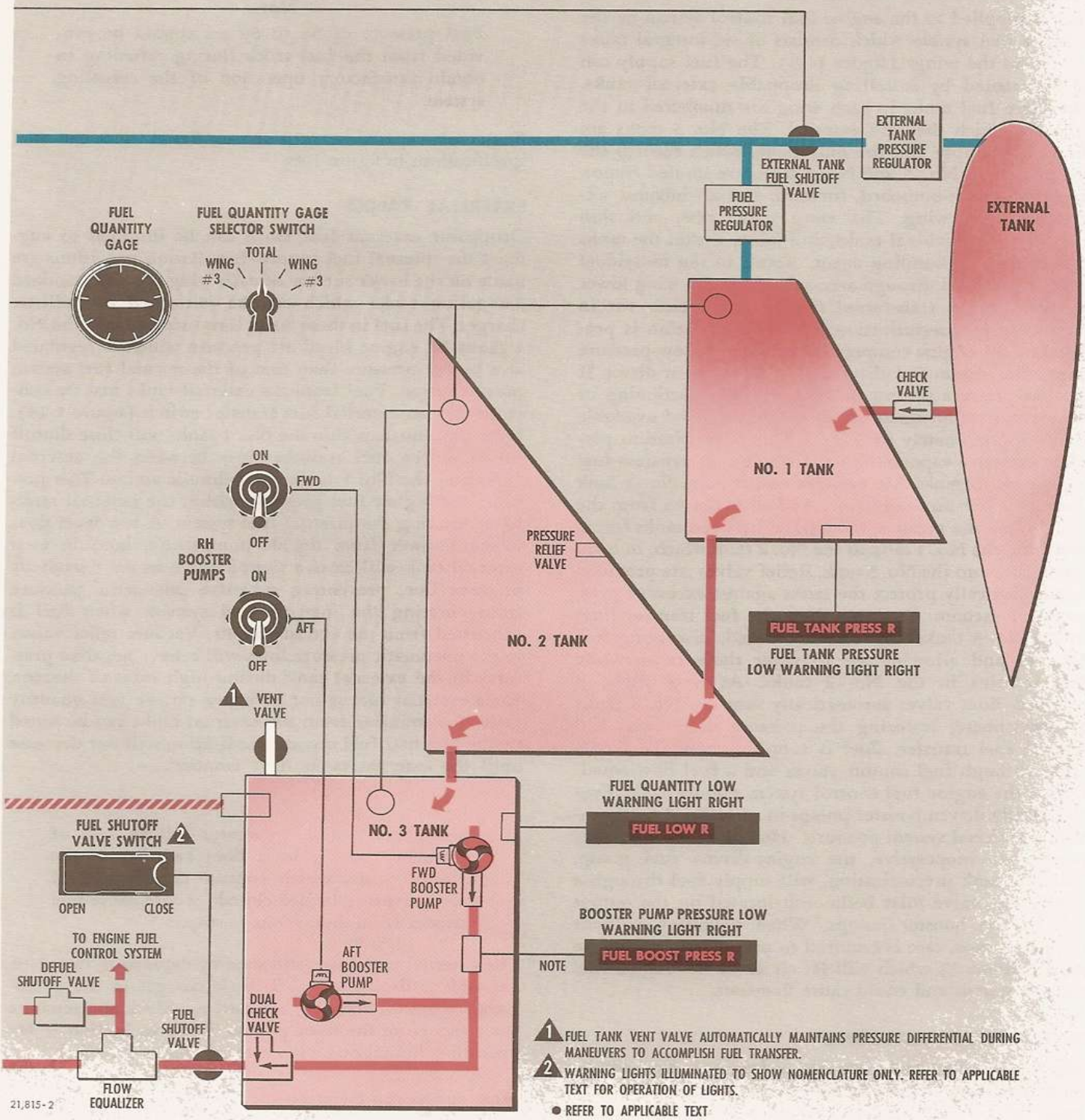

EXTERNAL TANK FUEL TRANSFER SWITCH
ON
OFF
FROM ENGINE COMPRESSOR (9TH STAGE)
FROM LOW-PRESSURE PNEUMATIC SYSTEM (16TH STAGE)
EXTERNAL TANK PRESSURE REGULATOR
EXTERNAL TANK FUEL SHUTOFF VALVE
FUEL PRESSURE REGULATOR
EXTERNAL TANK
CHECK VALVE
NO. 1 TANK
PRESSURE RELIEF VALVE
NO. 2 TANK
FUEL TANK PRESS L
FUEL TANK PRESSURE LOW WARNING LIGHT LEFT
ON
FWD
OFF
LH BOOSTER PUMPS
ON
AFT
OFF
1 VENT VALVE
SINGLE POINT REFUELING RECEPTACLE
FUEL LOW L
FUEL QUANTITY LOW WARNING LIGHT LEFT
NO. 3 TANK
FWD BOOSTER PUMP
CLOSE
OPEN
FUEL SHUTOFF VALVE SWITCH 2
FUEL BOOST PRESS L
BOOSTER PUMP PRESSURE LOW WARNING LIGHT LEFT
AFT BOOSTER PUMP
DUAL CHECK VALVE
FUEL SHUTOFF VALVE
FUEL FLOW
REFUEL
PRESSURE
VENT LINE
ELECTRICAL ACTUATION
21,815-1

Figure 1-13

fuel supply system



1 FUEL TANK VENT VALVE AUTOMATICALLY MAINTAINS PRESSURE DIFFERENTIAL DURING MANEUVERS TO ACCOMPLISH FUEL TRANSFER.

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

● REFER TO APPLICABLE TEXT

and displays "OIL PRESS" when the engine oil pressure is below approximately 35 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.

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 1-13). 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 the engine compressor through the low-pressure pneumatic system and also from the compressor direct. If the low-pressure pneumatic system is malfunctioning or is inoperative, adequate pressurization will be available above approximately 85% rpm. This pressurization prevents excessive vaporization at altitude and transfers fuel from tank to tank. 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 1-3). Two electrically driven booster pumps in each No. 3 tank provide additional system pressure. However, if the booster 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 booster pumps. When operating without booster 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 provided on some airplanes.

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 1-15 and fuel specifications in figure 1-44.

EXTERNAL TANKS

Droppable external fuel tanks can be installed 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 ballistic charges. 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. Fuel from the external tanks may be controlled by an external fuel transfer switch (figure 1-14). 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.

Note

In the event of dc generator failure, loss of power to the low-level float switches in the external tanks which control the air shutoff valves, (spring-loaded closed) would cause fuel transfer from these tanks to stop.

The external tanks are jettisoned by depressing the external tank jettison button. Ballistic charges in the pylon-installed ejector assembly are used to unlock and separate the tanks from the fixed pylons. Refer to Section V for operating limitations with external tanks installed.

FUEL BOOSTER PUMPS

Two electrically driven, submerged dual impeller booster pumps are installed in each No. 3 fuel tank (figure 1-13)

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 intakes 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 booster 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. 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 booster 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 booster 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. A separate fuel shutoff valve switch is provided for each valve. The fuel shutoff valves are normally open throughout a flight. With a malfunction where the power setting cannot be changed (such as throttle hang-up that has been experienced during penetration with throttle at or near idle setting) the only controlled fuel shutoff is with the fuel shutoff valve switches. Tests conducted to find the required time to flameout after closing the fuel shutoff valves provided the following results:

Power Setting	Time to Flameout
Full Military	5 seconds
80%	10 seconds
70%	20 seconds
Idle	25 seconds

At full military power, the fuel flow and rpm started dropping back at two seconds after the switches were closed; the fuel flow was at zero in five seconds. In all tests, the fuel flow and rpm started dropping off from two to three seconds before flameout occurred. The fuel shutoff valves are powered by the 28-volt dc essential bus.

WARNING

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

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 booster pump failure within one wing or malfunction of the flow equalizer.

FUEL SHUTOFF VALVE SWITCHES

The left- and right-hand fuel shutoff valves are controlled by individual switches located at the bottom of the fuel control panel (figure 1-14). 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 booster pumps.

FUEL BOOSTER PUMP SWITCHES

Four two-position fuel booster pump switches (figure 1-14) are located on the fuel control panel to provide individual control of the fuel booster pumps. The pairs of switches are placarded "LH Booster Pumps" and "RH Booster Pumps" with ON and OFF positions which control the booster pumps accordingly. Individual switches are placarded "Fwd" and "Aft" to identify the specific pump within each fuel tank. All fuel booster pump 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 1-29) 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 depressed, electrically fired ballistic charges unlock and separate the tanks from the pylons. On TF airplanes, the button is located on the pilot's landing gear control panel only. The jettison circuits receive power from the 28-volt dc essential bus.

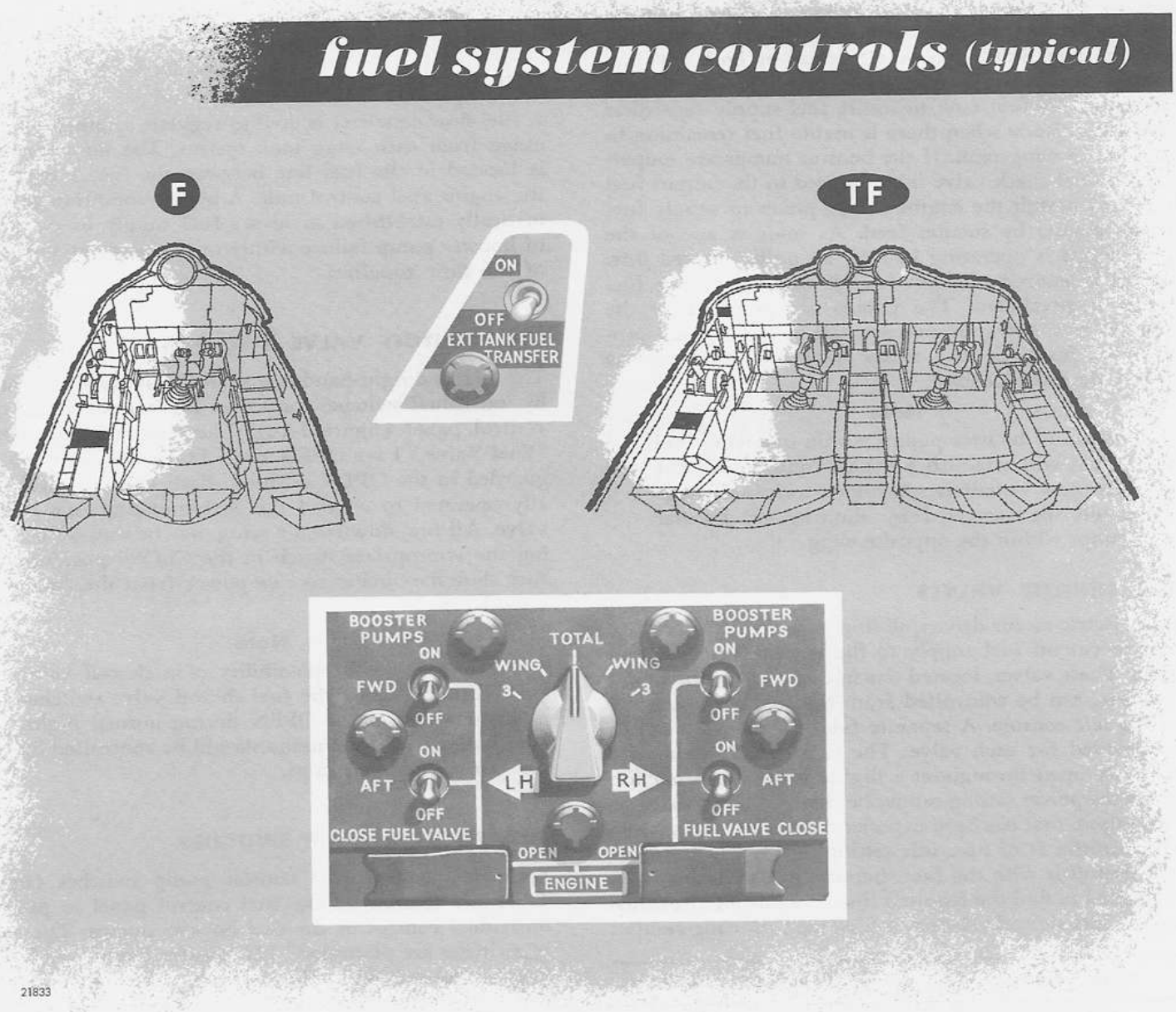


Figure 1-14

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 (figure 1-14) is installed 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.

EXTERNAL TANK GROUND SAFETY PIN

The external tank ground safety pin can be installed from either side of the tank pylon into the tank ejector rack.

fuel quantity data table (typical)

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

STANDARD ATMOSPHERE

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

21,816

Figure 1-15

WARNING

The external tank ground safety pins shall be installed and removed from the inboard side of the pylons. Removing the pins from the inboard side will provide protection against injury if a tank jettisons when the pin is pulled out.

The safety pin is a ball-locking type pin. It safeties the ballistic jettison system and provides a mechanical lock to prevent inadvertent separation of the tank from the rack. The pin is equipped with a remove-before-flight streamer.

FUEL QUANTITY GAGE SELECTOR SWITCH

The fuel quantity gage selector switch (figure 1-14), located on the fuel control panel, connects the fuel quantity gage to various tank quantity indicating circuits. 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 115-volt ac essential bus.

Note

If the fuel quantity gage selector switch is placed in an intermediate position, the fuel quantity gage will indicate zero.

FUEL QUANTITY GAGE

The fuel quantity gage (figure 1-7) 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.

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 ac essential bus.

FUEL QUANTITY-LOW WARNING LIGHTS

Two fuel quantity-low warning lights (figure 1-36), 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 on a standard day). The fuel quantity-low warning circuits receive power from the 28-volt dc essential bus.

FUEL BOOSTER PRESSURE-LOW WARNING LIGHTS

Two fuel booster pressure-low warning lights (figure 1-36), 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 if a fuel shutoff valve is not in fully open position. The appropriate light will remain illuminated until the booster pump pressure exceeds 12 psi if the illumination was due to booster pump failure. The warning light will not illuminate if only one booster pump fails. The fuel booster 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 (figure 1-36), 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 shortly 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 primary dc power source is the 28-volt, 200-ampere hour generator. The generator supplies power to dc powered equipment through the cockpit generator switch. From the switch, power is routed to the dc essential bus, the dc nonessential bus through the nonessential bus tie relay, and to the emergency dc bus through the dc bus changeover relay. The dc essential bus is connected to a 24-volt, 24 ampere hour battery (36 ampere hour battery on some airplanes*) through a battery switch having power positions of ON or TR FAIL. With the switch ON, the battery will charge, providing generator voltage exceeds battery voltage. A transformer-rectifier, powered by the ac essential bus, is provided as a standby power source for the emergency dc bus. With generator failure, the nonessential bus tie relay opens and the nonessential dc bus becomes inoperative. The emergency dc bus changeover relay opens and allows the transformer rectifier to power the emergency dc bus. The battery powers the essential dc bus and illuminates the dc power failure warning light (providing the battery switch is in ON or TR FAIL). If either the transformer rectifier or ac essential bus are inoperative, the battery switch may be placed in the TR FAIL position to re-energize the emergency dc bus changeover relay, allowing the battery power to flow from the dc essential bus to the dc emergency bus.

AC ELECTRICAL POWER DISTRIBUTION

The primary ac power source (figure 1-17) is the 115/200-volt, 400 cycle, 3-phase, 30-kva generator. The generator supplies power to ac powered equipment through the cockpit generator switch. From the switch, power is routed through the dc actuated ac power control relay to the ac nonessential bus, and through the emergency

**Airplanes modified by TCTO 1F-102-1215.*

ac power control relay to the three ac essential buses and the transformer rectifier. Emergency ac power is supplied by the ac emergency generator which is driven by secondary hydraulic system pressure. With main ac generator failure, all ac powered equipment becomes inoperative and the ac power failure warning light illuminates. The ac emergency generator must be manually selected. Selection of the ac emergency generator energizes and connects the generator to the three ac essential busses and will supply power to the transformer rectifier.

ELECTRICALLY OPERATED EQUIPMENT

See figures 1-16 and 1-17 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.
- Airplane ac power will not be available while external power is connected to the airplane.

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.

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 1-19) located at the aft end of the left and right consoles and at the forward side of the left console. Figure 1-20 shows the direct effect of certain circuit breakers on the operation of airplane components or systems. Fuses are used in the booster pump motor circuit, and spare fuses for this circuit are located adjacent to the booster pump relays on the aft side of each main wheel well.

MASTER SWITCH

The guarded master switch (figure 1-18), 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 three positions: NORMAL, OFF, and TRIP. In the NORMAL position, normal generator and battery power is supplied to the airplane buses. In the OFF position battery and emergency ac generator power is removed from the buses but the ac and dc generators are still on the line. The TRIP position is a momentary position spring-loaded to OFF, which will remove power from the ac and dc generator fields, thus lessening the danger of fire in an emergency situation. The switch receives power from the 28-volt dc essential bus.

BATTERY SWITCH

The guarded battery switch (figure 1-18), located on the electrical power control panel, controls the battery circuits. The switch is placarded "Bat" and has three positions: ON, OFF, and TR FAIL. The ON and OFF positions control the circuit accordingly. The TR FAIL position energizes the emergency dc bus changeover relay directly from the battery. This 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 the emergency ac generators have failed. The battery control circuit receives power directly from the battery.

CAUTION

With dc generator and transformer-rectifier unit failed, complete electrical failure will occur when battery power is depleted.

DC GENERATOR SWITCH

The guarded dc generator switch (figure 1-18), 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 TRIP, 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 1-18), located on the electrical power control panel, is used to control ac

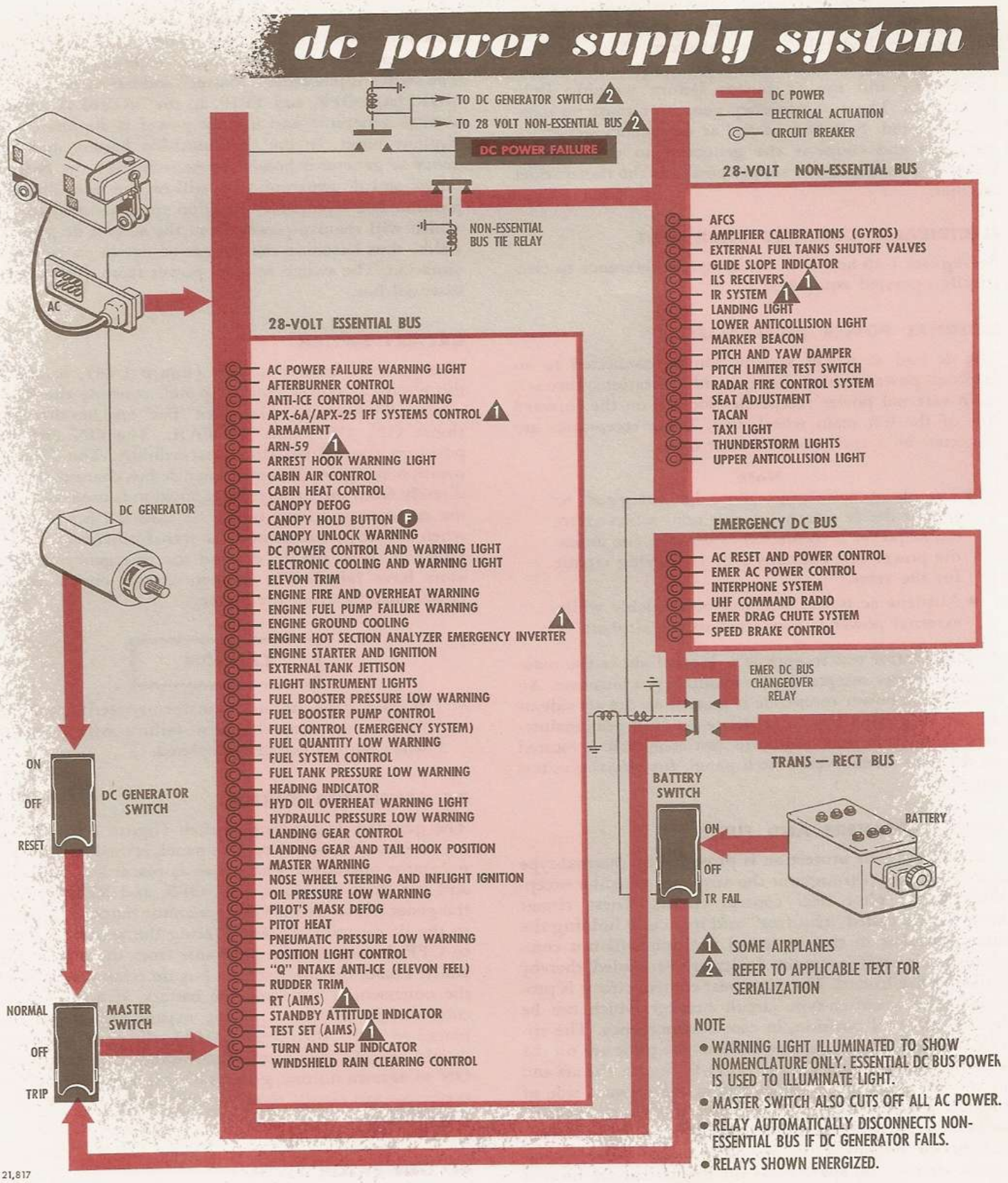
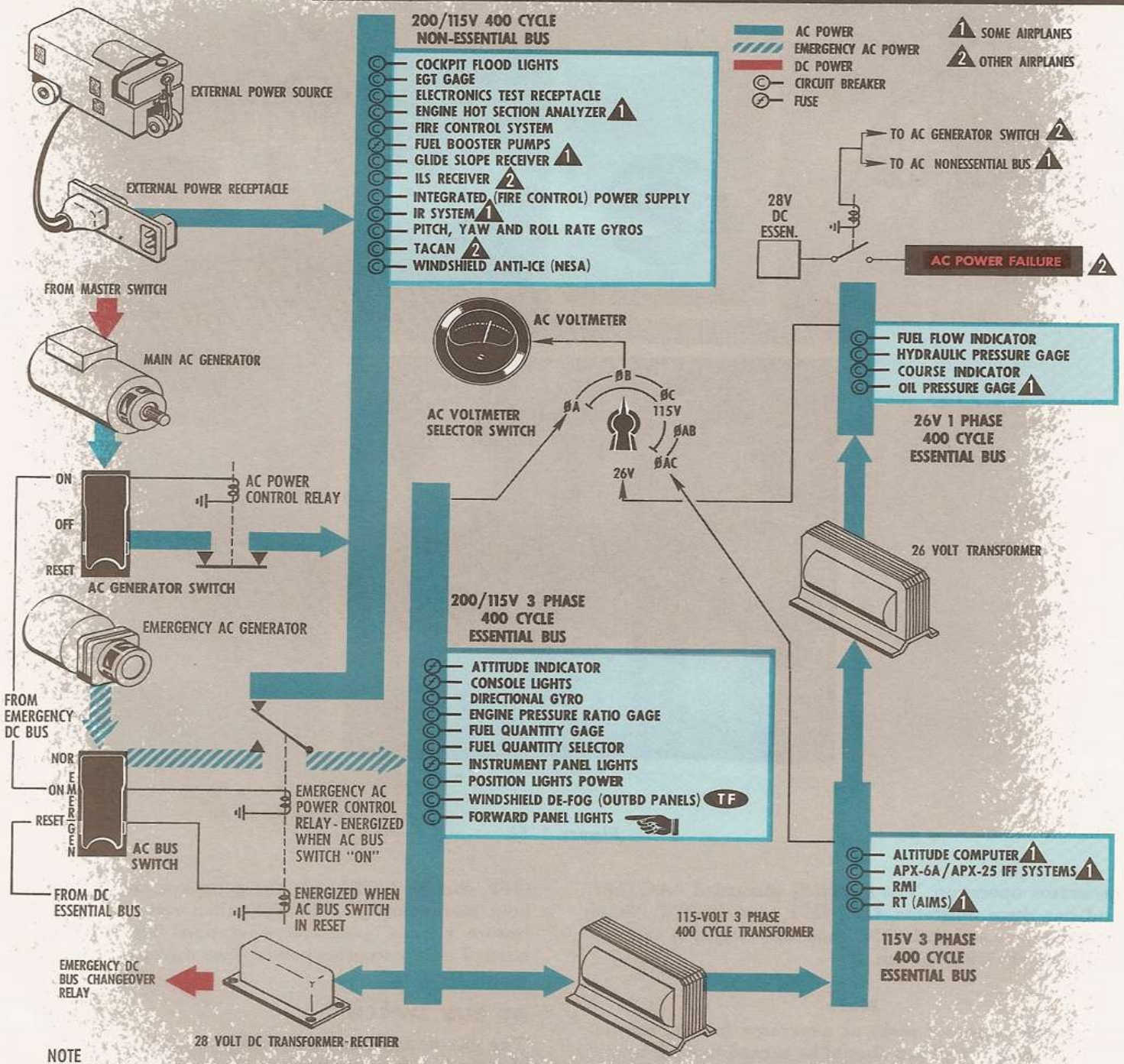

dc power supply system
TO DC GENERATOR SWITCH 2
TO 28 VOLT NON-ESSENTIAL BUS 2
DC POWER FAILURE
DC POWER
ELECTRICAL ACTUATION
CIRCUIT BREAKER
28-VOLT NON-ESSENTIAL BUS
NON-ESSENTIAL BUS TIE RELAY
AFCS
AMPLIFIER CALIBRATIONS (GYROS)
EXTERNAL FUEL TANKS SHUTOFF VALVES
GLIDE SLOPE INDICATOR
ILS RECEIVERS 1
IR SYSTEM 1
LANDING LIGHT
LOWER ANTICOLLISION LIGHT
MARKER BEACON
PITCH AND YAW DAMPER
PITCH LIMITER TEST SWITCH
RADAR FIRE CONTROL SYSTEM
SEAT ADJUSTMENT
TACAN
TAXI LIGHT
THUNDERSTORM LIGHTS
UPPER ANTICOLLISION LIGHT
AC
28-VOLT ESSENTIAL BUS
AC POWER FAILURE WARNING LIGHT
AFTERBURNER CONTROL
ANTI-ICE CONTROL AND WARNING
APX-6A/APX-25 IFF SYSTEMS CONTROL 1
ARMAMENT
ARN-59 1
ARREST HOOK WARNING LIGHT
CABIN AIR CONTROL
CABIN HEAT CONTROL
CANOPY DEFOG
CANOPY HOLD BUTTON F
CANOPY UNLOCK WARNING
DC POWER CONTROL AND WARNING LIGHT
ELECTRONIC COOLING AND WARNING LIGHT
ELEVON TRIM
ENGINE FIRE AND OVERHEAT WARNING
ENGINE FUEL PUMP FAILURE WARNING
ENGINE GROUND COOLING
ENGINE HOT SECTION ANALYZER EMERGENCY INVERTER 1
ENGINE STARTER AND IGNITION
EXTERNAL TANK JETTISON
FLIGHT INSTRUMENT LIGHTS
FUEL BOOSTER PRESSURE LOW WARNING
FUEL BOOSTER PUMP CONTROL
FUEL CONTROL (EMERGENCY SYSTEM)
FUEL QUANTITY LOW WARNING
FUEL SYSTEM CONTROL
FUEL TANK PRESSURE LOW WARNING
HEADING INDICATOR
HYD OIL OVERHEAT WARNING LIGHT
HYDRAULIC PRESSURE LOW WARNING
LANDING GEAR CONTROL
LANDING GEAR AND TAIL HOOK POSITION
MASTER WARNING
NOSE WHEEL STEERING AND INFLIGHT IGNITION
OIL PRESSURE LOW WARNING
PILOT'S MASK DEFOG
PITOT HEAT
PNEUMATIC PRESSURE LOW WARNING
POSITION LIGHT CONTROL
"Q" INTAKE ANTI-ICE (ELEVON FEEL)
RUDDER TRIM
RT (AIMS) 1
STANDBY ATTITUDE INDICATOR
TEST SET (AIMS) 1
TURN AND SLIP INDICATOR
WINDSHIELD RAIN CLEARING CONTROL
DC GENERATOR
EMERGENCY DC BUS
AC RESET AND POWER CONTROL
EMER AC POWER CONTROL
INTERPHONE SYSTEM
UHF COMMAND RADIO
EMER DRAG CHUTE SYSTEM
SPEED BRAKE CONTROL
EMER DC BUS CHANGEOVER RELAY
TRANS — RECT BUS
BATTERY SWITCH
ON
OFF
TR FAIL
BATTERY
ON
OFF
RESET
DC GENERATOR SWITCH
NORMAL
OFF
TRIP
MASTER SWITCH
1 SOME AIRPLANES
2 REFER TO APPLICABLE TEXT FOR SERIALIZATION
NOTE
• WARNING LIGHT ILLUMINATED TO SHOW NOMENCLATURE ONLY. ESSENTIAL DC BUS 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.
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Figure 1-16

ac power supply system



NOTE

- WHEN THE AC SWITCH IS IN THE "NORM" POSITION, THE ESSENTIAL AND NONESSENTIAL BUSES ARE CONNECTED TO THE MAIN AC GENERATOR. MOVING THE SWITCH TO THE "ON" POSITION CONNECTS THE EMERGENCY AC GENERATOR TO THE ESSENTIAL BUSES, IF THE EMERGENCY DC BUS HAS POWER.
- AC CONTROL CIRCUITS USE DC POWER.

WARNING LIGHTS ILLUMINATED TO SHOW NOMENCLATURE ONLY. ESSENTIAL DC BUS POWER IS USED TO ILLUMINATE LIGHTS.

Figure 1-17

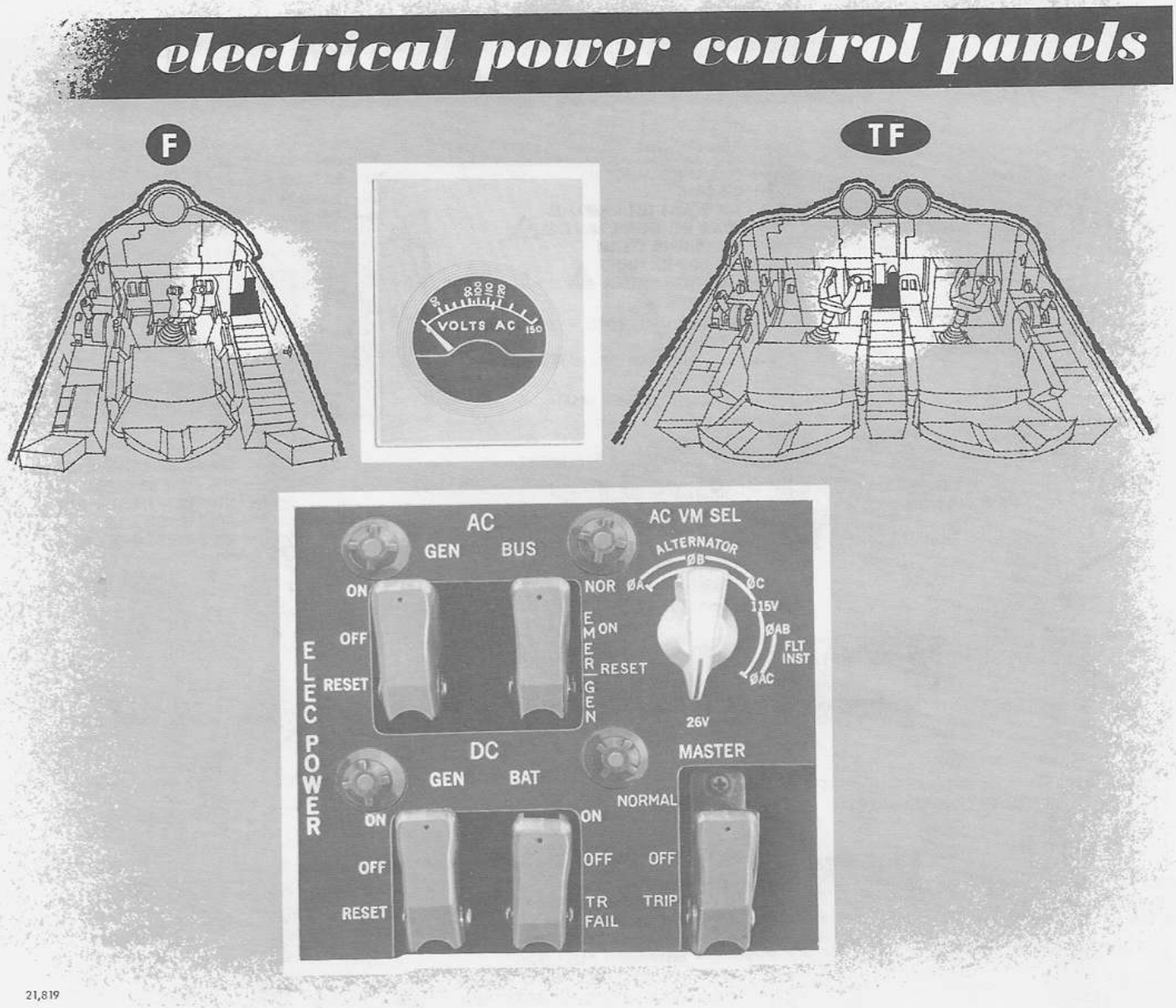


Figure 1-18

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.

Note

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 TRIP, then 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 emergency dc bus.

AC BUS SWITCH

The guarded ac bus switch (figure 1-18), located on the electrical power control panel, is used to energize the hydraulically driven emergency ac generator. The 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. In the ON position, the emergency generator shutoff valve and the emergency ac power control relay are energized (if the emergency dc

bus is energized) to connect the emergency ac generator to the ac essential buses. The ON position of the switch receives power through the emergency dc bus. When emergency dc bus power is not available for actuation of the emergency ac generator in the ON position, actuation of the switch to the spring-loaded RESET position completes a circuit from the dc essential bus to the emergency generator shutoff valve and the emergency ac power control relay. The completed circuit starts the emergency ac generator, ties it to the ac essential buses, and re-energizes the emergency dc bus through the transformer-rectifier. When released, the ac bus switch will return to the ON position which ties the emergency generator shutoff valve and the emergency ac power control relay to the emergency dc bus. The RESET position receives power directly from the dc essential bus.

Note

If the switch is in the ON position and secondary hydraulic failure occurs, the switch must be returned to NORMAL for the ac generator to supply power to the ac essential bus.

AC VOLTMETER AND SELECTOR SWITCH

The ac voltmeter (figures 1-8 and 1-9), 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 1-18), 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 (figure 1-36), 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 F* and TF** airplanes, the warning light will illuminate 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 switches on the electrical control panel. In the event of both dc generator and battery failure, dc power will be provided by the transformer-rectifier for the control of the ac system. If the transformer-rectifier fails, complete electrical failure will result.

AC POWER FAILURE WARNING LIGHT

The dc power failure warning light (figure 1-36), 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 F* and TF** airplanes, the warning light will illuminate when the battery switch is ON or the external power unit is connected for starting and will remain illuminated 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 1-21) 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 pump 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. A hydraulic fluid overheat warning light is also provided to warn of overheated hydraulic 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 1-44 for hydraulic fluid specification.

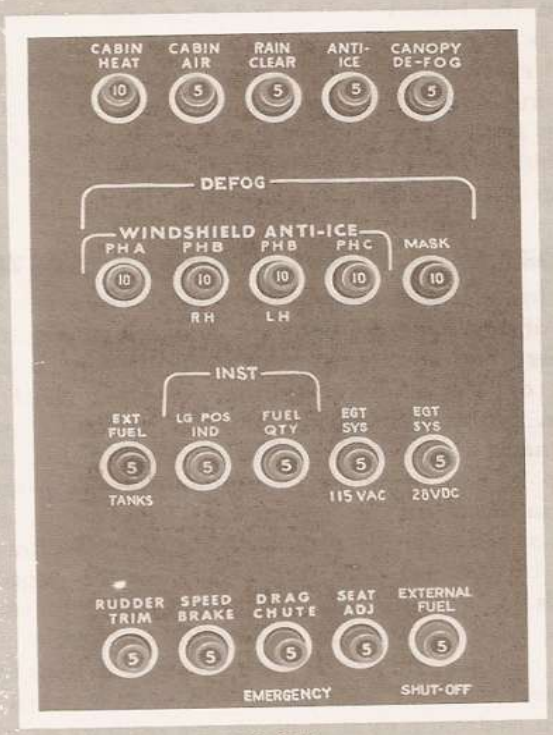
PRIMARY HYDRAULIC SYSTEM

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. 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 except when engine is operating below 80% rpm. Pressure is then furnished by the high-pressure system. The piston-type accumulator contains a pressure gage (not accessible in flight) for ground checking the preload pressure.

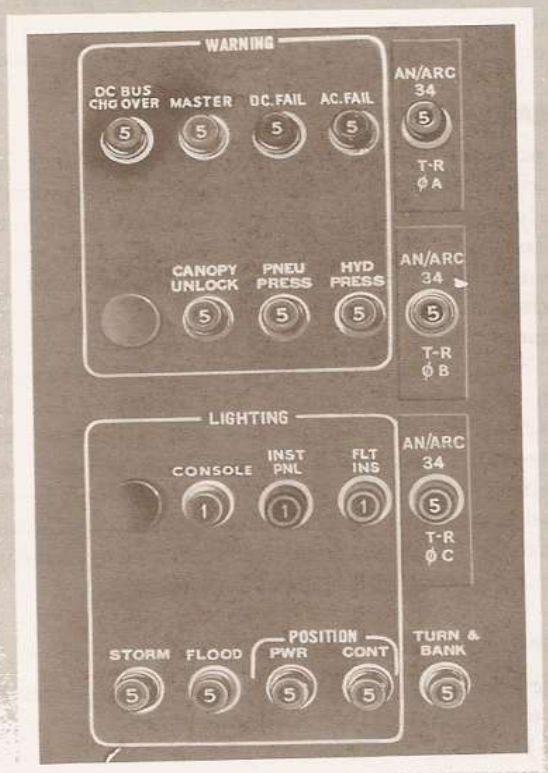
SECONDARY HYDRAULIC SYSTEM

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

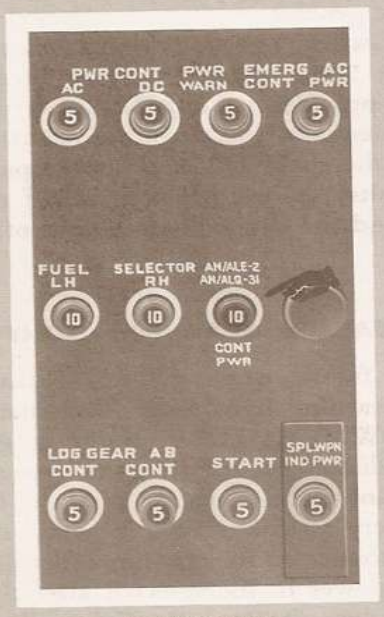
**AF 56-1275 thru -1316, -1332, & on.*
***AF 55-4057 & on.*



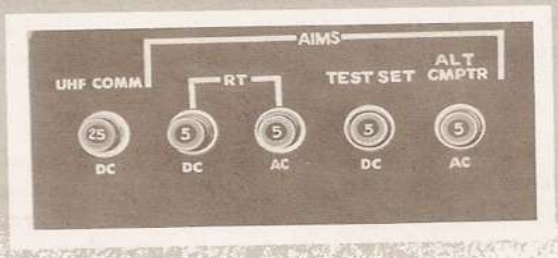
LH AFT



RH AFT



LH FORWARD



RH FORWARD

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F

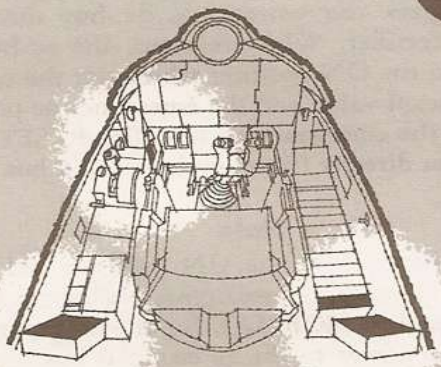
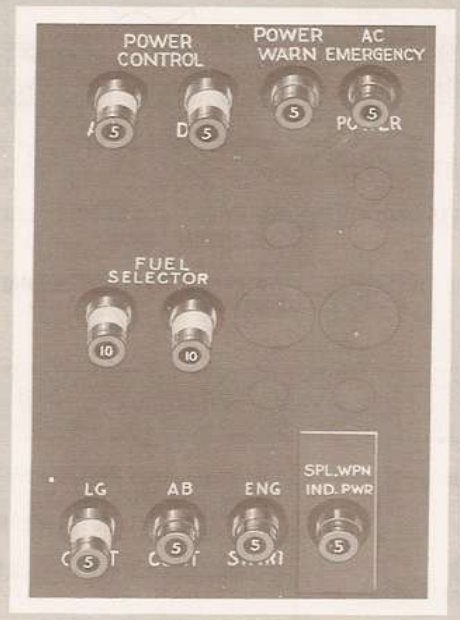


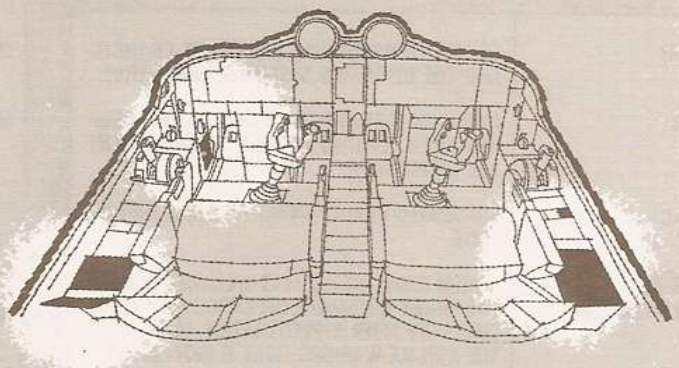
Figure 1-19

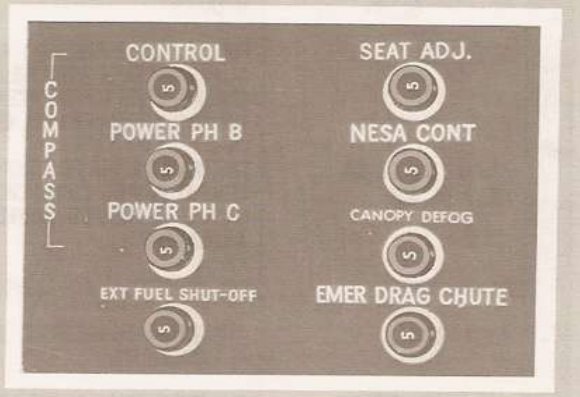
circuit breaker panels (typical)

TF



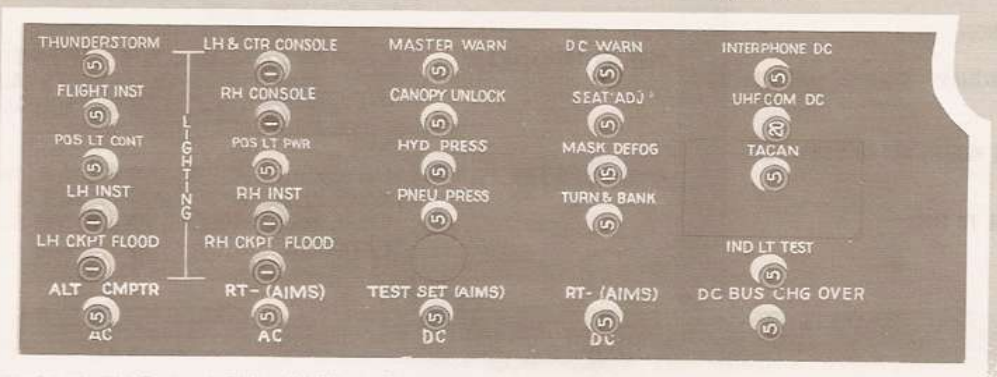
LH FORWARD





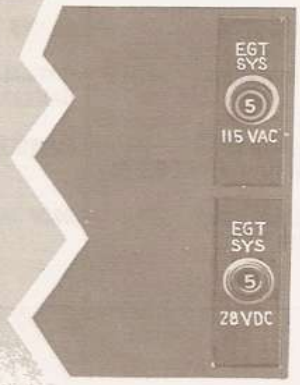


LH AFT



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RH AFT



RH CONSOLE

CIRCUIT BREAKER	EFFECT ON CIRCUIT WITH CIRCUIT BREAKER OUT
LEFT HAND AFT PANEL	
CABIN HEAT	COCKPIT TEMPERATURE CANNOT BE CHANGED AND THE DEFOGGING SYSTEM IS INOPERATIVE.
CABIN AIR	CABIN AIR SYSTEM IS SHUT OFF. THE JET PUMP AND THE ELECTRONIC COOLING WARNING LIGHT ARE INOPERATIVE.
RAIN CLEAR	WINDSHIELD RAIN CLEARING SYSTEM IS INOPERATIVE.
ANTI-ICE F ANTI-ICE CONT TF	SYSTEM TURNS OFF IF IT WAS OPERATING IN MANUAL. UNDER AUTOMATIC OPERATION THE SYSTEM TURNS OFF AFTER APPROXIMATELY 60 SECONDS. THE ANTI-ICE WARNING LIGHT IS INOPERATIVE.
CANOPY DEFOG	CANOPY DEFOG SYSTEM IS INOPERATIVE.
WINDSHIELD ANTI-ICE ØA, ØB(LH), ØB(RH), & ØC F	NESA PANEL HEATING IS INOPERATIVE. IF THE RAIN CLEAR SWITCH IS IN STBY, THE WINDSHIELD RAIN CLEARING SYSTEM OPERATES WHEN ØA CIRCUIT BREAKER IS OUT.
NESA CONT TF	NESA PANEL HEATING IS INOPERATIVE.
EXT FUEL TANKS F WING TANK RELEASE TF	EXTERNAL FUEL TANKS WILL NOT JETTISON.
LG POS IND F LG POS TF	THE THREE GREEN LANDING GEAR POSITION INDICATOR LIGHTS, LANDING GEAR WARNING, AND ARREST HOOK LIGHTS ARE INOPERATIVE.
FUEL QTY	THE FUEL QUANTITY INDICATOR NEEDLE IS NOT DAMPENED.
EGT SYS (115 VAC) F	THE ENGINE HOT SECTION ANALYZER SHOULD STILL OPERATE ON DC POWER.
EGT SYS (28 VDC) F	THE ENGINE HOT SECTION ANALYZER SHOULD STILL OPERATE ON AC POWER.
RUDDER TRIM	RUDDER TRIM IS INOPERATIVE.
SPEED BRAKES	NORMAL AND AUXILIARY SPEED BRAKES ARE INOPERATIVE.
DRAG CHUTE F EMER DRAG CHUTE TF	EMERGENCY SPEED BRAKES ARE INOPERATIVE.
EXTERNAL FUEL SHUTOFF	DROP TANKS CEASE TO FEED.

CIRCUIT BREAKER	EFFECT ON CIRCUIT WITH CIRCUIT BREAKER OUT
RIGHT HAND CONSOLE PANEL	
EGT SYS (115 VAC) TF	THE ENGINE HOT SECTION ANALYZER SHOULD STILL OPERATE ON DC POWER.
EGT SYS (28 VDC) TF	THE ENGINE HOT SECTION ANALYZER SHOULD STILL OPERATE ON AC POWER.
RIGHT HAND AFT PANEL	
DC BUS CHG OVER	THE TRANSFORMER-RECTIFIER UNIT IS CONNECTED TO THE EMERGENCY DC BUS, PROVIDING THE BATTERY SWITCH IS NOT IN THE TR FAIL POSITION.
MASTER F MASTER WARN TF	THE MASTER WARNING LIGHT IS INOPERATIVE.
DC FAIL F DC WARN TF	THE DC POWER FAILURE WARNING LIGHT IS INOPERATIVE.
AC FAIL F	THE AC POWER FAILURE WARNING LIGHT WILL ILLUMINATE EVEN WITH THE AC GENERATOR OPERATING. (THIS CIRCUIT BREAKER IS LOCATED IN THE NOSE WHEEL WELL ON TF AIRPLANES.)
AN/ARC 34 T-R ØA, ØB, & ØC F	THE TRANSFORMER-RECTIFIER UNIT WILL NOT RECEIVE AC POWER. (THESE CIRCUIT BREAKERS ARE LOCATED IN THE NOSE WHEEL WELL ON TF AIRPLANES.)
CANOPY UNLOCK	THE CANOPY UNLOCK LIGHT IS INOPERATIVE.
PNEU PRESS	THE PNEUMATIC PRESSURE LOW WARNING LIGHT IS INOPERATIVE.
ALT CMPTR TF	THE SERVO MODE OF THE AAU-19/A ALTIMETER IS INOPERATIVE AND THE ALTIMETER IS OPERATING IN STANDBY MODE.
TEST SET (AIMS) TF	IN ADDITION TO THE AN/APX-72 TRANSPONDER TEST SET BEING INOPERATIVE, THE PILOT'S AAU-19/A ALTIMETER VIBRATOR VOLTAGE WILL NOT BE AVAILABLE.
RT (AIMS) TF	IN ADDITION TO LOSS OF 28-VOLT DC TO THE AN/APX-72 RT, THE INSTRUCTOR PILOT'S AAU-19/A ALTIMETER VIBRATOR VOLTAGE WILL NOT BE AVAILABLE.

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Figure 1-20

circuit breaker functions

RIGHT HAND AFT PANEL (CONTINUED)

CIRCUIT BREAKER	EFFECT ON CIRCUIT WITH CIRCUIT BREAKER OUT
HYD PRESS	THE HYDRAULIC FAIL WARNING LIGHT IS INOPERATIVE. THE HYDRAULIC PRESSURE GAGE FUNCTIONS NORMALLY.
CONSOLE (F)	THE CONSOLE LIGHTS ARE INOPERATIVE.
LH & CTR CONSOLE (TF)	THE LEFT HAND AND CENTER CONSOLE LIGHTS ARE INOPERATIVE.
RH CONSOLE (TF)	THE RIGHT HAND CONSOLE LIGHTS ARE INOPERATIVE.
INST PNL (F)	THE INSTRUMENT PANEL LIGHTS ARE INOPERATIVE.
LH INST (TF)	THE LEFT HAND INSTRUMENT PANEL LIGHTS ARE INOPERATIVE.
RH INST (TF)	THE RIGHT HAND INSTRUMENT PANEL LIGHTS ARE INOPERATIVE.
FLT INS (F) FLIGHT INST (TF)	THE INSTRUMENT LIGHTS ARE INOPERATIVE. THE WARNING LIGHTS LOSE THEIR DIMMING FEATURE.
STORM (F) THUNDERSTORM (TF)	THE STORM LIGHTS ARE INOPERATIVE. THE WARNING LIGHTS WILL STILL BRIGHTEN IF THE SWITCH IS ACTIVATED WITH FLIGHT INSTRUMENT LIGHTS ON.
FLOOD (F)	THE FLOODLIGHTS ARE INOPERATIVE.
LH CKPT FLOOD (TF)	THE LEFT HAND COCKPIT FLOODLIGHTS ARE INOPERATIVE.
RH CKPT FLOOD (TF)	THE RIGHT HAND COCKPIT FLOODLIGHTS ARE INOPERATIVE.
POSITION PWR (F) POS LT PWR (TF)	THE WING TIP, FUSELAGE, AND TAIL LIGHTS ARE INOPERATIVE.
POSITION CONT (F) POS LT CONT (TF)	THE BEACON LIGHTS ARE INOPERATIVE.
TURN & BANK	THE TURN NEEDLE IS INOPERATIVE.
UHF COM DC (TF)	THE UHF IS INOPERATIVE.

RIGHT HAND FORWARD PANEL

CIRCUIT BREAKER	EFFECT ON CIRCUIT WITH CIRCUIT BREAKER OUT
UHF COMM DC (F)	THE UHF IS INOPERATIVE.
ALT CMPTR (F)	THE SERVO MODE OF THE AAU-19/A ALTIMETER IS INOPERATIVE AND THE ALTIMETER IS OPERATING IN STANDBY MODE.
TEST SET (AIMS) (F)	IN ADDITION TO THE AN/APX-72 TRANSPONDER TEST SET BEING INOPERATIVE, THE AAU-19/A ALTIMETER VIBRATOR VOLTAGE WILL NOT BE AVAILABLE.

LEFT HAND FORWARD PANEL

CIRCUIT BREAKER	EFFECT ON CIRCUIT WITH CIRCUIT BREAKER OUT
AC PWR CONT (F) AC POWER CONT (TF)	THE AC GENERATOR IS INOPERATIVE AND THE AC POWER FAILURE WARNING LIGHT IS ILLUMINATED. (THE CIRCUIT BREAKER IS POWERED FROM THE EMERGENCY DC BUS.)
DC PWR CONT (F) DC POWER CONT (TF)	THE RESET POSITION OF THE AC BUS SWITCH AND TR FAIL POSITION OF THE BATTERY SWITCH ARE INOPERATIVE.
PWR WARN (F) POWER WARN (TF)	THE AC POWER FAILURE WARNING LIGHT IS INOPERATIVE.
EMER AC CONT PWR (F) AC EMERGENCY POWER (TF)	THE ON POSITION OF THE AC BUS SWITCH IS INOPERATIVE. (THE CIRCUIT IS POWERED FROM THE EMERGENCY DC BUS.)
FUEL SELECTOR RH & LH	THE FUEL BOOST PUMPS ARE INOPERATIVE AND THE FUEL SHUTOFF VALVES CANNOT BE REPOSITIONED.
LDG GEAR CONT (F) LG CONT (TF)	POWER IS REMOVED FROM LANDING GEAR HANDLE SWITCH.
AB CONT	THE AFTERBURNER IS INOPERATIVE.
START (F) ENG START (TF)	GROUND START CANNOT BE OBTAINED. AIRSTART CANNOT BE OBTAINED USING GROUND START PROCEDURES. INFLIGHT IGNITION IS NOT AFFECTED.

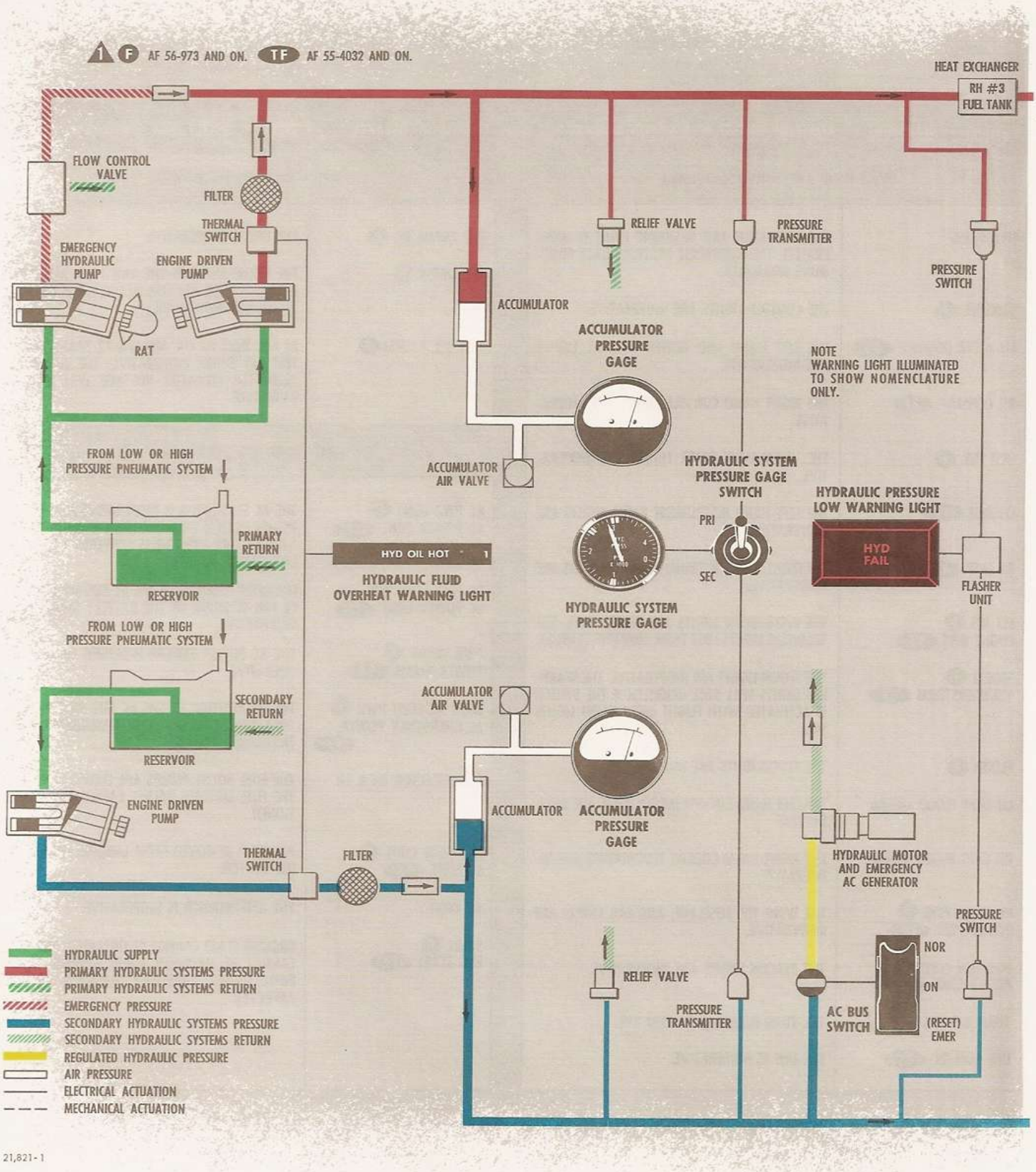
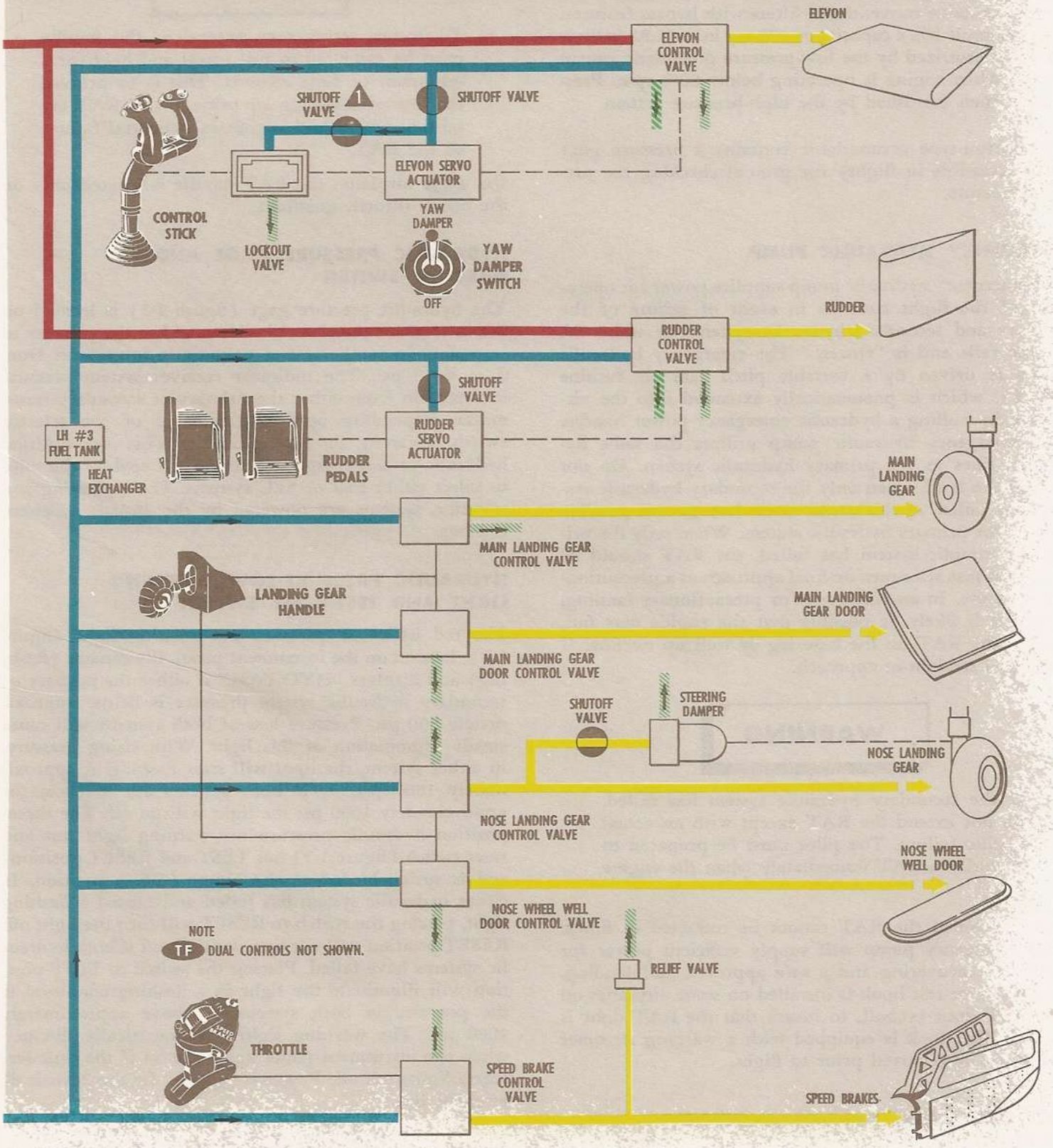

1 F AF 56-973 AND ON. TF AF 55-4032 AND ON.
HEAT EXCHANGER
RH #3 FUEL TANK
FLOW CONTROL VALVE
FILTER
THERMAL SWITCH
EMERGENCY HYDRAULIC PUMP
ENGINE DRIVEN PUMP
RAT
RELIEF VALVE
PRESSURE TRANSMITTER
PRESSURE SWITCH
ACCUMULATOR
ACCUMULATOR PRESSURE GAGE
NOTE
WARNING LIGHT ILLUMINATED TO SHOW NOMENCLATURE ONLY.
FROM LOW OR HIGH PRESSURE PNEUMATIC SYSTEM
ACCUMULATOR AIR VALVE
HYDRAULIC SYSTEM PRESSURE GAGE SWITCH
HYDRAULIC PRESSURE LOW WARNING LIGHT
PRIMARY RETURN
HYD OIL HOT 1
HYDRAULIC FLUID OVERHEAT WARNING LIGHT
PRI
SEC
HYD FAIL
FLASHER UNIT
RESERVOIR
HYDRAULIC SYSTEM PRESSURE GAGE
FROM LOW OR HIGH PRESSURE PNEUMATIC SYSTEM
SECONDARY RETURN
ACCUMULATOR AIR VALVE
RESERVOIR
ENGINE DRIVEN PUMP
ACCUMULATOR
ACCUMULATOR PRESSURE GAGE
HYDRAULIC MOTOR AND EMERGENCY AC GENERATOR
THERMAL SWITCH
FILTER
PRESSURE SWITCH
NOR
ON
(RESET) 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
21,821-1

Figure 1-21

hydraulic power supply system



21,821-2

brakes, damper system, 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. 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 except when engine is operating below 80% rpm. Pressure is then furnished by the high-pressure system.

The piston-type accumulator contains a pressure gage (not accessible in flight) for ground checking the preload pressure.

EMERGENCY HYDRAULIC PUMP

The emergency hydraulic pump 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. The emergency hydraulic pump utilizes the same hydraulic lines as the primary hydraulic system. Do not 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. In an emergency or precautionary landing, where it is likely or possible that the engine may fail, extend the RAT on the base leg or well up on final if making a straight-in approach.

WARNING

If the secondary hydraulic system has failed, do not extend the RAT except with an actual engine failure. The pilot must be prepared to extend the RAT immediately when the engine fails.

Once extended the RAT cannot be retracted in flight. This emergency pump 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 1-4), located in the outboard side of the throttle quadrant, is used to energize the emergency hydraulic pump. Pulling the handle (approximately two inches) mechanically selects the pneumatic pressure to extend the RAT into the airstream.

CAUTION

To insure satisfactory extension, the handle must be extended to full travel and held for a minimum of four seconds. This is to prevent the loss of snubbing air prior to full RAT extension which could result in structural failure of the RAT.

On TF airplanes the RAT handle is located only on the pilot's throttle quadrant.

HYDRAULIC PRESSURE GAGE AND SELECTOR SWITCH

The hydraulic pressure gage (figure 1-7) is located on the instrument panel 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 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 (figure 1-7), 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 1-7) 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 (figure 1-36) located on the warning ligh. 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 differentiare 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; to supply warm air for anti-icing, rain-clearing and defogging; and to operate the afterburner exhaust nozzles.

HIGH-PRESSURE PNEUMATIC SYSTEM

The high-pressure pneumatic system (figure 1-22) stores air under pressure in four spherical flasks and in the main landing gear drag braces. The 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 aft side of the left main wheel well. System pressure is indicated on a pressure gage, located on the forward side of the left main wheel well above the external power receptacle. 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. A priority valve is installed between the priority and nonpriority air flask and 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, normal and emergency drag chute deployment, barrier probe extension, and landing gear extension. For servicing requirements see figure 1-44.

PNEUMATICALLY OPERATED EQUIPMENT

See figures 1-22 and 4-1 for complete reference to pneumatically operated equipment.

PNEUMATIC SHUTOFF VALVE (RIGHT MAIN WHEEL WELL)

A manually controlled pneumatic shutoff valve is installed in the right wheel well to shut off high pressure air to the canopy counterbalance (F), armament bay doors, armament launchers, rudder feel system, and to the hydraulic reservoir pressurization system. The valve is used as a means of shutting off the source of high pressure pneumatic system air when maintenance is being performed on the systems listed. In the OPEN position, this valve has no effect on the operation of the high-pressure pneumatic system.

PNEUMATIC PRESSURE-LOW WARNING LIGHT

The pneumatic pressure-low warning light (figure 1-36), located on the warning light panel, illuminates to display "PNEU PRESS." The warning light indicates that the system contains 1400 psi pressure at the time of light illumination. The status of the system will be in one of the three following conditions:

- Pressure of 1400 psi throughout system (with normal system usage).
- Pressure of 1400 psi in priority side (with leakage in non-priority side).
- No air pressure in system (with leakage in priority side of system).

Note

Illumination of PNEU PRESS warning light in most cases indicates that the system contains sufficient air pressure for deployment of priority actuated components.

FLIGHT CONTROL SYSTEM

The flight control system (figures 1-23 and 1-24) 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 bell-crank 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

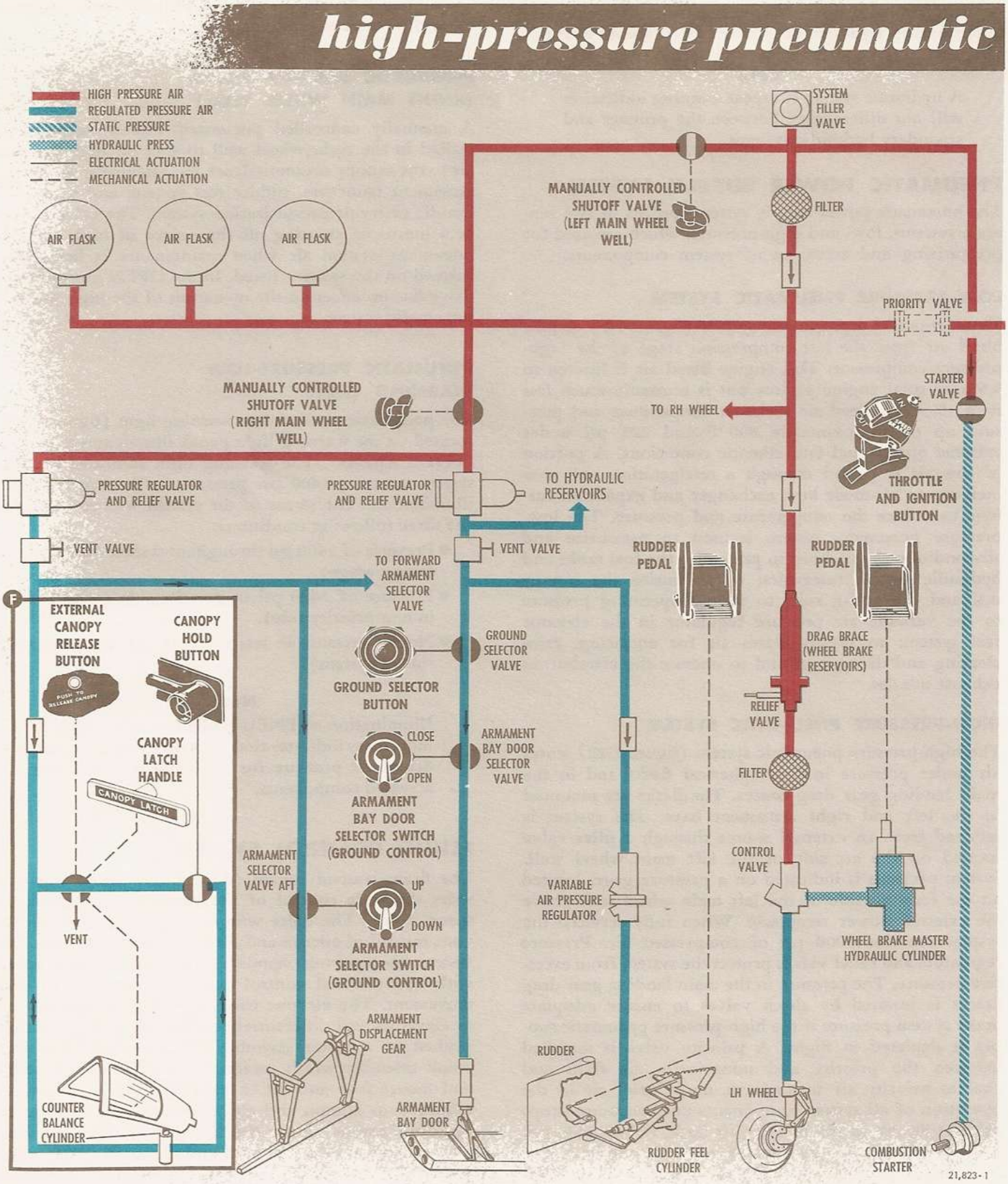
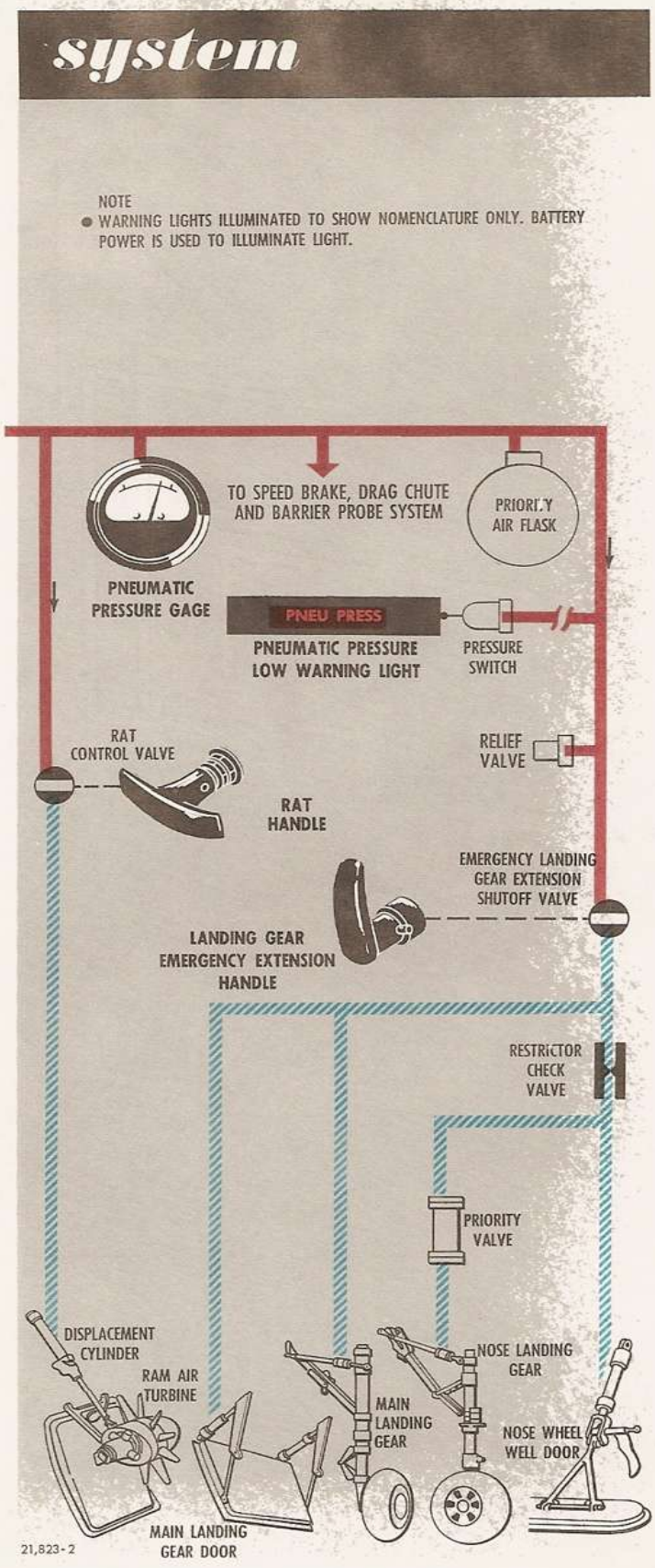

high-pressure pneumatic
HIGH PRESSURE AIR
REGULATED PRESSURE AIR
STATIC PRESSURE
HYDRAULIC PRESS
ELECTRICAL ACTUATION
MECHANICAL ACTUATION
AIR FLASK
AIR FLASK
AIR FLASK
SYSTEM FILLER VALVE
MANUALLY CONTROLLED SHUTOFF VALVE (LEFT MAIN WHEEL WELL)
FILTER
PRIORITY VALVE
STARTER VALVE
MANUALLY CONTROLLED SHUTOFF VALVE (RIGHT MAIN WHEEL WELL)
TO RH WHEEL
THROTTLE AND IGNITION BUTTON
PRESSURE REGULATOR AND RELIEF VALVE
PRESSURE REGULATOR AND RELIEF VALVE
TO HYDRAULIC RESERVOIRS
VENT VALVE
VENT VALVE
RUDDER PEDAL
RUDDER PEDAL
TO FORWARD ARMAMENT SELECTOR VALVE
EXTERNAL CANOPY RELEASE BUTTON
PUSH TO RELEASE CANOPY
CANOPY HOLD BUTTON
GROUND SELECTOR VALVE
GROUND SELECTOR BUTTON
DRAG BRACE (WHEEL BRAKE RESERVOIRS)
RELIEF VALVE
CANOPY LATCH HANDLE
CANOPY LATCH
CLOSE
OPEN
ARMAMENT BAY DOOR SELECTOR VALVE
ARMAMENT BAY DOOR SELECTOR SWITCH (GROUND CONTROL)
FILTER
CONTROL VALVE
ARMAMENT SELECTOR VALVE AFT
UP
DOWN
VARIABLE AIR PRESSURE REGULATOR
WHEEL BRAKE MASTER HYDRAULIC CYLINDER
VENT
ARMAMENT SELECTOR SWITCH (GROUND CONTROL)
ARMAMENT DISPLACEMENT GEAR
RUDDER
LH WHEEL
COUNTER BALANCE CYLINDER
ARMAMENT BAY DOOR
RUDDER FEEL CYLINDER
COMBUSTION STARTER
21,823-1

Figure 1-22



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. 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 the primary system 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° (F); 4½° (TF)), 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 a feel force regulator 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 feel force regulator is operated by low-pressure pneumatic system pressure and controls the ram air pressure from the large and small impact tubes 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

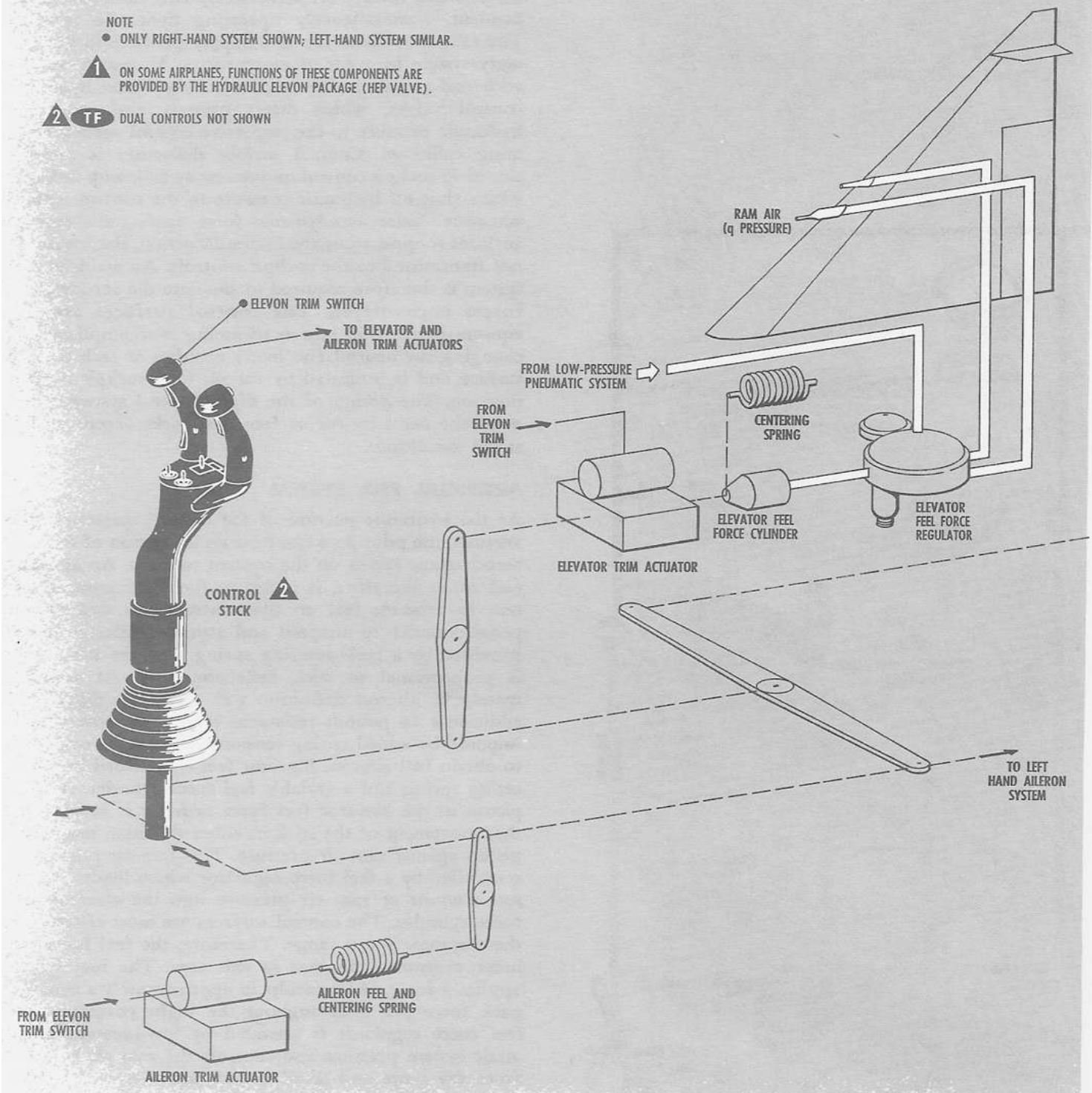
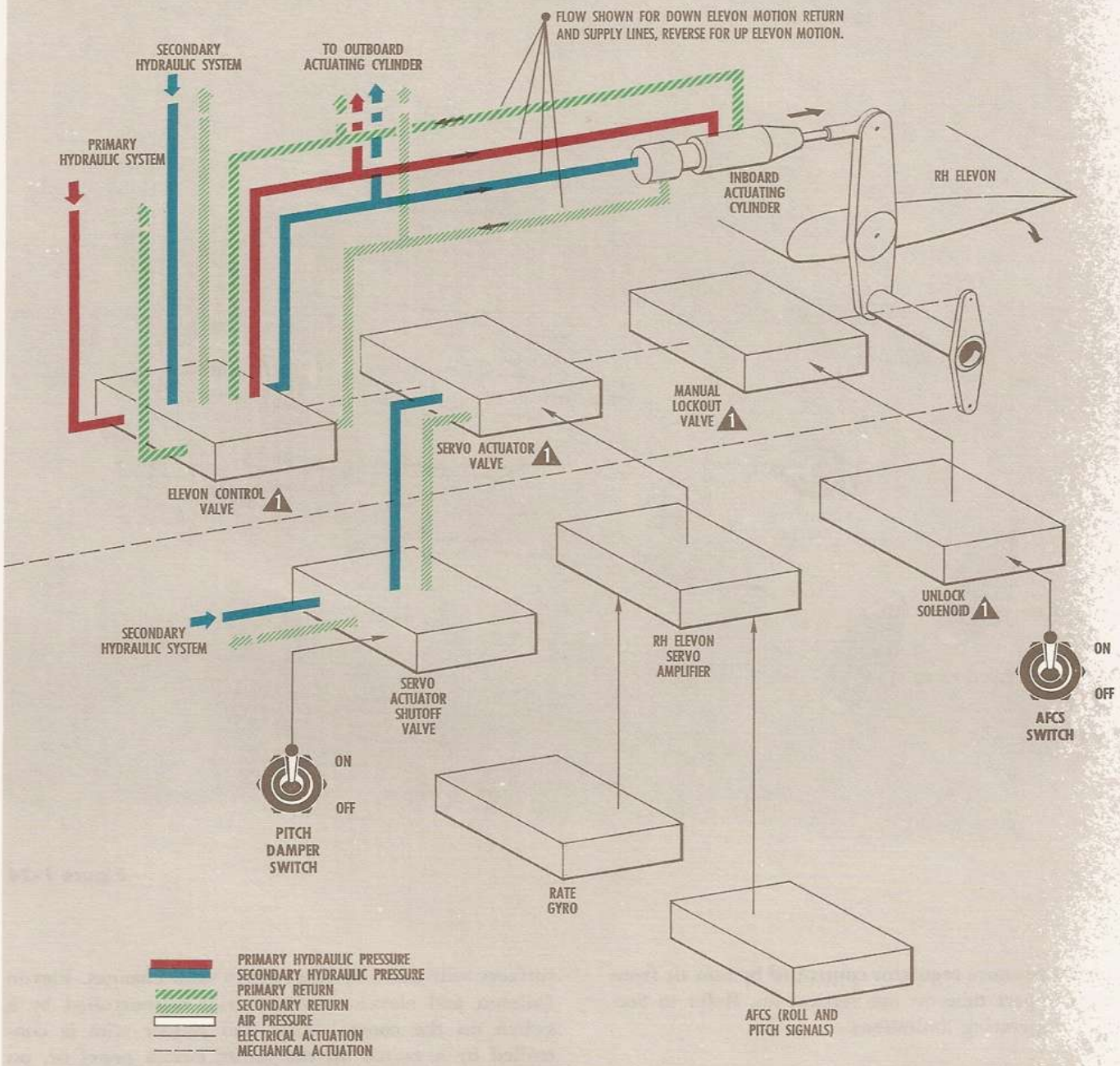

NOTE
ONLY RIGHT-HAND SYSTEM SHOWN; LEFT-HAND SYSTEM SIMILAR.
1 ON SOME AIRPLANES, FUNCTIONS OF THESE COMPONENTS ARE PROVIDED BY THE HYDRAULIC ELEVON PACKAGE (HEP VALVE).
2 TF DUAL CONTROLS NOT SHOWN
RAM AIR (q PRESSURE)
ELEVON TRIM SWITCH
TO ELEVATOR AND AILERON TRIM ACTUATORS
FROM LOW-PRESSURE PNEUMATIC SYSTEM
FROM ELEVON TRIM SWITCH
CENTERING SPRING
ELEVATOR FEEL FORCE CYLINDER
ELEVATOR FEEL FORCE REGULATOR
ELEVATOR TRIM ACTUATOR
CONTROL STICK 2
TO LEFT HAND AILERON SYSTEM
AILERON FEEL AND CENTERING SPRING
FROM ELEVON TRIM SWITCH
AILERON TRIM ACTUATOR
21824-1

Figure 1-23

elevon control system



21824-2

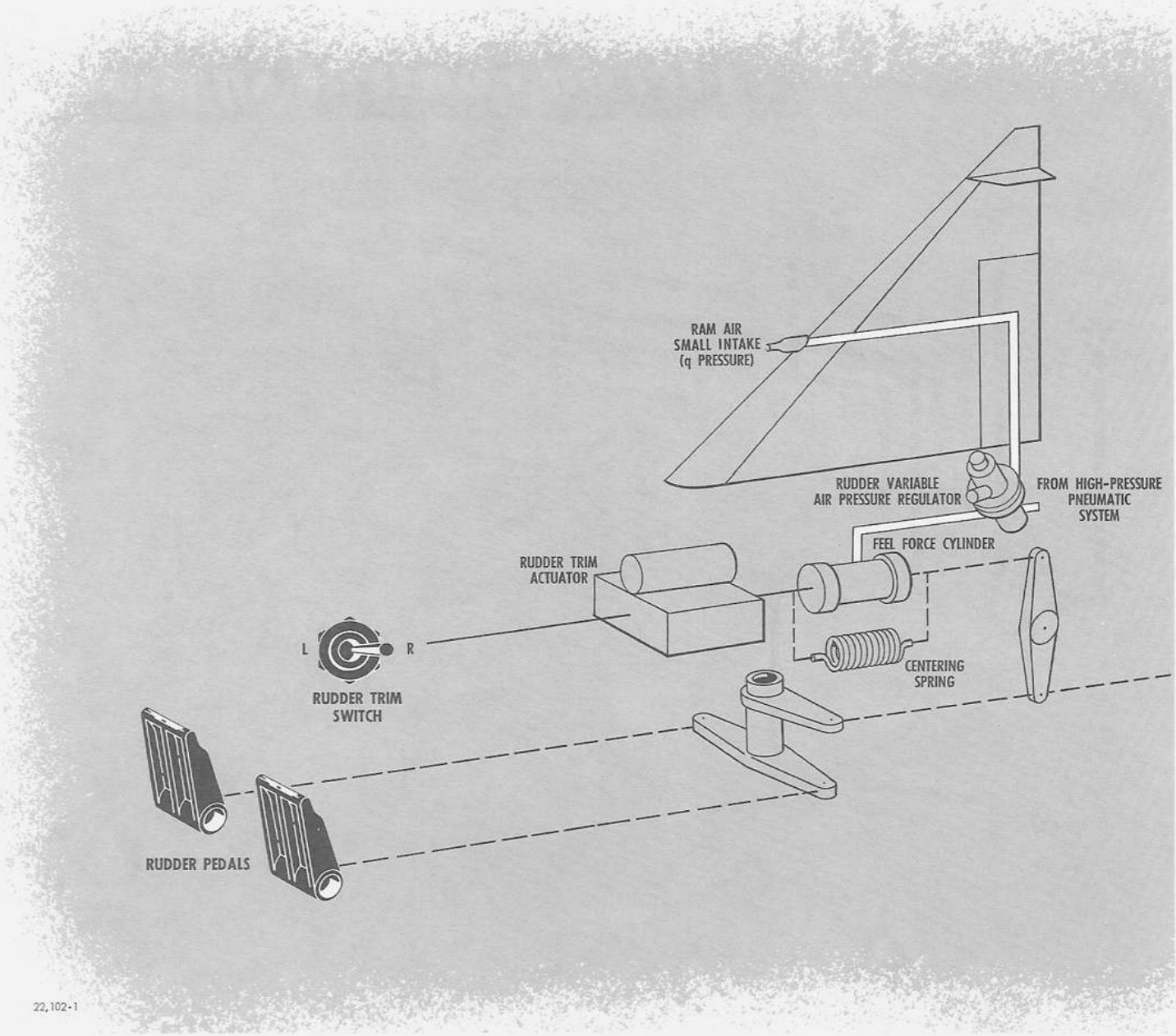


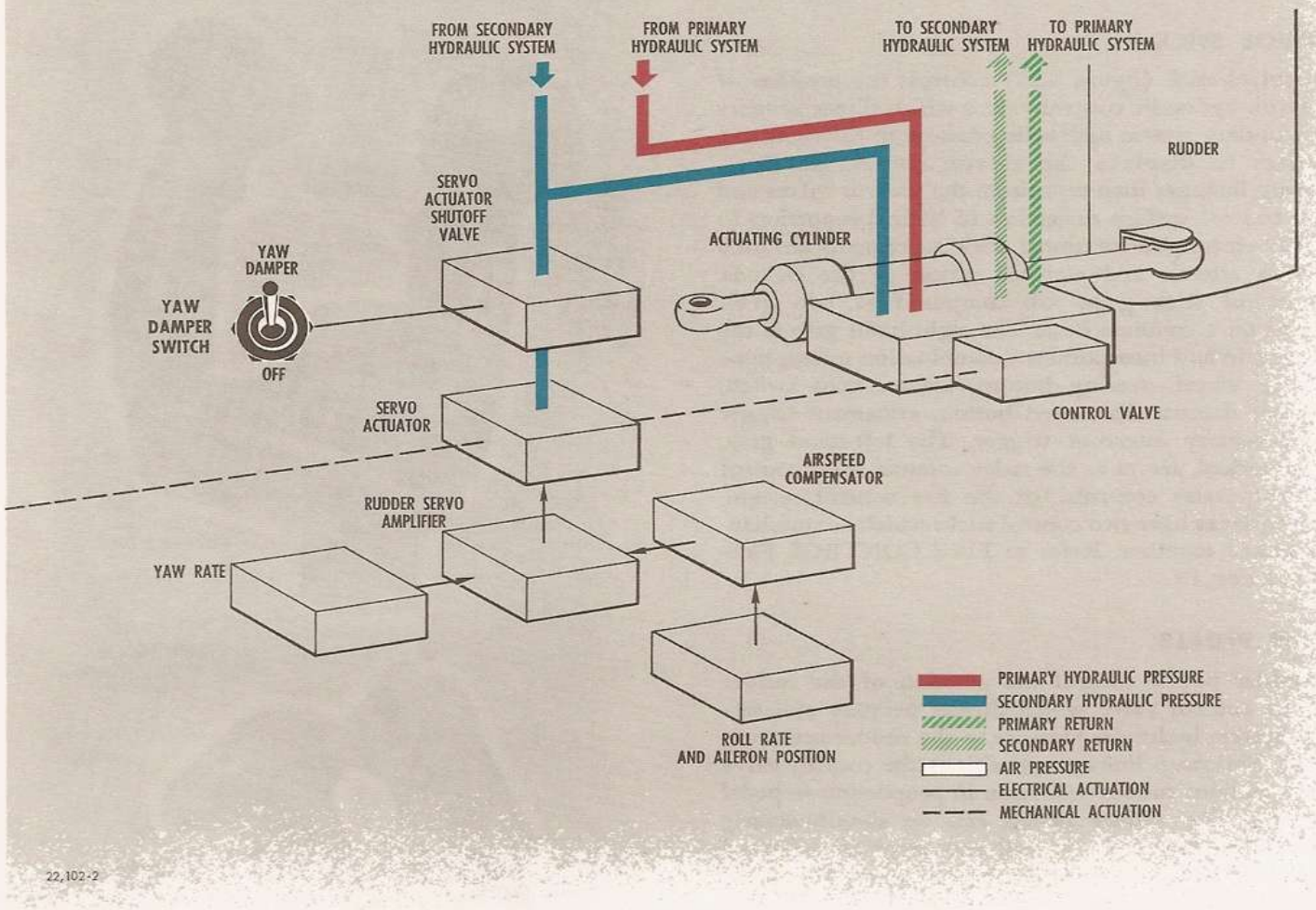
Figure 1-24

variable air pressure regulator controlled by ram air from the small 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 bus.

rudder control system



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 (F) and (TF) airplanes or, on other 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 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.

CONTROL STICK

The control stick (figure 1-25) controls the position of the elevon hydraulic control valves which direct primary 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. TF airplanes have two control sticks which are mechanically linked together. 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. TF airplanes have dual rudder pedals which are mechanically linked together. Refer to WHEEL BRAKE SYSTEM and NOSE WHEEL STEERING SYSTEM, this Section.

ELEVON TRIM SWITCH

Lateral and longitudinal trim is controlled by a five-position elevon trim switch (figure 1-25), 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. TF airplanes have an elevon

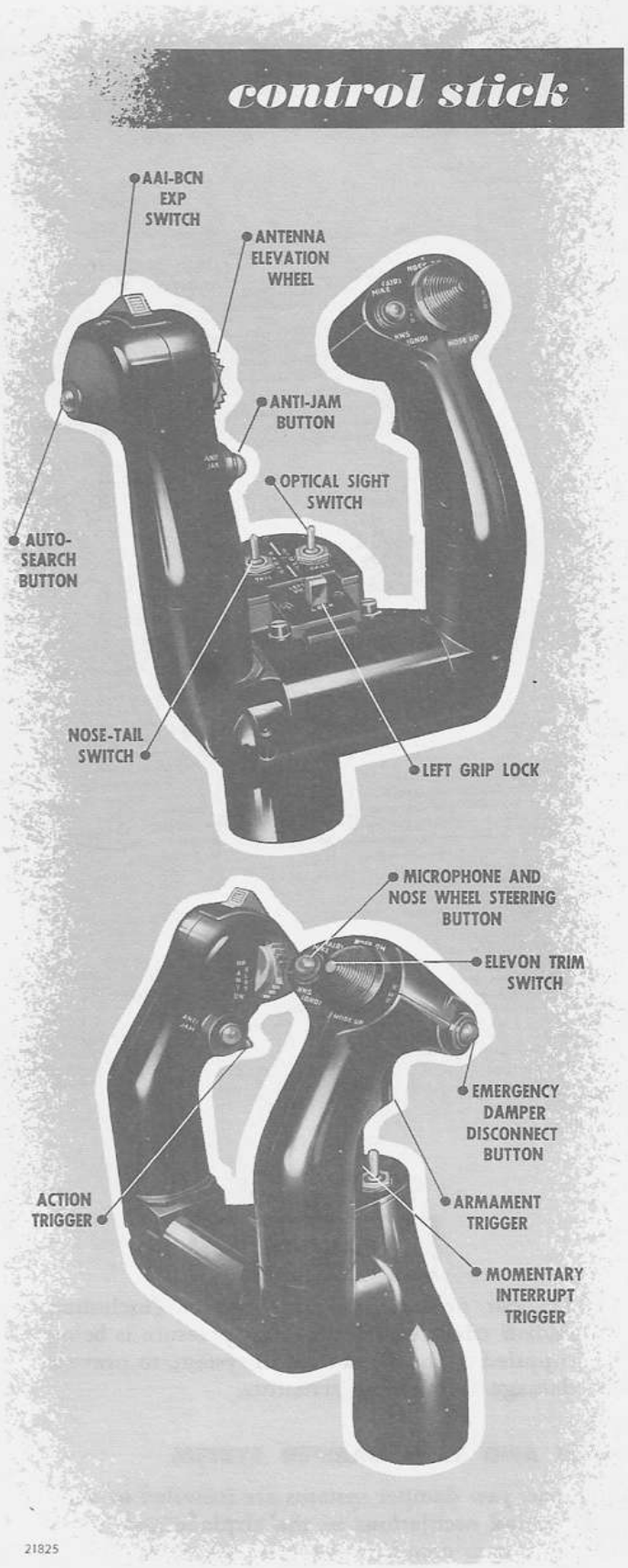


Figure 1-25

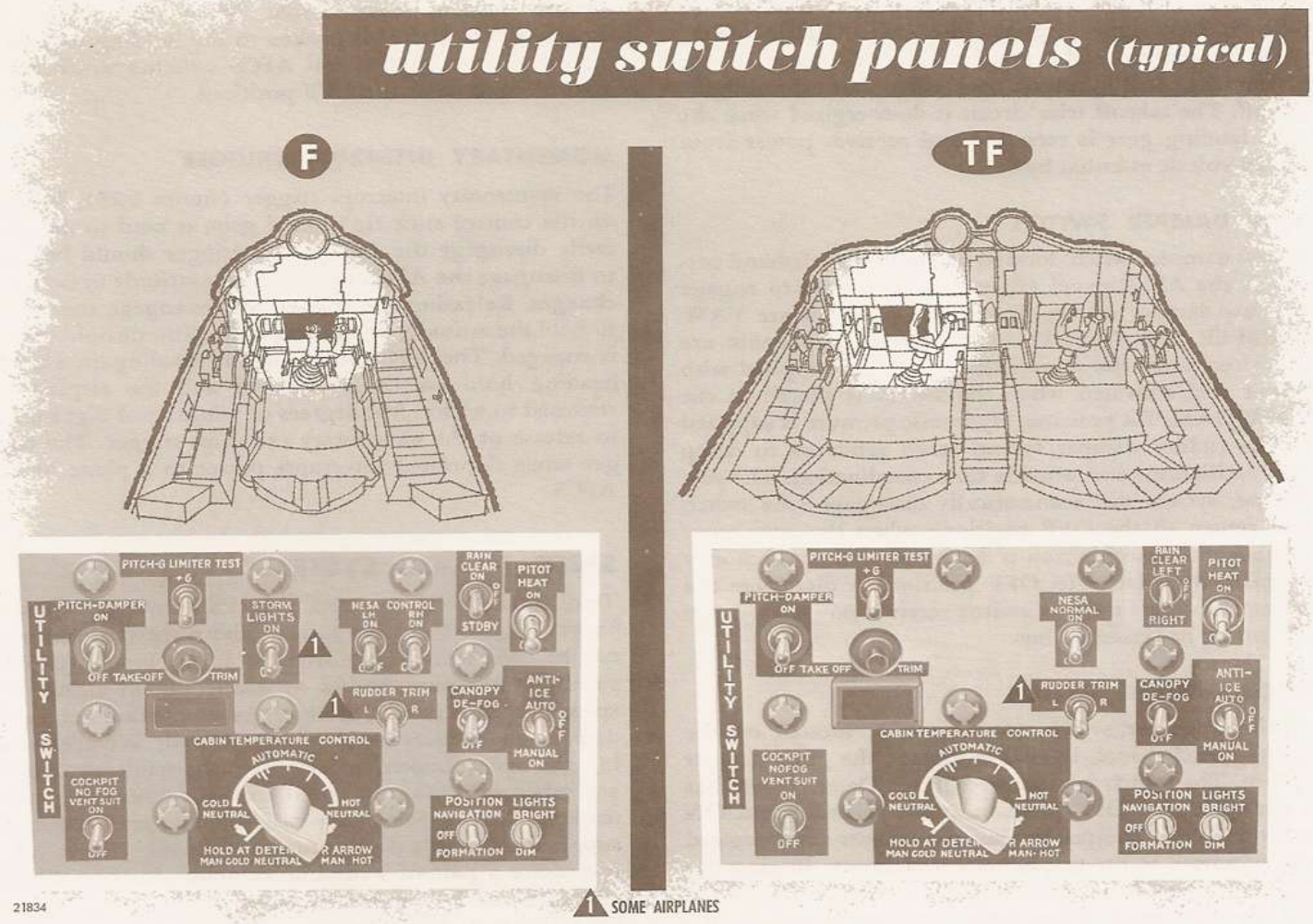


Figure 1-26

trim switch on each right-hand control stick grip. 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 AFTO 781 with a red cross.

RUDDER TRIM SWITCH

The three-position rudder trim switch (figure 1-4 or figure 1-26), 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. On TF airplanes, the rudder trim switch is located on the utility switch panel on some airplanes and on the lower right-hand side of the pilot's throttle quadrant on other airplanes. The trim switch receives power from the 28-volt dc essential bus.

TAKEOFF TRIM BUTTON AND INDICATOR LIGHT

The takeoff trim button (figure 1-26), 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 de-energized when the nose landing gear is retracted and receives power from the 28-volt dc essential bus.

YAW DAMPER SWITCH

A yaw damper switch, located in the lower left-hand corner of the AFCS panel (figure 4-19), is used to engage the yaw damper system. 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 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 the OFF position) when the emergency damper disconnect button is depressed or the switch may be manually placed to 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 1-26), located on the utility switch panel, is used to engage the pitch damper system. The switch is placarded "Pitch Damper" and has 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 YAW DAMPER position prior to placing the pitch damper switch to ON. When the yaw damper switch is in 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 1-25), 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 damper system and AFCS switches which automatically return to the OFF positions.

MOMENTARY INTERRUPT TRIGGER

The momentary interrupt trigger (figure 1-25), 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 altitude, if the altitude hold switch is engaged. The AFCS will return to heading hold if the heading hold switch is actuated and the airplane is returned to within five degrees of wings level flight prior to release of the momentary interrupt trigger. The trigger when depressed interrupts power to a phase of the AFCS.

SPEED BRAKES SYSTEM

Two hydraulically operated, electrically controlled speed brakes (figure 1-27) 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 nose-up pitch tendency occurs, and when retracting the speed brakes, a nose-down pitch tendency 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 two 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

The speed brake emergency opening system 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 165 KCAS.

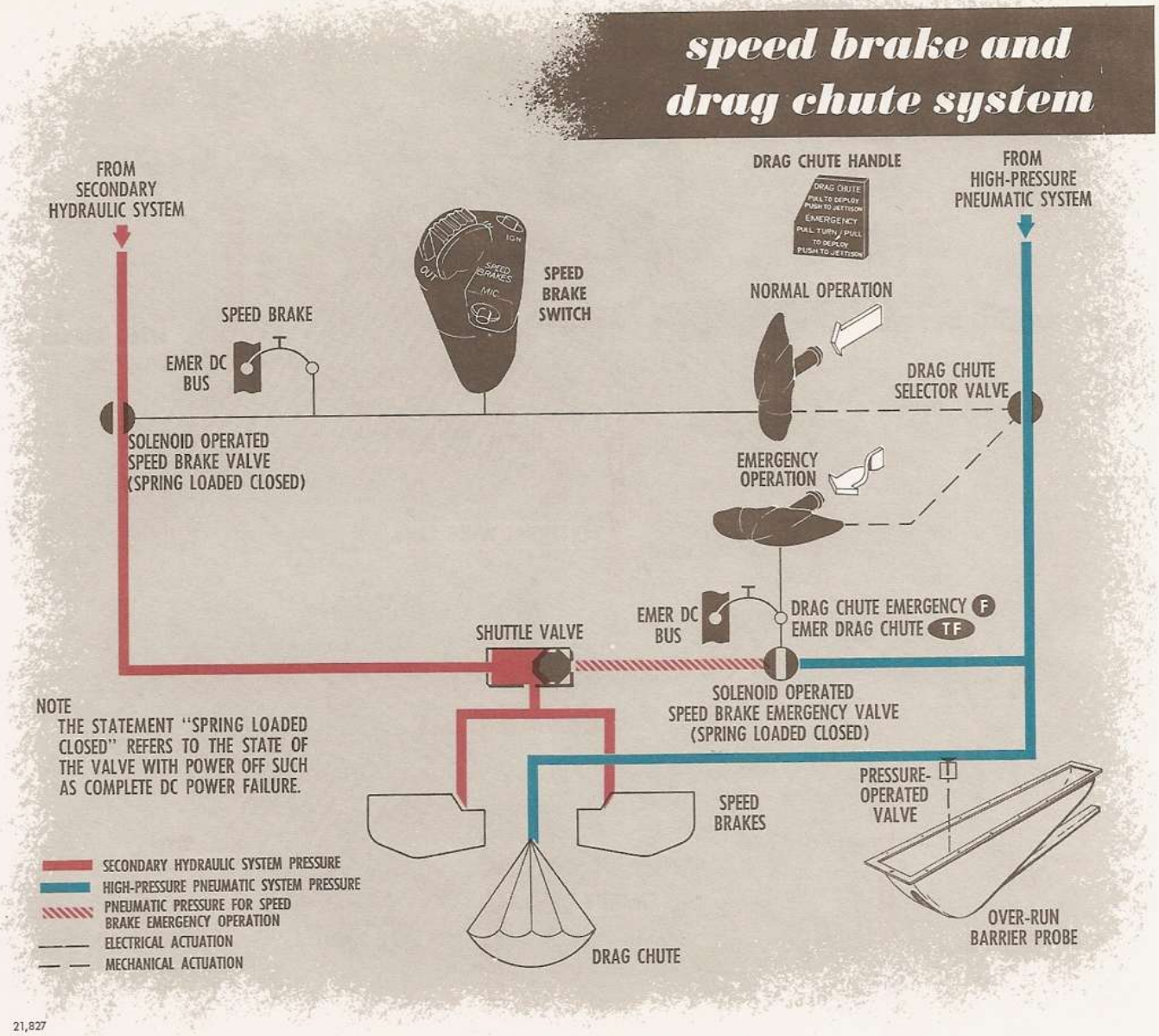


Figure 1-27

Note

The speed brakes cannot be opened without emergency dc bus power, either by the normal or emergency systems. Normal closing requires both secondary hydraulic power and emergency dc bus power.

SPEED BRAKES GROUND SAFETY LOCKS

Ground maintenance safety locks (figure 1-28) may be installed on the speed brakes actuating cylinders when the brakes are extended and must be removed before flight.

SPEED BRAKES SWITCH

The three-position speed brakes switch (figure 1-4), 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.

WARNING

Return the speed brakes switch to neutral after opening or closing of the speed brakes. This is done to relieve hydraulic pressure on the speed brake cylinders and to prevent secondary hydraulic system failure in event a leak develops in the speed brake area.

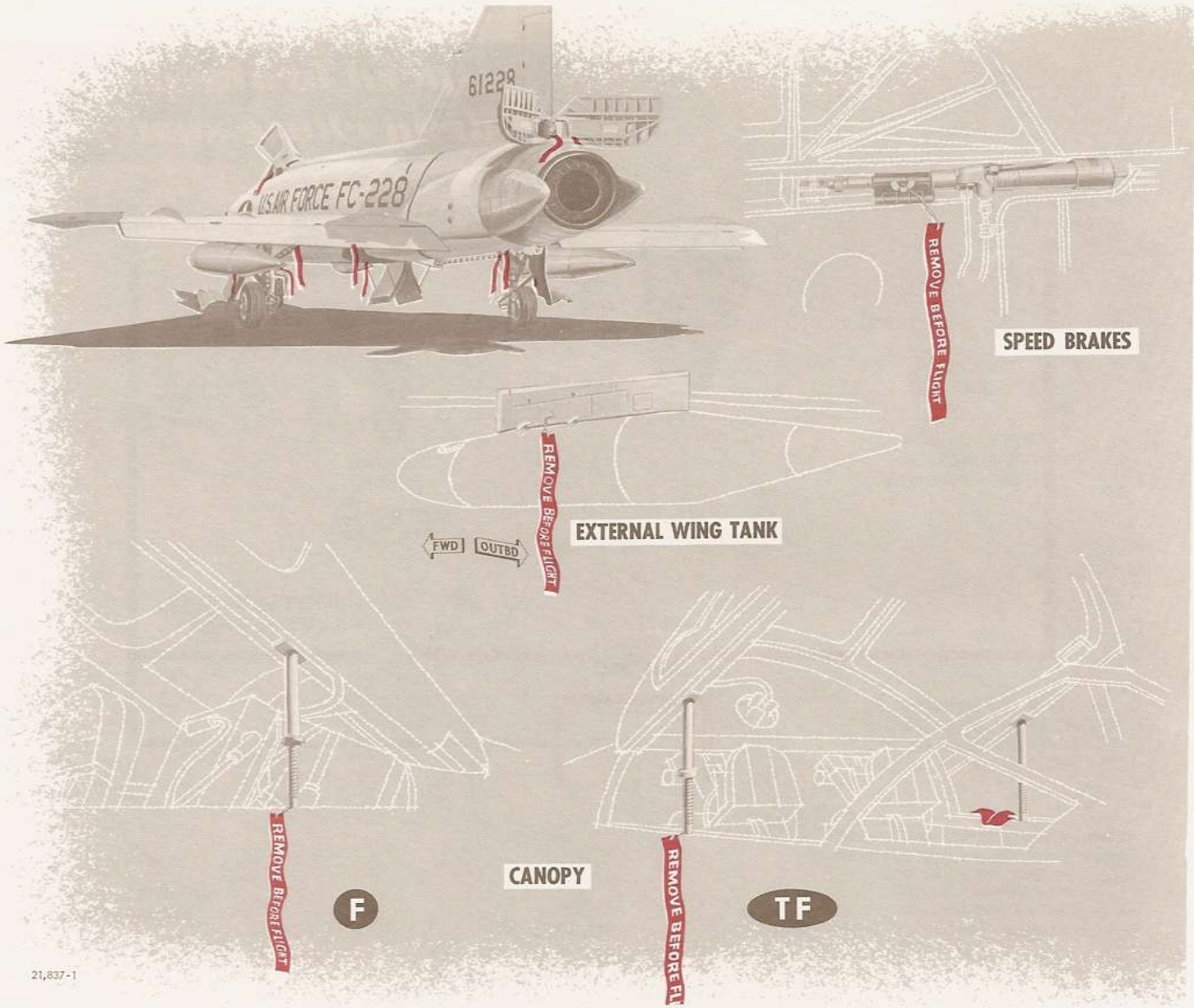


Figure 1-28

The NEUTRAL position is indicated by a white alignment mark on the switch guide. When the switch is in the NEUTRAL position, the control valve is closed and the speed brakes are held in the selected position. On TF airplanes, a speed brakes switch is located on top of each throttle. The speed brakes switch on the instructor's throttle has three positions: IN, OUT, and a center NEUTRAL position, and is spring-loaded to the NEUTRAL position. To extend or retract the speed brakes, the switch must be held in the IN or OUT position until the speed brakes are extended or retracted.

Note TF

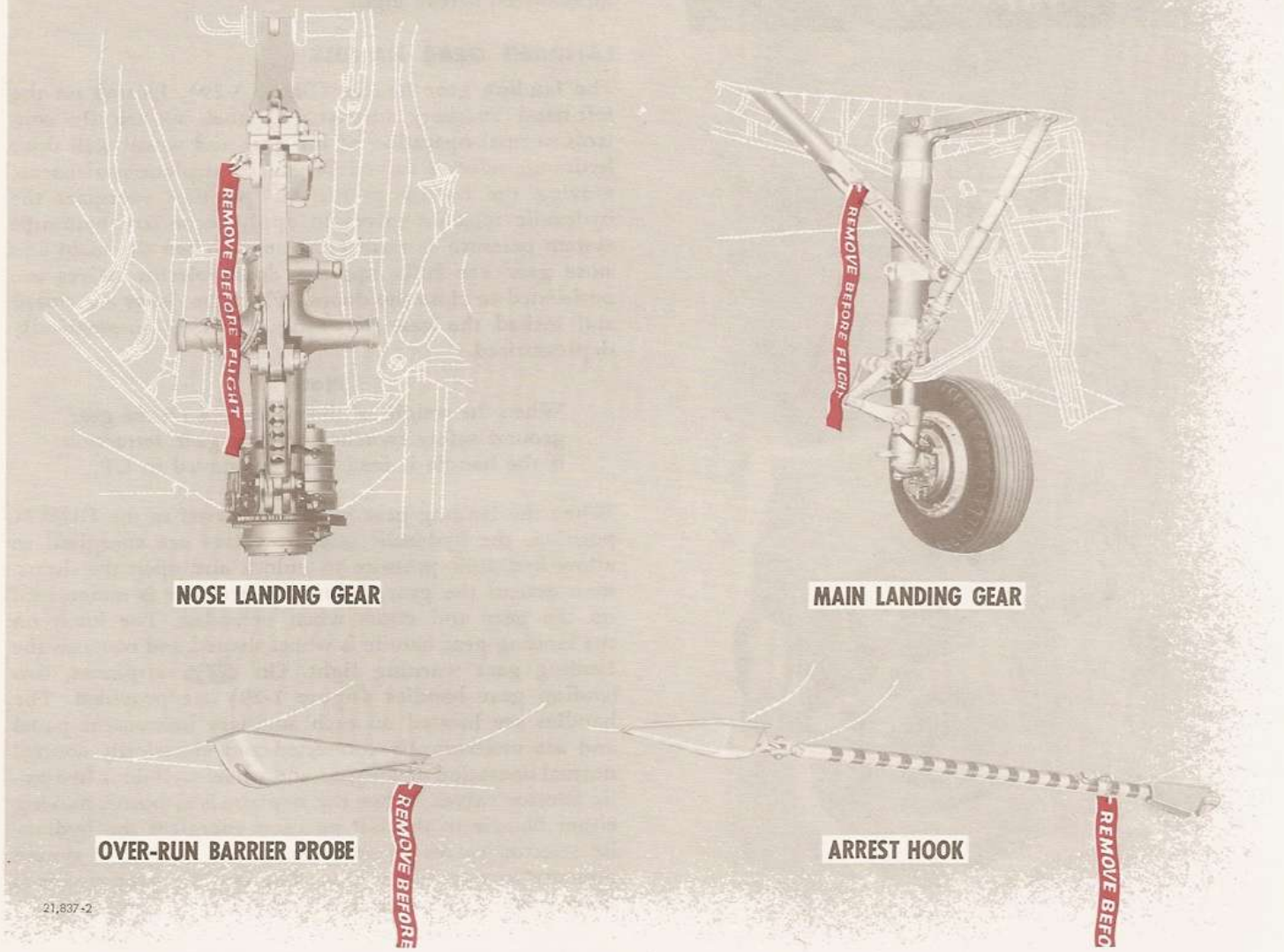
In order for the instructor's speed brakes switch to function normally, the pilot's switch must be in the NEUTRAL position. If the pilot's switch is in other than the NEUTRAL position when the instructor's speed brakes switch is released, the speed brakes will return to the position selected by the pilot's switch.

The speed brakes switch receives power from the emergency dc bus.

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

ground safety locks



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 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. The main landing gear, 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.

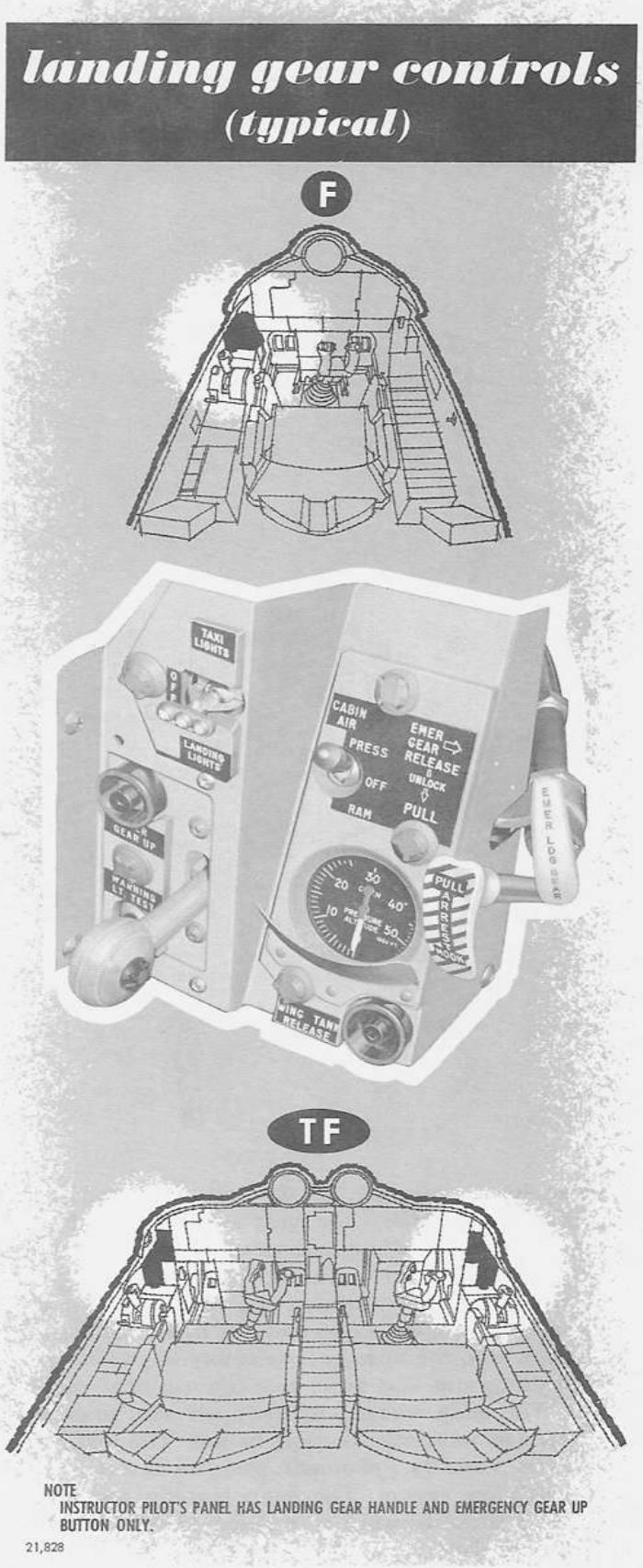


Figure 1-29

LANDING GEAR GROUND SAFETY LOCKS

Removable ground safety locks (figure 1-28) 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 1-29), 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 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. On TF airplanes, two landing gear handles (figure 1-29) are provided. The handles are located on each auxiliary instrument panel and are mechanically connected and electrically control normal operation of the gear and wheel well door hydraulic selector valves. When the airplane is airborne, moving either handle to the UP position energizes the hydraulic selector valves to apply secondary hydraulic system pressure to retract the gear. The landing gear circuit is powered by the 28-volt dc essential bus.

LANDING GEAR EMERGENCY-UP BUTTONS

A landing gear emergency-up button is located on the left-hand auxiliary instrument panel F and on each auxiliary instrument panel TF (figure 1-29). The button is placarded "Emer Gear Up" and can be used to retract the landing gear when the airplane is in the air or on the ground. On the ground, holding the button depressed bypasses the ground safety switches and the gear will retract if the normal landing gear handle is in the UP position. When airborne, depressing the button momentarily will allow the landing gear control circuit to bypass the ground safety switches in the event the switches malfunction, and retract the gear with the normal control handle in the UP position. To insure positive emergency retraction after takeoff, the gear emergency-up button should be held depressed until the gear completely retracts. After emergency gear extension has been accomplished, the gear emergency-up button will be inoperative

until the landing gear emergency extension handle is pushed in. The landing gear audible warning signal, the red "gear unsafe" warning bar light, and the red light in the gear handle operating in conjunction with illumination of the three green indicator lights indicate a ground safety switch malfunction. Only under these conditions should the emergency-up button be used while 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 1-29), 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.

Note

The handle must remain fully extended to assure that the landing gear remains down and locked.

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.

LANDING GEAR POSITION INDICATORS

Three green landing gear position indicator lights (figures 1-8 and 1-9), located on the left side of the instrument panel, show the down and locked position of the main and nose landing gear. These lights illuminate only when the corresponding gear is down and locked. Power to the landing gear position indicator lights is supplied by the 28-volt dc essential bus.

LANDING GEAR WARNING LIGHT

A red "gear unsafe" warning bar light, located on the left side of the instrument panel, and a red warning light located within the wheel-shaped knob on the landing gear control handle (figure 1-29), simultaneously illuminate at any time landing gear is not in the position selected by the landing gear control handle. Both lights will also illuminate when the landing gear is not down and locked at an altitude of 13,500 (±1,000) feet climbing or 9,500 (±1,000) feet descending if the throttle is retarded below 80% (+2, −0) rpm and airspeed is less than 295 KCAS maximum on acceleration or less than 210 (±10) KCAS (KIAS) on deceleration. Both "gear unsafe" warning lights may be dimmed by pressing the bar light. The audible warning signal operates in conjunction with the warning lights to warn of gear unsafe conditions. Refer to LANDING GEAR AUDIBLE WARNING SIGNAL AND CUTOUT BUTTON, this Section. The lights receive power from the 28-volt dc essential bus. The lights and horn are tested with the master warning system.

LANDING GEAR AUDIBLE WARNING SIGNAL AND CUTOUT BUTTON

An audible signal is provided to sound in the pilot's radio headset to warn of gear unsafe conditions. The audible signal operates in conjunction with the landing gear warning lights and will be heard in the pilot's radio headset at any time the landing gear warning lights illuminate. Refer to LANDING GEAR WARNING LIGHT, this Section. The audible signal will also be heard when the master warning test and reset switch is moved to the TEST position. A landing gear audible warning signal cutout button (figure 1-4) is installed on the aft portion of the throttle quadrant, to cut off the audible signal. The audible signal will not be heard in the radio headset unless the throttle is retarded to approximately 80% rpm or an unsafe gear condition exists. The button is placarded "Horn Cut-Out" and is spring-loaded to the OUT position. When depressed momentarily, the cutout button will silence the audible signal in the radio headset. However, the warning lights will remain illuminated. If the button has been used to silence the audible signal and the throttle has been advanced beyond approximately 80% rpm, the audible warning signal relay will reset. If the throttle is again retarded to approximately the 80% rpm position or less, the audible signal will again be heard in the radio headset. The audible signal cutout button is powered by the 28-volt dc essential bus through the landing gear warning light circuit.

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 1-25), 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. Momentarily depressing and releasing the button will engage nose wheel steering. Depressing the button again will disengage the system. Nose wheel steering is available on the ground any time the button is held depressed and the nose wheel does not exceed 50° from center in either direction. On TF airplanes, a nose wheel steering button is located on each control stick right-hand grip and depressing either button will engage or disengage nose wheel steering. 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.

WARNING

Depressing the brake pedals in flight may result in locked brakes during landing.

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.

Relief valves are installed to protect the brake system against excessive pressure. No emergency, antiskid, or parking brake system is provided. On TF airplanes, dual rudder pedals are 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 pneumatic 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 emergency dc bus power is available, high-pressure pneumatic system pressure will open the brakes for drag chute deployment. The drag chute handle must be pulled fully out, rotated 90° to the right and pulled out again.
- 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 165 KCAS. An extendible overrun barrier probe is installed and will extend simultaneously with the drag chute activation.

DRAG CHUTE HANDLE

The drag chute handle (figure 1-8), 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 handle 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. On TF airplanes, a drag chute handle (figure 1-9) is located above the forward end of both the pilot's and instructor's consoles. Either handle will deploy or jettison the drag chute. The handles are not interconnected.

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.
- If the emergency drag chute system is actuated, air will be introduced into the secondary hydraulic system. This requires an entry in AFTO 781 and purging prior to the next flight.

Pulling the drag chute handle will pneumatically extend the overrun barrier probe as the drag chute is activated.

PITOT-STATIC SYSTEM

The pitot-static system (figure 1-30) 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. On some airplanes, static pressure is also supplied to the CPU-46/A altitude computer for operation in conjunction with the servo barometric 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.

Note

The pitot static head installed on airplanes having the AIMS modification results in CAS being equal to IAS.

INSTRUMENTS

Note

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

PRIMARY ATTITUDE INDICATOR

The MM-3 or ARU-13/A attitude indicator (figure 1-31) is a remote attitude indicating system which provides visual pitch and roll indications of the airplane's flight attitude. This system consists of an attitude indicator on the flight instrument panel, an indicator amplifier, and MD-1 gyro control, and an MC-1 rate gyro. The indicator contains internal lighting which maintains the appearance of a daylight presentation under night-lighted conditions. On TF airplanes, an attitude indicator is installed on both the pilot's and instructor's instrument panels. Attitude indications are presented by a graduated attitude sphere, miniature airplane, a bank pointer, and a bank index scale. The airplane attitude is shown accurately through 360 degrees of roll and 82 degrees of climb or dive. Pitch and roll attitudes are shown by the circular motion of a universally mounted sphere, displayed as the background for a miniature reference airplane. The miniature airplane is always in proper physical relationship to the earth, horizon bar, and sky areas of the background sphere. On the sphere, the horizon is represented by a solid fluorescent line, the sky by a light gray area, and the earth by a dull black area. Horizontal lines (pitch reference scale) with five degrees of separation on the face of the sphere show accurate airplane attitudes up to 82 degrees of climb or dive. Bank angles are easily determined by reading the bank pointer in relation to the bank index scale on the upper half of the instrument. A pitch trim knob on the lower right side of the attitude indicator permits alignment of the horizon bar with the miniature airplane to compensate for varied aircraft weights and airspeeds. A warm-up period of

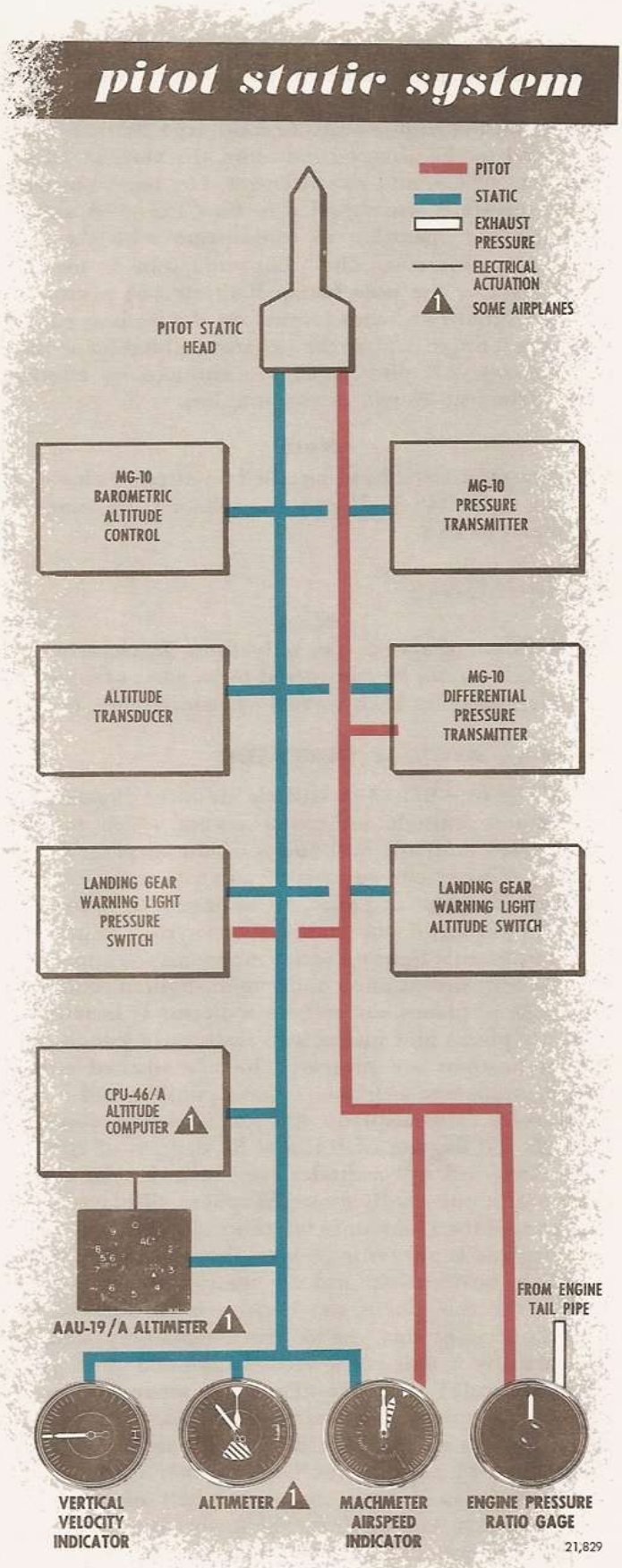


Figure 1-30

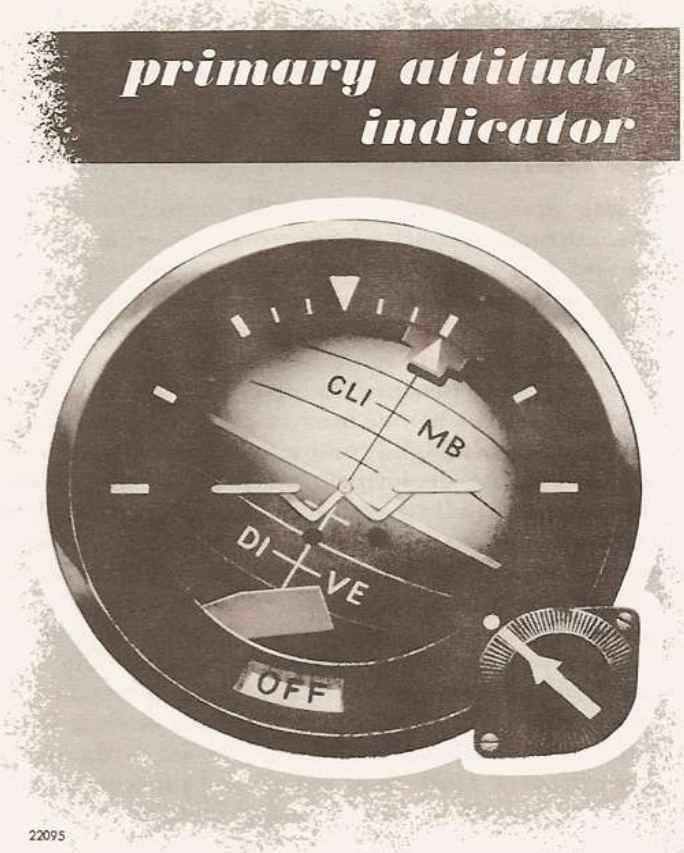


Figure 1-31

approximately 1½ minutes is required for the warning flag to disappear. There are no manual gyro caging provisions and it is normal for the attitude sphere to oscillate and vibrate during the warm-up period. The warning flag will appear any time ac essential power to the indicator is lost or interrupted.

WARNING

- If the warning flag requires longer than two minutes to disappear, or if the attitude sphere does not respond to the pitch trim knob, or if any oscillations of the sphere are noted on the indicator after the warning flag disappears, the possibility of a malfunction exists. Either of the above is cause for rejection of the attitude indicator and should be noted on AFTO 781. A slight reduction in power, or failure of certain electrical components within the system will not cause the warning flag to appear even though the system is not functioning properly. Therefore, periodically in flight, the attitude indications should be checked against other flight instruments, such as the artificial horizon on the radar scope, turn-and-slip, and vertical velocity.

- If a difference in indication is noted between the primary attitude indicator and the standby attitude indicator, cross-check other flight instruments and radar horizon (if available) to determine which is correct.

Turn and acceleration errors have been reduced in the attitude indicator by use of an improved displacement gyro (MD-1) and the MC-1 rate-switching gyro.

WARNING

A slight amount of pitch error on the 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 takeoff. The pitch error of the attitude indicator (MD-1 gyroscope) will be generated at a rate of 0.8 to 1.8 degrees per minute; therefore, the exact amount of error will depend upon the elapsed time of each period of acceleration or deceleration. The erection system will automatically remove the error after the acceleration/deceleration ceases.

If the pitch limit of 82 degrees is exceeded, the sphere will roll 180 degrees, reversing the normal position of the sky and earth reference area, but still providing an accurate indication of airplane attitude. As soon as the airplane returns below the 82 degree limit, the sphere rolls back to its normal position without introducing any errors into the system. The system is powered by the 115-volt, 400-cycle, 3-phase ac essential bus.

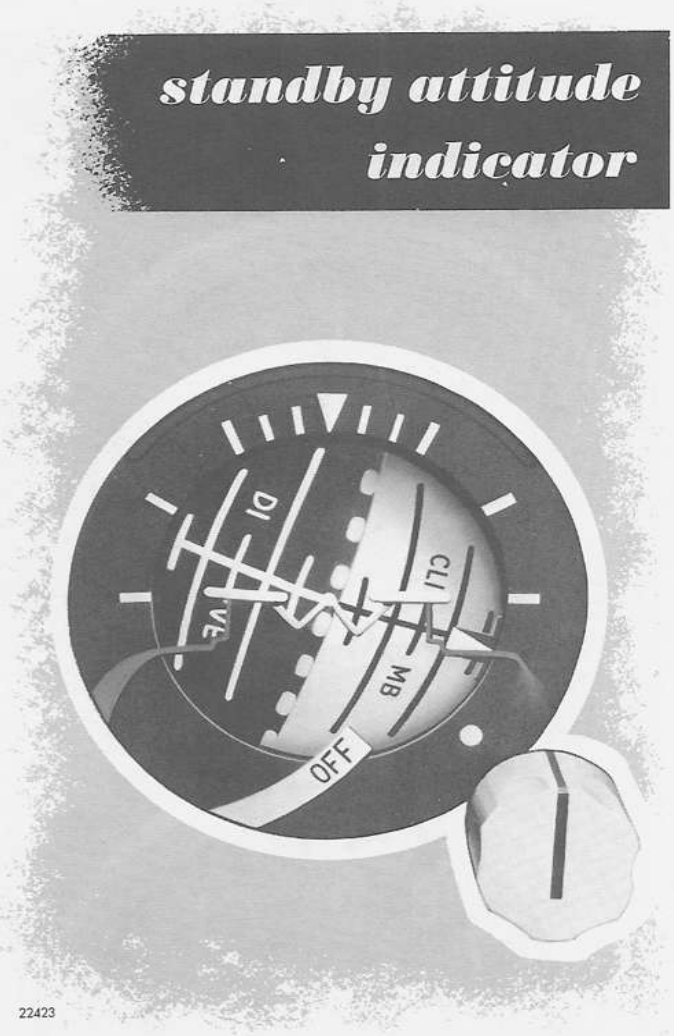


Figure 1-32

STANDBY ATTITUDE INDICATOR

The standby attitude indicator (figure 1-32), located on the upper right side of the pilot's instrument panel, functions as an alternate cockpit display and may be used as a check against the primary indicator, or as a backup indicator in the event of primary indicator malfunction. The standby indicator operates any time 28-volt dc essential power is available. The indicator provides indications of airplane attitude through 360° of bank and up to ±82° of pitch. The indicator is remotely operated from an MD-1 roll and pitch displacement gyro in the upper electronics compartment which is independent of the gyros which operate the primary attitude indicator. Climb, dive, and roll attitudes are shown by the circular motion of a universally mounted sphere displayed as the background for a miniature reference airplane. The miniature reference airplane is always in proper physical relationship to the simulated earth, horizon, and sky areas of the background sphere. The horizon is represented by a white horizontal bar, sky by a light grey area, and the earth by a black area. The pitch trim knob rotates the sphere to the proper position in relation to the miniature reference airplane to correct for variations of airplane level flight attitude. During initial gyro erection, and when power is off or is insufficient to keep the gyro stabilized, a flag placarded OFF appears in the bottom side of the indicator. A warm-up period of approximately 1½ minutes is required for the warning flag to disappear. The standby attitude indicator system receives ac power from a separate inverter by the 28-volt dc essential bus.

AIRSPEED-MACH INDICATOR

The airspeed-mach indicator (figure 1-33) 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

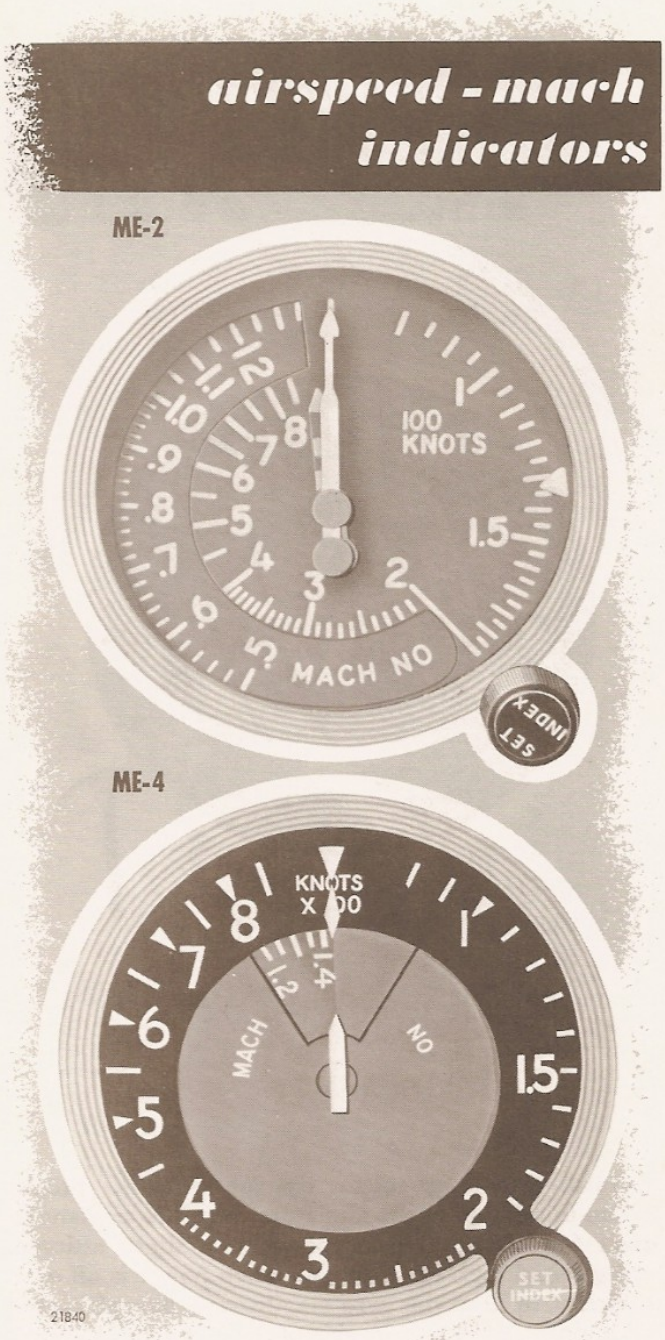


Figure 1-33

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. TF airplanes have an airspeed-mach indicator located on both the pilot's and instructor's instrument panel.

Note

Airplanes modified by TCTO 1F-102-1193 have an improved IFF/SIF radar transponder system (AIMS). This TCTO also incorporated an improved pitot static probe which essentially eliminated airspeed installation errors. Therefore, the readings on the airspeed-Mach indicator are now calibrated airspeed and true Mach number. Mach numbers above 0.5M have a small correction when extreme accuracy is required; however, under normal operation, they can be considered the same as true Mach. During the transonic speed range, some lag is encountered in airspeed readings; but it is not considered significant. A correction is provided in the event extreme accuracy is required. For airplanes not modified by TCTO 1F-102-1193 speeds will remain indicated airspeed (KAIS) and indicated Mach number (IMN) with appropriate corrections required to attain calibrated airspeed and true mach number. Airspeeds considered critical will be shown as XXX KCAS (XXX KIAS) to differentiate between configurations. Airspeeds not considered critical or airspeeds that are approximate will be listed as XX KCAS (KIAS) to show applicability to both configurations.

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. On TF airplanes, the accelerometer (if installed) is located at the lower left corner of the pilot's instrument panel.

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.

BAROMETRIC ALTIMETER

Some airplanes are equipped with a conventional barometric altimeter to indicate altitude. It incorporates the standard 100, 1000, and 10,000 foot pointers. It also has a low altitude warning symbol (striped area) which is visible at altitudes below 16,000 feet. An altimeter calibration card is provided in a holder located on the left side of the cockpit, above the console and forward of the throttle quadrant. On TF airplanes an altimeter is installed on both the pilot's and instructor's flight instrument panel.

WARNING

- It is possible to set the altimeter with a 10,000 foot error by rotating the barometric pressure set knob continuously in one direction until the barometric scale disappears, then reappears. If the correct altimeter setting is then established, the altimeter will read 10,000 feet in error.
- Large altimeter errors occur in flight at subsonic airspeeds. Refer to Appendix I for altimeter data.

AAU-19/A SERVO BAROMETRIC ALTIMETER

Note

Installation of the AAU-19/A altimeter is a part of the AIMS modification.

An AAU-19/A altimeter (figure 1-34) is located on the instrument panel. The altimeter has a counter-drum-pointer display. A single drum-type counter, labeled 100 FT, displays altitude in 100-foot increments. A dual digital counter, labeled 1000 FT, indicates altitude in 1000-foot increments. The first digit displays a barber-pole indication when below 10,000 feet. The dial face of the altimeter is marked in 100-foot increments around the periphery of the dial. A single pointer repeats the 100-foot indications of the drum and serves both as a vernier for the drum and as a quick indication of the rate and direction of altitude change. A four-digit counter, placarded IN HG, displays the barometric pressure setting which is set by an unmarked knob at the bottom left corner of the altimeter.



Figure 1-34

A reset-standby lever at the lower right corner of the instrument is used to select the operating mode of the altimeter. This lever is spring-loaded to an unmarked center position between positions marked RESET and STBY. The altimeter can be operated in either the servo mode or the standby mode. When operating in the standby mode, a standby flag indicator will display STBY in a window on the face of the altimeter.

Servo mode is the primary mode of operation. In this mode, the altimeter is electrically operated by a synchro signal from the CPU-46/A altitude computer and displays pressure altitude corrected for installation error. Servo mode is selected by holding the reset-standby lever in RESET position until the standby flag disappears. A failure monitor circuit automatically causes the system to revert to standby mode if the system becomes unreliable because of power loss or failure within the system. In event the system reverts to standby mode, an attempt may be made to reset to servo operation by holding the reset-standby lever in RESET position. If the malfunction was transient, the altimeter will reset and the standby flag will disappear. In the standby mode, the altimeter operates from the static pressure system and displays pressure altitude which is not corrected for installation error. The altimeter will be in standby mode when power is first applied and will remain there until it is reset to servo. Altimeter operation may be manually changed from servo mode to standby mode by holding the reset-standby lever in STBY position until the standby flag appears (one to three seconds).

The altimeter contains an internal vibrator which operates continuously and minimizes friction. This enables the instrument to provide a smooth display during altitude change. If the vibrator fails, the altimeter will continue to function but will operate less smoothly.

Instrument error can be checked under static conditions on the ground. With the correct barometric pressure setting in the instrument, the altimeter should read within ±75 feet of field elevation in either servo or standby mode. In addition, readings in the two modes should be within 75 feet of each other.

The altimeter vibrator receives 28-volt dc essential power. The CPU-46/A altitude computer uses 115-volt ac essential bus power and in turn supplies 115-volt and 26-volt power to the altimeter.

TURN-AND-SLIP INDICATOR

A conventional turn-and-slip indicator (figure 1-8) located on the instrument panel is rated for a four-minute turn (1½° per second). The turn-and-slip indicator is mounted perpendicular to the longitudinal axis of the airplane and presents turn indications. On TF airplanes, a turn-and-slip indicator (figure 1-9) is in-

stalled on both the pilot's and instructor's instrument panel. The turn-and-slip indicator is powered by the 28-volt dc essential bus.

MAGNETIC 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 compass light switch on the right side of the cockpit. The switch is wired in series with the flight instrument panel lights rheostat and the compass light will be dimmed or brightened accordingly if the compass light switch is in the ON position. On (TF) airplanes, the compass light switch is located on the warning light panel. The switch is powered by the 28-volt dc essential bus. A magnetic compass correction card and holder are located on the left side of the cockpit above the console and aft of the throttle quadrant.

CLOCK

The clock (figures 1-8 and 1-9) 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.

HEADING INDICATOR

Refer to NAVIGATION EQUIPMENT, Section IV.

EMERGENCY EQUIPMENT

INDICATOR AND WARNING LIGHTS

Refer to figures 1-35 and 1-36 for indicator and warning lights.

CANOPY KNIFE

On some airplanes, a canopy knife (figure 3-2) is provided to break through the canopy in the event the canopy cannot be raised or jettisoned after a crash landing. A knife is located on either the left or right canopy rail on (F) airplanes and on the canopy frame above each pilot's head on (TF) airplanes.

LIFE RAFT DEFLATION DEVICE

A life raft deflation device (figure 1-37) is installed on the control stick for the purpose of automatically deflating the life raft, should the raft inadvertently inflate during flight. A slight back or side movement of the control stick may be required when an automatic deflation does not occur.

MASTER WARNING SYSTEM

A warning light panel (figures 1-35 and 1-36), 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 (except for testing), canopy unlocked warning light, hydraulic pressure-low warning light, or fire and overheat warning light. On (TF) airplanes, the warning light panel is located on the center console.

Master Warning Test and Reset Switch

A three-position master warning test and reset switch (figure 1-36), 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 and the landing gear warning lights should illuminate and the audible gear warning should operate.

Note

Placing the switch to TEST is a 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 1-38) 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 (figure 1-38), 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 1-38), 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. On TF airplanes, a fire warning light is located on each instrument panel. The test switch (figure 1-38) is located to the right of the warning light on the pilot's instrument panel. When the test switch is actuated or an abnormally high temperature is indicated, both lights should illuminate.

ARREST HOOK SYSTEM

An arrest hook system (figure 1-39) is installed for use with "water squeezer" or "chain" type arresting barriers. The system consists of an arrest hook installed on the bottom aft portion of the fuselage, an arrest hook uplock, a self-contained air-and-oil hydraulic pressure system, and an all-mechanical cable arrest hook release system attached to a handle in the cockpit. An arrest hook light, powered from the 28-volt dc essential bus, is installed on the instrument panel, and when illuminated provides an indication that the arrest hook has been extended. Hydraulic pressure from the system actuates a hydraulic cylinder which lowers the arrest hook to the extended position. Hydraulic snubbing pressure holds the hook arm down and minimizes bounce of the hook. A pressure gage is provided to check the air pressure in the system and may be observed through the access door on the left side of the fuselage. A ground safety pin with a red streamer is installed on the hook arm to prevent inadvertent extension when the airplane is on the ground. See figure 1-44 for hydraulic fluid specification.

Arrest Hook Release Handle

On airplanes with a tailhook arresting system, an arrest hook release handle (figure 1-29) is installed on the left side of the cockpit below the landing gear emergency extension handle. The handle is formed to resemble a hook and is used to unlatch the arrest hook when required for barrier engagement. When the arrest hook release handle is pulled (approximately four inches) to the OUT (stop) position, a cable actuates the arrest hook uplock assembly unlatching the hook arm. Hydraulic pressure in the system accumulator then actuates a hydraulic cylinder which extends the hook for barrier engagement. If the arrest hook is inadvertently extended in flight, it will not introduce any problems while airborne. A safe landing can be made with the hook arm extended. There are no provisions for retracting the arrest hook in flight.

WARNING

- If the arrest hook is inadvertently extended in flight, and if there is a barrier on the approach end of the runway, the airplane touchdown point must be compensated to avoid inadvertent engagement with the barrier. When extended, the arrest hook may drag for 1000 feet before touchdown during a normal landing. When landing with the arrest hook extended, fly a high final approach and touch down well past the arresting cable on the approach end of the runway.
- The arrest hook release handle must be pulled straight out. Any sideload or twisting action on the handle may make it impossible to pull out.

SURVIVAL KIT

The survival kit (figure 1-40) is designed to fit in the ejection seat. The kit, fitted with a rubber cushion which snaps to the kit, is of fiber glass 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, communications leads, and the green knob for bailout bottle manual actuation. The oxygen regulator in the kit controls all oxygen used by the pilot 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, the bailout bottles provide breathing 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. 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

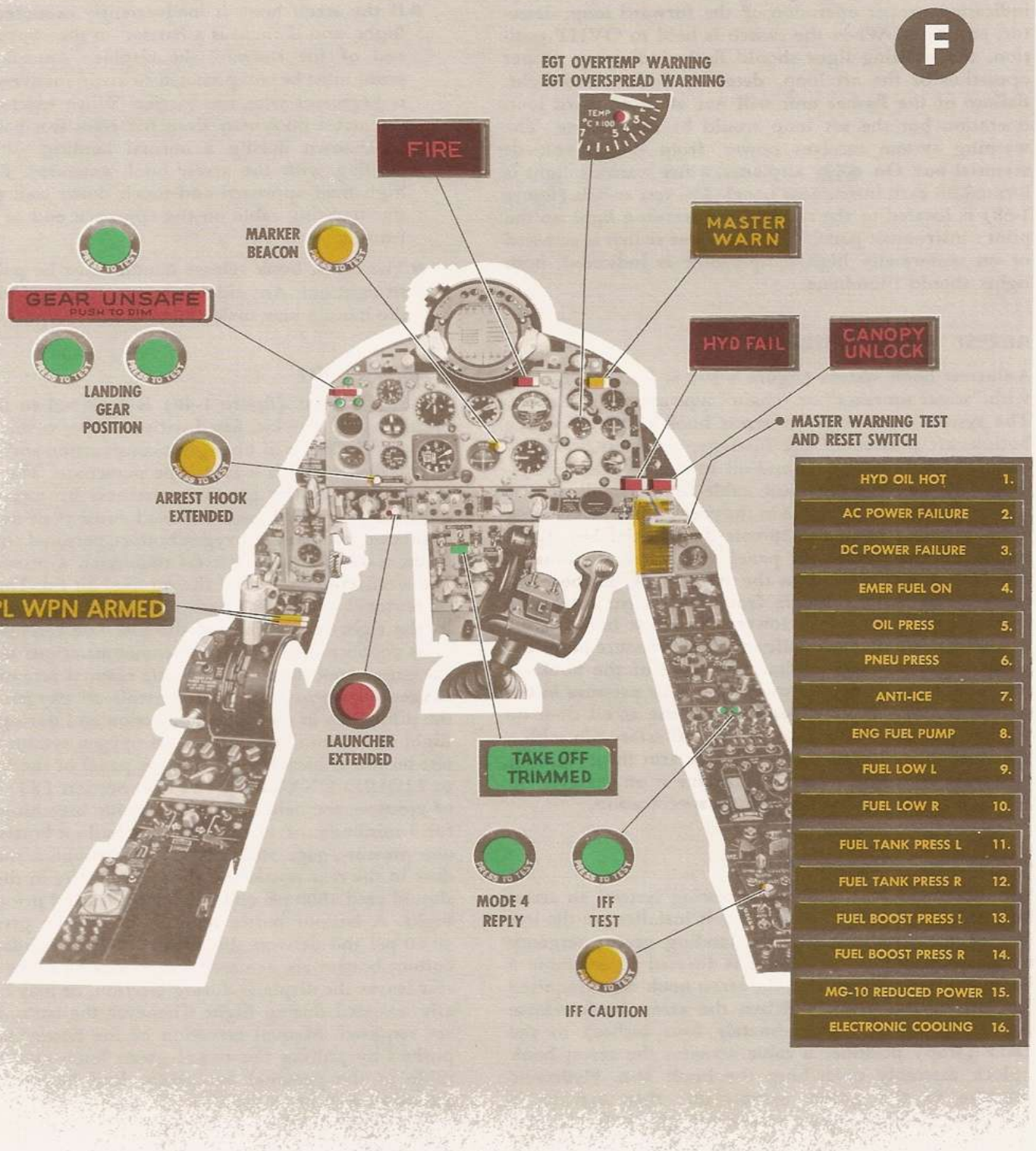
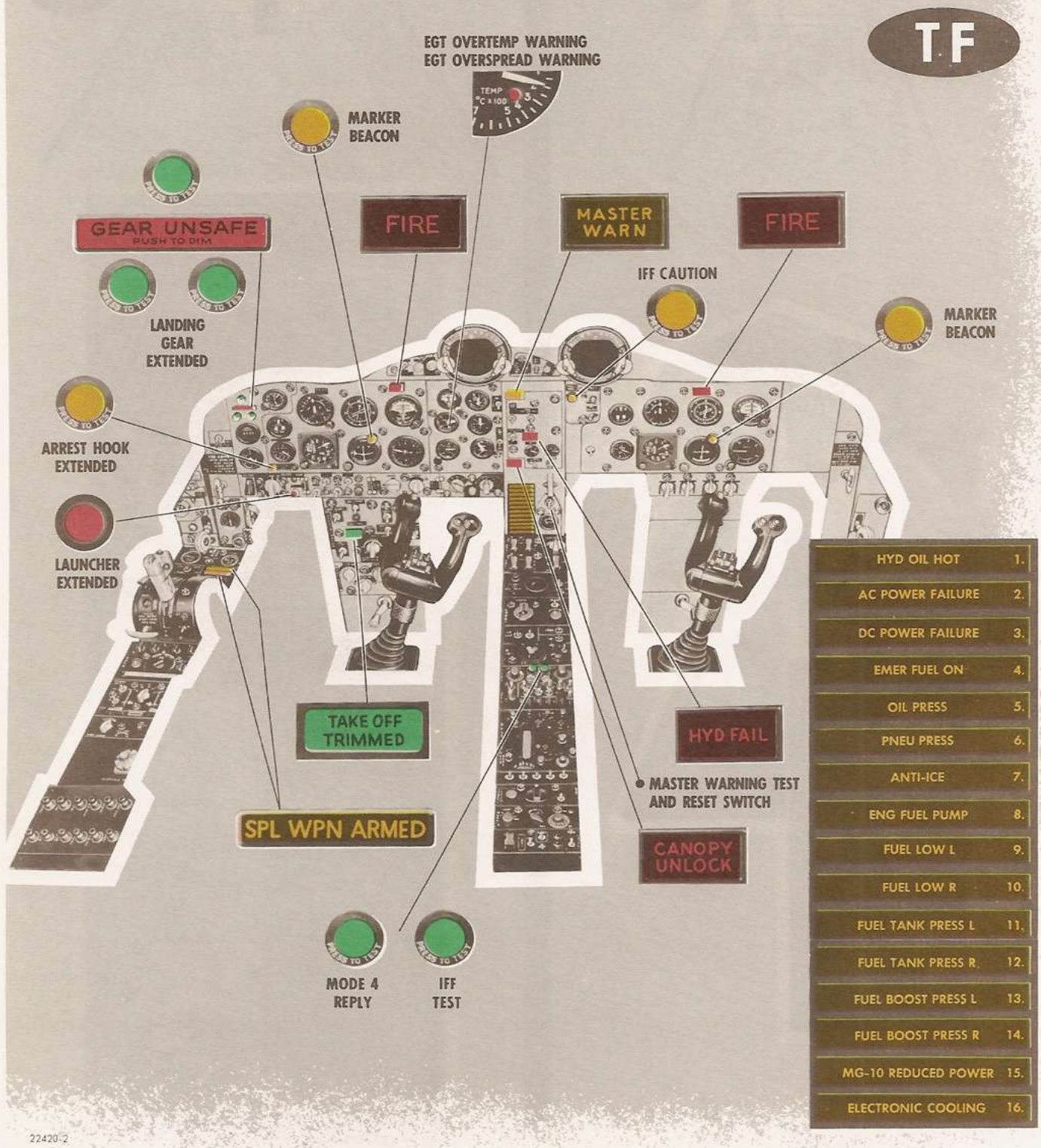

F
EGT OVERTEMP WARNING
EGT OVERSPREAD WARNING
FIRE
MASTER WARN
MARKER BEACON
GEAR UNSAFE
PUSH TO DIM
HYD FAIL
CANOPY UNLOCK
LANDING GEAR POSITION
MASTER WARNING TEST AND RESET SWITCH
ARREST HOOK EXTENDED
SPL WPN ARMED
LAUNCHER EXTENDED
TAKE OFF TRIMMED
MODE 4 REPLY
IFF TEST
IFF CAUTION
HYD OIL HOT 1.
AC POWER FAILURE 2.
DC POWER FAILURE 3.
EMER FUEL ON 4.
OIL PRESS 5.
PNEU PRESS 6.
ANTI-ICE 7.
ENG FUEL PUMP 8.
FUEL LOW L 9.
FUEL LOW R 10.
FUEL TANK PRESS L 11.
FUEL TANK PRESS R 12.
FUEL BOOST PRESS L 13.
FUEL BOOST PRESS R 14.
MG-10 REDUCED POWER 15.
ELECTRONIC COOLING 16.
22420-1

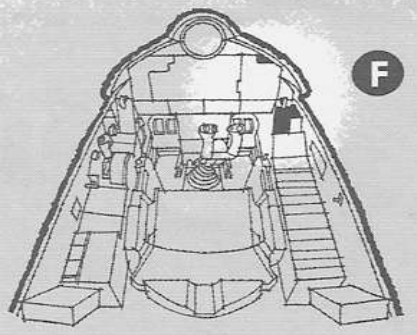
Figure 1-35

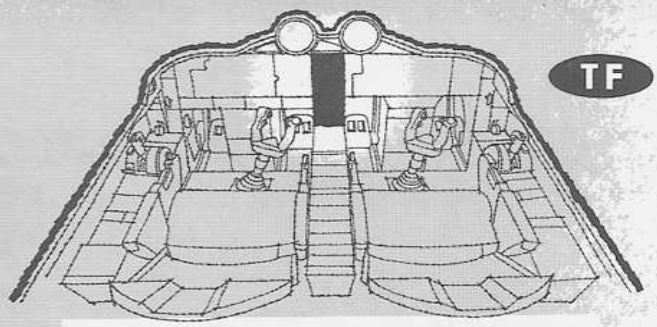
indicator and warning lights (typical)



22420-2

warning light panels (typical)

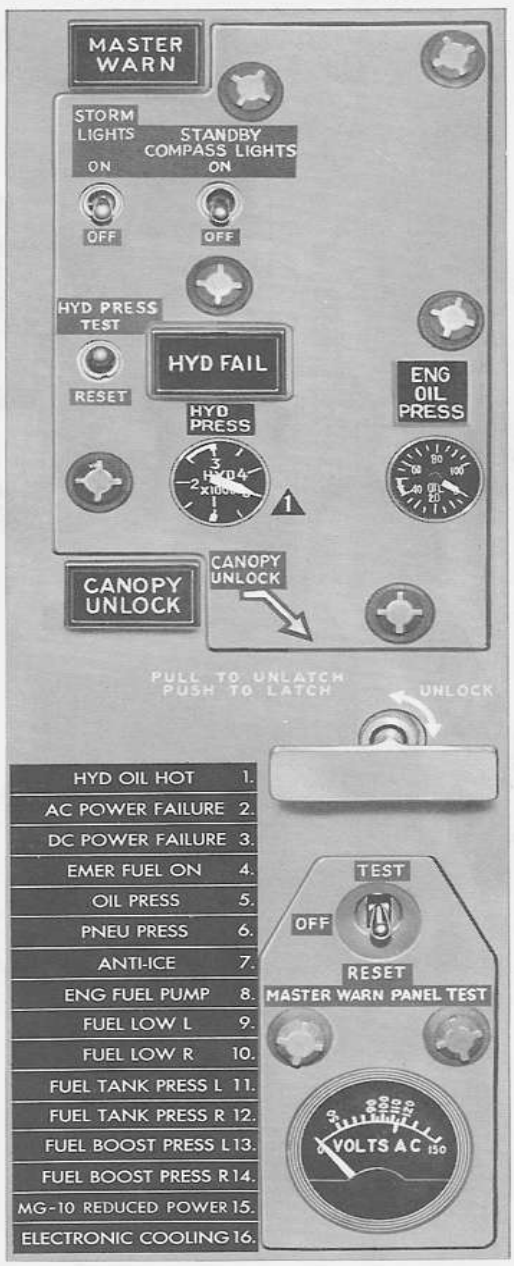








HYD OIL HOT	1.
AC POWER FAILURE	2.
DC POWER FAILURE	3.
EMER FUEL ON	4.
OIL PRESS	5.
PNEU PRESS	6.
ANTI-ICE	7.
ENG FUEL PUMP	8.
FUEL LOW L	9.
FUEL LOW R	10.
FUEL TANK PRESS L	11.
FUEL TANK PRESS R	12.
FUEL BOOST PRESS L	13.
FUEL BOOST PRESS R	14.
MG-10 REDUCED POWER	15.
ELECTRONIC COOLING	16.



HYD OIL HOT	1.
AC POWER FAILURE	2.
DC POWER FAILURE	3.
EMER FUEL ON	4.
OIL PRESS	5.
PNEU PRESS	6.
ANTI-ICE	7.
ENG FUEL PUMP	8.
FUEL LOW L	9.
FUEL LOW R	10.
FUEL TANK PRESS L	11.
FUEL TANK PRESS R	12.
FUEL BOOST PRESS L	13.
FUEL BOOST PRESS R	14.
MG-10 REDUCED POWER	15.
ELECTRONIC COOLING	16.

1 SOME AIRPLANES

21,899

Figure 1-36

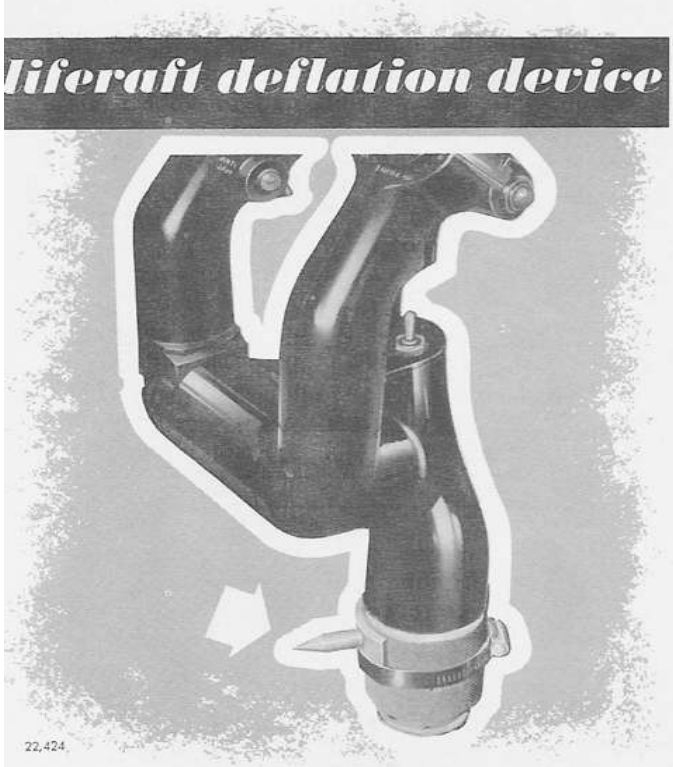


Figure 1-37

urvival kit and should contain the items necessary for urvival in the area of operation. On airplanes modified y TCTO 1F-102-1209, the survival kit is automatically leployed. During parachute deployment, tension is pplied to a cable between the parachute right riser and he automatic actuator in the survival kit which causes a our-second delay gas generator cartridge to fire. The gas ressure forces a piston in the multistage release to oper- te the release mechanism and the following events occur:

a. Lid unlocks.

b. Personal leads separate from the kit.

c. Kit harness release assemblies separate from kit.

d. Kit falls away (attached to the crew member on a 25-foot dropline).

e. Life raft inflates.

f. Multistage release handle remains in place on survival kit.

hould the automatic release mechanism malfunction, the urvival kit may be manually deployed by pulling the nultistage release handle.

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.

Note

An automatic deployment selector lever is located below the emergency release handle and is used to deactivate the automatic deployment feature when the kit is removed for maintenance. This lever should always be in the AUTO (forward) position during flight.

Survival Kit Emergency Release Handle

The yellow emergency release handle (figure 1-40) 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.

WARNING

The release handle must be pulled rapidly through the complete arc of travel to allow proper functioning of the survival kit system. Failure to follow this procedure could present a serious hazard to the aircrew during parachute landing.

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 left-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

- 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.

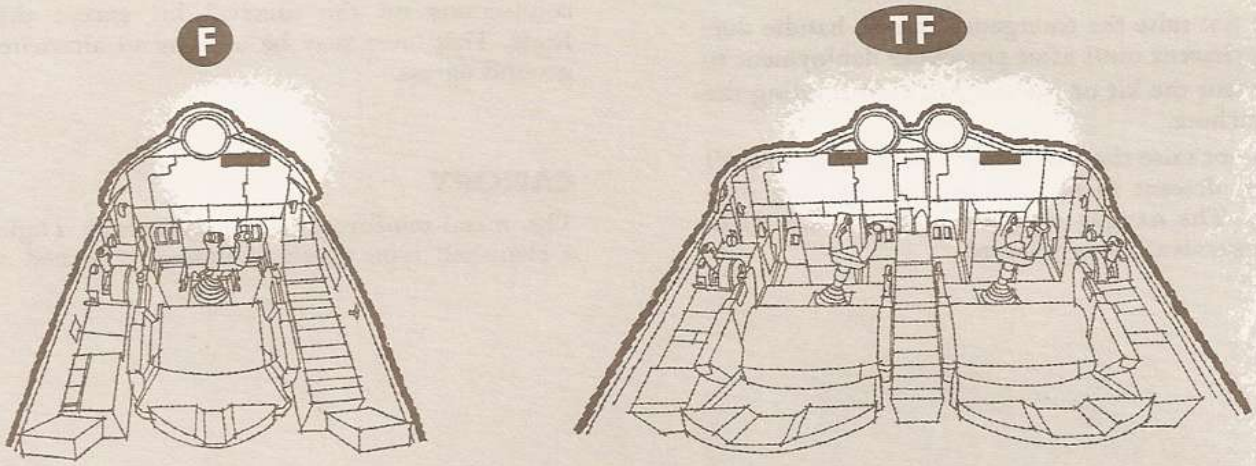
Survival Kit Harness Release Lever

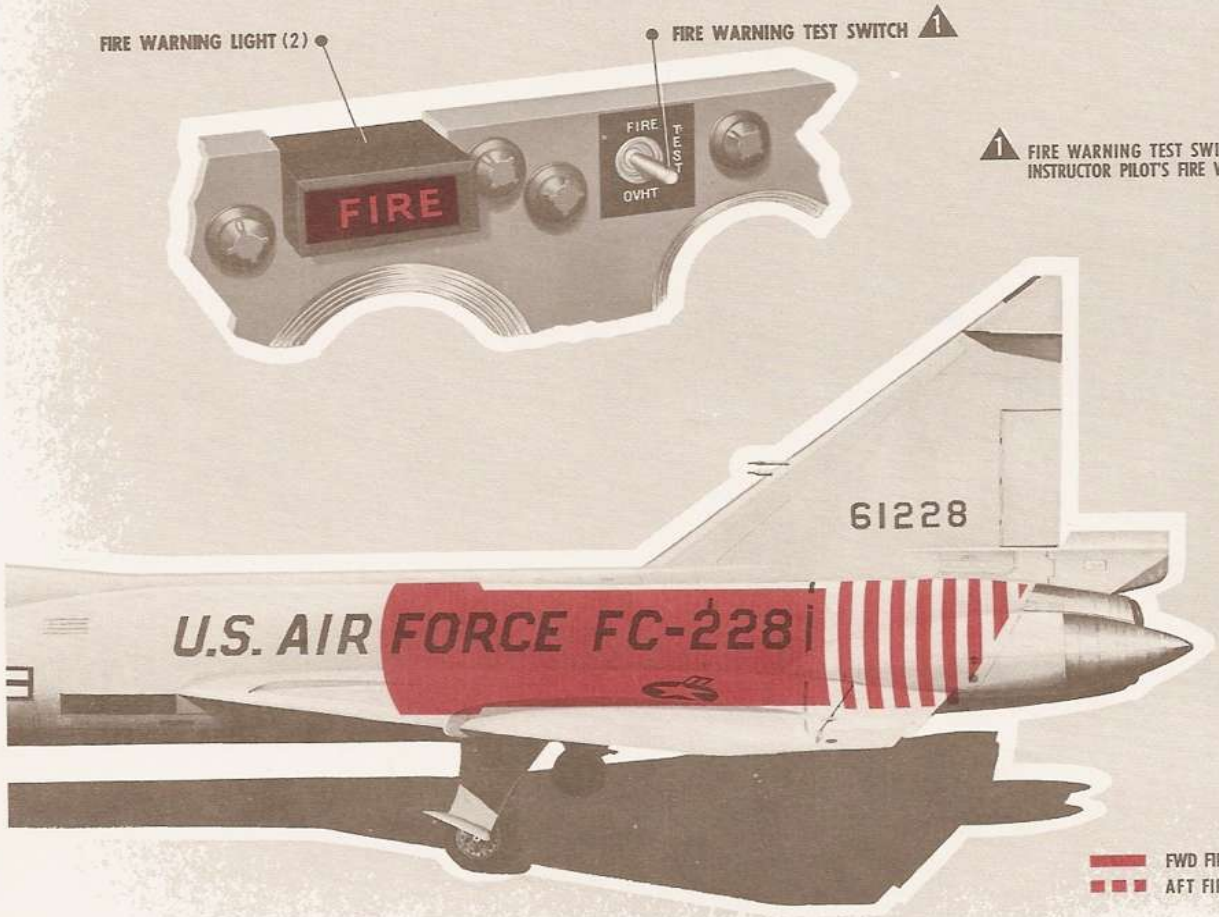
The harness release (figure 1-40) is a lever aft of the emergency release handle on the right-hand side of the survival kit. It is placarded "Harness Release" and has a hole into which a finger may be inserted for raising. The lever is hinged at the rear and, when raised, releases all connections on the survival kit except the personal leads. This lever may be used as an alternate means of ground egress.

CANOPY (F)

The metal-reinforced plexiglas canopy (figure 1-1) is a clamshell type which is manually opened and closed.

fire warning system (typical)





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Figure 1-38

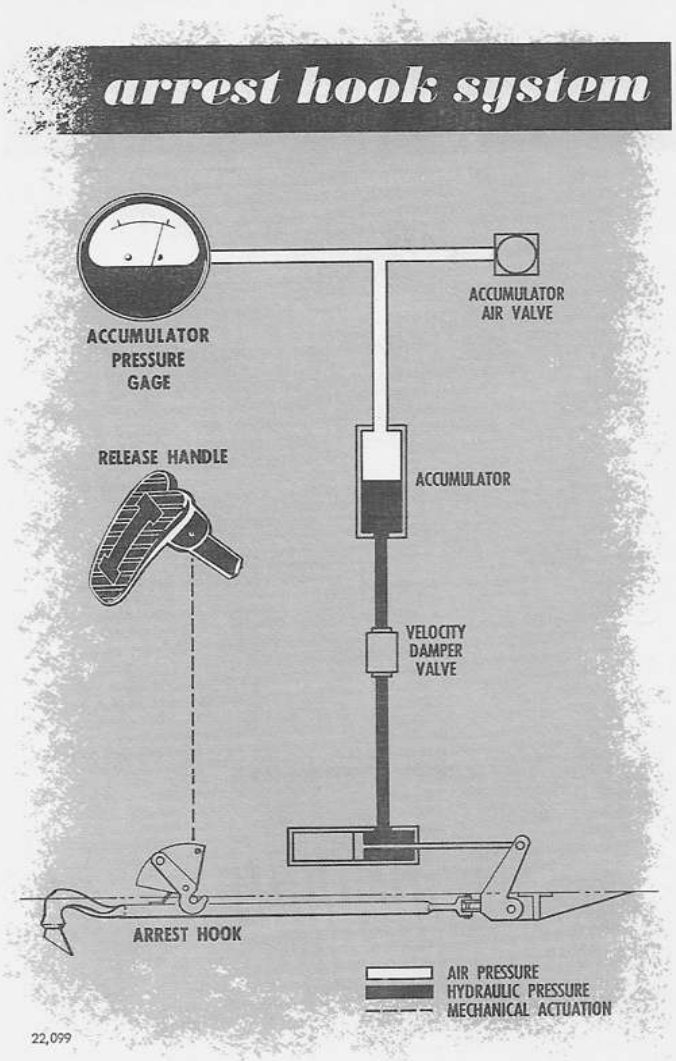


Figure 1-39

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. 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. 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.

WARNING

- The seat catapult initiator is armed mechanically when the canopy separates from the airplane.
- If the canopy is open during taxi operations, observe canopy limit speeds. Keep hands away from canopy sill unless the canopy support tool is in place.

CANOPY SEAL (F)

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 (F)

A removable canopy ground support tool (figure 1-28) 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 (F)

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

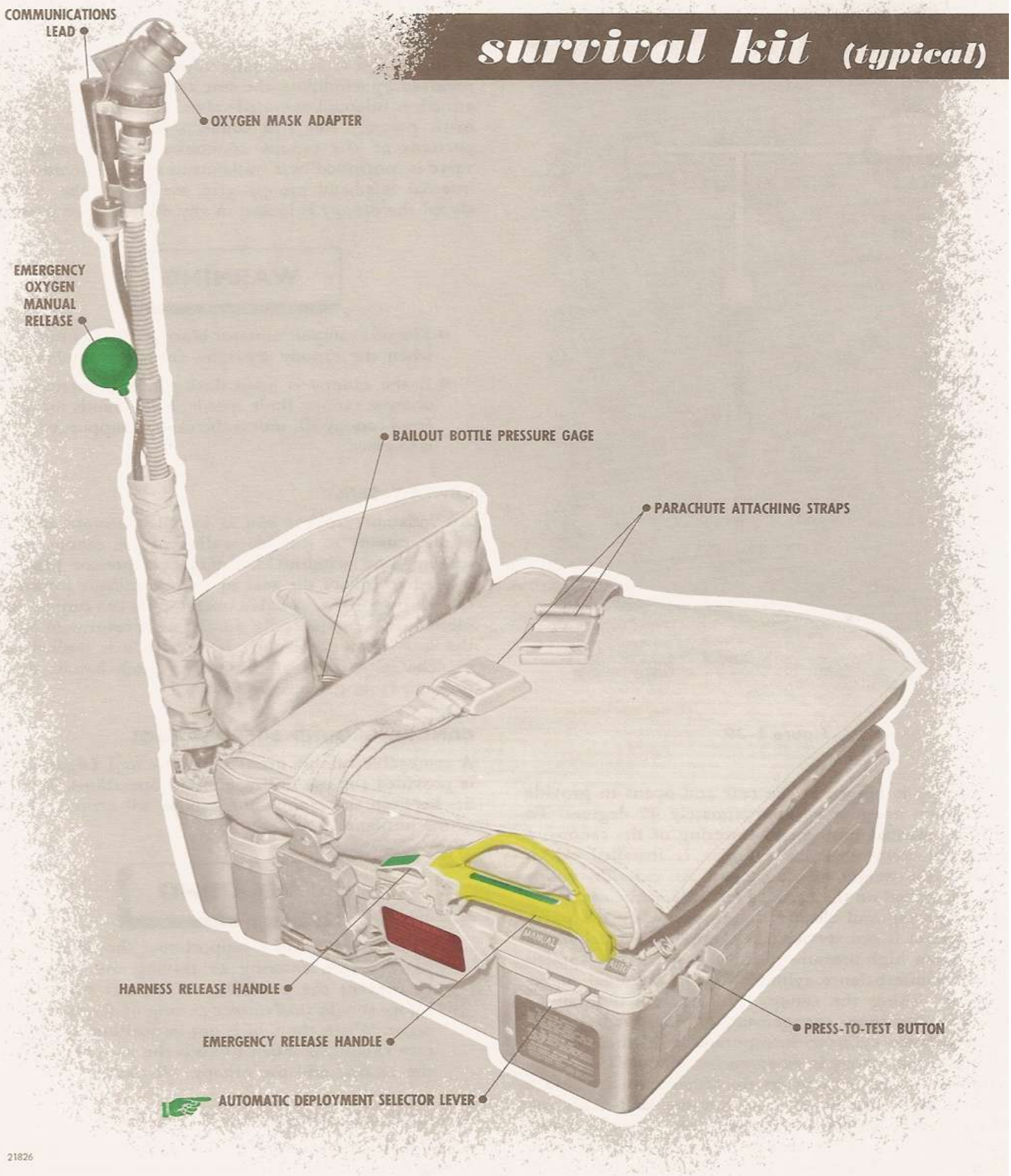

survival kit (typical)
COMMUNICATIONS LEAD
OXYGEN MASK ADAPTER
EMERGENCY OXYGEN MANUAL RELEASE
BAILOUT BOTTLE PRESSURE GAGE
PARACHUTE ATTACHING STRAPS
HARNESS RELEASE HANDLE
EMERGENCY RELEASE HANDLE
AUTOMATIC DEPLOYMENT SELECTOR LEVER
PRESS-TO-TEST BUTTON
MANUAL
AUTO
21826

Figure 1-40

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; therefore, ensure that the area beneath the canopy is clear to prevent personal injury or damage to equipment.

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 canopy opening 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.

WARNING

Depletion of high-pressure pneumatic system will allow the canopy to fall when battery power is turned off.

CANOPY EXTERIOR RELEASE BUTTON (F)

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 2-1) 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.

CANOPY LATCH HANDLE (F)

The T-shaped canopy latch handle (figure 1-8), 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 relieves canopy seal pressure. When the handle is fully out (approximately eight inches), the pneumatic control valve is in the counterbalance condition. The canopy can then be raised manually.

CANOPY HOLD BUTTON (F)

The canopy hold button (figure 2-1) 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 (F)

The yellow canopy jettison handle (figure 1-42) 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.

Note

The canopy can be jettisoned from any position, i.e., closed, cinchdown, partially open, or fully open.

EJECTION SEAT HANDGRIPS (F)

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

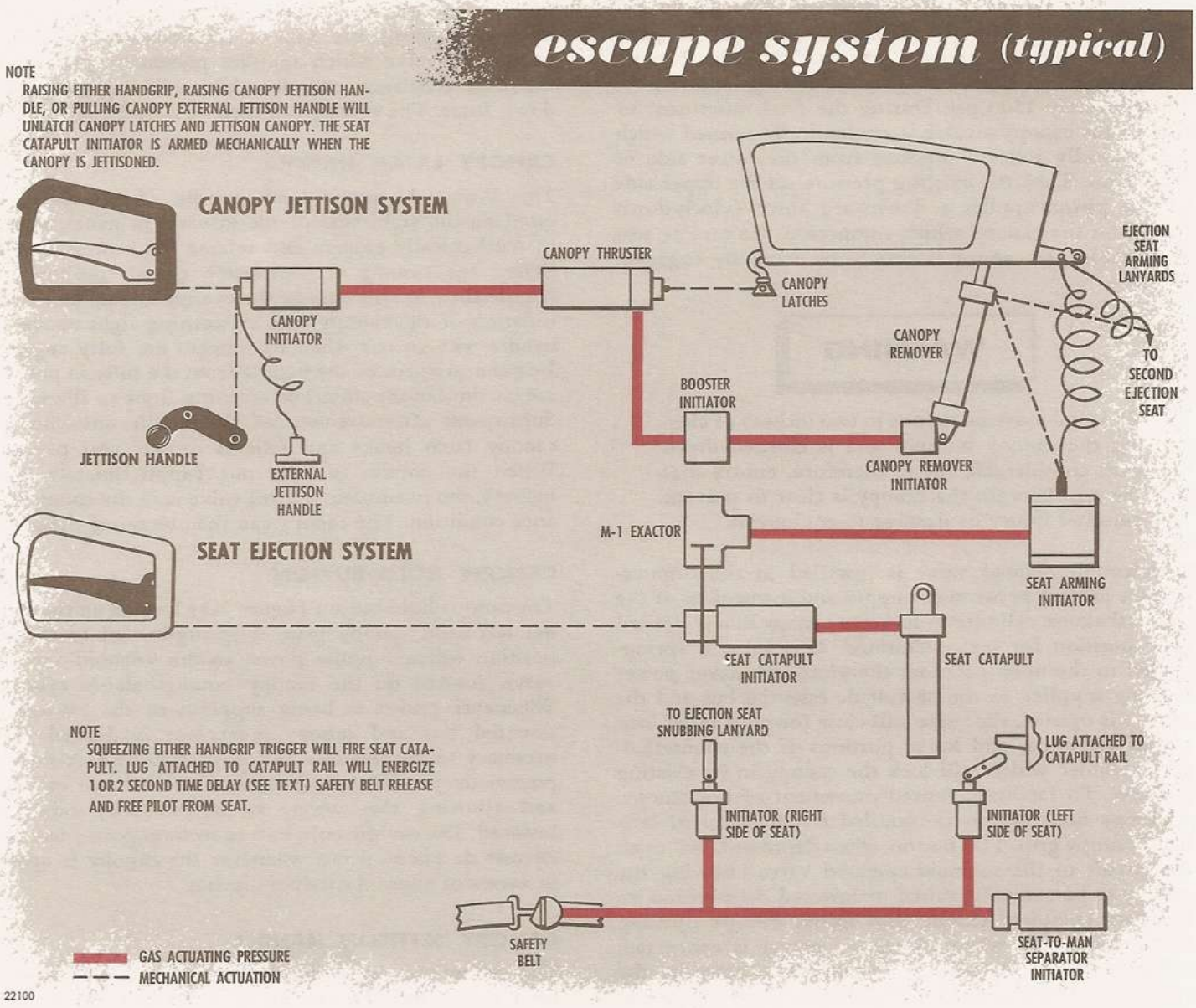


Figure 1-41

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 3-1). 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 3-1). 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 1-41) 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 (F)

The red canopy unlocked warning light (figure 1-8), 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.

CANOPY (TF)

The metal-reinforced plexiglas canopy (figure 1-1) 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°. To facilitate manual raising and lowering of the canopy, a pneumatic counterbalance cylinder is installed which equalizes the weight of the canopy. A clamping or cinch-down force (by pneumatic pressure stored within the cylinder) 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 (prior to seat ejection) is accomplished by raising either ejection seat handgrip on either seat. Emergency jettisoning of the canopy only should be accomplished by raising either canopy jettison handle, located on the forward end of the left-hand armrest), or by pulling the canopy external jettison handle. When the canopy is jettisoned by the above methods, it arms the ejection seat arming initiator. If the canopy cannot be jettisoned in flight and it is manually raised and allowed to be removed by windblast, the seat arming initiator is also armed.

WARNING

The seat catapult initiator is armed mechanically when the canopy separates from the airplane.

To facilitate taxiing with the canopy open, the canopy can be pneumatically locked at any desired angle of opening.

CAUTION

- To prevent small airloads or taxiing shocks from shifting canopy position and possible damage to the canopy counterbalance cylinder, do not taxi the airplane with the canopy open unless the cylinder is in the locked position (not counterbalance).
- To avoid abnormal stresses on the canopy when taxiing with the canopy open, observe canopy limit speeds as stated in Section V.

CANOPY SEAL (TF)

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 TOOLS (TF)

Two removable canopy ground support tools (figure 1-28) are provided for use during ground operation. The tools fit between the canopy and canopy sills on both sides of the airplane.

WARNING

The canopy ground support tools should be in place before entering or leaving the cockpit to prevent the possibility of serious personal injury should the canopy inadvertently close. When the tools are not in position, special care should be taken to keep clear of the area between the canopy and canopy sill.

CANOPY COUNTERBALANCE CYLINDER (TF)

The canopy counterbalance and remover cylinder functions as a canopy normal operation actuator; and also serves to jettison the canopy in an emergency. During

normal canopy operation the cylinder counterbalances the canopy to assist manual raising or lowering and applies a cinch-down force on the canopy to put pressure on the weather seal and facilitate engagement of the locking hooks. It also utilizes air under pressure to lock the canopy in the open position at any desired angle of opening. The cylinder functions pneumatically for canopy normal operation, and by means of a ballistic charge for emergency canopy jettisoning. For pneumatic operation the cylinder is precharged with nitrogen or dry air to a pressure of 1500 (±25) psi.

Note

The cylinder is not connected to the airplane's high-pressure pneumatic system, and is charged separately from the system.

The cylinder incorporates a pressure gage, two selector valves with actuating solenoids, a relief valve to prevent overcharging, a pressure regulating valve to regulate the venting of air during cinch-down operation, a filler valve through which the cylinder is charged, and a ballistic cartridge.

Note

If the air supply is not replenished when the pressure gage shows a low reading, the air supply in the cylinder will be depleted by frequent closings, to the point where low counterbalance pressure will make the canopy difficult to open.

The cylinder is fastened to the airplane structure and the piston rod is attached to the canopy. The piston divides the interior of the cylinder into two chambers. Exchange of air pressure between the chambers on each side of the piston is effected through external tubing, in which the solenoid-actuated selector valves are located. The 28-volt dc essential bus current (some airplanes*) or 28-volt dc nonessential current (other airplanes**) used to energize the selector valve solenoids is controlled by microswitches moved by the canopy latch handle and latch control linkage. Provisions are made for actuating the solenoid valves manually from either inside or outside the airplane without the use of electric power.

CANOPY LATCH HANDLE TF

The T-shaped canopy latch handle (figure 1-9), located between the pilot's and instructor's instrument panels, provides the normal means of engaging and releasing the canopy latches from inside the cockpit. Movement of the mechanical latch linkage by the handle automatically sequences microswitches controlling 28-volt dc nonessential bus power to the canopy selector valve solenoids. The canopy is opened by pulling on the latch handle. 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 relieves canopy seal pressure. When the handle is fully out (approximately eight inches), the pneumatic control valve is in the counterbalance condition. The canopy can then be raised manually. On some airplanes, the canopy can be locked at any desired angle of opening by rotating the latch handle 90° clockwise to horizontal position. Rotation of the handle actuates a microswitch to deenergize the counterbalance solenoid. The counterbalance selector valve goes to neutral position, thus creating an air lock on both sides of the counterbalance piston, and locking the canopy in place. To close the canopy, the latch handle is turned 90° counterclockwise to vertical position to energize the counterbalance solenoid and apply counterbalance pressure. The canopy is then lowered manually. As the canopy guide fittings lower into the slots in the canopy sill, a fitting on the right-hand side unlocks a mechanical safety lock and allows the canopy latch handle to be pushed in. As the handle is pushed in, a microswitch is actuated to energize the cinch-down solenoid and apply a cinch-down pressure to the canopy. At the same time, movement of the handle operates the latch linkage to engage the latches. An overcenter indicator is provided on the canopy latch linkage just forward of the latch on each side of the cockpit. When the latches are fully engaged, the indicator will point to the last of the three lines.

WARNING

Do not push the latch handle in until the canopy is resting on the seal. If the canopy is slightly open at the time cinch-down pressure is applied, there is danger of injury to hands or fingers if they are caught between the canopy and the windshield frame or the cockpit sill.

On 56-2324 and later airplanes, the canopy is released and counterbalanced in the same manner as on earlier airplanes when dc nonessential power is available. However, the canopy is locked open by placing the canopy latch handle inward approximately ¼ inch from the fully out position. To close the canopy, the handle is returned to the fully out position and the canopy lowered manually. A plunger on the aft right-hand side of the canopy actuates switches that cause a cinch-down force in the canopy cylinder. The canopy latches are hooked and the seal is pressurized when the canopy latch handle is pushed to the fully in position.

EXTERIOR CANOPY LATCH CONTROL TF

To gain access to a closed airplane in an emergency, the canopy latching mechanism and counterbalance cylinder selector valve switches can be operated from outside the cockpit. A control box in the latching system is located just below the canopy sill on the left side of the fuselage. A shaft having a square recess in the end projects through the fuselage skin from the control box. The end of the shaft is flush with the surface of the fuselage skin. To

*54-1351 thru 56-2323.
**56-2324 & on.

open the canopy from outside the airplane, a square-ended tool is inserted in the shaft. On some airplanes, the exterior canopy latch control releases the canopy latches when rotated 90° counterclockwise. Counterbalance is then automatically effected if either the airplane's battery or external dc power is available to operate the solenoid valves. The canopy can then be opened manually and locked open by rotating the cockpit canopy latch handle. To close the canopy from the outside, the cockpit canopy latch handle must be vertical. Without dc power, it is necessary for one man to operate the solenoid manual override handle in the intermediate electronics compartment while another man operates the exterior canopy latch control. On other airplanes, the exterior canopy latch control duplicates the functions of the cockpit canopy latch handle. A latch tool for operation of the control is provided in the latch tool stowage recess beneath the canopy on the left side of the airplane. When the tool is inserted in the exterior latch control socket and rotated 90° clockwise, the canopy is unlatched and in counterbalance if dc power is on. The canopy can be locked in any open position by backing off approximately ¼ inch on the exterior latch control. To close the canopy from the outside, the canopy is lowered manually and the latch tool rotated 90° counterclockwise. Cinch-down is automatically effected when the canopy plunger actuates the switches on the right-hand canopy sill. On these airplanes, a solenoid manual override handle and canopy cinch-down switch are provided in the latch tool stowage recess to permit one-man operation of the canopy when dc power is not available.

SOLENOID MANUAL OVERRIDE HANDLES TF

Two solenoid manual override handles are provided to permit operation of the canopy system when electrical power is not available. One handle (for interior use) is located in the cockpit aft of the pilot's seat. The other (for exterior use) is located below the canopy on the left side of the airplane and is accessible through an access door placarded "Canopy Control." The handles are placarded PULL FOR CBL (pull for counterbalance). With the canopy latches released, pulling either handle places the canopy cylinder in the counterbalance condition and permits manual opening and closing.

CANOPY CINCH-DOWN SWITCH TF

A canopy cinch-down switch is provided in the latch tool stowage recess. The switch operates a solenoid valve to provide cinch-down pressure in the canopy cylinder. Power for the exterior canopy cinch-down switch is taken directly from the airplane battery. Therefore, the switch is operative even when the automatic cinch-down feature is inoperative due to the airplane power supply being deenergized.

CANOPY JETTISON HANDLES TF

The yellow canopy jettison handles (figure 1-42) may be used to jettison the canopy by a ballistic charge. The handles are located on the forward end of the left armrest of each seat and incorporate 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 TF

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

CANOPY EXTERNAL JETTISON HANDLE TF

During emergency conditions the canopy can be jettisoned from outside the airplane for emergency rescue by the external jettison handle (figure 3-1). 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 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 1-41) which gas-fires the thruster. This releases the canopy latches and cinch-down 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, and 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 TF

The red canopy unlocked warning light (figure 1-9), 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.

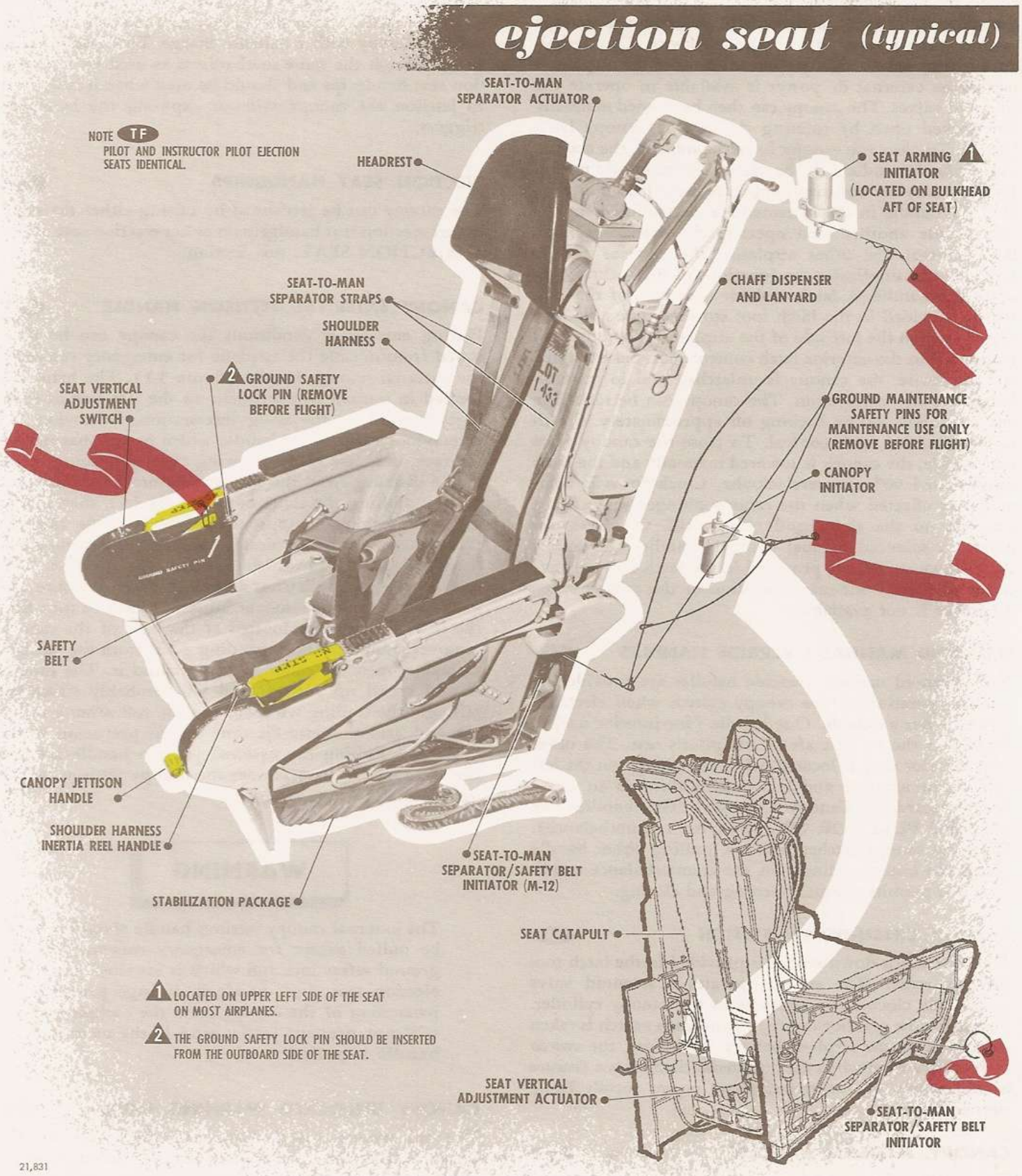

ejection seat (typical)
SEAT-TO-MAN SEPARATOR ACTUATOR
NOTE TF
PILOT AND INSTRUCTOR PILOT EJECTION SEATS IDENTICAL.
HEADREST
SEAT ARMING INITIATOR (LOCATED ON BULKHEAD AFT OF SEAT)
1
SEAT-TO-MAN SEPARATOR STRAPS
CHAFF DISPENSER AND LANYARD
SHOULDER HARNESS
2 GROUND SAFETY LOCK PIN (REMOVE BEFORE FLIGHT)
SEAT VERTICAL ADJUSTMENT SWITCH
GROUND MAINTENANCE SAFETY PINS FOR MAINTENANCE USE ONLY (REMOVE BEFORE FLIGHT)
CANOPY INITIATOR
SAFETY BELT
CANOPY JETTISON HANDLE
SHOULDER HARNESS INERTIA REEL HANDLE
SEAT-TO-MAN SEPARATOR/SAFETY BELT INITIATOR (M-12)
STABILIZATION PACKAGE
SEAT CATAPULT
1 LOCATED ON UPPER LEFT SIDE OF THE SEAT ON MOST AIRPLANES.
2 THE GROUND SAFETY LOCK PIN SHOULD BE INSERTED FROM THE OUTBOARD SIDE OF THE SEAT.
SEAT VERTICAL ADJUSTMENT ACTUATOR
SEAT-TO-MAN SEPARATOR/SAFETY BELT INITIATOR
21,831

Figure 1-42

EJECTION SEAT

The ejection seat (figure 1-42) permits bailout at high speeds and any flight attitude. The seat has a rigid seat-style survival and oxygen kit container, an automatic-opening safety belt, an ejection stabilization system, and accommodates a back-type parachute. No additional parachute support block is required as the survival kit container has its own support for the parachute.

CAUTION TF

Crew members should pay special attention to the lowering of their helmet visors before jettisoning the canopy on all airplanes equipped with the MK-1 rocket catapult. Live tests have shown that the remaining crew member is properly protected when wearing normal items of flight clothing, and will suffer no adverse effects when the adjacent seat is ejected.

WARNING

Do not use the A-5 cushion, or any similar sponge rubber cushion, with the 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. 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.

WARNING

Ground maintenance safety pins are inserted in the canopy initiator, the seat arming initiator and the seat-to-man separator/safety belt initiators 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.

EJECTION SEAT STABILIZATION SYSTEM

An ejection seat stabilization system is installed that is designed to stabilize the seat during ejection and to insure positive separation of the pilot from the ejection seat. The system is mechanical and operates automatically. The system is mounted on the bottom of the ejection seat with snubbing and stabilization lanyards attached to the airplane floor. During ejection, the lanyards begin to play out as the seat moves up the rails. During the first 16 feet of ascent, a slack line (which is an extension of the stabilization lanyard) plays out. At that point, the stabilization lanyard plays out for an additional 24 feet under controlled tension provided by a braking mechanism. This action stabilizes the seat and prevents tumbling. The snubbing lanyard continues to play out after the stabilization line is free of the seat. After lap belt release, the seat is snubbed at the end of an 80-foot lanyard providing positive separation to preclude seat/man entanglement. The seat may remain attached to the snubber line or, under certain conditions, it may separate after applying the required restraining force. The correct parachute configuration for optimum performance of this system is the BA-22 parachute equipped with a ballistic spreader gun and external pilot chutes.

SEAT-TO-MAN SEPARATOR

A seat-to-man separator (figure 1-42) is installed on the ejection seat to assure immediate pilot separation from the seat after ejection. A strap assembly is attached to the forward edge of the seat pan and extends aft under the survival kit, and up the forward side of the backrest to a gas operated rotary actuator located on a shelf back of the headrest. Upon ejection, gas pressure operates the rotary actuator to reel in the slack of the separator strap. As the separator strap is reeled in (approximately ¾ second) the pilot and seat are forcibly separated. The rotary actuator is operated by gas pressure from one of two initiators. One initiator is located on the left side of the seat and is armed by a lug on the seat rail as the seat moves up the rail. This initiator, which also operates the safety belt, incorporates a one-second delay. The other initiator, acting as a backup, is installed on the right side of the seat below the pan and also provides gas pressure for actuation of the rotary actuator and the safety belt. This initiator is fired by the first stretching action of the snubbing lanyard in the seat stabilization system.

CHAFF DISPENSER

On some airplanes, an automatic-opening chaff dispenser (figure 1-42) 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. Chaff dispenses to aid ground radar in locating the bailout area for rescue.

AUTOMATIC IFF ACTUATOR

The emergency mode of the identification radar (IFF) is automatically actuated when the airplane is airborne and the canopy is unlocked or jettisoned. When the airplane is on the ground, the IFF is deactivated by the main landing gear safety switch.

Note

On airplanes with AN/APX-6A, AN/APX-25 IFF, the IFF master control knob must be in NORM or LOW to automatically transmit the emergency return. On airplanes with AN/APX-72, emergency IFF may be activated automatically with switches in any powered position.

AUTOMATIC-OPENING SAFETY BELT

A HBU-2B/A automatic-opening safety belt (figure 1-43) 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 greater deceleration forces and the wind shock could open the parachute pack prematurely. The belt may be opened manually or by gas pressure from one of two automatically controlled initiators. One initiator is located on the left side of the seat and is armed by a lug on the seat rail as the seat moves up the rail. This initiator, which also operates the seat-to-man separator, incorporates a one-second delay. The other initiator is installed on the right side of the seat below the pan and also provides gas pressure for actuation of the safety belt release. This initiator is fired by the first stretching action of the snubbing lanyard in the seat stabilization system. This redundant system assures that the belt is opened prior to seat snubbing. The release incorporates a means of attaching a lanyard which connects the belt buckle to the static line to the timer of the automatic parachute. To latch the buckle, the parachute lanyard

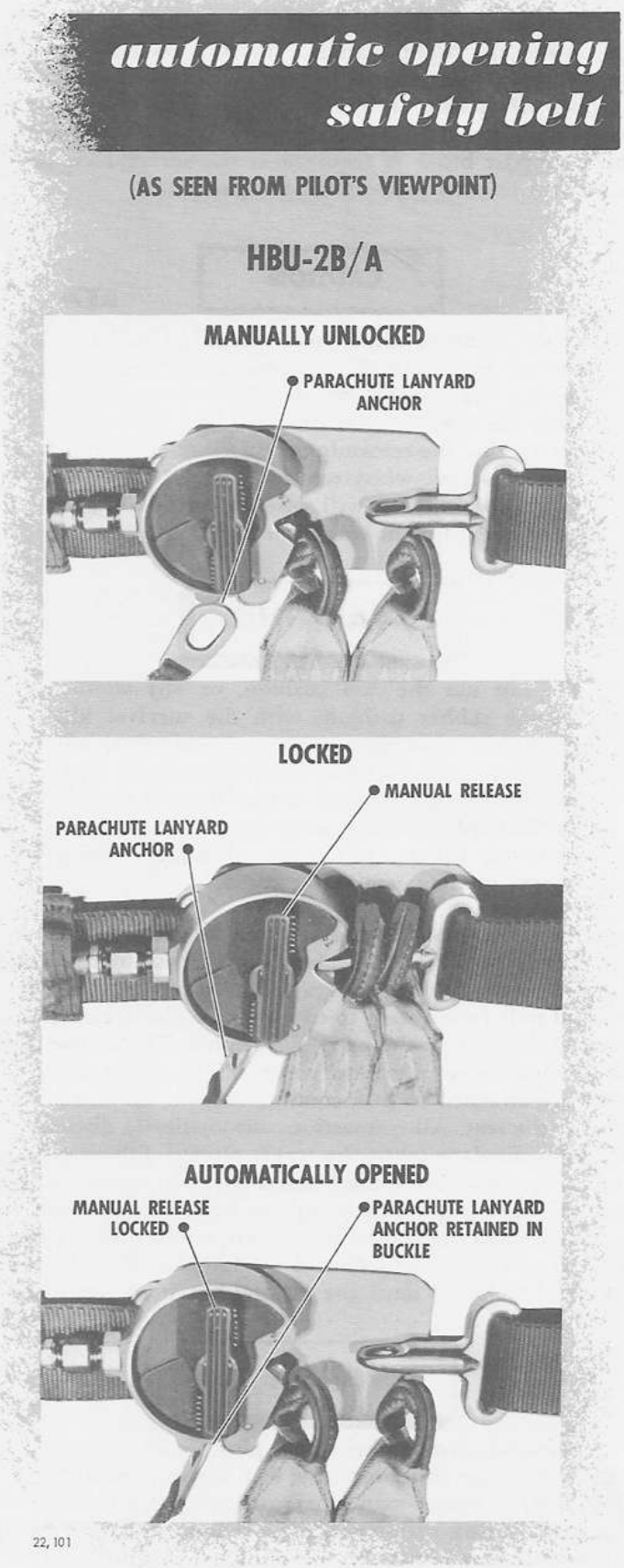


Figure 1-43

anchor (gold key) must first be inserted into the V-shaped receptacle in the top of the buckle. The link which retains the shoulder harness loops can then be latched into the buckle. Gas pressure actuation of the automatic release mechanism detaches the right strap link from the buckle assembly but retains the lanyard anchor. After the parachute release has been actuated by the automatic belt, a preset delay of 0.75 second 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 relay deployment until a preselected safe altitude is reached by free falling, then deployment begins after a 0.75-second delay. Automatic operation of the safety belt system 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.

EJECTION SEAT HANDGRIPS

Raising either seat handgrip (figure 1-42) 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; these gasses 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 1-42) 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

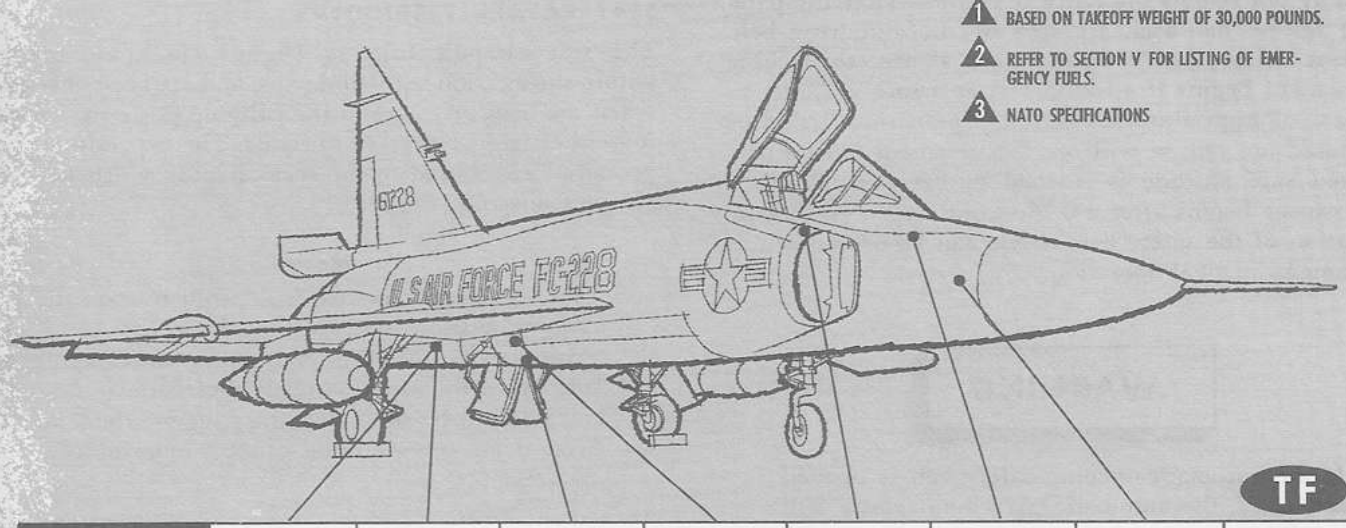
The seat ejection system is dependent upon the canopy jettison system. If raising the handgrips does not fire the canopy, the seat catapult initiator will not be armed; therefore, it cannot be fired by squeezing the triggers. Refer to Section III for alternate canopy removal procedures.

SEAT VERTICAL ADJUSTMENT SWITCH

The three-position seat vertical adjustment switch (figure 1-42), 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.

SHOULDER-HARNESS INERTIA REEL HANDLE

Manual control of the shoulder-harness inertia reel is provided by the shoulder-harness inertia reel handle (figure 1-42), 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, the harness retracts



NAME	Fuel Flask (Combustion Starter)	Constant Speed Drive Accumulator	Hydraulic System Reservoirs	Hydraulic System Accumulators (Filler Valves)	Survival Kit Emergency Oxygen Bottles	Infrared Receiver	Radome Anti-Ice	Canopy Counter Balance Cylinder
REPLENISHING AGENT	Engine Fuel	Dry Air Or Nitrogen	Red Hydraulic Fluid	Dry Nitrogen Or Clean Dry Air	Gaseous Oxygen	Dry Nitrogen	Water (40%) Glycol (60%) (By Volume)	Dry Air or Nitrogen
SPECIFICATION	Refer To Combustion Start Second Attempt, Section VII	Placarded 115 Psi (Std. Day)	Mil-H-5606 (3) H-515		Mil-O-27210 Grade A Type-I	Mil-P-27401	Mil-A-8243A	1500 ±25
CAPACITY			Fill To 1/4 Inch Below Full Mark	750 Psi At 21°C (70°F)	1800 Psi	Ref T.O. 1F-102A-2-12	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	(F) Forward Electronics Compartment Right Side; (TF) Right Side of Fuselage Aft of Canopy	Center of Bulkhead Aft of Seats

21,832-1

Figure 1-44

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

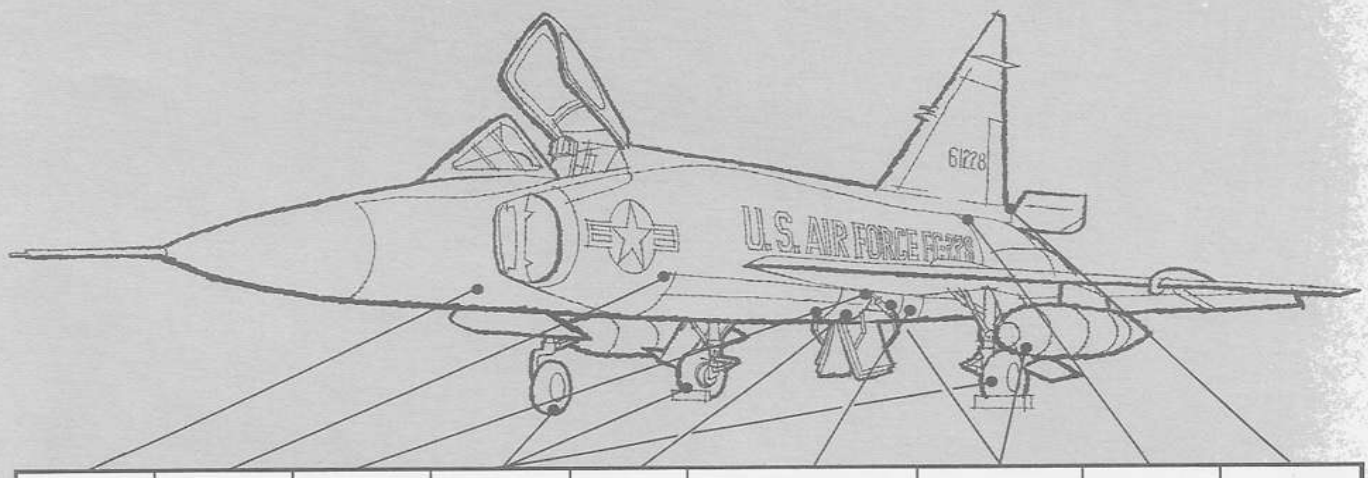
Note

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 (typical)



Battery	Oxygen System (Liquid)	High-Pressure Pneumatic System	Tires	External Power	Arresthook System Accumulator		Fuel Tanks	Engine Oil Tank	Drag Chute
Distilled Water	Oxygen	Pneumatic Pressure	Pneumatic Pressure	115/200V 400 Cps 3 Ph AC 28V DC	Hydraulic Fluid	Clean Dry Air Or Nitrogen	Fuel	Oil	Drag Chute Pack
	Mil-O-27210 Grade A Type II	Ma-1 Cart	Nose — 175 Psi Main — 195 Psi (1)		Mil-H-5606 (3) Mil-H-515		Grade JP 4 Mil-J-5624 (2) (3) F-40	Mil-L-7808 (3) Mil-L-2369	Pack In Accordance With T.O. 14D1-3-112
	Liters (F) 4-4½ (TF) 8-8½	3000 Psi				750 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	Left Engine Access Door		Main Wheel Well Aft Outboard (External) Left Side (Internal)	Upper Fuselage Left Side	Above Tail Cone

21,832-2



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SOUND JUDGMENT

This manual provides the best possible operating instructions under most circumstances, but is a poor substitute for sound judgment. Multiple emergencies, adverse weather, terrain, etc., may require modification of the procedures.

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 checklist items so as not to sacrifice clarity or standardization. External power is required to place the airplane in alert status and cocked for STARTING ENGINE (SCRAMBLE). The airplane when cocked is ready for STARTING ENGINE (SCRAMBLE) with either external power or battery. All other phases of the checklist 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

Information to complete Takeoff and Landing Data Cards in the Pilot's Abbreviated Checklist, T.O. 1F-102A-1CL-1, is contained 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

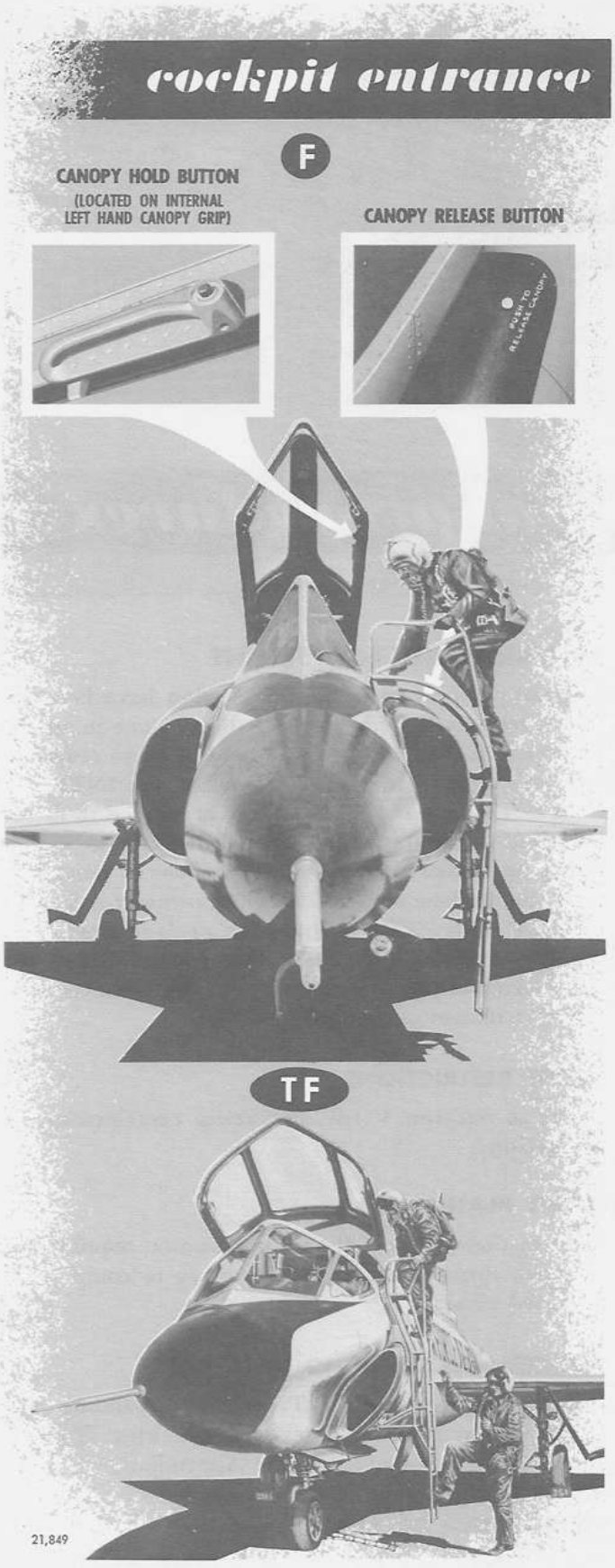


Figure 2-1

flight check the following:

a. Takeoff and anticipated landing gross weights.
b. Weight and balance clearance (Form 365F).

CHECKLISTS

The Flight Manual contains only amplified checklists; the abbreviated checklists have been issued as a separate technical order, T.O. 1F-102A-1CL-1 Refer to CHECKLISTS in the Introduction.

Note

The term "climatic," as used in the checklists, indicates equipment operation or settings which may be necessary for other than daylight VFR conditions. This includes IFR, night, cold weather, tropic, and desert conditions. The equipment operation or setting will vary depending on the prevailing conditions. In practice, the response to climatic items will be the required switch or control position.

PREFLIGHT CHECKS

It shall be the pilot's responsibility to accomplish interior and exterior visual inspection as outlined in the Flight Manual, and to ensure that the AFTO Form 781 indicates that the airplane preflight has been performed. Check that USAF Av Fuels Ident Plate is available.

Note

- The visual inspection procedures in this Section are predicated on the assumption that maintenance personnel have completed all the requirements of the Manual of Scheduled Inspection and Maintenance Requirements, T.O. 1F-102A-6. Therefore, duplicate inspections and operational checks of systems have been eliminated, except for certain items required in the interest of flying safety.
- Items coded (IP), or (P-IP), may be applicable to both F and TF airplanes but are coded to show the requirement for checking by instructor pilot or both pilots in TF airplanes.

ENTRANCE

On F airplanes, cockpit entry is accomplished from the left side of the airplane (figure 2-1) by use of a ladder. To open the canopy, remove external power, press the external canopy release button, manually raise the canopy to the full open position and install the canopy support tool.

WARNING

Do not attempt to open canopy externally with external power applied as severe personnel injury could result.

On TF airplanes, cockpit entry is accomplished from either side of the airplane (figure 2-1) by use of a ladder. Because of duct configuration, a separate ladder is required for each side. To open canopy refer to EXTERIOR CANOPY LATCH CONTROL TF, Section I.

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(s)—In place.
2. Ejection seat safety pin — Installed, streamer visible. (P-IP)
3. AFTO Form 781—Check.

   Check AFTO Form 781 for engineering status. Make certain the airplane has been serviced with fuel, hydraulic fluid, oxygen, air armament, nitrogen, and any special equipment required.
4. Canopy jettison handle—DETENT position release button OUT. (P-IP)
5. Canopy and safety belt ground safety pins—Check removed. (P-IP)

   If the M-12 safety belt initiator safety pin has not been removed, maintenance personnel should be consulted regarding the status of the ejection system before occupying the ejection seat.
6. Check ejection seat quick disconnect.
7. Ejection seat arming lanyard(s)—Coiled.

WARNING

If seat arming lanyard(s) is (are) not properly connected and coiled, jettisoning the canopy may not arm the ejection seat, preventing seat ejection.

TF 8. Canopy counterbalance pressure—Check. Check for 1500 (±25) psi.

9. Upper electronics compartment doors—Secured.
10. FLIP—Check.

    Check that flight information publications required for the mission are available.
11. 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.
12. Canopy knife—Installed. (P-IP)
13. Survival kit—Check. (P-IP)
    a. Emergency bailout bottle pressure — 1800 psi.
    b. Survival kit lid — Fully locked.
    c. Green knob assembly — Intact.
    d. Personal equipment bundle connection—Check.

Note

If the personal equipment leads become disconnected in flight, 100% oxygen from the converter will flow into the cockpit. Rapid loss of the normal oxygen supply will occur and emergency oxygen will not be available. UHF communications may also be lost.

    e. Automatic deployment selector lever — AUTO.
14. RAT handle—Fully in.
15. Throttle—OFF. (P-IP)
16. Igniter control switch—OFF (safetied if armament is aboard) (some airplanes); TRAINING (safetied if armament is aboard) (other airplanes).
17. Special weapon switch—DISARM (if installed).
18. Armed-safe switch—SAFE (safetied if armament is aboard).
19. Armament selector switch—SNAKE.
20. Battery switch—ON.

Note

Check all warning lights including engine fire warning light.

21. Battery switch—OFF.
22. Master switch—NORMAL (guard closed).
23. FCS master switch—OFF.

Note

Do not turn the FCS master switch from the OFF position until all phases of external ac power have been checked.

EXTERIOR INSPECTION

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

Note

While checking the items below, check for closed and secured doors and inspection plates, for fluid leaks, indications of defective or insecure installations, and general overall appearance of the airplane. An engine access door may be left open until after engine start for fire extinguisher access and leak inspection.

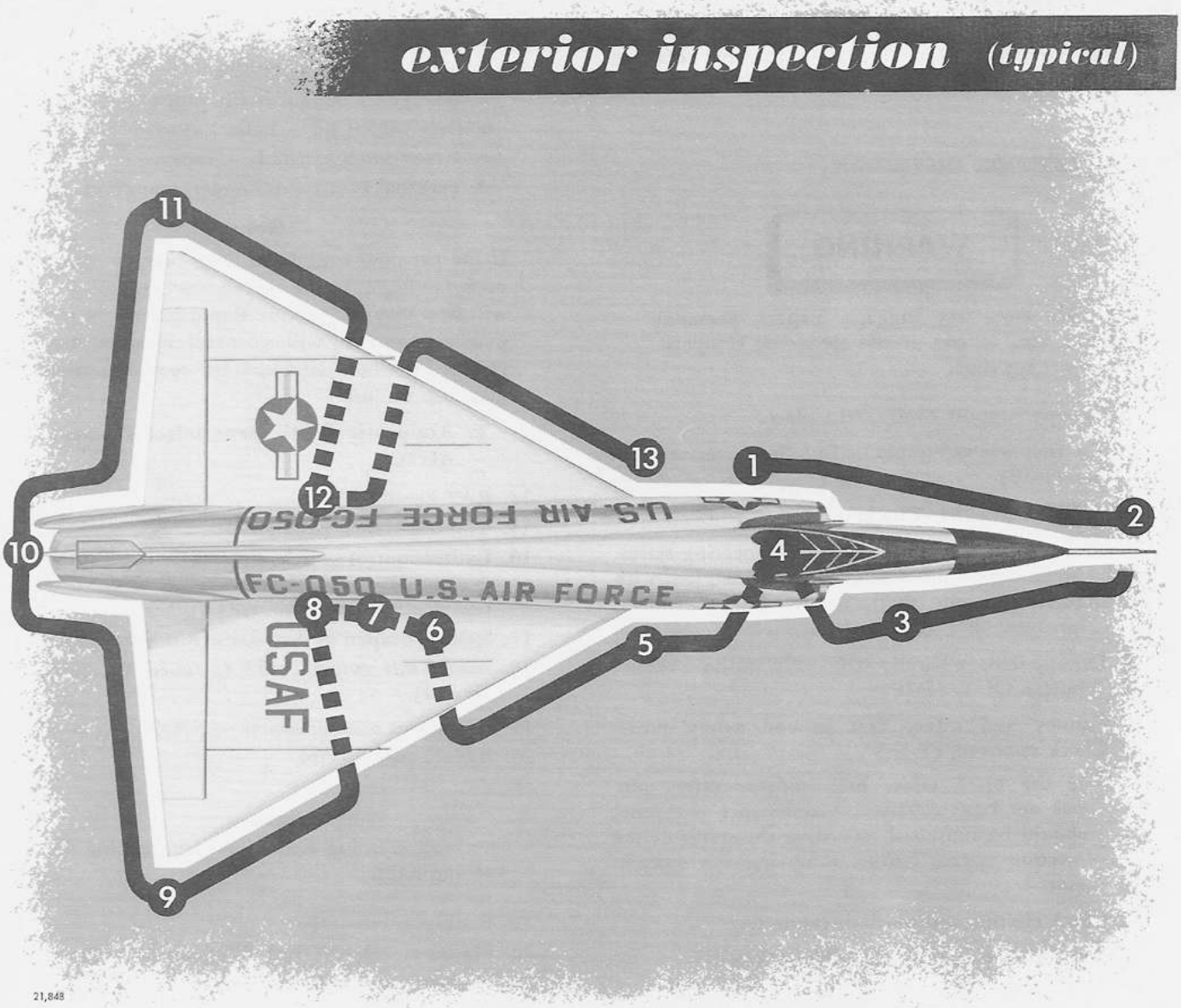


Figure 2-2

1. Forward left side—Check.

   a. Armament placard—Check for proper notation.

   b. Missile bay—Check for armament (if required). On all missions, the airplane will be considered "hot" until a visual inspection of the missile bays is accomplished. A visual inspection of the missile bays will be accomplished immediately prior to any flight which will involve missile bay door actuation, lowering of rails, WSEMS, or when special weapons are loaded. This inspection will include a check for the type and position of the armament aboard.

WARNING

- When the airplane is considered or verified as being "hot," the igniter and arm-safe switches will remain safety-wired. If the safety-wire is discovered broken, no flight will be made and a proper entry will be made in AFTO Form 781.
- If an inspection is accomplished without door locks installed, it will be from a position outside the missile bay to prevent possible injury if doors inadvertently close. Do not lower launchers for this inspection.

If an armament check is required, check the following:

(1) With armament doors open (launchers retracted), visually check missiles for cracks, dents, bent fins, and security of mounting. Check missile nose section for visible damage to guidance unit.

(2) Igniter disconnect — Connected and undamaged.

c. Intake and boundary layer ducts—Clear.

d. Forward electronic bay doors—Secure.

e. Rocket jump angle vane—Condition.

2. Nose area — Check.
   - (F) a. Static ports — Clear.
   - b. Radome — Condition & security.
   - c. Mast and pitot tube — Condition & Cover removed.
3. Forward right side — Check.
   - a. Forward electronic bay doors — Secure.
   - b. Boundary layer and intake duct — Clear.
4. Nose wheel well area — Check.
   - a. Tire — Check wear, cuts, and inflation.
   - b. Gear strut — Check five to six inches between torque link bolts, center to center.
   - c. Nose wheel steering scissors hinge pin — absence of lateral movement.
   - d. Left-hand aft circuit breakers — In.
   - e. Left-hand dust cover — Secure.
   - f. Nose gear door lock — Remove (if installed).
   - g. Radar ground cooling door — Secure.
   - h. Battery — Secure.
   - (F) i. Right-hand forward circuit breakers — In.
   - j. Right-hand dust cover — Secure.
   - k. Taxi light — Condition and security.
   - l. Nose door seal — Condition.
   - m. Nose landing gear safety lock pin — Removed.
5. Right side — Check.
   - (TF) a. Static port — Clear.
   - b. Intermediate electronic bay door — Secure (punch marks aligned with stripe).
   - c. Armament bay doors — Secured.
   - d. Position lights — Condition & security of glass.
   - e. Upper electronic compartment door — Secure.
6. Ram air turbine compartment — Check.

WARNING

Check the RAT door for free play to insure that pressure is shut off from the actuator.

   - a. Primary & secondary hydraulic fluid levels and accumulator pressures — Check.
     Check the hydraulic reservoir sight gages for correct fluid level and the primary and secondary accumulator air gages at 750 (±25) psi.
   - b. Caps on reservoirs, safety pins in and clipped (if applicable).
   - c. Quick disconnect line—Connected.
   - d. RAT rotor—Turns freely & condition of blades.
   - e. RAT door—Close.
     Close the RAT door by a gradual increase in force until click is heard (40-60 lbs).
   - f. RAT door test hook—Pull (approximately 80 lbs).
     Pull the RAT door test hook to check the RAT door for security.
   - g. RAT door test hook—Remove (if applicable).
7. Right main wheel well area — Check.
   - a. Circuit breakers — In.

WARNING

AN/ALQ-31 circuit breakers must be pulled out unless a chaff tank is installed on the associated pylons.

   - b. Emergency ac generator — Check.
   - c. Manually controlled (armament) pneumatic shutoff valve—Check open.
     If valve is not open, have system checked and restored to an operational configuration by qualified personnel.
   - d. Deleted.
   - e. Chocks—In place.
   - f. Tire—Check wear, cuts & inflation.
   - g. Brake & hydraulic lines—Check.
   - h. Gear strut extension—Check five-six inches extension between torque link bolts, center to center.
   - i. Landing gear fairing & door—Condition & security.
   - j. Landing light—Condition & security.
   - k. Gear ground safety lock pin—Removed.
   - l. Combustion starter exhaust port—Clear.
   - m. Oil overboard vent—Clear.
8. Engine access compartment — Check.
   - a. Constant-speed drive accumulator pressure — Check.
     The CSD accumulator pressure gage is placarded to give information for variations in ambient temperatures versus accumulator air pressure. If the gage has a numerical type dial, the gage should read 115 psi under standard day conditions (15°C/59°F). If the gage has a dial with pressure ranges indicated by color areas or bands, the gage should read within the green band under standard day conditions (15°C/59°F).
   - b. Hydraulic & fuel leaks — Check.
   - c. Elevator feel-force cable & quadrant — Check visually.
   - d. Elevator intelligence unit — Pin flush with case.

Note

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

   - e. Arrest hook system pressure gage — 750 psi (if installed).
   - f. Starter and duct installation—Free of contamination.

9. Right wing area — Check.
   a. Fuel drains—Unobstructed.
   b. External tank — Check (if installed).
      Visually check fuel level. Cap secured.
   c. External tank safety pin—Installed from inboard side.
   d. Wing leading edge & tip—Condition.
   e. Position lights—Condition & security of glass.
   f. Trailing edge & elevon—Condition.
   g. Hydraulic installations—Check for leakage.

WARNING

Any evidence of a hydraulic leak at the elevon actuator housing must be entered in AFTO Form 781 and corrected or certified within limits prior to flight.

10. Tail section — Check.
    a. Speed brakes — Condition.
    b. Drag chute—Stowage & pin (chute upright with two T's at top and red lower strap installed last).
    c. Rudder — Condition.
    d. Position lights — Condition & security of glass.
    e. Tailpipe & exhaust nozzles — Check.

CAUTION

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

    f. Arrest hook — Check, ground safety pin removed, condition and security of protective fairing.
       Check that the arrest hook is latched. Check hook point and arm attachments for condition, then remove the ground safety pin.
    g. Data link antenna—Check for condition and security (if installed).
11. Left wing area — Check.
    a. Trailing edge & elevon — Condition.
    b. Hydraulic installations—Check for leakage.

WARNING

Any evidence of a hydraulic leak at the elevon actuator housing must be entered in AFTO Form 781 and corrected or certified within limits prior to flight.

    c. External tank safety pin—Installed from inboard side.
    d. Position lights—Condition & security.
    e. Wing tip & leading edge—Condition.
    f. External tank—Check (if installed).
       Visually check fuel level. Cap secured.
    g. Fuel drains—Unobstructed.
12. Left main wheel well area — Check.
    a. Chocks — In place.
    b. Landing gear fairing & door — Condition & security.
    c. Landing light — Condition & security.
    d. Tire—Check wear, cuts & inflation.
    e. Brakes & hydraulic lines — Check.
    f. Gear strut extension — Check five-six inches extension between torque link bolts, center to center.
    g. Ground safety switch — Check.
    h. Armament control panel — Check.
    i. Circuit breakers — In.
    j. Fuel filler cap — Secure.
    k. Engine ignition disconnect switch — ON.
    l. Combustion starter manual air valve — GROUND (if external air source is available).
    m. High-pressure pneumatic system pressure — 3000 psi.
    n. Engine hot-section analyzer data recorder — Check reset to preflight condition (if installed).
    o. Gear ground safety lock pin — Removed.
    p. Overrun barrier probe & safety lock pin — Physically check in fully up and locked position. Remove lock pin.

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.

13. Left side — Check.
    a. Aft electronic bay door — Secure.
    b. Armament bay doors — Secure.
    c. Emergency canopy jettison lanyard — Stowed & secure.
    d. Position lights — Condition & security of glass.
    e. Ram air intakes — Covers removed & condition.
    f. Engine oil cap access door — Secure.
    g. Oxygen filler access door — Secured.
    TF h. Static port — Clear.
    TF i. Canopy external actuator door — Closed and secure.

INTERIOR INSPECTION (ALL FLIGHTS)

Note

Items marked with the symbol ▲ preceding the step shall not be performed if making the interior check without external power prior to battery start. These items (▲) should be checked after battery start. (Refer to INTERIOR CHECK AFTER BATTERY START, this Section.) The 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. (P-IP)

   Attach survival kit parachute attaching straps to parachute harness. Pull straps snug. Pull fastener end of straps; if slippage occurs, the attachment straps are installed backwards.

2. Spreader gun lanyard—Attached to right parachute harness above quick disconnect. (P-IP)
3. Safety belt & shoulder harness—Secure. (P-IP)

   Attach as follows:

   a. Parachute lanyard anchor—Insert into safety belt receptacle.

Note

Lanyard anchor must be attached first to assure proper connection.

   b. Shoulder harness straps—Place loops over safety belt link.
   c. Safety belt link—Insert into safety belt buckle and secure.
   d. Parachute lanyard anchor—Pull to ensure anchor is secure in receptacle.

WARNING

- Do not route the anti-g suit hose under the lap belt or in any manner which would interfere with emergency disconnecting of the hose if required. In case of a malfunctioning anti-g valve (where the anti-g suit fails to deflate after a maneuver), immediately disconnect the anti-g suit hose from the pressurization system by pulling on the hose.
- When fastening the left parachute attaching strap to the parachute, ascertain that the strap is not placed between the safety belt and lap belt initiator hose, as this will tie the seat to the pilot in case of ejection.
- Check that the flight suit left leg bottom zipper is zipped closed to preclude possible involvement with the canopy jettison handle.

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.

4. Personal equipment—Check. (P-IP)
   a. Personal equipment leads — Connect.
   b. Personal equipment lead bundle — Connected to parachute harness.
   c. Oxygen hose and communications cord—Route under right shoulder harness.
5. Throttle quadrant cover—Remove.
6. External tanks jettison button cover—Remove.
7. Circuit breakers—In.
8. Booster pumps—OFF.
9. AC and dc generator switches—OFF.
10. AC bus switch—NOR; guard closed.

▲11. External power—Connected (if available).

When external power is connected on some F* and TF** 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.

▲12. AC voltage—Check.

To determine that all three ac phases of the external power source are available for booster pump operation, place the voltmeter selector switch to ∅A, ∅B, and ∅C and note 115 (±10) volts. ∅AB and ∅AC should give readings in the 115 (±15) volt range, and the 26V position should read more than "0" but not more than 50 volts.

CAUTION

Do not operate the fuel booster pumps or radar if the voltmeter readings are not as specified.

13. Battery switch—ON (if no external power source is available).

▲14. Seat & rudder pedals—Adjust. (P-IP)

15. Wheel brakes—Check (crew chief visually checks pucks for movement).
16. Pitot heat switch—Check, then OFF.
    a. Pitot heat switch—ON (for cocking).
       Have crew chief check pitot tube for heating.

▲17. Landing & taxi lights—Check.

a. Taxi light—On (for cocking).

**AF 56-1275 thru -1316, -1332 & on.*
***AF 55-4057 & on.*

Left-Hand Console

1. LH aft circuit breakers — In.
2. IR cooling switch*—As required.
   a. IR cooling switch—ON for cocking.
3. Mask defog—Set.
   If partial pressure suit is worn, set mask defog as desired.
4. Fuel shutoff valve switches—OPEN.

WARNING

If the safety wire is broken, it must be reestablished that both valves are open prior to flight and before safety wire is replaced.

▲ 5. Fuel system—Check.
   a. Fuel booster pump pressure-low warning lights — On.
   b. Fuel booster pumps — Check.
      (1) Right-hand forward booster pump switch ON and check that right-hand booster pump pressure-low warning light extinguishes within five seconds, then place switch OFF. Repeat for the right-hand aft and left-hand forward booster pumps.

CAUTION

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

      (2) Left-hand aft booster pump switch ON and check that left-hand booster pump pressure-low warning light extinguishes within five seconds. If the light extinguishes within five seconds, leave the switch ON for a minimum of 30 seconds to insure the starter flask is full.

Note

The left-hand aft booster pump is used instead of the forward pump because of accessibility for maintenance.

▲ 6. Fuel quantity—Check.
   Select each position of the fuel quantity selector switch and check fuel quantity gage for correct readings.
7. Left-hand aft booster pump switch—OFF.
8. UHF radio—BOTH.

Note

ADF position of the UHF selector eliminates the guard receiver.

9. Speed brakes switch—NEUTRAL.
   a. Speed brakes switch — In (for cocking).

Note

The speed brakes may be closed on a cocked airplane to protect the drag chute from weather.

10. Fuel control switch—NORMAL (guard closed).
11. Oxygen system—Check.
   Check oxygen system as follows:
   a. Attach oxygen hose (see figure 4-16).
   b. Oxygen pressure—70 to 110 psi.
   (F) (1) Liquid content gage—Three liters minimum (with partial pressure suit), two liters minimum (without pressure suit).
   ▲ (TF) (2) Liquid content gage—Six liters minimum (with partial pressure suit), four liters minimum (without pressure suit, for crew of two). (AC power required for this check.)
   (TF) (3) Oxygen gage test button—TEST.
   c. Supply switch—ON.
   d. Breathe normally into mask; there should be mild resistance to exhalation.
   e. Supply switch—OFF.
12. Igniter control switch—OFF (safetied if armament is aboard) (some airplanes); TRAINING (safetied if armament is aboard) (other airplanes).
13. Special weapon switch—DISARM (if installed).
14. Armed-safe switch—SAFE (safetied if armament is aboard).
15. Armament selector switch—SNAKE.

**Airplanes modified by TCTO 1F-102-1170*

Left-Hand Auxiliary Panel

1. Landing gear handle and warning lights — DOWN and check.

   Check that both the landing gear unsafe bar warning light and control handle light are out.

2. Landing gear position indicators — Green lights.
3. LH forward circuit breakers—In.
4. Cabin air switch—PRESS.
5. Landing gear emergency extension handle—In & secure.
6. Arrest hook release handle—In.
7. Arrest hook warning light—Press-to-test.
8. Cabin pressure altitude gage—Check.
9. Drag chute handle—In & secure.
10. External tank fuel transfer switch—OFF.
11. LH & RH radar scope controls—Fully counterclockwise.

Instrument Panel

1. Clock — Set.

▲ 2. RMI (stabilizing).

▲ 3. Attitude indicators (primary & standby)—Check.

   a. Attitude warning flags for retraction within 2 minutes.

   b. Horizon bars for proper attitude and freedom from oscillation (energized).

   c. Horizon bars for response to trim knobs and set 5° nose low.

WARNING

If the attitude warning (OFF) flags require longer than 2 minutes to retract, if the horizon bars do not respond to the trim knobs, or if any oscillations are noted on the indicators after the attitude warning (OFF) flags retract, possibility of a malfunction exists. Either of the above is cause for rejection of the indicators and should be noted on AFTO Form 781.

4. Vertical velocity indicator, airspeed-mach indicator, and turn-and-slip indicator indicating proper static conditions.

▲ 5. Altimeter—Check; RESET and check disappearance of OFF flag (if AAU-19/A is installed).

   Set current altimeter setting and check altimeter for field elevation ± 75 feet. If AAU-19/A altimeter is installed, perform the error check in STBY then place selector to RESET and repeat; check for maximum of 75 feet difference between STBY and RESET modes.

Note

The computer operated RESET mode of the AAU-19/A altimeter requires electrical power for these checks. On airplanes having a conventional barometric altimeter, the instrument error check does not require electrical power.

6. Deleted.

7. Engine fire warning test switch—FIRE, then OVHT. (P-IP)

   Place engine fire test switch to FIRE and check fire warning light on steadily. Place switch to OVHT and check for flashing light.

▲ 8. Engine instruments—Check.

   Check tachometer, exhaust gas temperature gage, fuel flow indicator, pressure ratio gage, and oil pressure gage.

9. EGT power flag (airplanes with hot-section analyzer)—Not displayed.

10. Hydraulic pressure-low warning light—On.

Right-Hand Auxiliary Panel (F), Instrument Panel (Cont) (TF)

(TF) 1. Magnetic compass light — Check (if required).

▲ (TF) 2. Thunderstorm lights—Checked.

   a. Thunderstorm lights — On (for cocking).

Note

Thunderstorm lights provide added safety during alert scrambles. This insures full illumination of master warning panel lights. Switch may be turned OFF prior to takeoff.

3. Canopy unlocked warning light — On.

4. 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. The gear warning light should illuminate and the audible gear warning should be heard in the headset.

5. Oil pressure, engine fuel pump, and electronic cooling lights—Illuminated.

RIGHT-HAND CONSOLE (F), CENTER CONSOLE (TF)

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

   a. DC generator switch—ON (for cocking).

4. FCS master and mode selector switches—OFF.
   a. FCS master switch—WARM (for cocking).
5. TACAN — OFF.
   a. TACAN—T/R (for cocking).
6. ILS—OFF.
   a. ILS—POWER (for cocking).
7. TACAN/ILS function switch—As required.
▲ 8. J-4 compass system—Check.
   a. Compass fast slave—Check.
      (1) With the two-position function selector switch positioned to MAG, observe compass heading then position function selector to DG.
      (2) Position synchronizer SET switch to either DECR or INCR until compass card moves off heading 10° to 40° and release SET switch.
      (3) Position function selector switch to MAG and ensure that compass card fast slaves to within ±2° of correct heading. Check against known heading or magnetic compass.
9. IFF—Check and set.

TF 10. Interphone panel—Set. (P-IP)

F 11. Magnetic compass light—Check (if required).
   a. Magnetic compass light—On (for cocking).

F 12. RH forward & aft circuit breakers—In.

Utility Switch Panel

▲ F 1. Thunderstorm lights—Checked.
   a. Thunderstorm lights—On for cocking.

Note

Thunderstorm lights provide added safety during alert scrambles. This insures full illumination of master warning panel lights. Switch may be turned OFF prior to takeoff.

2. Nesa switch(es)—OFF.
   a. Nesa — NORMAL (for cocking).
3. Cabin temperature control knob—AUTOMATIC.
4. Canopy defog switch—Climatic.
5. Windshield rain clear switch—OFF.
6. Cockpit no-fog/vent suit switch—As required.
   a. Cockpit no-fog vent suit switch — ON (for cocking).
7. Anti-ice switch — OFF.
   a. Anti-ice switch — AUTO (for cocking).

CAUTION

If start or takeoff is delayed after the airplane is set up for scramble, the nesa, anti-ice, and pitot heat switches should be turned off to prevent damage to electrical circuits. These switches must then be turned on prior to takeoff.

Lighting Control Panel

▲ 1. Position lights — NAVIGATION.

Test position light switches for FORMATION and NAVIGATION.

▲ 2. Cockpit lights — Check (if required). (P-IP)
Check operation of cockpit lights and set rheostats as desired (OFF or ON).
   a. Cockpit lights—ON (for cocking).

Right-Hand Auxiliary Panel TF

1. Drag chute handle — IN & secure. (IP)
2. Landing gear handle — DOWN. (IP)

Right-Hand Console TF

1. Oxygen system—Check as outlined in this section. (IP)
2. Throttle — OFF. (IP)
3. Speed brakes switch — NEUTRAL. (IP)
4. Fuel control switch — NORMAL, guard closed. (IP)
5. Mask defog control knob — Set. (IP)
6. Anti-g suit valve — Set. (IP)
7. RH aft circuit breakers — In. (IP)

BEFORE STARTING ENGINE

An outside air source should be used (if available) for starting to conserve air supply.

Note

- If it is necessary to start and no external ac power source is available, a start may be accomplished by turning the booster pumps off and following the normal start procedure. There will be no indication of fuel flow, hydraulic or oil pressure, or fuel quantity. The oil pressure warning light will be operative.
- Remove canopy support tool prior to start.

CAUTION

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 and to preclude interference with airplane flight control operation.

STARTING ENGINE

Note

- 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, 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 fill if external power is applied and a left-hand fuel booster pump is turned on. However, if no external power is available, refer to START—SECOND ATTEMPT, Section VII.

Note

If practical, 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.

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.

Note

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

2. Danger areas — Clear.
   See figure 2-3.

WARNING

- 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.
- 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.

3. Throttle—START.
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 has stopped and is visually checked for trapped fuel.

5. RPM—Positive rpm or hydraulic pressure indication.

WARNING

If an rpm indication is not evident, do not move the throttle inboard to the OFF position with the ignition button depressed. This could result in disintegration of the combustion starter. If no rpm indication is evident, the starter has failed to engage the engine.

6. Throttle—OFF, then IDLE.
   As soon as a definite rpm indication is noted, 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

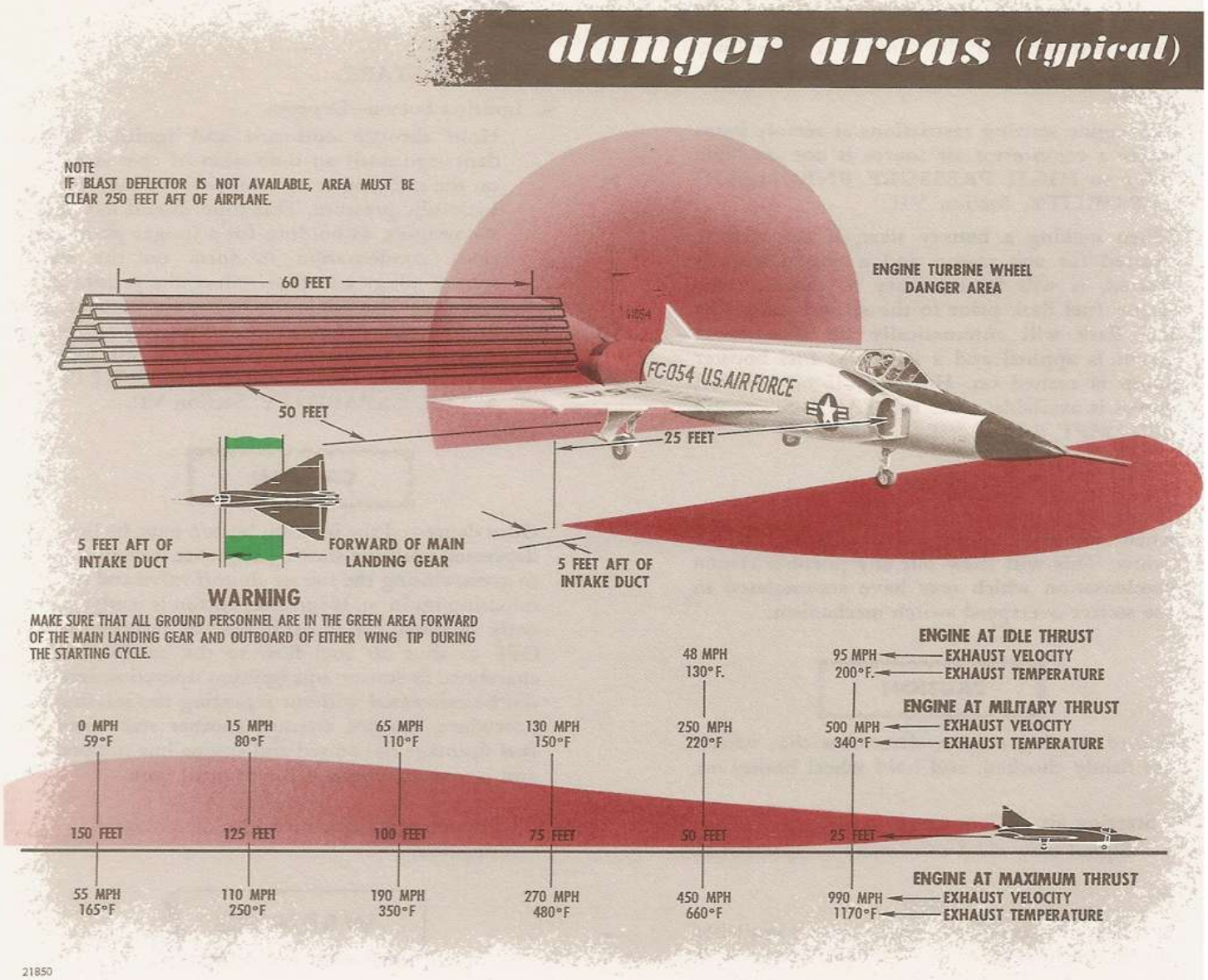


Figure 2-3

noted, the start should be aborted immediately in accordance with the unsuccessful start 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 ground 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 hot start can be anticipated by observing a rapid increase to 450°C and temperature is still rising. Discontinue the start and record on AFTO 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.

Note

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

13. Oil pressure gage—Check, 40 to 55 psi normal.
14. Engine fuel pump failure warning light—OFF.

CAUTION

- If the engine fuel pump failure light illuminates, 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.
- 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.

15. Electronic cooling warning light—Off.
16. Rain clearing system—Check.
    a. Rain clear switch—ON.
    b. Airflow—Check.
    c. It may be necessary to slightly increase rpm to detect airflow.

Note

Do not increase rpm above 75%.

    d. Rain clear switch—STBY (F), OFF (TF).
    e. Check airflow stopped.
    (F) f. Rain clear switch—OFF.
    g. Throttle—IDLE.

Note

Some (TF) airplanes have a 3-position switch labeled LEFT, OFF, and RIGHT. Both left and right functions should be checked.

WARNING

If rain clear does not function or requires more than 75% rpm prior to functioning, abort. Rain clear malfunction may indicate hot air bleed line/duct failure.

17. Compressed air and external power—Disconnected.

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 slowly 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.

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

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.

CAUTION

- Check for zero rpm before re-starting, to avoid shearing starter shaft.
- The following starter cooling periods must be observed to prevent damage to the starter from overheating:

  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.

  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.

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

Note

A start is defined as any time the starter ignites and fires.

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

- Parking brakes are not provided. If an engine runup is made for ground tests, make certain that wheel chocks are in place and hold wheel brakes on.
- 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, do not reopen the throttle.

BEFORE TAXIING

ELECTRICAL POWER SUPPLY SYSTEM

1. Battery switch — ON (check output).

   After external power has been disconnected, check that dc operated equipment or lights are operating and that the dc power failure warning light is illuminated.

2. Standby attitude indicator—Check.

   Check for proper operation and adjustment.

3. AC bus switch—ON (check "0" volts).

   With the ac bus switch in the ON position (do not hit the RESET position), check for "0" volts and that the ac power failure warning light is illuminated. If ac power is available, a malfunction is indicated.

4. AC bus switch—RESET, then ON (check 103 to 140 volts).

   Check operation of the emergency ac generator. If the transformer-rectifier is operating, all positions of the ac voltmeter selector switch should give voltmeter readings of 103 to 140 volts except that the 26v position should read above "0" but not over 50 volts. On TF airplanes, 103 volts may not be reached if NESA is on. If the transformer-rectifier is not operating, the ON position will give a "0" voltmeter reading.

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.

5. Attitude indicator (MM-3)—Check.

   Check for proper operation and adjustment. Check that MM-3 and standby attitude indicators show similar indications.

6. Radio magnetic indicator—Check.

   Check for proper heading.

7. AC bus switch—NOR (check "0" volts).

8. Battery switch—T/R FAIL.

   The T/R FAIL position of the battery switch reconnects the emergency dc bus to the dc essential bus.

9. AC generator switch—RESET, then ON (check voltage).

   After the ac generator switch has been placed in the RESET and ON positions, ∅A, ∅B, and ∅C positions should give voltmeter readings of

115 (±10) volts. ∅AB and ∅AC positions should give readings in the 115 (±15) volt range, and the 26V position should read more than "0" but not more than 50 volts.

Note

AC power control and reset are powered by the emergency dc bus. If the T/R 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.

10. DC generator switch—RESET, then ON (check output).
11. FCS master switch—STBY or as required.
12. IR cooling switch—As required.
13. TACAN—T/R.
14. ILS—POWER.
▲15. Fuel booster pump switches—ON (if external power was used for start).

    Fuel booster 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, booster pumps should not be turned ON until INTERIOR CHECK AFTER BATTERY START.

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 booster pump pressure-low warning lights—On.
3. Fuel booster pumps—Check.
   a. Left-hand forward booster pump switch ON and check that left-hand booster pressure-low warning light extinguishes within five seconds. Then place the booster pump switch OFF.

CAUTION

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

   b. Repeat step "a" for the left-hand aft, right-hand forward and right-hand aft booster pumps.
4. Booster pump switches—ON.

   The booster pump switches should be turned on one at a time allowing approximately five seconds for each pump to start.

5. Landing & taxi lights—Checked.

5A. AAU-19/A altimeter—RESET.

6. RMI—Check (stabilizing). (P-IP)
7. Attitude indicators—Check. (P-IP)
8. Engine instruments—Check. (P-IP)

   Check the tachometer, exhaust gas temperature gage, fuel flow indicator, pressure ratio gage, and oil pressure gage for normal operating limits.

9. Rain clearing system—Check.
10. Thunderstorm lights—Checked.
11. J-4 compass system—Check.
    a. Compass fast slave—Check.
       (1) With the two-position function selector switch positioned to MAG, observe compass heading then position function selector to DG.
       (2) Position synchronizer SET switch to either DECR or INCR until compass card moves off heading 10° to 40° and release SET switch.
       (3) Position function selector switch to MAG and ensure that compass card fast slaves to within ±2° of correct heading. Check against known heading or magnetic compass.
12. Deleted.
13. Cockpit lights—Check (if required).

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

14. Seat—Adjust. (P-IP)

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 NEUTRAL.
   Retract speed brakes and check with crew chief for proper speed brake operation. Secondary hydraulic system pressure may drop momentarily.
3. Hydraulic pressure-low warning light — TEST.
   Check for blinking light.
4. Yaw, then pitch dampers—Engage.
5. Flight control surface movement — Check.
   Check rudder surface movement. During this check bottom a rudder and then insure that the rudder pedals are easily returned to a neutral position. Then, with the stick in the fully aft position, check elevon surfaces for proper movement (aileron as well as elevator position).

Note

- Prior to performing the above check, insure that the seat is in the full down position to prevent the life raft deflation device from striking the survival kit sleeping bag. After completion of the check, readjust the seat and rudder pedals.
- Prior to performing the above check insure that nose wheel steering is disengaged.
- During the above check there may be jerky rudder pedal motions caused by the turn coordinator. This is a normal condition.

6. 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. When system pressure drops, the hydraulic pressure must return to system pressure within two seconds, or if it overshoots on recovery, it must return to normal within five seconds after control stick movement has stopped. Then place the hydraulic pressure selector switch in the SEC position and repeat the above procedures for .. check on the secondary hydraulic system.
7. AFCS—Check.

Note

This check is not required for scramble.

a. Artificial horizon—Present.
   Check for appearance of artificial horizon on scope and that the vertical gyro erects in about 60 seconds.
b. Yaw, then pitch damper systems—Engage.
c. AFCS switch—AFCS; AFCS should engage.
   Check that no objectionable stick movement occurs when the switch is placed to AFCS.
d. Pitch G limiter test switch—Check.
   Check +G and −G functions.
e. AFCS switch—AFCS.

8. Emergency damper disconnect button—Depress.
   Check that pitch and yaw dampers and AFCS disengage.
9. Trim—Check and set for takeoff trim.
   Check elevon and rudder trim in all trim parameters. Trim nose up, left wing down, and left rudder; then depress takeoff trim button. Control surfaces should return to takeoff trim position. Takeoff trim light should illuminate while the takeoff trim button is depressed.

GENERAL

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

1. Canopy—Close and lock, check light not illuminated.
   Check the canopy warning light out indicating the latches are fully engaged. Visually check the canopy latches after the warning light goes out. The canopy may be reopened for taxiing in hot weather after cinch down and locks have been checked.

WARNING

The final movement (1 to 2 inches) of closing the canopy is rapid and it is cinched down with considerable force; therefore, care should be taken that the area beneath the canopy is clear to prevent personal injury or damage to equipment.

CAUTION

- To ensure positive canopy locking (TF), push the canopy lock handle with a firm, steady motion and hold in for a minimum of three seconds. This procedure will eliminate springback of the teleflex cable. If the canopy is open for taxiing, taxi lock must be engaged.
- If the canopy is open during taxi operations, observe canopy limit speed.

(TF) 2. Canopy latch overcenter pointers—Check needle below middle mark for over-center. (P-IP)

3. Ejection seat safety pin—Remove. (P-IP)
   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.
4. FCS master switch—STBY or as required.
5. Engine pressure ratio gage—Set.
   Check outside air temperature and set engine pressure ratio gage to indicate takeoff limits.
6. Chocks—Removed.
7. External tank safety pins—Removed.

TAKEOFF CHECK TABLE

TEMP °F	TEMP °C	PRESSURE RATIO SETTING (F)	PRESSURE RATIO SETTING (TF)
−40.0	−40.0	2.63	2.65
−36.0	−37.5	2.62	2.64
−31.0	−35.0	2.61	2.62
−27.0	−32.5	2.59	2.61
−22.0	−30.0	2.57	2.59
−18.0	−27.5	2.55	2.57
−13.0	−25.0	2.52	2.55
− 9.0	−22.5	2.49	2.53
− 4.0	−20.0	2.47	2.51
0	−17.5	2.44	2.49
5	−15.0	2.42	2.47
9	−12.5	2.40	2.45
14	−10.0	2.38	2.43
18	− 7.5	2.36	2.41
23	− 5.0	2.34	2.39
27	− 2.5	2.32	2.36
32	0	2.30	2.34
37	2.5	2.28	2.32
41	5.0	2.26	2.30
45	7.5	2.24	2.28
50	10.0	2.22	2.26
54	12.5	2.20	2.24
59	15.0	2.18	2.22
63	17.5	2.16	2.20
68	20.0	2.15	2.18
72	22.5	2.13	2.16
77	25.0	2.11	2.15
81	27.5	2.09	2.13
86	30.0	2.07	2.11
90	32.5	2.05	2.09
95	35.0	2.04	2.07
99	37.5	2.02	2.06
104	40.0	2.00	2.04
108	42.5	1.98	2.02
113	45.0	1.96	2.00

STARTING ENGINE (SCRAMBLE)

1. All personal equipment—Attach. (P-IP)
2. Canopy support tool(s)—Remove.
3. Normal start.

BEFORE TAXIING (SCRAMBLE)

ELECTRICAL POWER SUPPLY SYSTEM (SCRAMBLE)

1. Battery switch—TR FAIL.
2. AC generator switch—RESET, then ON.
3. FCS master switch—STBY or as required.
4. IR cooling switch—As required.
5. Fuel booster pump switches—ON.
6. AAU-19/A altimeter—RESET.

HYDRAULIC POWER SUPPLY SYSTEM (SCRAMBLE)

1. Throttle — IDLE.
2. Speed brakes switch – IN, then NEUTRAL.
3. Flight controls – Check.
4. Hydraulic system recovery – Check.
5. Trim – Check and set for takeoff.

GENERAL (SCRAMBLE)

1. Canopy—Close and lock; check light not illuminated.
2. Ejection seat safety pin—Remove. (P-IP)
3. Engine pressure ratio—Set.
4. Chocks—Removed.
5. External tank safety pins—Removed.
6. Fuel control emergency system—Check.

PREFLIGHT ENGINE CHECK

1. Fuel control emergency system — Check.
   a. Throttle — IDLE.
   b. Fuel control switch — EMER.
   c. Emergency fuel control warning light—On.
   d. Fuel flow—Check for slight fluctuation in fuel flow during system changeover.
   e. Fuel control switch—NORMAL (guard closed).
   f. Emergency fuel control warning light—Out.

TAXIING

CAUTION

When taxiing with fully fueled external tanks and fully armed, the following should be closely observed and care must be taken to prevent damage to the landing gear structure.

Taxiing and ground handling qualities of the airplane are normal but special techniques are required when taxiing with full external tanks and fully armed:

a. Taxi speed should be held to the minimum practical.

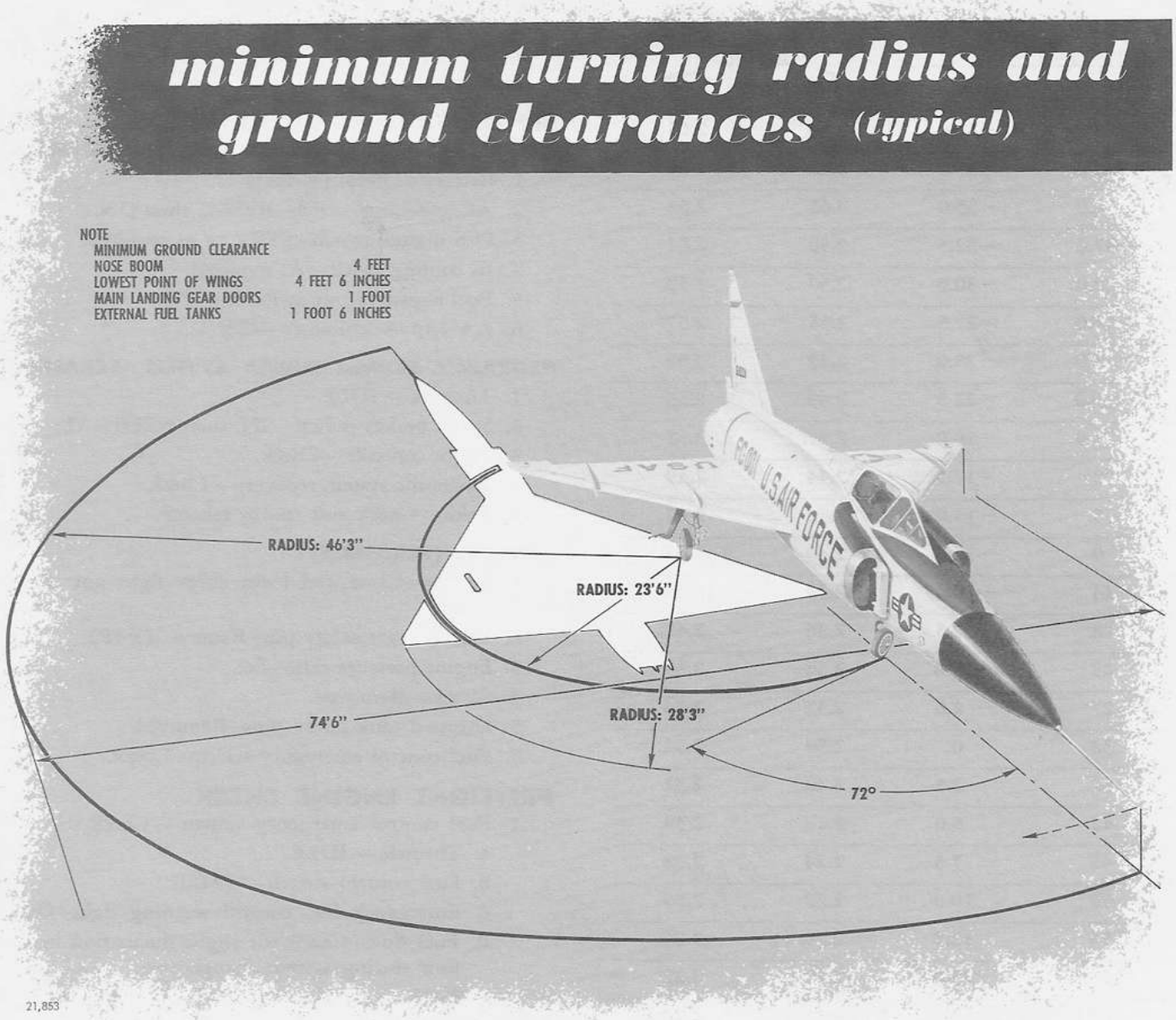


Figure 2-4

b. Do not apply wheel brakes during turns. Slow the airplane prior to reaching the turn point.

c. All turns should be made with as large a radius as practical and at a reduced speed.

There may be a slight delay in the braking between the pedal release and braking action release. Directional control utilizing differential braking is possible but is not recommended due to the loss in taxi speed and excess wear on tires and brakes. Normal idle thrust is sufficient for all taxi operations under most conditions. Do not taxi in the emergency fuel system as a standard practice because of poor engine oil scavenging at reduced rpm. The emergency fuel system should be used during taxiing only when reduced thrust is required during slippery (ice or snow) conditions. Operation in the emergency fuel system requires cautious throttle movement and egt monitoring to preclude engine overtemperature. 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.

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, lightly apply brakes to check for proper operation. See figure 2-4 for Minimum Turning Radius.

CAUTION

Excessive rudder pedal pressure with nose wheel steering engaged and the airplane not in motion may damage the steering mechanism.

2. Flight instruments — Check & set. (P-IP)

3. Navigation equipment — Check.

   Check TACAN ILS operation and the heading indicator (slaved) for indicator changes during taxiing.

BEFORE TAKEOFF

PREFLIGHT AIRPLANE CHECK

1. Canopy—Close and lock, check light not illuminated.

CAUTION

TF Do not use excessive force to latch canopy in closed position.

If canopy was opened for taxiing, visually recheck the canopy latches after the canopy warning light goes out.

2. Nesa switch(es)—NORMAL or ON.
3. Rain clear switch—As required.
4. Pitot heat switch—ON.
5. Anti-ice switch—AUTO.
6. Anticollision light—As required.

Note

The rotating anticollision light should be turned OFF during flight through reduced visibility conditions where the pilot could experience vertigo as a result of the rotating reflections of the light against the clouds. In addition, the light would be ineffective as an anticollision light during these conditions since it could not be observed by pilots of other airplanes.

7. Cockpit no-fog vent suit switch—ON.

WARNING

- Excessive moisture condensation may occur through the cabin pressurization system if the canopy no-fog vent suit switch is not on or the automatic temperature control is not set at one o'clock or higher. 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.

- If condensation occurs and the cockpit no-fog vent suit switch is in the ON position, place the cabin air switch to the RAM position.

8. Takeoff trim—Recheck.
9. Shoulder harness inertia reel handle—UNLOCKED. (P-IP)
10. Warning lights—Out.
11. IFF—Check and set.

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. Positive nose wheel steering is available by holding nose wheel steering button depressed.

TAKEOFF

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:

WARNING

Avoid wake turbulence. Allow a minimum of two minutes before takeoff behind a heavy aircraft or helicopter and four minutes behind extremely heavy aircraft such as C-5A and Boeing 747. With effective crosswinds of over five knots, the interval may be reduced. Attempt to remain above and upwind of the preceding aircraft's flight path.

1. Flight instruments check:
   a. Heading indicator—Check.
   b. Magnetic compass—Check.
   c. AAU-19/A altimeter—RESET.
2. Throttle—FULL MIL POWER.

CAUTION

Prior to advancing the throttle to FULL MIL POWER, ensure that the fuel control switch is in NORMAL. If the throttle is advanced routinely with the switch in EMERGENCY, engine overtemp and damage can occur without a correspondingly high EGT indication due to lag in the thermocouples.

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 egt limits are exceeded or that more than 15 seconds are required for the engine to accelerate from idle to maximum rpm. If more than 15 seconds are required, abort.

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.

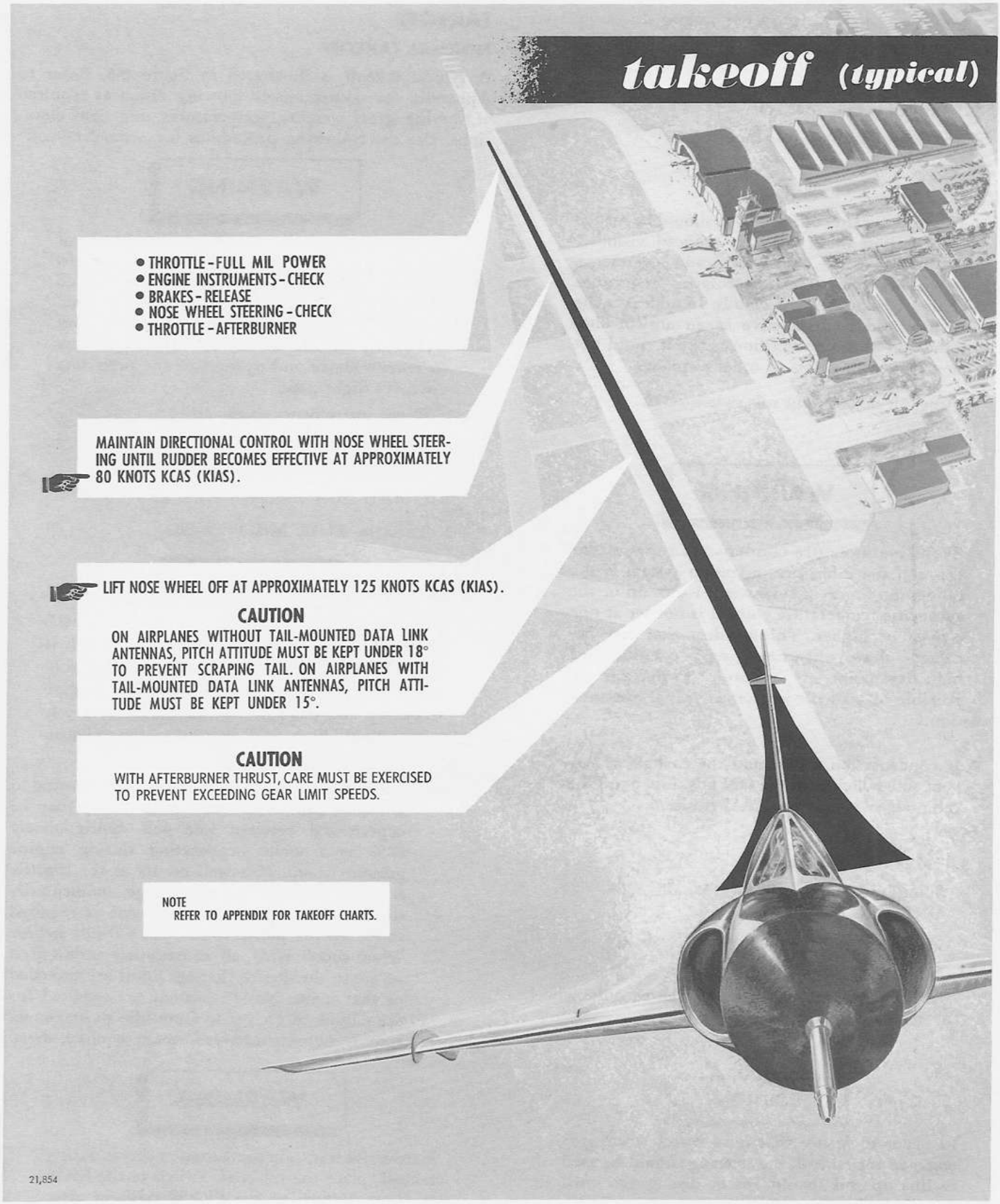

takeoff (typical)
• THROTTLE - FULL MIL POWER
• ENGINE INSTRUMENTS - CHECK
• BRAKES - RELEASE
• NOSE WHEEL STEERING - CHECK
• THROTTLE - AFTERBURNER
MAINTAIN DIRECTIONAL CONTROL WITH NOSE WHEEL STEERING UNTIL RUDDER BECOMES EFFECTIVE AT APPROXIMATELY 80 KNOTS KCAS (KIAS).
LIFT NOSE WHEEL OFF AT APPROXIMATELY 125 KNOTS KCAS (KIAS).
CAUTION
ON AIRPLANES WITHOUT TAIL-MOUNTED DATA LINK ANTENNAS, PITCH ATTITUDE MUST BE KEPT UNDER 18° TO PREVENT SCRAPING TAIL. ON AIRPLANES WITH TAIL-MOUNTED DATA LINK ANTENNAS, PITCH ATTITUDE MUST BE KEPT UNDER 15°.
CAUTION
WITH AFTERBURNER THRUST, CARE MUST BE EXERCISED TO PREVENT EXCEEDING GEAR LIMIT SPEEDS.
NOTE
REFER TO APPENDIX FOR TAKEOFF CHARTS.
21,854

Figure 2-5

3. Engine instruments—Check.

   Check tachometer, exhaust gas temperature gage, fuel flow indicator, engine pressure ratio gage and oil pressure gage, for normal operating limits.

Note

- The TAKEOFF CHECK TABLE, this section, has a +.03 pressure ratio tolerance over the minimum trim pressure ratio specified in T.O. 1F-102A-2-4 to permit overshoot on cold engines. Takeoff is permitted with stabilized engine conditions of .03 pressure ratio less than that shown in the TAKEOFF CHECK TABLE.
- Pressure ratio indication below normal may indicate activation of the anti-ice system.

4. EGT spread (airplanes with hot-section analyzer) —Check (140°C Max.).

   At military thrust in takeoff position, when desired engine pressure ratio (EPR) has been reached, depress EGT spread switch and check spread within limits; record rpm and spread readings.

5. Brakes—Release.

   Release brakes and establish a straight takeoff roll.

CAUTION

Feet must be positioned so that brakes will not be inadvertently actuated after release.

6. 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.

7. Throttle—Outboard to light afterburner.

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.
- If afterburner does not light on the first attempt, abort the takeoff.

8. Engine instruments—Check.

   Check pressure ratio gage for a drop and then a rise when lighting afterburner.

Maintain directional control with nose wheel steering until the rudder becomes effective at approximately 80 KIAS. Beginning at 125 KCAS (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

- Pitch attitude must be kept under 18° to prevent scraping the tail. On airplanes with tail-mounted data link antennas, pitch attitude must be kept under 15°.
- With afterburner thrust, care must be exercised to prevent exceeding gear limit speeds before landing gear is fully retracted.

If, without engine acceleration EGT increases and approaches allowable limits, reduce thrust to maintain temperature within limits. Monitor tachometer. Engine speed in excess of maximum rpm limits should be noted on AFTO Form 781.

WARNING

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

Pitch attitude must be kept under 18° to prevent scraping the tail. On airplanes with tail-mounted data link antennas, pitch attitude must be kept under 15°.

CROSSWIND TAKEOFF

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

AFTER TAKEOFF

When airplane is definitely airborne:

1. Landing gear handle—Up.

CAUTION

To prevent airloads from inflicting structural damage, the landing gear should be up and locked before limit speed is reached.

Note

Use care not to knock the throttle out of AFTERBURNER when retracting the landing gear.

2. Gear checked—Up.
   Check that landing gear warning lights extinguish and warning horn ceases after gear retracts.

CAUTION

- If gear requires longer than six seconds to fully retract, do not trouble-shoot the system by cycling the gear. Restrict further gear operation to extension for landing and make entry on AFTO 781 to this effect.
- If the landing gear warning light does not extinguish or the audible warning signal does not cease, refer to EMERGENCY LANDING GEAR RETRACTION, Section III.

CLIMB

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

1. EGT—Monitor.
   Adjust throttle setting as necessary to prevent engine overtemperature during climb.

WARNING

If EGT exceeds limits, reduce thrust and increase airspeed. Abort the mission and land as soon as practicable. Make an entry on AFTO 781 indicating time and temperature peaks of overtemperature operation.

2. Damper systems—Engage.

WARNING

- Initial engagement of the pitch damper should be made prior to exceeding 330 KCAS (320 KIAS) and at an altitude that will provide adequate ground clearance in event of any phasing errors that could occur during the initial engagement.
- For airplanes not equipped with HEP valves, flight operation at speeds in excess of 330 KCAS (320 KIAS) without pitch damper engaged requires extreme care to prevent severe pilot induced oscillations.

3. Cockpit no-fog vent suit switch—As required.
   With the cockpit no-fog vent suit switch in the OFF position, temperature control is accomplished by the automatic temperature controller.
4. Oxygen—Check. (P-IP)
5. IFF—Checked.

Note

As soon after takeoff as flight conditions permit, positive operation of the IFF should be established with an Air Traffic Control Facility if flight route requires an operative IFF. Consult appropriate FLIP documents for IFF traffic control requirements and procedures.

6. Battery switch—ON.
7. Armament safety check—Accomplish.
   a. Igniter control switch—OFF (safetied if armament is aboard) (some airplanes); TRAINING (safetied if armament is aboard) (other airplanes).
   b. Special weapon switch—DISARM (if installed).
   c. Armed-safe switch—SAFE (safetied if armament is aboard).
   d. Armament selector switch—TRIGGER SALVO.
   e. Armament triggers—FIRST DETENT.

WARNING

With the armament trigger in the FIRST DETENT position, if the armament doors start to open, release the trigger. Place the armament selector switch in the SNAKE position and do not use the radar during the mission.

8. Fuel feeding—Check symmetrical at 5500 pounds. When fuel quantity reaches 5500 pounds, check for symmetrical feeding prior to turning on external tanks.

Note

When a chaff dispenser or ECM pod is installed under one wing and an external fuel tank under the other wing, asymmetrical fuel feeding will occur after the external tank fuel transfer switch is turned on. For fuel management procedures, refer to FUEL FLOW EQUALIZER FAILURE, Section III.

9. External tank fuel transfer switch—ON, when internal fuel quantity indicates 5500 pounds or less.
10. Altimeter—Reset to 29.92 (as prescribed). (P-IP)

CRUISE

Note

- During flight, since the fuel quantity indicating system does not indicate individual tank fuel quantity, fuel feeding should be monitored to verify symmetrical fuel flow. Minor fuel imbalance can be expected due to tolerances within the fuel system. A fuel imbalance of 500 pounds or greater could be an indication of a fuel system malfunction that requires corrective action. Fuel imbalance can be recognized by fuel quantity gage readings, wing heaviness, or one external tank failing to feed. Should an imbalance be indicated, refer to FUEL FLOW EQUALIZER FAILURE, Section III.

- During operation in high humidity conditions (ground dew point 20°C or higher) operate canopy defog as necessary 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.

- Because of possible compressor stalls from engine inlet icing, the anti-ice switch should be in MANUAL ON when operating in conditions of visible moisture. Under these conditions, engine operation below approximately 91% rpm may not supply sufficient heat to keep the engine compressor inlet clear of ice. Also 91% rpm should approximate 96% of the data plate rpm for J-57 engines. The widest stall margin of J-57 engines is obtained at 96% of data plate rpm. When it is necessary to penetrate weather conditions conducive to compressor stalls, stabilize engine rpm at approximately 91%, if practicable.

- If EGT is increasing, without engine acceleration, so as to exceed allowable limits, reduce thrust to remain within limits.

- If afterburner light-up is not accomplished within two seconds at sea level (five seconds at altitude) to refill afterburner igniter flask after the throttle is moved outboard to the AFTERBURNER range, the throttle should again be moved inboard and retarded to not less than 89% and after two to five seconds, readvanced to full AFTERBURNER to recycle the igniter. The recirculating igniter in the system does not alter system function. Recycling, as previously described, is required if afterburner light-up is not accomplished on the first attempt.

Refer to Appendix for recommended cruise procedures. Refer to Section IX for recommended procedures for penetrating weather phenomena conducive to compressor stalls.

AUTOMATIC FLIGHT CONTROL SYSTEM NORMAL OPERATION

Engaging Procedure

WARNING

AFCS will not be used in close formation.

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. FCS master switch—STBY or ON.
2. Yaw, then pitch 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 interrupt 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, or if AILAS is engaged with glide-slope capture. 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. Yaw or pitch damper systems—Disengage.
3. Emergency damper disconnect button—Depress.
4. FCS master switch—WARM or OFF.

CAUTION

The FCS master switch should only be used in an emergency to disengage AFCS to preclude damage to the FCS vertical gyro.

FLIGHT CHARACTERISTICS

Refer to Section VI for flight characteristics.

WEAPON DELIVERY

Refer to T.O. 1F-102A-1A for weapon delivery information.

DESCENT

Note

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

1. Fuel quantity gage—Check.
2. Forward boost pumps—Check (if fuel quantity in either No. 3 tank is 570 pounds or less).
   Refer to Section III for fuel boost pump operation under low fuel condition.
3. UHF control switch—BOTH.
4. Armed-safe switch—SAFE (safetied if armament is aboard).
5. Igniter control switch—OFF (safetied if armament is aboard) (some airplanes); TRAINING (safetied if armament is aboard) (other airplanes).
6. Special weapon switch—DISARM.
7. Armament selector switch—SNAKE.
8. Altimeter—Set to station pressure setting. (P-IP)

8A. AAU-19/A altimeter (if installed)—Check servoed mode accuracy.

Crosscheck servoed mode reading against pneumatic mode. If above 20,000 feet MSL and the difference exceeds 250 feet, remain in STBY. If during landing approach the difference exceeds that observed during before takeoff ground checks, remain in STBY. Differences greater than these should be entered in the AFTO Form 781.

9. Hydraulic pressure gages—Check.
10. FCS master switch—As required.
11. IFF—Check and set.
12. Canopy defog switch—As required.
13. Cockpit no-fog vent suit switch—ON.

Cabin temperature control knob—AUTOMATIC, midway or hotter (airplanes without cockpit no-fog vent suit switch).

WARNING

If cockpit no-fog vent suit system is inoperative 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.

14. Throttle—Increase 5% (momentarily).

Note

At 5000 feet above desired level off altitude, momentarily advance throttle five percent to a minimum of 90% rpm. This will afford more time and altitude to initiate corrective action in the event of a malfunction of the fuel control, throttle linkage hang-up, or possible icing in the fuel system.

BEFORE LANDING

The normal landing pattern entry speed can vary over a wide range; however, the most desirable speed range is from 300 to 325 KCAS (KIAS). At the break, the speed brakes should be extended (if desired) and thrust reduced to obtain desirable gear-down speed. The downwind leg should be flown at 220 KCAS (KIAS). Prior to turning base, extend the landing gear and check for a safe indication. As the turn-on to base leg is started, a descent should be established and airspeed reduced on the base leg. Refer to the Landing Distance Charts in the Appendix for recommended base, approach, and touchdown speeds. See figure 2-6 for typical landing pattern.

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.

Note

With a speed brakes system malfunction, it is possible for one brake to extend while the other remains retracted, resulting in violent yawing. If this occurs, it is desirable to open the speed brakes early enough in the pattern to permit immediate retraction and recovery.

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.

1. Landing gear handle—Down; gear checked down; warning light out. (P-IP)

Note

The warning light will not illuminate and the audible warning signal will not sound unless the throttle is retarded to 80% rpm or below, airspeed is less than 210 (±10) KCAS (KIAS), and altitude is below 10,000 feet and the landing gear has not been extended or is in an unsafe condition.

- Refer to Section V for landing gear maximum extension speed and tire ground speed limits.
- Above 205 KCAS with landing gear extended a high frequency buffet occurs. This is normal and is due to airflow in wheel well areas.

2. Arrest hook light—Check not illuminated.
3. Speed brakes switch—OUT then neutral prior to landing.

LANDING

WARNING

Avoid wake turbulence. Allow a minimum of two minutes separation before landing behind a heavy aircraft or helicopter and four minutes behind extremely heavy aircraft such as C-5A and Boeing 747. With effective crosswinds of over five knots, the interval may be reduced. However, attempt to remain above and upwind of the preceding aircraft's flight path. Wake turbulence is most dangerous during the approach and flare prior to touchdown when it is calm or there are light crosswinds.

NORMAL LANDING

A typical landing pattern and recommended procedures are shown in figure 2-6. Refer to the 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:

CAUTION

On airplanes without tail-mounted data link antennas, pitch attitude must be kept under 18° to prevent scraping tail. On airplanes with tail-mounted data link antennas, pitch attitude must be kept under 15°.

a. Flare out.

CAUTION

The TF airplane, in contrast to the F airplane, requires a slower rate of thrust reduction during the landing flare due to increased drag caused by the cockpit configuration. Therefore, a small amount of thrust must be maintained until touchdown for normal landing configuration in the TF airplane.

F b. Throttle—IDLE during flareout.
TF Throttle—Reduce thrust 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.
- Be careful not to inadvertently depress the brake pedals as locked brakes and blown tires may result from very slight brake application on touchdown.

c. Touchdown.

(1) Drag chute—Deploy.
Drag chute may be deployed immediately after touchdown. There is a slight nose-down pitch when the drag chute is deployed.

(2) If drag chute fails to deploy hold nose up for maximum aerodynamic braking.

Note

Aerodynamic braking is relatively ineffective unless the nose is held in the takeoff attitude (approximately 14° to 15° nose-high).

(3) Lower nose wheel to runway.

Hold nose-high attitude to approximately 100 KIAS.

Note

- After the nose has been lowered to the runway and normal braking applied, additional aerodynamic braking may be obtained below 100 KCAS (KIAS) by holding maximum back stick (without raising the nose from the runway).
- 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 KCAS (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.

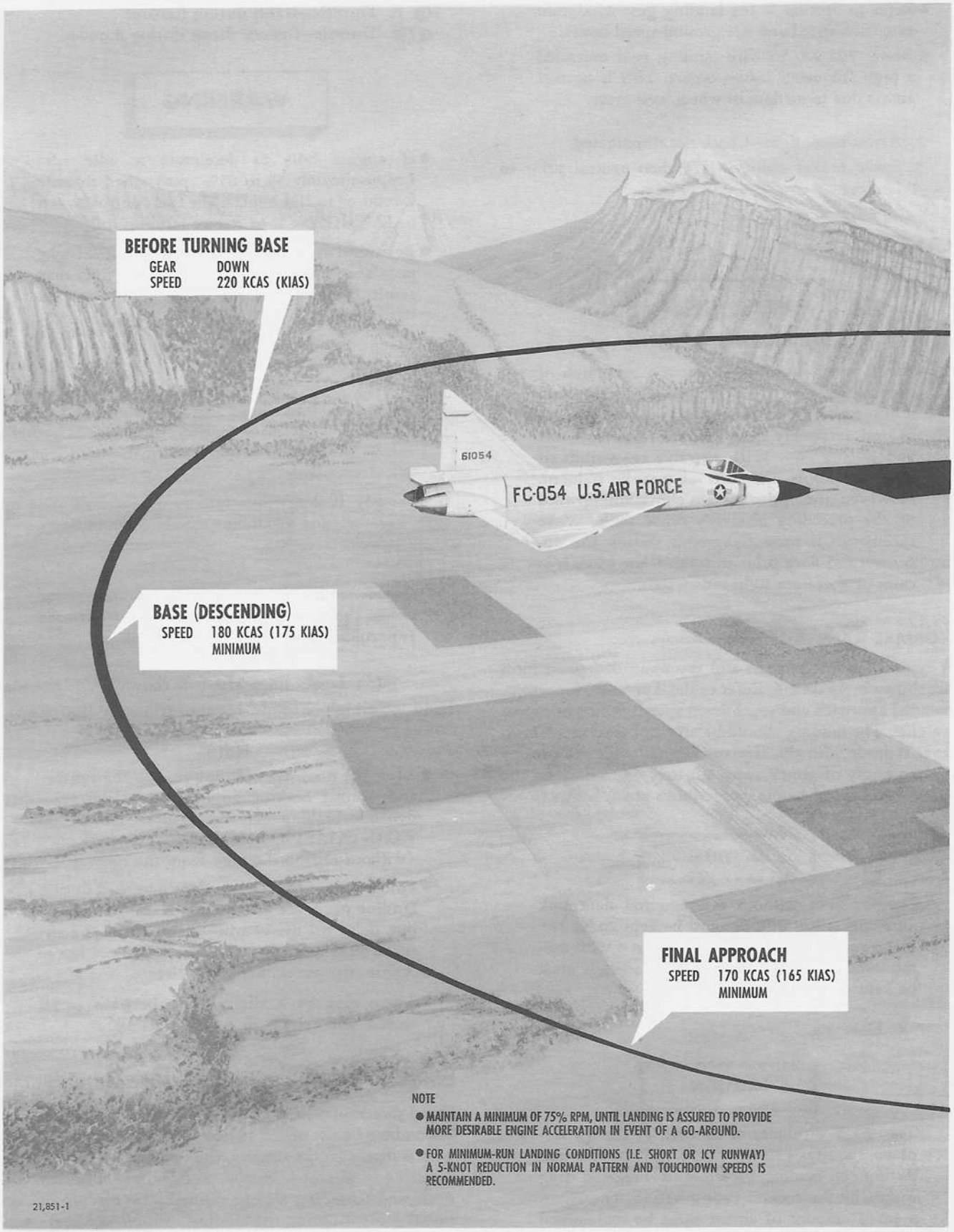
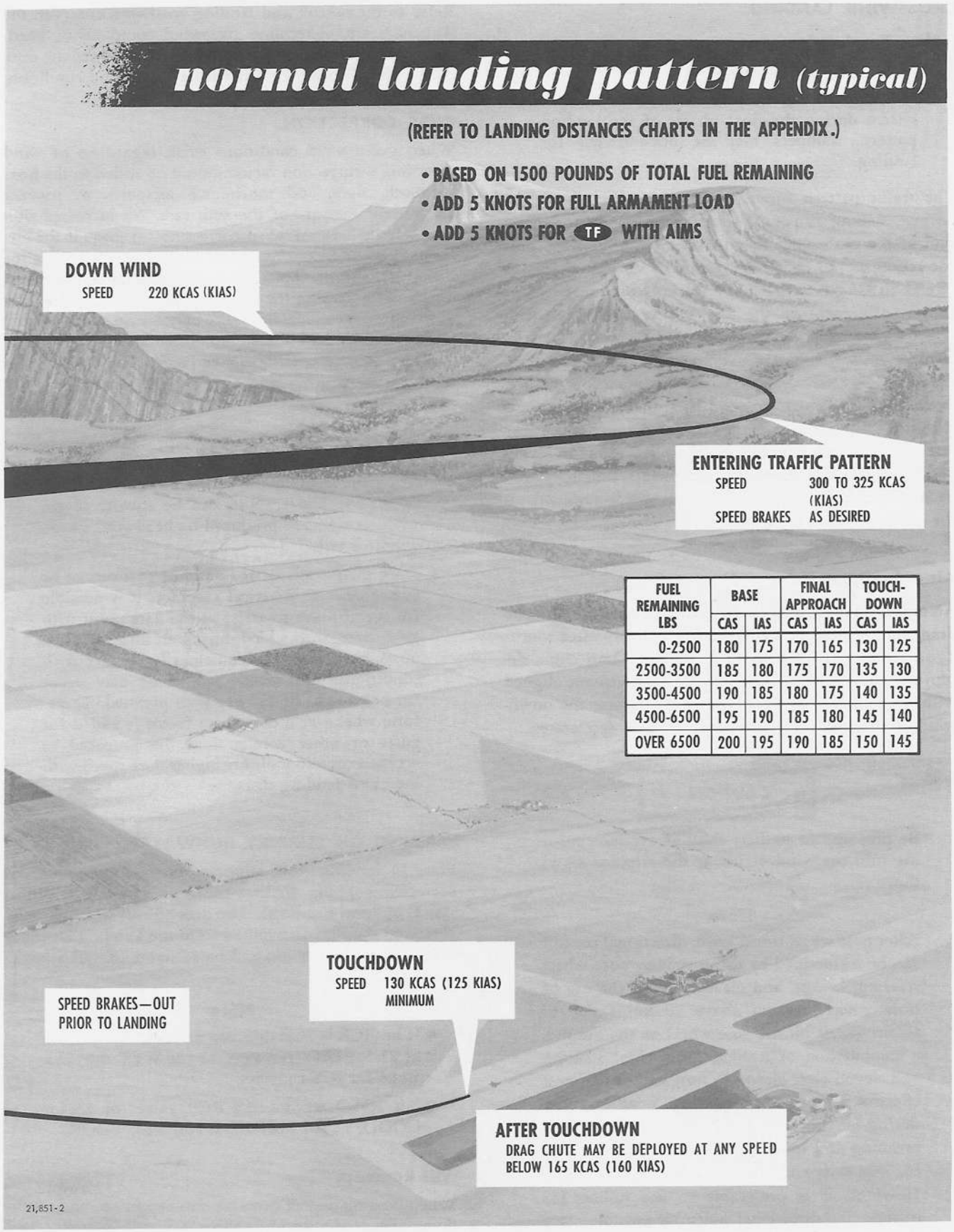

BEFORE TURNING BASE
GEAR DOWN
SPEED 220 KCAS (KIAS)
61054
FC-054 U.S. AIR FORCE
BASE (DESCENDING)
SPEED 180 KCAS (175 KIAS) MINIMUM
FINAL APPROACH
SPEED 170 KCAS (165 KIAS) MINIMUM
NOTE
● MAINTAIN A MINIMUM OF 75% RPM, UNTIL LANDING IS ASSURED TO PROVIDE MORE DESIRABLE ENGINE ACCELERATION IN EVENT OF A GO-AROUND.
● FOR MINIMUM-RUN LANDING CONDITIONS (I.E. SHORT OR ICY RUNWAY) A 5-KNOT REDUCTION IN NORMAL PATTERN AND TOUCHDOWN SPEEDS IS RECOMMENDED.
21,851-1

Figure 2-6

normal landing pattern (typical)

(REFER TO LANDING DISTANCES CHARTS IN THE APPENDIX.)

- BASED ON 1500 POUNDS OF TOTAL FUEL REMAINING
- ADD 5 KNOTS FOR FULL ARMAMENT LOAD
- ADD 5 KNOTS FOR TF WITH AIMS



FUEL REMAINING LBS	BASE		FINAL APPROACH		TOUCH-DOWN	
	CAS	IAS	CAS	IAS	CAS	IAS
0-2500	180	175	170	165	130	125
2500-3500	185	180	175	170	135	130
3500-4500	190	185	180	175	140	135
4500-6500	195	190	185	180	145	140
OVER 6500	200	195	190	185	150	145

CROSSWIND LANDING

Before Touchdown

Note

If it becomes necessary to override the damper system during the final phases of the landing pattern, dampers may be disconnected for landing.

The traffic pattern for a crosswind landing should be normal, making proper allowances for strength and direction of the crosswind. In gusty wind conditions, a gust correction should be applied to final approach airspeeds. (Refer to GUST CORRECTION, this Section.) Proper runway alignment on the final approach can be maintained by crabbing and/or dropping one wing. Since the airplane must not be allowed to touch down in a crab, the wing-low method is preferred. The wing-low method is considered adequate to compensate for all crosswinds within the allowable range for this airplane. Reduce sink rate to a minimum in order to accomplish a smooth touchdown. At increased crosswind components, sink rates must be minimized due to increased side loads imposed on the landing gear.

After Touchdown

Accomplish touchdown in the center of the runway. The drag chute should be deployed immediately after touchdown (below 165 KCAS (160 KIAS). Drag chute deployment at touchdown tends to keep the airplane aligned with the runway. The chute also counteracts the downwind weathervaning of the airplane as it decelerates.

CAUTION

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

Note

After nose wheel touchdown, directional control can be maintained by using rudder, nose wheel steering, brakes, and elevon. Avoid the use of nose wheel steering above 80 knots due to sensitivity. With the nose wheel on the ground, a combination of rudder, nose wheel steering and elevon (left elevon to turn left) is the most effective means of correcting weathervaning during landing roll. Elevon corrects for weathervaning and nose wheel steering compensates for side drift and maintains runway alignment. However, it is preferable to use rudder for maintaining directional control if possible, since the use of elevon imposes greater side loads on the landing gear.

Refer to the takeoff and landing crosswind charts in the Appendices to determine crosswind components, headwind components, and minimum touchdown or nose wheel lift-off speed for a particular crosswind condition.

GUST CORRECTION

When gusty wind conditions exist, regardless of wind direction, a correction factor should be added to the final approach, flare, and touchdown airspeeds to prevent inadvertent increase of the sink rate. An increased sink rate may be induced when correcting for gusts if the airspeed is at or below the minimum approach speeds for the gross weight. This gust correction factor is determined by increasing the final approach, flare, and touchdown speed by ½ the difference between steady wind and peak gust. Wind 10 knots, gusts to 20 knots — difference = 10 knots × ½ = 5 knots gust factor.) This gust factor is added to the gross landing weight airspeed.

Note

- The gust factor is added to provide a safety margin to maintain a desired sink rate while flying the airplane through a series of accelerations. The accelerations can be equally severe whether they are produced by headwind, crosswind, or tailwind.
- Since the frequency or timing of gust cannot be estimated with practical accuracy, it is possible for the airplane to arrive at the flare point with gust corrections added during an interval when gusts have stopped momentarily. Under such conditions, the touchdown point may occur further down the runway than planned. Therefore, whenever a correction factor is added for gusts or other accelerations, be prepared to accept a correspondingly higher flare speed with increased landing distance.

LANDING ON SLIPPERY RUNWAYS

Prior to landing on a slippery runway, determine the corrected stopping distance using the latest RCR (Runway Condition Reading). The ground roll which is to be corrected shall be determined from the Landing Distance Charts in the Appendix and based upon GOOD runway conditions.

Note

- If no RCR is available, use representative values of 23 for DRY runways, 12 for WET runways, and 5 for ICY runways.
- For ICAO reports, use RCR values of 23 for GOOD, 12 for MEDIUM, and 5 for POOR.

Wet Runways

When landing on wet runways, final approach and touchdown speeds should be reduced 5 knots from the final approach and touchdown speeds for the airplane gross weight as given on the landing distance charts in the

Appendix. Deploy the drag chute at touchdown. During the initial ground roll, maintain a 14° to 15° nose high attitude and use rudder and aileron for directional control. At approximately 90 KCAS (KIAS), lower the nose wheel to the runway and commence intermittent braking. Avoid use of nose wheel steering above 80 knots, due to sensitivity.

Note

- To reduce landing roll, move the throttle to IDLE and place fuel control switch to EMERGENCY.
- Do not reduce touchdown speed below the recommended minimum speed of 130 KCAS (125 KIAS).

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.

If the drag chute fails to deploy, hold the nose up to achieve maximum aerodynamic braking. At approximately 90 KCAS (KIAS), lower the nose wheel to the runway and commence intermittent braking at 80 KCAS (KIAS). See figure 7-2 for an illustration showing the benefits derived from aerodynamic barking.

Note

- Maximum aerodynamic braking is achieved only with the maximum angle of attack, when the nose is held to an attitude of 15° to 18°.
- Design and installation of the data link antennas are such that if the maximum angle of attack is exceeded during an emergency landing, only a minimum amount of structural damage will occur.

Icy Runways

The technique for landing on icy runways is the same as for wet runways. However, 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 reduce the touchdown speed 5 knots from that recommended for the particular airplane gross weight. At approximately 90 KCAS (KIAS), slowly lower the nose wheel to the runway and commence intermittent braking.

MINIMUM-RUN LANDING

For minimum-run landings on wet or icy runways, use the technique given for landings on wet or icy runways. For minimum-run landings on dry runways, reduce the final approach and touchdown speed 5 knots from those recommended on the landing distance charts in the Appendix. Touch down with the maximum pitch attitude which will not damage the tail cone. At touchdown, retard throttle to IDLE, place the fuel control switch to EMERGENCY, 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.
- Do not reduce touchdown speed below the recommended minimum speed of 130 KCAS (125 KIAS).

CAUTION

Braking is permissible above 90 KCAS (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.

HEAVY-WEIGHT LANDING

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

GO-AROUND

A go-around may be made from any point in the approach. See figure 2-7. Inasmuch as a relatively high engine rpm is maintained throughout most of the pattern, thrust will usually be readily available when the throttle is advanced. However, this may not be true when go-around is initiated after thrust has been reduced to near idle. Under this condition, or at any time when the go-around is initiated after landing flare has been started, the application of thrust may not be able to overcome the sink rate that has been established and the airplane may be committed to a touchdown. If such is the case, normal flare and touchdown speeds should be maintained resulting in a normal touchdown followed by a normal takeoff after go-around thrust has adequately increased airspeed. No appreciable trim change is experienced when applying thrust or retracting landing gear.

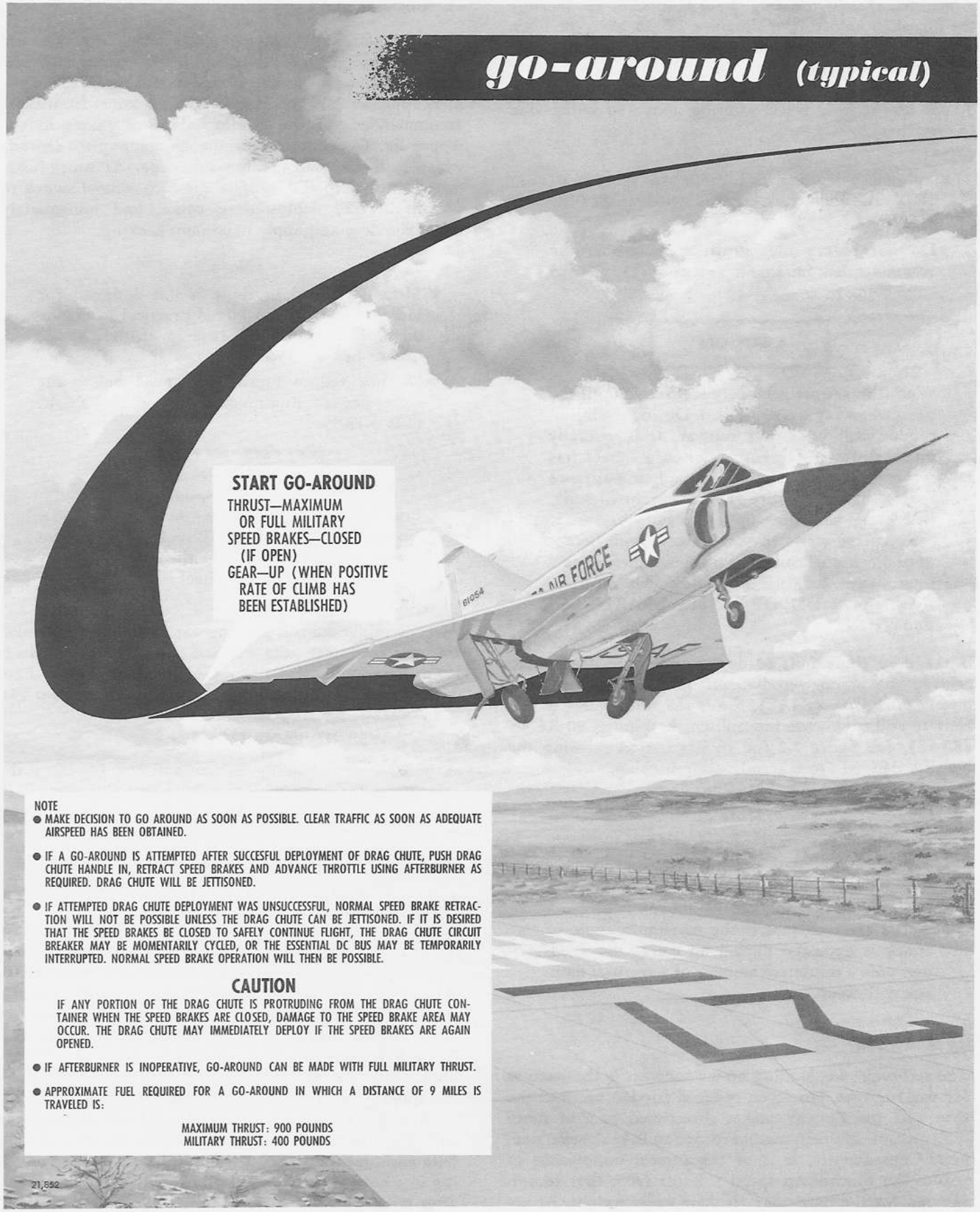

go-around (typical)
START GO-AROUND
THRUST—MAXIMUM OR FULL MILITARY
SPEED BRAKES—CLOSED (IF OPEN)
GEAR—UP (WHEN POSITIVE RATE OF CLIMB HAS BEEN ESTABLISHED)
NOTE
● MAKE DECISION TO GO AROUND AS SOON AS POSSIBLE. CLEAR TRAFFIC AS SOON AS ADEQUATE AIRSPEED HAS BEEN OBTAINED.
● IF A GO-AROUND IS ATTEMPTED AFTER SUCCESFUL DEPLOYMENT OF DRAG CHUTE, PUSH DRAG CHUTE HANDLE IN, RETRACT SPEED BRAKES AND ADVANCE THROTTLE USING AFTERBURNER AS REQUIRED. DRAG CHUTE WILL BE JETTISONED.
● IF ATTEMPTED DRAG CHUTE DEPLOYMENT WAS UNSUCCESSFUL, NORMAL SPEED BRAKE RETRACTION WILL NOT BE POSSIBLE UNLESS THE DRAG CHUTE CAN BE JETTISONED. IF IT IS DESIRED THAT THE SPEED BRAKES BE CLOSED TO SAFELY CONTINUE FLIGHT, THE DRAG CHUTE CIRCUIT BREAKER MAY BE MOMENTARILY CYCLED, OR THE ESSENTIAL DC BUS MAY BE TEMPORARILY INTERRUPTED. NORMAL SPEED BRAKE OPERATION WILL THEN BE POSSIBLE.
CAUTION
IF ANY PORTION OF THE DRAG CHUTE IS PROTRUDING FROM THE DRAG CHUTE CONTAINER WHEN THE SPEED BRAKES ARE CLOSED, DAMAGE TO THE SPEED BRAKE AREA MAY OCCUR. THE DRAG CHUTE MAY IMMEDIATELY DEPLOY IF THE SPEED BRAKES ARE AGAIN OPENED.
● 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
21,852

Figure 2-7

WARNING

After initiating a go-around, do not attempt to increase altitude until the required takeoff speed has been achieved. In ground effect, it is possible to lift off at speeds which can result in a critical angle-of-attack after leaving the ground effect. If this situation develops, there may be insufficient time to reduce up-elevator deflection and avoid excessive sink rate or post-stall gyrations.

Note

- If a go-around is attempted after successful deployment of the drag chute, push the drag chute handle in, retract speed brakes, and advance throttle using afterburner as desired. The drag chute will be jettisoned.
- If an attempted drag chute deployment was unsuccessful, normal speed brake retraction will not be possible unless the drag chute can be jettisoned from the airplane. If it is desired that the speed brakes be closed to safely continue flight, the drag chute circuit breaker may be momentarily cycled. Normal speed brake operation will then be possible.

CAUTION

- If any portion of the drag chute is protruding from the drag chute container when the speed brakes are closed, damage to the speed brake area may occur. The drag chute may immediately deploy if the speed brakes are again opened. Speed brakes should not be opened until after touchdown on subsequent landing.
- Be prepared to counteract yawing in the event a speed brakes malfunction causes unequal speed brakes retraction.

CLOSED TRAFFIC PATTERN

If a closed traffic pattern is anticipated, an airspeed of 220 to 230 KCAS (KIAS) should be obtained prior to starting the turn to downwind.

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. IR cooling switch—OFF.
2. RAT handle—Pull and hold for 4 seconds.
   The RAT should be extended while taxiing to prevent possible injury to ground personnel.
3. RAT handle—In.
   If the RAT handle did not return to the in position, push it in. If it binds or is stiff while pushing in, note this in AFTO 781.
4. Speed brakes switch—NEUTRAL. (P-IP)
   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.
5. Landing and taxi lights—As desired.
6. Drag chute—Jettison (when clear of runway).

Note

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

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, the chute being damaged from dragging on the ground, or airplane damage because of entanglement of the chute with the airplane. The drag chute should also be jettisoned in high wind or rain if the chute collapses.
- 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.

7. External tank fuel transfer switch—OFF.
8. Canopy—Open (if desired).

CAUTION

If the canopy is open during taxi operations, observe canopy limit speeds. Keep hands and arms away from canopy sill unless canopy support tool is in place.

9. Ejection seat safety pin—Installed, streamer visible (P-IP)
10. TACAN and ILS—OFF.

11. Mode 4 code selector knob—Hold for two to three seconds (if required).

11A. IFF master control knob—OFF.

Note

- If the mode 4 code selector knob has been moved to HOLD, wait at least 15 seconds before moving the IFF master control knob to OFF to prevent the code setting from being zeroized.

- Turn the IFF off as soon after landing as possible. This prevents signals from taxiing or parked airplanes which could block the controller's radar scope and interfere with the control of airborne airplanes.

12. Pitot heat switch—OFF.
13. Windshield rain clear switch—OFF.
14. Nesa switches—OFF.
15. Anti-ice switch—OFF.
16. Canopy defog switch—OFF.
17. Position lights – NAVIGATION.
18. Cockpit no-fog vent suit switch—OFF.
19. Takeoff trim—Set.
20. Dampers—Disengage.

ENGINE SHUTDOWN

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. 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 if it overshoots on recovery, it must return to normal within five seconds 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.
4. FCS master switch—OFF.
5. FCS scope intensity controls—Fully counterclockwise.
6. RPM—75% or above for 30 seconds.
7. Booster pump switches—OFF.
8. Battery switch—T/R FAIL.
   Placing battery switch to T/R FAIL will provide UHF operation.
9. AC bus switch—ON.
10. AC and dc generators—OFF.
11. Canopy—Open.
12. Throttle—OFF; check fuel flow to zero.

Note

Check that engine decelerates freely by listening for any excessive engine noises during shutdown.

13. Speed brakes switch—OUT. (P)
    The speed brakes should be left in the extended position to facilitate turnaround.
14. AC emergency generator—Check output (103 to 140 volts) down to 15% rpm.
15. AC bus switch—NOR (15% rpm).
16. Position lights—Off.
17. Canopy support tool(s)—In place.
18. Armed-safe switch—SAFE (safetied if armament is aboard).
19. Igniter control switch—OFF (safetied if armament is aboard) (some airplanes). TRAINING (safetied if armament is aboard) (other airplanes).
20. Special weapon switch—DISARM (if installed).
21. Armament selector switch—SNAKE.
22. UHF—OFF.
23. Move elevons to remove hydraulic system pressure.

Note

While bleeding off hydraulic pressure check for blinking hydraulic pressure-low warning light followed by a steady light.

24. Battery switch—OFF.

WARNING

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

BEFORE LEAVING AIRPLANE

1. Ejection seat safety pin—Recheck installed, streamer visible. (P-IP)
2. Oxygen supply switch — OFF before removing mask. (P-IP)
3. Personal equipment leads — Disconnect. (P-IP)

CAUTION

If wearing an automatic opening parachute that has a key attached to the aneroid arming lanyard, the key should be stowed to prevent the chute from being opened inadvertently when leaving the cockpit.

4. Throttle quadrant cover—Installed.
5. Survival kit—Detach from parachute. (P-IP)
6. Engine hot-section analyzer data recorder—Check (if installed).

   Check recorder for temperature performance data. If overtemperature is observed, record in AFTO Form 781.
7. AFTO Form 781—Complete.

CAUTION

- Make appropriate entries on AFTO 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, emergency gear extension, emergency drag chute deployment, etc.
- To assure that airplane servicing may be accomplished safely, ensure the security of the airplane (ground safety pins, including external tank ground safety pins, chocks, etc.) before leaving the area.

STRANGE FIELD TURN-AROUND PROCEDURES

The following procedures provide servicing instructions when ground crew personnel are not familiar with the F and TF airplanes. These procedures 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. 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 accessory gear case in sufficient quantities to prevent an accurate oil quantity check.*
- *If other than MIL-L-7808 oil is used, refer to T.O. 42B2-1-1.*

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.
2. Service with oil until visible.
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. Armed safe switch—SAFE (safetied if armament is aboard).
   c. Igniter control switch—OFF (safetied if armament is aboard) (some airplanes); TRAINING (safetied if armament is aboard) (other airplanes).
   d. Special weapon switch—DISARM (if installed).
   e. Armament selector switch—SNAKE.

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. Check armament pneumatic shut-off valve open.
5. Armament reset circuit breaker—IN.
   Armament power circuit breaker—OUT.
6. Actuate armament door switch—CLOSE.
   Actuate launcher selector switches—UP.
7. Momentarily actuate armament rest switch (right-hand reset switch).

Note

Armament reset light will illuminate if system resets.

8. Armament door switch—OPEN (doors should open).
9. Armament reset circuit breaker—OUT.
10. External power—Disconnect (or) battery switch—OFF.
11. 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. Check armament pneumatic shut-off valve open.
2. Armament bay door safety locks—Removed (if installed).
3. Armament reset circuit breaker—In.
4. Actuate armament door switch—OPEN, then release.
   Actuate launcher switches—UP, then release.
5. Momentarily actuate armament reset switch (right-hand reset switch).

Note

Armament reset light illuminate if system resets.

6. Armament door switch — Close (doors should close).
7. External power — Remove (or) battery switch — OFF.

ARREST HOOK RECOCKING

The arrest hook system has a self-contained air-and-oil hydraulic pressure system which normally does not require servicing; however, after recocking the arrest hook arm, check the system pressure gage through the engine access door on the left side of the airplane for approximately 750 psi. To recock the arrest hook arm, proceed as follows:

1. When preparing to stow the tail hook, assure that the cockpit release handle is pushed fully forward.
2. Manually raise hook arm into the stowed position (one man can, by exerting a force of 60 pounds, lift the hook arm to the stowed position).
3. Install safety pin.

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

Note

Immediately after shutdown, the fluid level in both hydraulic reservoirs will normally be above the full mark due to thermal expansion of the fluid. Drain fluid from reservoirs only when fluid is at ambient temperature.

Primary Reservoir

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

Note

Open armament doors using procedures previously outlined.

1. Disconnect reservoir pressurization line quick-disconnect fitting located at the top center of the RAT compartment.
2. Depressurize reservoir by depressing button or top of reservoir cap or loosening cap.
3. Remove drain line cap cover.
4. Hold button on top of reservoir depressed or remove cap to allow excess fluid to drain.
5. Drain until flluid level is ¼ inch below full mark.
6. Release button. Replace reservoir cap.
7. Replace drain line cap cover and check for leaks.
8. Reconnect pressurization line quick-disconnect fitting and recheck fluid level.

Note

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

9. 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 ⅛ inch when the reservoir is pressurized.

SERVICING HYDRAULIC RESERVOIRS 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 or below, service with hydraulic fluid to ¼ inch below full mark by connecting servicing hose to quick-disconnect fitting on reservoir cap.

Note

- The airplane is unacceptable for flight if the fluid level in the primary reservoir decreases over ¼ inch or the secondary decreases over ⅛ inch after pressurization.
- The high-pressure pneumatic system should be serviced to 1500 psi or above to provide air for hydraulic reservoir 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 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.

TIRE SERVICING

GROSS WEIGHT (POUNDS	MAIN TIRES (PSI)	NOSE TIRES (PSI)
25,001 to 28,000	185	175
28,001 to 31,000	210	175
31,001 to 33,000	225	205

LIQUID OXYGEN

1. Service with liquid oxygen specification BB-O-925, Grade A, Type II.

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

On (F) airplanes, the radome anti-icing tank is located in the right forward electronics compartment. On (TF) airplanes, the radome anti-icing tank is located on the right side of the fuselage below aft end of canopy.

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

CAUTION

Accomplish all steps in sequence to assure satisfactory drag chute operation. Failure to do so will result in damage to equipment or drag chute deployment malfunction.

1. Speed brakes switch—OPEN.
2. Insure that barrier probe is retracted with safety pin installed, or that probe is in full extended position.
3. Inspect drag chute for serviceability, ascertain that cone is free of nicks and burrs.
4. Inspect ripcord pin for burrs, nicks, and distortion.
5. Inspect drag chute canister for cleanliness, damage, foreign objects, and moisture.
6. Insert safety pin (6.75-inch jettison pin cable) in safety switch.
7. Push "D" ring into jaw mechanism, assure that "D" ring is not cocked in the jaws and is aligned with the "D" ring guide.
8. Place ripcord pin in ripcord guide leg.
9. Pull drag chute handle to OUT position, assure that ripcord pin travels a minimum of two inches.
10. Place drag chute in drag chute canister with yellow T's at the top.
11. Push drag chute handle to IN position.
12. Install drag chute restraining straps (red lower strap to be installed last).
13. Align cone hole vertically with airplane so pin will pull out freely. Insert ripcord pin in cone and ripcord swaged ball in guide leg.
14. Remove temporary packing pin (red streamer attached).

CAUTION

Failure to remove the packing pin will render the drag chute inoperative.

15. Reset barrier probe.

CANOPY COUNTERBALANCE CYLINDER SERVICING

TF

The canopy counterbalance cylinder is not connected to the airplane high-pressure pneumatic system, and is charged separately. The servicing valve is located in the rear upper canopy area, adjacent to the counterbalance cylinder. Canopy must be in counterbalance.

WARNING

Hold the solenoid manual override handle in the up position throughout the entire servicing operation. If the handle is allowed to return to the down position during servicing, serious injury and extensive damage may result.

Note

- Ensure personnel are clear of canopy rail.
- Install all safety pins in the canopy jettison and seat ejection system.

1. Remove valve cap.
2. Connect low-pressure air source.
3. Loosen hex nut one-half turn.
4. Charge with dry air or nitrogen to 1500 (±25) psi.
5. Tighten hex nut and remove air source.
6. Replace cover and check for leaks.

START — SECOND ATTEMPT

The following procedure should be followed to manually replenish the starter fuel flask if external ac power is not available. 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.

ENGINE HOT-SECTION ANALYZER RECORDER CONDITIONING

1. Check recorder for recorded engine temperature performance data; enter data in AFTO Form 781.
2. Energize ac or dc essential bus. Check that EGT gage power failure flag is not on display.
3. Remove recorder reset button caps; reset recorder over temperature flags and clocks as required.

Note

- Do not reset recorder using pencil lead or other object which could break and enter the interior of the unit.
- Do not reset factor units counter.

4. Reinstall reset button caps; turn off electrical power.



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Note

Procedures containing boldfaced steps are shown in capital letters.

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INTRODUCTION

This section includes procedures to be followed to correct an emergency condition. The procedures, if followed, will insure safety of the pilots and aircraft until a safe landing is made or other appropriate action is accomplished. Multiple emergencies, adverse weather, and other peculiar conditions may require modification of these procedures. Therefore, it is essential that pilots determine the correct course of action by use of common sense and sound judgment. Procedures appearing in bold face capital letters are considered critical action. Procedures appearing in small letters are considered noncritical action. Each is defined as follows:

Critical Actions: Those actions which must be performed immediately without reference to a checklist if the emergency is not to be aggravated and injury or damage is to be avoided. These critical steps will be committed to memory.

Noncritical Actions: Those actions which contribute to an orderly sequence of events, and where there is time available to consult a checklist.

To assist the pilot when an emergency occurs, three basic rules are established which apply to most emergencies occurring while airborne. They should be remembered by each pilot. The rules follow:

1. Maintain aircraft control.
2. Analyze the situation and take proper action.
3. Land as soon as practicable.

GROUND OPERATIONS

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 shutoff valve switches—CLOSE.

   If the safety wire is broken, it must be reestablished that both valves are open before the next flight and before safety wire is replaced.
3. Fire fighting equipment—Summon.
4. Master switch—TRIP, then OFF.

   Allow three to five seconds for fuel shutoff valves to close prior to placing switch to TRIP, then OFF.

Note

When starting the aircraft with external power connected, the external power must be turned off or disconnected to remove external power from the aircraft.

EXCESSIVE EXHAUST GAS TEMPERATURE DURING START OR ENGINE FIRE AFTER SHUTDOWN

On some airplanes, excessive EGT is indicated by the exhaust gas overtemperature warning light.

1. **THROTTLE—OFF.**

   Summon fire fighting equipment.
2. 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 opening the manual shutoff valve in the left main landing gear wheel well.

3. Engine ignition disconnect switch—OFF.

   While in the left 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 starter may be fired but the engine will not light up since the engine ignition has been disconnected.

4. Fuel shutoff valve switches—CLOSE.

   If the safety wire is broken, it must be reestablished that both valves are open before the next flight and before safety wire is replaced.
5. Throttle—START.
6. Ignition button—Depress.

   To air-motor the starter hold the ignition depressed until rpm indication is evident on the tachometer.
7. 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 starter (if fuel is available in the starter flask) and aid in clearing the engine.
8. Ignition button—Release.

   Release the ignition button after the engine clears or if the fire cannot be extinguished by clearing.
9. Master switch—TRIP, then OFF.

   Allow three to five seconds for fuel shutoff valves to close prior to placing switch to TRIP, then OFF.

Note

When starting the aircraft with external power connected, the external power must be turned off or disconnected to remove external power from the aircraft.

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 3-1.

TAKEOFF

ABORT

An aborted takeoff should always be made with the landing gear extended and in accordance with those procedures necessary to stop the airplane. Use the following abort procedures:

1. **THROTTLE—IDLE OR OFF.**

Note

With the throttle in the OFF position, hydraulic pressure for nose wheel steering may not be adequate.

2. **DRAG CHUTE—DEPLOY.**

3. **ARREST HOOK—EXTEND** (if required).

WARNING

The arrest hook release handle must be pulled straight out. Any side load or twisting action on the handle may make it impossible to pull out.

4. **TANKS—JETTISON** (if required).
   External tanks must be jettisoned for successful barrier engagement unless the arrest hook is to be used (barrier webbing either lowered or absent).

5. Brakes—As Required.

6. Fuel shutoff valve switches—CLOSE.
   The fuel valves take three to five seconds to close and require electrical power during this time.

7. Master switch—TRIP, then OFF.

Note

With the master switch in the OFF position, nose wheel steering will be inoperative.

8. 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.

9. Canopy—Retain.
   Refer to CANOPY RETENTION VS JETTISONING.

RUNWAY OVERRUN BARRIER ENGAGEMENT (WITHOUT ARREST HOOK)

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. An extendible barrier probe is 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 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.



REMOVE ACCESS DOOR.

1



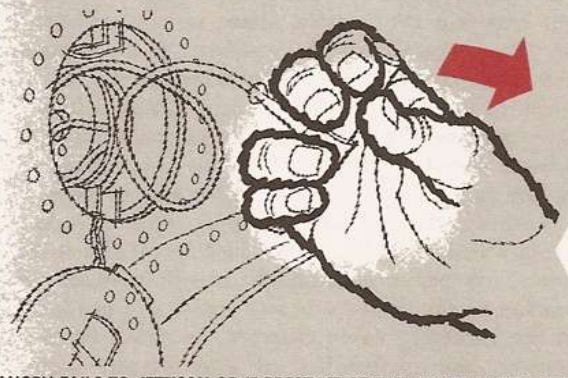
GRASP "T" HANDLE AND PULL OUTBOARD APPROXIMATELY 6 FEET TO JETTISON CANOPY.

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.



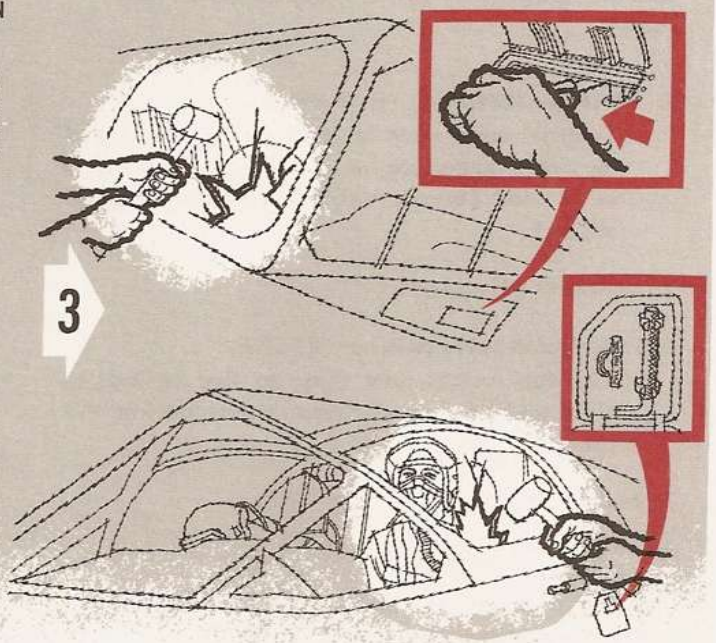
2

IF CANOPY FAILS TO JETTISON OR IF PRESENCE OF FUEL FUMES MAKES JETTISON INADVISABLE, GAIN ENTRANCE EITHER BY:

(F) 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.

(TF) a. INSERTING A ⅜ INCH SQUARE TOOL IN THE EXTERNAL CANOPY LATCH RELEASE (LEFT HAND SIDE BELOW CANOPY) AND TURN COUNTERCLOCKWISE. PLACE MANUAL CYLINDER VALVE HANDLE IN COUNTERBALANCE POSITION AND RAISE CANOPY. ON AIRPLANES 56-2324 AND SUBSEQUENT, OPEN LATCH ACCESS DOOR (LEFT HAND SIDE BELOW CANOPY TO OBTAIN THE ⅜ INCH SQUARE TOOL. RELEASE LATCH, PULL OUT THE "PULL FOR CBAL" HANDLE AND RAISE 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. BREAKING OF CANOPY GLASS MAY BE AIDED BY CHILLING WITH CO_2 EXTINGUISHER.



3

21,871-1

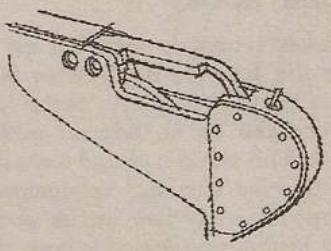
Figure 3-1

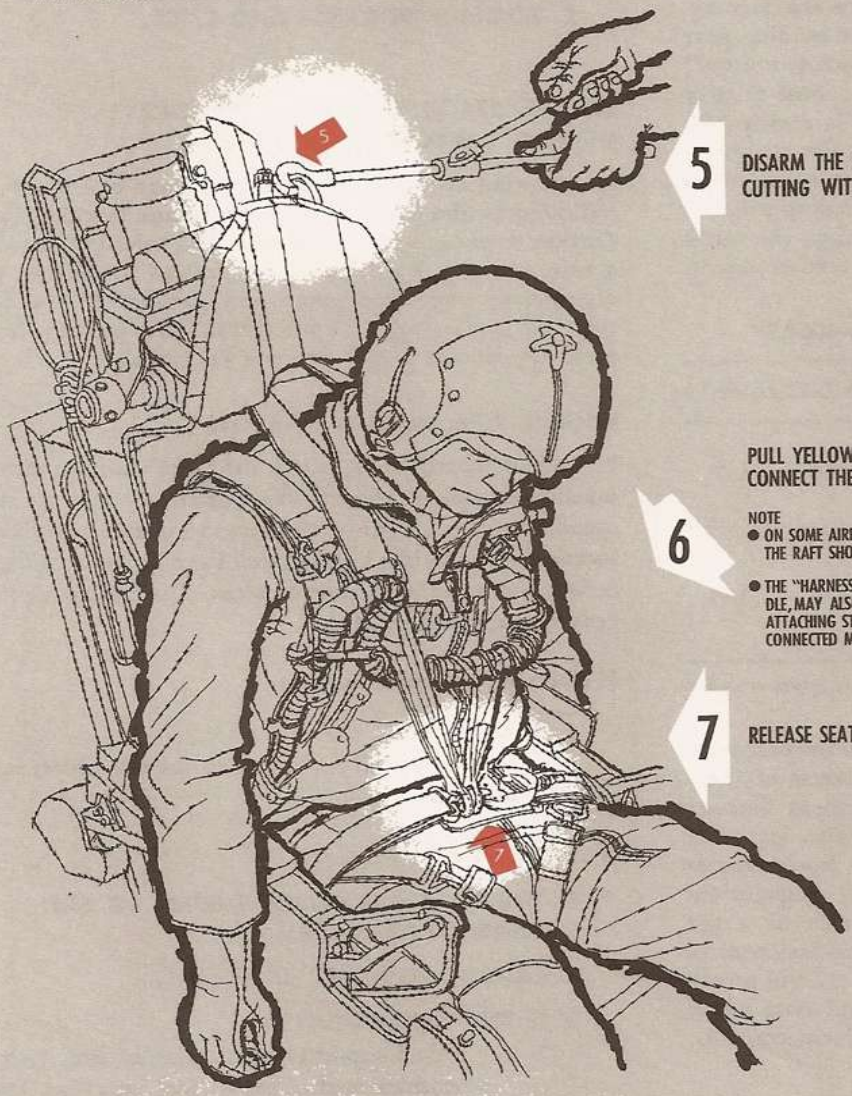
emergency entrance

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.

4





5 DISARM THE SEAT BY PUSHING QUICK DISCONNECT FITTINGS, OR BY CUTTING WITH HEAVY DUTY CABLE CUTTERS AS SHOWN.

6 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.

7 RELEASE SEAT BELT AND SHOULDER HARNESS.

21871-2

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.

RUNWAY OVERRUN BARRIER ENGAGEMENT (WITH ARREST HOOK)

The arrest hook system is installed on some airplanes for use with the BAK-9, BAK-12 or BAK-13 overrun barrier, or the MA-1 or MA-1A "chain" overrun barriers which are modified to include an additional "across-runway" cable. This cable is placed some distance in front of the landing gear webbing engagement cable and is supported approximately two inches above the runway. If the barrier installation includes both the landing gear barrier webbing cable and the tailhook "across-runway" cable, the tailhook cable will provide the most reliable operation. In case a barrier engagement is anticipated, external wing tanks should be jettisoned prior to lowering the gear for landing; however, in the event of an aborted takeoff, or an emergency on the landing roll, the external wing tanks should be retained when the arrest hook is to be used with webbing either lowered or absent.

BARRIER ENGAGEMENT SPEEDS—KCAS*

	MA-1/1A	BAK-9	BAK-12	BAK-13
Maximum Gross Weight	115	168	160	137
Normal Landing Gross Weight	115	178	170	145

Refer to Section V for engagement speeds at other gross weights.
**Subtract 6 knots for IAS.*

Safe off-center engagements may be made at lower speeds. Engagements can be made at speeds in excess of those listed; but as speed or off-center engagement distance increases, the risk of equipment failure also increases. There is no minimum speed for arrest hook barrier engagement. Excessive braking prior to engagement should be avoided to reduce the possibility of a tire blowout, which could cause loss of directional control and possibly prevent barrier engagement. During arrestment, airplane behavior is satisfactory, and nose wheel steering or brakes may be used for directional control.

CAUTION

Above 80 knots ground speed, nose wheel steering becomes sensitive and directional control is very difficult to maintain.

Steer the airplane toward the center of the runway and attempt to engage perpendicular to the cable. Deploy the arrest hook as soon as possible. Just prior to engagement, the throttle should be in the OFF position. Be prepared to correct for yaw after engagement which might cause the airplane to swerve off the edge of the overrun.

ENGINE FAILURE DURING TAKEOFF

If Decision Is Made To Stop

1. **ABORT.**

If Takeoff Is Continued

1. **ZOOM IF POSSIBLE AND EJECT.**

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 (up to full military power); however, afterburner power will not be available.

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:

If Decision Is Made To Stop

1. **ABORT.**
   If runway and overrun area permit, abort; refer to ABORT.

If Committed To Takeoff

1. **THROTTLE—MAXIMUM THRUST TO SAFE EJECTION ALTITUDE.**
2. **TANKS—JETTISON** (if required).
3. **IF FIRE EXISTS—EJECT.**
   Check for positive indication of fire, such as abnormal engine instrument readings, trailing smoke or flame, smoke in cockpit, or request a visual check from the ground or another airplane.
4. If fire cannot be confirmed, make decision to land or eject.

Note

Refer to EJECTION for emergency minimum altitudes.

TIRE FAILURE DURING TAKEOFF

If Below Nose Wheel Liftoff Speed

1. Abort.

If Takeoff Is Continued

1. Do not retract landing gear.

WARNING

Do not retract the landing gear with a blown tire. This will prevent jamming the gear in the up or intermediate position.

The instant the decision is made to abort and while performing the abort procedure, engage nose wheel steering, use opposite rudder, aileron, and braking to maintain directional control. Use opposite braking and aileron with nose wheel steering to maintain directional control after nose wheel steering is engaged. As the speed decreases, if vibration or shimmy increases, lock the brake on the wheel with the blown tire, but release prior to crossing barrier cable.

WARNING

The rudder pedals must be aligned with the nose wheel direction and the nose strut must be compressed to engage the nose wheel steering.

If a nose wheel tire has blown out on takeoff and the decision is made to abort, the brakes and rudder should be used for primary directional control and the drag chute should be deployed to reduce ground roll and nose wheel shimmy. Because of the extreme shimmy of the nose wheel with a blown nose wheel tire, nose wheel steering may be uncontrollable. Up elevator should be used to relieve as much pressure as possible from the nose wheel.

WARNING

With a blown tire, directional control is more difficult and braking efficiency is greatly reduced at high gross weights.

Landing gross weight should be reduced, if possible, and landing accomplished as directed in TIRE FAILURE ON LANDING.

AFTERBURNER FAILURE DURING TAKEOFF

Afterburner failure may be noted by a sudden loss of acceleration.

If Decision Is Made To Stop

1. **ABORT.**

   If sufficient runway remains to make a safe stop or insufficient runway available to continue takeoff at military thrust, abort takeoff; refer to ABORT.

If Committed To Takeoff

1. **THROTTLE—MILITARY.**

   If there is insufficient runway available to make a safe stop and takeoff can be safely continued at military thrust, move 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 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.

AFTERBURNER EXHAUST NOZZLE FAILURE DURING TAKEOFF

If the afterburner exhaust nozzle fails to open at the time of afterburner ignition, a rapid rise in engine pressure ratio and 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 engine pressure ratio and exhaust gas temperature rise rapidly and rpm drops or if any directional change is noted, abort the takeoff. (Refer to ABORT.)

LANDING GEAR EMERGENCY RETRACTION

The landing gear emergency-up button can be used to retract the landing gear when the airplane is in the air or on the ground.

1. Landing gear handle—Up.
2. Landing gear emergency-up button—Depress (hold until gear collapses or comes up and locked).

Note

If this procedure is used in the air and is unsuccessful, do not troubleshoot the system by cycling the landing gear handle. Put the landing gear handle in the down position, verify three green lights, and leave the landing gear down.

CANOPY RETENTION VERSUS JETTISONING

The canopy should be retained (not jettisoned) during landing emergencies (except ditching), aborted takeoff, and barrier engagements. Under such emergency conditions, probability of survival is enhanced with canopy retention. The canopy is protection against being saturated with flaming fuel and is temporary protection against the direct effects of fire; also, retention of the canopy may prevent barrier cables and wires from entering the cockpit.

Note

- The canopy can be jettisoned from any position, i.e., closed, cinchdown, partially open, or full open.
- On TF airplanes, when 28-volt dc nonessential power is not available, the solenoid manual override handle must be used before the canopy can be raised.

The application of these considerations does not preclude the pilot from exercising sound judgment in jettisoning the canopy when he deems it necessary.

EMERGENCY GROUND EGRESS

The following procedures prescribe the fastest method of egress from a disabled airplane on the ground. They also provide pilot protection from fire. After the airplane is stopped, proceed as follows:

1. **OXYGEN SUPPLY SWITCH—OFF.**
2. **SURVIVAL KIT—EMERGENCY RELEASE.**

Note

- Pull survival kit emergency release handle rapidly through complete arc of travel.
- Maintain downward body pressure on the survival kit while actuating the survival kit emergency release handle to assure complete separation from the kit and kit lanyard.

3. **RELEASE LAP BELT AND SHOULDER HARNESS.**
4. **CANOPY—JETTISON/OPEN/BREAK.**
   Refer to Figure 3-2 for use of canopy knife.
5. Abandon airplane.

WARNING

- If the nose gear is extended, there is approximately an eight foot drop to the ground. Exercise caution to prevent an incapacitating injury.
- The retention of helmet, mask (visor down, oxygen mask hooked), and parachute reduces egress time and provides protection from fire.

INFLIGHT

EJECTION

Note

For considerations affecting the decision to eject, refer to EJECTION VS FORCED LANDING.

Every emergency in which ejection is considered will have its particular set of circumstances, involving such factors as airplane speed, altitude 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.

EJECTION AT LOW ALTITUDES

Ejection fatalities attributed to insufficient terrain clearance indicate delayed decisions to eject. Emergency minimum altitudes for ejection are last resort altitudes and ARE NOT to be used as a basis for delaying the decision to eject. Ejection statistics continue to show that the MAJORITY of low altitude ejections (below 500 feet) terminate in fatalities, and that there is a progressive decrease in successful ejections as altitude decreases below 2000 feet. Do not confuse Emergency Minimum Altitudes for Ejection with the Recommended Minimum Altitudes for Ejection. Recommended Minimum Altitudes for Ejection are as follows:

a. Under level flight conditions, eject above 2000 feet if possible.

b. Under uncontrolled conditions, eject above 12,000 feet.

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). The zoom

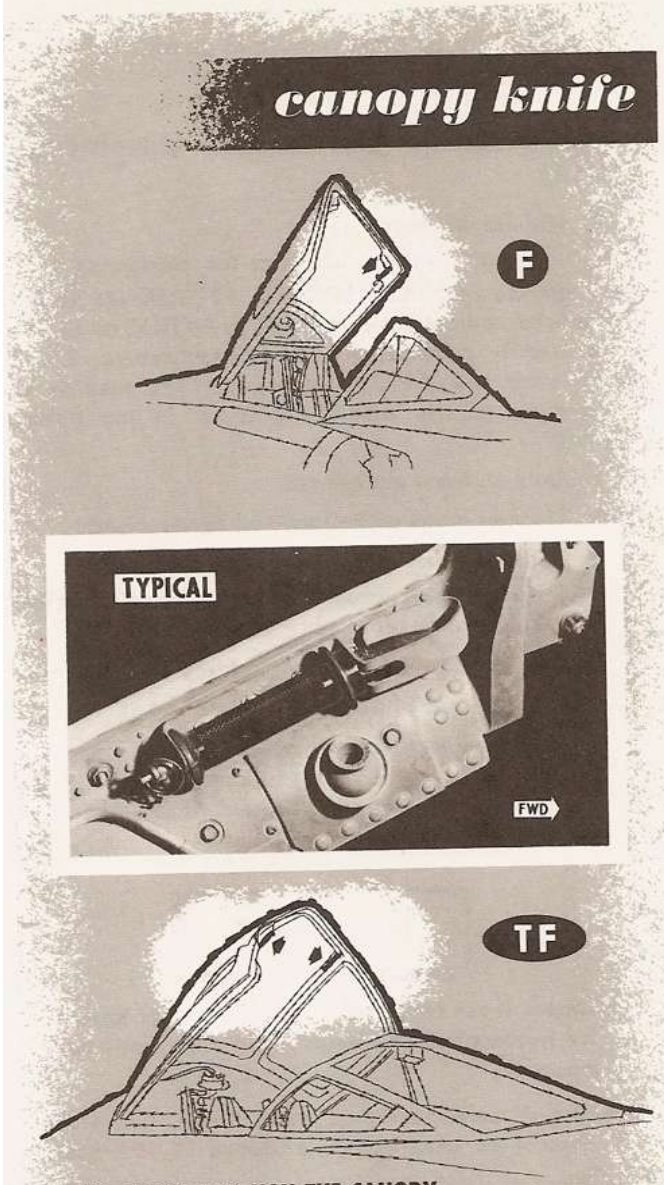


Figure 3-2

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 95 KCAS (90 KIAS) is recommended to assure rapid deployment of the chute.

WARNING

- **Under level flight conditions, eject at least 2000 feet above the terrain whenever possible. Do not delay ejection below 2000 feet in futile attempts to start the engine or for other reasons that may commit you to an unsafe ejection or a dangerous flameout landing. Statistics emphatically show a progressive decrease in successful ejections as altitude decreases below 2000 feet.**
- **Under uncontrolled conditions, eject at least 12,000 feet above the terrain whenever possible.**
- Eject at the lowest practical airspeed above 95 KCAS. (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 regardless of altitude. If the automatic seat belt is opened manually, the automatic opening feature of the parachute is eliminated and seat separation may be too rapid at high speeds. Care must be taken to ensure that flight clothing, such as sleeves, will not catch and release the lap belt during ejection.
- If possible, prior to ejection, 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 (wings level), if at all possible and use excess speed to gain altitude.

EJECTION SPEEDS

If possible, ejection should be made at a speed between 95 and 450 KCAS. The airplane should be slowed as much as possible if at high airpeeds, as forces on the body and injury hazard are decreased when airspeed is reduced. There is no need to quote maximum airspeeds for ejection. If the airplane is controllable, airspeed should be reduced to the lowest practical airspeed. This 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.

WARNING

Do not delay ejection if aircraft is in an uncontrolled condition.

EMERGENCY MINIMUM ALTITUDES FOR EJECTION (LEVEL FLIGHT)

In the event of a low-level emergency in which even zooming the airplane is not possible, it may be necessary to eject at or near the emergency minimum altitude. Normally, every attempt must be made to eject prior to descending below 2000 feet above the terrain. The emergency minimum altitude for ejection is based on ideal conditions of wings level and zero sink rate plus perfect operation of the ejection system. The emergency minimum altitude is listed in the following table:

EMERGENCY MINIMUM ALTITUDES FOR EJECTION (WINGS LEVEL AND ZERO SINK RATE)

HBU-2B/A Automatic Safety Belt, BA-22 pack modified with external pilot chute and ballistic spreader gun, DART/Seat Snubber, and C-9 Canopy
Zero Airspeed and Zero Altitude

WARNING

Emergency minimum altitude and airspeed figures for ejection 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 seat, etc. These figures are quoted only to show minimums 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 above terrain since accident statistics show a progressive decrease in successful ejections as altitude decreases below 2000 feet above terrain. Therefore, whenever possible, eject above 2000 feet above terrain. Since survival from an extremely low altitude depends primarily on the airplane attitude, airspeed, and altitude, the decision to eject under this condition must be left to the discretion of the pilot. Factors such as g-loads, high sink rate, and airplane attitudes other than level or slightly nose high decrease chances for survival.

BEFORE EJECTION (IF TIME AND CONDITIONS PERMIT)

1. IFF—Emergency.

   Canopy jettison activates the emergency IFF. On airplanes with AN/APX-6A, AN/APX-25 IFF, the IFF master control knob must be in NORM or LOW to automatically transmit the emergency return. On airplanes with AN/APX-72, emergency IFF may be activated automatically with the switches in any powered position.

2. Stow all loose equipment.
3. Tighten strap of helmet and lower visor.
4. Cabin air switch—RAM.

   To prevent explosive decompression at high altitude, place cabin air switch to RAM.

5. Aim airplane toward uninhabited impact area.
6. Throttle—OFF.
7. Sit erect, brace arms in armrests, hold upper arms and elbows tightly against body, head back hard against headrest with chin tucked in and feet back against the seat.

WARNING

- **Under level flight conditions, ejection should be accomplished above 2,000 feet whenever possible.**
- **Under uncontrolled conditions, ejection should be accomplished above 12,000 feet.**
- Eject at the lowest practical airspeed above 95 KCAS. (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.
- When ejecting at low altitude (below 2000 feet above terrain), pull the nose of the airplane above the horizon, if at all possible, and use excess speed to gain altitude.

EJECTION PROCEDURES

1. **HANDGRIPS—RAISE.**

   Raise either or both handgrips to jettison the canopy and expose the trigger(s).

2. **TRIGGERS—SQUEEZE.**

   Use of either or both triggers will complete the ejection sequence.

ALTERNATE CANOPY REMOVAL

If Canopy Fails to Jettison (F)

1. Canopy jettison handle—Raise.

If Canopy Is Still On (F)

1. Master switch—TRIP, then OFF.
2. Pull canopy latch handle full out.
3. Push canopy up; airstream will rip canopy from airplane.
4. Master switch—ON at pilot's discretion.

If Canopy Fails to Jettison (TF)

1. Canopy jettison handles—Raise.

If Canopy Is Still On (TF)

1. Pull canopy latch handle full out.
2. Push canopy up; airstream will rip canopy from airplane.

Note

- If the canopy ballistic jettison system fails and you are unable to open the canopy, hold the solenoid manual override handle firmly up against the stop (while holding the latch handle full out), and push the canopy up into the airstream.
- The solenoid manual override handle, located aft between the two seats, is spring-loaded to the horizontal position. The handle moves up approximately one inch of arc travel and must be held firmly up to override the solenoid and obtain canopy counterbalance.
- The canopy can be jettisoned from either seat.

AFTER EJECTION

On ejection, the seat and pilot will have a component of thrust provided by the airplane. The ejection seat stabilization and snubbing system, automatic opening belt, seat-to-seat separator, and parachute are timed to provide the most desirable sequence (figure 3-3) 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. As a backup, try to manually open the seat belt, push free of the seat by pushing and releasing the handgrips, and pull parachute ripcord AFTER ejection. This is only a precautionary measure since it will be impossible to "beat" the automatic opening features if the system functions properly. 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.

AFTER PARACHUTE DEPLOYMENT

Note

On airplanes modified by TCTO 1F-102-1209, the survival kit is automatically deployed. In the event the automatic deployment feature malfunctions, the survival kit may be manually deployed by pulling the multistage release handle.

After parachute is fully deployed and stabilized, release the survival kit by grasping the survival kit emergency release handle and pulling sharply up and back to separate the handle from the kit. This will suspend the life raft and survival kit below pilot and prevent injury on landing. 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. The kit should be released for all parachute landings except when it is evident that the landing will be in trees.

WARNING

- The survival kit emergency release handle must be pulled rapidly through the complete arc of travel to allow proper functioning of the survival kit system. Failure to follow this procedure could present a serious hazard to the aircrew during parachute landing.
- Do not actuate the survival kit emergency release handle until after parachute deployment to prevent the kit or lanyard from fouling the parachute.
- Do not actuate 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.

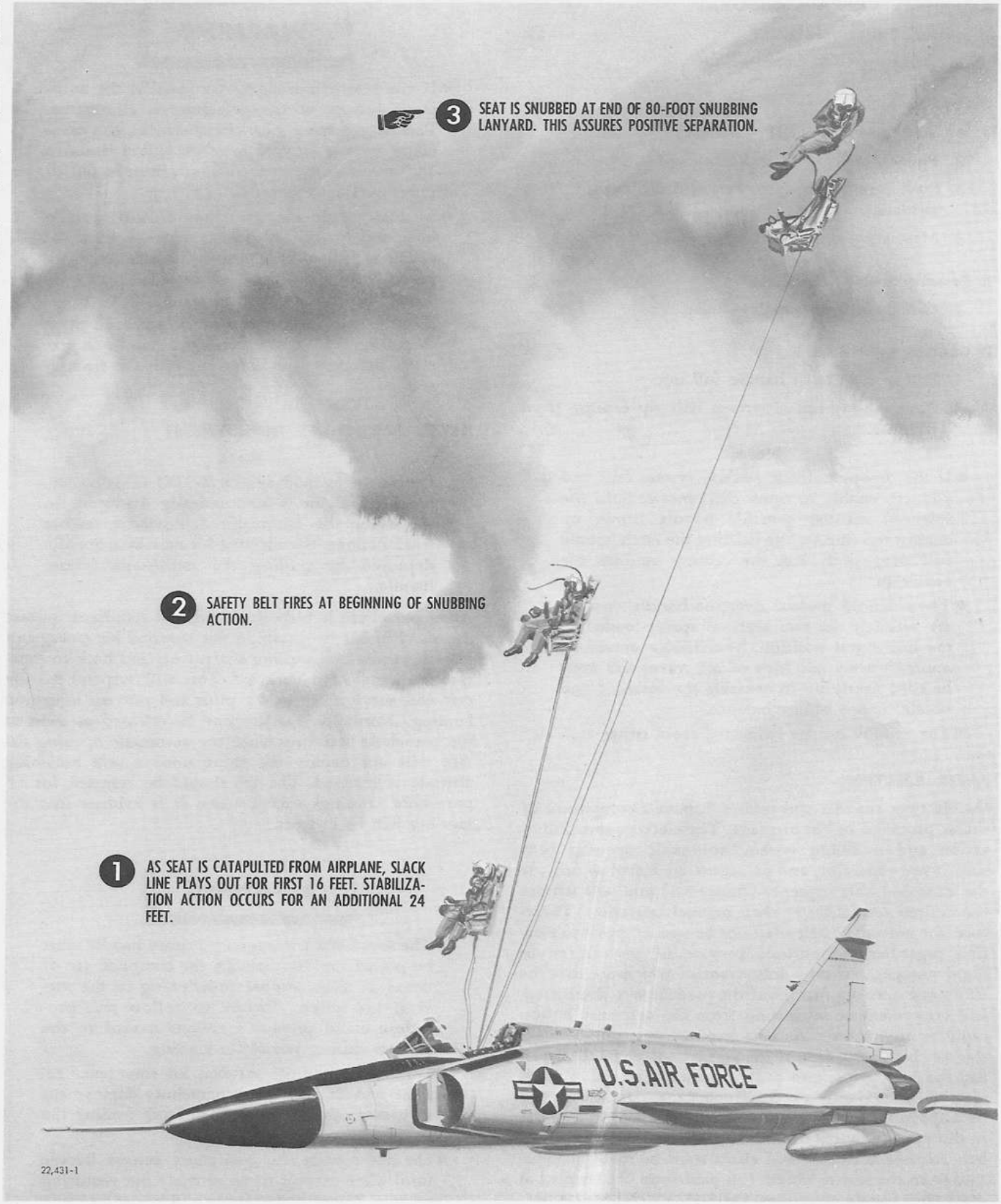
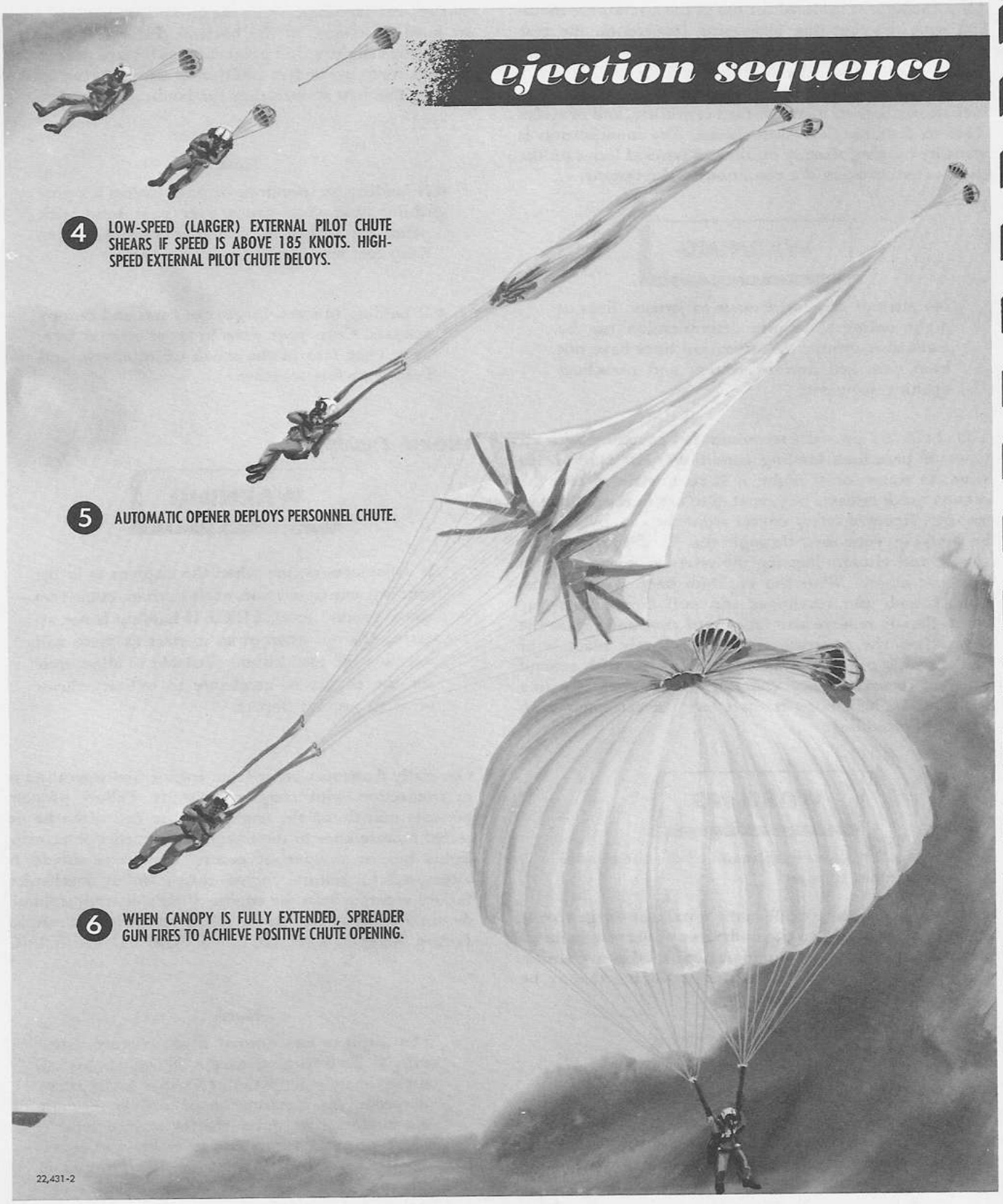

3 SEAT IS SNUBBED AT END OF 80-FOOT SNUBBING LANYARD. THIS ASSURES POSITIVE SEPARATION.
2 SAFETY BELT FIRES AT BEGINNING OF SNUBBING ACTION.
1 AS SEAT IS CATAPULTED FROM AIRPLANE, SLACK LINE PLAYS OUT FOR FIRST 16 FEET. STABILIZATION ACTION OCCURS FOR AN ADDITIONAL 24 FEET.
U.S. AIR FORCE
22,431-1

Figure 3-3

ejection sequence



4 LOW-SPEED (LARGER) EXTERNAL PILOT CHUTE SHEARS IF SPEED IS ABOVE 185 KNOTS. HIGH-SPEED EXTERNAL PILOT CHUTE DELOYS.

5 AUTOMATIC OPENER DEPLOYS PERSONNEL CHUTE.

6 WHEN CANOPY IS FULLY EXTENDED, SPREADER GUN FIRES TO ACHIEVE POSITIVE CHUTE OPENING.

22,431-2

All parachutes authorized for use in the aircraft are modified with the four-line jettisoning lanyard on the rear risers. This modification may be used at any altitude and during oscillations. The four-line modification forms a scallop in the rear of the canopy skirt which significantly reduces oscillation, increases turn capability, and provides three to four knots forward speed. The modification is made by tugging sharply on the red lanyard loops on the risers after checking the condition of the canopy.

WARNING

No attempt should be made to jettison lines at night unless a definite determination can be made that canopy or suspension lines have not been damaged during ejection and parachute opening sequence.

T.O. 14D1-2-1 provides recommended procedures for all types of parachute landing conditions. For landings in wind, in water, or at night, it is recommended that the canopy quick releases be prepared to use in the following manner. Remove safety covers from the canopy releases by hooking your arm through the V of the riser, fist closed, and visually locating the safety cover underneath the bent elbow. With the opposite hand, reach across using thumb and forefinger and pull down the safety cover. Slowly remove arm from V of riser and repeat the process on the opposite side. Immediately place your hands on the canopy releases. Upon contact with ground or water (actual physical contact) operate releases with a sharp jerk. The canopy releases have an extremely high degree of reliability.

WARNING

Once safety covers are removed do not remove hands from releases.

If the landing is to be made in the water, the oxygen mask should be removed and the underarm preservers inflated. After a water landing, entanglement in the suspension lines is a possibility. It is recommended the canopy be discarded rather than retained.

WARNING

When the feet touch the water, immediately operate canopy releases to spill the parachute. In any water landing, altitude is difficult to determine. Do not release the canopy until the feet touch the water, no matter how close you think you are.

For ground landings, watch the ground at a shallow angle or keep your eyes on the horizon. Place feet together, bend knees slightly, and point the toes toward the ground. Do not force leg or feet position as tenseness can lead to injury. The best preparation for landing is to relax.

Note

- If landing on telephone or power wires is a possibility, put your hands over your head with palms flat against the inside of the front risers. Keep feet together.

- If landing in trees, forget the risers and canopy releases. Cross your arms in front of your face, bury your face in the crook of an elbow, and keep your feet together.

ENGINE FAILURE

WARNING

If a flameout occurs when the airplane is in the landing configuration at less than 2000 feet above ground level, EJECT if landing is not assured. Do not attempt an airstart as there will not be sufficient altitude available to allow time for the engine to accelerate to military thrust prior to ground impact.

Generally flameouts occur from engine fuel starvation or in connection with compressor stalls. 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. Pilot's discretion should determine the action to be taken following an engine failure. Refer to EJECTION VS FORCED LANDING.

Note

The airplane has normal flight characteristics with a dead engine except above speeds of approximately 220 KCAS (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 speed and is only slightly perceptible at best glide speed of 220 KCAS (KIAS).

ENGINE FAILURE DURING FLIGHT AT LOW ALTITUDE

Note

An immediate air start can be attempted regardless of airspeed or altitude by moving the throttle inboard if in afterburner, placing the fuel control switch to emergency, and depressing the engine ignition button.

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 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.

Note

- EGT should be monitored during airstarts to preclude possible engine overtemperature.
- An immediate air start can be attempted regardless of airspeed or altitude by moving the throttle inboard if in afterburner, placing the fuel control switch to emergency, and depressing the engine ignition button.

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 KCAS (KIAS) will provide this rpm between sea level and approximately 35,000 feet. Above this altitude, at 220 KCAS (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 KCAS (KIAS) be used for all air starts, regardless of altitude. The probability of a relight at 220 KCAS (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. For an engie air start, proceed as follows:

1. **THROTTLE—INBOARD.**
2. **FUEL CONTROL SWITCH—EMERGENCY.**

   Check that emergency fuel control warning light illuminates.

Note

Due to excessive altitude loss at other than best glide speed, 220 KCAS (KIAS) is recommended for all air starts. The probability of a relight at 220 KCAS (KIAS) increases below approximately 35,000 feet.

3. **THROTTLE AND IGNITION BUTTON—AS REQUIRED AND DEPRESS.**

WARNING

Engine overtemperature and compressor stalls can occur due to the introduction of excessive fuel if the throttle position does not approximately correspond to the engine rpm. Attempt to match throttle position with engine rpm and simultaneously depress ignition button.

4. RPM—60 to 80%, then release ignition button.
5. Throttle—OFF (relight not obtained).
6. AC bus switch—RESET, then ON.

   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 and to supply power for the emergency dc bus.

Note

Loss of main ac generator will cause loss of booster pump operation.

7. Fuel control panel—Check.
   Fuel shutoff valve switches—OPEN.
8. RPM—15 to 30%.

   Air starts should be made with engine windmilling between 15 and 30% rpm. An airspeed of 220 KCAS (KIAS) 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 ram air turbine.

9. Ignition button—Depress and hold.
10. Throttle—IDLE.

Note

Do not move throttle past IDLE position until rpm reaches 60 to 80%.

11. RPM—60 to 80%, then release ignition button.

AFTER RESTART

1. DC generator switch—RESET, then ON.
2. Radar master switch—OFF.
3. Booster pump switches—OFF.
   Turn booster pumps off to reduce the load on the ac generator.
4. NESA switch—OFF.
5. AC generator switch—RESET, then ON.
6. Booster pump switches—ON (one at a time).
7. NESA switch—NORMAL (some airplanes), ON (other airplanes).
8. Radar master switch—As required.
9. AC bus switch—NORMAL.

UNSUCCESSFUL AIR START

1. Throttle—OFF.
   Retard the throttle to OFF if during an air start any one of the following conditions prevail:
   a. Lightup 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. Starter circuit breaker—Check.
3. Attempt another air start (try both normal and emergency fuel control using ground start procedure).

Note

If dc power is lost, make subsequent start attempts below 35,000 feet.

   a. Throttle—START.
   b. Ignition button—Depress and hold.

Note

The ignition button must be held depressed until the starting sequence is complete to avoid breaking the ignition circuit. If the ignition button is inadvertently released, the entire starting sequence must be reaccomplished.

   c. Throttle—OFF, then IDLE.
   d. If engine instruments do not indicate normal engine operation, retard throttle to OFF and attempt another start.

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. The transformer-rectifier is connected to the emergency dc bus when dc generator output is lost. The emergency ac system is self-sustaining and will not be affected by battery power depletion.

- 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.

WARNING

During flameouts where RPM and fuel flow drop suddenly to zero and the fuel flow does not respond to throttle setting, a failure of the accessory drive train is strongly indicated. AC and DC power will be lost immediately and flight control stiffening will follow. Placing the battery switch to TR FAIL position under this condition will provide power for UHF position report or intercommunications. No attempt should be made to change channels on the command radio. If change of channels is attempted, a loss of communications will occur. The RAT must be extended for continued flight control operation. Repeated air start attempts will be futile and decision to make a flameout landing. or eject, should be made immediately.

Note

If engine rpm does not respond to throttle movement and rpm is above 70%, select afterburner to attempt to obtain sufficient thrust to recover the airplane. If rpm is below 70%, afterburner thrust may not be enough to maintain level flight.

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

MAXIMUM GLIDE

Maximum glide distances with a windmilling engine are obtained with an airspeed of 220 KCAS (KIAS) in a clean configuration. See figure 3-4 for additional information.

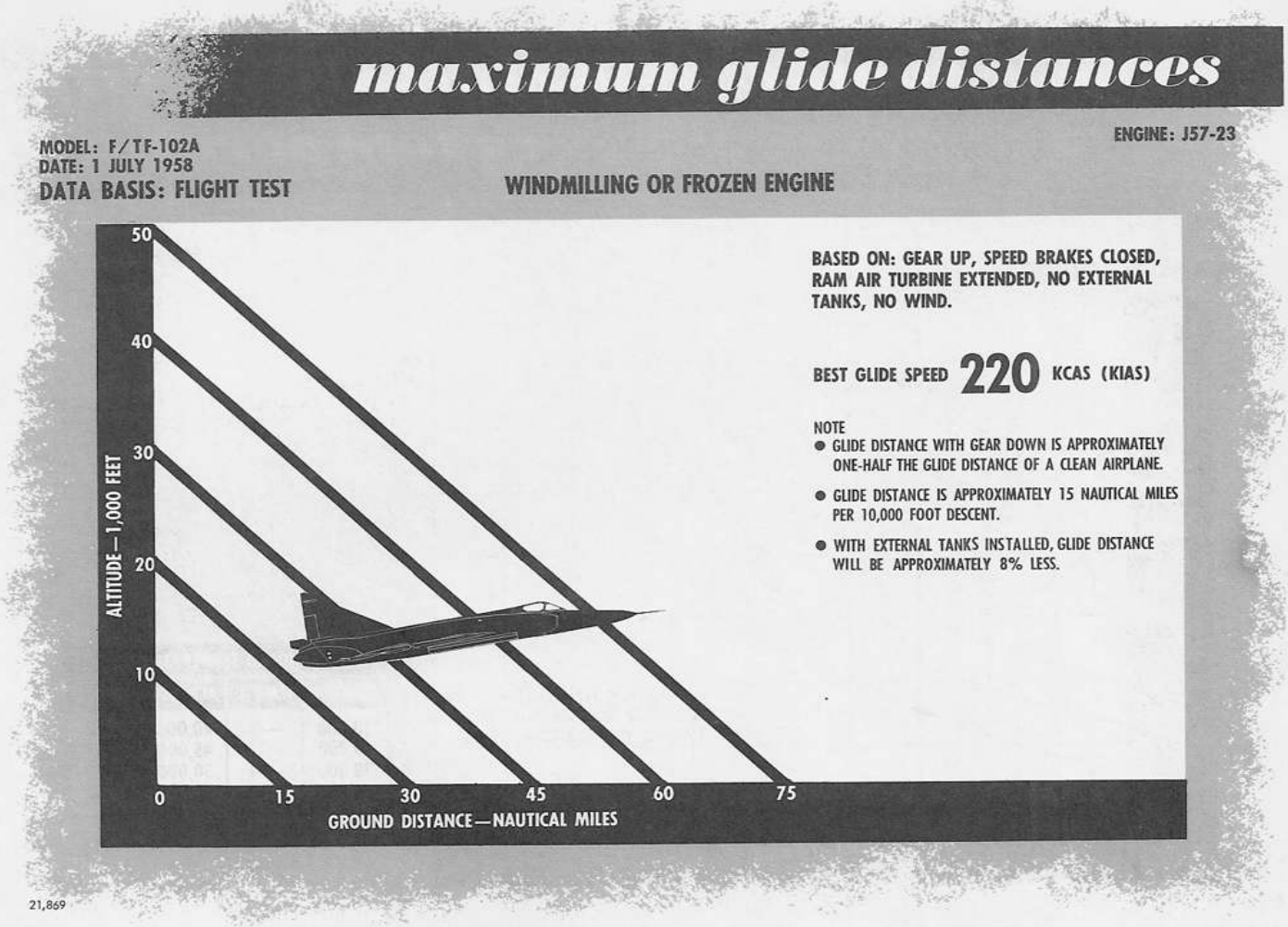


Figure 3-4

COMPRESSOR STALL

An undesirable but inherent characteristic of most air compressors, including those of the axial-flow multistage turbojet design, is that airflow instability may occur as a result of adverse compressor inlet or exit conditions. This characteristic is more pronounced in high-performance compressors of the type used in the J-57 engine. This unstable condition has been referred to as surge, pulsation, chugging, "choo-choo," or explosions, but is usually described as compressor stall in a turbojet engine because it results from separation of airflow from the surfaces of the compressor blades just as separation of airflow from wing surface results in an airplane stall. Compressor stalls may vary in severity, and may occur momentarily or be cyclic. Compressor stalls may be anything from the first indication of an impending engine failure to a simple breakdown of airflow across the compressor face. Compressor stalls will be accompanied by one, or any combination, of the following characteristics:

a. Noise. The noise associated with compressor stalls may take the form of an individual loud report or a series of loud reports. However, it is possible to have a compressor stall without noise.
b. Engine instruments. Engine instrument indications of compressor stall may include reduction or fluctuation of engine presure ratio at a constant throttle setting, rise in exhaust gas temperature, and failure of rpm to increase with an increased throttle setting.
c. Compressor pulsations felt throughout the airplane.
d. Emission of vapor from engine air intake duct.
e. Flame may be seen coming out of the bleed air door on the side of fuselage or out of the intake ducts when a severe compressor stall is experienced.

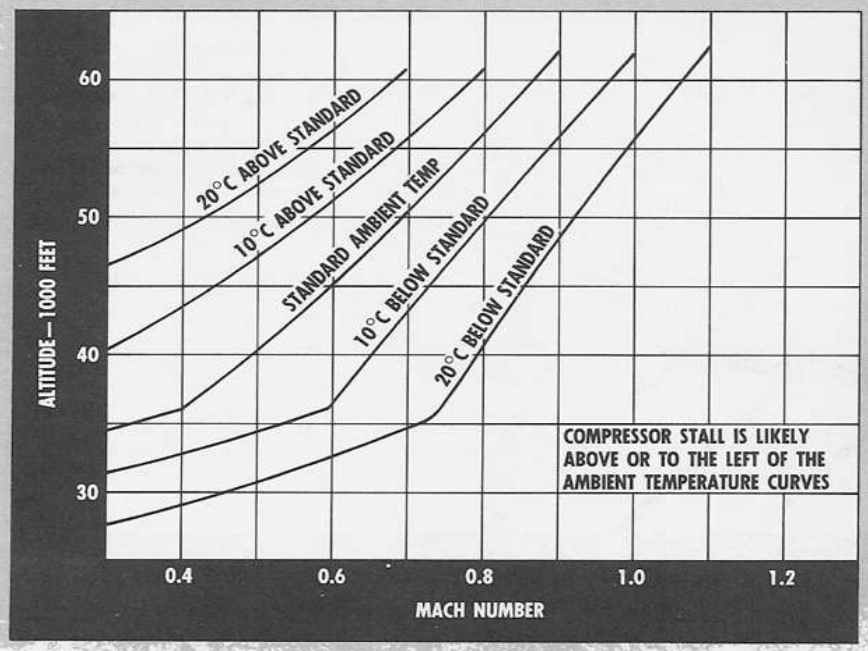
WARNING

When temperature limits are exceeded, heat damage to the engine can occur rapidly and may result in engine failure.

compressor stall sensitivity vs temperature

MODEL: F-102A
DATE: JULY 1970
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
FUEL GRADE: JP-4



NOTE
- CLEAR AIR
- VALID AT THROTTLE SETTINGS ABOVE MINIMUM AFTERBURNER STOP WITH THROTTLE INBOARD OR OUTBOARD

STANDARD TEMPERATURES			
ALTITUDE	TEMP°C	ALTITUDE	TEMP°C
20,000	—25	40,000	—57
25,000	—35	45,000	—57
30,000	—44	50,000	—57
35,000	—54	55,000	—57

22,428

Figure 3-5

Note

- The engine can sustain numerous rapid-order compressor stalls without engine damage, provided exhaust gas temperature limits are not exceeded.
- Record on AFTO Form 781 any compressor stall and indicate duration and peak temperatures of any overtemperature operation.

Each of the following factors contributes to narrowing the stall margin of the J-57 engine and makes it susceptible to compressor stalls. A combination of the factors further increases the probability of compressor stalls, especially at high altitude.

a. Weather phenomena that are conducive to compressor stalls are:

(1) Abnormally low outside air temperature at high altitude. Compressor stall sensitivity as a function of temperature is shown in figure 3-5. This information can be used for flight planning to avoid areas where compressor stalls are likely.

(2) Flying in moisture, ice crystals, turbulence, and in or near thunderstorms.

b. A severe stall or a series stalls may be the result of one, or a combination, of the following malfunctions:

(1) Excessive fuel scheduling by the fuel control unit can cause stalls during engine acceleration.

(2) Unsatisfactory operation of the intercompressor bleed valve may result in stalls during engine acceleration or deceleration.

(3) If the exhaust nozzle is slow-acting or will not open, an extremely violent compressor stall may occur when afterburner is selected or terminated.

c. Compressor stalls are sometimes induced during normal operations as follows:

(1) Low airspeed maneuvering may induce compressor stall because of distortion of intake duct airflow. During spins, continuous stalls will probably be encountered with a steady increase in exhaust temperature because of severe engine airflow distortion.

(2) During high-altitude, low-airspeed flight conditions, the engine compressor is operating closer to the stall region because low-density air has a greater tendency to separate from the compressor blades. Consequently, engine acceleration stalls may be induced at high altitude with low airspeeds.

(3) During high-altitude air starts, mild compressor stalls may occur as the engine accelerates to idle. If persistent stalls occur, they may be eliminated by descending to increase airspeed.

(4) Erratic throttle movements can induce compressor stalls. An example of this is when the throttle is retarded and then advanced while the engine is still decelerating.

(5) The emergency fuel control system does not provide automatic fuel scheduling to meet engine acceleration requirements nor compencate for altitude variations above 30,000 feet. Consequently, compressor stalls may be experienced whenever rapid throttle movements are made using the emergency fuel system.

COMPRESSOR STALL PROCEDURES

1. **THROTTLE—INBOARD; IGNITION BUTTON—DEPRESS AND HOLD; EGT GAGE—MONITOR.**

   At first indication of compressor stall, simultaneously shut off afterburner, depress and hold ignition button, and monitor exhaust gas temperature gage for overtemperature.

CAUTION

Continuous use of the ignition system in excess of ten minutes, or consecutive 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 stalls, but may aid in preventing flameout.

2. **THROTTLE—RAPIDLY TO IDLE** (if stall cannot be broken or EGT reaches 670°C).

3. **THROTTLE—OFF** (if stall cannot be broken or EGT reaches 670°C).

Note

With the throttle in the OFF position, duct rumble will be present and should not be interpreted as a compressor stall.

4. **AC BUS SWITCH—RESET, THEN ON.**

Note

- Engine RPM below 40% will be insufficient to provide normal ac generator operation.
- Engine windmilling RPM at 220 KCAS (KIAS) will be insufficient to provide normal ac generator operation and may not be sufficient at 250 to 275 KCAS (KIAS).
- Reducing altitude and increasing airspeed will improve compressor stall recovery potential or will possibly preclude re-entering a stall.

5. Make an air start.

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

Note

Past experience indicates that an immediate air-start attempt may not be successful because the engine may still be in a stalled condition.

6. AFTO Form 781—Record any compressor stall.

   Record any compressor stall and indicate duration and peak temperatures of any overtemperature operation.

FIRE

A steady illumination of the fire warning light is an indication of a fire in the forward engine compartment. A flashing 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 90% rpm) 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.

ENGINE FIRE IN FLIGHT

1. **THROTTLE—MINIMUM PRACTICABLE THRUST.**

   Reduce thrust to minimum necesary 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 FIRE EXISTS—EJECT.**
   Check for positive indication of fire, such as abnormal engine instrument readings, trailing smoke or flame, smoke in the cockpit, or request a visual check from the ground or another airplane.
3. If warning light extinguishes:
   a. Check warning light.
   b. Continue flight at minimum safe thrust.
   c. Land as soon as practicable.
4. If warning light remains on and fire not evident by check:
   a. Continue flight at minimum safe thrust.
   b. Land as soon as possible.

EXHAUST GAS OVERTEMPERATURE WARNING LIGHT ILLUMINATED DURING FLIGHT

1. Thrust—Reduce (if practical).
   The exhaust gas overtemperature warning light is set to illuminate at 665°C. A reduction in thrust should extinguish the light.
2. If the light extinguishes—Continue mission and monitor instruments.
   Monitor the exhaust gas temperature and other engine instruments during the remainder of the mission.
3. If the light stays on—Monitor exhaust gas temperature and land as soon as practicable.

Note

Illumination of the exhaust gas overtemperature warning light in flight could be caused by several factors. Among these are engine fuel control failure, afterburner exhaust nozzle failure, compressor stall, or improper throttle management under certain flight conditions.

HOT AIR BLEED LINE/DUCT FAILURE

Problems detected in the air conditioning and pressurization systems, electronic cooling, and electrical system (when persistent), or any combination of these problems, may be regarded as sound indications of possible hot air bleed line/duct failure. Some symptoms of these problems may be a mild explosion, smoke and/or fumes in the cockpit, increasing cockpit temperature, cockpit pressure fluctuations, cycling of electronic cooling warning light, loss of UHF radio, and vibration of rudder pedals. When hot air bleed line/duct failure is suspected on the ground, the flight should be aborted and the engine shut down. When inflight failure is suspected, this condition should be treated as an emergency requiring minimum power and landing as soon as possible. In a malfunction caused by rupture of the hot air bleed line, continued flight could be extremely hazardous. If hot air bleed line/duct failure is suspected:

1. Cabin air switch—RAM (below 25,000 feet if possible).
2. Thrust—Minimum for recovery.
3. Jettison canopy if smoke and/or fumes become severe.
4. Land as soon as possible.

WARNING

After landing, when clear of the runway, open the canopy, shut down the engine, and abandon the airplane. Do not turn off any electrical switches to preclude the possibility of explosion of fuel fumes.

ELECTRONIC COOLING WARNING LIGHT ILLUMINATED

If electronic cooling warning light illuminates and remains on:

1. Prior to takeoff:
   a. Radar—OFF.
   b. Abort.
2. After takeoff:
   a. Thrust—Minimum for recovery.
   b. Radar—OFF.
   c. Land as soon as practicable.

Refer to ELECTRONIC COOLING WARNING LIGHT, Section IV.

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, proceed as follows:

1. AC bus switch—RESET, then ON.
   While energizing the emergency ac generator, check ac voltage for a momentary drop followed by a buildup indicating that emergency ac power is available.
2. AC and dc generator switches—OFF (if not essential for flight).

Note

If emergency as power is not available, and tl flight instruments are essential for IFR flight, do not turn the ac generator OFF. Return the ac bus switch to the NOR position.

3. All equipment not necessary for flight—OFF.
4. If fire continues, master switch—TRIP, then OFF.

Note

It will be necessary to turn the master switch on long enough to facilitate normal landing gear and speed brakes operation if landing is to be made and drag chute operation is desired.

5. If fire persists and becomes severe—eject.
6. Land as soon as possible if fire subsides.

SMOKE, FUMES, OR FOG ELIMINATION

Note

Any unidentified fumes shall be considered toxic.

Should smoke, fumes, or fog enter the cockpit, proceed as follows:

1. Cabin air switch—RAM (below 25,000 feet if possible).
2. Jettison canopy if smoke becomes severe.

WARNING

If the canopy is jettisoned below 180 KCAS while airborne, it may strike the fin but probably will not result in loss of control.

SPIN RECOVERY

WARNING

If a spin occurs below 12,000 feet above the ground — EJECT. Do not attempt to recover the airplane.

1. **CONTROLS—NEUTRALIZE.**
   a. Emergency damper disconnect button—Depress.

WARNING

- The flight controls must be held in the neutral position to prevent aileron movement due to inertial forces acting on the control stick.
- Any aileron displacement may aggravate the spin or post-stall gyration.
- If in a spin or post-stall gyration, maintain the neutral position until the direction of rotation can be definitely established by use of the turn needle.

2. **THROTTLE—IDLE.**
   Retarding the throttle will reduce the possibility of engine overtemp and compressor stall.
3. **AIRPLANE—CLEAN CONFIGURATION.**
   a. Retract landing gear.
   b. Jettison external loads.

WARNING

Failure to jettison external tanks severely reduces the possibility of recovery.

   c. Close armament bay doors.
   d. Close speed brakes.
4. **APPLY ONE-HALF AILERON IN THE DIRECTION OF ROTATION** (after direction of rotation is definitely established).

Note

- Definitely establish direction of rotation visually or by reference to the turn needle and apply aileron in the same direction as the turn needle.
- 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.

5. **CONTROLS—NEUTRALIZE (WHEN SPIN IS BROKEN).**
   Begin a dive recovery only when airspeed has reached 135 KCAS (KIAS).

WARNING

If spin rotation has not been stopped by 12,000 feet above the terrain, EJECT.

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 AFTO Form 781 so that the engine can be inspected for damage. Refer to STALLS AND SPINS, Section VI.

ARMAMENT JETTISON

The TRIGGER SALVO selection is provided to insure jettisoning of all armament in an emergency. The aft bay missile is jettisoned prior to the forward missile. If the missile is being prepared for launch by the AWCS and TRIGGER SALVO is selected, missile preparation is immediately disrupted and discontinued. Although the missile battery cannot be deactuated once activated, the internal power is shut off, and the unlocking command

signals are removed, if present. If a missile misfires during salvo, the misfire warning light will not illuminate because the source of power to the light is removed when TRIGGER SALVO is selected. Since normal attack preparations are also bypassed during TRIGGER SALVO, the firing cycle will continue as if no malfunction had occurred. To salvo the armament, use the following procedure:

1. Armament selector switch—TRIGGER SALVO.
2. Armament safety switch—ARMED.
3. Igniter control switch—ON (some airplanes); TACTICAL (other airplanes).
4. Special weapon switch—DISARM.
5. Armament triger—Depress to second detent and hold.

   When the trigger is depressed, the missile bay doors open, the armament is jettisoned, and the doors close. The trigger must be depressed throughout the cycle.

AFTERBURNER FAILURE DURING FLIGHT

1. Throttle—Inboard.

   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 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.
- If throttle motion does not cut off the afterburner, the afterburner control circuit breaker may be pulled to terminate afterburning.

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) and/or the oil pressure gage. Illumination of the light or loss of oil pressure as indicated by the oil pressure gage 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, proceed as follows:

1. Throttle—Set optimum thrust for Recovery.

Note

- Once the throttle is set for optimum thrust for recovery, additional throttle movement could result in the engine freezing. Subsequent throttle movement should be used only in the interest of safety.
- If it is necessary to use power beyond the minimum afterburner throttle stop, select afterburner rather than a higher rpm setting. This will reduce engine temperatures and pressures and will result in longer engine operation than will the use of a higher rpm.

2. Airspeed—345 KCAS (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.

WARNING

- In any emergency or precautionary landing, where it is likely or possible that the engine may fail or "freeze," extend the RAT on the base leg or well up on final if making a straight-in approach.
- If the secondary hydraulic system has failed, do not extend the RAT except with an actual engine failure. The pilot must be prepared to extend the RAT immediately when the engine fails.

3. External tanks—Jettison (if necessary).
4. Avoid high g maneuvers.

5. If engine vibrations become excessive during flight, attempt to reduce vibrations with a change of thrust.
6. If excessive vibrations continue, shut down the 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.

7. Land as soon as possible.

   A flameout pattern may 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.

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.

ENGINE FUEL CONTROL FAILURE

Engine fuel control failure may be recognized by a drop in exhaust gas temperature, rpm, fuel flow, and failure of rpm to respond to throttle movement.

Note

- Failure may also be evidenced by rapid increases in exhaust gas temperature and fuel flow. During emergency conditions, switching from NORMAL to EMER may be accomplished at any throttle position. Recovery using the emergency system should then be made. In the event emergency conditions warrant return to the normal fuel control, the throttle must be at IDLE prior to positioning the fuel control switch to NORMAL to prevent pressure surges.
- Refer to FUEL FLOW FLUCTUATIONS, Section V.

If fuel control system fails during takeoff, 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 excess fuel introduction if throttle position does not approximately correspond to engine rpm prior to emergency fuel system selection.

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. If the afterburner is operating at the time of the engine stage failure, afterburner operation will terminate automatically and the exhaust nozzles will close. The throttle should be placed inboard. Land as soon as possible.

COMPLETE FUEL BOOSTER PUMP FAILURE

If complete booster 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 tanks at takeoff. With complete fuel booster pump failure, observe the following:

WARNING

With reduced power settings and high rates of descent, engine instruments should be closely monitored to detect loss of engine thrust.

1. Avoid negative g maneuvers.

2. Operate in a nose-high attitude (normal traffic pattern attitude is nose high).
3. Avoid uncoordinated maneuvers.
4. Avoid rapid decelerations (combinations of large thrust changes, speed brakes, and landing gear extensions).
5. Land as soon as practicable.

The above steps are to prevent uncovering the aft fuel intakes. A complete fuel booster pump failure will occur with loss of ac nonessential or dc essential power.

FUEL BOOSTER PRESSURE-LOW WARNING

Fuel booster pressure L or R warning light will illuminate under the following conditions:

- An empty No. 3 tank.
- A dual failure of booster pumps on one side.
- Fuel shutoff valve not opened. (An electrical interlock makes booster pump operation dependent upon a fully open shutoff valve.)
- Loss of ac nonessential power.
- Fuel selector right and left circuit breakers out.

Note

Boost pumps will be inoperative with loss of dc essential power and the warning lights will be inoperative.

If the fuel booster pressure-low warning light illuminates in flight, proceed as follows:

1. Plan flight for one-half the indicated fuel being available.

   If the warning is due to a fuel valve which is not fully open, initial action should be based on the assumption that only one-half of the total fuel on board will be available.
2. Fuel shutoff valve switches—OPEN.
3. Fuel selector circuit breakers—Check.
4. 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 moving the appropriate fuel shutoff valve switch to CLOSE position.

WARNING

- When the left No. 3 tank is empty, the total fuel aboard may be 200 to 300 pounds less than indicated on the fuel quantity gage.
- If operation with a known empty tank is required, the fuel shutoff valve on the empty side must 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. (Refer to FUEL BOOSTER PUMP LOW FUEL LEVEL OPERATION.)

   b. If the No. 3 tank indicates fuel remaining:
      (1) Fuel shutoff valve switches—OPEN.
      (2) Failed booster 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 booster 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 BOOSTER PUMP FAILURE paragraph above. Prior to penetration, turn good booster pumps ON.

FUEL BOOSTER 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 booster 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 booster 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 booster pump low-pressure light, the fuel valve to the empty side will be closed.

Note

Momentary illumination of the fuel low level warning light may occur during negative-g maneuvers or rapid maneuvering flight.

In this condition, the concern is with the forward pump and "T" check valve on the side supplying fuel. In flight, booster pump failures are few but unawareness of probable consequences is a hazard. A booster 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. Fuel booster pumps—Check.

   Whenever a penetration is made with a comparatively low fuel level in the No. 3 tanks, 570 pounds or less per tank, then the positive operation of the forward pumps should be established. If the aft booster pumps are momentarily deactivated and the booster pump low-pressure warning light does not illuminate, this is an indication of proper functioning of respective forward booster pump.

2. If either forward pump is not available—Maintain nose-high attitude.

FUEL FLOW EQUALIZER FAILURE

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 BOOSTER PRESSURE-LOW WARNING. To preclude the above, fuel loading may be balanced as follows:

1. Fuel shutoff valve switches—OPEN.
2. Fuel quantity—Check (select each position on the fuel quantity selector switch).

   If 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. Booster pump switches—OFF on the low side. This will permit high rates of flow from the high side.
4. Booster 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.

WARNING

In the event fuel will not feed from the high side, a fuel system icing condition may be indicated. A fluctuating fuel flow may or may not be observed. All booster pumps should be turned on and a high power setting, preferably at 90% rpm or above, should be maintained for the remainder of the flight. Land as soon as practicable.

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 booster 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 booster 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 tank. It is possible to monitor the No. 3 tanks and determine if fuel is feeding from the other two tanks without tank pressurization. The flight should be adjusted accordingly in event that only No. 3 tank is feeding the engine. Illumination of the fuel tank pressure-low warning light may occur during rapid descent. If the warning light illuminates steadily, proceed as follows:

1. Depend on No. 3 tank fuel only.
2. Throttle—85% or above; check light.
   If the light remains on, accomplish the following steps:
3. Cabin air switch and cabin air circuit breaker—Check.
4. Monitor low level warning lights and fuel quantity in each wing.
5. Do not use afterburner.
6. Land as soon as practicable.

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 5 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.

A dc generator failure followed by an ac generator failure results in loss of all electrical power (except battery). 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 (5 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:

WARNING

AC power failure can be an indication of loss of engine oil. Either an illumination of the oil pressure low warning light or a low oil pressure gage reading following an ac power failure should be interpreted as a depletion of engine oil. Follow ENGINE OIL SYSTEM FAILURE procedure and land as soon as possible.

1. **AC BUS SWITCH—RESET, THEN ON.**
2. FCS master switch—OFF.
3. Booster pump switches—OFF.
4. Nesa switch—OFF.
5. Rain clear switch—RAINCLEAR.

Note

If windshield fogging is not anticipated, the rainclear switch can be turned OFF.

6. AC generator switch—RESET, then ON.

Note

If generator will not reset, check circuit breakers.

If warning light extinguishes:

1. Booster pump switches—ON (one at a time).
2. Nesa switch—NORMAL (some airplanes); ON (other airplanes).
3. FCS master switch—As required.
4. AC bus switch—NOR.
5. Rainclear switch—As required.
6. Dampers—On.

If warning light remains on:

1. AC generator switch—OFF.
2. Land as soon as practicable.
3. Refer to FUEL BOOSTER PUMP FAILURE. With the loss of the ac generator, the fuel booster pumps will be inoperative.

Note

In the event of main ac generator failure TACAN will not be available. However, the following primary flight aids, in addition to the pitot-static instruments, will be operable:

All gyro flight instruments
Gyro compass
Interphone, UHF command, and marker beacon radio
IFF radar

Note

If both ac generators are inoperative, the following primary equipment will be inoperative:

Primary attitude indicator
Fire control system
Fuel booster pumps
Fuel flow indicator
Fuel quantity gage
IFF system
Hydraulic pressure gage
RMI
Oil pressure gage
Engine pressure ratio gage.
ILS glide slope

Refer to Section I for detailed list of electrically operated equipment.

DC GENERATOR FAILURE

A dc voltage monitor causes the dc power failure warning light to illuminate if dc generator output drops below 25 (±0.5) volts. The battery will continue to power all three dc busses at some level of reduced efficiency. Eventual depletion of the battery will result in complete electrical failure. This failure results in gradual reduction in performance of some and/or all dc powered components plus cabin temperature full-hot, yaw oscillations, UHF radio cycling failure. Individually, these symptoms are not sufficient to indicate power loss; occurring in conjunction, they are a most positive indication of dc generator failure. If the generator shaft has sheared, the generator will not reset when the dc generator switch is placed in the reset position and the dc power failure light will illuminate. In the event of failure, use the following procedure:

1. DC generator switch—Reset, then on.
2. If warning light remains on, ac bus switch—Reset, then on.

   Accomplish this step to energize the emergency dc bus with the emergency ac generator to preclude complete electrical failure in event of main ac generator failure after battery depletion.

Note

- If unable to reset the ac bus, check dc power control and emergency ac power control circuit breakers. The dc power control circuit breaker controls the RESET position of the ac bus switch, and the emergency ac power control circuit breaker controls the ON position.

- The instrument lights may dim when the dc generator fails.

Note

UHF radio may drop off the line for 20 to 30 seconds following resetting of the emergency ac bus. As long as the ac voltmeter shows proper operation of the emergency generator, use of battery TR FAIL position to regain UHF should be delayed for at least 45 to 60 seconds.

3. DC generator switch—OFF.
4. Turn off all equipment not needed.
5. Battery switch—As required.

Note

- If dc essential bus power is not required, the battery may be turned off to conserve battery power for the landing phase of flight. The emergency dc bus will be powered by the TR if the ac bus switch has been reset. Warning lights will not be available with the battery switch off.
- 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 5 to 20 minutes.
- Because of reduced electrical power, use emergency landing gear lowering procedures unless positive normal extension can be confirmed.

6. Land as soon as practicable.

Note

- Should subsequent emergency ac generator failure occur, return the ac bus switch to NOR, place the battery switch to T/R FAIL, reset the main ac generator, and place the battery switch back to ON.
- After failure of the dc generator, the following primary flight aids will be available until battery power depletion:

  Heading indicator
  IFF radar.

Note

With dc generator and battery failure (TR activated), the following primary equipment will be inoperative:

AFCS
Afterburner control
External tank jettison
Fuel control switch
Heading indicator
Hot-section analyzer emergency inverter
IFF
Inflight ignition
Landing gear normal extension
Landing gear position indicators
Pitot heat
Standby attitude indicator
Warning lights
Windshield rain clearing control
Elevon and rudder trim
EGT gage
Boost pump control.

Note

With complete dc power failure and TR failed, complete electrical failure will occur. Refer to Section I for a detailed list of electrically operated equipment.

AC AND DC GENERATOR FAILURE

1. **AC BUS SWITCH—RESET, THEN ON.**
2. If both normal ac and dc generators fail, follow the same procedures prescribed for individual generator failure.

Note

Failure of the TR unit followed by dc generator failure will also cause the ac and dc power failure warning lights to illuminate. If the essential ac bus drops off after releasing the ac bus switch from RESET to ON, place the battery switch to T/R FAIL. Either or both ac generators may then be turned on. The emergency ac generator may be operating in an overloaded condition. Therefore, all equipment not needed should be turned off.

Emergency AC generator Or Transformer-rectifier Failure

1. Turn off all unnecessary electrical equipment.
2. Battery switch—TR FAIL.

Note

Placing the battery switch in TR FAIL position connects the dc emergency bus to dc essential bus (battery). When operating from battery only, the UHF will not function after approximately five minutes due to lowering of battery power. No attempt should be made to change channels on the command radio. If change in channels is attempted, a loss of communications will occur. When the UHF stops functioning, it should be shut off to conserve battery power.

3. Proceed with ac and dc generator failure procedure.

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, oil pressure gage, fuel flow indicator, heading indicator, engine pressure ratio gage and primary 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.

FAILURE OF ONE HYDRAULIC SYSTEM

Note

It should be noted on AFTO Form 781 if the RAT has been extended in flight.

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. 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. Dampers—OFF.
3. 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.

4. 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.

5. Land as soon as practicable.
6. If primary hydraulic system has failed, RAT handle—Pull (just after turn on final approach) and hold for four seconds.

   Hold the handle in fully extended position for four seconds to insure a satisfactory extension of the RAT.

WARNING

In any emergency or precautionary landing, where it is likely or possible that the engine may fail or "freeze," extend the RAT just after turn on final or at final approach turn altitude, if making a straight-in approach.

7. If secondary hydraulic system has failed:
   a. Emergency landing gear extension handle—Pull (in landing pattern).
      If secondary hydraulic system has failed, use the emergency gear extension system to extend gear. Use the normal gear extension if primary hydraulic system has failed.
   b. Drag chute handle—Pull, rotate 90° then pull fully out (after touchdown).

WARNING

- If secondary hydraulic system failure has been caused by a known malfunction in the speed brake system, asymmetrical speed brake extension may occur if the drag chute is deployed. Therefore, a no-drag-chute landing should be planned if runway conditions, length, and barrier availability permit. If use of drag chute is necessary, nose wheel should be lowered to the runway prior to deploying the drag chute.
- If the secondary hydraulic system has failed, do not extend the RAT except with an actual engine failure. Be prepared to extend the RAT immediately when the engine fails.

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 and the hydraulic pressure gage indicates double failure, use the following procedure:

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.

1. **RAT—EXTEND.**

   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. Accomplish deceleration by moving the flight controls a minimum amount to establish a flight attitude that will provide deceleration.

2. **IF FLIGHT CONTROL OPERATION IS NOT POSSIBLE—EJECT.**

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 hydraulic system failure is the result of a frozen engine, the hydraulic pressure gage will be inoperative due to complete ac power failure, and 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 warning light will be the only instrument for indication of hydraulic pressure.

4. If flight control is still possible but the pressure gage indicates both systems are below 1000 psi and the hydraulic failure warning light is still illuminated steadily—Eject.

5. 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 assures adequate hydraulic pressure for normal elevon operation during the flareout.

   d. Drag chute handle—Pull, rotate 90°, then pull fully out (after touchdown).

"HYD OIL HOT" WARNING LIGHT ILLUMINATION

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.

In flight—

1. **AIRSPEED—REDUCE TO 220-240 KCAS (KIAS) BY REDUCING ENGINE RPM.**
2. Flight controls and other hydraulically operated components—Use minimum necessary to maintain safe operating conditions.
3. Dampers—OFF.
4. Do not extend the RAT if minimum (1000 psi) hydraulic pressure is available.

Note

- Extension of the RAT will serve only to increase hydraulic temperatures in the primary system.
- It may be possible to determine the overheating system by noting a drop in hydraulic pressure on that system.

5. Land as soon as practicable.
6. Do not open speed brakes with speed brake switch.

   Speed brakes will open when drag chute is deployed.

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. A warning light, located on the master warning light panel, is provided to give an indication of hydraulic system overtemperatures. If uncontrollable oscillations are encountered, proceed as follows:

1. **AIRSPEED—REDUCE TO 220-240 KCAS (KIAS) BY REDUCING ENGINE RPM.**
2. **DAMPERS—OFF.**

   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 with the damper system disengaged, are phenomena not related to flight control system oscillations.

3. Do not extend the RAT.

Note

RAT extension will only increase hydraulic temperature in the primary system.

4. Select alternate flight modes (dampers disengaged, dampers engaged, and AFCS), 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 level flight attitude.

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 while evaluating the nature of the emergency.

Note

If oscillations were caused by overheating, 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. Loss of dc essential bus 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 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, internal failure of the feel force regulator, 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 either high or low stick and/or rudder forces, or pilot-induced oscillations. Internal failure of the elevator feel force regulator normally directs unregulated ram air into the elevator feel-force cylinder which will produce higher stick forces but the airplane remains controllable at restricted speeds. 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 stick force will be normal below Mach .88, and high above that range. 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 approximately normal during approach and landing. In the event of artificial feel system failure, proceed as follows:

1. **AIRSPEED—REDUCE TO 220-240 KCAS (KIAS) BY REDUCING ENGINE RPM.**
2. Use minimum control stick movement; attempt to fly hands off for at least 30 seconds.
3. Dampers—OFF.

WINDSHIELD FAILURE

If an outer layer cracks in flight, cabin pressure need not be dumped. However, if the panel becomes white or opaque or the inner layer shatters (indicating failure of the stress-bearing part of the windshield), then cabin pressure should be dumped immediately.

In the event of windshield failure, proceed as follows:

1. **CABIN AIR SWITCH—RAM (IF INNER LAYER SHATTERS OR BECOMES OPAQUE).**
2. Nesa switches—OFF.
3. Lower visor.

Note (F)

- 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. Before accomplishing the following procedures, determine whether the nature of the emergency is such that flight may be continued safely without equipment powered by the nonessential ac bus as they will be rendered inoperative. 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 RESET, then ON, and placing the ac generator switch to OFF.

(TF)

- 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 or pulling the 5-amp "Nesa Control" circuit breaker on the left-hand aft circuit breaker panel may not remove electrical power from the windshield heating elements. Under these conditions, power can be removed from the inboard windshield panels by placing the ac bus switch to RESET and ON, and placing the ac generator switch to OFF. Before moving the ac bus switch to RESET and ON, 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. The outboard panel heaters receive power from the ac essential bus and will not be cut off by switching to the emergency ac generator.

CANOPY WARNING LIGHT ILLUMINATED IN FLIGHT

Illumination of the canopy unlocked warning light while in flight indicates that the canopy latch hooks may not be fully engaged and loss of the canopy could ensue. If the light illuminates in flight, observe the following:

1. Do not move the canopy latch handle.
2. Reduce airspeed to 220-240 KCAS (KIAS).
3. Cabin air switch to OFF.
4. Lower visor.
5. FCS master switch—OFF.
6. Land as soon as practicable.

HIGH-PRESSURE PNEUMATIC SYSTEM WARNING LIGHT ILLUMINATED

If the "PNEU PRESS" warning light illuminates in flight prior to using the high-pressure pneumatic system, land as soon as practical. Either one or both wheel brakes or the drag chute may not be available after landing. If a high-pressure pneumatic failure is suspected, land as short as possible and engage nose wheel steering as soon as the airplane has been slowed sufficiently and before applying wheel brakes. Apply brakes cautiously to determine if both brakes operate. Refer to HIGH PRESSURE PNEUMATIC SYSTEM, Section I, for equipment affected by system loss.

ANTI-ICE SYSTEM WARNING LIGHT ILLUMINATED

1. Anti-ice switch—MANUAL ON.

EXTERNAL TANK JETTISONING

Jettisoned external tanks sometimes "fly" laterally or climb immediately after separation, rather than falling as normally expected. Jettisoning while in a turn or yaw greatly increases the tendency of the tanks to "fly." Prior to jettisoning, the pilot shall warn adjacent airplanes of his intentions. Pilots of adjacent airplanes shall exercise watchfulness and take positions above and forward of the jettisoning airplane.

CHAFF DISPENSER JETTISONING

Altitude and emergency conditions permitting, chaff pods should be jettisoned at airspeeds between 220 and 375 KCAS. If emergency jettisoning is necessary below 220 KCAS, it is probable that the pods will fly up and into the wings. Refer to Section V for chaff dispenser weight limitations.

MG-10 REDUCED POWER WARNING LIGHT ILLUMINATED

Illumination of the MG-10 reduced power warning light indicates that the MG-10 is operating at a reduced output. For procedure refer to T.O. 1F-102A-1A.

LAUNCHER-EXTEND WARNING LIGHT ILLUMINATED (MISFIRE OR HANGFIRE)

If the intervalometer delivers the fire signal and firing does not occur, the push-to-retract button will illuminate. There is a negligible chance that the missile motor may ignite after a five minute wait period. However, in emergencies the armament may be retracted after 30 seconds with only a low probability of the missile motor igniting.

Note

- A misfire occurs if armament has not moved on the rails.
- A hangfire occurs if armament has moved on the rails.

If either a hangfire or a misfire condition exists, accomplish the following after the escape maneuver:

1. Auto search button—depress to break lockon.
2. Armed-safe switch—SAFE (guard down).
3. Armament selector switch—SNAKE.
4. Ignitor control switch—OFF (or TRAINING) (guard down).
5. SPL WPN switch—DISARM (guard down).

CAUTION

Normally, no attempt should be made to retract the launchers. There is a possibility that the missile has moved forward on the launcher sufficiently to cause structural damage to the airplane and/or missile, if retraction is attempted. Except for flight conditions existing wherein a clean airplane configuration is required, the launcher should not be retracted unless a visual check confirms that the missile will not contact the airplane structure upon retraction. If a decision is reached to retract the launchers, the launchers should be retracted after a five minute wait period. Only in emergencies should the launchers be retracted after 30 seconds.

6. Push-to-retract button—Depress if required.
   If the launchers do not retract when the push-to-retract button is depressed, an alternate method of retraction may be attempted as follows:
   a. Armament selector switch—MISSILE RAD, ALL or OPT (five seconds).

CAUTION

If either MISSILES TRIGGER SALVO or any ROCKETS is selected, armament bay doors will close immediately with possible interference with an extended launcher. Avoid these selections if armament bay doors are to remain open.

b. Armed-safe switch—ARMED (five seconds), then SAFE.
c. Armament selector switch—SNAKE.

INCOMPLETE ARMAMENT CYCLE

If the launching racks remain extended or the missile bay doors do not close for any reason other than a misfire or hangfire as described above, the push-to-retract button will not illuminate. Use the following procedure in attempting to close doors:

CAUTION

When the push-to-retract button is not illuminated, do not use the push-to-retract button to attempt to retract launchers.

WARNING

The armament trigger should not be depressed at any time during this procedure, to preclude the possibility of armament firing.

1. Auto search button—Depress to break lockon; wait 15 seconds.
2. Armed-safe switch—SAFE (guard down).
3. Ignitor control switch—OFF (or TRAINING) (guard down).
4. SPL WPN switch—DISARM (guard down).
5. Armament selector switch—Another position.
6. Armed-safe—ARMED for five seconds; then SAFE.
7. Armament selector switch—SNAKE.

CAUTION

The drag chute should be used (if required) when landing with the armament bay doors open; however, with the aft center rail extended, deployment of the drag chute may result in damage to WSEM or missile aboard when the barrier probe extends.

If this procedure does not result in closing the doors, land with doors open. Refer to LANDING WITH ARMAMENT DOORS OPEN.

WARNING

If armament doors remain open and launchers remain extended, the range of the airplane will be reduced by approximately 40%.

UNSNUBBED LAUNCHER EXTENSION

1. Armed-safe switch—SAFE.
2. Armament selector switch—SNAKE.
3. Reduce airspeed to 220-240 KCAS (KIAS).
4. Do not attempt to retract launchers.
5. Avoid congested areas.
6. Check for condition of WSEM/missile.

   Determine condition and security of WSEM or missile by a report from another aircraft or from the ground. To establish whether the WSEM/missile is secure for landing, impose positive g loads on the airplane.

CAUTION

Do not exceed the acceleration limitations for missile bay doors open.

Additionally check for security by actuating speed brakes in straight and level flight.

7. Land as soon as practicable.

Note

When the aft center rail extended, deployment of the drag chute will result in damage to WSEM/missile when the barrier probe extends.

LANDING

LANDING GEAR UNSAFE ON LANDING GEAR EXTENSION

If the landing gear does not indicate down and locked after landing gear normal extension, recycle the landing gear handle. After the recycling attempt, if the landing gear warning light does not go out, the landing gear green position indicator lights do not illuminate, or the audio warning signal continues with the gear handle in the DOWN position, use the EMERGENCY GEAR EXTENSION procedure, this section. If time and conditions permit, accomplish the following before using the emergency gear extension system:

1. Gear handle—DOWN.
2. Landing gear control and landing gear position circuit breakers—In.
3. Landing gear emergency extension handle—In.
4. Landing gear position indicators—Press-to-test.

Note

Main gear position indicators illuminate on test regardless of gear position. The nose gear position indicator illuminates only with the nose gear locked up or down.

5. Landing light switch—LANDING LIGHTS. Illumination of landing lights indicates nose gear

down and in a position safe for landing. Illumination does not indicate main gear position although it may assist in determining the position of the main gear.

6. Obtain a visual check of landing gear position from another airplane or tower flyby.

EMERGENCY GEAR EXTENSION

1. Airspeed—Below 240 KCAS (KIAS).
2. Gear handle—UP.
   Use of the emergency gear extension system with the gear in the up position provides the best chance for obtaining a safe indication. If conditions do not permit having the gear handle UP, accomplish emergency extension with the gear handle DOWN.
3. Landing gear emergency extension handle—Push down (unlock) then pull approximately 2 inches until fully extended.

CAUTION

Land as soon as possible after landing gear emergency extension. Emergency extension may be followed by HYD OIL HOT warning light, flight control system oscillations, or secondary hydraulic system failure.

Note

- Emergency extension should not require more than 10 seconds.
- Nose wheel steering is inoperative when the gear is extended by the emergency system.

4. Landing gear position indicators—Green.
5. Gear handle—DOWN to deactivate the landing gear warning lights and audible warning signal.

WARNING

The landing gear handle must remain down for the remainder of the flight. Recycling the handle to the UP position introduces a large quantity of air into the secondary hydraulic system which may render it inoperative.

6. Make a normal landing and stop on the runway to have the gear safety pins installed before making any turns.

If the emergency gear lowering procedure does not result in a safe indication, yaw the aircraft with alternate rudder applications and apply symmetrical g to attempt to lock the gear down. If the faulty gear is in the up position, an electrical system malfunction may be causing the problem. If conditions permit flight without electrical power, the following procedure may be attempted:

a. Landing gear emergency extension handle—Leave fully extended.
b. Gear handle—UP.
c. DC generator and battery switches—OFF.

WARNING

Fuel boost pumps will be inoperative after the dc generator and battery switches are turned off. For a low fuel condition, maintain wings level in a nose high attitude to prevent engine flameout.

d. Gear handle—DOWN.
e. Obtain visual confirmation of gear position from a chase airplane or tower flyby since position indicators will not be available without dc power.
f. If this procedure is successful, do not turn the dc generator and battery back on until gear safety pins have been installed.

If a safe indication cannot be obtained on the main landing gear, then an approach end barrier engagement should be considered. With only a nose gear unsafe, an approach end barrier engagement is not recommended.

RUNWAY APPROACH END ARRESTMENT (BAK-9, -12, or -13)

Engagement of the BAK-9, -12, or -13 type arresting gear at the approach end of the runway should be considered when a condition occurs which is likely to cause

loss of directional control after touchdown or when extended runout can cause progressive failure, such as a blown main gear tire, landing with one or both main gear up or unlocked.

WARNING

Do not attempt to engage the MA-1A type barrier on the approach end as injury and damage will result. If time allows, the MA-1/A cable should be removed before an approach end engagement of either the BAK-9, BAK-12, or BAK-13 barrier is attempted. If time does not allow for removal of the MA-1/1A cable, the tail hook must be deployed after passing the MA-1/1A barrier cable(s).

CAUTION

Approach end arrestments are not recommended if the only problem is failure of the nose gear to extend. Emergency landings utilizing a normal approach and landing roll, have experienced no control difficulty and only minor damage.

Engagement of a BAK-9, -12, or -13 barrier at the approach end of the runway will provide the following:

a. Additional directional stability.
b. A reduction in time that the pilot and airplane are exposed to possible injury or damage.
c. Limit airplane rollout or uncontrolled skid to 1500 feet or less.
d. A specific location where emergency vehicles can plan to meet the airplane.

The BAK-12 system is capable of arresting an F-102 at maximum gross weight at an engaging ground speed of 154 knots. The BAK-9 system is capable of arresting an F-102 at maximum gross weight at an engaging ground speed of 162 knots. The BAK-13 is capable of arresting an F-102 at maximum gross weight at an engaging ground speed of 131 knots. The yield strength of the F-102 arresting hook will not be exceeded with any of the above speed/weight combinations. The decision to attempt an approach end arrestment must be dictated by local conditions. Where the arresting cable is installed at the extreme end of the runway or in the overrun, the ability of the overrun area to accommodate touchdown loads should be considered. Touchdown on poorly paved overrun may prevent an arrestment through loss of directional control, hook skip, or damage to the airplane. Late lowering of the nose results in a severe nose down reaction at engagement. If it is decided that an approach end barrier arrestment is to be made, proceed as follows:

1. Expend excess fuel to lighten airplane and minimize fire hazard. Retain empty drop tanks.

Note

- Alert ground personnel and request BAK-12, BAK-9, or BAK-13 arresting gear be installed on approach end of runway.
- Request runway be foamed in arresting gear runout area to decrease fire hazard and landing gear side loads (time and conditions permitting). (BAK-13, BAK-12, and BAK-9 runout distance is 950 feet).

2. Perform airborne procedures listed in the flight manual for the respective conditions.
3. Arrest hook handle—Pull straight out. Arrest hook light—Illuminated.

WARNING

The arrest hook release handle must be pulled straight out. Any side load or twisting action on the handle may make it impossible to pull out.

CAUTION

Do not deploy in flight where obstructions such as power lines, fences, approach lights, etc., could be engaged by the arrest hook.

Note

- Plan touchdown point according to type of emergency. A normal, smooth landing flare and touchdown is of primary importance. Therefore, do not "force on" or "hold off" airplane to achieve planned touchdown point.
- Touchdown point should be sufficiently short of the arresting cable to permit lowering of nose gear to runway before arrestment.

CAUTION

Premature touchdown and lowering of nose will cause wing to drop, resulting in loss of directional control which may permit airplane to veer off runway before reaching the arresting cable.

4. Shoulder harness—Lock.
5. Drag chute—Deploy.
6. Nose wheel steering—Engage.
7. Avoid using brakes just prior to engagement.

CAUTION

Do not attempt to lock brakes on wheel with flat tire until after crossing arresting cable. The wheel should be rolling when passing over the cable to avoid snagging or cutting the cable.

8. Throttle—IDLE or OFF.

Note

Throttle advancement may occur during rapid deceleration if the pilot has his hand on the throttle.

CAUTION

When landing with a flat main gear tire, keep engine running until fire equipment arrives. Stopcocking the engine allows fuel to vent overboard near the wheel brakes, thus creating a fire hazard.

9. Fuel valve switches—CLOSE.

   Approximately three to five seconds are required for the shutoff valves to close.

10. Master switch—TRIP, then OFF (if required).

CAUTION

Master switch must be ON if nose wheel steering is required.

RUNWAY OVERRUN BARRIER ENGAGEMENT

Refer to TAKEOFF for overrun barrier engagement procedures.

LANDING WITH ARREST HOOK EXTENDED (LIGHT ILLUMINATED)

The arrest hook release system is all mechanical, and once locked in the up position is unlikely to extend until deployed. If the arrest hook is inadvertently extended in flight, it will not introduce any problems while airborne. A safe, hook-extended touchdown can be made.

WARNING

If the arrest hook is inadvertently extended in flight, and there is a prone barrier on the approach end of the runway, the airplane touchdown point must be compensated to avoid inadvertent barrier engagement. When extended, the arrest hook may drag 1000 feet before touchdown during a normal landing. When landing with the arrest hook extended, fly a high final approach and touch down well past the arresting cable on the approach end of the runway.

WARNING

When the arrest hook is dragged on the runway for a prolonged landing rollout, the hook will become overheated. Prior to stopcocking the throttle, keep engine running until fire equipment arrives and cools the hook.

TIRE FAILURE ON LANDING

When a blown main gear tire is realized, or the instant the airplane starts to pull off to one side, lower the nose wheel to the runway and engage nose wheel steering. Simultaneously with the instant lowering of the nose wheel, perform the abort procedure. Use opposite rudder, aileron and braking to maintain directional control as the nose wheel is being lowered to the runway. Use opposite braking and aileron with nose wheel steering to maintain direction control after nose wheel steering is engaged. If the external tanks are empty, they need not be jettisoned unless a barrier engagement is likely. External tanks must be jettisoned for successful barrier engagement, unless the arrest hook is to be used (barrier webbing either lowered or absent). As speed decreases, if vibration increases, lock the brake on the wheel with the blown tire.

WARNING

- A locked brake should be released when barrier engagement becomes imminent to avoid severing barrier cables.
- The rudder pedals must be aligned with the nose wheel direction and the nose strut must be compressed to engage the nose wheel steering.
- 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.

Main Wheel Tire Failure On Landing.

1. Nose wheel—Lower immediately.
2. Nose wheel steering—Engage.
3. Drag chute—Deploy.
4. Arrest hook—Deploy (if necessary).

Nose Wheel Tire Failure On Landing.

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 extreme nose wheel shimmy with a blown nose wheel tire, nose wheel steering may be uncontrollable. Up elevator should be used to relieve as much as pressure as possible from the nose wheel.

WARNING

When it is known before landing that the nose wheel tire is flat, maintain a nose-high attitude until airspeed decreases to minimum allowable to make a controlled nose wheel touchdown.

ANY GEAR UP OR PARTIALLY EXTENDED

If landing is to be made with partial gear extension, proceed as follows:

Note

- Approach end barrier arrestment should be considered when a condition exists which is likely to cause loss of directional control after touchdown. Refer to RUNWAY APPROACH END ARRESTMENT, this Section.
- 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. they should be retained to cushion the impact.
2. Make normal approach.
3. Speed brake switch—As required.
   Open speed brakes prior to touchdown to ensure drag chute deployment.
4. Landing gear handle—DOWN.
5. Shoulder harness inertia reel handle—LOCKED.

CAUTION

All switches not readily accessible should be positioned before moving the shoulder harness inertia reel handle to LOCKED position.

6. Immediately before touchdown:
   a. Throttle—OFF.
   b. Fuel shutoff valve switches—CLOSE.
      Approximately three to five seconds are required for the shutoff valves to close.
   c. Canopy—Retain.
      Refer to CANOPY RETENTION VS JETTISONING.
7. After touchdown:
   a. Drag chute—Deploy.
   b. Master switch—TRIP, then OFF.
   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.

FOAMING THE RUNWAY

To minimize the possibility of fire when landing with an unsafe landing gear and no approach end barrier available, the runway should be foamed in the area where the wing tip or nose will contact the runway. With an unsafe nose gear, the foam strip should be laid on the runway centerline 8 feet wide and 5000 feet long beginning 3000 feet from the approach end of the runway. With one main gear unsafe the strip should be started 1500 feet from the approach end of the runway.

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, they should be retained to cushion the impact.
2. Make normal approach.
3. Speed brakes switch—As required.
   Open speed brakes prior to touchdown to ensure drag chute deployment.
4. Shoulder harness inertia reel handle—LOCKED.

CAUTION

All switches not readily accessible must be positioned before moving the shoulder harness inertia reel handle to LOCKED position.

5. Immediately before touchdown:
   a. Throttle—OFF.
   b. Fuel shutoff valve switches—CLOSE.
      Approximately three to five seconds are required for the shutoff valves to close.
   c. Canopy—Retain.
      Refer to CANOPY RETENTION VS JETTISONING.

6. Normal landing attitude for touchdown.
7. Drag chute—Deploy.
8. Master switch—TRIP, then OFF.

EJECTION VS. FORCED LANDING

Note

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.

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 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 like 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 are *proficient* in 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 a satisfactory 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 equivalent of "low key" altitude.

AIR START ATTEMPTS DURING FORCED LANDING PATTERN

It is suspected that a contributing factor in some recent unsuccessful flameout landings may have been that the pilot did not devote his full attention to the maintenance of the required flameout pattern once high key was reached. The distracting influence in these instances was air start attempts. The solution to this problem is not simply to prohibit air start attempts once high key is reached. Past experience has shown that flameout landings are dangerous and should be attempted only under ideal conditions. Accordingly, it would be dangerously misleading to create the impression that air starts should be abandoned at any specific time and all concentration be directed towards a flameout landing. If several air starts have already been attempted or if the nature of the flameout indicates that any further air start attempts will be futile, then obviously no further attempts should be made even long before the high key point is reached. But if flameout has occurred near high key point the pilot may attempt an air start below high key provided he is not dangerously distracted. Regaining power must be afforded a high priority since, as already pointed out in previous instructions, flameout landings should be avoided to the extent that ejection is normally considered the best course of action. If it is decided to attempt a forced landing, the following considerations should be observed:

a. In the event of a flameout, attempt to complete all air start efforts before high key is reached so that full attention may be devoted to accomplishing a successful flameout landing.

b. If the circumstances of flameout have precluded conclusive air start attempts prior to high key, further air starts may be attempted but primary attention should be devoted to proper execution of the flameout landing.

c. Do not attempt air starts after low key is reached.

Note

This does not preclude air start attempts when flameout occurs below low key.

d. These instructions in no way alter previously established requirements for ejection versus forced landing. (Refer to EJECTION VS FORCED LANDING.)

FORCED LANDING (WINDMILLING OR FROZEN ENGINE)

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 may 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 unsuitable or whenever the landing gear cannot be extended for forced landing, eject. Refer to EJECTION VS FORCED LANDING.

Note

The helmet visor should be lowered during any emergency landing. The visor provides eye protection from impact or flying object and from wind blast if the canopy is jettisoned.

The handling characteristics of the airplane during a flameout landing are satisfactory. 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

Normal speed brakes

Normal drag chute deployment

Emergency gear retraction

Nose wheel steering

Normal landing gear extension.

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 high key in the forced landing pattern.

WARNING

- In any emergency or precautionary landing, where it is likely or possible that the engine may fail or "freeze," extend the RAT on the base leg or well up on final if making a straight-in approach.
- If the secondary hydraulic system has failed, do not extend the RAT except with an actual engine failure. The pilot must be prepared to extend the RAT immediately when the engine fails.

Gear extension should be accomplished when entering the 10,000-foot high-key point but not later than low-key point.

Note

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, a power setting of 76 percent rpm with speed brakes fully extended will simulate flameout conditions. Figure 3-6 contains recommended procedures and techniques for a forced landing.

Execution of the overhead approach may be impossible or impractical. Initial conditions of airspeed, altitude, distance, or orientation—any single factor or combination of factors—may preclude attainment of a satisfactory high- or low-key position. A straight-in approach, on the other hand, may afford the only means of landing the airplane. The technique of a straight-in approach is similar to aiming a gun at a target. The airplane is the gun, and the desired landing point is the target. At the beginning of the approach, with landing gear down and while maintaining best glide speed, take precise, explicit note of what part of the airplane (pitot tube, glycol ring, IR seeker head, windshield apex, etc.) lies in the line of sight to the desired landing point and monitor this sight picture for changes. If the selected landing point appears to rise above the airplane reference point, the airplane will be short and ejection is the only course of action. If the selected landing point remains aligned with the airplane reference point, the airplane will reach the selected landing point. If the landing point appears to fall below the airplane reference point, the airplane will land long unless the pilot dissipates the excess altitude by adjusting his pattern with S-turns or establishes a modified base leg.

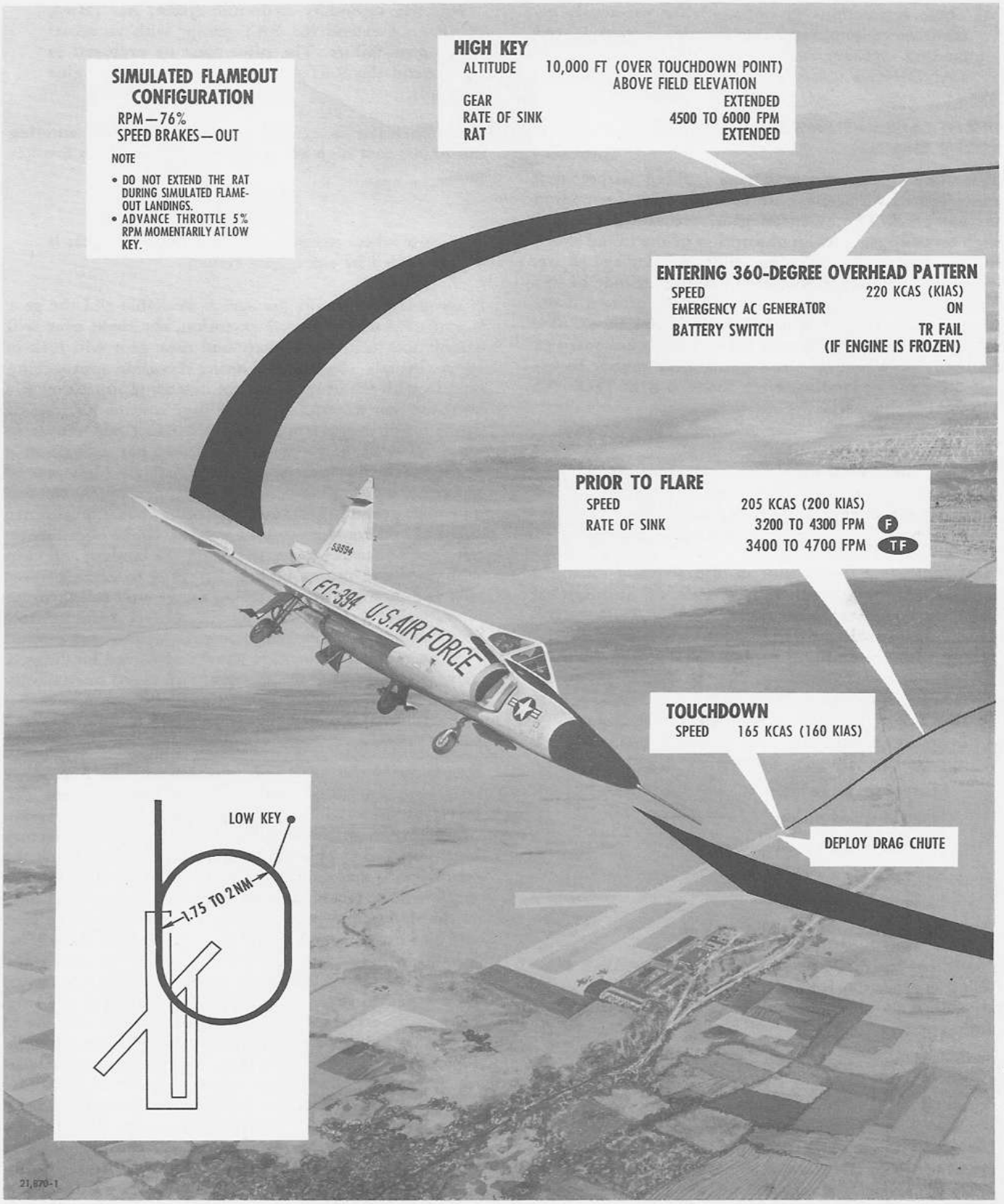


Figure 3-6 (Sheet 1)

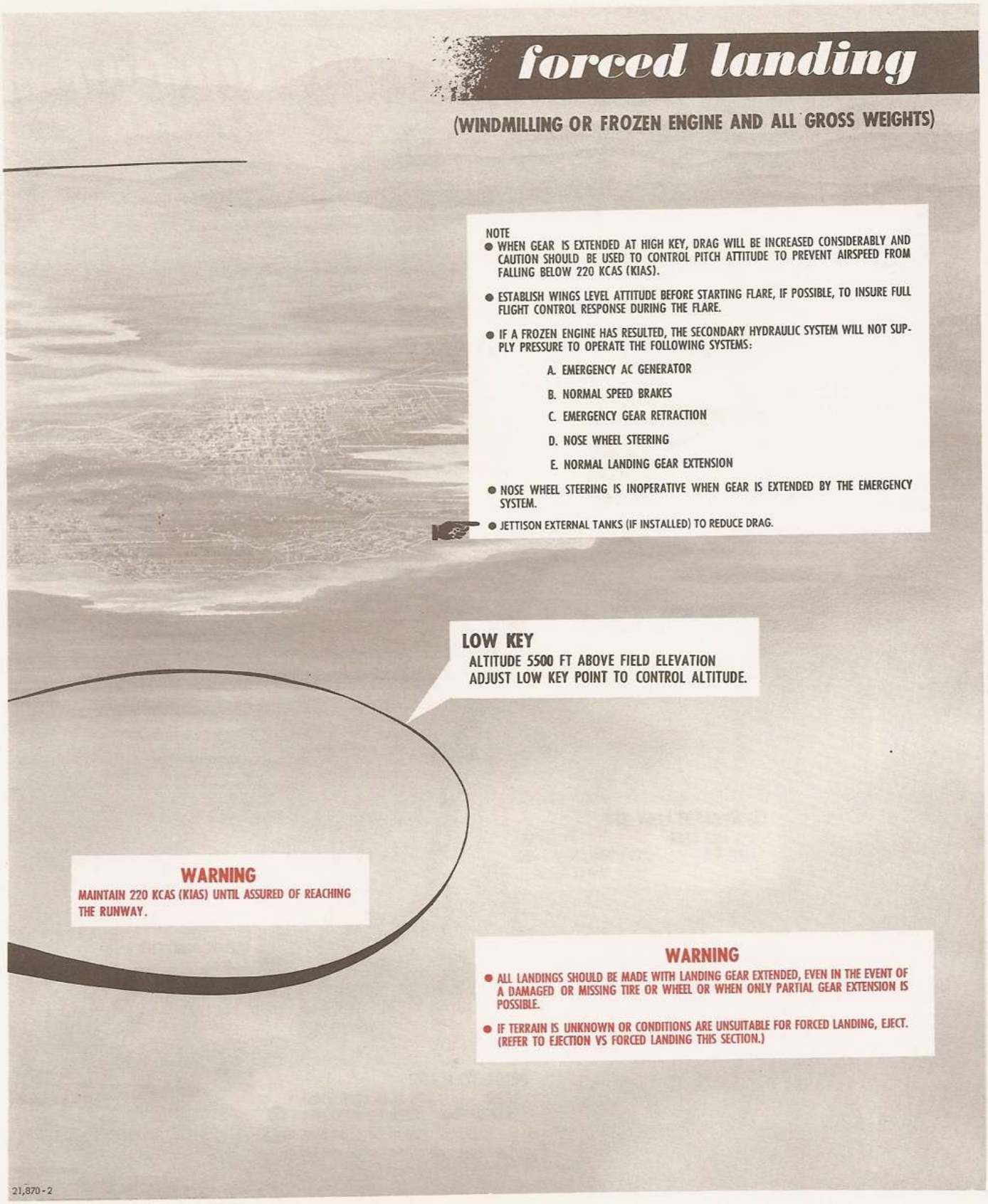


Figure 3-6 (Sheet 2)

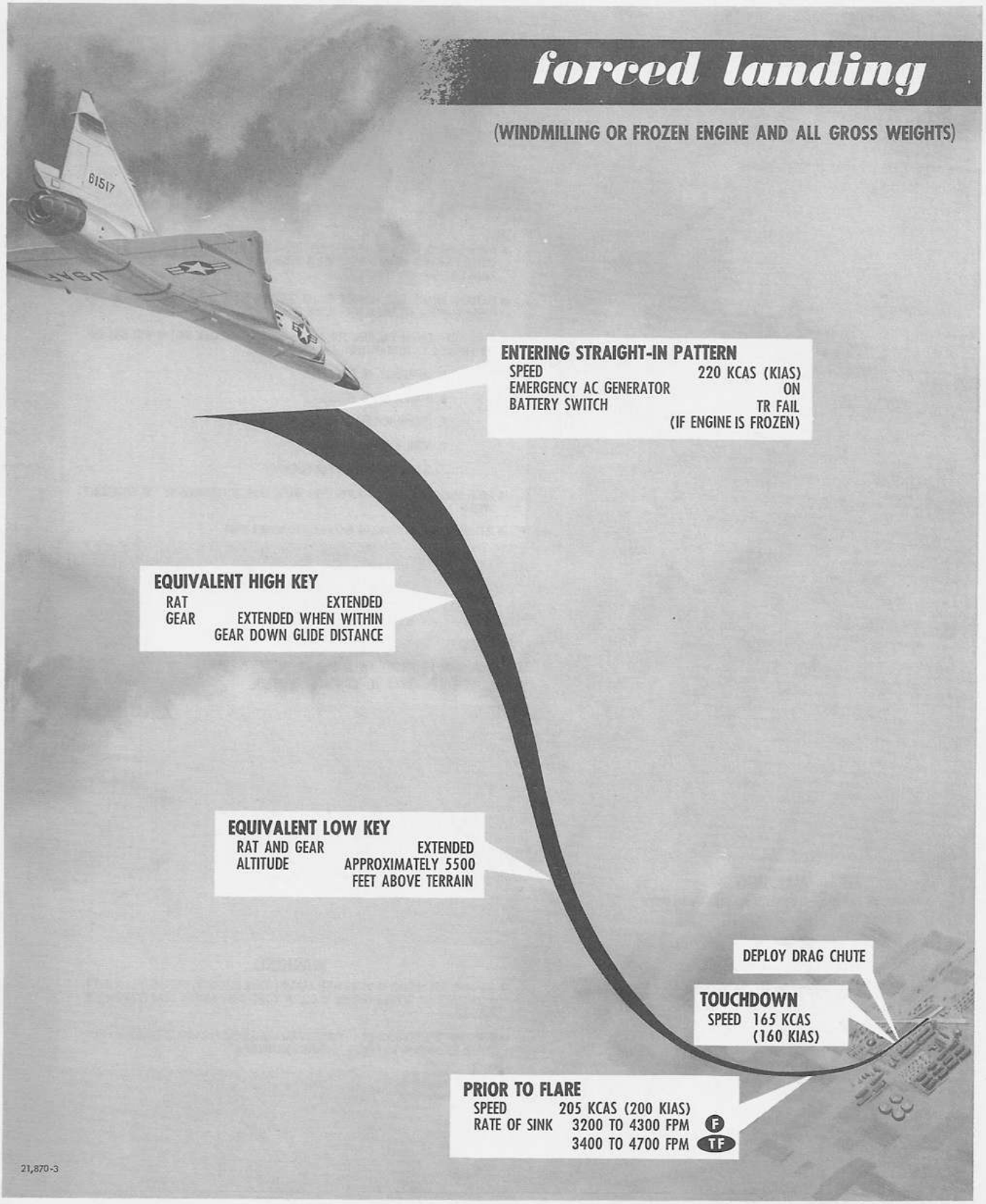


Figure 3-6 (Sheet 3)

The landing gear should not be extended until within gear-down glide distance. Make use of any available navigational aids to help determine distance to the desired landing point.

PRECAUTIONARY LANDING

A precautionary landing pattern should be flown whenever an inflight emergency requires landing as soon as possible and a forced landing pattern is either undesirable due to the type of malfunction experienced or unattainable due to darkness, weather, etc. Because of the many variables involved, it is impossible to prescribe a specific pattern to be used in all cases. The pilot must determine the most suitable pattern from his judgment and evaluation of all factors. In most cases, a slightly larger than normal rectangular pattern is desirable in order to reduce abrupt flight control and power control movements. The pattern may be entered on downwind or base leg, or flown as a straight-in approach. Pattern altitude and size may be adjusted as necessary. Airspeed prior to final approach should be 10 to 20 knots higher than normal recommended speeds. Specific actions, such as lowering the RAT, should be accomplished at the times specified in the emergency procedures for the malfunction encountered. A normal 2.5° to 3.0° glide path and final approach course should be intercepted at least two miles from the approach end of the runway, using precision instrument landing aids if available and practical. In some cases a slightly steeper final approach might be flown in order to increase the chances for either safe ejection or reaching the field should a complete loss of power occur. Excess airspeed should be slowly bled off while on final approach to allow touchdown at the recommended speed within the normal touchdown zone.

LANDING WITH ARMAMENT BAY DOORS OPEN

If it becomes necessary to land with the armament bay doors open due to an inflight 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.

CAUTION

The drag chute should be used (if required) when landing with the armament bay doors open; however, with the aft center rail extended, deployment of the drag chute will result in damage to armament aboard when the barrier probe extends.

LANDING WITH SPEED BRAKE STRUCTURAL FAILURE

In event of a speed brake malfunction with or without an associated secondary hydraulic system problem, a no-speed-brake, no-drag-chute landing should be planned if runway conditions, length, and barrier availability permit. If use of the drag chute is necessary, the nose wheel should be lowered to the runway prior to opening the speed brakes for drag chute deployment. The pilot should be prepared to overcome the aerodynamic effect of asymmetric speed brake operation by using a combination of flight controls and nose wheel steering (if available).

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. Landing gear handle—Up.
2. Follow radio distress procedure.
3. External fuel tanks button—Depress.
4. RAT handle—Pull (if necessary) & hold for four seconds.

   Hold the handle in the fully extended position for four seconds to ensure a satisfactory extension of the turbine.
5. Shoulder harness inertia reel handle—LOCKED.
6. Master switch—TRIP, then OFF.
7. Canopy jettison handle—Raise just before touchdown.
8. 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 settles into the water, the airplane may tend to flip over due to the intake ducts acting as water scoops.
9. Throttle—OFF at touchdown.
10. 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.

Note

The bailout bottle can be used under water. Pulling the "green knob" will supply oxygen for approximately 12 minutes.



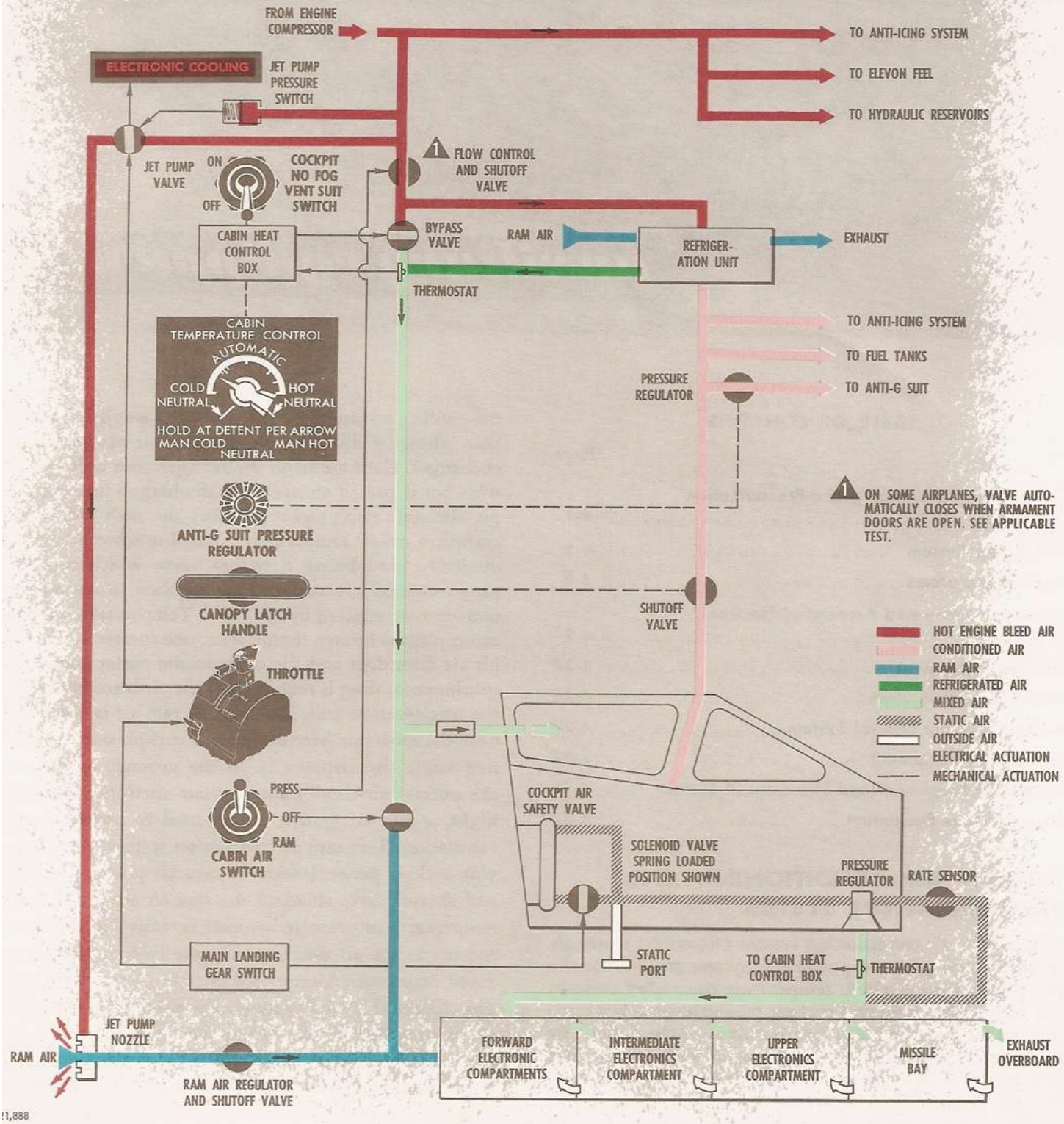
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COCKPIT AIR-CONDITIONING AND PRESSURIZATION SYSTEM

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 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 contin-

low-pressure pneumatic and air conditioning systems (typical)



21,888

Figure 4-1

ues to climb, the cockpit altitude remains at 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 below approximately 15,000 feet in a speed range of 200 to 400 KCAS (KIAS).

Cockpit pressurization is not shut off and some smoke or fumes in the cockpit may be experienced during armament firing.

CABIN AIR SWITCH

The cabin air switch (figure 1-29) 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. See figure 4-3 for switch positions and their effect on systems using low pressure pneumatic air. The cabin air switch receives power from the 28-volt dc essential bus. Loss of dc essential power will close the flow control and shutoff valve.

CABIN TEMPERATURE CONTROL KNOB

The rotary cabin temperature control knob (figure 1-26) on the utility switch panel provides for automatic control of cockpit temperature within a range of 45°F to 100°F 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.

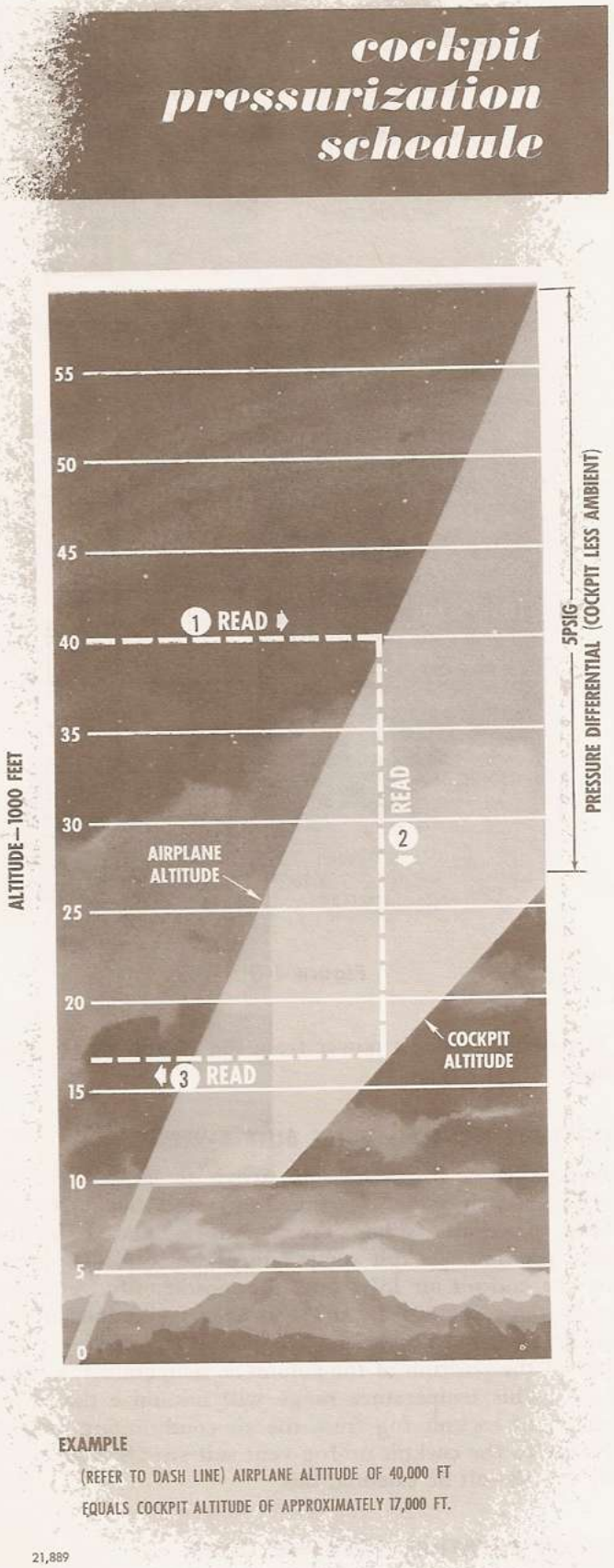
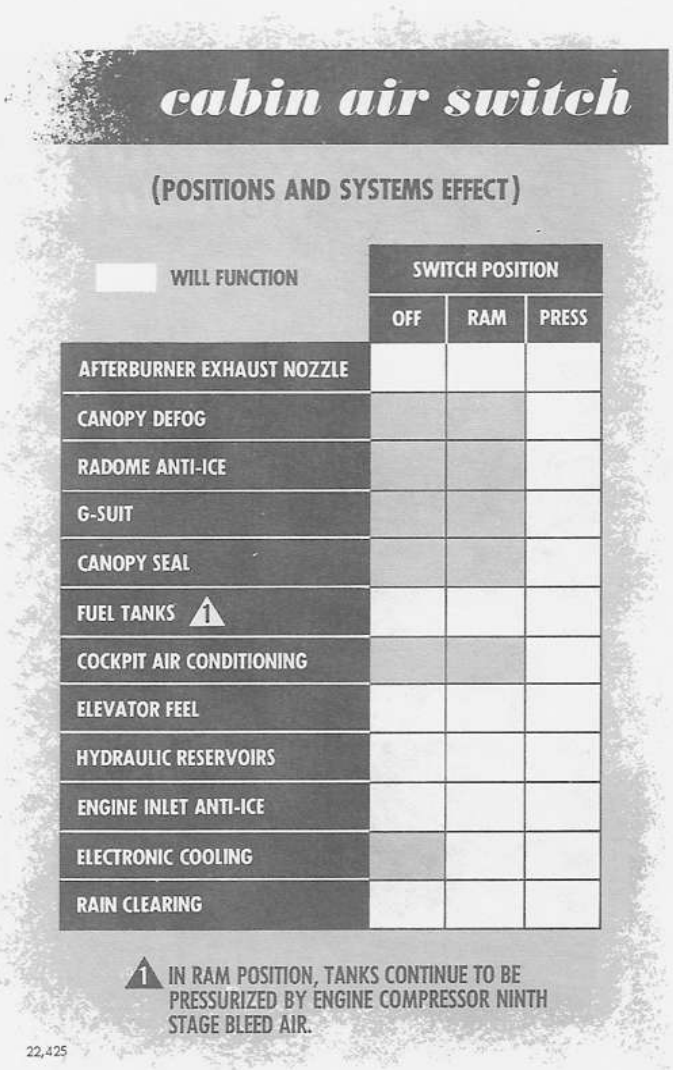


Figure 4-2



cabin air switch

(POSITIONS AND SYSTEMS EFFECT)

WILL FUNCTION	SWITCH POSITION		
	OFF	RAM	PRESS
AFTERBURNER EXHAUST NOZZLE			
CANOPY DEFOG			
RADOME ANTI-ICE			
G-SUIT			
CANOPY SEAL			
FUEL TANKS 1			
COCKPIT AIR CONDITIONING			
ELEVATOR FEEL			
HYDRAULIC RESERVOIRS			
ENGINE INLET ANTI-ICE			
ELECTRONIC COOLING			
RAIN CLEARING			

1 IN RAM POSITION, TANKS CONTINUE TO BE PRESSURIZED BY ENGINE COMPRESSOR NINTH STAGE BLEED AIR.

22,425

Figure 4-3

The switch controls power from the 28-volt dc essential bus.

COCKPIT NO-FOG VENT SUIT SWITCH

A cockpit no-fog vent suit switch is located on the utility switch panel (figure 1-26) 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 (figure 1-29) 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. Cabin air switch – PRESS.
2. Canopy – Close.
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, 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 at or above approximately 3000 feet.

DEFOGGING SYSTEM

Warm air from the low-pressure pneumatic system is used to defog the canopy panels and, on some F 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 1-26) 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 F 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 as necessary 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.

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

ANTI-ICING SYSTEMS

The airplane is equipped with an automatic anti-icing system (figure 4-4) 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.

Note

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

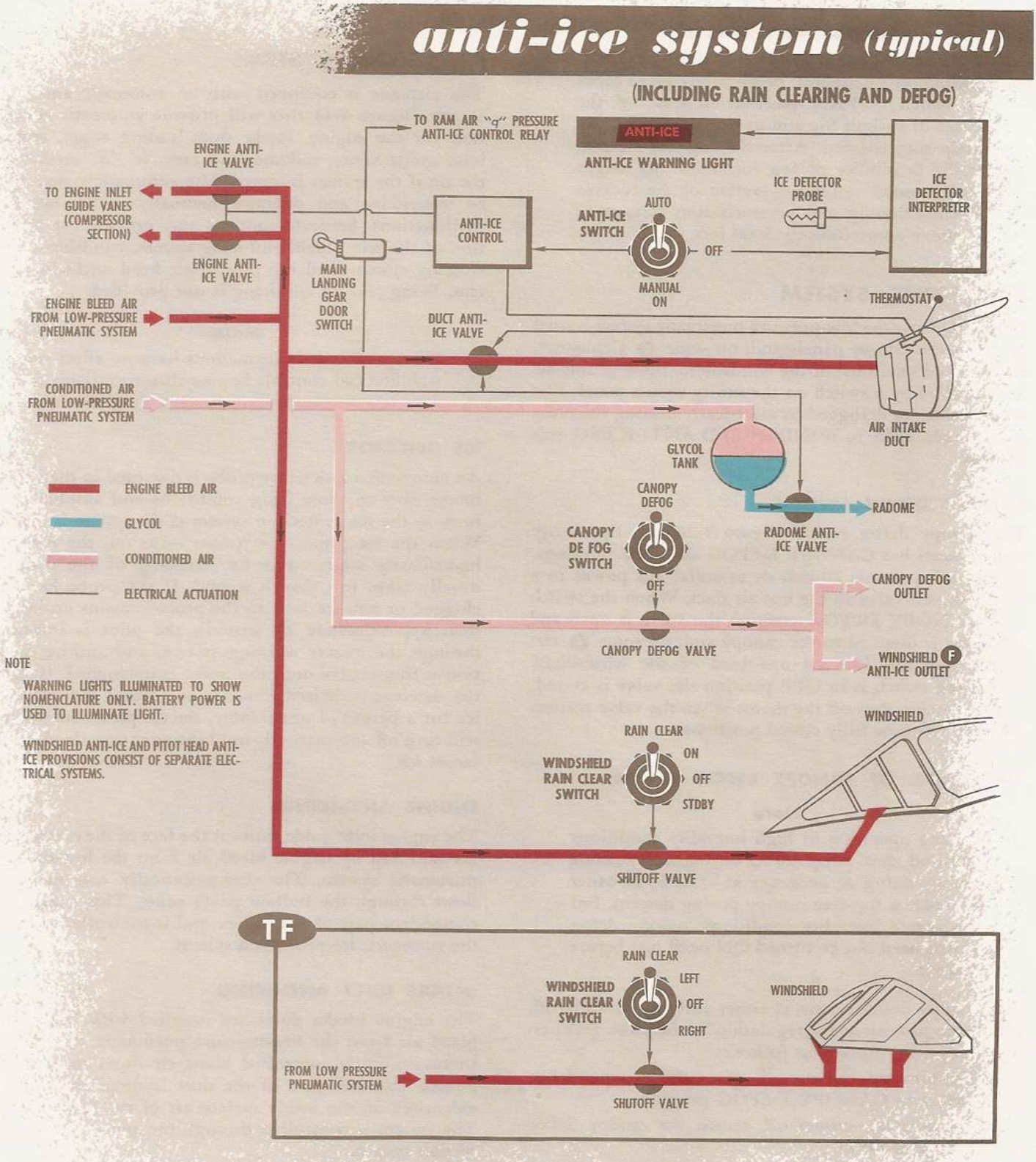

anti-ice system (typical)
(INCLUDING RAIN CLEARING AND DEFOG)
TO RAM AIR "q" PRESSURE ANTI-ICE CONTROL RELAY
ANTI-ICE
ANTI-ICE WARNING LIGHT
ENGINE ANTI-ICE VALVE
TO ENGINE INLET GUIDE VANES (COMPRESSOR SECTION)
ENGINE ANTI-ICE VALVE
ANTI-ICE CONTROL
ANTI-ICE SWITCH
AUTO
OFF
MANUAL ON
ICE DETECTOR PROBE
ICE DETECTOR INTERPRETER
MAIN LANDING GEAR DOOR SWITCH
ENGINE BLEED AIR FROM LOW-PRESSURE PNEUMATIC SYSTEM
DUCT ANTI-ICE VALVE
THERMOSTAT
AIR INTAKE DUCT
CONDITIONED AIR FROM LOW-PRESSURE PNEUMATIC SYSTEM
GLYCOL TANK
CANOPY DEFOG
CANOPY DE FOG SWITCH
OFF
RADOME
RADOME ANTI-ICE VALVE
CANOPY DEFOG OUTLET
CANOPY DEFOG VALVE
WINDSHIELD ANTI-ICE OUTLET
F
ENGINE BLEED AIR
GLYCOL
CONDITIONED AIR
ELECTRICAL ACTUATION
NOTE
WARNING LIGHTS ILLUMINATED TO SHOW NOMENCLATURE ONLY. BATTERY POWER IS USED TO ILLUMINATE LIGHT.
WINDSHIELD ANTI-ICE AND PITOT HEAD ANTI-ICE PROVISIONS CONSIST OF SEPARATE ELECTRICAL SYSTEMS.
WINDSHIELD
RAIN CLEAR
WINDSHIELD RAIN CLEAR SWITCH
ON
OFF
STDBY
SHUTOFF VALVE
TF
RAIN CLEAR
WINDSHIELD RAIN CLEAR SWITCH
LEFT
OFF
RIGHT
WINDSHIELD
FROM LOW PRESSURE PNEUMATIC SYSTEM
SHUTOFF VALVE
21,891

Figure 4-4

used to force the glycol from the tank and through the porous 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 1-44 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.

WINDSHIELD ANTI-ICING AND DEFOGGING (F)

Anti-icing and defogging of the aft two-thirds of the windshield panels are 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 are 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 ANTI-ICING AND DEFOGGING (TF)

Anti-icing and defogging of the two inboard windshield panels and defogging of the two outboard windshield panels are accomplished by electrical heating. The airplane's 220/115-volt, three-phase, ac power is stepped up by transformers to provide the required voltages. A separate transformer and control unit is provided for each panel. Temperature sensing elements imbedded in the Nesa panels control power to prevent overheating of the glass.

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. On (TF) airplanes, 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 either the left or right windshield panels.

Note

Use of the rain clearing system should be limited to 30 seconds on and one minute off for all conditions except actual rain and/or loss of nesa.

ANTI-ICE SWITCH

The anti-ice switch (figure 1-26) 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. It is necessary to pull the toggle switch out and then down to place the switch in MANUAL ON. On some (F)* and (TF)** 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 (F)

The windshield anti-ice and defog system is controlled by switches located on the utility switch panel (figure 1-26). 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 ANTI-ICE (NESA) SWITCH (TF)

The windshield anti-ice and defog switch is located on the utility switch panel (figure 1-26). The switch is placarded "Nesa" and has positions NORMAL and OFF. In the NORMAL position, 28-volt dc essential control power actuates a relay to deliver 200/115-volt ac nonessential power to the inboard windshield panel transformers, and 200/115-volt ac essential power to the outboard windshield panel transformers.

WINDSHIELD RAIN CLEAR SWITCH (F)

The windshield rain clear switch is located on the utility switch panel (figure 1-26). The switch is labeled "Rain Clear" and has three positions, ON, STDBY 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. STDBY position provides for automatic operation of the windshield rain clearing system if the Nesa system fails due to an ac power failure. On some airplanes, it is necessary to pull the toggle switch out before changing switch position.

**AF 56-1045 & on.*
***AF 55-4044 & on.*

WINDSHIELD RAIN CLEAR SWITCH (TF)

The windshield rain clear switch is located on the utility switch panel (figure 1-26). 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 and provides hot air to the right and left windshield panels. On other airplanes, the switch has three positions: LEFT, OFF, and RIGHT. On these airplanes, only one side of the windshield receives the hot air blast. On some airplanes, it is necessary to pull the toggle switch out before changing switch position.

PITOT HEAT SWITCH

The pitot-static head anti-icing switch (figure 1-26) 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. 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.

WARNING

No warning circuit is available. An electrical malfunction in this system could cause the switch to return to the OFF position. During icing conditions this could result in loss of pitot pressure. Periodic checks of the switch position should be made during IFR operation.

ANTI-ICE SYSTEM WARNING LIGHT

A light on the warning light panel (figure 1-36) 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.

NORMAL OPERATION OF ANTI-ICING SYSTEMS

The windshield anti-icing system, which also provides defogging, should be turned on prior to takeoff and left on continuously to ensure proper defogging.

1. Anti-ice switch – AUTO. Under known or suspected icing conditions, place anti-ice switch in MANUAL ON.
2. Windshield anti-ice (Nesa) switch – NORMAL (some airplanes); both Nesa switches – ON (other airplanes).
3. Rain clear switch – As required.
4. Pitot heat switch – ON.

EMERGENCY OPERATION OF ANTI-ICING SYSTEMS

Automatic Anti-Icing System

1. Anti-ice switch – MANUAL ON.

Windshield Anti-Icing System (F)

If nonessential ac power to the laminated windshield is lost or if condensation has formed, windshield defogging may be obtained as follows:

1. Windshield rain clear switch – ON.

   If windshield is heated and nonessential ac power failure is noted, the rain clearing system should be turned on immediately. This will provide an effective "heat block" for the heat energy stored in the left windshield panel at the time of nonessential ac power failure. If ac power failure is not noted until condensation has formed, placing the rain clear switch to RAIN CLEAR will initiate a heat addition process on the left windshield outer surface.
2. Cockpit no-fog vent suit switch – OFF.
3. Cabin temperature control knob – MAN HOT.

   Regulate cabin air temperature to maximum tolerable to start a heat addition process on the inside surface of the windshield.

Note

- Two to four minutes are required to heat a cold windshield using the above procedure.
- If the rain clear switch is in STDBY position (some airplanes), the rain clearing system will come on automatically when nonessential ac power to the laminated windshield is lost.

Windshield Anti-Icing System (TF)

If nonessential ac power to the laminated windshield is lost or if condensation has formed, windshield defogging may be obtained as follows:

1. Windshield rain clear switch – RAIN CLEAR (some airplanes); LEFT or RIGHT (other airplanes).

   If windshield is heated and nonessential ac power failure is noted, the rain clearing system should be turned on immediately. This will provide an effective "heat block" for the heat energy stored in the left and right windshield panels at the time of nonessential ac power failure. If ac power failure is not noted until condensation has formed, placing the rain clear switch to RAIN CLEAR (some airplanes), or LEFT or RIGHT (other airplanes), will initiate a heat addition

process on the windshield outer surfaces (the selected surface, some airplanes).

2. Cockpit no-fog vent suit switch — OFF.
3. Cabin temperature control knob — MAN HOT.
   Regulate cabin air temperature to maximum tolerable to start a heat addition process on the inside surface of the windshield.

Note

Two to four minutes are required to heat a cold windshield using the above procedure.

COMMUNICATIONS AND ASSOCIATED ELECTRONIC EQUIPMENT

TABLE OF COMMUNICATION AND ASSOCIATED ELECTRONIC EQUIPMENT

See figure 4-5.

INTERPHONE — AN/AIC-10 (F)

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 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 1-4) and on the control stick grip (figure 1-25). There is no cockpit interphone control panel. The system is on whenever power is available from the emergency dc bus.

INTERPHONE SYSTEM — AN/AIC-10 (TF)

The interphone system provides audio communication between the pilot and instructor at all times, and, when the airplane is on the ground with ground interphone equipment connected, the system allows communication between persons in the cockpit and others outside the airplane. The system also interconnects with the audio outputs of the radio and navigation systems. The pilot's and instructor's microphones and headsets are part of the interphone system. Separate control panels (figure 4-6), located on the instructor's console and the center console, mount facility switches which allow the pilot or instructor to select various modes of operation. A receptacle on the nose wheel well switch panel provides for connection to the interphone system by the ground crew when the airplane is on the ground. The 170-volt dc interphone dynamotors, built into each control panel amplifier, are powered by the 28-volt dc essential bus or, on some airplanes, by the emergency dc bus.

FACILITY SWITCHES (TF)

A rotary facility switch in the center of each control panel (figure 4-6), can be set to CALL, INTER, COMM INTER, or COMM positions. On CALL position, the other station can be called, regardless of how the facility switch for that station may be set. On INTER position the pilot and instructor can communicate by interphone, by depressing the microphone buttons and, if ground interphone is connected, persons in the cockpit can communicate with the ground interphone station. On COMM INTER position, the interphone functions as a "hot mic" (the microphone button does not need to be depressed to communicate on interphone). Command radio transmission on this position can be obtained by using the microphone buttons. On COMM position, the pilot or instructor can receive and transmit on the UHF command radio.

MIXER TOGGLE SWITCHES (TF)

Five mixer toggle switches are located across the top of each control panel (figure 4-6). Only four of the five switches are used on this airplane. They are placarded "Comm," "Inter," "ILS," and "Tacan," and are used to monitor the UHF command radio, the interphone, or the navigation receiver when the facility switch is set for some facility other than that for which the toggle switch is placarded.

NORMAL-AUXILIARY LISTEN SWITCH (TF)

A toggle switch to the right of the facility switch, with NORMAL and AUX LISTEN positions, is safety-wired in the NORMAL position. If the interphone amplifier fails, the switch can be placed in the AUX LISTEN position to bypass the amplifier for listening. Transmission is not possible if the amplifier fails.

VOLUME CONTROL KNOB (TF)

The volume control knob on the left side of the interphone control panel (figure 4-6) is normally set at a comfortable volume level for listening, but can be turned clockwise past the 12-o'clock position to make use of a high-fidelity noise reducing circuit. Turning the volume control knob to this position will increase the volume of the radio sets being heard through the interphone, so that volume controls on the individual sets will have to be adjusted to establish the desired volume. The volume control knob has no effect when the normal-auxiliary listen switch is in the AUX LISTEN position.

MICROPHONE BUTTONS

Push-to-talk microphone buttons are located on the throttle grip(s) (figure 1-4) and on the right-hand grip of each control stick (figure 1-25). The nose wheel steering buttons on the control sticks are normally used to control nose wheel steering with the airplane on the ground, but can serve as secondary microphone switches in flight, or on the ground, if the nose wheel is turned more than 50° right or left of center. The microphone button is inoperative in flight if the landing gear is extended on (F) airplanes only.

communications and associated elec-

TYPE	DESIGNATION	FUNCTION	PRIMARY OPERATOR (F)	PRIMARY OPERATOR (TF)	RANGE
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.	Pilot's and ground crew.	Cockpit to ground crew.
UHF Command Radio	AN/ARC-34	Communications from airplane to airplane and from airplane to ground by UHF communications radio.	Pilot	Pilot	Line of sight.
Tacan	AN/ARN-21	Used in conjunction with Tacan surface beacon to provide bearing and range information.	Pilot	Pilots	Approximately 195 miles, line of sight.
ILS Receiver	AN/ARN-31	Receives ILS localizer and glide path transmission.	Pilot	Pilots	Localizer approximately 25 miles, glide path approximately 15 miles.
Marker Beacon Receiver	AN/ARN-12 or AN/ARN-32	Receives location marker signal on navigational beam.	Pilot	Pilots	
IFF	AN/APX-6A, AN/APX-25, or AN/APX-72	Automatic radar identification returned if challenged by surface or airborne sets. Can transmit emergency identification signal.	Pilot	Pilots	Line of sight.

21909-1

Figure 4-5

NORMAL OPERATION OF INTERPHONE SYSTEM (TF)

a. To transmit or receive on UHF command radio, place facility switch in COMM INTER position and use the push-to-talk microphone buttons when transmitting. (Interphone communication can be carried on at the same time, without using the microphone buttons.)

b. For communication between personnel in the cockpit, or between personnel in the cockpit and ground personnel (when the ground interphone is connected), place all mixer toggle switches in OFF position, the facility switch in INTER position, and depress the microphone button.

c. To call another interphone station (cockpit or ground), place the selector switch in CALL position. Voice signals from the interphone will override whatever other signals are being received by the other person.

d. To monitor interphone when the facility switch is set for any position other than INTER or COMM INTER, use the mixer toggle switch placarded "Inter."

Note

Selection of the ADF position prevents monitoring of guard receiver.

e. To monitor the UHF command receiver, navigation receiver, and use interphone communication at the same time, use the "Comm" and Navigation mixer toggle switches, and place the facilities switch in INTER.

tronic equipment

F	TF
LOCATION OF CONTROLS	LOCATION OF CONTROLS
Ground crew connection amplifier in the nose wheel well.	Center console. Ground crew connection and amplifier in the nose wheel well.
Left-hand console.	Left-hand console.
Right-hand console. Indicators on instrument panel.	Center console. Indicators on instrument panel.
Right-hand console. Indicator on instrument panel.	Center console. Indicator on each instrument panel.
Indicator light on instrument panel.	Indicator light on each course indicator.
Right-hand console.	Center console.

21909-2

EMERGENCY OPERATION OF INTERPHONE SYSTEM (TF)

In the event of amplifier failure as evidenced by silence or unintelligible signals from all facilities, proceed as follows:

1. Place normal-auxiliary listen switch in AUX LISTEN position. (This will necessitate breaking the safety wire.)
2. Place all mixer toggle switches in OFF position and turn facility switch to the desired listening facility.
3. If the desired listening facility cannot be selected by the facility switch, turn on the appropriate mixer toggle switch. If more than one mixer toggle switch is on, only the switch farthest left will have effect.

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 4-7) is on the left console and permits selection of any of 20 frequencies which can be preset in any order. A remote indicator (figures 1-8 and 1-9) on the instrument panel, will indicate the frequency of the manually tuned channel or the number of the selected preset channel. 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.

Note

Selection of the ADF position prevents monitoring of guard receiver.

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 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

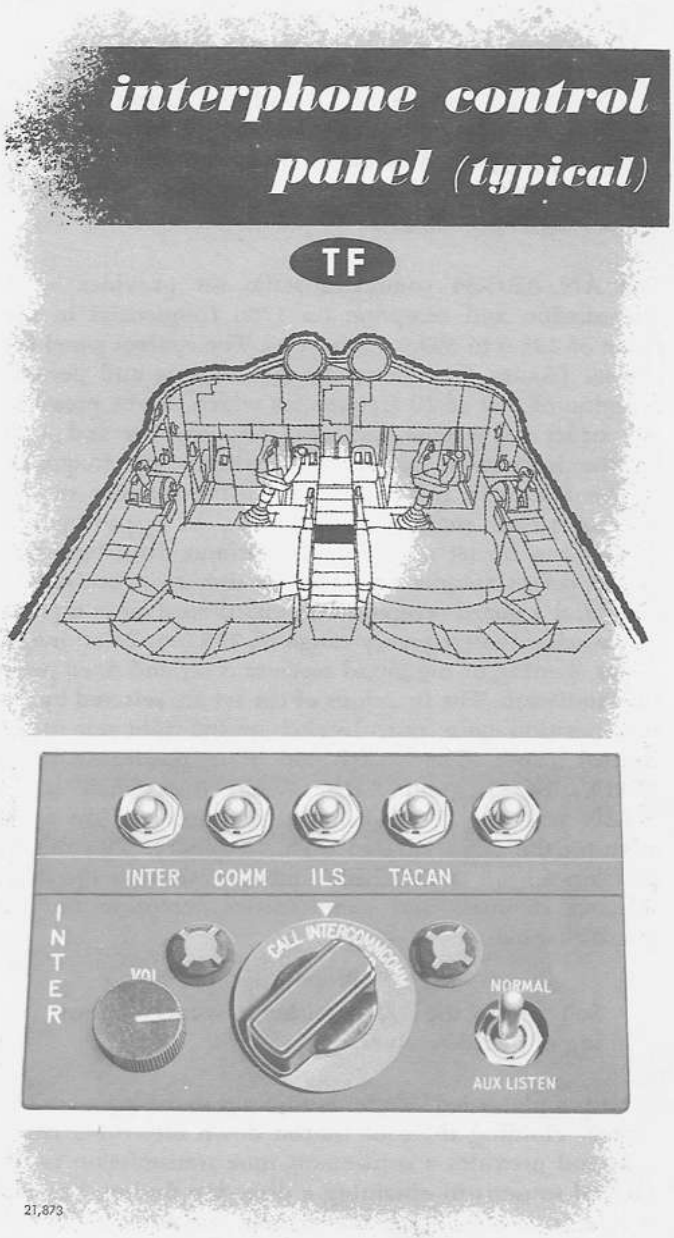


Figure 4-6

current for operation. The command radio is powered by the emergency dc bus.

Operation of Command Radio

1. Select PRESET position with the selector control (sliding button).
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. 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.) The 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.
- If the command radio cycles continuously, selection of a different frequency may stop the cycling. If cycling continues, 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.

KY-28

Some airplanes* have KY-28 equipment installed which is used with the AN/ARC-34 UHF command radio to provide wide band communications. Controls and indicators for the KY-28 are provided on the console-installed ciphony control panel (figure 4-8). The KY-28 is installed in the ram air turbine compartment.

Power Switch

The power switch is a two-position toggle switch on the ciphony control panel and has PWR and OFF positions. The PWR position provides power to the KY-28 for operation of the wideband facility. The OFF position removes all power from the KY-28 and permits only normal operation of the UHF command radio.

**Airplanes modified by TCTO 1F-102-1124.*

uhf command radio control panel

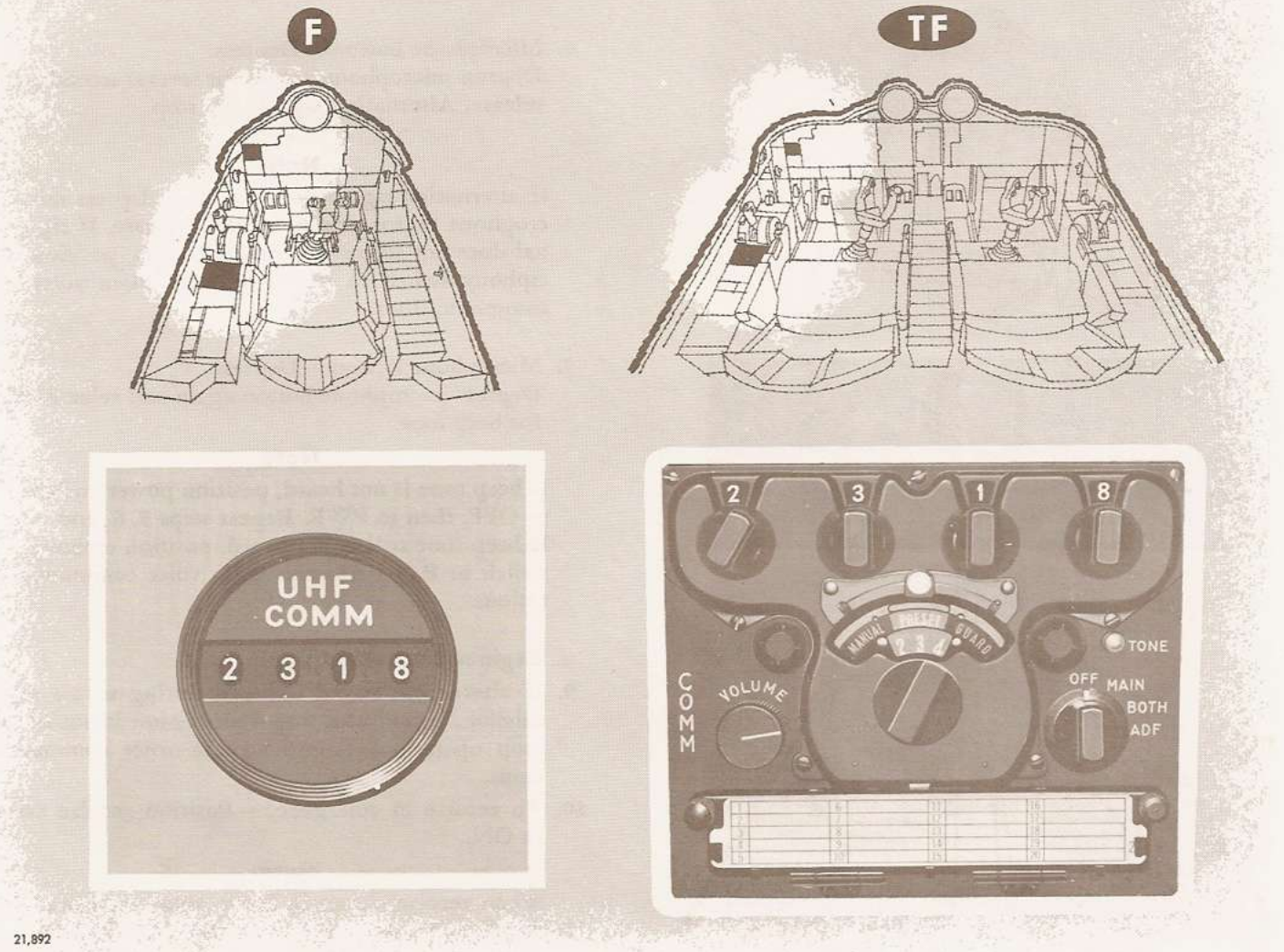


Figure 4-7

Delay Switch

The delay switch is a two-position toggle switch on the ciphony control panel and has IN and OUT positions. The IN position provides a short delay for use during wideband operations with a relay radio. The OUT position disables the delay function.

Zeroize Switch

The zeroize switch is a two-position toggle switch on the ciphony control panel and has ON and OFF positions. The switch is guarded and is spring loaded to the OFF position. The ON position returns certain circuits in the KY-28 to a zero condition. The OFF position disables the zeroize circuits.

Ciphony Switch

The ciphony switch is a two-position toggle switch on the ciphony control panel and has P (plain) and C (cipher) positions. The C position provides wideband speech operation with the UHF command radio. The P position provides normal UHF command radio communications.

Ciphony Indicator Light

The ciphony indicator light is a green press-to-test light on the ciphony control panel. When illuminated, the light indicates that the ciphony switch is positioned to C (cipher).

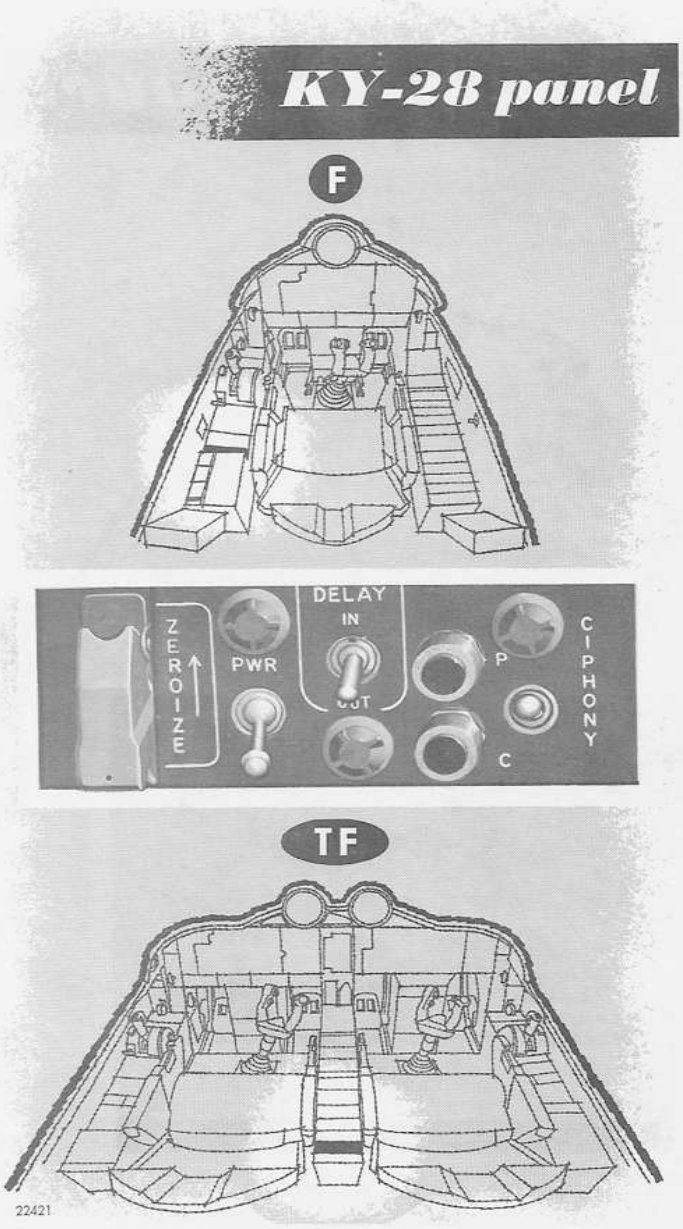


Figure 4-8

Plain Indicator Light

The plain indicator light is an amber press-to-test light on the ciphony control panel. When illuminated, the light indicates that the ciphony switch is positioned to P (plain).

Normal Operation

For normal operation on a preset command frequency and on the fixed guard frequency, proceed as follows:

1. UHF selector control — PRESET.
2. Channel selector knob — Select desired channel.
3. Power switch — PWR.
4. Establish plain voice communication.
5. Ciphony switch — C.
   Steady tone, then alternating two-tone signal will be heard.

Note

Prolonged steady tone indicates that transmission is not secure. Switch to plain voice and transmit plain voice communications.

6. Microphone button — Depress.
   Depress microphone button for several seconds, then release. Alternating signal will stop.

Note

If alternating signal does not stop, depress microphone button and hold; then release. If signal does not stop after three attempts, position ciphony switch to P and transmit plain voice communications.

7. Microphone button — Depress.
   Depress microphone button again and release. Wait for beep tone.

Note

If beep tone is not heard, position power switch to OFF, then to PWR. Repeat steps 5, 6, and 7. If beep tone still is not heard, position ciphony switch to P and transmit plain voice communications.

8. Begin secure transmission.
9. If alternating signal is heard during secure transmission, it indicates that transmission is not secure; stop operation. Transmit plain voice communications.
10. To zeroize in emergency — Position zeroize switch to ON.

Note

When zeroize occurs in flight, secure transmission is no longer possible until KY-28 has been reset on the ground.

TACAN — AN/ARN-21

The TACAN equipment is designed to operate in conjunction with a surface navigation beacon to form a radio navigation system called TACAN (TACtical Air Navigation). This system provides continuous indications of 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 indicator, a range indicator, the course deviation indicator (CDI) 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 range 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 is powered by the 28-volt dc nonessential bus and the 115-volt ac nonessential bus.

Note

All TACAN functions are powered by the ac and dc nonessential busses. Failure of either or both will result in loss of the marker beacon light and TACAN information. (Magnetic heading will be presented since it is powered by the essential busses.)

TACAN Control Panel

The TACAN control panel (figure 4-9), located on the right-hand console, has a power selector knob with OFF, REC, and T/R positions, two channel selector controls, a channel window and a volume-control knob. With the power selector in the REC position, the range function of the set is disabled, and only bearing information is available. With the power selector in the T/R position, both bearing and range information is displayed on the indicators. The large channel selector control selects the first two figures of the TACAN beacon channel number, and the small channel selector handle selects the third number. The volume control knob is used to adjust the volume of identification signals received from the TACAN surface beacon. On TF airplanes, the TACAN control panel is located on the center console.

Instrument Selector Switch

A two-position instrument selector switch labeled INST SELECT is located on the right hand console (figure 4-10). On TF airplanes, the instrument selector switch is located on the center console. The two positions are labeled TACAN and ILS. With the switch in the TACAN position, the word TACAN will be illuminated and TACAN course information will be displayed on the course indicator. With the switch in the ILS position, ILS will be illuminated and ILS localizer course information will be displayed on the course indicator. The intensity of these lights is controlled by the console light rheostat.

Note

The position of the switch has no effect on TACAN range and bearing display.

Radio Magnetic Indicator

The radio magnetic indicator (RMI) (figures 1-8 and 1-9) includes a rotating compass card and two TACAN bearing pointers. The rotating compass card is actuated by the heading indicator system. The single-barred bearing pointer circuit is wired in parallel with the

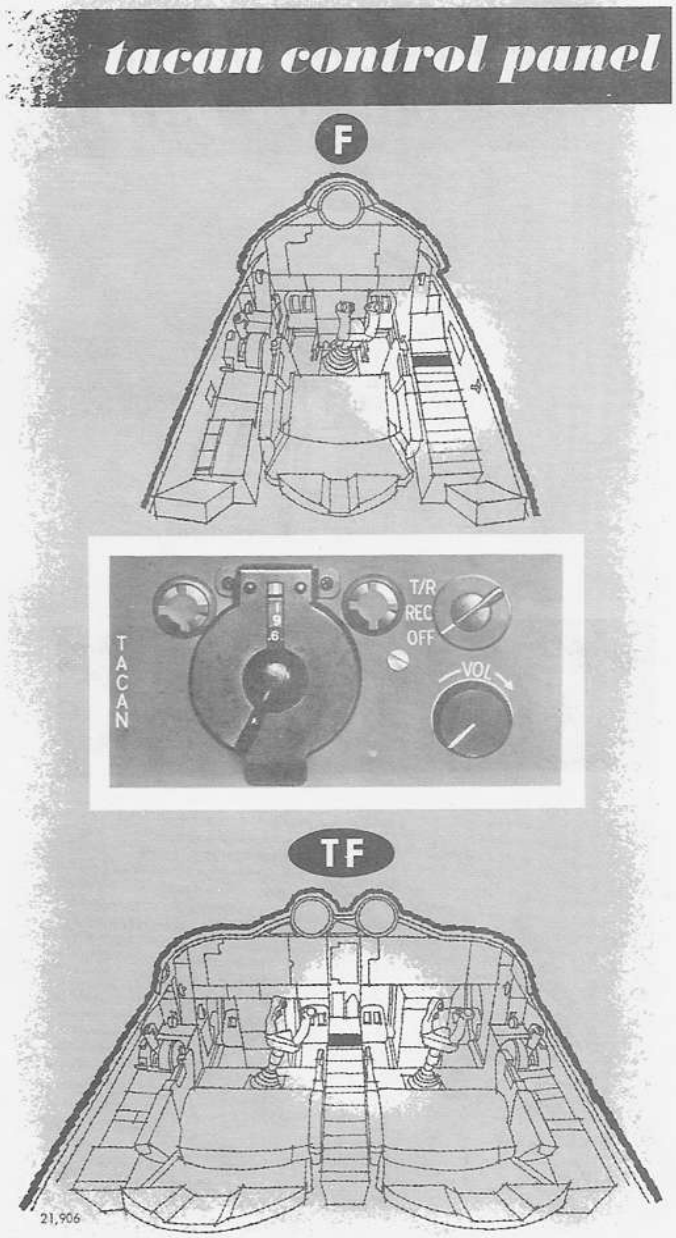


Figure 4-9

double-barred bearing pointer and both (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 bearing pointers to indicate the magnetic bearing to 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 indicator.

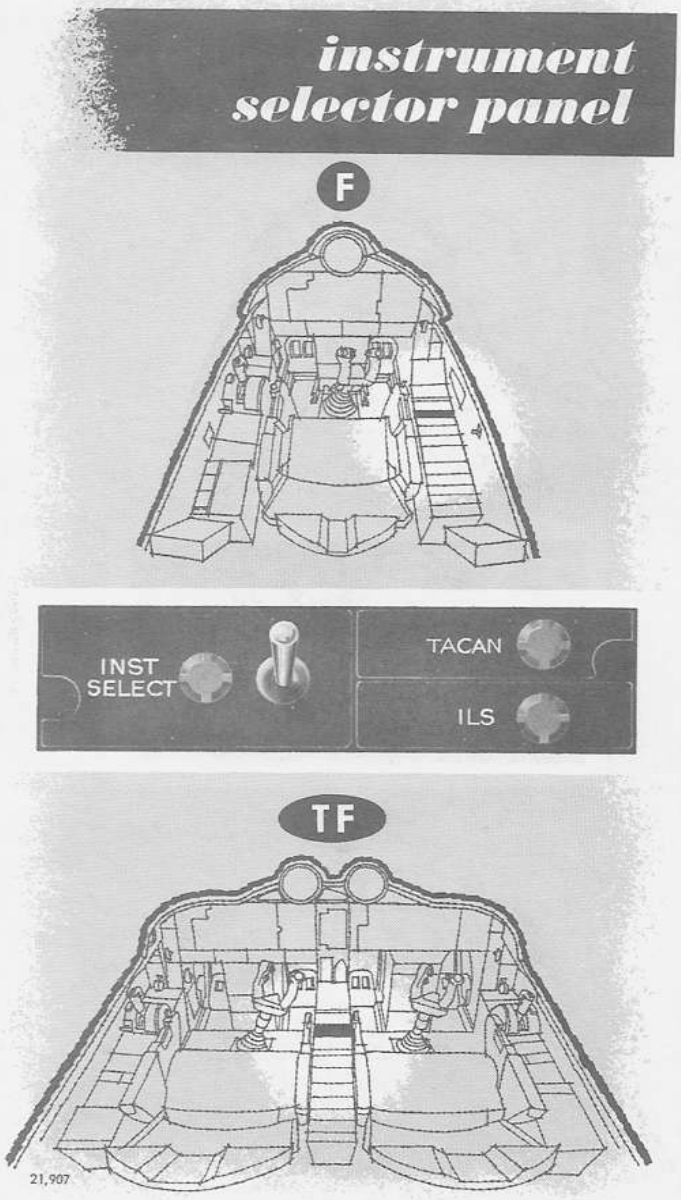


Figure 4-10

TACAN Range Indicator

The TACAN range indicator panel (figure 4-11) is installed on the center console on TF airplanes and on the instrument panel on F airplanes. Night lighting is provided by two panel edge lights, and on some airplanes, a light mounted on the upper right side of the range indicator. The lights are controlled by the console lights rheostat or the flight instrument panel lights rheostat respectively. 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 transmission 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 range indicator is partially covered by a range warning flag, which warns against reading incorrect range indications.

Course Indicator

The course indicator (figures 1-8 and 1-9) has a course set knob, a course selector window, a "TO-FROM" indicator, a course deviation indicator (CDI), a glide-slope indicator, a heading pointer and a marker beacon light. The course set knob permits selection of a desired course, which is displayed in the course window. The "TO-FROM" indicator indicates whether the course selected (if intercepted and flown) is to or from the station. The CDI shows the position of the airplane in relation to the selected course. The glide-slope indicator is used when flying an instrument landing system approach. The heading pointer, actuated by the heading indicator system, shows the heading of the airplane relative to the selected course. The marker beacon light is illuminated when signals are received by the marker beacon receiver. When the CDI is centered, the airplane is on the selected course. If the heading 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 heading scale. When radio signals are unreliable or too weak, a course warning (OFF) flag appears at the top end of the CDI or a glide slope warning flag appears at the left end of the glide slope indicator, depending on which system is unreliable. On TF airplanes, a course indicator is installed on each instrument panel. On some airplanes, the instructor's course indicator does not have a course set knob, a course selector window, or a "TO-FROM" indicator. On other airplanes, the pilot's and instructor pilot's course indicators are identical.

Operation of AN/ARN-21

Note

It is possible that improperly adjusted or malfunctioning ground or airborne TACAN equipment may lock-on to a false bearing. The error will probably be plus or minus 40° but can be any value which is a multiple of 40° and can be to either side of the correct bearing. The possibility of a wrong 40° lock-on is inherent in the TACAN system and can be completely avoided only by the use of other navigation equipment in addition to TACAN. Therefore, attempt to verify TACAN bearing information whenever possible. After takeoff, the TACAN should be cross-checked with ground radar, airborne radar, or known checkpoint. When using TACAN for instrument departures, penetrations or let-downs, utilize airborne radar monitor or ground

radar monitor when possible to verify TACAN bearing information. When improper lock-on is detected, switching to another channel and back again or turning the set off, then back to on, will recycle the search mode. This will probably result in a correct lock-on.

1. Instrument selector switch to TACAN.
2. Power switch to T/R or REC. Expect a 100 second delay for TACAN to warm up.
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 is not available if the first channel selected is used.

4. Select desired beacon channel.
5. Adjust volume to desired level.
6. To turn equipment off, position power switch to OFF.

ILS RECEIVERS — AN/ARN-31

AN/ARN-31 is installed on some airplanes and 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 localizer and glide-slope guidance to a pilot making an instrument landing system approach. The localizer and glide-slope operate on 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, one at the upper end of the CDI and one at the left end of the glide-slope indicator, are visible 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.

ILS Control Panel

The ILS control panel (figure 4-12) 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. On TF airplanes, the ILS control panel is located on the center console.

Instrument Selector Panel

A two-position switch (figure 4-10), on the right-hand console, controls relays that connect the course indicator

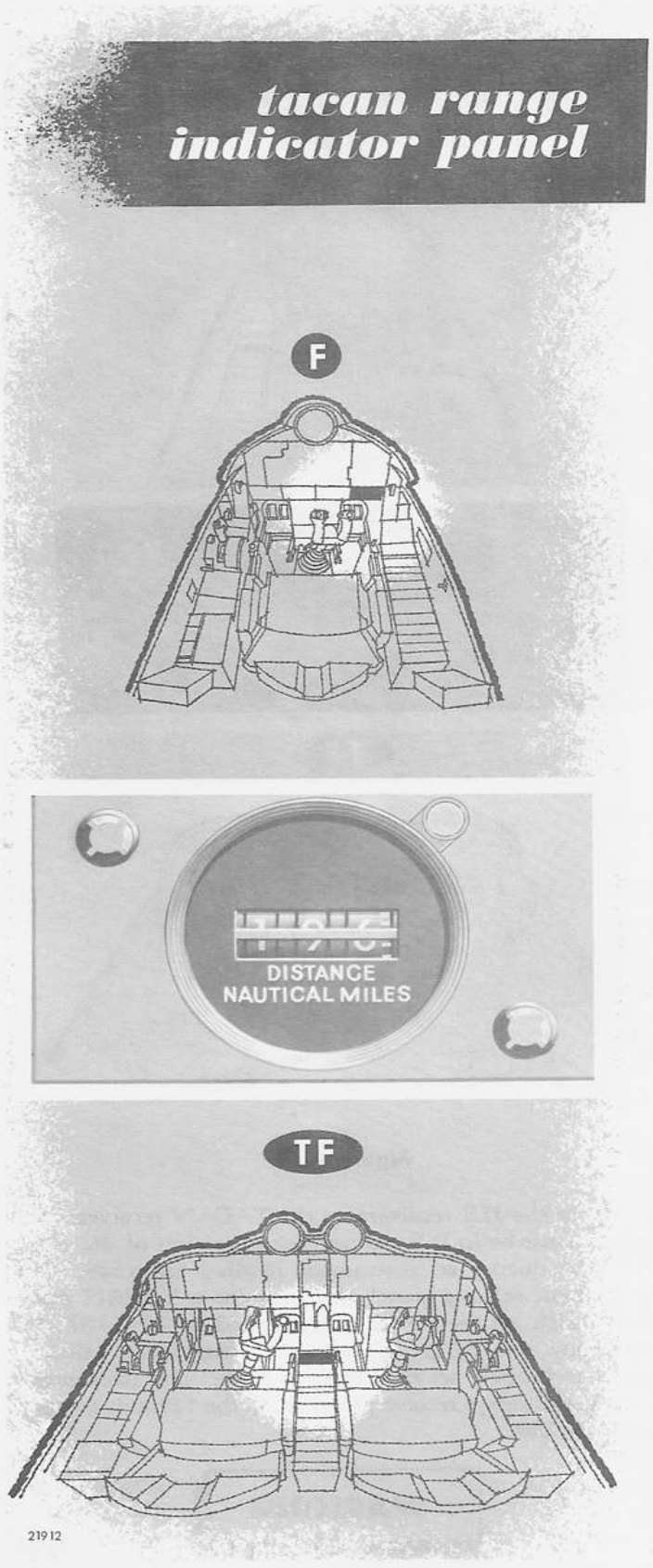


Figure 4-11

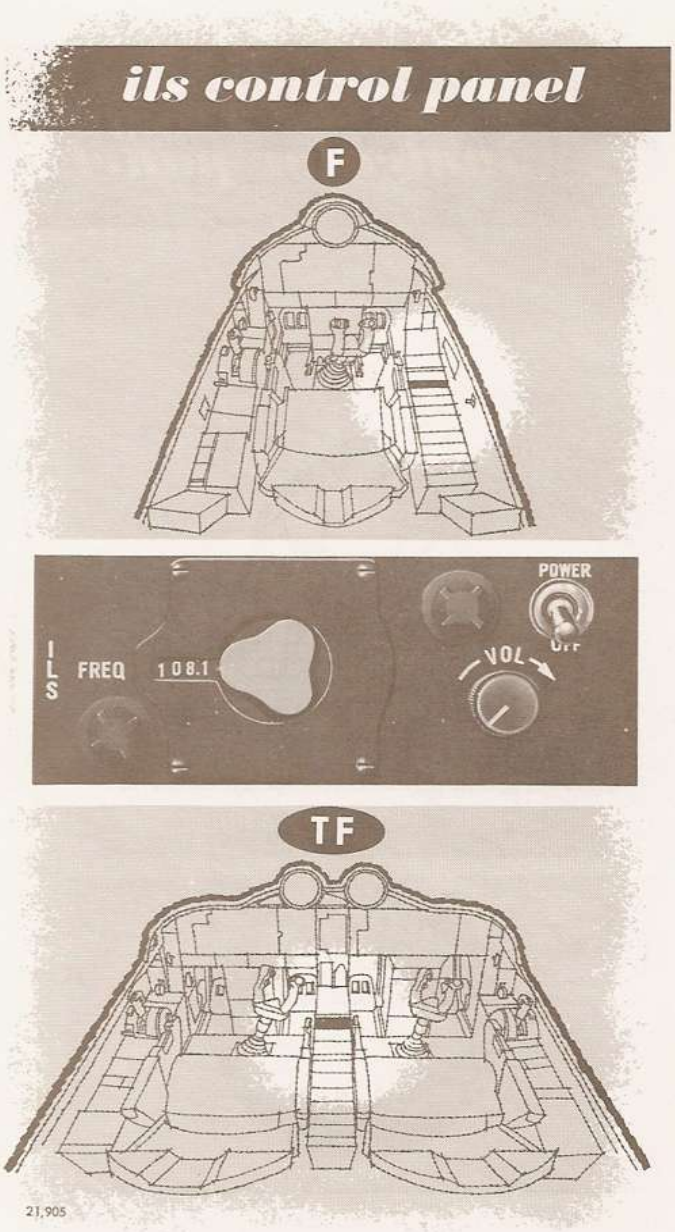


Figure 4-12

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 CDI to respond to AN/ARN-21 functions. On TF airplanes, the instrument selector panel is located on the center console. 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 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 course 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 course set knob a few degrees away from the inbound course. If the CDI responds to the course change, the signal was being received from a TACAN station and not the localizer. If the CDI does not respond to the inbound course 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 glide-slope warning (OFF) flag appearing and the glide-slope indicator will center itself and remain centered regardless of airplane movement. It will be impossible for TACAN signal information to be inadvertently displayed on the CDI if a power failure occurs in the ILS power circuit with the TACAN ILS indicator selector switch in the ILS position.

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 course in course selector window.
5. Set is ready for operation when warning flags are no longer visible.
6. 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. On TF airplanes, a marker beacon light is installed on both the pilot's and instructor's course indicators. 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 (SOME AIRPLANES)

The AN/APX-6A, AN/APX-25 radar sets identify the airplane automatically according to prearranged modes of operation when challenged by suitably equipped ground or airborne units. Interrogation signals are decoded and coded signals sent in return. If the IFF master control knob is positioned to NORM or LOW, an emergency

distress signal is transmitted when the canopy is jettisoned or unlocked. 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 4-13) 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

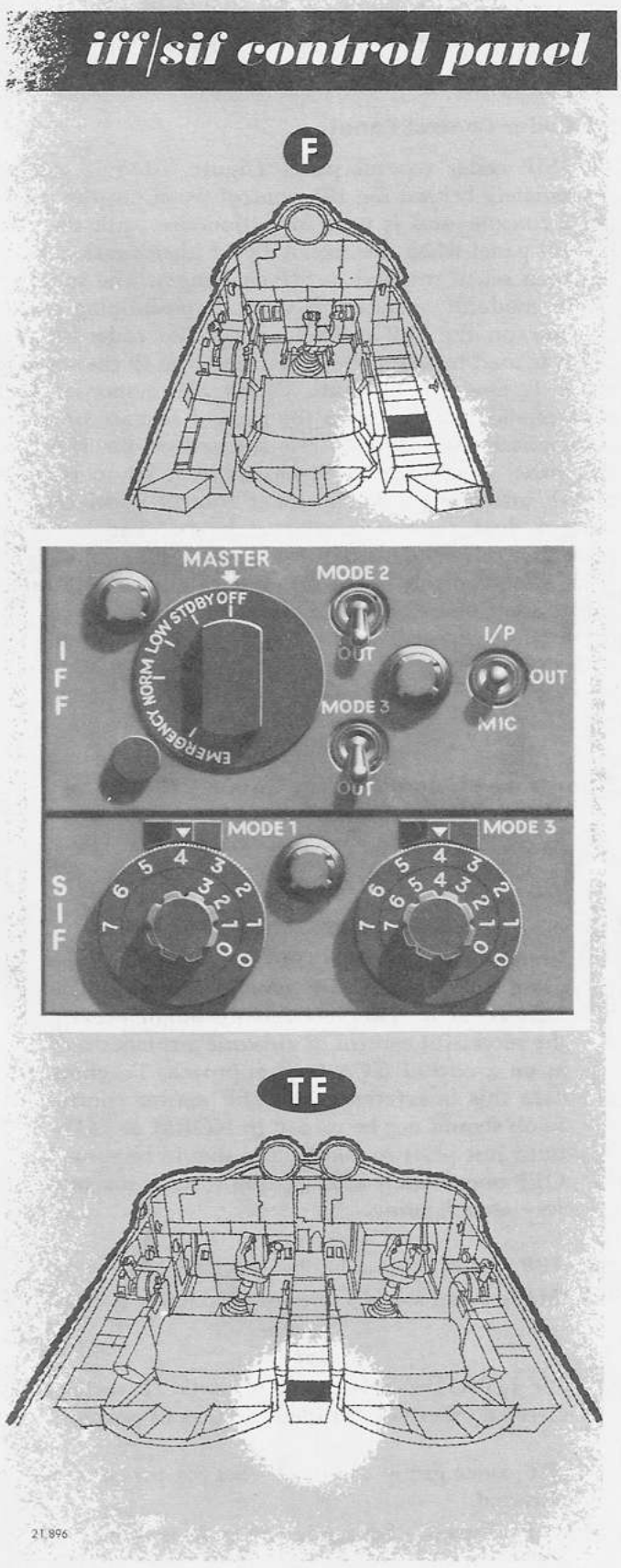


Figure 4-13

aid ground and airborne air-controllers with special position identification problems and should be used only as directed. On TF airplanes, the IFF control panel is located on the center console.

SIF Coder Control Panel

The SIF coder control panel (figure 4-13) 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 control 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. On TF airplanes, the SIF coder control panel is located on the center console.

Operation of Identification Radar (IFF-SIF)

WARNING

Improper use of the IFF/SIF equipment can cause interference with ground radar displays such as GCA. This interference might prevent the successful control of airborne airplanes such as on a critical GCA final approach. To eliminate this interference, the IFF master control knob should not be turned to NORM or LOW until just prior to takeoff and should be turned OFF immediately after taxiing clear of the runway after landing.

1. IFF master control knob — As desired.
2. Mode 2 and mode 3 selector switches — As required.

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 — Set mode and code as directed, press dial stop plunger button and rotate master switch to EMERGENCY position. Set will then automatically transmit distress signals when interrogated.

Note

If the master control knob is in NORM or LOW, the emergency distress signal is transmitted when the canopy is jettisoned or unlocked.

AN/APX-72 IFF SYSTEM (SOME AIRPLANES)

Note

Installation of the AN/APX-72 IFF system is a part of the AIMS modification.

The AN/APX-72 IFF system provides automatic radar identification, emergency identification, and automatic altitude reporting. The system receives challenges and automatically transmits responses to suitably equipped interrogating facilities within the line-of-sight operating range of the system. An ejection relay is activated if the canopy is unlocked and the left main landing gear is in the up position. Regardless of APX-72 control positions (other than power off), energizing the ejection relay will: (1) switch the APX-72 from low to high sensitivity; (2) enable modes 1, 2, and 3/A; (3) set emergency mode and mode 3/A to code 7700; and (4) zeroize mode 4 codes. The IFF control panel (figure 4-14) is located on the right console on F airplanes and on the center console on TF airplanes. The panel contains the master control knob; enabling switches for modes 1, 2, 3/A, 4, and C; radiation test switch, modes 1 and 3/A code switches, identification switch, mode 4 reply indication switch, mode 4 code selector knob, mode 4 reply light, and a test light. The system receives power from the 28-volt dc essential bus and the 115-volt ac essential bus to provide operation in the following modes.

Modes 1, 2, and 3/A

Modes 1, 2, and 3/A are the modes which perform the conventional IFF functions of the system. There are provisions for selection of multiple codes within each of these operational modes. Mode 1 incorporates a total of 32 different codes with selection being made on the IFF control panel. Modes 2 and 3/A provide 4096 codes each. Mode 2 code selection is made on the front panel of the receiver-transmitter. Mode 3/A codes are selected on the IFF control panel. A self-test function is available in these modes.

Mode C

Mode C provides an altitude reporting capability. This mode enables the system to report the aircraft pressure altitude to the interrogating agency using data supplied by the altitude computer CPU-46/A. A self-test function is available in mode C.

Mode 4

Mode 4 provides a military secure (encrypted) IFF capability. A mode 4 interrogation is received and processed through a special transponder computer which encodes

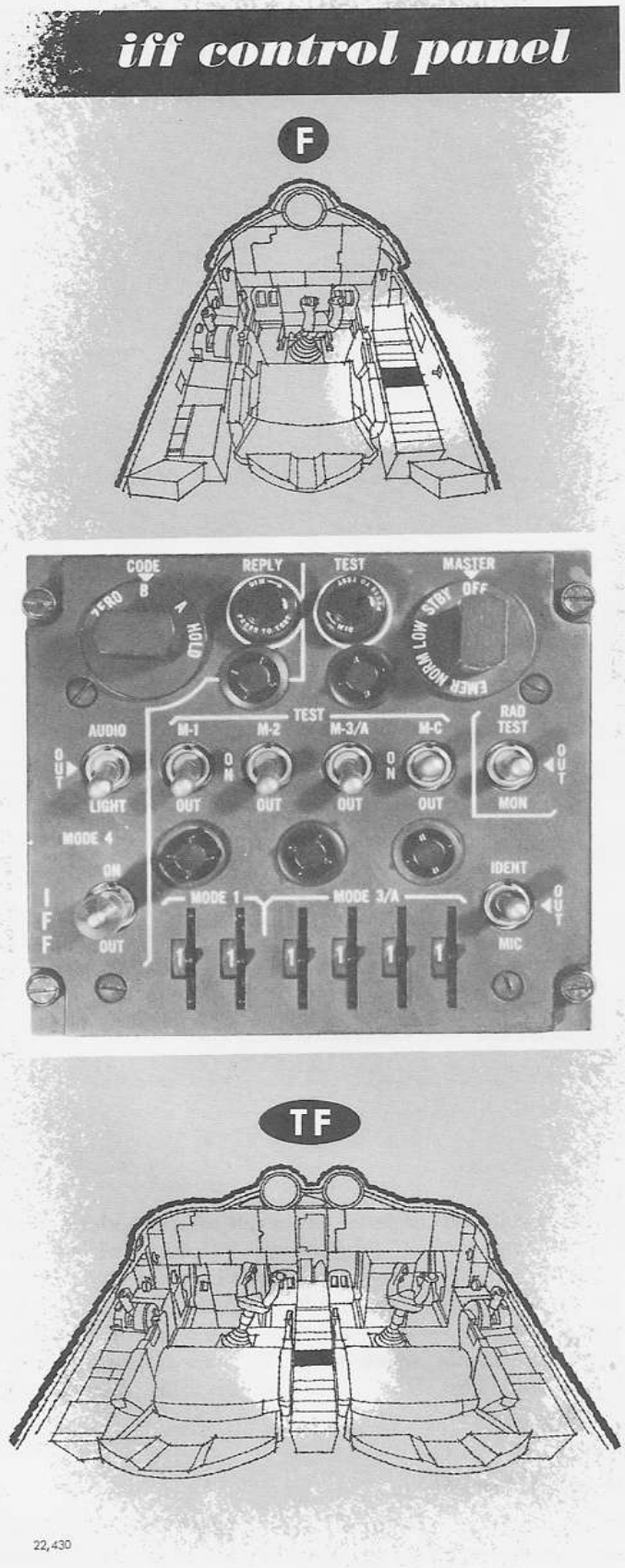


Figure 4-14

and triggers the proper identification response signal. Mode 4 operates in either of two codes. The codes, A and B, are inserted into the transponder computer and are changed periodically by ground personnel. The mode 4 code settings are normally zeroized when power is removed from the system. However, if it is desired to retain the code settings (such as for additional flights during a specified period) a code-hold function is available. When the landing gear is down, the code-hold function may be selected and (if power is left on the system for 15 seconds) the codes will be retained after power is off. Mode 4 codes also zeroize in event of ejection or may be zeroized manually by switch positioning.

IFF Master Control Knob

The IFF master control knob (figure 4-14) is a rotary switch placarded MASTER and has positions of OFF, STBY, LOW, NORM, and EMER. In the OFF position, all electrical power is removed from the AN/APX-72 system. The STBY position enables the receiver and computer but does not enable the transmitter. The LOW position provides operation in any mode but with reduced receiver sensitivity. In this position the system responds only to strong, local interrogation. The NORM position provides operation in any mode at normal receiver sensitivity. The EMER position provides automatic transmission of emergency response signals to modes 1, 2, or 3/A interrogations regardless of the mode switch settings. To prevent accidental selection of the OFF or EMER positions, the knob must be raised to clear stops before it can be moved to either of these positions.

Mode Switches

The mode switches (figure 4-14) are placarded M-1, M-2, M-3/A, and M-C and have positions of OUT, ON, and TEST. The ON position enables the receiver-transmitter to reply to interrogations for the selected mode. The OUT position disables the selected mode. Positioning to the spring-loaded TEST position triggers the transponder test set to generate a simulated mode interrogation for self-test of the selected mode. Correct receiver-transmitter response is indicated by illumination of the system test indicator light.

Radiation Test Monitor Switch

The radiation test monitor switch (figure 4-14) has placarded positions of RAD TEST, MON, and OUT. Switch positioning to RAD TEST permits the transponder to be interrogated by signals from test equipment. The MON position enables the monitoring circuits of the transponder test set. The OUT position disables the switch functions.

Mode 1 Code Switches

The mode 1 code selectors (figure 4-14) consist of two thumbwheel switches that are placarded MODE 1. The switches provide a selection of any one of 32 (00 through

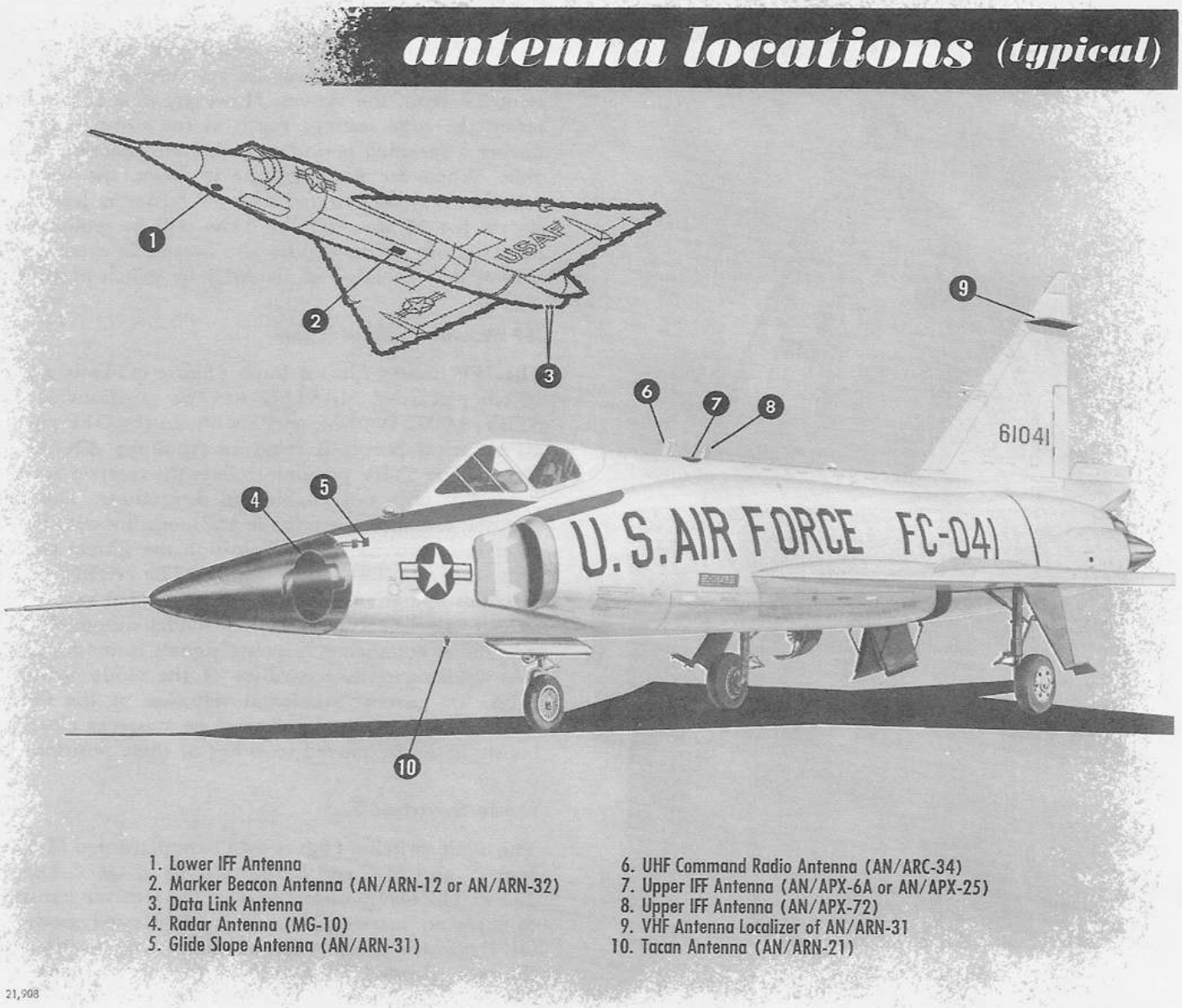


Figure 4-15

73) codes. The selected code number is displayed in windows adjacent to the switches.

Mode 3/A Code Switches

The mode 3/A code selectors (figure 4-14) are four thumbwheel switches that are placarded MODE 3/A. The switches provide a selection of any one of 4096 (0000 through 7777) codes. The number of the selected code is displayed in windows adjacent to the switches.

Identification Switch

The identification switch (figure 4-14) has placarded positions of IDENT, OUT, and MIC. The switch, which selects an identifier pulse, does not affect modes C and 4. Momentary switch positioning to the spring loaded IDENT position causes an identification reply pulse for 15 to 30 seconds on the mode and code selected. The MIC position causes an identification reply pulse of the selected mode and code when the UHF transmitter button is depressed. The OUT position disables the identifier feature.

Mode 4 Switch

The mode 4 switch (figure 4-14) is placarded MODE 4 and has placarded positions of ON and OUT. Switch positioning to the ON position enables the receiver-transmitter to reply to mode 4 interrogations. The OUT

position disables the mode 4 operation. To prevent accidental selection of the OUT position, the switch lever must be raised to clear a stop before it can be moved to OUT.

Mode 4 Reply Indication Switch

The mode 4 reply indication switch (figure 4-14) has placarded positions of AUDIO, LIGHT, and OUT. The AUDIO position enables both the aural monitoring (short bursts of aural tone in the headset) of mode 4 interrogations and reply light monitoring of mode 4 replies. The LIGHT position provides only mode 4 reply light monitoring. The OUT position disables mode 4 aural and visual monitoring.

Mode 4 Code Selector Knob

The mode 4 code selector knob (figure 4-14) is placarded CODE and has positions of HOLD, A, B, and ZERO. Positioning of the knob to HOLD, and holding against the spring load, overrides the zeroizing function and enables the transponder computer to retain mode 4 code settings when aircraft power is removed. The landing gear must be down and locked when the hold function is engaged. Knob position A enables the system to respond to code A interrogations. Knob position B enables the system to respond to code B interrogations. Knob position to ZERO cancels mode 4 code settings from the transponder computer providing other operational requirements are met. To prevent accidental selection of the ZERO position, the knob must be raised to clear a stop before it can be moved to ZERO.

Mode 4 Reply Light

The mode 4 reply light (figure 4-14) is placarded REPLY and lights green when the receiver-transmitter has transmitted a proper reply to a mode 4 interrogation. The lamp has a press-to-test feature.

Test Light

The test light is placarded TEST (figure 4-14). The lamp lights green when a proper reply is made to a mode 1, 2, 3/A, or C test interrogation. The lamp has a press-to-test feature.

IFF Caution Light

The amber colored IFF caution light is located on the F-102A right console or the top left side of the instructor's instrument panel on TF-102A airplanes. The lamp is a press-to-test type and is placarded IFF CAUTION. The lamp lights to indicate that the AN/APX-72 system has failed to respond to a valid mode 4 interrogation at any time the system is in a mode 4 operating configuration. Specifically, the light illuminates if mode 4 codes are zeroized, the transponder fails to reply to proper interrogation, or the automatic self-test function of the transponder computer reveals a malfunction.

Note

Since power for the caution light is routed through the transponder computer, the computer must be physically installed in the airplane before the caution light will operate.

IFF Antenna Selector Switch

The IFF antenna switch is located on the F-102A right console and on the top left side of the instructor's instrument panel on the TF-102A. The switch is placarded IFF ANT and has positions UPPER, BOTH, and LOWER. On F-102A airplanes, the switch positions are left, right and center. On TF-102A airplanes, the switch is moved up, down, and center to select the desired positions. The switch selects, through the antenna switching unit, the antenna to be used for transmitting and receiving IFF signals. The normal position is BOTH. However, UPPER or LOWER may be selected to assure the best orientation toward the interrogating facility.

Normal Operation of AN/APX-72 IFF System

1. IFF master control knob—STBY for minimum of two minutes warmup time.
2. Mode 1 and mode 3/A code switches—As desired.
3. IFF master control knob—NORM.
4. IFF antenna selector switch—BOTH.
5. Identification switch—OUT.
6. System self-test:
   a. Radiation test switch—OUT.
   b. Momentarily hold mode 1 switch to TEST, check for illumination of the test light, then return switch to OUT. Repeat for modes 2, 3/A, and C.
7. Mode switches—As desired.
8. Radiation test switch—OUT or MON as desired.
9. Mode 4—Select desired operation.
   a. Prior to flight assure that the KIT-1A/TSEC transponder computer is installed and properly keyed.
   b. Mode 4 switch—ON.
   c. Mode 4 code selector knob—A or B as desired.
   d. Mode 4 reply indication switch—AUDIO or LIGHT as required.
   e. To hold mode 4 codes after landing—Mode 4 code selector knob to HOLD for five seconds prior to removing power from the system.
10. To terminate system operation, master control knob —OFF.

Emergency Operation of AN/APX-72 IFF System

For emergency operation, pull upward on the master control knob and rotate to EMER position. Modes 1, 2, and 3/A are enabled automatically. Modes 1 and 2 will reply

to interrogations with the reply code selected by their respective code switches. Mode 3/A reply code will be 7700 regardless of the position of the mode 3/A code switches.

LIGHTING EQUIPMENT

EXTERIOR LIGHTING

Exterior lighting consists of two anticollision lights on the fuselage, two wing tip lights, two upper and two lower fuselage lights, four tail lights (two on each side of the tail cone fairings), two landing lights, and one taxi light. The upper tail light on each side is amber; the lower tail light on each side is white. One of the anticollision lights is located on the upper centerline of the fuselage aft of the cockpit; the other is located on the lower centerline of the fuselage aft of the main wheel well. The anticollision lights remain extended at all times. The four fuselage lights, located aft of the cockpit, are installed in upper left and right, and lower left and right positions. Placing the position lights switch to the NAVIGATION position will cause both anticollision lights to illuminate and rotate. When the position lights switch is placed in the FORMATION position, the anticollision lights cease rotating. 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 are operative if the nose landing gear is up and locked. The landing and taxi lights are controlled by a single selector switch.

Position Light Switches

Two position light switches (figure 1-26) are located at the bottom right corner of the utility switch panel. The left switch has three positions: NAVIGATION, OFF (center), and FORMATION. In NAVIGATION position the anticollision lights illuminate bright and rotate, the wing tip lights illuminate bright-steady, the four fuselage and four tail lights are off, and the BRIGHT-DIM switch will be ineffective. In FORMATION position, the anticollision lights (dim, non-rotating), wing tip lights, four fuselage lights, and the four tail lights illuminate; the BRIGHT-DIM switch will be effective for all but the beacon. The NAVIGATION position controls 115-volt ac essential bus power to the transformer for operation of wing tip, fuselage, and tail lights, and 28-volt dc essential bus power for operation of the external lights power relay and nonessential bus power for the beacon. The FORMATION position controls 115-volt ac essential bus power to the transformer for illumination of wing tip lights, fuselage lights, white and amber tail lights; and 28-volt dc essential bus power for illumination of the anticollision 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 1-29) 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. On TF airplanes, the switch is locked on the pilot's landing gear control panel.

INTERIOR LIGHTING

Interior lighting equipment includes the edge-lighting for the instrument panels, switch panels and consoles, the thunderstorm lights, cockpit floodlights, and the magnetic 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. On TF airplanes, a hand light with flexible cord, which can be used for map reading or troubleshooting, is located on the rear bulkhead between the pilot's and instructor's seats. The light has its own control switch and adjustable red filter, and gets its power from the 28-volt dc nonessential bus.

Flight Instrument Panel Lights Rheostat

The flight instrument panel rheostat (figures 1-8 and 1-9) 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 magnetic compass light, when the compass light switch is in the ON position. Power to this rheostat comes from the 28-volt dc essential bus. On (TF) airplanes, the rheostat controls the intensity of the lights that illuminate the pilot's flight instrument group.

Magnetic Compass Light Switch

The magnetic compass light switch has ON and OFF positions and is located on the right-hand console. On (TF) airplanes, the compass light switch is located on the warning light panel between the pilot's and instructor's instrument panels. The switch is wired in series with the flight instrument panel lights rheostat circuit.

Forward Panel Lights Powerstat (F)

A powerstat placarded "Fwd Panels" at the top right corner of the lighting control panel (figure 1-8) 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, and 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, the TACAN range indicator 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.

Forward Panel Lights Powerstat (TF)

A powerstat placarded "Fwd Panels" at the top right corner of the lighting control panel (figure 1-9) controls the intensity of the lighting for the engine instrument group on the pilot's 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 pilot's radar scope, the far left side of the pilot's instrument panel, the warning light 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.

Instructor's Instrument Panel Lights Powerstat (TF)

A powerstat placarded "Instrument" with positions OFF and BRT, located on the lighting control panel (figure 1-9), controls the intensity of the lighting for the instructor's flight instruments. Power source for this powerstat is the 115-volt ac essential bus.

Console Lights Powerstat (F)

The console lights powerstat, located at the lower right corner of the lighting control panel (figure 1-8), 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.

Console Lights Powerstats (TF)

The pilot's console lights powerstat, located at the lower right corner of the lighting control panel (figure 1-9), is placarded "Console." Turning the knob clockwise from OFF to BRT controls the intensity of the lights on the left and center consoles, and also the lights on the upright panels at the forward ends of the two consoles. Power to this powerstat comes from the 115-volt ac essential bus. The instructor's console lights powerstat is located on the small panel below the instructor's instrument panel.

Cockpit Floodlights Powerstats (F)

Two floodlight powerstats are located at the left side of the lighting control panel (figure 1-8). 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 nonessential bus.

Cockpit Floodlights Powerstats (TF)

Two floodlight powerstats with positions OFF and BRT are located at the left side of the lighting control panel (figure 1-9). 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 the left and center consoles. Power to these powerstats comes from the 115-volt ac nonessential bus. Two similar powerstats located on the small panel below the instructor's instrument panel control the floodlights for the instructor's instrument panel and console.

Thunderstorm Lights Switch

A switch placarded "Storm Lights" is located above the canopy unlocked warning light on (TF) airplanes (figure 1-9), and on the utility switch panel on (F) airplanes (figure 1-26). The switch 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, 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. 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.

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 in the seat cushion-survival kit. 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 4-16. 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 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. 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. It is necessary to use an adapter to reduce the amount of oxygen pressure between the regulator and the oxygen mask. The oxygen system is controlled by a single switch on the oxygen control panel.

Liquid Oxygen Control Panel

The liquid oxygen control panel (figure 4-17), located on the left-hand console, 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 oxygen flow from the airplane supply to the oxygen regulator. The liquid oxygen content gage provides an indication of the storage container content and is calibrated in liters from 0 to 5.

Note

The liquid oxygen content gage will read between 4 to 5 liters on (F) airplanes and 8 to 10 liters on (TF) airplanes when the system is fully charged.

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 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 used with the oxygen mask. 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.

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.
3. After landing when oxygen is no longer desired, turn oxygen supply switch OFF before unfastening the mask.

liquid oxygen duration chart-hours

USING PRESSURE BREATHING OXYGEN MASK TYPE MS22001 (A-13A)

CABIN ALTITUDE—FEET	LIQUID CONTENT—LITERS										
	10	9	8	7	6	5	4	3	2	1	
35,000 AND ABOVE	31.4	28.3	25.2	22.0	18.9	15.7	12.6	9.4	6.3	3.1	EMERGENCY BELOW 1 LITER DESCEND TO ALTITUDE NOT REQUIRING OXYGEN
30,000	22.7	20.4	18.1	15.8	13.6	11.3	9.0	6.8	4.5	2.3	
25,000	17.5	15.7	14.0	12.2	10.5	8.8	7.0	5.2	3.5	1.8	
20,000	13.3	12.0	10.7	9.3	8.0	6.7	5.3	4.0	2.7	1.3	
15,000	10.7	9.6	8.6	7.5	6.4	5.3	4.3	3.2	2.2	1.1	
10,000	8.6	7.7	6.9	6.0	5.2	4.3	3.4	2.6	1.7	.9	
5,000	6.8	6.3	5.4	4.9	4.1	3.5	2.7	2.1	1.4	.7	
0	5.5	5.0	4.4	3.8	3.3	2.8	2.2	1.7	1.1	.5	
F	5	4.5	4	3.5	3	2.5	2	1.5	1		BELOW 1
TF CREW—2	10	9	8	7	6	5	4	3	2	1	

21,898

Figure 4-16

WARNING

The oxygen supply switch should neither be turned ON until immediately after the mask is closed nor turned OFF until immediately before opening of the mask. This will prevent a rapid depletion of the oxygen supply and also prevent the low temperatures from damaging various components of the liquid oxygen system and causing personal injury.

Emergency Operation of Liquid Oxygen System

1. If symptoms of hypoxia develop, check oxygen hose connections and check supply switch ON.
2. If the airplane 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 when emergency oxygen supply is depleted.

3. If wearing an oxygen mask and cabin pressure is lost, an immediate descent to 25,000 feet or below is mandatory.

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-10 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

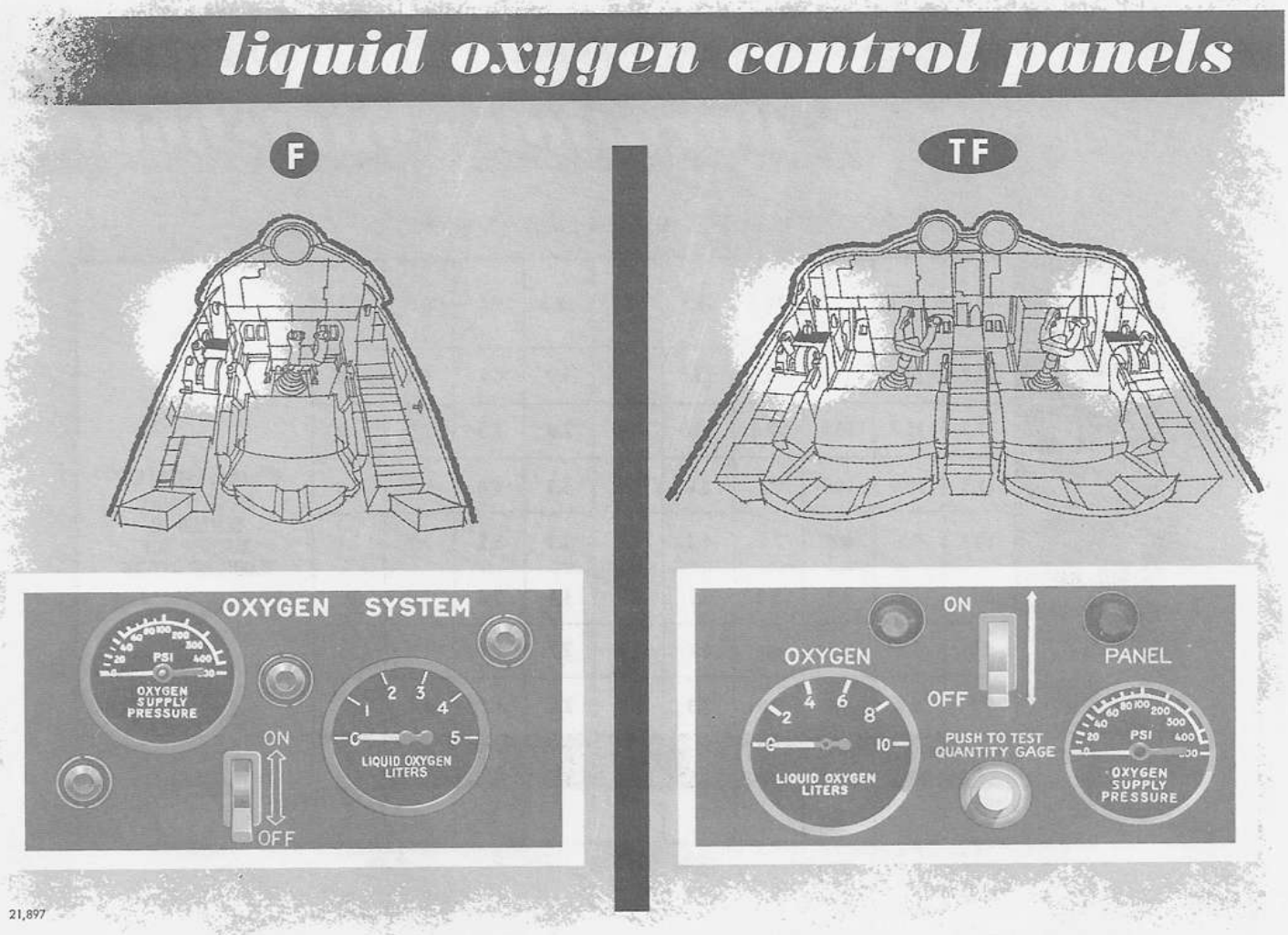


Figure 4-17

hydraulic actuators, to steer the desired course. The airplane damping system must be engaged and the FCS 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

The AFCS is 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. 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

oxygen mask connection

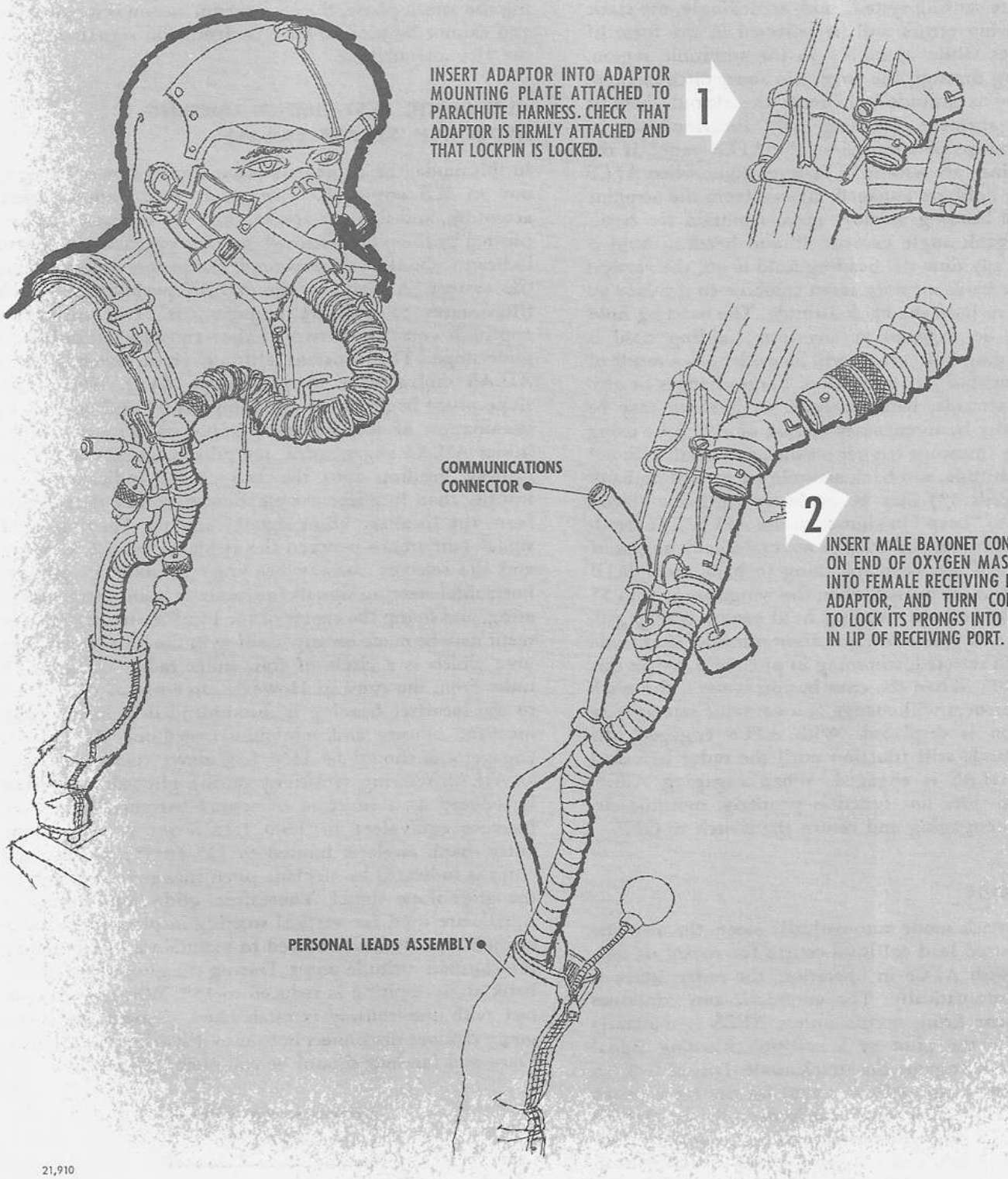


Figure 4-18

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. 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. 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 or altitude hold) mode. During automatic operation, electronic limiting provides g limits of 4.3 g to 0 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 radius 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 response 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, depress the emergency damper disconnect button to disengage the AILAS. Flare and landing should be completed manually.

CAUTION

Flare and landing must be accomplished manually as the landing gear will not withstand the impact at 175 KCAS 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 1-26) on the utility switch panel placarded "Pitch G Limiter Test" has positions +G, −G, and a spring-loaded (center) position. When the AFCS switch is engaged, wait for positive indication of engagement, then, placing the g limiter 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.

Note

If there is no positive indication of AFCS engagement, actuating the g limiter test switch to either +G or −G position will not actuate the g limiter. Only the AFCS switch will return to the OFF position.

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 to actuate the limit system. In the −G position, an input signal equivalent to −2g's is fed to the limiter to actuate the limit system. The pitch g limiter test switch is operated by the 28-volt dc nonessential bus.

AUTOMATIC FLIGHT CONTROL SYSTEM SWITCH

The AFCS switch (figure 4-19), 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 4-19), located on the AFCS control panel, is a two-position switch with ALTI-

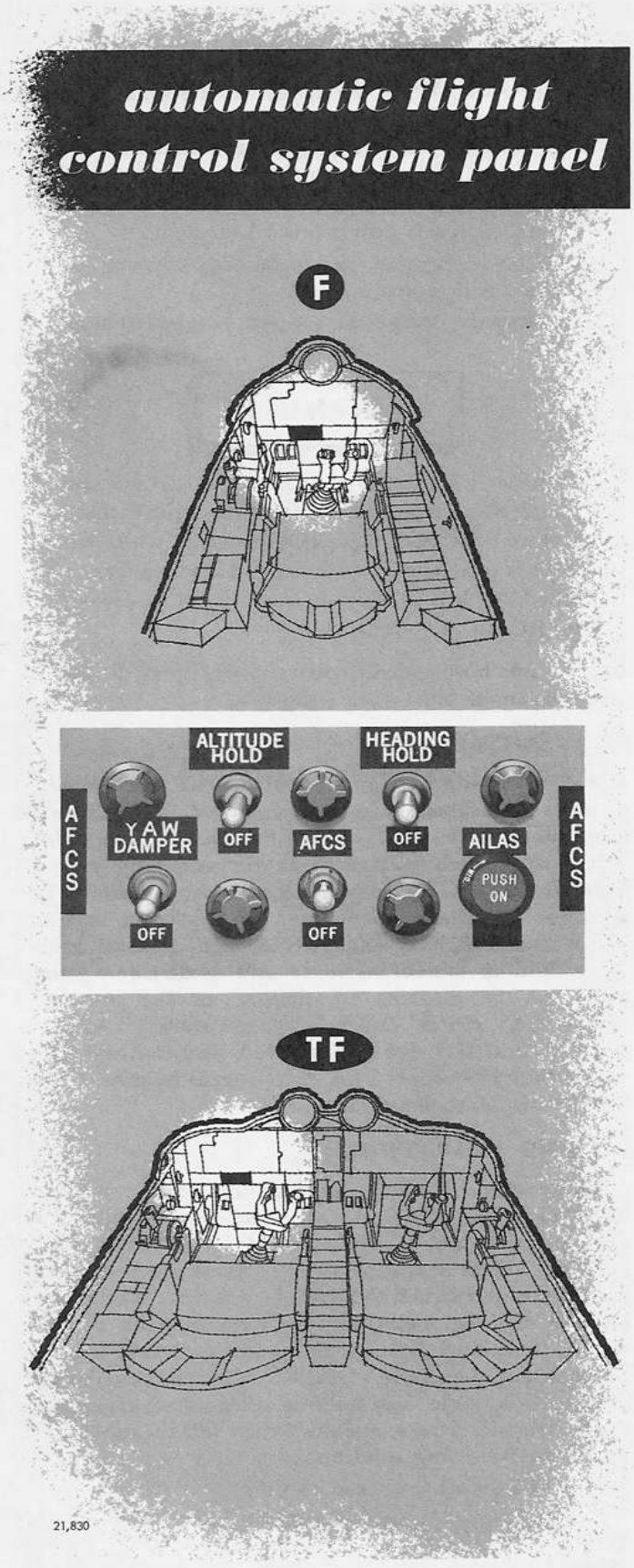


Figure 4-19

TUDE 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. The airplane is beep-trimmed in pitch.
2. AILAS is engaged with glide-slope capture and the switch will return to OFF.
3. Emergency damper disconnect button is depressed.

WARNING

If the switch is inadvertently turned to OFF and the airplane trimmed in pitch, the altitude hold feature will disengage. If this occurs while the pilot is preoccupied with the radar scope, an unscheduled loss of altitude could result in descent to ground or water level.

The altitude hold switch receives power from the 28-volt dc nonessential bus.

AILAS BUTTON

The AILAS button (figure 4-19) 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

A two-position heading hold switch (figure 4-19) is installed on the AFCS control panel. The switch has an OFF position and a solenoid-held HEADING HOLD position. When operating in pilot assist mode and with the switch in HEADING HOLD position, the airplane heading will be maintained as long as bank angle is within 5° of level flight. If the switch is in OFF position or bank angle exceeds 5°, AFCS will maintain the prevailing bank angle. The holding solenoid which engages the switch will release, and the switch will return to OFF, under the following conditions:

1. AFCS switch returned to OFF.
2. Emergency damper disconnect button depressed.

The heading hold switch is powered by the 28-volt dc nonessential bus.

AUTOMATIC FLIGHT CONTROL SYSTEM PREFLIGHT

Refer to Section II for this procedure.

AUTOMATIC FLIGHT CONTROL SYSTEM NORMAL OPERATION

Refer to Section II for this procedure.

NAVIGATION EQUIPMENT

MAGNETIC COMPASS

Refer to INSTRUMENTS in Section I.

HEADING 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 4-20) 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 magnetic mode of the J-4 compass system also serves as a directional reference for the automatic flight control system and supplies information to the VOR/TACAN 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 (figures 1-8 and 1-9). The system is energized whenever power is supplied to the essential buses. On TF airplanes, the J-4 compass control panel is located on the center console.

Heading 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 4-20), located on the J-4 compass control panel, has positions

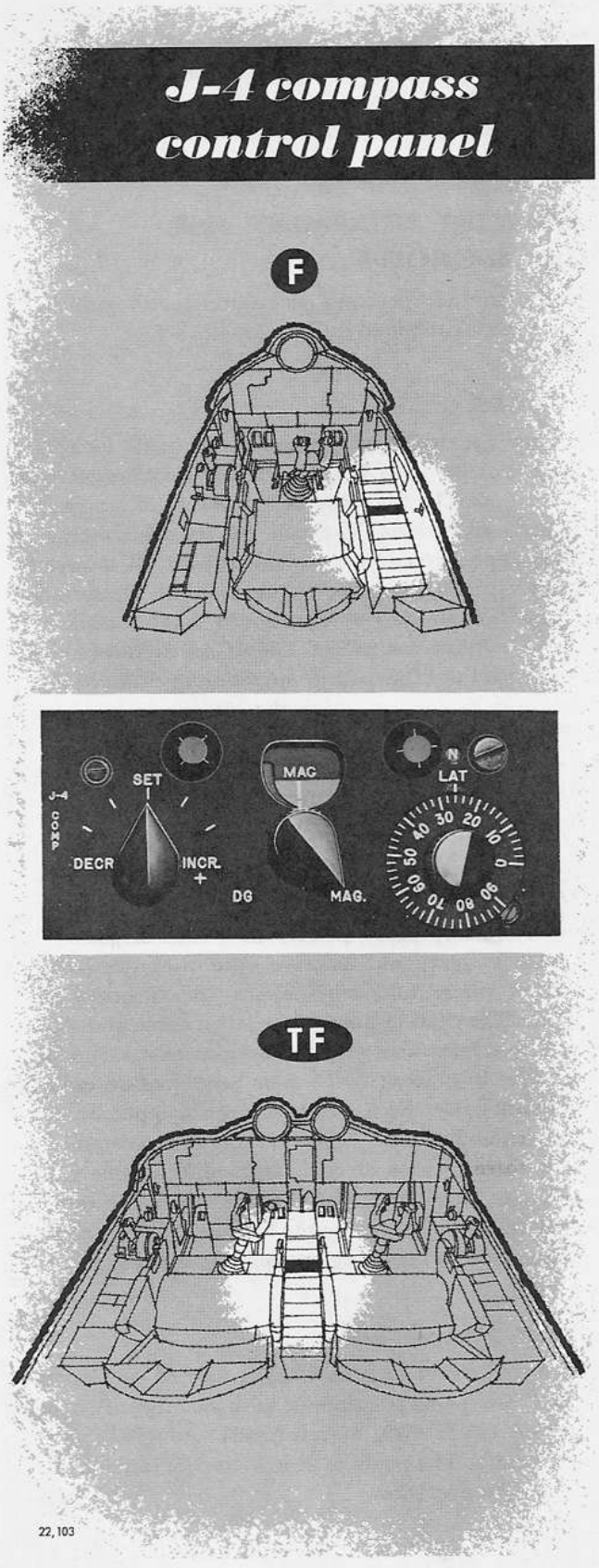


Figure 4-20

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 4-20), 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 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 4-20) 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 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 4-20) on the compass control panel is to select the operating airplane's hemisphere. A small window beside the screw displays "N" or "S" to indicate the hemisphere selected.

Latitude Selector Switch

The latitude selector switch (figure 4-20) 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

Fast Slave Ground Check

Refer to Section II for this procedure.

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, 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 AND FIRE CONTROL SYSTEM

Description and operating procedures for the armament equipment is contained in T.O. 1F-102A-1A.

ECM PODS

On some (F) airplanes* provisions are made for the installation of the AN/ALQ-72 electronic countermeasures (ECM) pod. The pod may be carried on either or both external tank pylons. The pod has its own internal ac power supply driven by a ram air turbine. DC power for control is provided by the dc essential bus. Operation of the pod is controlled by the C-6175/ALQ control indicator. A minimum speed of 250 KCAS should be maintained with the ECM pod in operation in order to insure satisfactory power output by the ram air turbine. Refer to T.O. 12P3-ALQ-72-2 for specific information regarding operation of the pod. Speed and acceleration limitations are the same as with external tanks installed and empty.

AN/ALE-2 CHAFF DISPENSER

On some (F) airplanes**, AN/ALE-2 chaff dispensers are provided to carry and dispense chaff packages into the airstream where turbulence opens and disperses their contents. The chaff dispensers are jettisonable pods which are 14 feet long and weigh approximately 375 pounds each when fully loaded. One may be carried on each external tank pylon. Each unit contains a stripper, an intervalometer, and a chaff hopper capable of accommodating two or three cartons of chaff, depending on the size of the cartons. Flush latches provide for removal of the nose and tail sections for loading the hopper and preparing the stripper without removing the unit from the airplane.

AN/ALE-2 Control Panel

The control panel (figure 4-21) is located on the right console. The panel is equipped with two sets of controls placarded UNIT NO. 1 (left-hand) and UNIT NO. 2 (right-hand). The panel is illuminated by internal lighting for night operation.

**Airplanes modified by TCTO 1F-102A-625 and TCTO 1F-102A-646.*

***Airplanes modified by TCTO 1F-102A-646.*

Auto-Manual Control Switch

The auto-manual control switch serves as the power control for the dispensing unit. When the switch is placed in the MANUAL position, the unit begins to dispense chaff at continuous speeds, automatic intervals, or programmed patterns as selected by the speed control. The AUTO position is not used on this airplane. Moving the switch to the OFF position de-energizes the dispensing unit, discontinuing chaff dispensing.

Speed Selector Knob

The speed selector knob is used to select the method and the chaff dispensing rate. The knob has 12 positions placarded around the dial in numerals 1 through 6, and letters A through F. The numerical indexes provide 6 choices of tape speed for continuous chaff dispensing. The alphabetical indexes provide selection of the time pattern desired for intervalometer operation.

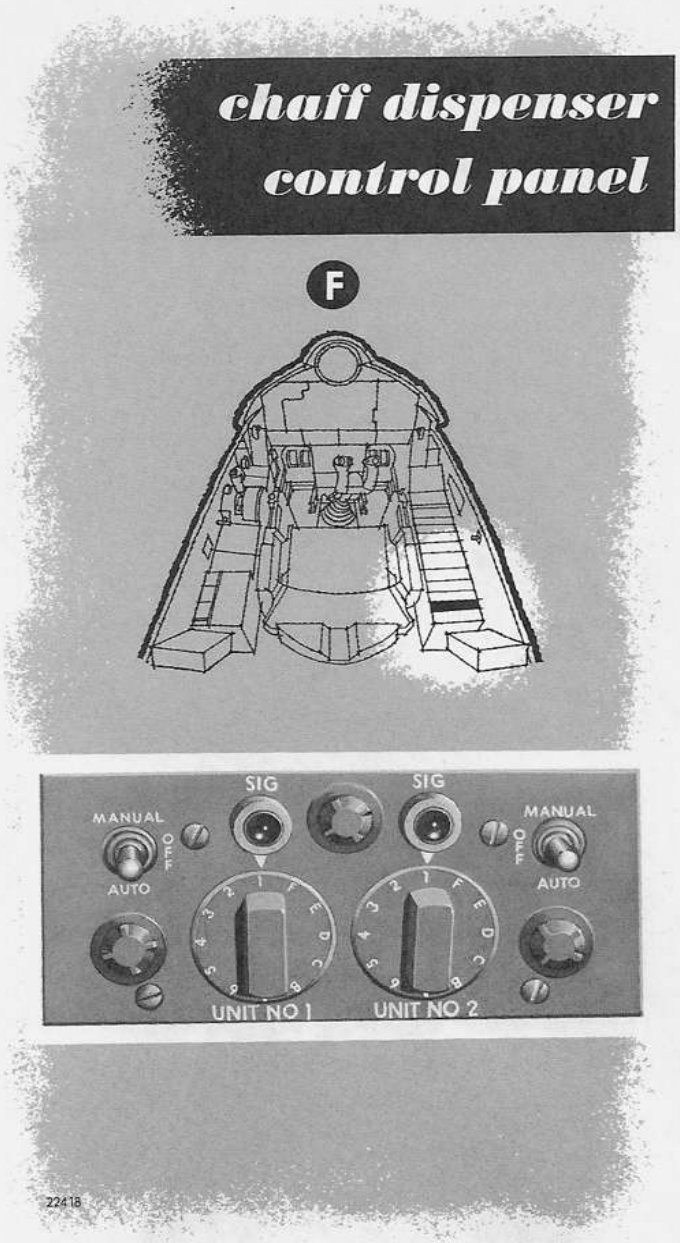


Figure 4-21

Signal Light

An amber light, placarded SIG, illuminates when the dispenser is in operation. Each blink of the light indicates the dispensing of one unit of chaff.

AN/ALE-2 Operation

To operate the chaff dispensing units, proceed as follows:

1. Preflight the dispensers. Check the load in the hopper and all latches, connections, and attachments for security. Remove the red nose and tail covers and check the main wheel well circuit breakers in.

CAUTION

Do not take off unless the red nose and tail covers are removed.

2. Adjust the speed selector knob according to the requirements of the mission.
3. When ready to dispense chaff, place the auto-manual control switch in the MANUAL position (according to the requirements of the mission) and observe the indicator light.

Note

AN/ALE-2 control panel circuit breaker is located in the left forward circuit breaker panel.

4. Place the auto-manual control switch in the OFF position to stop dispensing chaff. This should be done when countermeasure protection is no longer desired or any time the signal light stops blinking, indicating that the chaff supply is exhausted.

INFRARED COOLING

IR equipped aircraft contain a separate IR cooling switch in the cockpit. This switch, in conjunction with the FCS master power switch, controls operation of the closed cycle IR compressor. It must be in the ON position prior to operation of IR equipment.

ELECTRONIC COOLING

During normal flight conditions, the electronic compartments are cooled by a combination of ram air from the boundary layer ram air duct and cockpit exhaust air. This mixed air is circulated in the forward electronics compartment, through the nose wheel well, and into the intermediate and upper electronics compartments. A portion of the ram air is ducted directly to the upper and intermediate electronic compartments to provide adequate cooling. All expended cooling air flows into the armament bay and escapes overboard through the door hinges. A pressure and temperature controlled shutoff valve in the ram air duct regulates the pressure in the duct and shuts off the flow of ram air if its temperature exceeds 160-165°F. The aft electronics compartment is cooled by ram air from the engine air inlet ducts. Ground cooling of the electronic compartments with the engine operating is provided by a reverse flow of air in the ram air distribution ducts. The engine compressor creates a partial vacuum in the engine air inlet ducts which draws outside air from the main wheel well through the aft electronics compartment for cooling and from the open nose wheel well through the intermediate and upper electronics compartments and out a valve in the upper electronics compartment floor. The forward electronics compartment is cooled by a reverse flow of air caused by operation of the jet pump located in the boundary layer ram air duct. When the engine bleed air pressure falls

mask defog and anti-g suit control panels

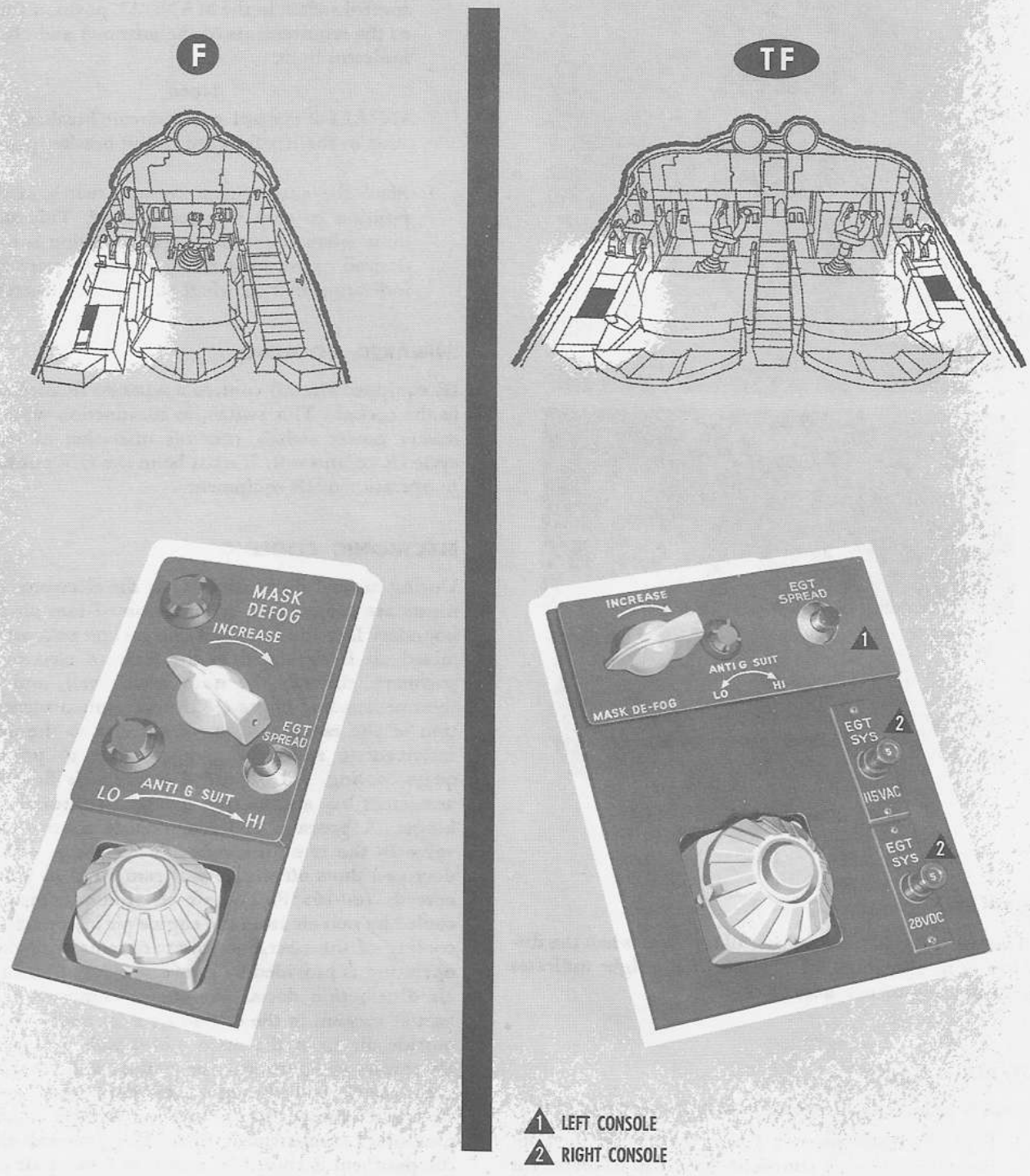


Figure 4-22

below 30-35 psi (engine rpm approximately 72% or below), the jet pump valve opens and allows engine bleed air to flow forward and overboard through the boundary layer ram air duct. A partial vacuum is thus created in the ram air duct drawing outside air from the open nose wheel well through the forward electronics compartment. Additional outside air is also drawn through the intermediate and upper electronics compartments to aid in their cooling. Operation of the jet pump is normally audible to the pilot. A pressure switch senses bleed air duct pressure and closes the jet pump valve when the duct pressure exceeds 30-35 psi (engine rpm above approximately 72%). Structure overheat detection probes, near the jet pump nozzles, sense excessive fuselage skin temperatures and automatically close the jet pump valve when excessive engine bleed air temperatures exist. Malfunction of the jet pump system is indicated by an electronics cooling light on the warning panel. During ground radar operations with the engine not running, a fitting in the nose wheel well provides for connection of a ground cooling unit.

ELECTRONIC COOLING WARNING LIGHT

An electronic cooling warning light is provided which will illuminate and display "ELECTRONIC COOLING" (figure 1-36) to indicate a malfunction in the electronic cooling system. Any time the light illuminates, it indicates a malfunction either in the electronic cooling air flow or in the electrical circuit of the warning light itself. It does not provide any indication of the actual temperature of the electronic components. Illumination of the light in flight could indicate the following cooling system malfunctions:

- Ram air regulator valve closed due to overpressure in the boundary layer ram air intake duct. (The valve must be reset manually by ground maintenance personnel.)
- Boundary layer ram air temperature above 160-165°F and ram air regulator valve failed to close.
- Failure of the ram air regulator motor allowing the valve to close.

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.

During ground operation, illumination of the electronic cooling warning light indicates a malfunction with one exception. When electrical power is on the airplane and the engine is shut down, it is normal for the light to illuminate steady. If there is no malfunction, the light should go out when external cooling is applied or the engine is started. (Refer to Section III for ELECTRONIC COOLING WARNING LIGHT ILLUMINATED procedure, and to ELECTRONIC COOLING, Section V, for limitations.)

MISCELLANEOUS EQUIPMENT

ANTI-G SUIT PROVISIONS

Anti-g suit provisions are installed on some airplanes. 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 4-22) 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.

SPARE LAMPS

Spare lamps are stowed in holders at the aft end of the left console. The holder is placarded "Spare Lamps."

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. On TF airplanes the data case is located on the aft bulkhead, between the two seats.

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.



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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.

MINIMUM CREW REQUIREMENTS TF

The minimum crew is a pilot and one other person who does not need to be rated. Although one person can physically operate the airplane from the left seat, both seats should be occupied to provide optimum visual clearance of the areas both to the left and right of the airplane. Flight may be accomplished with a crew of one person only when accompanied by a "chase" or "pacer" airplane.

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).

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.
- 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 ACCELERATION AND EGT LIMITATIONS

The J-57 engine has a tendency to overshoot in thrust and EGT during accelerations from IDLE to FULL MIL POWER when the engine is cold. A cold engine is defined as one which has just been started or has been permitted to cool at idle rpm for at least five minutes. On the

90% TO 98% CONTINUOUS OPERATION.
102% MAXIMUM PERMISSIBLE RPM.

TACHOMETER



250 KCAS (240 KIAS) MAXIMUM GEAR DOWN
655 KCAS (KIAS) MAXIMUM AIRSPEED
REFER TO TEXT FOR MAXIMUM SPEED LIMITS.

AIRSPEED-MACH INDICATOR

NOTE
MAXIMUM SPEED POINTER DOES NOT INDICATE ANY LIMIT.



55 PSI MAXIMUM
40 TO 55 PSI NORMAL OPERATION
40 PSI MINIMUM FOR IDLE

OIL PRESSURE

NOTE
MINOR FLUCTUATIONS OF THE OIL PRESSURE GAGE NEEDLE WILL NORMALLY OCCUR AND ARE PERMISSIBLE.



21936-1

Figure 5-1

ground the engine should accelerate smoothly from idle to military rpm in a maximum of 15 seconds. Additionally, if an EGT overshoot occurs, the EGT should peak out and begin to decrease within two minutes. Following an engine-acceleration EGT overshoot, the EGT must decrease to 630°C or below within a total elapsed time of five minutes, beginning at first throttle movement, or the engine is unacceptable. At 35,000 feet, at a minimum of 250 KCAS (KIAS), straight and level flight, the engine should accelerate smoothly from idle to military rpm in a maximum of 26 seconds. These limitations, although most frequently approached with cold engines, shall apply to all accelerations regardless of rpm change. The EGT overshoot time limit of two minutes (figure 5-2) is the period between advancing the throttle and the moment that the EGT is first observed to start falling after reaching its peak. The performance of the engine is not acceptable if either the EGT limit or time limit is exceeded.

ENGINE PRESSURE RATIO

The military and maximum thrust checks are based on the takeoff thrust index marker being set according to the outside air temperature. When making the military thrust check during the before takeoff engine check, the gage pointer should be within the arc of the takeoff thrust index marker. Engine rpm should be stabilized. The maximum thrust check is made at the start of takeoff roll and is the same as the military thrust check. During takeoff roll, the gage reading will increase above that noted during the maximum thrust check.

instrument markings



EXHAUST TEMPERATURE

J57-23 ENGINE

- 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)
- 140°C MAXIMUM FOR EGT SPREAD 1

SEE 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

1 SOME AIRPLANES

21936-2

ENGINE OVERSPEED

The maximum permissible engine speed is 102% rpm. Any engine speed in excess of 102% should be noted on AFTO 781. If the engine speed exceeds 102% rpm, it must be inspected for damage. Speeds in excess of 104% rpm necessitate engine overhaul.

FUEL FLOW FLUCTUATIONS

Fuel flow fluctuation of 500 pph or less is acceptable provided that there is no associated increase or decrease in rpm or EGT or that no engine surge is evident.

COOLING LIMITATIONS

Inflight Operations

Whenever the aircraft is flown for the maximum allowable time using maximum or military thrust (see figure 5-2), an equal amount of time should be flown at or below the maximum continuous limit prior to returning to the maximum or military thrust condition. For example, after flying 15 minutes at maximum power, the engine should be operated for 15 minutes at or below 610°C above 30,000 feet or 580°C below 30,000 feet, prior to returning to maximum thrust.

Note

It may be possible to exceed time limits at military or maximum thrust without exceeding the maximum continuous EGT limit. Failure to reduce thrust within the time limits prescribed in figure 5-2 (even though maximum continuous EGT limits are not exceeded) will greatly reduce engine life.

engine operating limits

J57-23/23A	MAXIMUM OBSERVED EXHAUST GAS TEMPERATURE (DEGREES C)				TIME LIMITS	
OPERATING CONDITIONS	S.L. TO 20,000 FT.	20,000 TO 25,000 FT.	25,000 TO 30,000 FT.	30,000 FT. AND ABOVE	GROUND OPERATION	FLIGHT OPERATION
MAXIMUM (EXCEPT ENGINE ACCELERATION) 1	640	650	660	670	1 MINUTE	15 MINUTES
MILITARY 1	630	640	650	660	5 MINUTES 2	30 MINUTES
MAXIMUM CONTINUOUS	580	590	600	610	5 MINUTES 2	CONTINUOUS
IDLE (GROUND OPERATION)	340 3	—	—	—	—	—
ACCELERATION 4	680	680	680	680	2 MINUTES	2 MINUTES
STARTING	450	—	—	—	—	—

1. WHEN THE AIRCRAFT IS FLOWN FOR THE MAXIMUM ALLOWABLE TIME USING MAXIMUM OR MILITARY THRUST, AN EQUAL AMOUNT OF TIME SHOULD BE FLOWN AT OR BELOW THE MAXIMUM CONTINUOUS LIMIT PRIOR TO RETURNING TO MAXIMUM OR MILITARY THRUST CONDITION.
2. REFER TO "COOLING LIMITATIONS" TEXT, THIS SECTION.
3. NORMAL OPERATING TEMPERATURE; NOT A LIMITATION.
4. REFER TO "ACCELERATION LIMITATIONS" TEXT, THIS SECTION. TIME LIMIT IS MEASURED FROM THROTTLE ADVANCEMENT UNTIL TEMPERATURE BEGINS TO DECREASE. EGT MUST RETURN TO RESPECTIVE MILITARY LIMIT WITHIN FIVE MINUTES TOTAL ELAPSED TIME.

NOTE
IF TEMPERATURE LIMITS ARE EXCEEDED, MAKE ENTRY IN AFTO 781 STATING DURATION AND PEAK TEMPERATURES.

21,939

Figure 5-2

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 75% rpm for 30 seconds to scavenge the oil system. Make entry on AFTO Form 781 if temperature limits are exceeded and state the time and temperature peaks.

FUELS

Specified Fuel

The fuel recommended for normal operation is JP-4, MIL-T-5624 (NATO Symbol F-40). This fuel has been proved to operate satisfactorily under all conditions and is the only fuel containing an icing inhibitor to prevent fuel filter icing. Performance data is based on the use of JP-4 fuel.

Alternate Fuel

Alternate fuels are those fuels approved for use when the specified fuel is not available and which may cause a reduction in engine performance and additional aircraft restrictions. Although alternate fuels will not cause permanent engine damage, their use may require an engine retrim to achieve rated engine performance. If the engine is not retrimmed, both ground and air starts may be more difficult and higher EGT may be reached at lower thrust settings than when operating on JP-4.

CAUTION

Engine operating limitations when using alternate fuels are the same as those established for JP-4. Care should be taken to insure that EGT and rpm limits are not exceeded when moving the throttle.

Alternate fuels do not contain an icing inhibitor; therefore, aircraft altitude is limited to that at which the outside air temperature is above the freeze point of the fuel. The fuel weight differential plus the fuel quantity system and flow indication errors are not significant and may be disregarded. Approved alternate fuels are listed below in their preferred order of use:

Type Fuel	NATO Symbol	Freeze Point
Commercial JET B	None	−49°C (−56°F)
JP-5 (MIL-T-5624)	F-44	−46°C (−51°F)
Commercial JET A-1	F-35	−48°C (−54°F)
Commercial JET A	None	−38°C (−36°F)

Emergency Fuel

Aviation gasoline (MIL-G-5572) is the only approved emergency fuel and is limited to one-time ferry missions only. Use of emergency fuel causes undesirable lead deposits in the engine and may damage the fuel control and fuel pump because of its poor lubricating properties. The same general operating limitations and cautions apply during use of emergency fuels as during use of alternate fuels. Approved emergency fuels are listed below in their preferred order of use:

Type Fuel	NATO Symbol	Freeze Point
Grade 80/87	F-12	−60°C (−76°F)
Grade 100/130	F-18	−60°C (−76°F)
Grade 115/145	F-22	−60°C (−76°F)

Note

When aviation gasoline is used, three percent lubricating oil, specification MIL-T-5624 must be added to improve its lubricity characteristic.

Center of Gravity Control When Using Alternate Fuels

Since all alternate fuels except JET B are heavier than JP-4, special action is required to maintain the airplane

center of gravity within limits. In the armament-out configuration, 1600 pounds of JP-5 and 1475 pounds of JET A or JET A-1 internal fuel must be burned off prior to takeoff.

WARNING

Failure to observe the fuel burnoff instructions will cause the c.g. to exceed the aft limit. This can cause over-rotation on take-off and possible loss of control during flight.

In the armament-in configuration, no special instructions are required for c.g. control and all alternate fuels may be used normally. Partial armament loads must be considered to be the armament-out configuration.

ENGINE OIL

The maximum allowable engine oil consumption is two quarts per hour.

AIRSPEED LIMITATIONS

MAXIMUM ALLOWABLE AIRSPEEDS

The maximum allowable airspeed is 655 KCAS (KIAS) or Mach 1.5, whichever occurs first.

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 250 KCAS (240 KIAS) with the landing gear extended.

DRAG CHUTE DEPLOYMENT SPEED

The maximum allowable airspeed at which the drag chute may be deployed is 165 KCAS (160 KIAS). If the drag chute is deployed at speeds above this limit it will be automatically released from the airplane.

RAT LIMIT SPEED

The maximum allowable airspeed at which the RAT can be extended or operated is 345 KCAS (KIAS). Extension of the RAT at speeds above 345 KCAS (KIAS) is likely to cause overspeed or structural damage. Satisfactory emergency hydraulic pressure will not be maintained at airspeeds below 125 KCAS (KIAS).

OTHER OPERATIONAL LIMITATIONS

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS)

Operation of the attack mode of AFCS is prohibited when external tanks containing fuel are installed.

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.

CAUTION

When engine rpm is below 72%, operation of the radar with the FCS master switch in STBY or ON is limited to 20 minutes. Operation of the radar with the FCS master switch in WARM is limited to 20 minutes when engine rpm is above 72%, and unlimited when engine rpm is below 72%.

On the ground, operation of the radar with FCS master switch in STBY or ON for more than 30 seconds is prohibited, with engine rpm above 72%, except during takeoff or if external air is connected. With engine rpm below 72%, operation of the radar in STBY or ON, is limited to 20 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 FCS master switch in WARM, place master switch to ON and wait 30 seconds prior to commencing takeoff roll.
2. If airplane is taxied with FCS master switch in STBY or ON (engine rpm below 72%) takeoff should be made within 20 minutes or the FCS 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 FCS master switch in ON or STANDBY positions. If flight at these throttle settings exceeds seven minutes, the FCS master switch must be in OFF or WARM positions to prevent excessive heat from damaging electronics equipment. In flight operation of the radar with the master switch in WARM is unlimited.

Note

- The FCS 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 FCS 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.

Maneuvering Limitations

With external tanks installed and containing fuel, maximum aileron deflection is limited to approximately ⅓ aileron (3°) above 270 KCAS (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 435 KCAS (KIAS) or Mach .95, whichever occurs first. External tanks may be jettisoned at any speed within the operating speed range.

CAUTION

Do not use full or abrupt rudder movement during flight with external tanks containing fuel, in order that the design limit of the airplane is not exceeded.

CHAFF DISPENSER WEIGHT LIMITATION

Maximum weight for each chaff dispenser including chaff is 375 pounds.

ARMAMENT EQUIPMENT

Armament may be fired (within the existing external tank and armament equipment limitations) with external tanks installed, providing the aft cg limits are not exceeded.

Note

- The following internal fuel remaining quantities provide earliest armament firing limits for the most adverse (aft) CG conditions, regardless of fuel quantity in external tanks.

Armament	Fuel Remaining
24 Rockets	6500 pounds
12 Rockets	6500 pounds
3 Missiles (aft bay)	6500 pounds
3 Missiles (fwd bay)	6300 pounds
6 Missiles	6200 pounds
Salvo	5500 pounds

- It may be possible to fire armament at a higher internal fuel load on some aircraft, depending on basic aircraft weight and balance and the type of armament carried.
- Armament door operation and launcher extension is permissible when external tanks are installed regardless of fuel quantity contained in the external tanks.

Armament Door 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.

CAUTION

It is 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 or aft 2.75-inch rockets are loaded.

TIRE LIMITATIONS

The maximum allowable main gear ground speed with 22-ply fabric tread tires is 217 knots (217 knots printed on the side wall).

ELEVATOR FEEL-FORCE REGULATOR

For unrestricted flight, the pin on the elevator feel force intelligence unit should be flush (±0.06 inch; 1/16 inch). If the pin is retracted more than 0.06 inch, the airplane should not be flown in excess of .90 Mach. If the pin is extended more than 0.06 inch, the regulator must be replaced prior to flight.

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.

CANOPY (F)

When taxiing with canopy open, do not exceed 60 knots ground speed or 90 KCAS (KIAS), whichever is less. Taxi speeds in excess of the above may result in damage to the canopy or aiplane structure.

CANOPY (TF)

When taxiing with canopy open, do not exceed 50 knots ground speed or 80 KCAS (KIAS), whichever is less. Taxi speeds in excess of the above may result in damage to the canopy or airplane structure.

MAXIMUM BARRIER ENGAGEMENT SPEEDS

For F/TF-102A gross weight up to 28,000 pounds, M-21 dial setting 6, limit engaging velocity to 85 knots dry or wet tapes. Aborted takeoff gross weight of 31,500 pounds, M-21 dial setting 6, limit engaging velocity to 80 knots dry or wet tapes. The velocity is limited by the aircraft hook strength.

BAK-9 (USN E-27)

Gross Weight Pounds	Ground Speed-Knots	G-Load
20,000	174	2.70
22,000	172	2.45
24,000	170	2.25
26,000	168	2.10
28,000	166	1.90
30,000	164	1.80
32,000	162	1.70

BAK-12

Gross Weight Pounds	Ground Speed-Knots	G-Load
20,000	166	2.70
22,000	164	2.45
24,000	162	2.25
26,000	160	2.10
28,000	158	1.90
30,000	156	1.80
32,000	154	1.70

BAK-13 (USN E-28)

Gross Weight Pounds	Ground Speed-Knots
20,000	140
22,000	139
24,000	138
26,000	136
28,000	134
30,000	132
32,000	131

PROHIBITED MANEUVERS

SPINS

Intentional spins are prohibited due to excessive loss of altitude and the probability of exceeding engine temperature limits. (Refer to SPINS, Sections III and VI.)

NEGATIVE G MANEUVERS

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 KCAS (KIAS).

1. Yaw damper and turn coordinator on:
   a. Above 450 KCAS (KIAS) below 20,000 feet—¾ aileron.
   b. All other conditions—No restriction.
2. Yaw damper and turn coordinator off:
   a. Above 450 KCAS (KIAS) below 20,000 feet—½ aileron.
   b. All other conditions—¾ aileron except for takeoff and landing.

Yaw Damper and Turn Coordinator Off

With the yaw damper and turn coordinator off, the following restrictions shall be observed to prevent exceeding the 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.

acceleration limitations

CONFIGURATION	LIMITS F		LIMITS TF	
	SYMMETRICAL	ROLLING PULLOUTS	SYMMETRICAL	ROLLING PULLOUTS
CLEAN— LESS THAN 4200 POUNDS FUEL	−3.0 TO +6.85 G	0 TO +5.0 G	−3.0 TO +6.0 G	0 TO +3.9 G
CLEAN— MORE THAN 4200 POUNDS FUEL	−2.7 TO +5.3 G	0 TO +3.9 G	−2.5 TO +5.1 G	0 TO +3.9 G
WITH EXTERNAL TANKS CONTAINING FUEL	−1.0 TO +3.0 G	0 TO +2.3 G	−1.0 TO +3.0 G	0 TO +2.3 G
WITH EXTERNAL TANKS EMPTY	−2.7 TO +5.3 G	0 TO +3.9 G	−2.5 TO +5.1 G	0 TO +3.9 G
MISSILE BAY DOORS OPEN	−1.0 TO +3.0 G		−1.0 TO +3.0 G	

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Figure 5-3

FLIGHT WITH PITCH DAMPER OFF

High speed, low-altitude flight without the pitch damper can cause control problems and should be avoided except where required to accomplish the assigned mission.

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 forces.

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.

ACCELERATION LIMITATIONS

Asymmetrical (rolling pullout) maneuvers impose higher loads on the airplane and require more restrictive g limitations than those required for symmetrical loads. Figure 5-3 shows symmetrical and asymmetrical maneuvering limitations, figure 5-4 represents operating flight limits for symmetrical loads.

Note

Maximum acceleration limits for all airplanes except Category I are four g to zero g. Category I airplanes are airplanes containing no wing spar cracks or ones that have cracks that have been repaired. These airplanes are identified as Category I by cockpit placards. Category I airplane limits are as indicated in figure 5-3.

CENTER OF GRAVITY LIMITATIONS

The only inflight 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

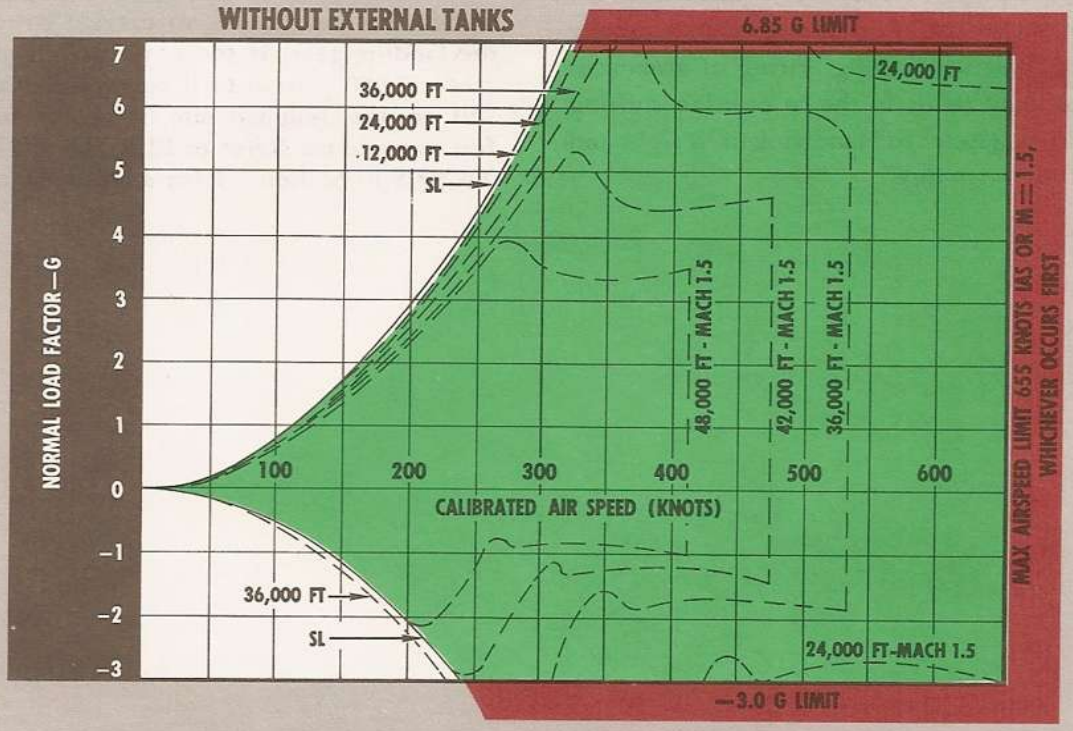
The gross weight limit of 32,859 pounds is based on the structural limitations of the landing gear and applies to both the F-102A and the TF-102A. To preclude exceeding this limit, the basic weight of the F-102A must not exceed 21,158 pounds and the TF-102A must not exceed 20,928 pounds. At landing weights (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.

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

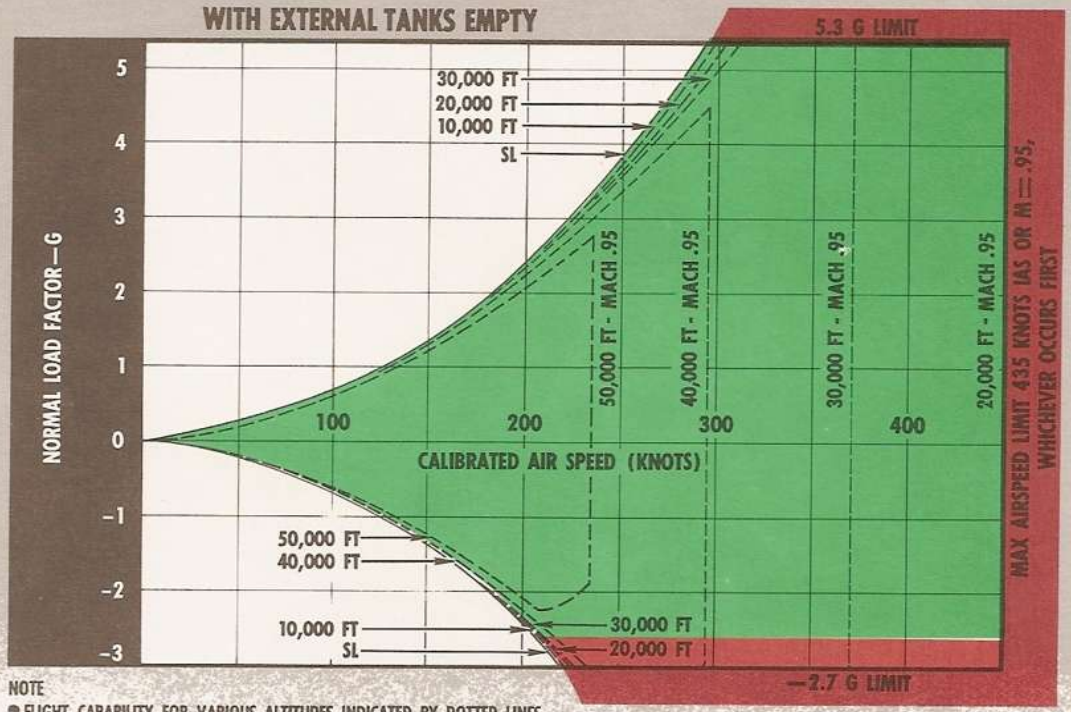
ENGINE: J57-23
FUEL GRADE: JP-4

F

WITHOUT EXTERNAL TANKS



WITH EXTERNAL TANKS EMPTY



NOTE

- FLIGHT CAPABILITY FOR VARIOUS ALTITUDES INDICATED BY DOTTED LINES.
- REFER TO TEXT FOR MAXIMUM ALLOWABLE AIRSPEEDS AND LOAD FACTORS.

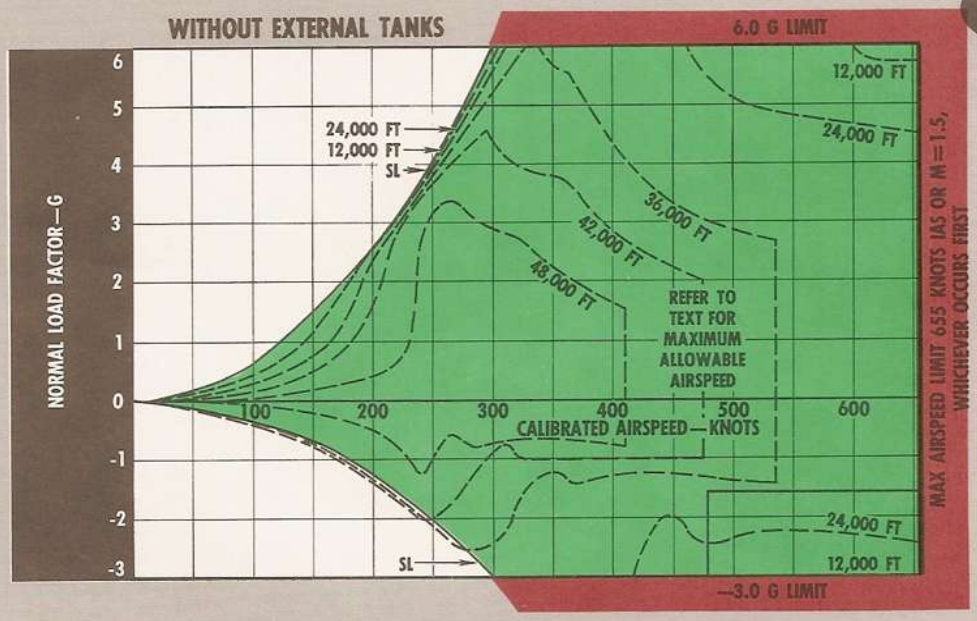
21937-1

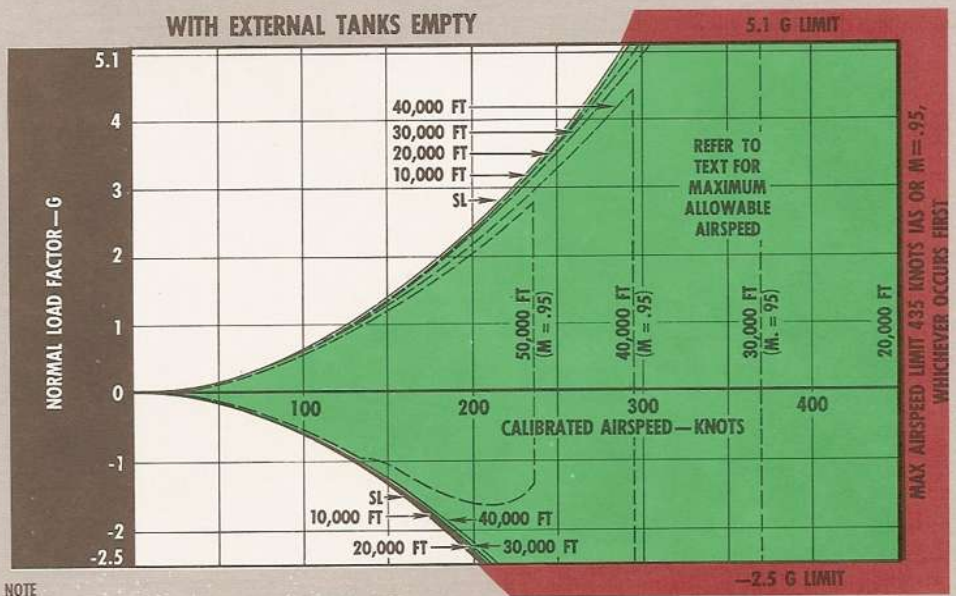
operating flight limits

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
FUEL GRADE: JP-4

TF





NOTE

- FLIGHT CAPABILITY FOR VARIOUS ALTITUDES INDICATED BY DOTTED LINES.
- REFER TO TEXT FOR MAXIMUM ALLOWABLE AIRSPEEDS AND LOAD FACTORS.

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Level Flight Characteristics	6-7
Maneuvering Flight	6-7
Dives	6-10
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Vortex Generators	6-13

Note

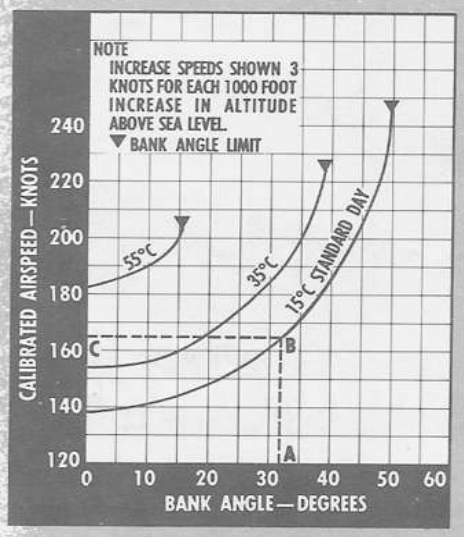
Airspeeds and Mach numbers shown in this Section are calibrated airspeed and true Mach number.

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. 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 225 KCAS, 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 225 KCAS. On TF airplanes, a buffet is also encountered between approximately Mach .90 to .95 in 1-g flight (straight and level) due to a disturbed airflow over the canopy. The buffet is also encountered during accelerated flight (g forces applied) and the range in which it is encountered is increased as g forces are increased. The canopy buffet may be quite startling when first experienced but should create no unusual characteristics, and as speed is increased beyond the buffet range it will completely disappear. On some airplanes, a loud bang may be heard when passing through Mach .85. This noise results from wrinkling of a cockpit panel below the left seat and does not affect the structural integrity of the airplane.

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

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 =32°.
B OUTSIDE AIR TEMPERATURE =15°C.
C MINIMUM AIRSPEED =165 KNOTS CAS.

ALTITUDE REQUIRED TO RECOVER FROM DESCENT IN ADJUSTING THRUST FROM THAT SPECIFIED TO MILITARY THRUST

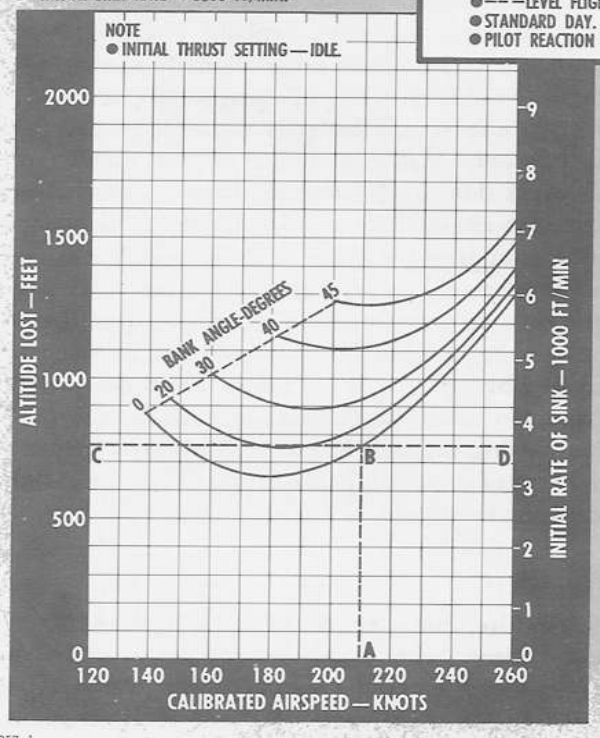
EXAMPLE

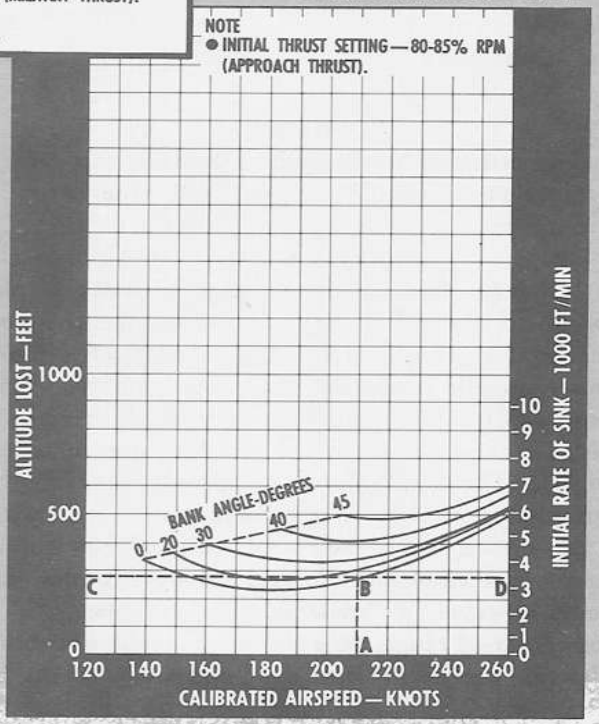
A INITIAL AIRSPEED =210 KNOTS CAS.
B INITIAL BANK ANGLE =0°.
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT =760 FEET.
D INITIAL SINK RATE =3600 FT/MIN.

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 =210 KNOTS CAS.
B INITIAL BANK ANGLE =0°.
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT =280 FEET.
D INITIAL SINK RATE =3,400 FT/MIN.





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Figure 6-1 (Sheet 1)

F minimum speeds

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

ENGINE: J57-23
FUEL GRADE: JP-4

CONDITIONS

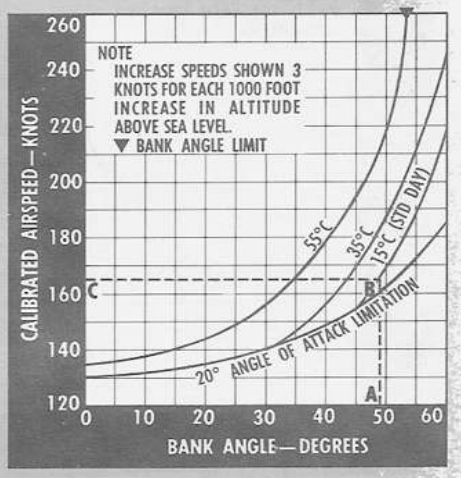
- GROSS WEIGHT EQUALS 27,500 LBS
- C G 27.5%
- AMBIENT TEMPERATURE AS INDICATED
- ALTITUDE—SEA LEVEL

CONFIGURATION

- LANDING GEAR DOWN
- SPEED BRAKES OUT

LEVEL FLIGHT MINIMUM SPEEDS WITH MAXIMUM THRUST

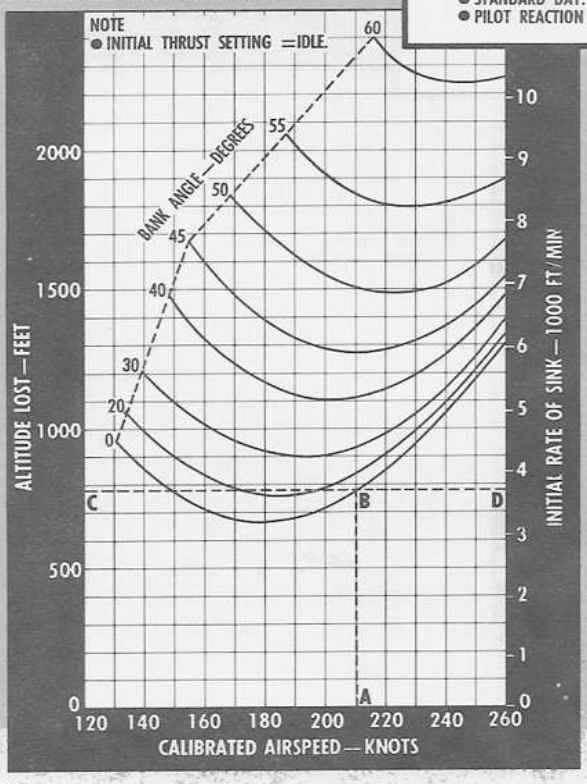
EXAMPLE
A INITIAL BANK ANGLE =49°.
B OUTSIDE AIR TEMPERATURE =15°C.
C MINIMUM AIRSPEED =165 KNOTS CAS.



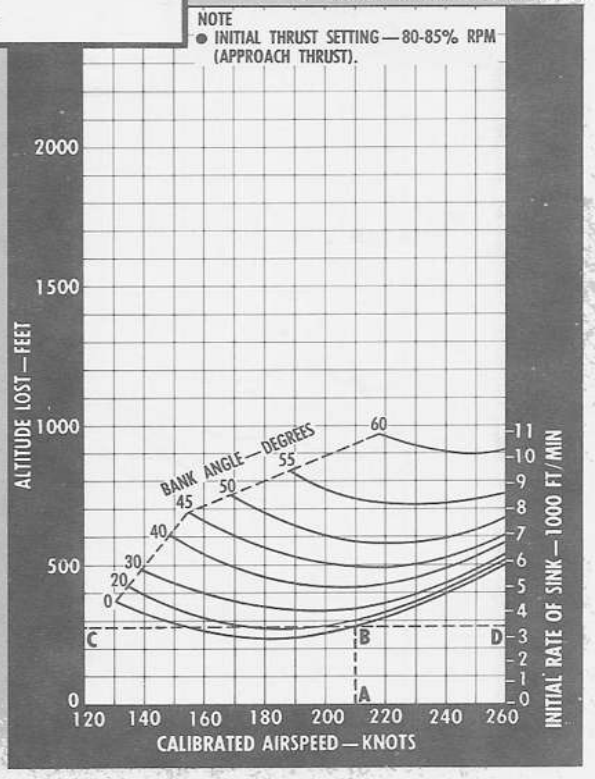
ALTITUDE REQUIRED TO RECOVER FROM DESCENT IN ADJUSTING THRUST FROM THAT SPECIFIED TO MAXIMUM THRUST

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 =210 KNOTS CAS.
B INITIAL BANK ANGLE =0°.
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT =780 FEET.
D INITIAL SINK RATE =3700 FEET/MIN.

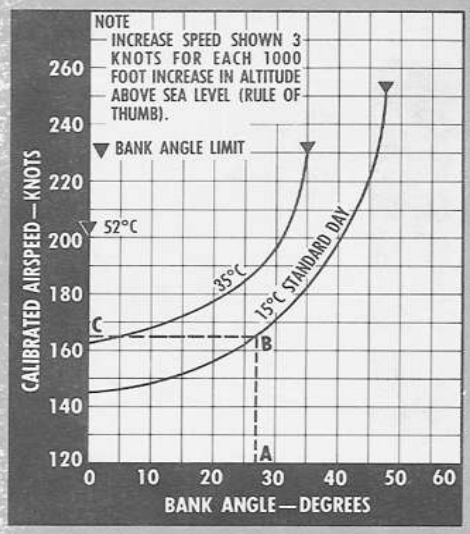


EXAMPLE
A INITIAL AIRSPEED =210 KNOTS CAS.
B INITIAL BANK ANGLE =0°.
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT =280 FEET.
D INITIAL SINK RATE =3,400 FEET/MIN.



21,957-2

Figure 6-1 (Sheet 2)



LEVEL FLIGHT MINIMUM SPEEDS WITH MILITARY THRUST

EXAMPLE

A INITIAL BANK ANGLE = 27°
B OUTSIDE AIR TEMPERATURE = 15°C
C MINIMUM AIRSPEED = 165 KNOTS CAS

BANK ANGLE	LOAD FACTOR
0°	1.0G
30°	1.2G
45°	1.4G
60°	2.0G

ALTITUDE REQUIRED TO RECOVER FROM DESCENT IN ADJUSTING THRUST FROM THAT SPECIFIED TO MILITARY THRUST

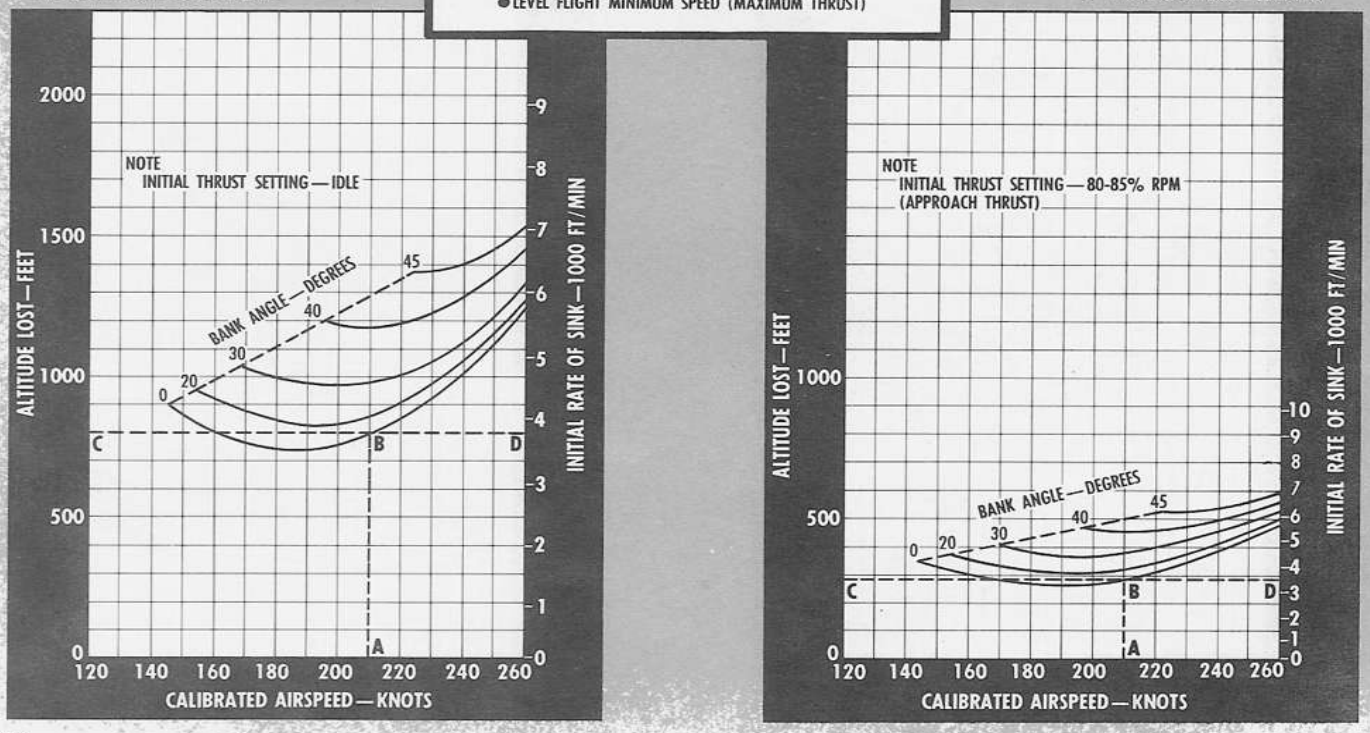
EXAMPLE

A INITIAL AIRSPEED = 210 KNOTS CAS
B INITIAL BANK ANGLE = 0°
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT = 800 FEET
D INITIAL SINK RATE = 3800 FT/MIN

NOTE
- ALTITUDE LOST IS HIGHLY DEPENDENT ON PILOT TECHNIQUE AND CURVES ARE THEREFORE APPROXIMATIONS
- STANDARD DAY
- PILOT REACTION TIME — 2 SECONDS
- LEVEL FLIGHT MINIMUM SPEED (MAXIMUM THRUST)

EXAMPLE

A INITIAL AIRSPEED = 210 KNOTS CAS
B INITIAL BANK ANGLE = 0°
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT = 290 FEET
D INITIAL SINK RATE = 3500 FEET



21,957-3

Figure 6-1 (Sheet 3)

TF minimum speeds

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
FUEL GRADE: JP-4

CONDITIONS

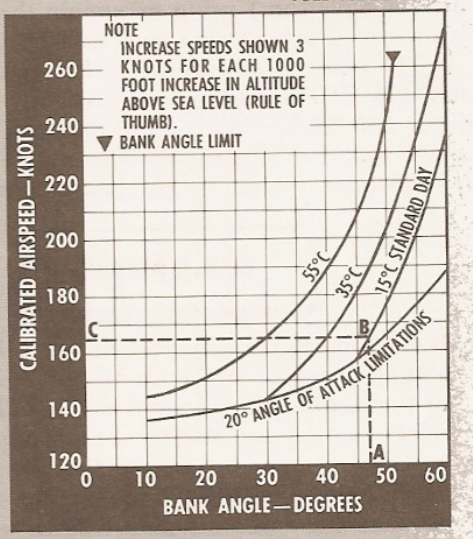
- GROSS WEIGHT EQUALS 27,500 LBS
- C.G. @ 27.5%
- AMBIENT TEMPERATURE AS INDICATED
- ALTITUDE—SEA LEVEL

CONFIGURATION

- LANDING GEAR DOWN
- SPEED BRAKES OUT

LEVEL FLIGHT MINIMUM SPEEDS WITH AFTERBURNER THRUST

EXAMPLE
A INITIAL BANK ANGLE = 47°
B OUTSIDE AIR TEMPERATURE = 15°C
C MINIMUM AIRSPEED = 165 KNOTS CAS

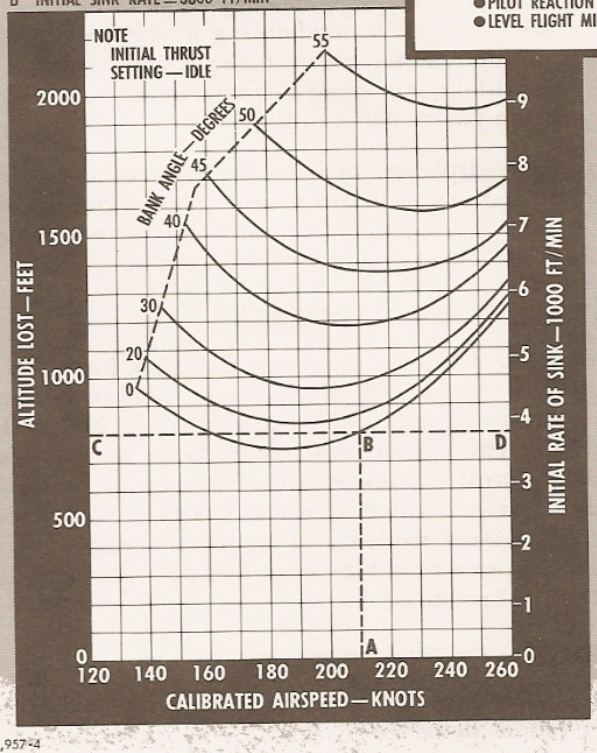


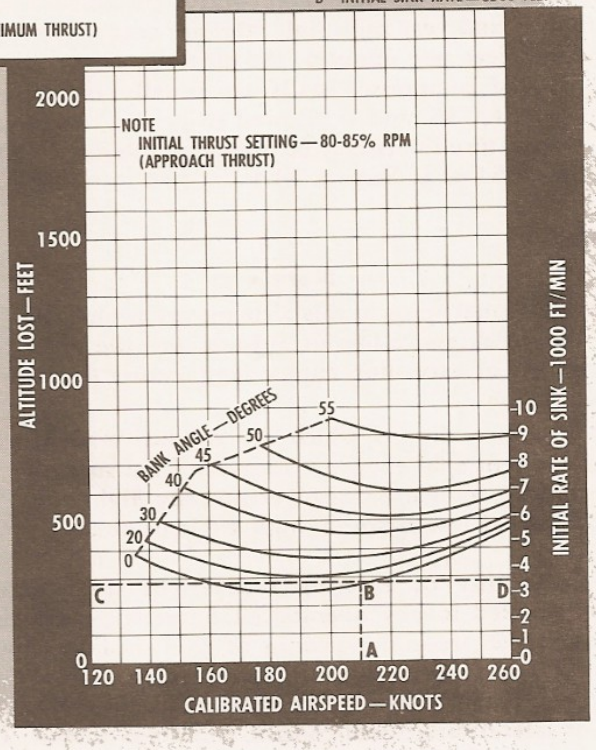
ALTITUDE REQUIRED TO RECOVER FROM DESCENT IN ADJUSTING THRUST FROM THAT SPECIFIED TO AFTERBURNER THRUST

EXAMPLE
A INITIAL AIRSPEED = 210 KNOTS CAS
B INITIAL BANK ANGLE = 0°
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT = 800 FEET
D INITIAL SINK RATE = 3800 FT/MIN

NOTE
- ALTITUDE LOST IS HIGHLY DEPENDENT ON PILOT TECHNIQUE AND CURVES ARE THEREFORE APPROXIMATIONS
- STANDARD DAY
- PILOT REACTION TIME—2 SECONDS
- LEVEL FLIGHT MINIMUM SPEED (MAXIMUM THRUST)

EXAMPLE
A INITIAL AIRSPEED = 210 KNOTS CAS
B INITIAL BANK ANGLE = 0°
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT = 290 FEET
D INITIAL SINK RATE = 3500 FEET





21,957-4

Figure 6-1 (Sheet 4)

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 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 or military thrust for recovery. During level flight, at a gross weight of 25,000 pounds, a warning buffet starts at approximately 175 KCAS, and increases slightly to a moderate buffet at 130 KCAS. With military thrust at 30,000 feet, a rate-of-sink develops at approximately 165 KCAS, and very high rates-of-sink are encountered at lower speeds.

WARNING

If the airspeed is reduced below 115 knots, the ailerons should not be used for lateral control of the airplane as the adverse yaw induced may cause a spin entry. Rudder control alone should be used to maintain lateral directional control to stall speed of 100 KCAS. 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 130 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 120 KCAS, 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.
- Flight conditions which are conducive to inadvertent spin entry are recovery from snap-up attacks, acrobatic maneuvers, or any flight condition of low airspeed, high angle of attack.

The possibility of entering inadvertent spins cannot be overlooked in any operational fighter-type airplane, although the pre-spin warnings of this airplane 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 if 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. Aileron application during either of these critical flight conditions greatly increases the probability of entering a spin. 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. 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. 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. (Refer to SPIN RECOVERY, Section III.)

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 pitch-up tendency occurs which becomes more pronounced at higher airspeeds. When retracting the speed brakes, a pitch-down tendency 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

Caution should be used during accelerated flight conditions as some elevator stick force lightening will occur as speed decreases in the transonic region (Mach 0.95 to 1.05).

On TF airplanes, due to the effect of the existing counterbalance installed to balance the dual control sticks, an estimated five-pound forward stick force is required to hold the stick in the takeoff trim position.

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 0.95 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

Control effectiveness at low speeds is exceptionally good. Flight at airspeeds in the approach and landing speed ranges is characterized by a relatively high angle of attack. The pilot should be alert during low-speed flight, since the high angle and effective control response make it possible to develop a high rate-of-sink without being aware of it. Once this condition develops at low altitude, it may be impossible to recover with full thrust applied before contact with the ground. Although the design of the airplane provides 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. The recommended airspeeds for takeoff, approach and landing, and their relationship to gross weight, ambient air temperature and altitude contained in this manual 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 nose-high-attitude 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 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 nose-high-attitude maneuvering flight. A speed higher than the minimum is advisable at least until more complete familiarity with the airplane characteristics is attained. See figure 6-1 for minimum airspeeds and figure 6-2 for characteristics at low speeds. Refer to the Appendix for takeoff and landing speeds.

MEDIUM SPEED

Medium-speed flight may be defined as from the low speed range to the transonic range. No unusual flight characteristics are encountered in the medium-speed range.

HIGH SPEED

Whenever flight through the transonic and 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 malfunctioning. 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 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.

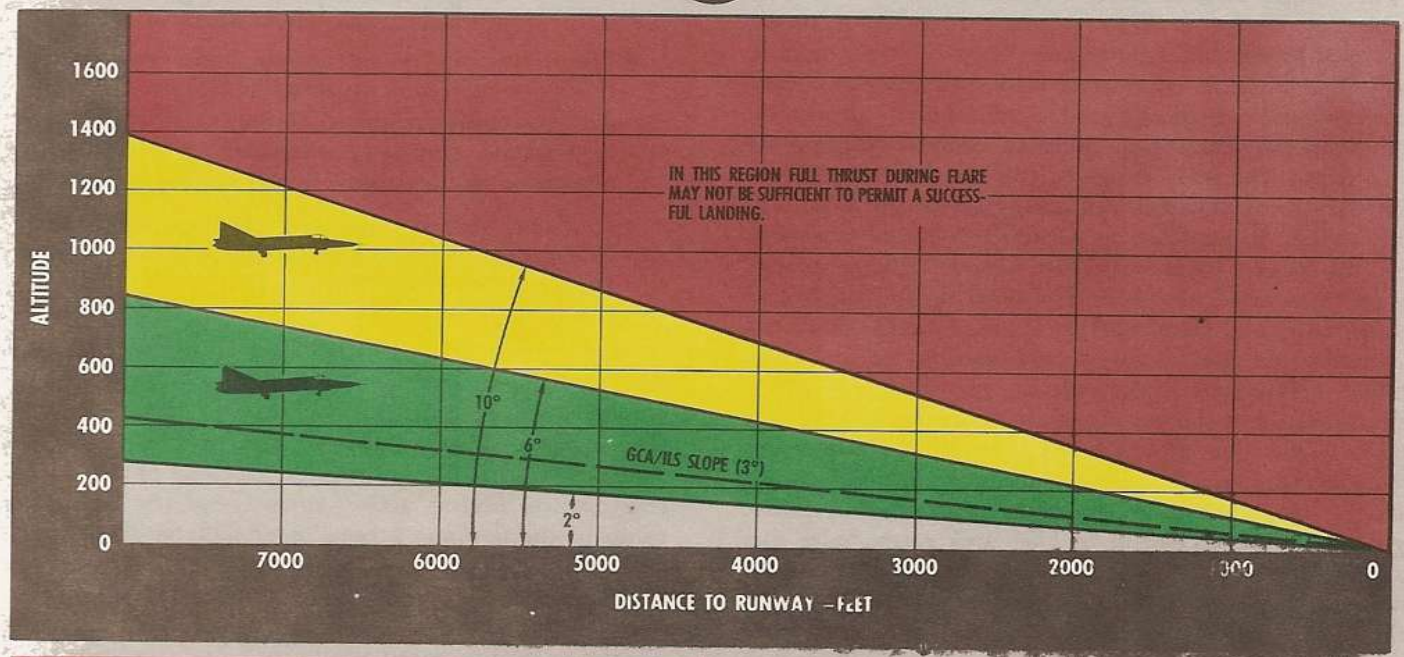
INERTIA COUPLING

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

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

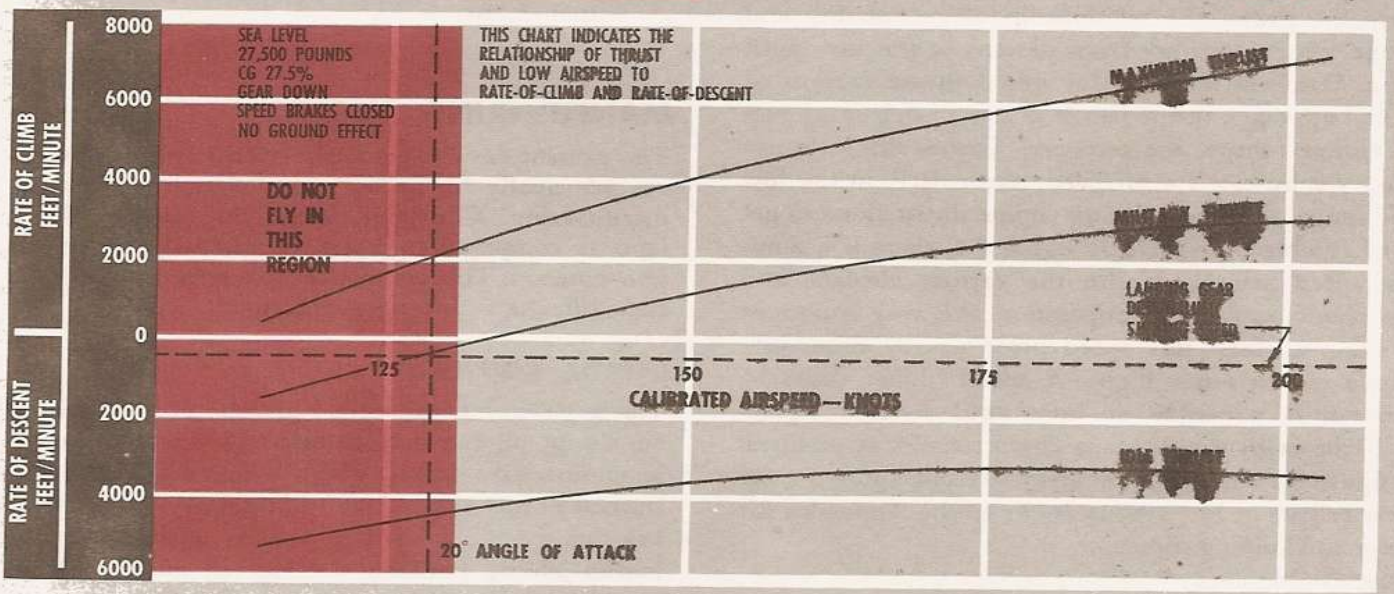
F

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



21,960-1

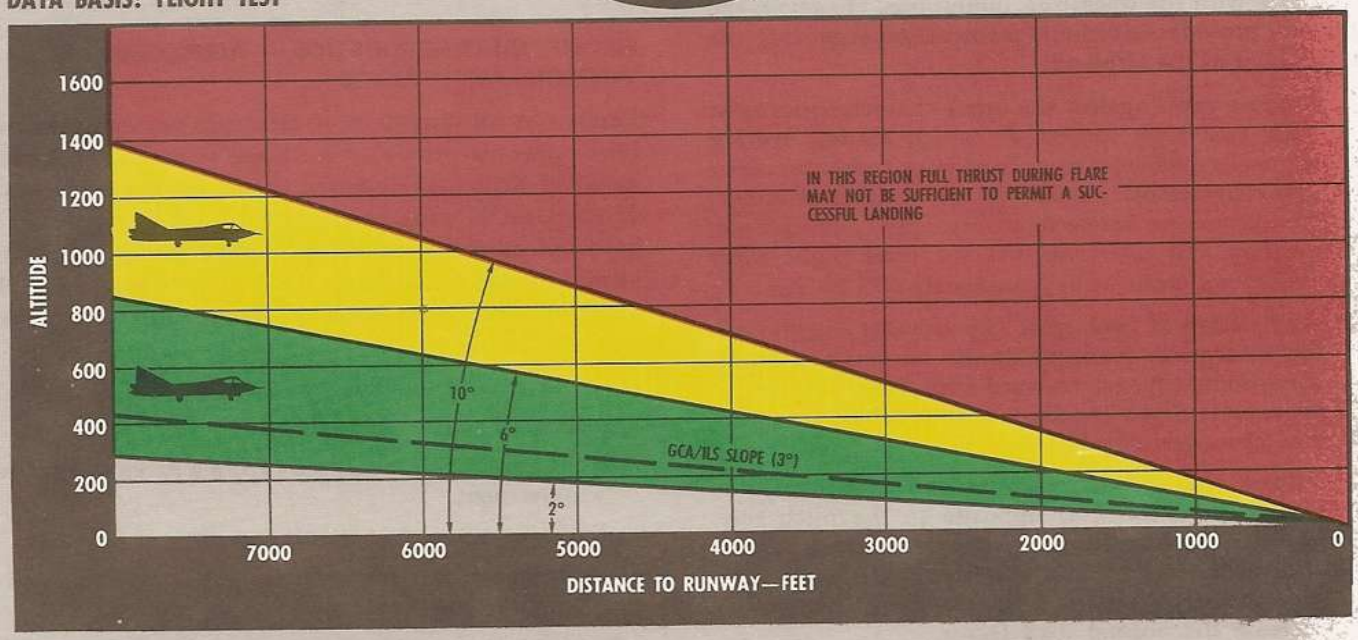
Figure 6-2

characteristics at low speeds

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

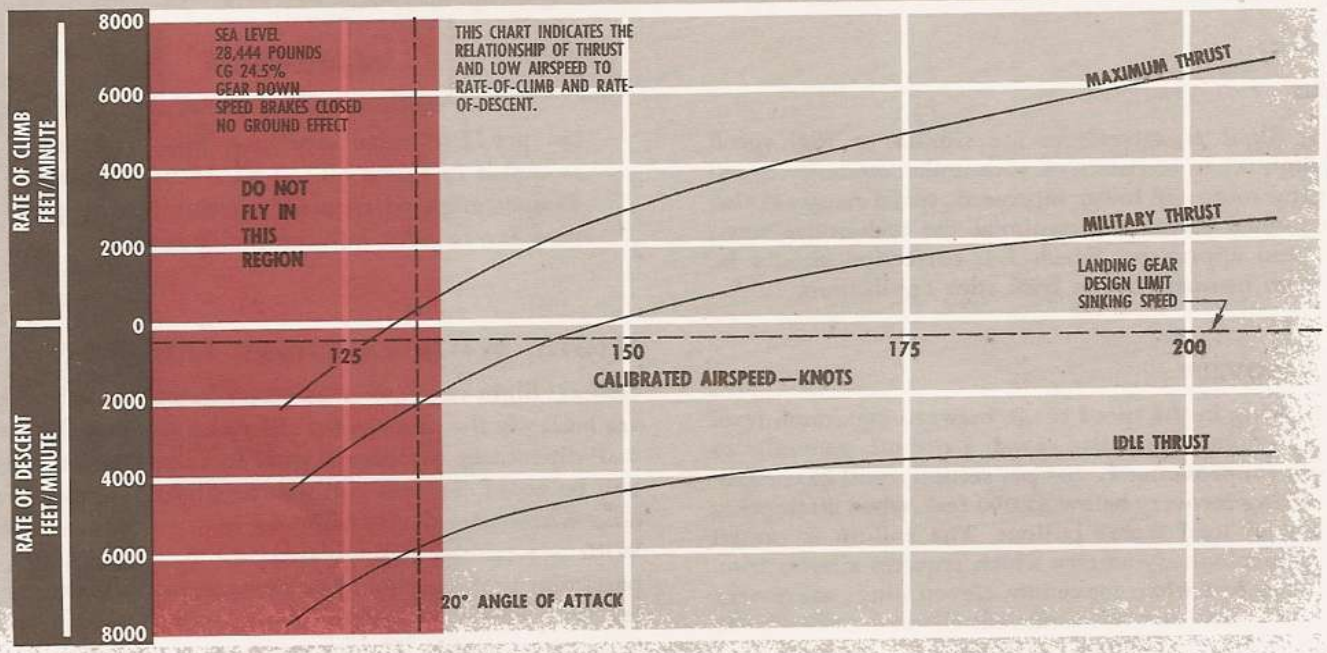
TF

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



21,960-2

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 the abrupt aileron maneuvers are performed in accordance with 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 275 KCAS.

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 complementary 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. 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 dive, minimum altitude loss is experienced if afterburner is terminated and speed brakes are opened.

FLIGHT CHARACTERISTICS — AIRPLANES WITHOUT HEP VALVES

Flight control operation of airplanes not equipped with HEP valves is primarily the same as on the HEP-valve-installed airplanes. However, pre-HEP valve airplanes are of the Case X wing configuration and therefore do not have the inherently stable feeling of the Case XX wing airplanes. Some conditions which may occur on pre-HEP valve airplanes as a result of service time accumulation are as follows:

a. Nose heavy tendency on takeoff.

b. Higher control stick forces required.

c. Sluggish or slow response from a given control stick movement.

d. Oversensitive feel to roll and pitch movement.

e. Creeping pitch trim, particularly nose down beyond neutral.

f. Difficulty in maintaining formation flying position.

g. Difficulty in maintaining a level trimmed attitude.

h. Damper/AFCS response time slightly slower.

WARNING

On pre-HEP valve airplanes, flight operation at speeds in excess of 330 KCAS without pitch damper engaged requires extreme care to prevent severe pilot induced oscillations.

FLIGHT WITH EXTERNAL TANKS

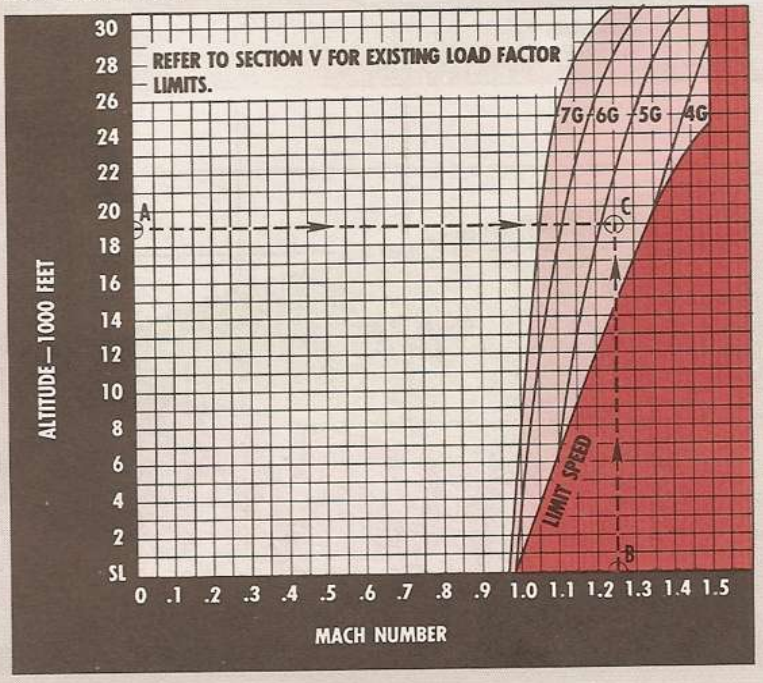
General flight characteristics with external tanks installed are basically the same as for the clean airplane. The normal differences associated with increased gross weight will be noted. Takeoff roll will be slightly increased and nose wheel lift-off speed should be a few knots higher. With one tank 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 permissible. 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

hinge moment limitation

MODEL: F/ TF-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



F TF

IDLE
THRUST

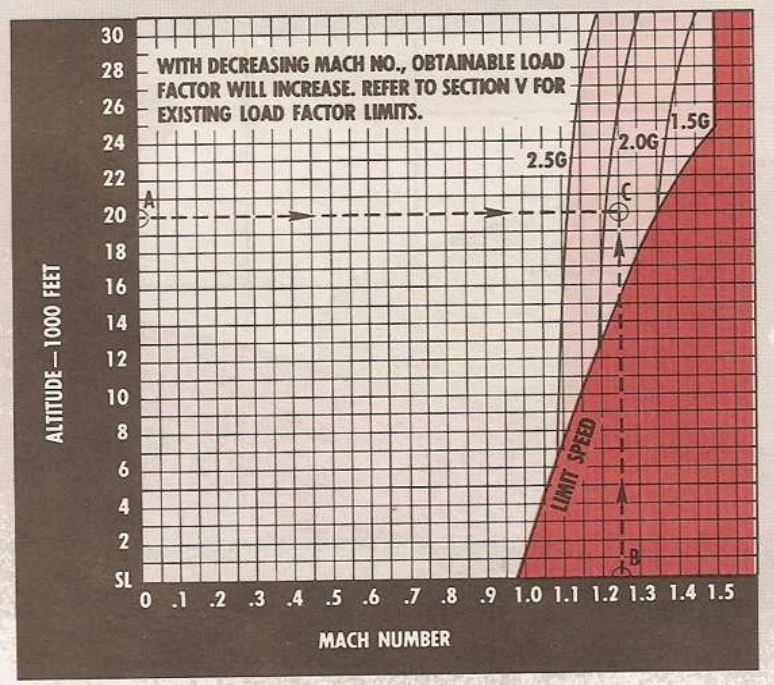
OBTAINABLE LOAD FACTORS
(WITH BOTH PRIMARY AND SECONDARY HYDRAULIC SYSTEM OPERATING)

EXAMPLE:

A = 19,000 FEET ALTITUDE WHEN LOAD FACTOR IS INDUCED.
B = MACH 1.25 WHEN LOAD FACTOR IS INDUCED.
C = 4.7 g AVAILABLE LOAD FACTOR BEFORE ENCOUNTERING ROLL-OFF.

NOTE

PLAN DIVE RECOVERY IN SUCH A MANNER AS NOT TO EXCEED HINGE MOMENT LIMITATIONS.



OBTAINABLE LOAD FACTORS
(WITH ONLY PRIMARY OR SECONDARY HYDRAULIC SYSTEMS OPERATING)

EXAMPLE:

A = 20,000 FEET ALTITUDE WHEN LOAD FACTOR IS INDUCED.
B = MACH 1.25 WHEN LOAD FACTOR IS INDUCED.
C = 1.75 g AVAILABLE LOAD FACTOR BEFORE ENCOUNTERING ROLL-OFF.

21958A

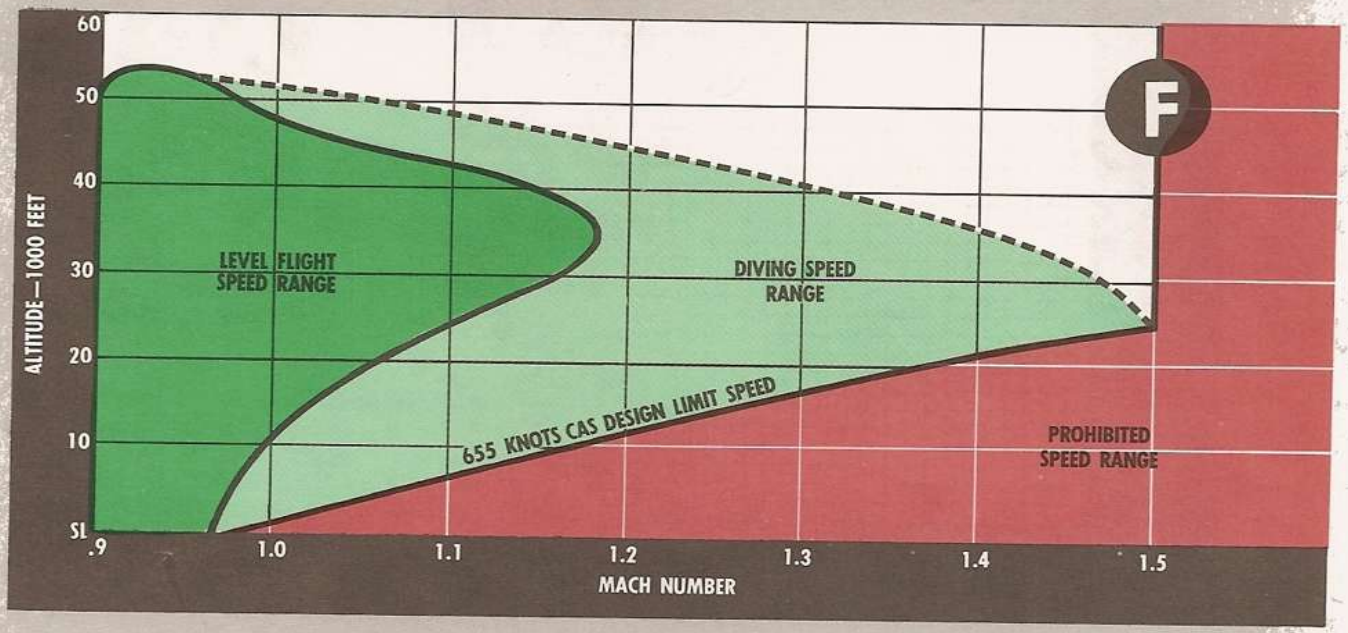
Figure 6-3

maximum speed capabilities

MODEL: F-102A
DATE: 23 AUGUST 1956
DATA BASIS: FLIGHT TEST

MAXIMUM THRUST
NO EXTERNAL LOAD

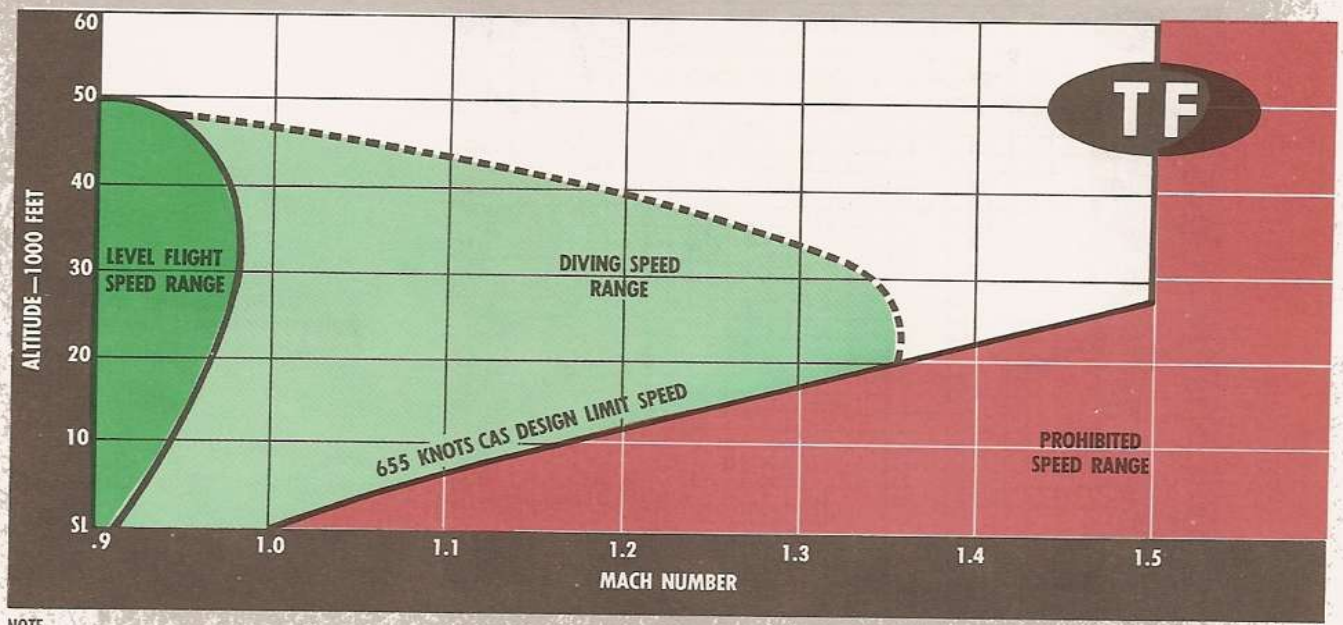
ENGINE: J57-23
FUEL GRADE: JP-4



MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

MAXIMUM THRUST
NO EXTERNAL LOAD

ENGINE: J57-23
FUEL GRADE: JP-4



NOTE
REFER TO SECTION V FOR EXISTING AIRSPEED LIMITATIONS

——— MAXIMUM LEVEL FLIGHT SPEED
– – – – MAXIMUM DIVE SPEED

22,105

Figure 6-4

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.

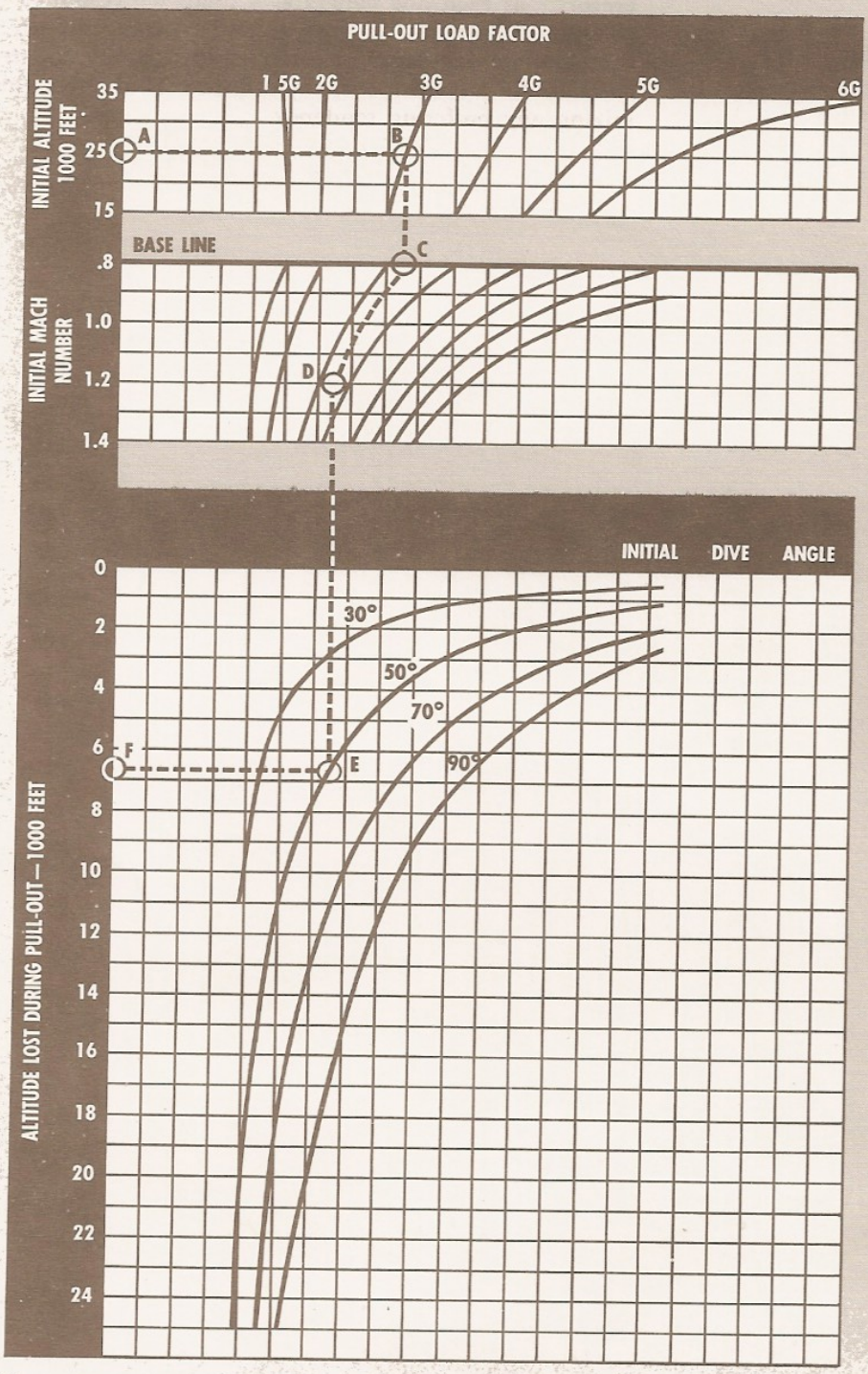
VORTEX GENERATORS (TF)

A series of small vanes are installed on the leading and trailing edges of the canopy to minimize buffet caused by a slow-moving layer of air over the canopy. Installation of these vanes eliminates the airflow which produces a buffet between approximately Mach 0.90 and 1.0, and at the same time reduces drag by providing a smoother airflow. These vanes are called "vortex generators," as each one generates a small whirlpool of air, commonly known as a vortex. Each vortex will stir up the slow-moving layer of air over the canopy and tend to maintain a continuous, smooth flow of air, thus minimizing any buffeting tendency.

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: SPEED BRAKES OPEN
(GROSS WT.: 24,500 LB.)
STANDARD ATMOSPHERE

ENGINE: J57-23
FUEL GRADE: JP-4



21959-1

F

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).

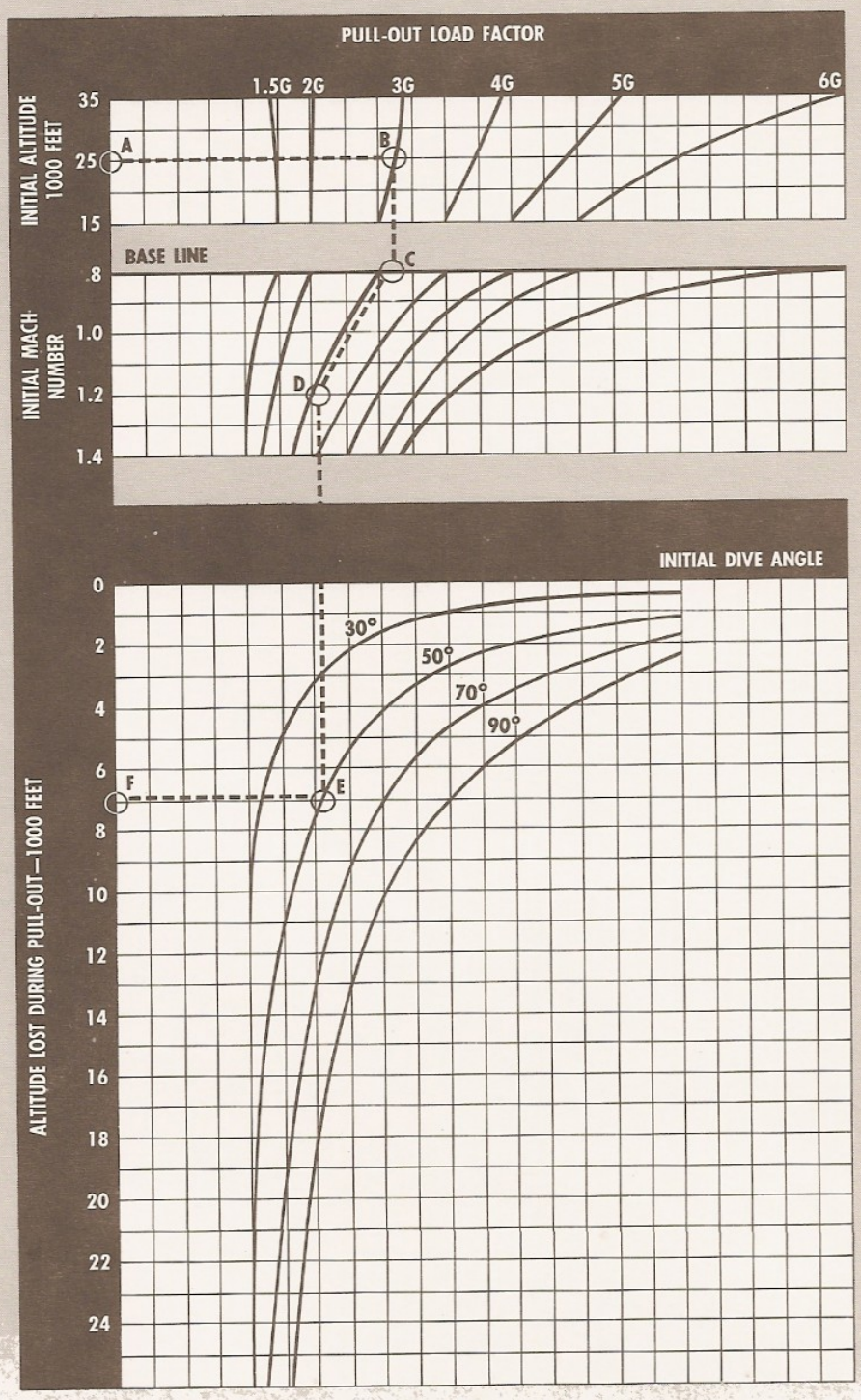
Figure 6-5

dive recovery chart

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: SPEED BRAKES OPEN
(GROSS WEIGHT=25,000 LB)
STANDARD ATMOSPHERE

ENGINE: J57-23



TF

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 (6,950 FEET).

21959-2



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START — SECOND ATTEMPT

If a second attempt to start is to be made 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. The forward (low-speed) compressor supplies a greater volume of air than can be used by the aft compressor. Excess air is vented to the atmosphere from the 9th stage manifold through a bleed valve and ducted into openings in the upper fuselage skin. A compressor bleed governor, driven by the low-speed compressor, senses changes 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.

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 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 motoring 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 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.

Under all other conditions, air motoring of the 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 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. Refer to EXCESSIVE EXHAUST GAS TEMPERATURE DURING START OR ENGINE FIRE AFTER SHUTDOWN, Section III. 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.

EMERGENCY LANDING GEAR EXTENSION SYSTEM

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 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.

Note

When the requirement for emergency gear extension is due to failure of the secondary hydraulic system, or engine failure, do not attempt to lower the gear with the normal gear handle unless the emergency gear extension system fails. 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.

Actuation of the emergency extension handle 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. The emergency extension system should be utilized whenever the normal extension system fails to extend the gear, in event of complete dc essential 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. Emergency extension should not require more than 10 seconds.

high-pressure pneumatic capability

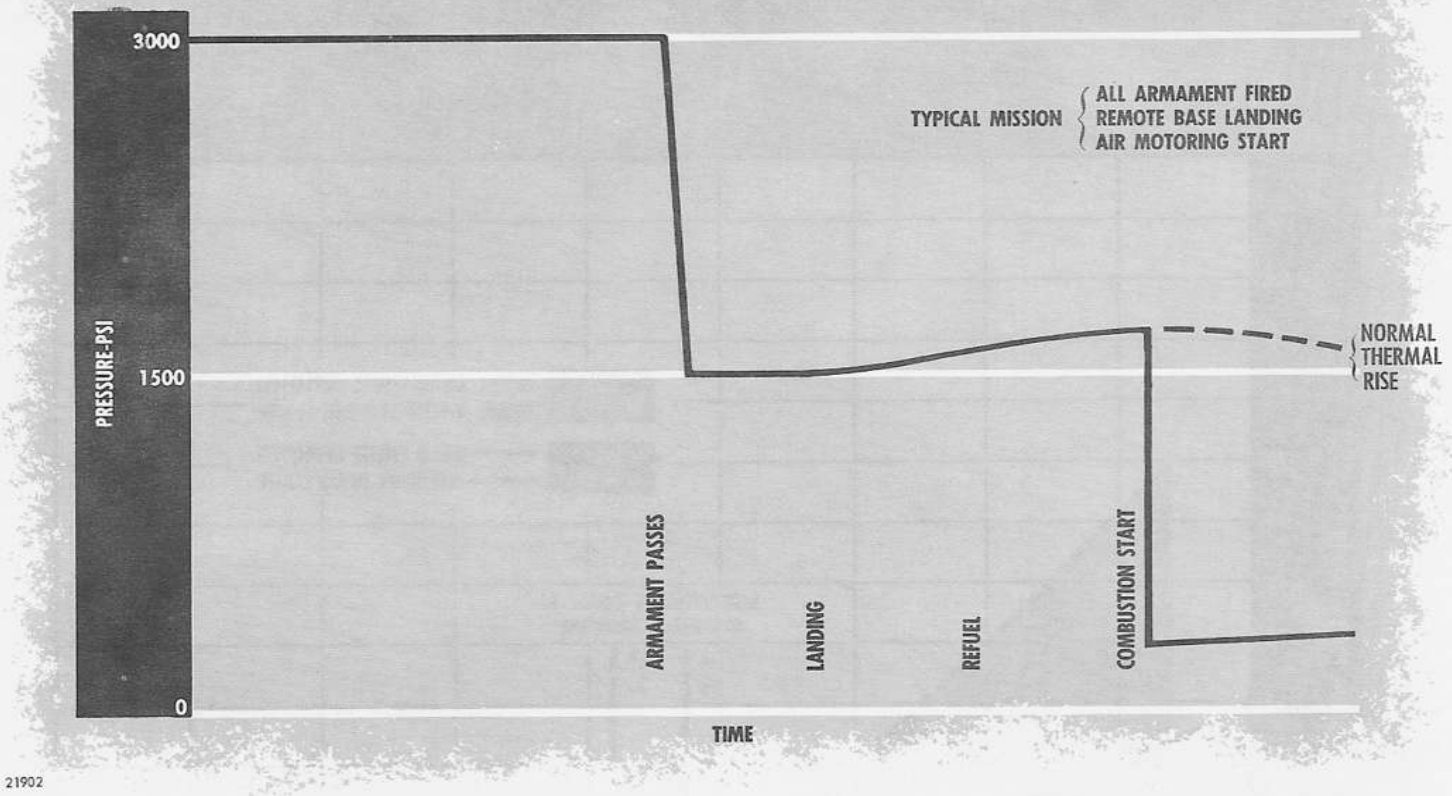


Figure 7-1

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.

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 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. If use of the pneumatic system fails to result in a safe gear down and locked indication and the secondary hydraulic system is still in operation, the hydraulic system may be used to help secure a down and locked condition. This is done by pushing the landing gear emergency extension handle IN (normal landing gear handle DOWN).

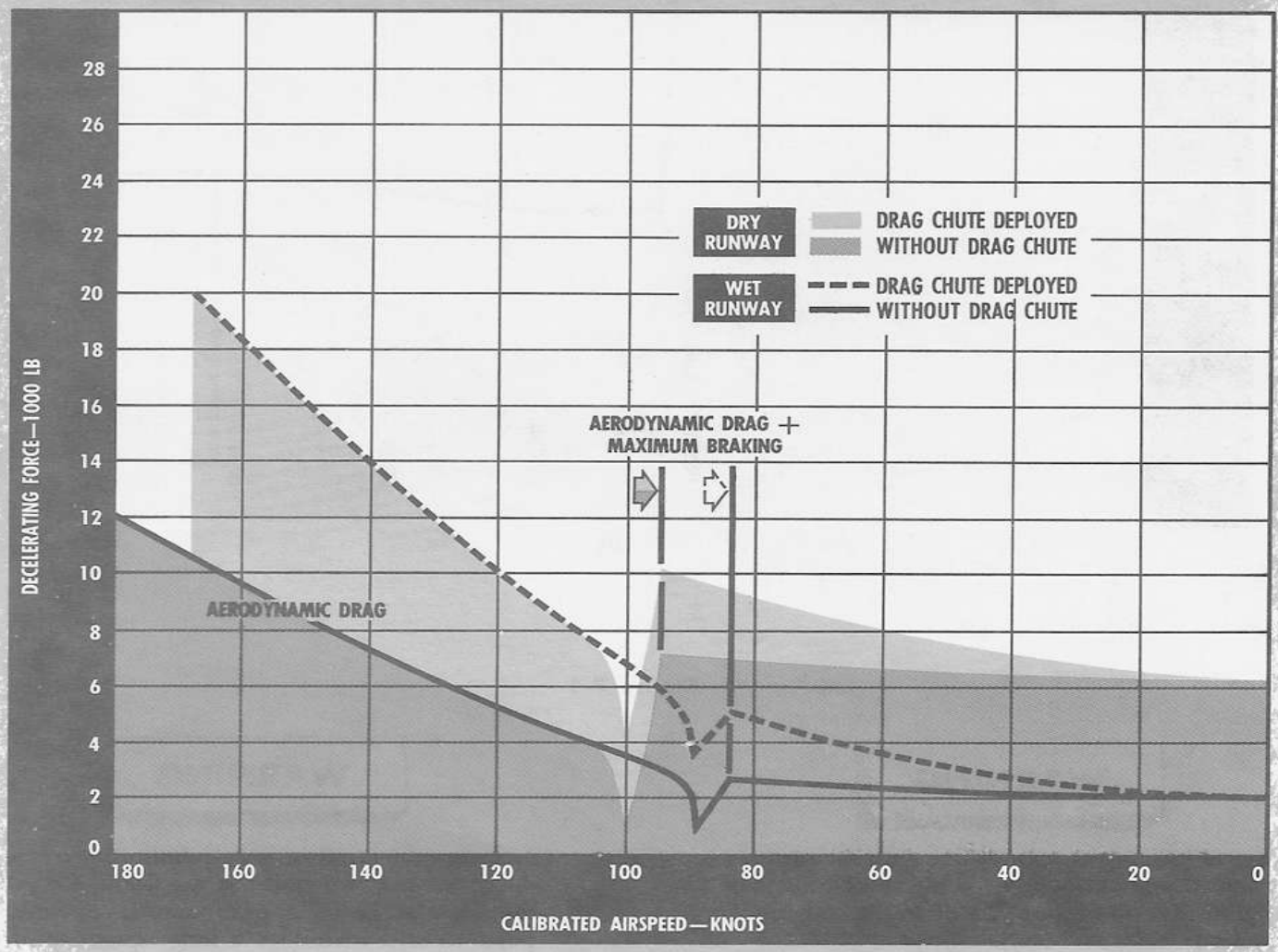
WARNING

The normal landing gear handle must remain down for the remainder of the flight. Recycling the normal landing gear handle to the UP position will introduce a large quantity of air into the secondary hydraulic system which may render it inoperative. This procedure should not be attempted in the event of primary hydraulic system failure.

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 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

landing deceleration chart normal landing



CONDITIONS

- SEA LEVEL
- STANDARD DAY
- (F) 21,500 LB GROSS WEIGHT
- (TF) 22,500 LB GROSS WEIGHT
- BRAKING COEFFICIENT OF FRICTION (μ) = 0.3 DRY RUNWAY, 0.1 WET RUNWAY
- NOSE HIGH ATTITUDE TO 105 KNOTS FOR DRY RUNWAY AND TO 95 KNOTS FOR WET RUNWAY
- MAXIMUM BRAKING BEGUN AT 100 KNOTS FOR DRY RUNWAY AND AT 90 KNOTS FOR WET RUNWAY
- SPEED BRAKES OPEN
- NO WIND

21,920

Figure 7-2

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.

WARNING

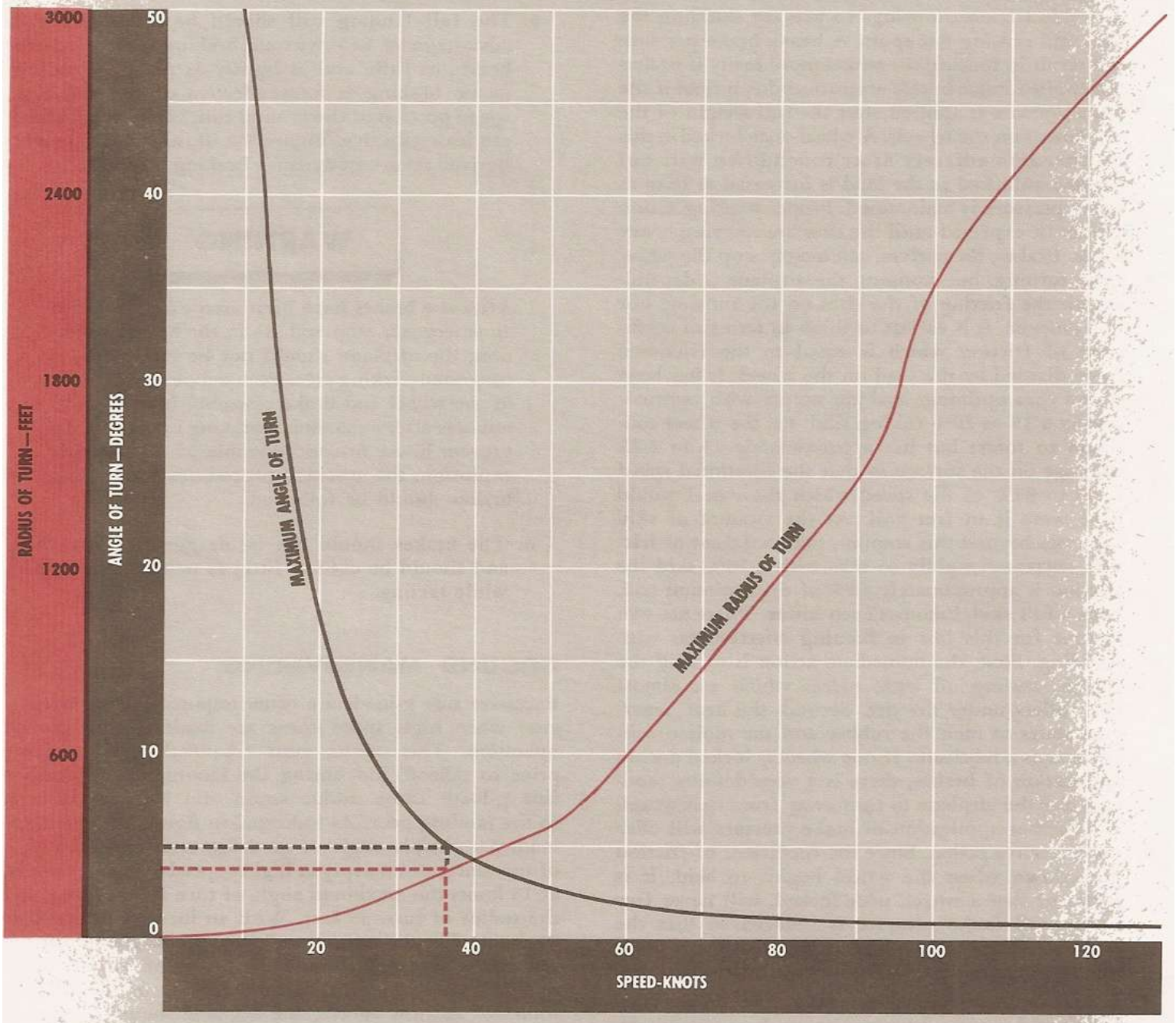
After the brakes have been used excessively for an emergency stop and are in the heated condition, the airplane should not be taxied into a congested parking area. Peak temperatures occur in the wheel and brake assembly from 5 to 15 minutes after a maximum braking operation. To prevent brake fire and possible wheel assembly explosion, the specified procedures for cooling brakes should be followed.

6. 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 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

ground maneuvering chart



EXAMPLE:
(SEE DOTTED LINE)
KNOTS: 37
ANGLE OF TURN: 5 DEGREES
RADIUS OF TURN: 230 FEET

21917

Figure 7-3

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.

SECTION VIII—CREW DUTIES

Not applicable to this airplane.



SECTION IX

all weather operation

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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.
- This manual 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.

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, Radar, and Tacan.

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 airplane heading (RMI) and magnetic compass against runway heading. Set attitude indicators at 5° nose-down. This setting will provide a more accurate indication as it will tend to offset both the nose low attitude of the airplane on the ground and 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 heading indicator is primarily for heading control, but reference should be made to runway centerline and runway lights if possible. Beginning at 125 KCAS (KIAS) apply back pressure and rotate the airplane from 5° nose-down pitch attitude to 10° nose-up pitch on the attitude indicator. Airplane will become airborne with pitch change. Raise landing gear when a positive climb indication is noted on both the vertical velocity indicator and the altimeter. Maintain approximately 7° 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 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

Upon reaching climb schedule, establish pitch attitude at approximately 6° to 8° for military thrust climb and 10° to 12° for maximum thrust climb on the attitude indicator. If the tactical situation permits, use military climb schedule when turbulence or rain are encountered.

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. Descents in instrument conditions can be made at any airspeed; however, for ease of handling, a constant airspeed descent at 275 KCAS (KIAS) is recommended.

WARNING

- Flameout and/or throttle hangup has been experienced during penetration with throttle at or near idle setting. Probable causes for this are ice formation in the fuel control, ice formation in the burner line, and icing of the airplane teleflex control system and fuel system. To reduce the possibility of flameout and unresponsive throttle movement, advance the throttle 5 per cent momentarily to a minimum of 90% rpm at 5000 feet above desired level-off altitude.
- 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.

Jet Penetration

Jet penetrations can be made by using various techniques. For ease of operation, penetrations using 82% rpm, 275 KCAS (KIAS), gear up and speed brakes out are recommended. At 5000 feet above the desired level-off altitude, momentarily advance the throttle 5 per cent to a minimum of 90% rpm. The exact procedure for jet penetrations in the "FLIP Terminal (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. Penetrations 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.

Holding

Holding airspeed for all altitudes is 250 KCAS (KIAS).

INSTRUMENT APPROACHES

Note

Under TERP's criteria, the F-102A and TF-102A are Category E airplanes.

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 KCAS (165 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 applied. Establish approximately 7° pitch attitude on the indicator and when a definite climb has been established, retract the landing gear, close the speed brakes and, if another approach is to be attempted, maintain 220 to 240 KCAS (KIAS) with approximately 90% power.

Tacan—Straight-In

Establish configuration and recommended final approach airspeed prior to reaching the final approach fix. Thrust required for level flight is approximately 90% rpm. Upon reaching the final approach fix, reduce thrust as necessary, and descend to minimum altitude.

Tacan — Circling Approach

With the recommended base leg airspeed (landing gear down and speed brakes closed) proceed from the TACAN final approach fix and/or designated point to begin a descent to the circling approach altitude. Maintain recommended base leg airspeed (landing gear down and speed brakes closed) during maneuvering in the circling approach until descent from the circling approach altitude for landing or missed approach is initiated. When this descent is initiated, open the speed brakes and reduce airspeed to final approach airspeed. See figure 9-5 for one type of circling approach.

Note

AFM 51-37 depicts various methods to maneuver for circling approaches.

ILS Approach

Plan the ILS approach carefully using the appropriate FLIP terminal instrument approach chart. Position the instrument selector switch to ILS and set the localizer front course in the course selector window. Intercept the localizer course and establish final approach airspeed with gear extended prior to glide slope interception. See figure 9-4.

Note

During ILS approaches, the FCS master switch should be in STBY to position the radar antenna downward clear of the ILS glide slope antenna.

WARNING

During an ILS approach, if disappearance of the glide-slope 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 course 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 course set knob a few degrees away from the inbound course. If the CDI responds to the course change, the signal was being received from a TACAN station and not the localizer. If the CDI does not respond to the inbound course 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 glide-slope warning (OFF) flag appearing and the glide-slope indicator will center itself and remain centered regardless of airplane movement. It will be impossible for TACAN signal information to be inadvertently displayed on the CDI if a power failure occurs in the ILS power circuit with TACAN ILS indicator selector switch in the ILS position.

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. Use the following procedures for an automatic approach:

1. FCS master switch—STBY. This positions radar antenna downward clear of the ILS glide-slope antenna.
2. AFCS engaged and operational in pilot assist mode.
3. Set ILS localizer course in course selector window.
4. Set function selector knob on J-4 compass control panel to MAG.
   This is necessary or the heading indicator will be phased to another heading reference.
5. ILS receiver (AN/ARN-31) on, frequency set, and instrument selector switch to ILS.
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 radius, 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 KCAS (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, establish final approach configuration (gear down and speed brakes as required) and reduce to final approach speed. 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.

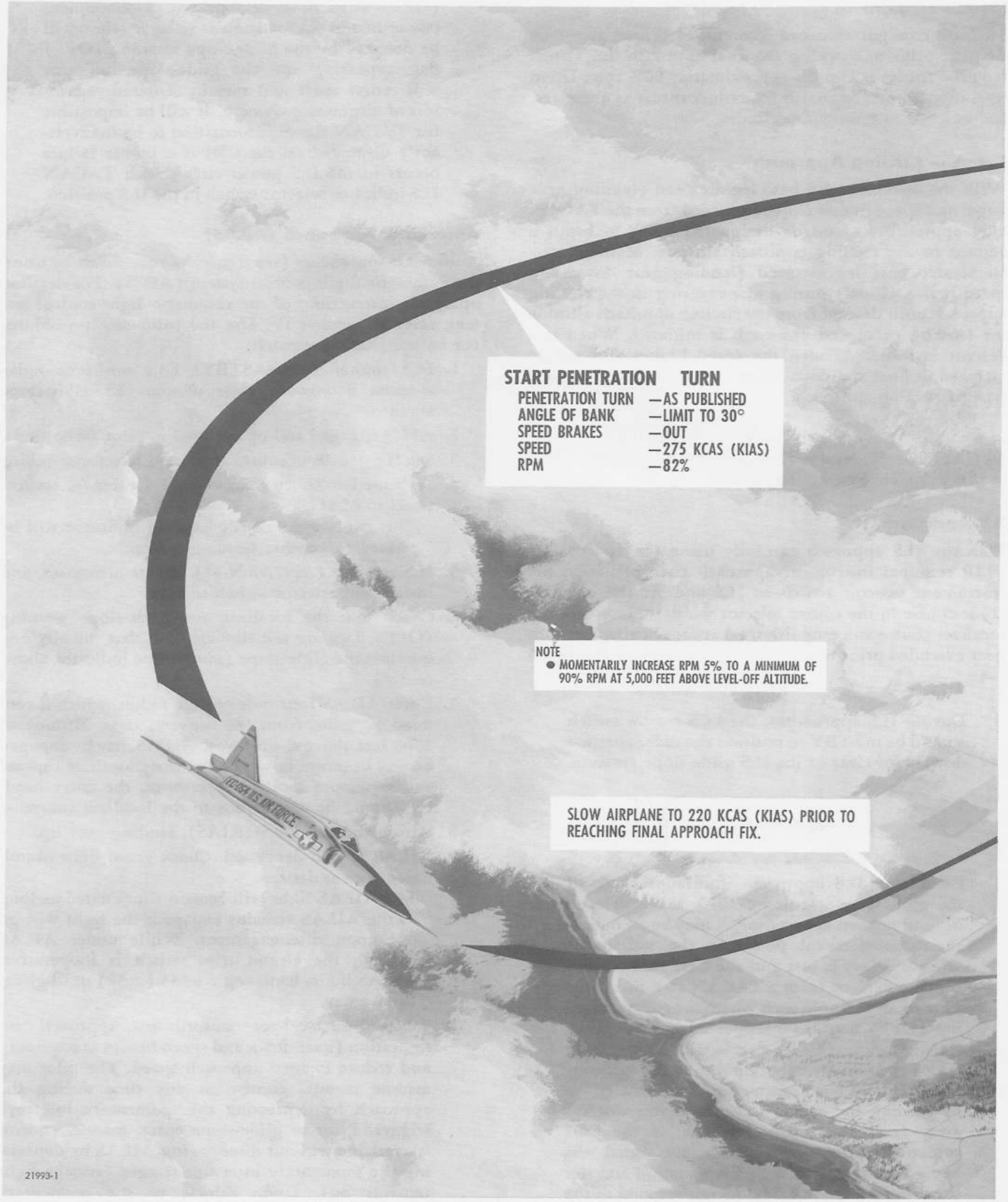
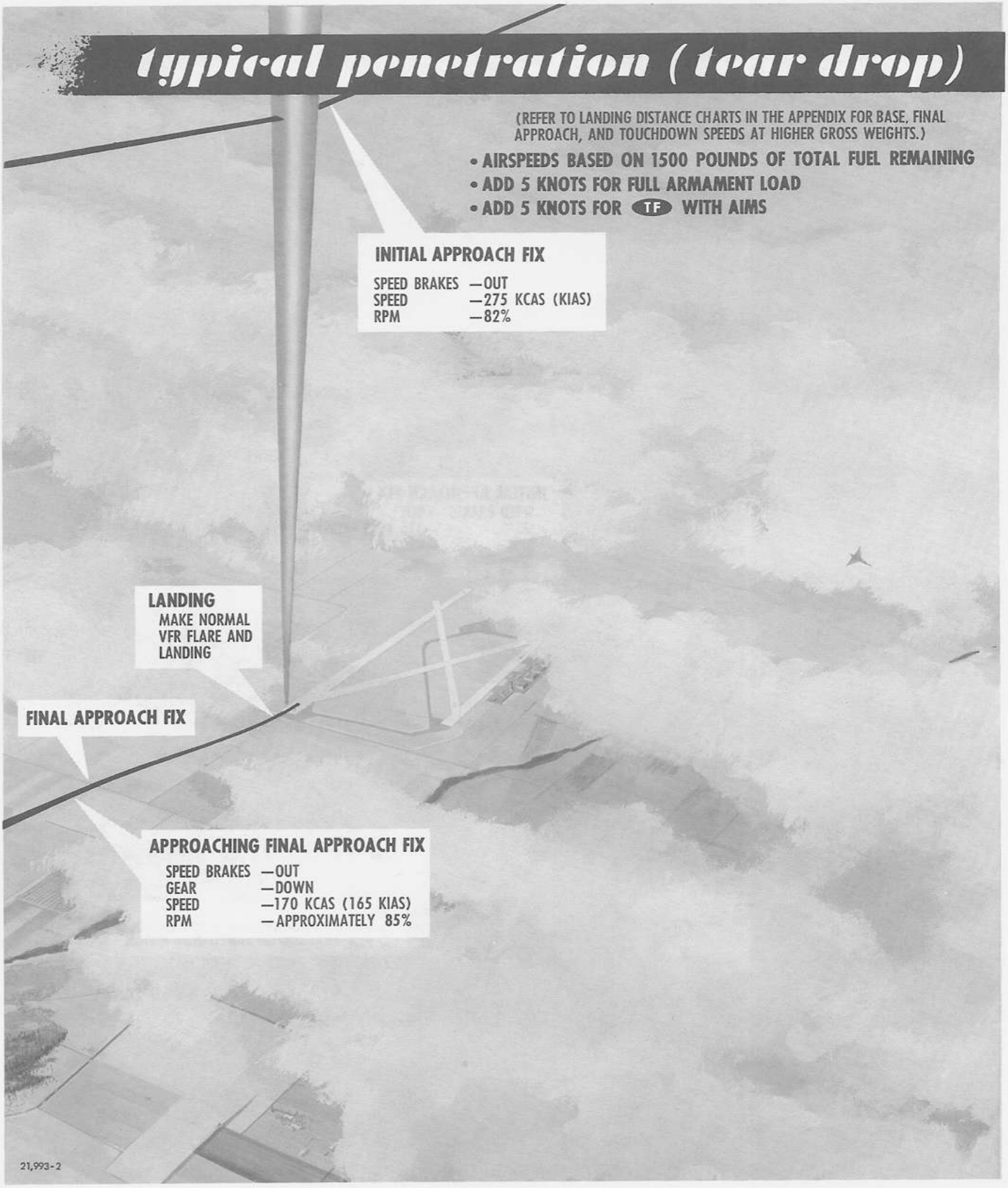

START PENETRATION TURN
PENETRATION TURN —AS PUBLISHED
ANGLE OF BANK —LIMIT TO 30°
SPEED BRAKES —OUT
SPEED —275 KCAS (KIAS)
RPM —82%
NOTE
● MOMENTARILY INCREASE RPM 5% TO A MINIMUM OF 90% RPM AT 5,000 FEET ABOVE LEVEL-OFF ALTITUDE.
SLOW AIRPLANE TO 220 KCAS (KIAS) PRIOR TO REACHING FINAL APPROACH FIX.
21993-1

Figure 9-1

typical penetration (tear drop)



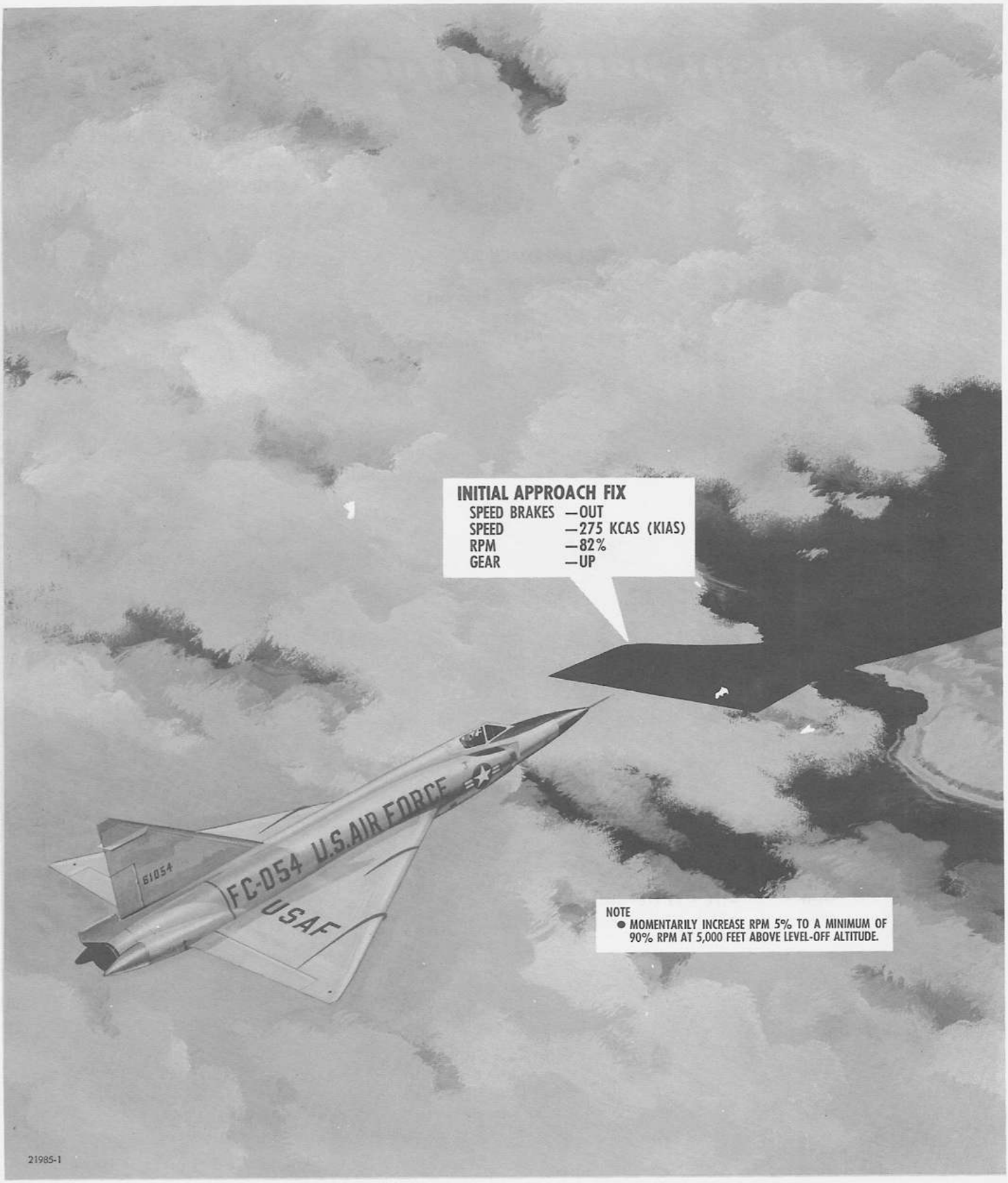

INITIAL APPROACH FIX
SPEED BRAKES —OUT
SPEED —275 KCAS (KIAS)
RPM —82%
GEAR —UP
NOTE
● MOMENTARILY INCREASE RPM 5% TO A MINIMUM OF 90% RPM AT 5,000 FEET ABOVE LEVEL-OFF ALTITUDE.
FC-054 U.S.AIR FORCE
USAF
61054
21985-1

Figure 9-2

typical penetration (straight in)

(REFER TO LANDING DISTANCE CHARTS IN THE APPENDIX FOR BASE, FINAL APPROACH, AND TOUCHDOWN SPEEDS AT HIGHER GROSS WEIGHTS.)

- AIRSPEEDS BASED ON 1500 POUNDS OF TOTAL FUEL REMAINING
- ADD 5 KNOTS FOR FULL ARMAMENT LOAD
- ADD 5 KNOTS FOR (TF) WITH AIMS

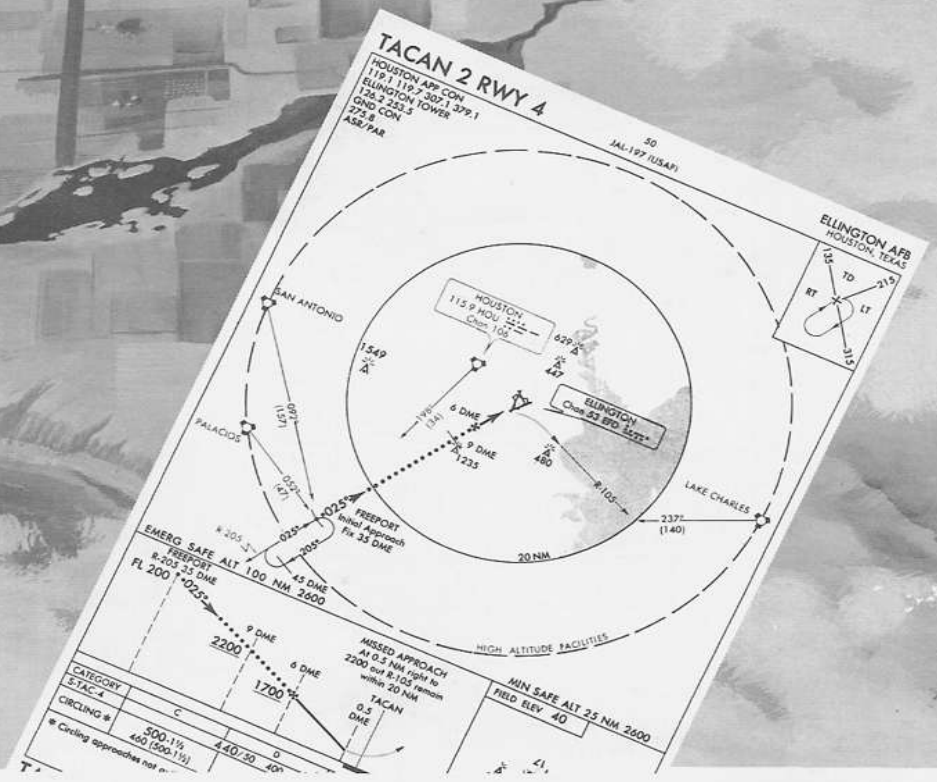
FINAL APPROACH FIX

LANDING
MAKE NORMAL VFR FLARE AND LANDING

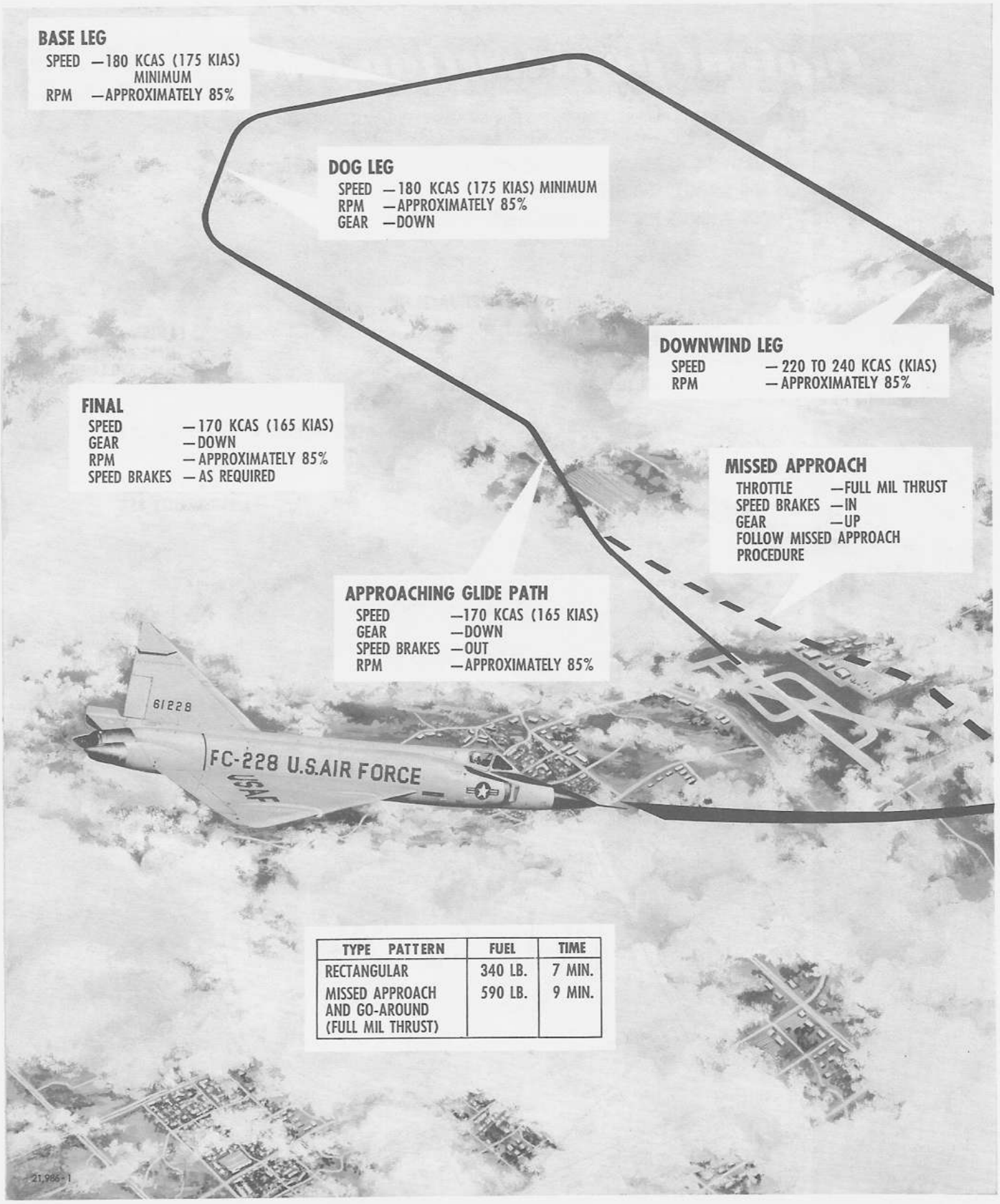
SLOW AIRPLANE TO 220 KCAS (KIAS) PRIOR TO REACHING FINAL APPROACH FIX.

APPROACHING FINAL APPROACH FIX

SPEED BRAKES	—OUT
GEAR	—DOWN
SPEED	—170 KCAS (165 KIAS)
RPM	—APPROXIMATELY 85%



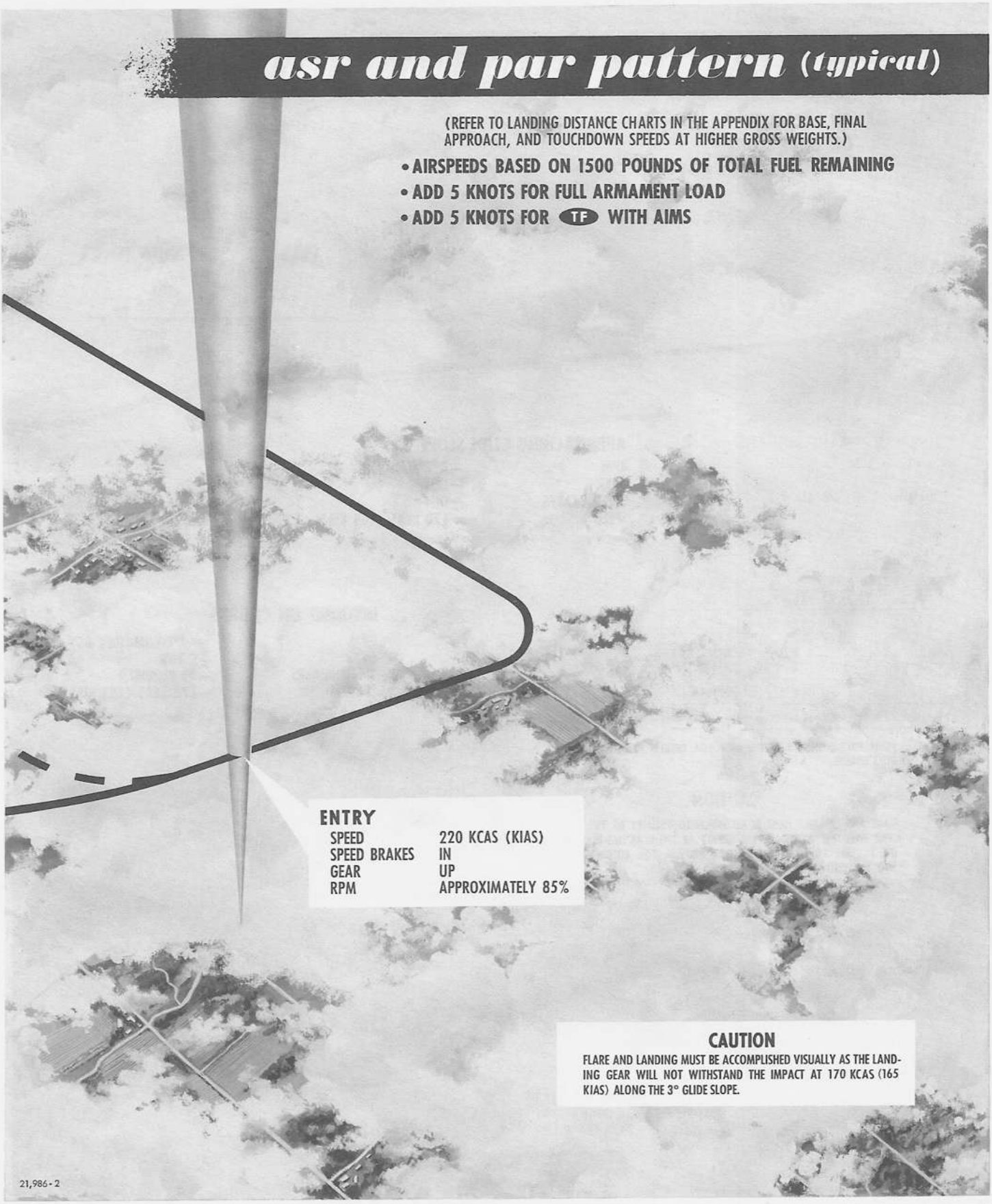
21,985-2



TYPE PATTERN	FUEL	TIME
RECTANGULAR	340 LB.	7 MIN.
MISSED APPROACH AND GO-AROUND (FULL MIL THRUST)	590 LB.	9 MIN.

Figure 9-3

asr and par pattern (typical)



(REFER TO LANDING DISTANCE CHARTS IN THE APPENDIX FOR BASE, FINAL APPROACH, AND TOUCHDOWN SPEEDS AT HIGHER GROSS WEIGHTS.)

- AIRSPEEDS BASED ON 1500 POUNDS OF TOTAL FUEL REMAINING
- ADD 5 KNOTS FOR FULL ARMAMENT LOAD
- ADD 5 KNOTS FOR TF WITH AIMS

ENTRY

SPEED	220 KCAS (KIAS)
SPEED BRAKES	IN
GEAR	UP
RPM	APPROXIMATELY 85%

CAUTION

FLARE AND LANDING MUST BE ACCOMPLISHED VISUALLY AS THE LANDING GEAR WILL NOT WITHSTAND THE IMPACT AT 170 KCAS (165 KIAS) ALONG THE 3° GLIDE SLOPE.

21,986-2

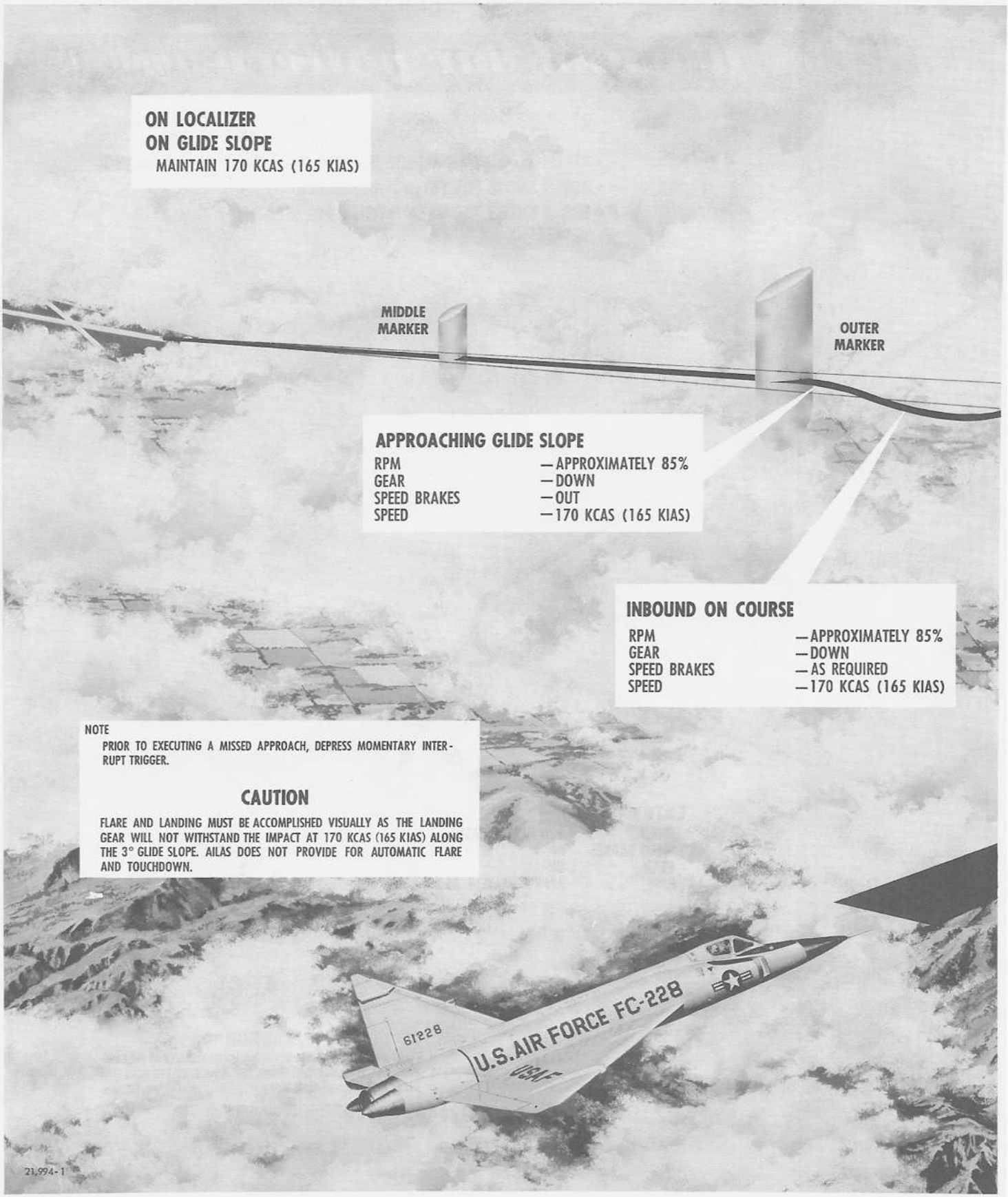
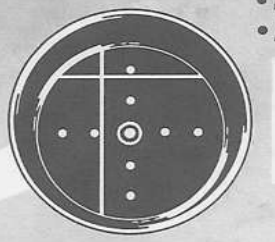

ON LOCALIZER
ON GLIDE SLOPE
MAINTAIN 170 KCAS (165 KIAS)
MIDDLE MARKER
OUTER MARKER
APPROACHING GLIDE SLOPE
RPM —APPROXIMATELY 85%
GEAR —DOWN
SPEED BRAKES —OUT
SPEED —170 KCAS (165 KIAS)
INBOUND ON COURSE
RPM —APPROXIMATELY 85%
GEAR —DOWN
SPEED BRAKES —AS REQUIRED
SPEED —170 KCAS (165 KIAS)
NOTE
PRIOR TO EXECUTING A MISSED APPROACH, DEPRESS MOMENTARY INTERRUPT TRIGGER.
CAUTION
FLARE AND LANDING MUST BE ACCOMPLISHED VISUALLY AS THE LANDING GEAR WILL NOT WITHSTAND THE IMPACT AT 170 KCAS (165 KIAS) ALONG THE 3° GLIDE SLOPE. AILAS DOES NOT PROVIDE FOR AUTOMATIC FLARE AND TOUCHDOWN.
61228
U.S.AIR FORCE FC-228
USAF
21,994-1

Figure 9-4

automatic approach (ailas)

(REFER TO LANDING DISTANCE CHARTS IN THE APPENDIX FOR BASE, FINAL APPROACH, AND TOUCHDOWN SPEEDS AT HIGHER GROSS WEIGHTS.)

- AIRSPEEDS BASED ON 1500 POUNDS OF TOTAL FUEL REMAINING
- ADD 5 KNOTS FOR FULL ARMAMENT LOAD
- ADD 5 KNOTS FOR (TF) WITH AIMS

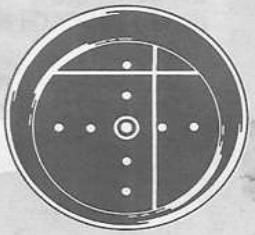


LOCALIZER OVERSHOOT

DEPENDENT UPON ENGAGE HEADING AND DISTANCE FROM RUNWAY.

INTERCEPT LOCALIZER

AUTOMATIC 45° HEADING TO LOCALIZER UNTIL LOCALIZER BEAM INTERCEPTION.



45° ENGAGE
220 KCAS (KIAS)

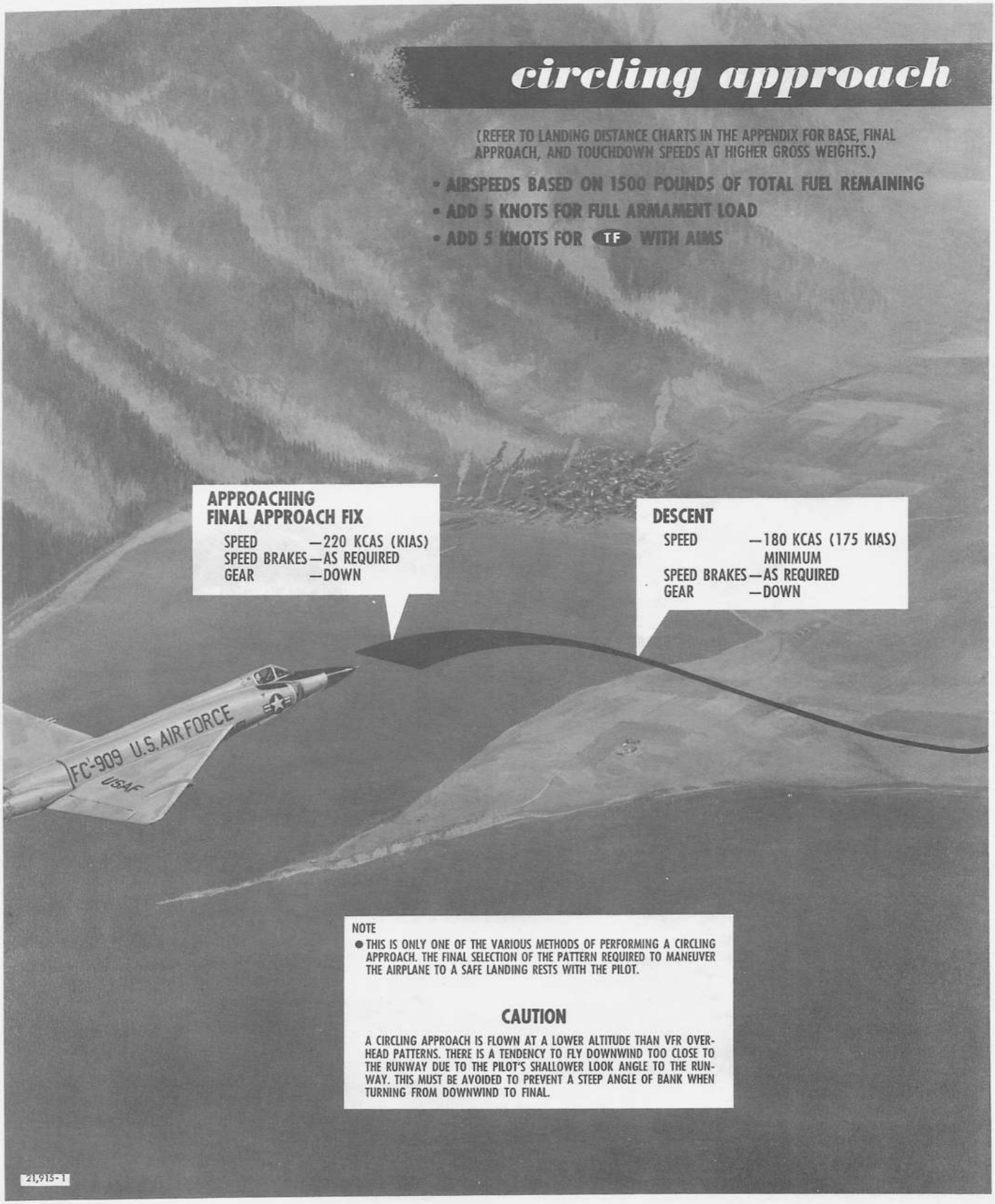
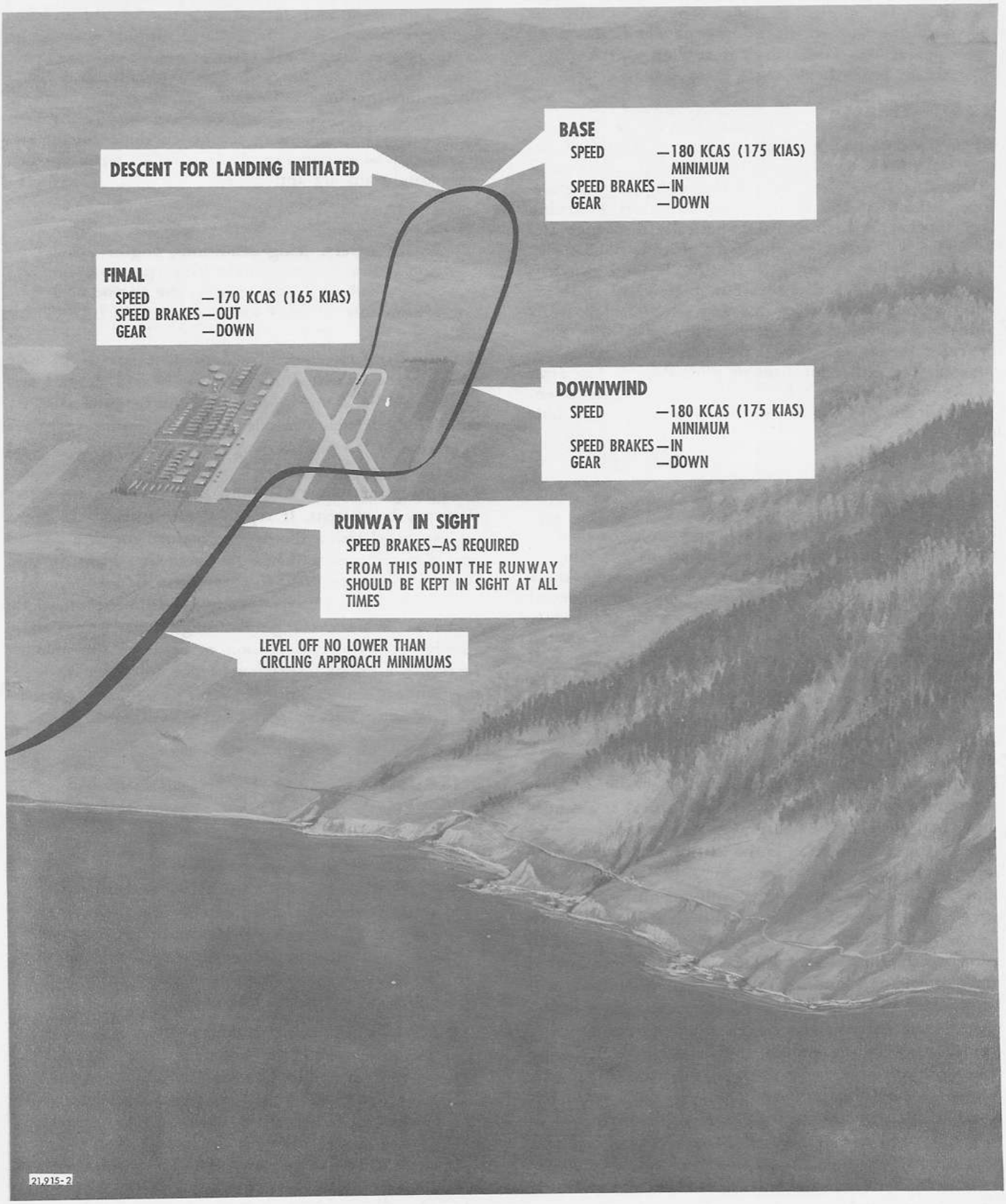

circling approach
(REFER TO LANDING DISTANCE CHARTS IN THE APPENDIX FOR BASE, FINAL APPROACH, AND TOUCHDOWN SPEEDS AT HIGHER GROSS WEIGHTS.)
• AIRSPEEDS BASED ON 1500 POUNDS OF TOTAL FUEL REMAINING
• ADD 5 KNOTS FOR FULL ARMAMENT LOAD
• ADD 5 KNOTS FOR TF WITH AIMS
APPROACHING FINAL APPROACH FIX
SPEED —220 KCAS (KIAS)
SPEED BRAKES—AS REQUIRED
GEAR —DOWN
DESCENT
SPEED —180 KCAS (175 KIAS) MINIMUM
SPEED BRAKES—AS REQUIRED
GEAR —DOWN
FC-909 U.S. AIR FORCE
USAF
NOTE
● THIS IS ONLY ONE OF THE VARIOUS METHODS OF PERFORMING A CIRCLING APPROACH. THE FINAL SELECTION OF THE PATTERN REQUIRED TO MANEUVER THE AIRPLANE TO A SAFE LANDING RESTS WITH THE PILOT.
CAUTION
A CIRCLING APPROACH IS FLOWN AT A LOWER ALTITUDE THAN VFR OVERHEAD PATTERNS. THERE IS A TENDENCY TO FLY DOWNWIND TOO CLOSE TO THE RUNWAY DUE TO THE PILOT'S SHALLOWER LOOK ANGLE TO THE RUNWAY. THIS MUST BE AVOIDED TO PREVENT A STEEP ANGLE OF BANK WHEN TURNING FROM DOWNWIND TO FINAL.
21,915-1

Figure 9-5


DESCENT FOR LANDING INITIATED
BASE
SPEED —180 KCAS (175 KIAS) MINIMUM
SPEED BRAKES —IN
GEAR —DOWN
FINAL
SPEED —170 KCAS (165 KIAS)
SPEED BRAKES —OUT
GEAR —DOWN
DOWNWIND
SPEED —180 KCAS (175 KIAS) MINIMUM
SPEED BRAKES —IN
GEAR —DOWN
RUNWAY IN SIGHT
SPEED BRAKES—AS REQUIRED
FROM THIS POINT THE RUNWAY SHOULD BE KEPT IN SIGHT AT ALL TIMES
LEVEL OFF NO LOWER THAN CIRCLING APPROACH MINIMUMS
21,915-2

11. 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.
12. Landing lights—ON.

When visual contact with the runway is established, depress the emergency damper disconnect button, flare out and touch down. With AILAS engaged prior to glide-slope capture, if the localizer signal is lost (warning flags appear), AILAS will automatically disengage and the AFCS will revert to pilot assist altitude hold if altitude hold switch is engaged. If altitude hold switch is not engaged, AFCS will revert to pilot assist (attitude hold) maintaining heading in course selector window. At glide-slope entry, the altitude hold switch will automatically return to "OFF" if engaged. If AILAS is engaged, and 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).

ICE AND RAIN

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 drag from surface ice buildups. Surface icing will reduce 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°) at sea level to −24°C (−12°F) at 20,000 feet. Above 20,000 feet, due to the inability of the 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 radar or ILS approach) 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. The possibility of this occurring 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 operated in MANUAL ON position when icing conditions are anticipated.

Note

Under adverse icing conditions, engine operation below approximately 91% rpm may not supply sufficient heat to keep the engine compressor inlet clear of ice.

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.

TURBULENCE AND THUNDERSTORMS

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, lightning strike, or turbulence.
- 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 "ice crystal" 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 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 KCAS (KIAS) not to exceed Mach .92 (.88 IMN) 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. 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.

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.
- When it is necessary to penetrate weather conditions conducive to compressor stalls, stabilize engine rpm at approximately 91%, if practicable.

NIGHT FLYING

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.

COLD WEATHER PROCEDURES

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. The 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.

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 as necessary 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.



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Part 1 INTRODUCTION

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INTRODUCTION

Note

- Appendix I contains performance data for operation of (F) and (TF) airplanes with or without external fuel tanks. The appendix is divided into nine parts, each pertaining to a specific phase of flight, and contains all charts (clean or with tanks) pertinent to that phase. The information is applicable to airplanes with either the Case X or the Case XX wing. In general, performance of the two configurations is similar. However, at high altitudes, airplanes of the Case XX configuration will provide significantly better performance than those having the Case X wing. Flight tests have demonstrated that the Case XX wing provides a 4200-foot increase in maximum combat ceiling and a Mach .06 speed increase at 50,000 feet, over the case X wing configuration.
- The data throughout this Appendix are based on aircraft having the AIMS modification. Conversion data is provided for converting indicated Mach and airspeed to true Mach and calibrated airspeed for use with these charts.

The charts in this section present the flight-tested performance for this airplane. These data are based on operation with the J57-P-23 engine and are representative of airplanes in the production configuration. Performance data will vary among airplanes and pilots. Performance deteriorates slightly with age of the airplane. The engine, for example, even though it is trimmed to thrust output specified, will show an increase in fuel flow with an accumulation of hours. Performance of the airplane also deteriorates with improperly fitting door panels, damaged or dented wing and fuselage skin areas. The term "clean configuration" as applied to these charts implies no optional external stores (wing tanks) and no drag producing components extended, such as speed brakes or landing gear. In all cases, the ICAO standard atmosphere is used (except on takeoff and landing charts where conditions are generalized).

MACH NUMBER CORRECTION (AIMS)

To determine either true or indicated Mach number, see figures A1-1 and A1-2 and enter at left with true Mach number. Follow across to diagonal, then read down to obtain indicated Mach number. Reverse procedure to obtain true Mach number. Example: True Mach number equals 0.900. Enter from left at 0.900, follow across to diagonal, and read down for indicated Mach number, 0.904. The reverse procedure for an indicated Mach number of 0.950 gives a true Mach number of 0.945. These relationships are true regardless of altitude or free air temperature.

MACH NUMBER CORRECTION (NON-AIMS)

To determine either true or indicated Mach number, see figures A1-3 and A1-4 and enter at left with true Mach number. Follow across to diagonal, then read down to obtain indicated Mach number. Reverse procedure to obtain true Mach number. Example: True Mach number equals 0.700. Enter from left at 0.700, follow across to diagonal, and read down for indicated Mach number, 0.680. The reverse procedure for an indicated Mach number of 0.700 gives a true Mach number of 0.725. These relationships are true regardless of altitude or free air temperature. Above Mach 1.025, true Mach number equals indicated Mach number.

AIRSPEED CORRECTION (AIMS)

Figures A1-5 and A1-6 show the airspeed correction in terms of indicated airspeed and indicated altitude.

Note

Indicated airspeed (IAS) is the instrument reading. Calibrated airspeed (CAS) is the IAS corrected for installation error. Equivalent airspeed (EAS) is the CAS corrected for compressibility effects, and true airspeed (TAS) is the EAS corrected for atmospheric density effects.

Indicated Mach number is the instrument reading and true Mach number is the indicated Mach number corrected for installation error. In all cases, IAS and indicated Mach number are instrument readings without instrument corrections applied.

AIRSPEED CORRECTION (NON-AIMS)

Figures A1-3 and A1-4 show the airspeed correction in terms of indicated airspeed and indicated altitude.

Note

Indicated airspeed (IAS) is the instrument reading. Calibrated airspeed (CAS) is the IAS corrected for installation error. Equivalent airspeed (EAS) is the CAS corrected for compressibility effects, and true airspeed (TAS) is the EAS corrected for atmospheric density effects.

Indicated Mach number is the instrument reading and true Mach number is the indicated Mach number corrected for installation error. In all cases, IAS and indicated Mach number are instrument readings without instrument corrections applied.

ALTITUDE CORRECTION (AIMS)

No altitude correction for installation error is required with the altimeter in RESET (servo mode). The altitude correction for STANDBY is presented in two forms. Figures A1-9 and A1-10 show the altitude correction in terms of KIAS and figures A1-11 and A1-12 show the correction in terms of indicated Mach number. Indicated altitude is the instrument reading without instrument error considered. True pressure altitude is obtained from figures A1-9 and A1-10 by entering the column for IAS and reading down to the line for indicated altitude. The corrective increment is added to the indicated altitude. Example: Entering figure A1-9 with 350 KIAS at an altitude of 25,000 feet gives an increment of −40 feet. The true pressure altitude is then 24,960 feet. If the chart was entered with true pressure altitude, the reverse would be true. Example: Entering the chart with 350 KIAS and a true pressure altitude of 25,000 feet gives the same increment of −40 feet, but this time the increment is subtracted. The indicated altitude would then be 25,040 feet. These corrections are also available on an indicated Mach number basis from figure A1-11. Read down the column for indicated Mach number to the line for indicated altitude and read the correction. Example: Indicated Mach number equals 0.83 and indicated altitude equals 25,000 feet. A linear interpolation between 0.80 and 0.85 indicated Mach number gives a correction of −40 feet. True pressure altitude is then 24,960 feet. If the chart was entered with true pressure altitude, the correction would be reversed and the indicated altitude would be 25,040 feet.

ALTITUDE CORRECTION (NON-AIMS)

For non-AIMS aircraft, the altitude correction for installation error is given in terms of KIAS in Figures A1-13 and A1-14, and in terms of indicated Mach number in Figures A1-15 and A1-16. Chart use is the same as described previously for altitude corrections with AIMS (altimeter in standby).

COMPRESSIBILITY CORRECTION TO CALIBRATED AIRSPEED

Figure A1-17 converts calibrated airspeed to equivalent airspeed. CAS $-\Delta V_c =$ EAS. Mach number may be determined by interpolation. True airspeed may be obtained by multiplying EAS times $1/\sqrt{\sigma}$. The term, $1/\sqrt{\sigma}$, may be obtained from standard atmosphere tables. For this case an ambient temperature greater than standard would result in a conservative true airspeed. The opposite is true for a temperature lower than standard. True airspeed may be determined more accurately by use of figure A1-18, density altitude chart. As noted by guide lines the value of $1/\sqrt{\sigma}$ is determined for a known pressure altitude and ambient temperature.

DENSITY ALTITUDE CHART

Density altitude (figure A1-18) may be found for a given temperature and pressure altitude condition.

Sample Problem

a. Outside air temperature— −30°C.

b. Pressure altitude—35,000 feet.

c. $\frac{1}{\sqrt{\sigma}} = 1.90$.

d. Density altitude—37,500 feet.

This chart also provides a $\frac{1}{\sqrt{\sigma}}$ value necessary to change equivalent airspeed to true airspeed. Enter the chart with the given temperature condition, proceed vertically to the pressure altitude, read horizontally to the right to obtain $\frac{1}{\sqrt{\sigma}}$ value. True airspeed may then be

obtained from equivalent airspeed by multiplying the given equivalent airspeed by the $\frac{1}{\sqrt{\sigma}}$ value.

TRUE AIRSPEED — TRUE MACH NUMBER AND TEMPERATURE CHART

Figure A1-19 shows true airspeed as a function of temperature and true Mach number. The total temperature scale represents the direct dial reading minus instrument correction, if any. The lines of constant temperature are true ambient conditions (adiabatic temperature rise removed). This chart ties together four functions: true airspeed, true Mach number, total air temperature, and ambient air temperature. Any two of these functions will determine the remaining two. Example: At an ICAO standard pressure altitude of 30,000 feet, the ambient temperature is −44.4°C (figure A1-19 or figures A1-20 and A1-21). At a true Mach number of 0.95 the true airspeed is 559 knots and the total temperature is −5.6°C. Conversely, an indicator reading of −5.6°C total temperature at a true Mach number of 0.95 shows the true airspeed to be 559 knots.

F

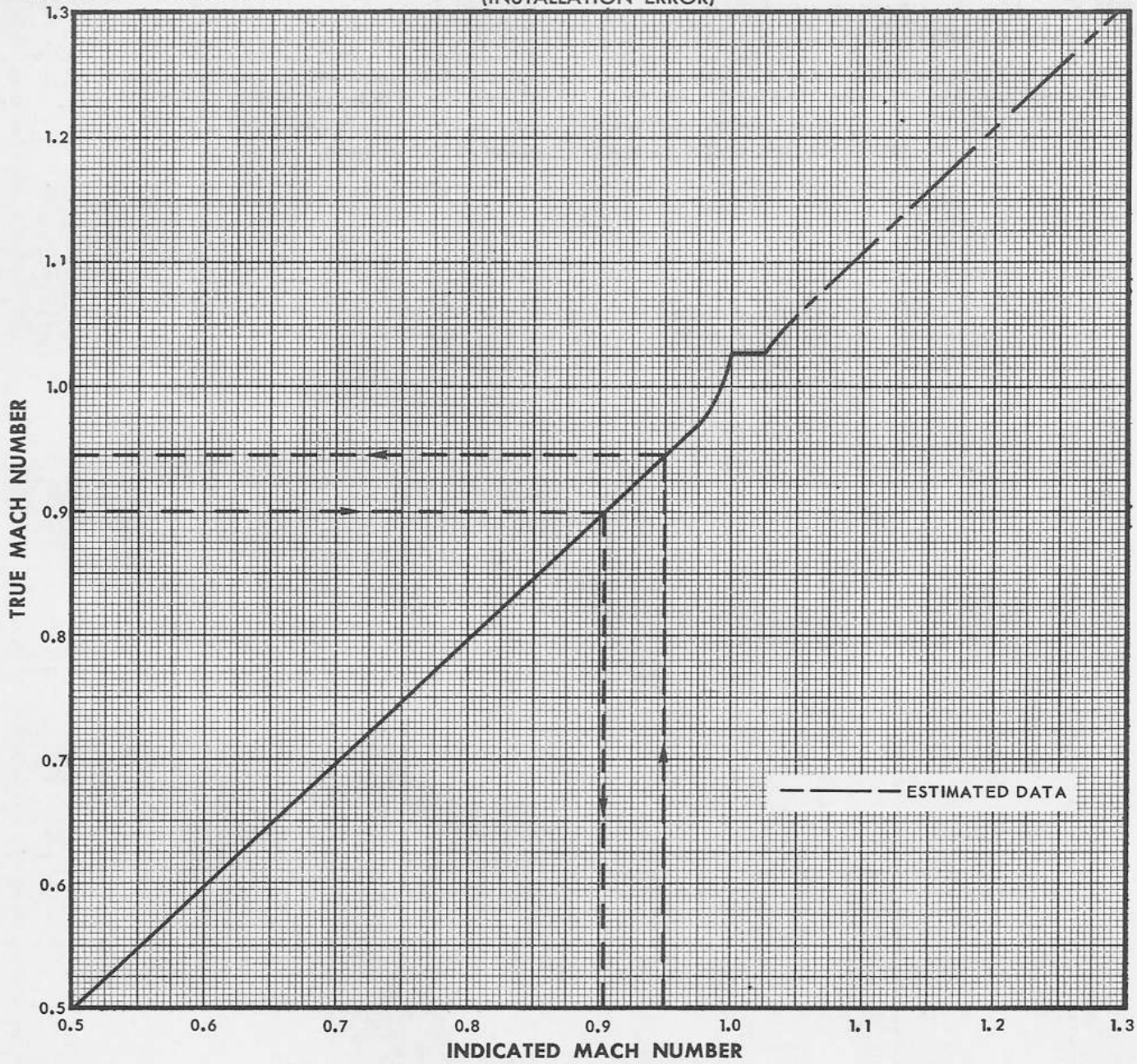
mach number correction

MODEL: F-102A
DATE: 5 AUGUST 1971
DATA BASIS: FLIGHT TEST

AIMS AIRPLANES

(INSTALLATION ERROR)

ENGINE: J57-23



NOTE

ENTER AT LEFT WITH TRUE MACH NUMBER THEN READ DOWN TO OBTAIN INDICATED MACH NUMBER

ENTER AT BOTTOM WITH INDICATED MACH NUMBER THEN READ LEFT FOR TRUE MACH NUMBER

AT INDICATED MACH NUMBERS ABOVE 1.30 ADD .01 TO OBTAIN TRUE MACH NUMBER

BELOW MACH 0.50, TRUE AND INDICATED MACH ARE EQUAL

22,001E

Figure A1-1

TF

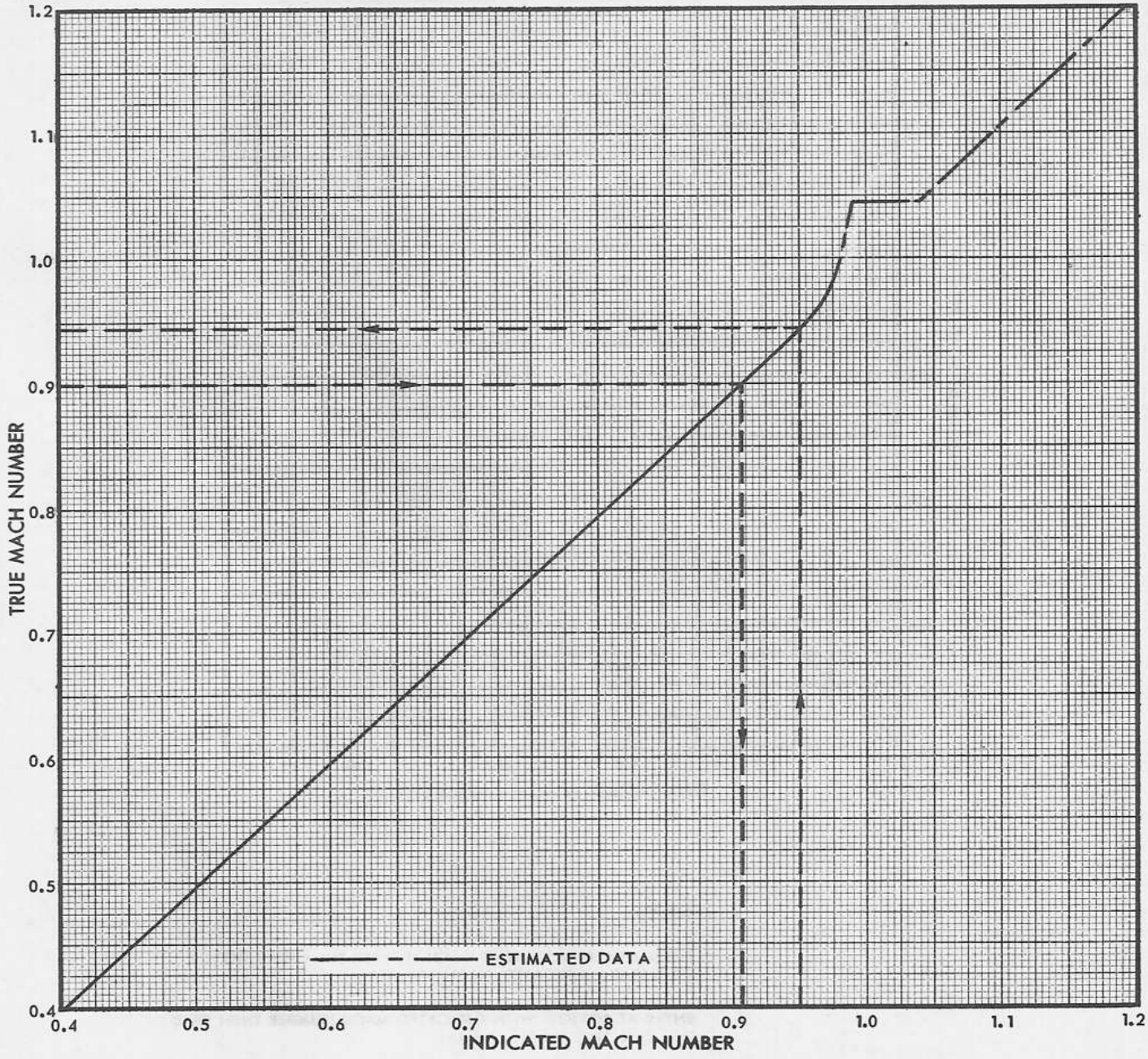
mach number correction

AIMS AIRPLANES

(INSTALLATION ERROR)

MODEL: TF-102A
DATE: 5 AUGUST 1971
DATA BASIS: FLIGHT TEST

ENGINE: J57-23



NOTE

ENTER AT LEFT WITH TRUE MACH NUMBER THEN READ DOWN TO OBTAIN INDICATED MACH NUMBER

ENTER AT BOTTOM WITH INDICATED MACH NUMBER THEN READ LEFT FOR TRUE MACH NUMBER

AT INDICATED MACH NUMBERS ABOVE 1.20 ADD .01 TO OBTAIN TRUE MACH NUMBER

BELOW MACH 0.40, TRUE AND INDICATED MACH ARE EQUAL

22,161B

Figure A1-2

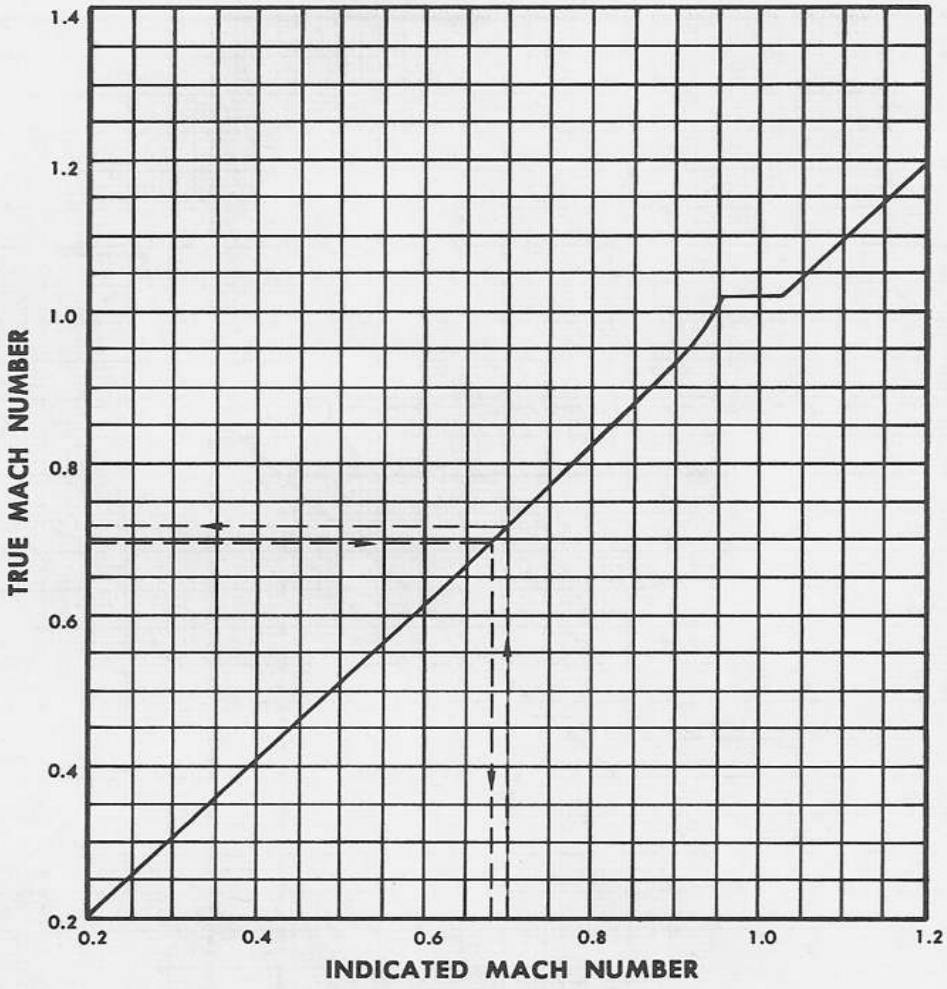
F

mach number correction

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

NON-AIMS AIRPLANES
(INSTALLATION ERROR)

ENGINE: J57-23



NOTE

ENTER AT LEFT WITH TRUE MACH NUMBER THEN READ DOWN TO OBTAIN INDICATED MACH NUMBER

ENTER AT BOTTOM WITH INDICATED MACH NUMBER THEN READ LEFT FOR TRUE MACH NUMBER

AT MACH 1.025 AND ON, TRUE AND INDICATED MACH ARE EQUAL

22,432A

Figure A1-3

TF

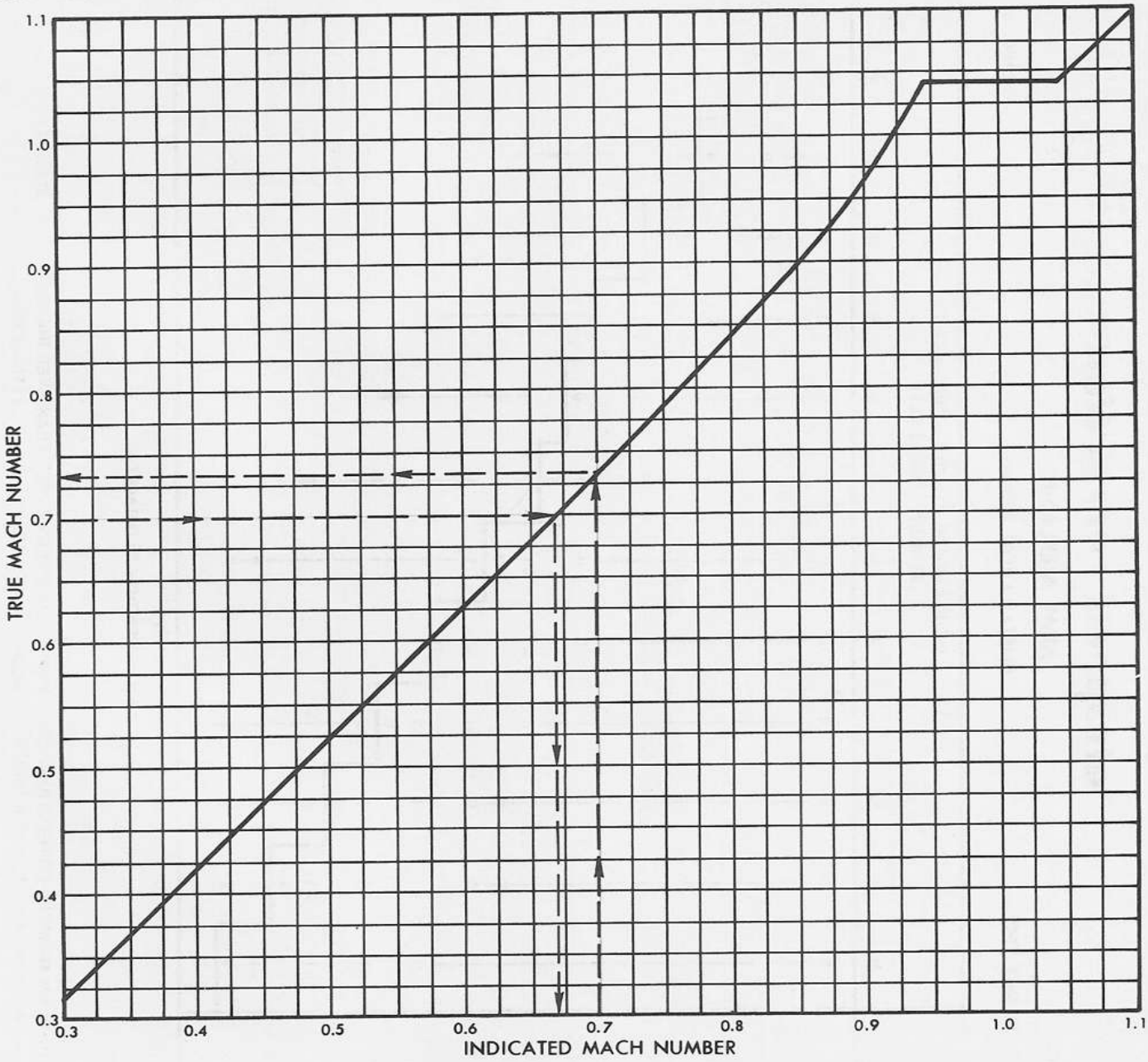
mach number correction

NON-AIMS AIRPLANES

(INSTALLATION ERROR)

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

ENGINE: J57-23



NOTE

ENTER AT LEFT WITH TRUE MACH NUMBER THEN READ DOWN TO OBTAIN INDICATED MACH NUMBER.

ENTER AT BOTTOM WITH INDICATED MACH NUMBER THEN READ LEFT FOR TRUE MACH NUMBER.

AT MACH 1.045 AND ON, TRUE AND INDICATED MACH ARE EQUAL.

22,433A

Figure A1-4

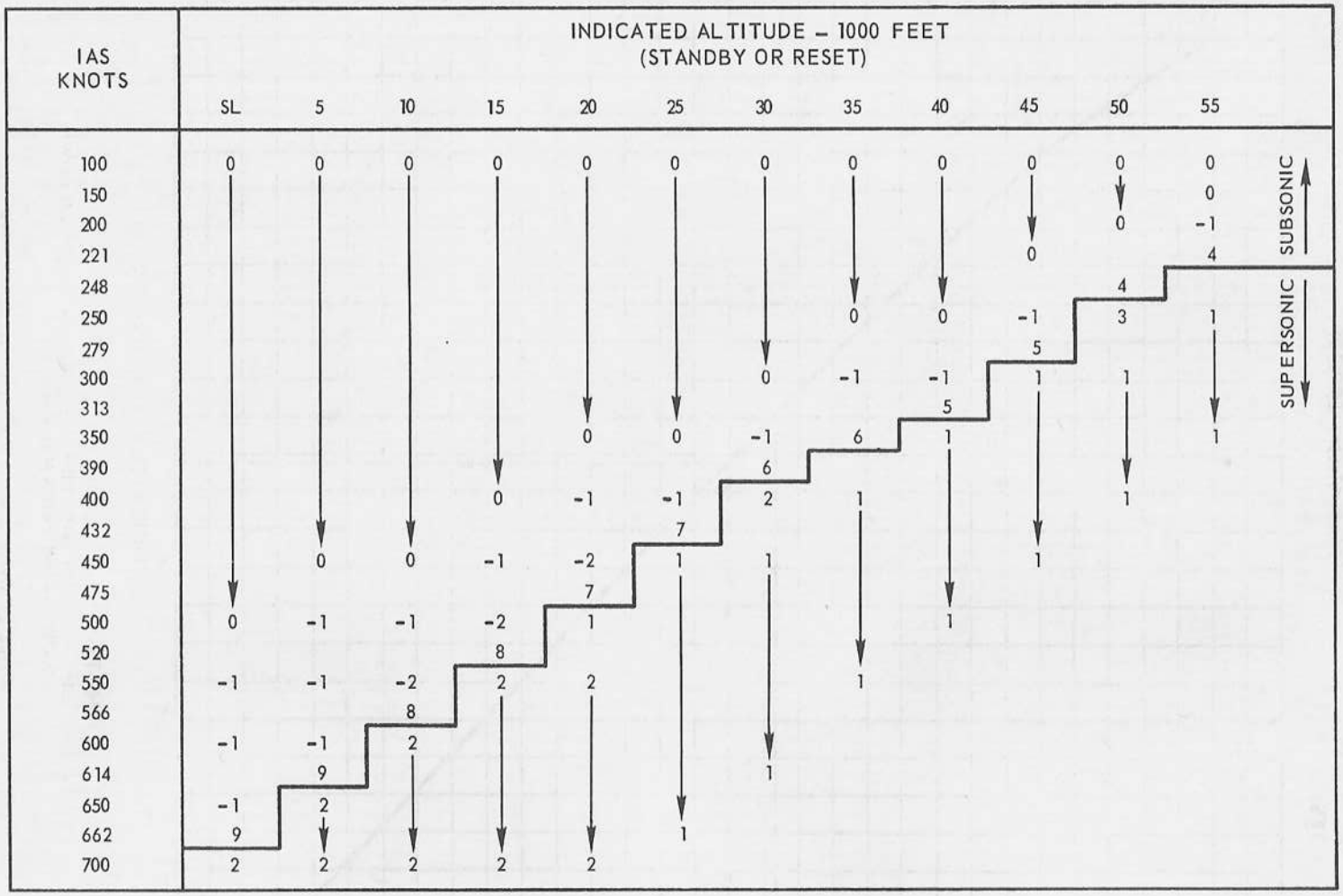
airspeed correction

MODEL: F-102A
DATE: 5 AUGUST 1971
DATA BASIS: FLIGHT TEST

AIMS AIRPLANES

(INSTALLATION ERROR)

ENGINE: J57-23



ADD CORRECTION TO OBTAIN CAS

NOTE

1. MAXIMUM AIRSPEED CORRECTIONS ARE GIVEN AT AIRSPEEDS OTHER THAN EVEN INCREMENTS OF 50 KNOTS.
2. AIRSPEED CORRECTIONS FOR INDICATED MACH NUMBERS ABOVE 1.05 ARE ESTIMATED.

22,002E

Figure A1-5

F

airspeed correction

MODEL: TF-102A
DATE: 5 AUGUST 1971
DATA BASIS: FLIGHT TEST

AIMS AIRPLANES

(INSTALLATION ERROR)

ENGINE: J57-23

IAS KNOTS	INDICATED ALTITUDE – 1000 FEET (STANDBY OR RESET)												
	SL	5	10	15	20	25	30	35	40	45	50	55	
100	0	0	0	0	0	0	0	0	0	0	0	0	SUBSONIC
150	-1	-1	-1	-1	-1	-1	-1	-1	-1	-1	-1	-1	
200	-1	-1	-1	-1	-1	-1	-1	-1	-1	-1	-1	-1	
218												8	
245											9		
250	-1	-1	-1	-1	-1	-1	-1	-1	-1	-1	6	1	SUPERSONIC
275										10			
300	-1	-1	-1	-1	-1	-1	-2	-2	-1	1	1	1	
309									11				
346								12					
350	-1	-1	-2	-2	-2	-2	-2	10	1	1	1	1	
385							13						
400	-2	-2	-2	-2	-2	-2	6	1	1	1	1		
427						14							
450	-2	-2	-2	-2	-2	3	1	1	1	1			
470					15								
500	-2	-2	-2	-1	1	1	1	1	1				
515				16									
550	-3	-2	1	2	1	1	1	1					
560			18										
600	-3	2	2	2	2	2	2						
607		19											
650	5	2	2	2	2	2							
695	19												
700	2	2	2	2	2								

ADD CORRECTION TO OBTAIN CAS

NOTE

1. MAXIMUM AIRSPEED CORRECTIONS ARE GIVEN AT AIRSPEEDS OTHER THAN EVEN INCREMENTS OF 50 KNOTS.
2. AIRSPEED CORRECTIONS FOR INDICATED MACH NUMBERS ABOVE 1.05 ARE ESTIMATED.

22,162B

Figure A1-6

TF

airspeed correction

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

NON-AIMS AIRPLANES

(INSTALLATION ERROR)

ENGINE: J57-23

IAS KNOTS	INDICATED ALTITUDE — 1000 FEET												
	SL	5	10	15	20	25	30	35	40	45	50	55	
100	5	5	4	4	4	4	4	4	4	3	3	3	
150	6	6	5	5	5	5	5	5	5	4	4	4	SUBSONIC
200	7	7	6	6	6	6	6	6	6	6	6	6	
210												10	
237											11	0	SUPERSONIC
250	8	8	7	7	7	7	7	7	7	7	0		
265										12			
300	9	8	8	8	8	8	8	8	13	0			
333								15	0				
350	9	9	9	9	9	9	9	0					
372							16						
400	10	10	10	10	10	12	0						
412						18							
450	11	11	11	11	19	0							
500	12	12	12	21	0								
541			22	0									
550	13	14	0										
587		23											
600	14	0											
634	25												
650	0												
700													

ADD CORRECTION TO OBTAIN CAS

NOTE

MAXIMUM AIRSPEED CORRECTIONS ARE GIVEN AT AIRSPEEDS OTHER THAN EVEN INCREMENTS OF 50 KNOTS

22,434A

Figure A1-7

F

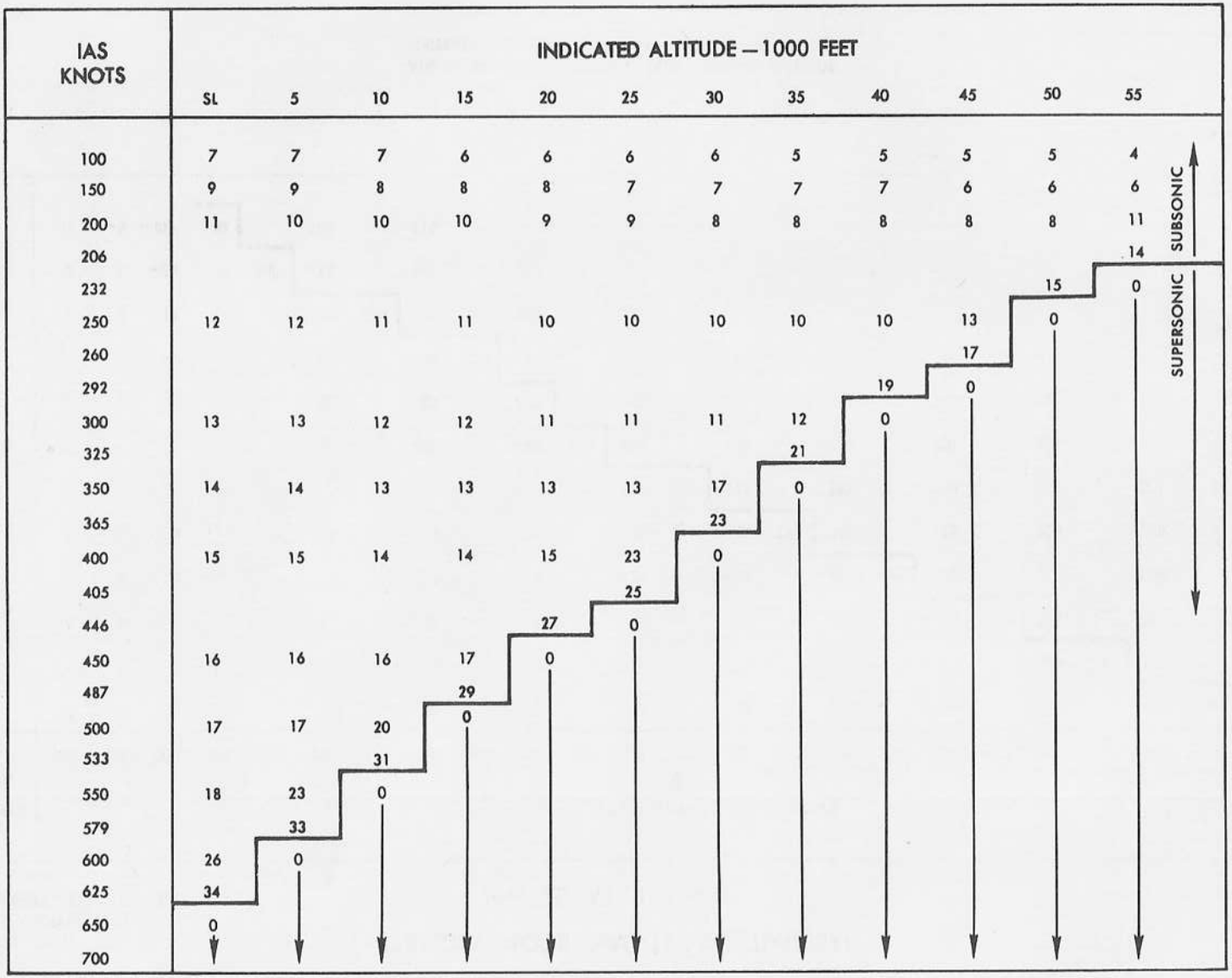
airspeed correction

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

NON-AIMS AIRPLANES

(INSTALLATION ERROR)

ENGINE: J57-23



ADD CORRECTION TO OBTAIN CAS

NOTE MAXIMUM AIRSPEED CORRECTIONS ARE GIVEN AT AIRSPEEDS OTHER THAN EVEN INCREMENTS OF 50 KNOTS

22,435A

Figure A1-8

TF

altitude correction

MODEL: F-102A
DATE: 5 AUGUST 1971
DATA BASIS: FLIGHT TEST

STANDBY MODE (AAU-19/A ALTIMETER)

(INSTALLATION ERROR)

ENGINE: J57-23

TRUE OR INDICATED ALTITUDE FEET	INDICATED AIRSPEED - KNOTS																								
	100	150	200	221	248	250	279	300	313	350	390	400	432	450	475	500	520	550	566	600	614	650	662	700	
SL	0	0	5			10		10		10		5		0		-15		-50		-114		-60	850	190	SUBSONIC
5,000	0	0	5			10		10		10		5		-15		-45		-150		-135	820	175		210	
10,000	0	0	5			10		10		5		-10		-35		-90		-150	790	165		205		240	
15,000	0	0	5			5		5		0		-25		-75		-150	765	160		200		235		270	SUPERSONIC
20,000	0	0	5			5		5		-15		-60		-145	735	145		195		225		255		300	
25,000	0	0	5			5		-5		-40		-120	705	115		180		220		255		300			
30,000	0	0	5			0		-30		-95	675	175		170		250		250		300					
35,000	0	0	5			-10		-60		645		165		200		240		290							
40,000	0	0	0			-40		-130	640	155		195		240											
45,000	0	0	-10			-90	640	145		185		230													
50,000	0	0	-40		640	475		175		225															
55,000	0	-5	-100	640		160		215																	

ADD CORRECTION TO OBTAIN TRUE PRESSURE ALTITUDE
SUBTRACT CORRECTION TO OBTAIN INDICATED ALTITUDE

NOTE

1. MAXIMUM ALTITUDE CORRECTIONS ARE GIVEN AT AIRSPEEDS OTHER THAN EVEN INCREMENTS OF 50 KNOTS.
2. ALTITUDE CORRECTIONS FOR INDICATED MACH NUMBERS ABOVE 1.05 ARE ESTIMATED.

22,003G

Figure A1-9

altitude correction

STANDBY MODE (AAU-19/A ALTIMETER)

(INSTALLATION ERROR)

MODEL: TF-102A
DATE: 5 AUGUST 1971
DATA BASIS: FLIGHT TEST

ENGINE: J57-23

TRUE OR INDICATED ALTITUDE FEET	INDICATED AIRSPEED – KNOTS																									
	100	150	200	218	245	250	275	300	309	346	350	385	400	427	450	470	500	515	550	560	600	607	650	655	700	SUBSONIC
SL	-5	-15	-20			-25		-40			-55		-75		-100		-135		-170		-200		700	1860	180	
5,000	-10	-15	-20			-30		-45			-60		-90		-120		-160		-190		210	1795	170		215	
10,000	-10	-15	-25			-35		-50			-75		-105		-150		-180		85	1730	165		210		240	SUPERSONIC
15,000	-10	-20	-30			-40		-65			-90		-135		-170		-100	1665	160		205		235		265	
20,000	-15	-20	-35			-50		-80			-115		-160		-145	1600	150		190		225		255			
25,000	-15	-25	-40			-65		-100			-140		-160	1540	310		185		225		250					
30,000	-20	-30	-50			-80		-120			-160	1475	700		175		210		250							
35,000	-20	-35	-60			-100		-145		1410	1210		215		200		240									
40,000	-20	-40	-70			-125		-70	1410		150		295		240											
45,000	-20	-50	-95			-150	1410	140			180		230													
50,000	-25	-65	-125		1410	1010		180			225															
55,000	-30	-90	-150	1410		160		210																		

ADD CORRECTION TO OBTAIN TRUE PRESSURE ALTITUDE
SUBTRACT CORRECTION TO OBTAIN INDICATED ALTITUDE

NOTE

1. MAXIMUM ALTITUDE CORRECTIONS ARE GIVEN AT AIRSPEEDS OTHER THAN EVEN INCREMENTS OF 50 KNOTS.
2. ALTITUDE CORRECTIONS FOR INDICATED MACH NUMBERS ABOVE 0.98 ARE ESTIMATED.

22,163D

Figure A1-10

altitude correction

STANDBY MODE (AAU-19/A ALTIMETER)

(INSTALLATION ERROR)

MODEL: F-102A
DATE: 5 AUGUST 1971
DATA BASIS: FLIGHT TEST

ENGINE: J57-23

TRUE OR INDICATED ALTITUDE FEET	INDICATED MACH NUMBER															
	.20	.40	.60	.80	.85	.90	.95	.975	1.00	1.03	1.05	1.10	1.20	1.30	1.40	1.50
SL	0	5	5	-35	-65	-105	-165	-145	850	80	175	210				
5,000	0	5	5	-35	-60	-105	-160	-140	820	80	165	200	235			
10,000	0	5	5	-30	-60	-100	-155	-135	790	75	160	195	225	270		
15,000	0	5	5	-30	-55	-95	-145	-130	765	70	155	185	215	260	290	
20,000	0	5	5	-30	-55	-90	-140	-125	735	70	150	180	210	250	280	330
25,000	0	5	5	-30	-55	-90	-135	-120	705	65	145	170	200	240	270	320
30,000	0	5	5	-30	-50	-85	-130	-115	675	65	140	165	190	230	260	305
35,000	0	5	5	-25	-50	-80	-125	-110	640	60	130	160	185	220	245	290
40,000	0	5	5	-25	-50	-80	-125	-110	640	60	130	155	180	215	245	290
45,000	0	5	5	-25	-50	-80	-125	-110	640	60	130	155	180	215	245	290
50,000	0	5	5	-25	-50	-80	-125	-110	640	60	130	155	180	215	245	290
55,000	0	5	5	-25	-50	-80	-125	-110	640	60	130	155	180	215	245	290

ADD CORRECTION TO OBTAIN TRUE PRESSURE ALTITUDE
SUBTRACT CORRECTION TO OBTAIN INDICATED ALTITUDE

NOTE
ALTITUDE CORRECTIONS FOR MACH NUMBERS ABOVE 1.05 ARE ESTIMATED

22,004G

Figure A1-11

F

altitude correction

MODEL: TF-102A
DATE: 5 AUGUST 1971
DATA BASIS: FLIGHT TEST

STANDBY MODE (AAU-19/A ALTIMETER)

(INSTALLATION ERROR)

ENGINE: J57-23

TRUE OR INDICATED ALTITUDE FEET	INDICATED MACH NUMBER													
	.20	.40	.60	.80	.85	.90	.95	.97	.98	.99	1.05	1.10	1.20	1.30
SL	-5	-30	-70	-155	-180	-195	-180	0	285	1860	175	210		
5,000	-5	-30	-70	-150	-175	-190	-170	0	275	1795	165	200	235	
10,000	-5	-25	-65	-145	-170	-185	-165	0	265	1730	160	195	225	270
15,000	-5	-25	-65	-140	-165	-175	-160	0	255	1665	155	185	215	260
20,000	-5	-25	-60	-135	-155	-170	-155	0	245	1600	150	180	210	250
25,000	-5	-25	-60	-130	-150	-165	-150	0	235	1540	145	170	200	240
30,000	-5	-25	-55	-125	-145	-155	-140	0	225	1475	140	165	190	230
35,000	-5	-20	-55	-120	-135	-150	-135	0	215	1410	130	160	185	220
40,000	-5	-20	-55	-120	-135	-150	-135	0	215	1410	130	155	180	215
45,000	-5	-20	-55	-120	-135	-150	-135	0	215	1410	130	155	180	215
50,000	-5	-20	-55	-120	-135	-150	-135	0	215	1410	130	155	180	215
55,000	-5	-20	-55	-120	-135	-150	-135	0	215	1410	130	155	180	215

ADD CORRECTION TO OBTAIN TRUE PRESSURE ALTITUDE
SUBTRACT CORRECTION TO OBTAIN INDICATED ALTITUDE

NOTE

ALTITUDE CORRECTIONS FOR MACH NUMBERS ABOVE 0.98 ARE ESTIMATED.

22.164B

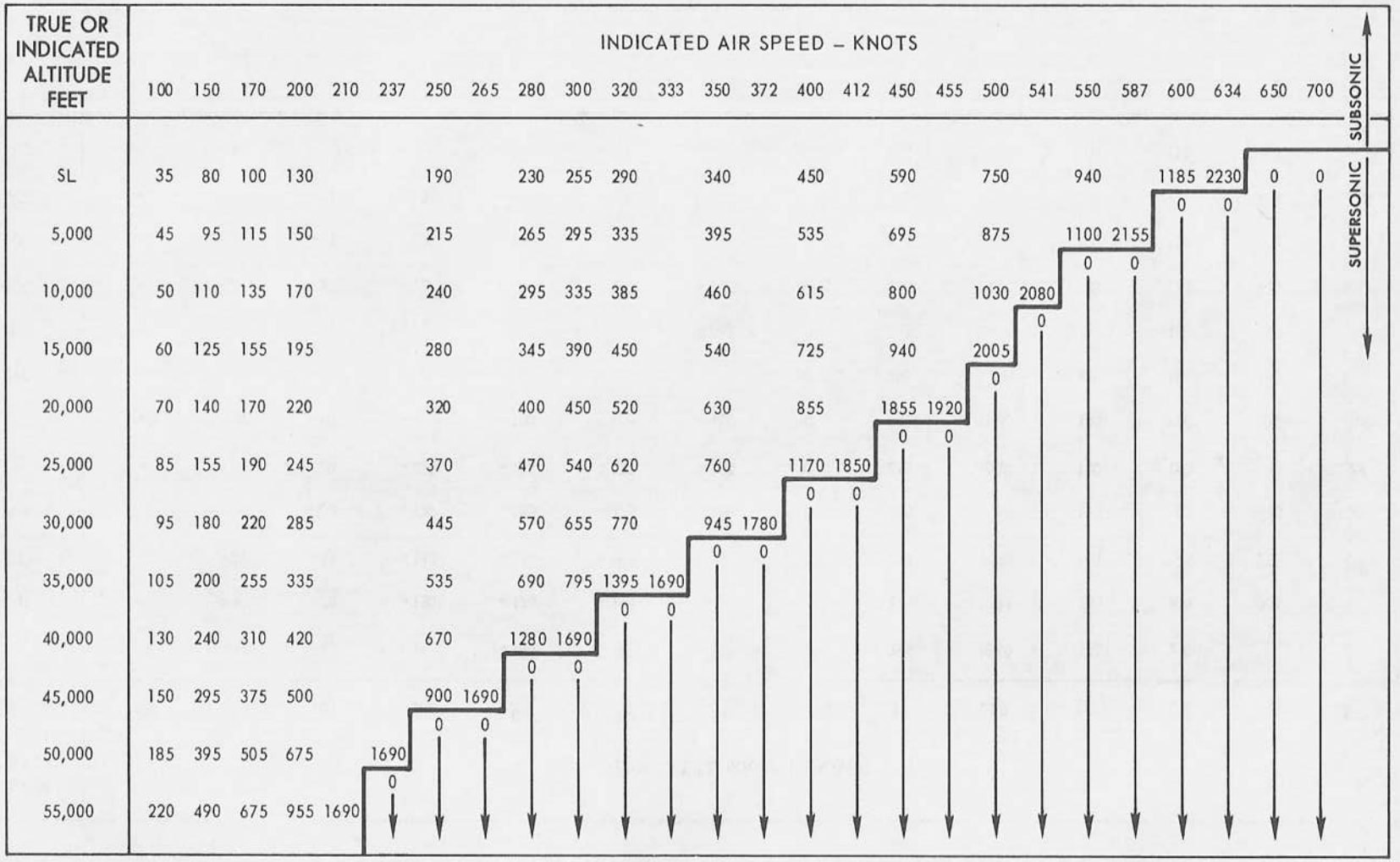
Figure A1-12

altitude correction–non-aims

(INSTALLATION ERROR)

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

ENGINE: J57-23



ADD CORRECTION TO OBTAIN TRUE PRESSURE ALTITUDE
SUBTRACT CORRECTION TO OBTAIN INDICATED ALTITUDE

NOTE MAXIMUM ALTITUDE CORRECTIONS ARE GIVEN AT AIRSPEEDS OTHER THAN EVEN INCREMENTS OF 50 KNOTS

22.436

Figure A1-13

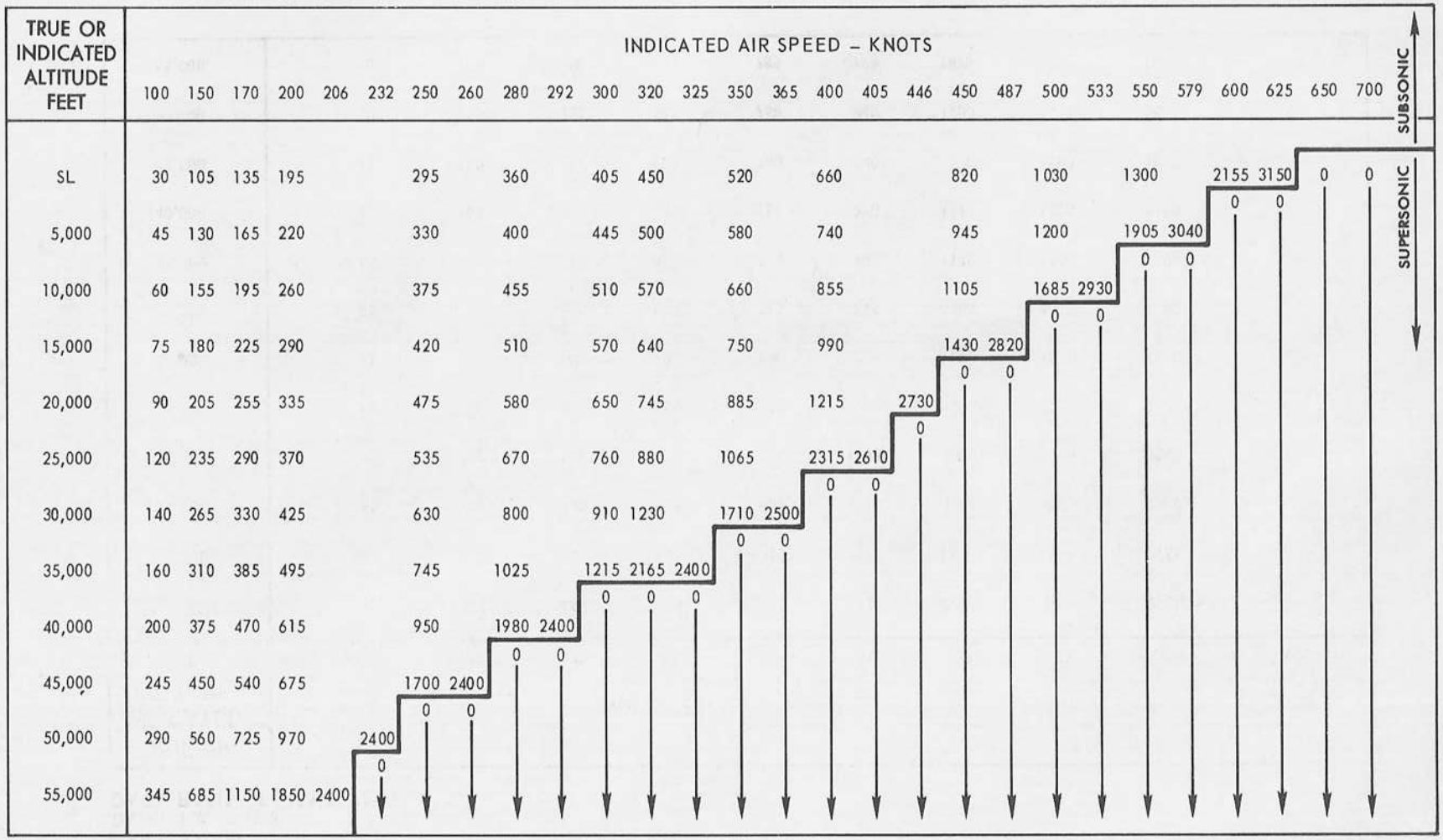
F

altitude correction–non-aims

(INSTALLATION ERROR)

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

ENGINE: J57-23



ADD CORRECTION TO OBTAIN TRUE PRESSURE ALTITUDE
SUBTRACT CORRECTION TO OBTAIN INDICATED ALTITUDE

NOTE MAXIMUM ALTITUDE CORRECTIONS ARE GIVEN AT AIRSPEEDS OTHER THAN EVEN INCREMENTS OF 50 KNOTS

Figure A1-14

22,437

TF

altitude correction–non-aims

(INSTALLATION ERROR)

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

ENGINE: J57-23

TRUE OR INDICATED ALTITUDE - FT	INDICATED MACH NUMBER									
	.20	.40	.60	.80	.85	.90	.925	.95	.954	1.026
SL	65	205	440	865	1025	1230	1490	2040	2230	0
5,000	65	202	440	845	980	1180	1400	1940	2155	0
10,000	65	199	420	815	950	1140	1355	1880	2080	0
15,000	65	197	405	790	920	1105	1300	1790	2005	0
20,000	65	187	385	760	890	1080	1270	1720	1920	0
25,000	65	179	370	725	850	1035	1220	1670	1850	0
30,000	65	173	355	695	815	995	1200	1610	1780	0
35,000	65	164	335	655	775	940	1170	1590	1690	0
40,000	65	164	325	645	765	940	1170	1590	1690	0
45,000	65	160	315	625	740	940	1170	1590	1690	0
50,000	30	155	350	645	750	940	1200	1610	1690	0
55,000	0	129	350	670	785	995	1670	1680	1690	0

ADD CORRECTION TO OBTAIN TRUE PRESSURE ALTITUDE
SUBTRACT CORRECTION TO OBTAIN INDICATED ALTITUDE

Figure A1-15

altitude correction—non-aims

(INSTALLATION ERROR)

MODEL: TF-102A
DATE: 1 NOVEMBER 1959

DATA BASIS: FLIGHT TEST

ENGINE: J57-23

TRUE OR INDICATED ALTITUDE — FEET	INDICATED MACH NUMBER										
	.20	.30	.40	.50	.60	.70	.80	.85	.90	.944	1.04
SL	150	250	360	500	690	915	1200	1420	2000	3150	0
5,000	145	240	345	480	655	875	1155	1370	1875	3040	
10,000	140	230	330	460	620	830	1110	1320	1750	2930	
15,000	130	215	310	435	595	795	1070	1265	1675	2820	
20,000	125	200	290	410	565	755	1030	1210	1600	2730	
25,000	115	185	270	385	540	725	985	1155	1545	2610	
30,000	100	170	250	360	510	690	940	1100	1490	2500	
35,000	85	150	230	335	480	670	905	1065	1380	2400	
40,000	↓	↓	↓	↓	↓	↓	↓	↓	↓	↓	↓
45,000											
50,000											
55,000											

ADD CORRECTION TO OBTAIN TRUE PRESSURE ALTITUDE
SUBTRACT CORRECTION TO OBTAIN INDICATED ALTITUDE

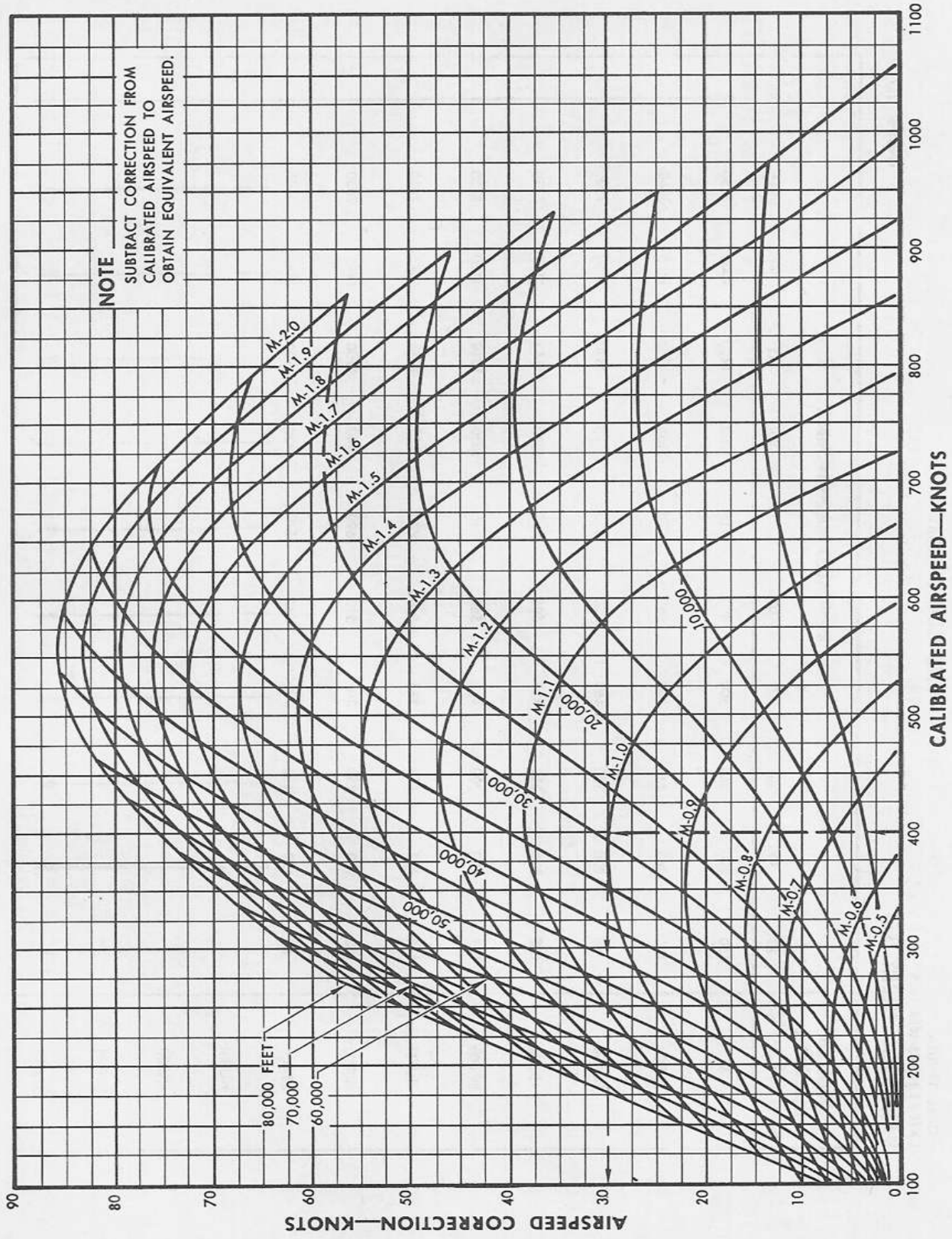
Figure A1-16

22,439

TF

F TF

compressibility correction to calibrated airspeed



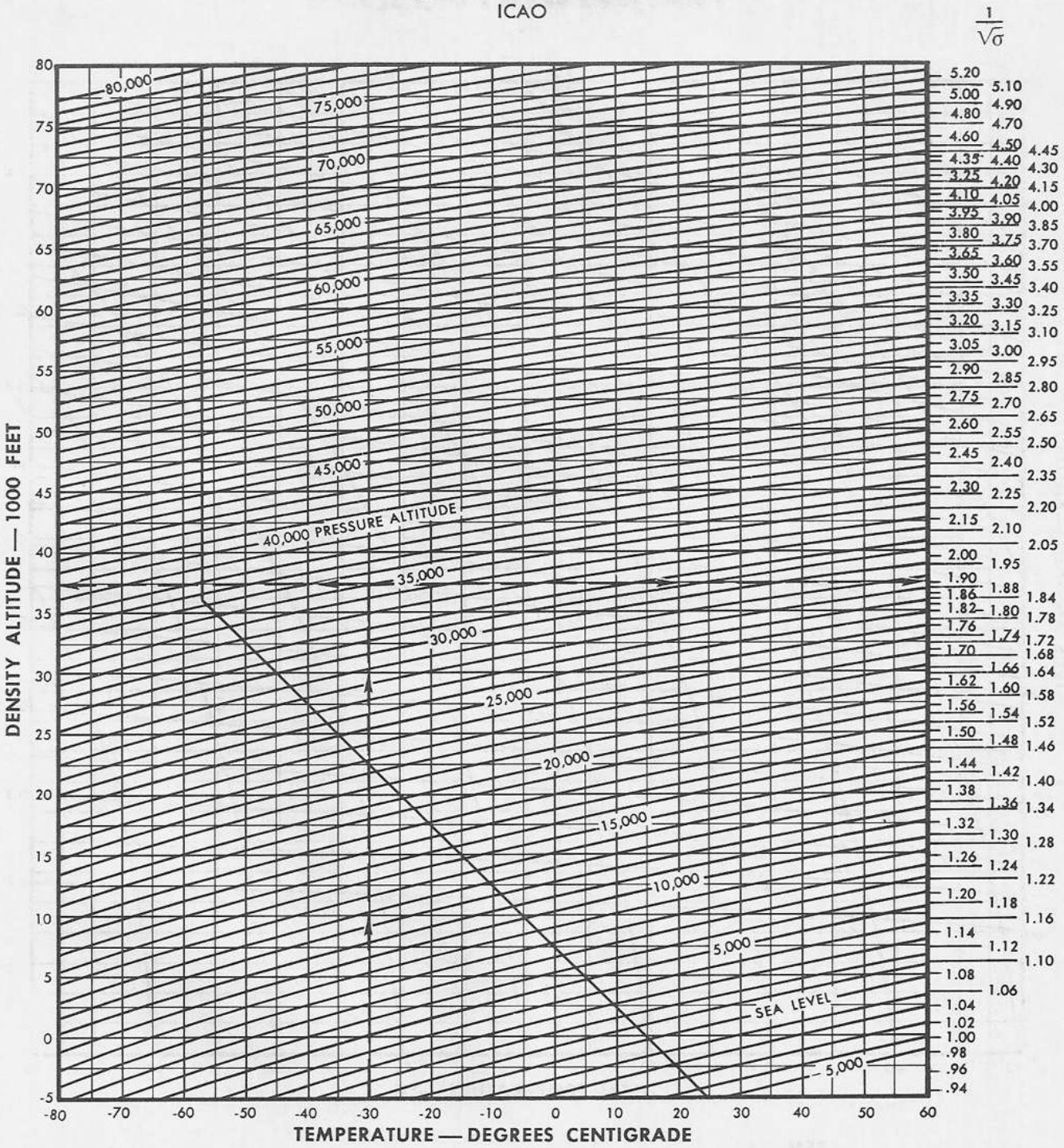
22,005D

Figure A1-17

F TF

density altitude chart

ICAO

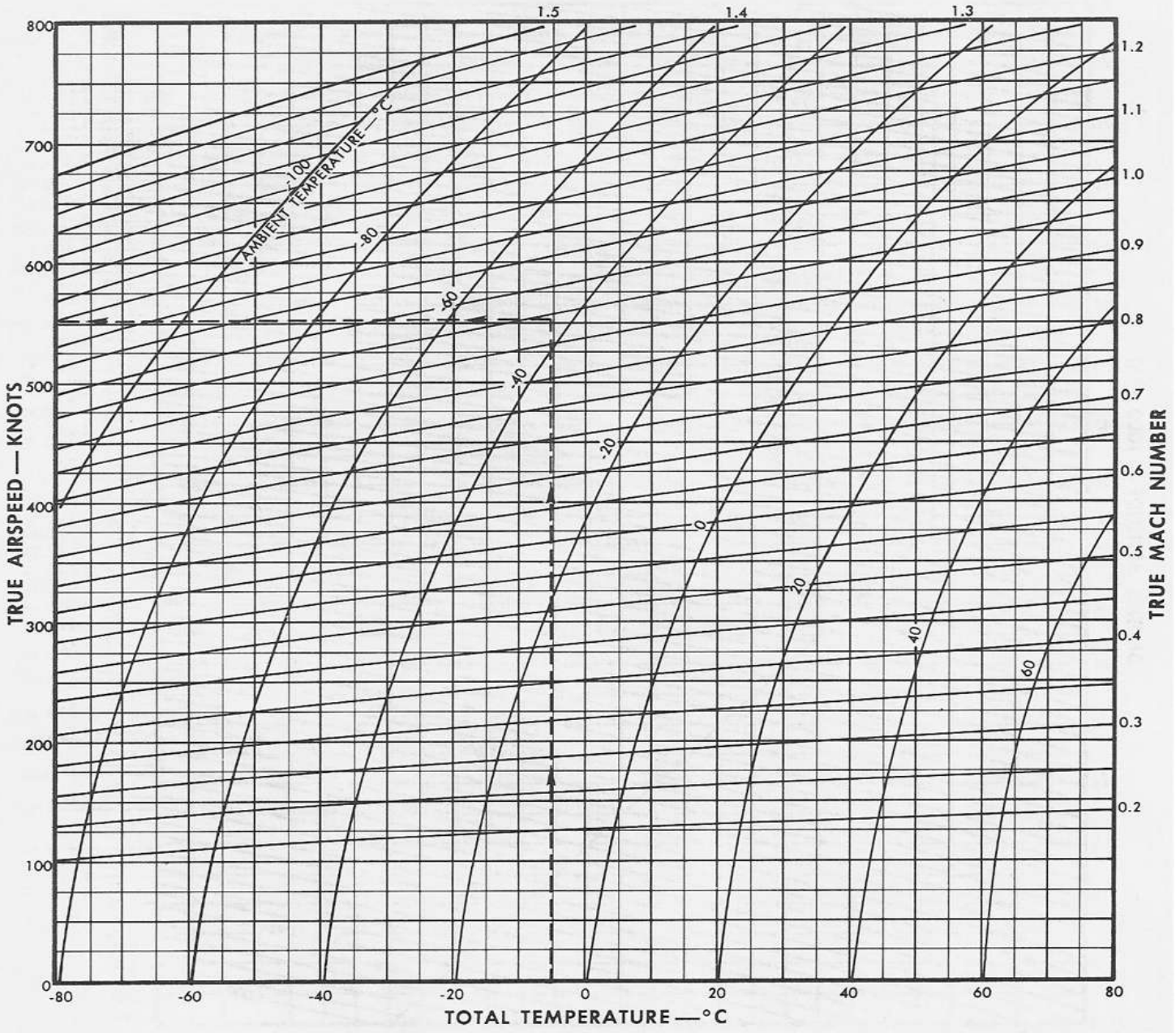


22,006D

Figure A1-18

F TF

true airspeed · mach number temperature chart



REMARKS

(1) TEMPERATURE RECOVERY FACTOR = 0.95

$T_{AMBIENT} = T_{TOTAL} / 1 + .19\ M^2$

REFER TO TEXT FOR EXAMPLE

22,007D

Figure A1-19

F TF

standard atmosphere table

STANDARD SL CONDITIONS:
TEMPERATURE=15°C (59°F)
PRESSURE=29.921 IN. Hg(2116.216 LB/SQ FT)
DENSITY=.0023769 SLUGS/CU FT
SPEED OF SOUND=1116.89 FT/SEC(661.7 KNOTS)

CONVERSION FACTORS:
1 IN. Hg=70.727 LB/SQ FT
1 IN. Hg=0.49116 LB/SQ IN.
1 KNOT=1.151 MPH
1 KNOT=1.688 FT/SEC

ALTITUDE FEET	DENSITY RATIO σ	$\frac{1}{\sqrt{\sigma}}$	TEMPERATURE °C	TEMPERATURE °F	SPEED OF SOUND KNOTS	PRESSURE IN Hg	PRESSURE RATIO δ
SL	1.000	1.0000	15.000	59.000	661.7	29.921	1.0000
1,000	.9711	1.0148	13.019	55.434	659.5	28.856	.9644
2,000	.9428	1.0299	11.038	51.868	657.2	27.821	.9298
3,000	.9151	1.0454	9.056	48.302	654.9	26.817	.8962
4,000	.8881	1.0611	7.076	44.735	652.6	25.842	.8637
5,000	.8617	1.0773	5.094	41.169	650.3	24.896	.8320
6,000	.8359	1.0938	3.113	37.603	648.7	23.978	.8014
7,000	.8106	1.1107	1.132	34.037	645.6	23.088	.7716
8,000	.7860	1.1279	—0.850	30.471	643.3	22.225	.7428
9,000	.7620	1.1456	—2.831	26.905	640.9	21.388	.7148
10,000	.7385	1.1637	—4.812	23.338	638.6	20.577	.6877
11,000	.7155	1.1822	—6.793	19.772	636.2	19.791	.6614
12,000	.6932	1.2011	—8.774	16.206	633.9	19.029	.6360
13,000	.6713	1.2205	—10.756	12.640	631.5	18.292	.6113
14,000	.6500	1.2403	—12.737	9.074	629.0	17.577	.5875
15,000	.6292	1.2606	—14.718	5.508	626.6	16.886	.5643
16,000	.6090	1.2815	—16.699	1.941	624.2	16.216	.5420
17,000	.5892	1.3028	—18.680	—1.625	621.8	15.569	.5203
18,000	.5699	1.3246	—20.662	—5.191	619.4	14.942	.4994
19,000	.5511	1.3470	—22.643	—8.757	617.0	14.336	.4791
20,000	.5328	1.3700	—24.624	—12.323	614.6	13.750	.4595
21,000	.5150	1.3935	—26.605	—15.889	612.1	13.184	.4406
22,000	.4976	1.4176	—28.587	—19.456	609.6	12.636	.4223
23,000	.4806	1.4424	—30.568	—23.022	607.1	12.107	.4046
24,000	.4642	1.4678	—22.549	—26.588	604.6	11.597	.3876
25,000	.4481	1.4938	—34.530	—30.154	602.1	11.103	.3711
26,000	.4325	1.5206	—36.511	—33.720	599.6	10.627	.3552
27,000	.4173	1.5480	—38.492	—37.286	597.1	10.168	.3398
28,000	.4025	1.5762	—40.474	—40.852	594.6	9.725	.3250
29,000	.3881	1.6052	—42.455	—44.419	592.1	9.297	.3107
30,000	.3741	1.6349	—44.436	—47.985	589.5	8.885	.2970
31,000	.3605	1.6654	—46.417	—51.551	586.9	8.488	.2837
32,000	.3473	1.6968	—48.398	—55.117	584.4	8.106	.2709
33,000	.3345	1.7291	—50.379	—58.683	581.8	7.737	.2586
34,000	.3220	1.7623	—52.361	—62.249	579.2	7.382	.2467
35,000	.3099	1.7964	—54.342	—65.816	576.6	7.041	.2353
36,000	.2981	1.8315	—56.323	—69.382	574.0	6.712	.2243
36,089	.2791	1.8347	—56.500	—69.700	573.7	6.683	.2234
37,000	.2843	1.8753	↓	↓	↓	6.397	.2138
38,000	.2710	1.9209				6.097	.2038
39,000	.2583	1.9677				5.811	.1942
40,000	.2462	2.0155	—56.500	—69.700	573.7	5.538	.1851

22,008-1C

Figure A1-20

F TF

standard atmosphere table

STANDARD SL CONDITIONS:
TEMPERATURE=15°C (59°F)
PRESSURE=29.921 IN. Hg 2116.216 LB/SQ FT
DENSITY=.0023769 SLUGS/CU FT
SPEED OF SOUND=1116.89 FT/SEC 661.7 KNOTS

CONVERSION FACTORS:
1 IN. Hg=70.727 LB/SQ FT
1 IN. Hg=0.49116 LB/SQ IN.
1 KNOT=1.151 MPH
1 KNOT=1.688 FT/SEC

ALTITUDE FEET	DENSITY RATIO σ	$\frac{1}{\sqrt{\sigma}}$	TEMPERATURE °C	TEMPERATURE °F	SPEED OF SOUND KNOTS	PRESSURE IN Hg	PRESSURE RATIO δ
41,000	.2346	2.0654	—56.500	—69.700	573.7	5.278	.1764
42,000	.2236	2.1148				5.030	.1681
43,000	.2131	2.1662				4.794	.1602
44,000	.2031	2.2189				4.569	.1527
45,000	.1936	2.2728				4.355	.1455
46,000	.1845	2.3281				4.151	.1387
47,000	.1758	2.3848				3.956	.1322
48,000	.1676	2.4428				3.770	.1260
49,000	.1597	2.5022				3.593	.1201
50,000	.1522	2.5630				3.425	.1145
51,000	.1451	2.6254				3.264	.1091
52,000	.1383	2.6892				3.111	.1040
53,000	.1318	2.7546				2.965	.09909
54,000	.1256	2.8216				2.826	.09444
55,000	.1197	2.8903				2.693	.09001
56,000	.1141	2.9606				2.567	.08578
57,000	.1087	3.0326				2.446	.08176
58,000	.1036	3.1063				2.331	.07792
59,000	.09877	3.1819				2.222	.07426
60,000	.09414	3.2593				2.118	.07078
61,000	.08972	3.3386				2.018	.06746
62,000	.08551	3.4198				1.924	.06429
63,000	.08150	3.5029				1.833	.06127
64,000	.07767	3.5881				1.747	.05840
65,000	.07403	3.6754				1.665	.05566
66,000	.07055	3.7649				1.587	.05305
67,000	.06724	3.8564				1.513	.05056
68,000	.06409	3.9502				1.442	.04819
69,000	.06108	4.0463				1.374	.04592
70,000	.05821	4.1447				1.310	.04377
71,000	.05548	4.2456				1.248	.04171
72,000	.05288	4.3488				1.190	.03976
73,000	.05040	4.4545				1.134	.03789
74,000	.04803	4.5633				1.081	.03611
75,000	.04578	4.6738				1.030	.03442
76,000	.04363	4.7874				0.982	.03280
77,000	.04158	4.9039				0.935	.03126
78,000	.03963	5.0231				0.892	.02980
79,000	.03777	5.1454				0.850	.02840
80,000	.03600	5.2706	—56.500	—69.700	573.7	0.810	.02707

22,008-2C

Figure A1-21

Part 2 TAKEOFF

TABLE OF CONTENTS

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Takeoff and Landing Data Card	A2-1
Takeoff Check Chart	A2-5
Takeoff and Landing Crosswind Chart	A2-6
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Maximum Refusal Speed	A2-15

TAKEOFF AND LANDING DATA CARD

Prior to each takeoff compute all data contained on the Takeoff and Landing Data Card as shown below.

Note

If takeoff distance is less than 50% of runway length, computation of takeoff speed, refusal speed, and refusal distance is not required.

F/TF-102A TAKEOFF AND LANDING DATA CARD

CONDITIONS

	TAKEOFF	LANDING
Runway Length	_____	_____
Wind	_____	_____
Outside Air Temp	_____	_____
Pressure Altitude	_____	_____
Fuel Remaining	_____	_____

TAKEOFF

Engine Pressure Ratio		_____
Acceleration Check	_____ Kts at	_____ Ft
Takeoff Distance		_____ Ft
Takeoff Speed		_____ Kts
Refusal Speed & Distance	_____ Kts at	_____ Ft
Initial Climb Speed		_____ Kts

LANDING

	IMMEDIATELY AFTER TAKEOFF	FINAL LANDING
Final Approach Speed	_____	_____
Touchdown Speed	_____	_____
Landing Ground Roll:		
Wheel Brakes Only	_____	_____
Drag Chute Deployed	_____	_____

Completion of both takeoff and landing data is necessary before takeoff, in event of emergency landing after takeoff. "Landing Immediately After Takeoff" data should be based on the takeoff gross weight. Takeoff and Landing Data Card nomenclature definitions are:

1. **Conditions.**
   a. Runway Length. Usable length of runway in feet.
   b. Wind. 100% of the wind component parallel to the runway.
   c. Outside Air Temperature. Runway air temperature in degrees centigrade.
   d. Pressure Altitude. Field pressure altitude obtained by setting altimeter to 29.92 inches Hg and reading altimeter dial.
   e. Gross Weight. Gross weight of airplane in pounds at start of takeoff roll or at start of final approach, whichever is applicable.

2. **Takeoff Data.**
   a. Engine Pressure Ratio. The pressure ratio of pitot pressure and engine turbine discharge pressure. The pressure ratio setting or the maximum and minimum limits are obtained from the Takeoff Check Chart, figure A2-1.
   b. Takeoff Distance. Ground roll to takeoff speed in feet (figures A2-3 through A2-6).
   c. Takeoff Speed. Indicated airspeed in knots at which the airplane leaves the ground (figures A2-7 through A2-10).
   d. Initial Climb Speed. Indicated airspeed in knots at start of climb for maximum rate-of-climb. (See figure A3-1 or A3-2 without external tanks; or figure A3-3 or A3-4 with external tanks.)

3. **Landing Immediately After Takeoff.**
   a. Approach Speed. Recommended indicated airspeed in knots for final approach (figures A7-1 through A7-4).
   b. Touchdown Speed. Indicated airspeed in knots at which the airplane contacts the runway.
   c. Landing Ground Roll:
      (1) Wheel Brakes Only. Distance in feet from touchdown to full stop with no drag chute and speed brakes open, using wheel brakes only (figures A7-1 and A7-2).
      (2) Drag Chute Deployed. Distance in feet from airplane touchdown to full stop with drag chute (inflated at touchdown), speed brakes open, and using wheel brakes (figures A7-3 and A7-4).

TAKEOFF CHECK CHART

Figure A2-1 shows the pressure ratio of pitot pressure and turbine discharge pressure. The minimum and maximum limits represent the range of pressure ratio that must be obtained for takeoff. The target limit represents that pressure to be set in the takeoff window of the pressure ratio gage. When this limit is set in the takeoff

window, the extreme ends of the reference mark will indicate the minimum and maximum limits. When the throttle is advanced to FULL MIL POWER, the pressure ratio gage should indicate within the extreme ends of the reference mark, with the exhaust gas temperature and rpm remaining within limits.

Use

Engine pressure ratio is determined by entering the ambient temperature scale for existing temperature condition and proceeding vertically to the minimum target, and maximum takeoff limits. From this point, read to the left for the pressure ratio setting.

TAKEOFF DISTANCE

Maximum thrust takeoff distances are shown in figures A2-3 and A2-4 and military thrust takeoff distances are in figures A2-5 and A2-6. The takeoff distance charts show both ground run distance and total distance over a 50-foot obstacle for takeoff from a hard-surface runway. Various conditions of gross weight (fuel remaining), ambient air temperature, pressure altitude, and wind are shown. These data are valid with or without external fuel tanks within the weight range shown.

Note

Empty gross weight is based on empty weight plus the weight of the external tanks. This basis gives the clean airplane slightly larger, but insignificant, ground roll values.

Use

Enter chart with ambient air temperature and read-up to pressure altitude. Read horizontally to the right to fuel remaining and then down to the ground roll distance with zero wind. Correct ground roll for wind and proceed down to appropriate base lines and read to the left for the total distance to clear 50-foot obstacle.

Sample Problem

Find the maximum thrust takeoff distance for an airplane with 6000 pounds of fuel, a temperature of 35°C and a 40-knot headwind:

A. Enter figure A2-3 with the ambient temperature—35°C.

B. Read up to the pressure altitude—4000 feet.

C. Read horizontally to 6000 pounds fuel remaining.

D. Proceed down to the zero wind base line.

E. Correct ground run for headwind by following parallel to a wind guide line to 40-knot headwind. Read ground roll distance for 40-knot headwind—2000 feet.

F. Read down to baseline for total distance to clear a 50-foot obstacle and 40-knot headwind.

G. Read horizontally to the left for total distance to clear a 50-foot obstacle with 40-knot headwind—3220 feet.

TAKEOFF SPEEDS

Maximum thrust takeoff speeds are shown in figures A2-7 and A2-8 and military thrust takeoff speeds are shown in figures A2-9 and A2-10. These charts provide the normal takeoff speeds for various gross weights, ambient air temperatures, and pressure altitudes. Tire limit speeds are also provided on the charts.

Use

Enter chart with ambient air temperature and read up to pressure altitude. Read horizontally to the right to total fuel remaining, then down to the takeoff speed.

Sample Problem

Find the takeoff speed for a maximum thrust takeoff at 4000 feet field elevation with 6000 pounds of fuel. Ambient temperature is 24.0°C.

A. Enter figure A2-7 at ambient temperature—24.0°C.

B. Read up to pressure altitude—4000 feet.

C. Read across to 6000 pounds fuel remaining.

D. Read down to a takeoff speed of 142 KCAS.

Note

The takeoff speed is less than tire limit speed. Do not take off when takeoff speed is higher than tire limit speed.

TAKEOFF AND LANDING CROSSWIND CHART

The Takeoff and Landing Crosswind Chart (figure A2-2) is used to resolve a surface wind of known velocity and direction into its headwind and crosswind components. The chart's green, yellow and light red areas are used to emphasize areas of varying degrees of risk. The green area includes crosswind components which present no particular problem regarding airplane controllability or landing gear strength. The yellow area covers takeoffs and landings of progressively increased risk. The light red area covers the crosswind components where takeoffs and landings are not recommended.

CAUTION

Due to the increased side loads on the landing gear in increased crosswinds, landings in the yellow area should be executed with extreme caution. Sink rate should be minimized and all crab should be removed prior to touchdown.

Use

1. To resolve a surface wind of known velocity and direction into its headwind and cross-wind components, enter the chart on the wind angle line and follow it to the wind velocity circle using maximum gust velocity for crosswind component computation (if gusts are present) and steady wind velocity for headwind component computation. Then proceed horizontally to the left scale and read the velocity of the headwind component; then vertically downward and read the velocity of the crosswind component.
2. When the crosswind component falls below 25 knots, use the recommended touchdown speeds (corrected for gusts) (figures A7-1 through A7-4) or the recommended takeoff speeds (figures A2-7 through A2-10).
3. If the crosswind component is above 25 knots, takeoff and landing is not recommended.

Sample Problem

Given: A steady wind velocity of 21 knots, gusts to 27 knots, 65° relative to runway heading.

Determine: The headwind and crosswind components.

A. Enter figure A2-2 on the wind angle line—65°.
B. Proceed to the steady wind velocity circle—21 knots.
C. Move horizontally to the left and read the headwind component—8.9 knots.
D. Proceed along the wind angle line to the maximum gust velocity circle—27 knots.
E. Move vertically downward and read the crosswind component—24.4 knots.

Crosswind Takeoff

The minimum nose gear liftoff speed for a given crosswind condition is determined from the Takeoff and Landing Crosswind Chart (figure A2-2).

Use

The use of the chart in figure A2-2 is explained in the section titled "Takeoff and Landing Crosswind Chart."

Sample Problem

Given: A wind of 21 knots with gusts to 27 knots, 65° relative to runway heading.

Determine: The headwind and crosswind components and the minimum nose wheel liftoff speed with 6000 pounds of fuel.

A. Enter figure A2-2 on the wind angle line—65°.
B. Proceed to the steady wind velocity circle—21 knots.
C. Move horizontally to the left and read the headwind component 8.9 knots (used to obtain takeoff distance from takeoff charts).
D. Proceed along the wind angle line to the maximum gust velocity circle—27 knots.
E. Move vertically downward to the crosswind component—24.4 knots.

The recommended takeoff speed sea level, 15°C with 6000 pounds of fuel is 142 KCAS (figure A2-7).

MAXIMUM REFUSAL SPEED

Maximum Refusal Speed Charts, figure A2-11 and A2-12, show the highest speed to which the airplane can be accelerated (assuming normal acceleration) and still be stopped within the remaining runway length. The refusal speed charts take into account gross weight, pressure altitude, temperature, available runway length and pilot reaction time. They are presented for a dry, hard surface runway with a correction factor noted for a wet runway. Line speed is used to allow the takeoff to be monitored at marked runway distances to check if acceleration is normal.

Use

To determine the refusal speed, enter the chart at the upper left block with the airplane gross weight and proceed to the right to intersect the pressure altitude line. Read down to the ambient temperature line in the block directly below. Then read to the right along the atmospheric conditions line and intersect the applicable available runway length. At this point read down to the maximum refusal speed. Return to the atmospheric conditions line and continue right to the runway marker line that will be used for an acceleration line check. Read down to the speed that should be attained at this point in the takeoff roll. To determine refusal distance, enter the right block (line check speed) with the refusal speed (not including wind vector) and read up to a point that intersects the continuation of the atmospheric conditions line. At this point interpolate between the acceleration line check distances for maximum refusal distance.

Sample Problem

Find the maximum refusal speed, the line check speed at the 1000-foot runway marker, and the refusal distance for a 7000-foot runway. Temperature is 20°C at 2000 feet field elevation. The airplane gross weight is 26,000 pounds, and the takeoff is to be made with maximum thrust:

A. Enter figure A2-11 with the airplane gross weight—26,000 pounds.

B. Proceed right to intersect the pressure altitude—2000 feet.

C. Read down to the ambient air temperature, 20°C or 68°F.

D. Read right to the available runway length—7000 feet.

E. Read down to the maximum refusal speed—151 KCAS.

F. Return to "D" above, then continue right to intersect the desired line check marker—2000 feet.

G. Read down to the line check speed—133 KCAS.

To determine refusal distance:

H. Enter the right block (line check speed) with refusal speed—151 KCAS.

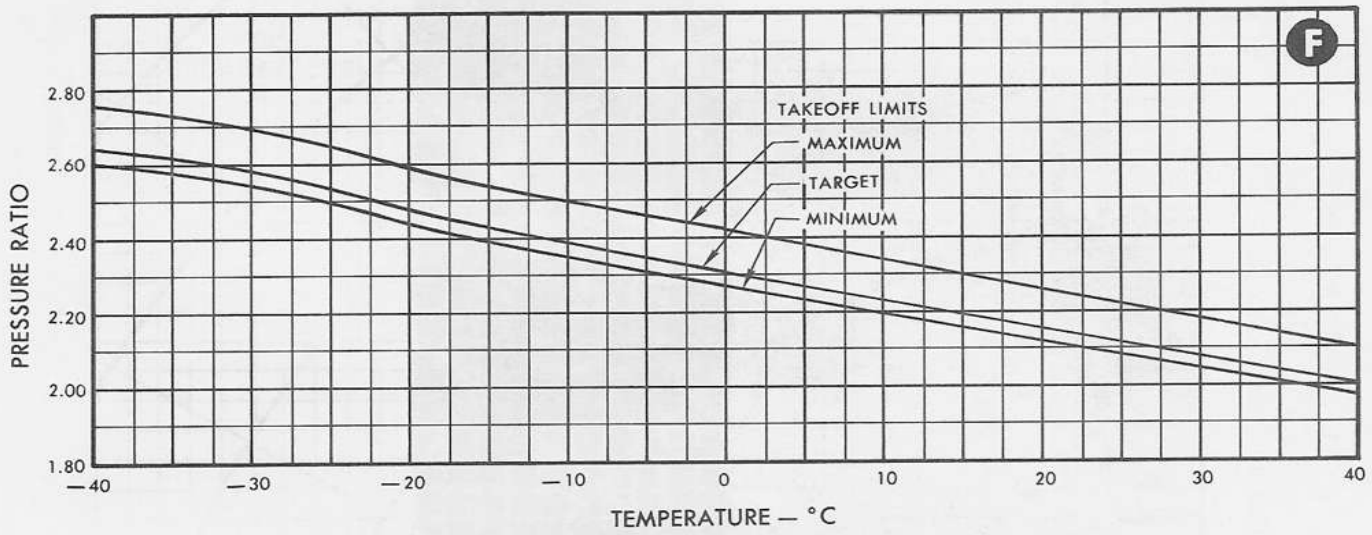
I. Read up to a point that intersects the continuation of the atmospheric conditions line. Interpolate between acceleration line check distances for maximum refusal distance—2700 feet.

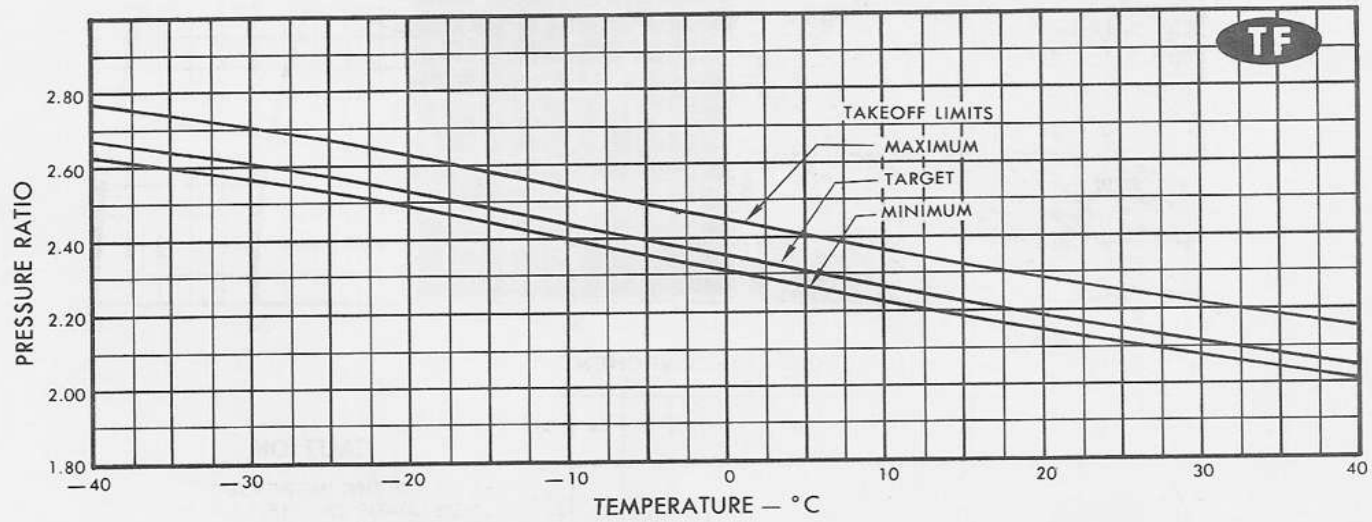
takeoff check chart

MILITARY THRUST • (NON-AFTERBURNING)

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

ENGINE: J57-23
FUEL GRADE: JP-4





22,009F

Figure A2-1

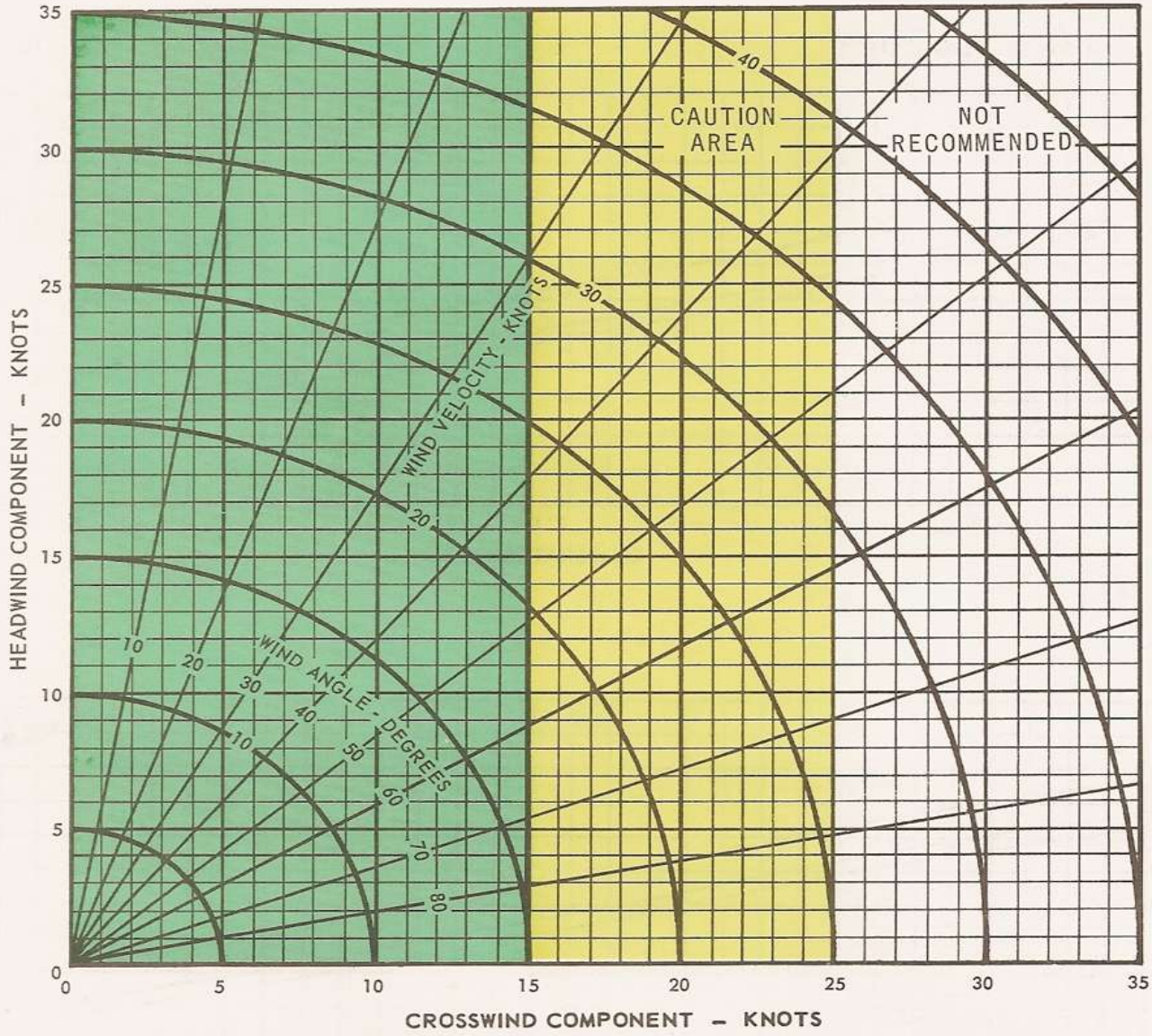
F TF

takeoff and landing crosswind chart

MODEL: F/TF-102A
DATE: 1 OCTOBER 1962
DATA BASIS: ESTIMATED

STANDARD ATMOSPHERE

ENGINE: J57-23



CROSSWIND COMPONENT KNOTS	25	26	27	28	29	30	31	32	33	34	35
MINIMUM TOUCHDOWN OR NOSEWHEEL LIFTOFF SPEED KNOTS (CAS)	136	141	146	152	158	164	170	176	181	188	194

NOTE

ENTER CHART WITH MAXIMUM GUST VELOCITY FOR CROSSWIND COMPONENT COMPUTATION AND STEADY WIND VELOCITY FOR HEADWIND COMPONENT COMPUTATION. REFER TO TAKEOFF SPEEDS CHART FOR TIRE GROUND LIMIT SPEED.

USE SPEEDS FROM TAKEOFF AND LANDING SPEED CHARTS FOR CROSSWIND COMPONENTS BELOW 25 KNOTS.

SUBTRACT 5 KNOTS FOR IAS.

CAUTION

DUE TO THE INCREASED SIDELOADS ON THE LANDING GEAR IN INCREASED CROSSWINDS, LANDINGS WITH A CROSSWIND COMPONENT ABOVE 15 KNOTS SHOULD BE EXECUTED WITH EXTREME CAUTION. SINK RATE SHOULD BE MINIMIZED AND ALL CRAB SHOULD BE REMOVED PRIOR TO TOUCHDOWN.

22,064B

Figure A2-2

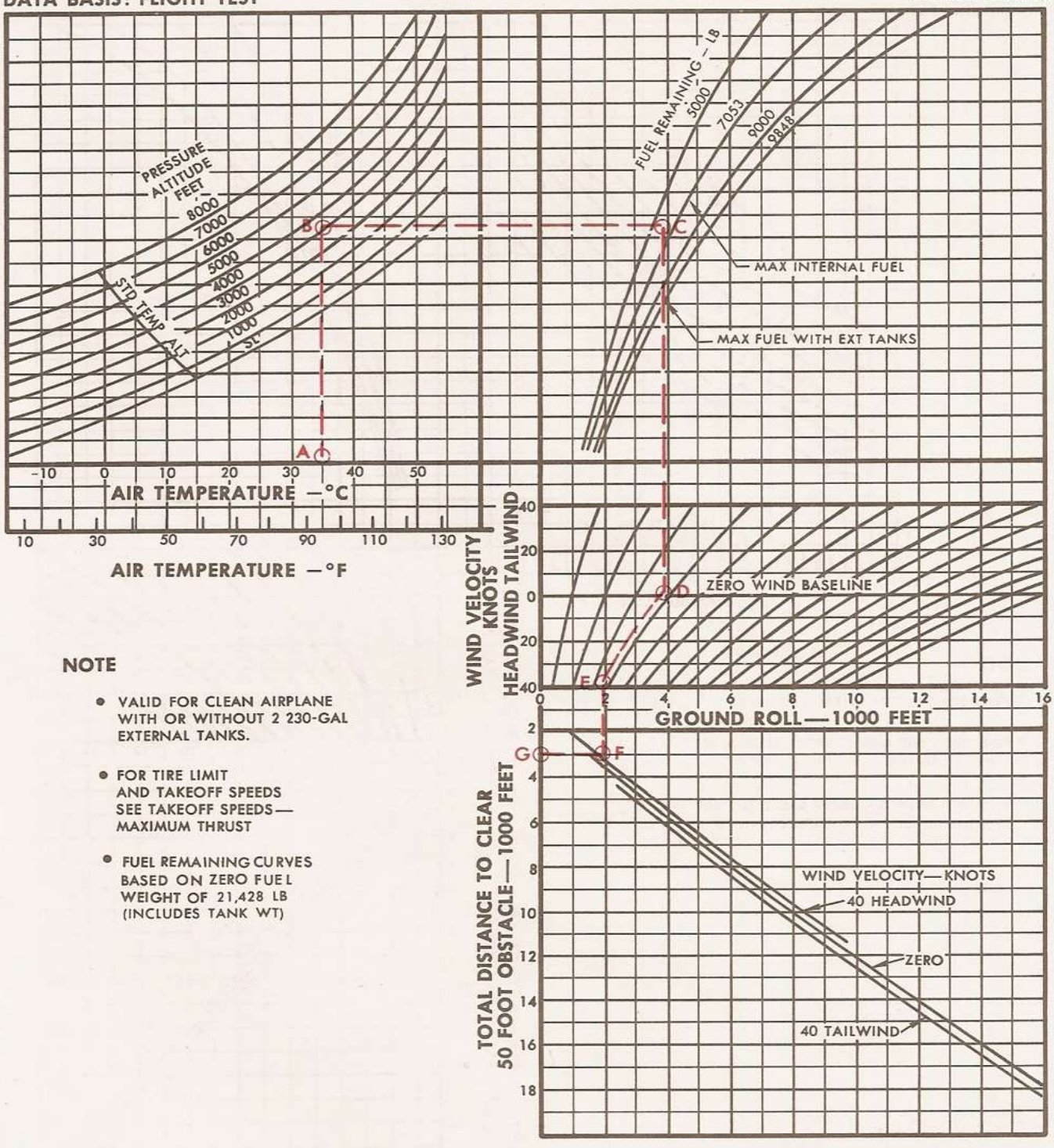
F

takeoff distances — maximum thrust

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

HARD SURFACE RUNWAY

ENGINE: J57-23
ALL FUEL GRADES



NOTE

- VALID FOR CLEAN AIRPLANE WITH OR WITHOUT 2 230-GAL EXTERNAL TANKS.
- FOR TIRE LIMIT AND TAKEOFF SPEEDS SEE TAKEOFF SPEEDS—MAXIMUM THRUST
- FUEL REMAINING CURVES BASED ON ZERO FUEL WEIGHT OF 21,428 LB (INCLUDES TANK WT)

22,010G

Figure A2-3

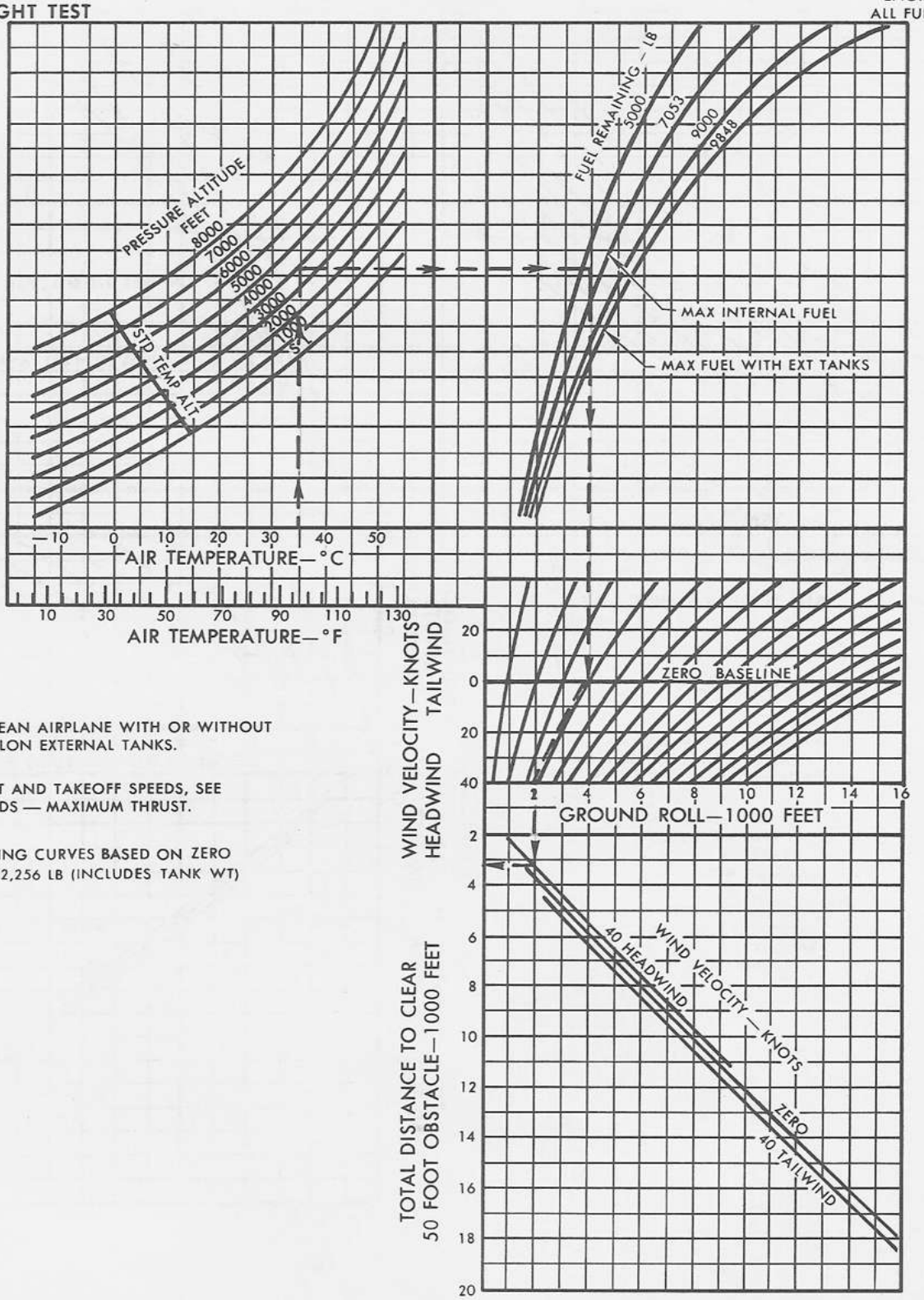
TF

takeoff distances—maximum thrust

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

HARD SURFACE RUNWAY

ENGINE: J57-23
ALL FUEL GRADES



NOTE

- VALID FOR CLEAN AIRPLANE WITH OR WITHOUT TWO 230-GALLON EXTERNAL TANKS.
- FOR TIRE LIMIT AND TAKEOFF SPEEDS, SEE TAKEOFF SPEEDS — MAXIMUM THRUST.
- FUEL REMAINING CURVES BASED ON ZERO WEIGHT OF 22,256 LB (INCLUDES TANK WT)

22,176A

Figure A2-4

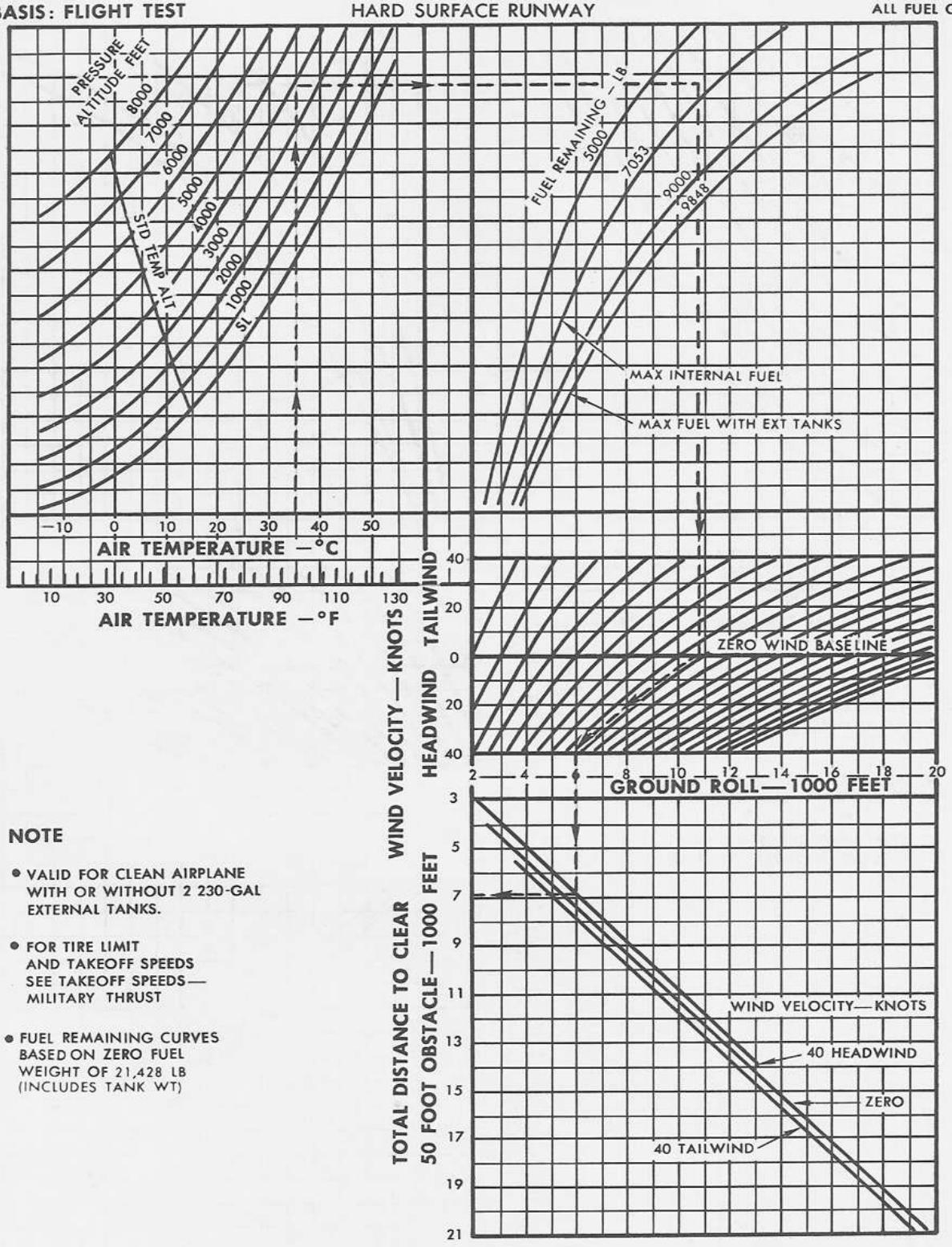
F

takeoff distances
military thrust

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

HARD SURFACE RUNWAY

ENGINE: J57-23
ALL FUEL GRADES



NOTE

- VALID FOR CLEAN AIRPLANE WITH OR WITHOUT 2 230-GAL EXTERNAL TANKS.
- FOR TIRE LIMIT AND TAKEOFF SPEEDS SEE TAKEOFF SPEEDS— MILITARY THRUST
- FUEL REMAINING CURVES BASED ON ZERO FUEL WEIGHT OF 21,428 LB (INCLUDES TANK WT)

22,057F

Figure A2-5

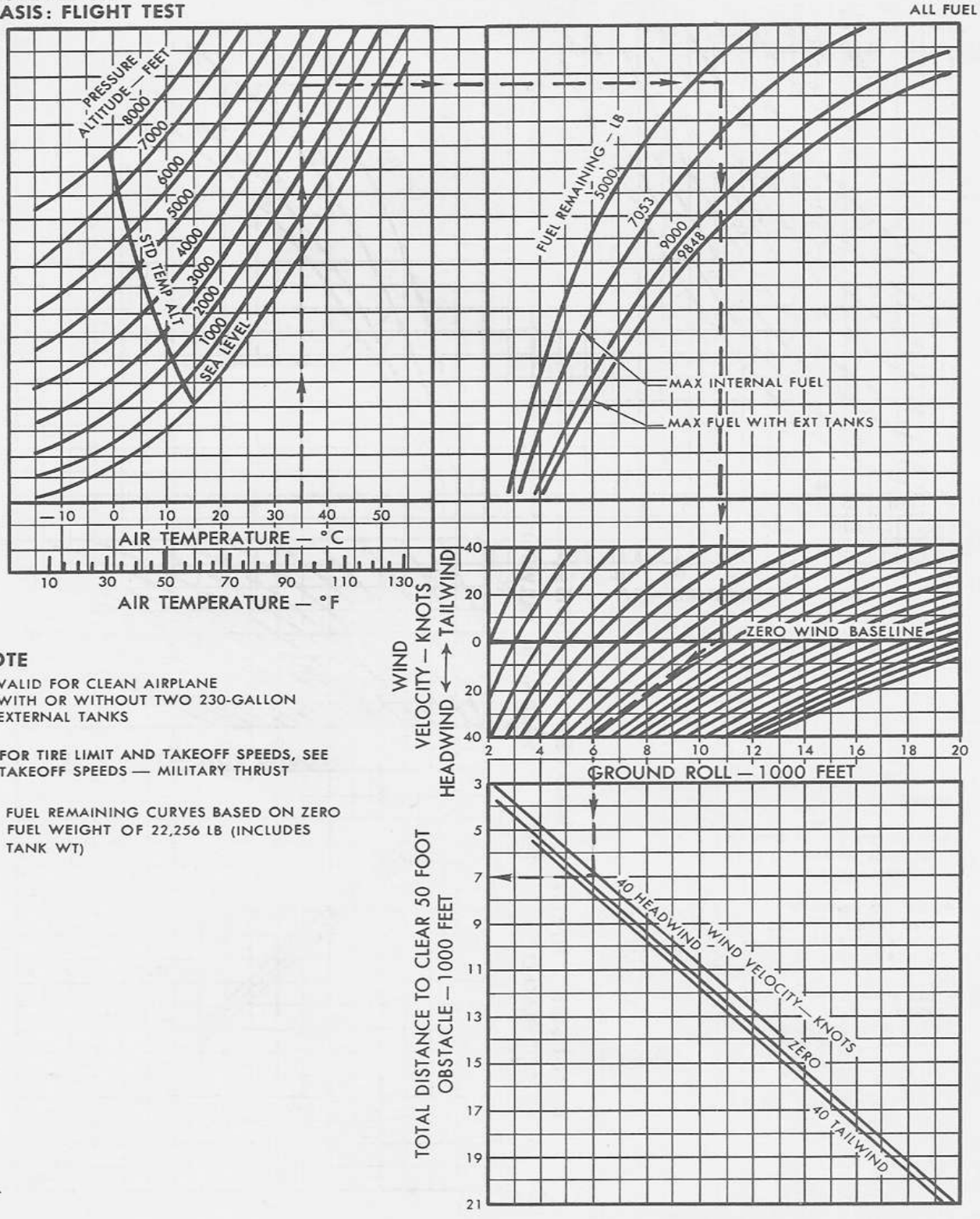
TF

takeoff distances—military thrust

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

HARD SURFACE RUNWAY

ENGINE: J57-23
ALL FUEL GRADES



NOTE

- VALID FOR CLEAN AIRPLANE WITH OR WITHOUT TWO 230-GALLON EXTERNAL TANKS
- FOR TIRE LIMIT AND TAKEOFF SPEEDS, SEE TAKEOFF SPEEDS — MILITARY THRUST
- FUEL REMAINING CURVES BASED ON ZERO FUEL WEIGHT OF 22,256 LB (INCLUDES TANK WT)

22,177A

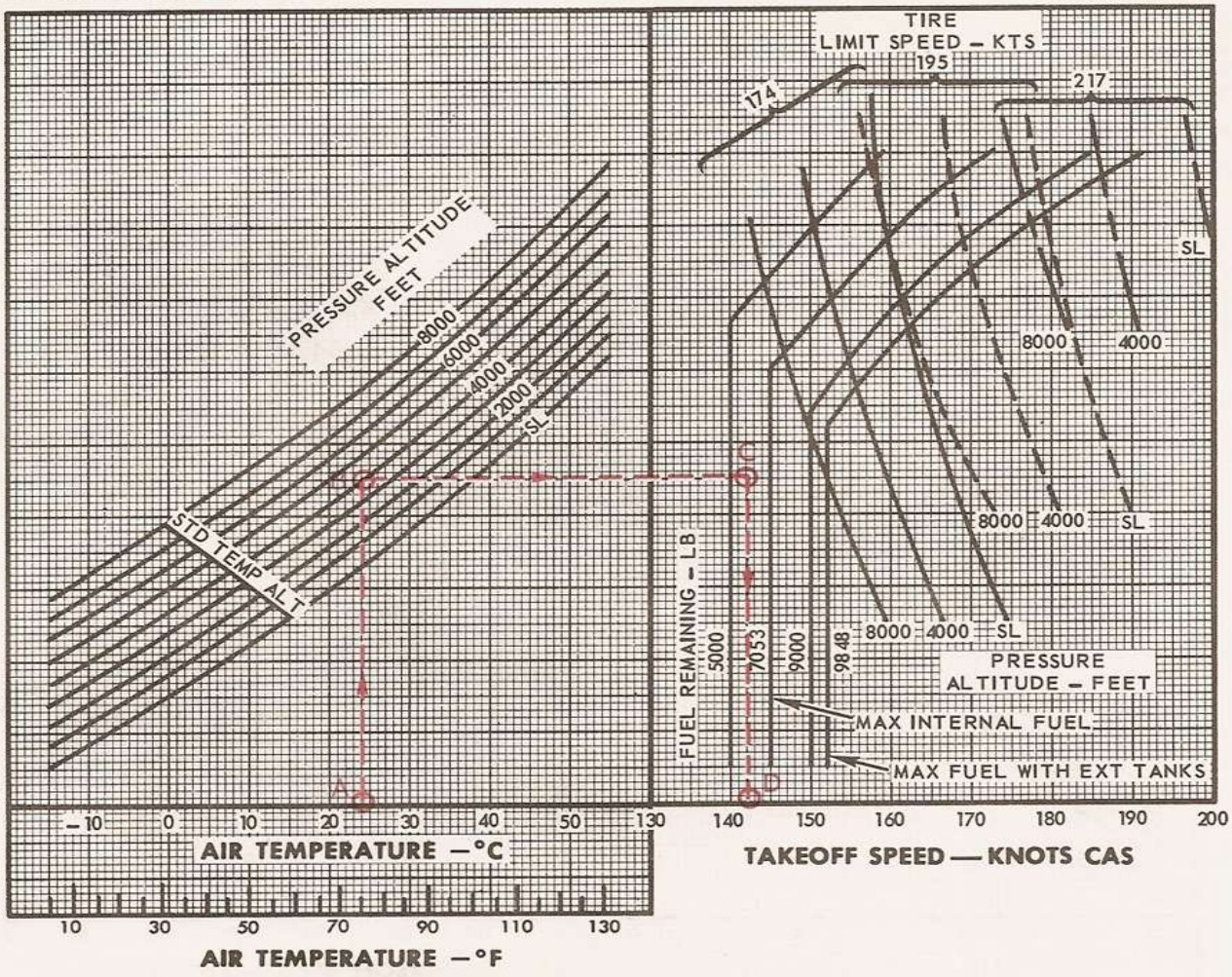
Figure A2-6

F

takeoff speeds - maximum thrust

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

ENGINE: J57-23
ALL FUEL GRADES



NOTE

- TO DETERMINE TAKEOFF SPEED LIMITATION, ADD HEADWIND OR SUBTRACT TAILWIND FROM ZERO WIND TIRE GROUND LIMIT SPEED.
- FUEL REMAINING CURVES BASED ON ZERO FUEL WEIGHT OF 21,428 LB (INCLUDES TANK WT)

22,063E

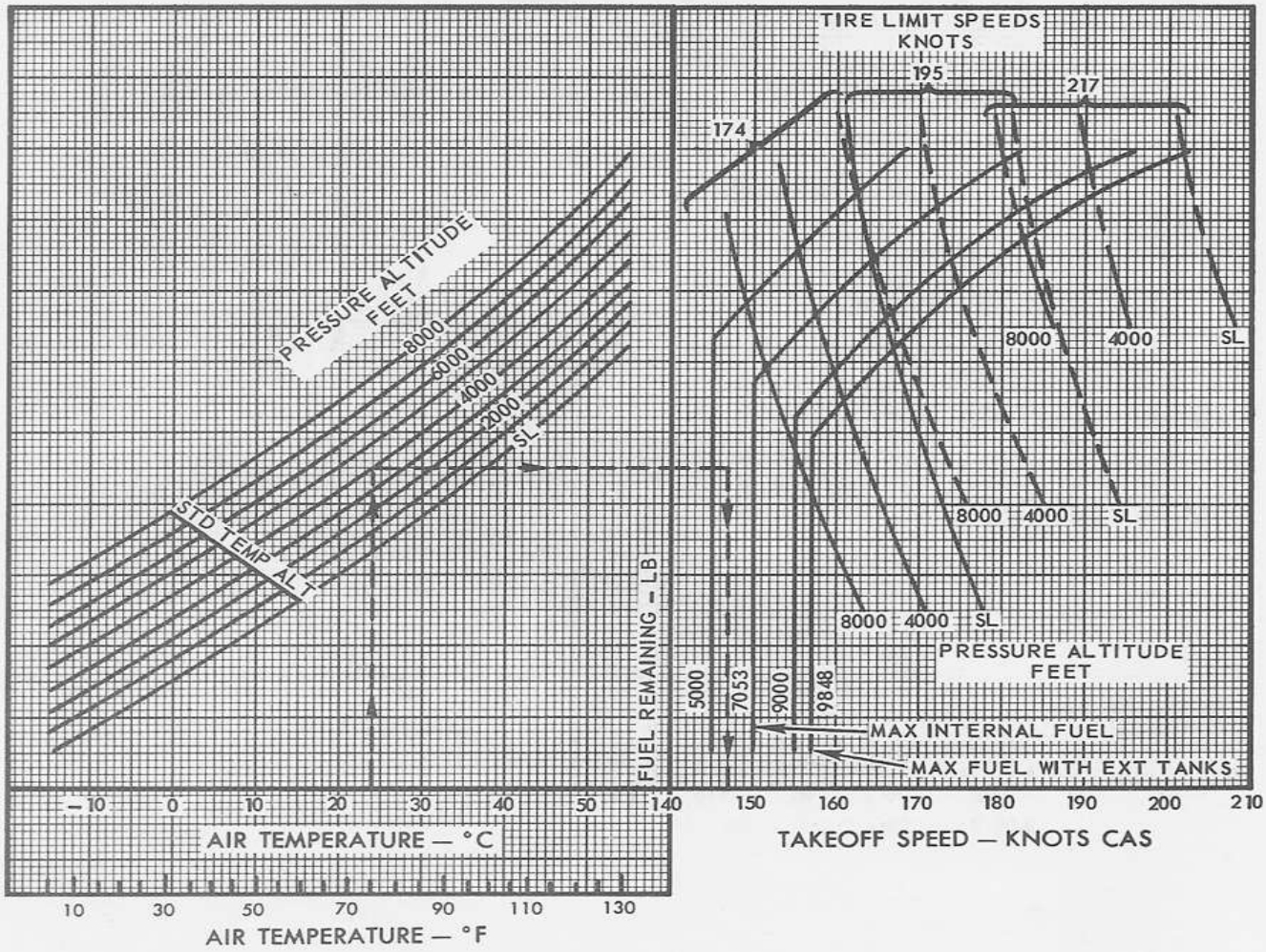
Figure A2-7

TF

takeoff speeds—maximum thrust

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
ALL FUEL GRADES



NOTE

- TO DETERMINE TAKEOFF SPEED LIMITATION, ADD HEADWIND OR SUBTRACT TAILWIND FROM ZERO WIND TIRE GROUND LIMIT SPEED.
- FUEL REMAINING CURVES BASED ON ZERO FUEL WEIGHT OF 22,256 LB (INCLUDES TANK WT)

22,178C

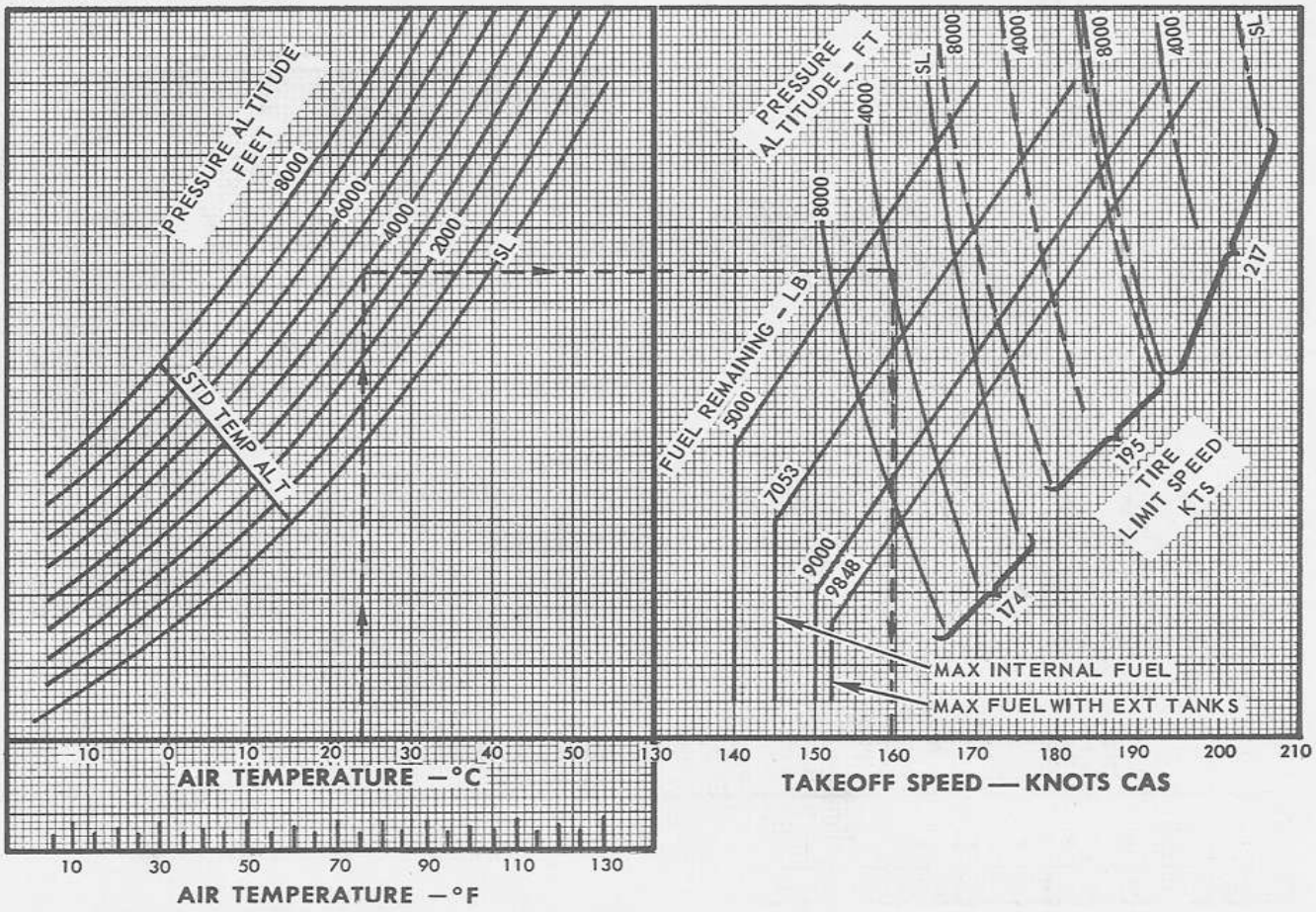
Figure A2-8

F

takeoff speeds – military thrust

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

ENGINE: J57-23
ALL FUEL GRADES



NOTE

- TO DETERMINE TAKEOFF SPEED LIMITATION, ADD HEADWIND OR SUBTRACT TAILWIND FROM ZERO WIND TIRE GROUND LIMIT SPEED.
- FUEL REMAINING CURVES BASED ON ZERO FUEL WEIGHT OF 21,428 LB (INCLUDES TANK WT)

22.062E

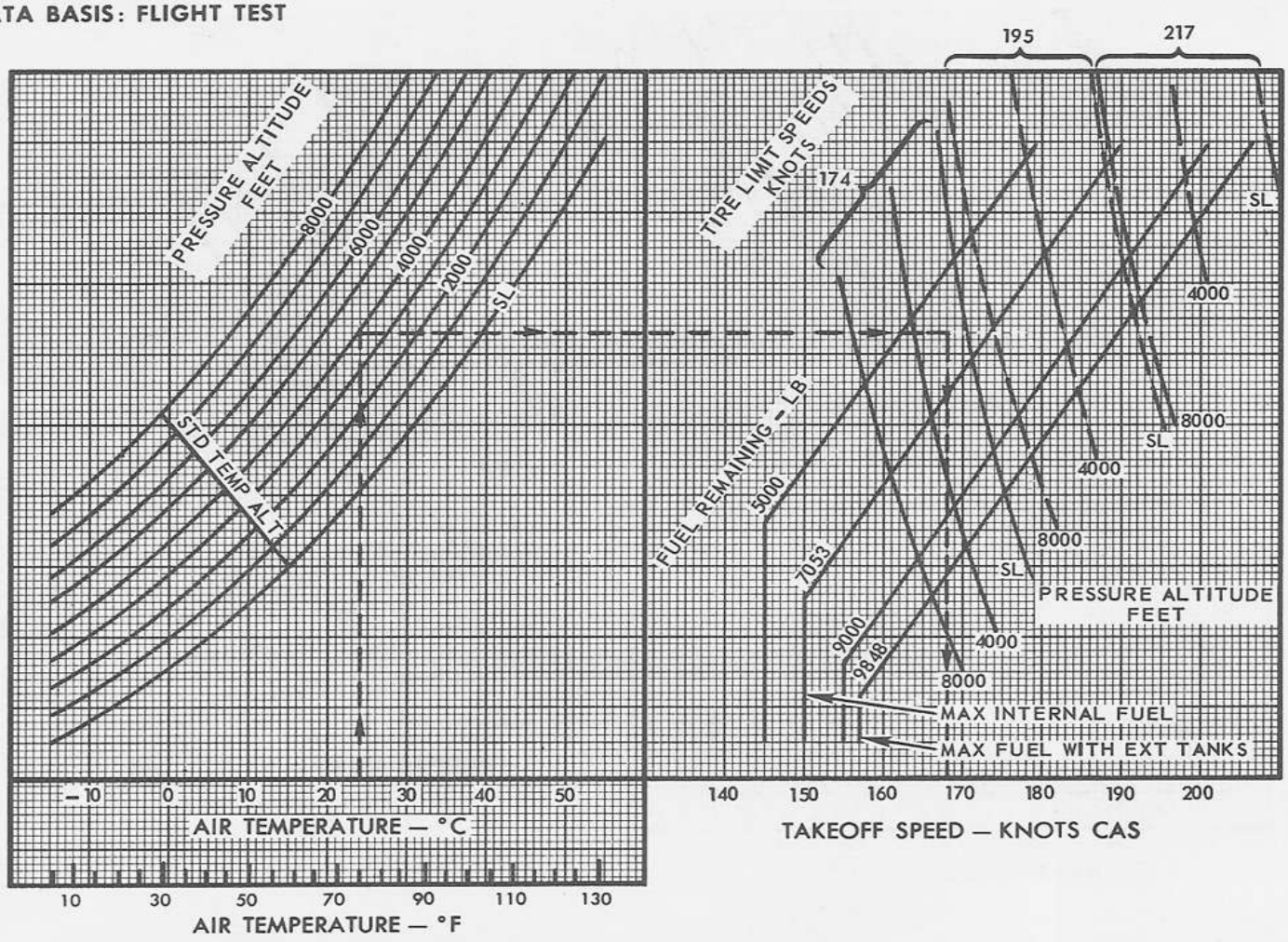
Figure A2-9

TF

takeoff speeds—military thrust

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
ALL FUEL GRADES



NOTE

- TO DETERMINE TAKEOFF SPEED LIMITATION, ADD HEADWIND OR SUBTRACT TAILWIND FROM ZERO WIND TIRE GROUND LIMIT SPEED.
- FUEL REMAINING CURVES BASED ON ZERO FUEL WEIGHT OF 22,256 LB (INCLUDES TANK WT)

22,179C

Figure A2-10

F TF

maximum refusal speed

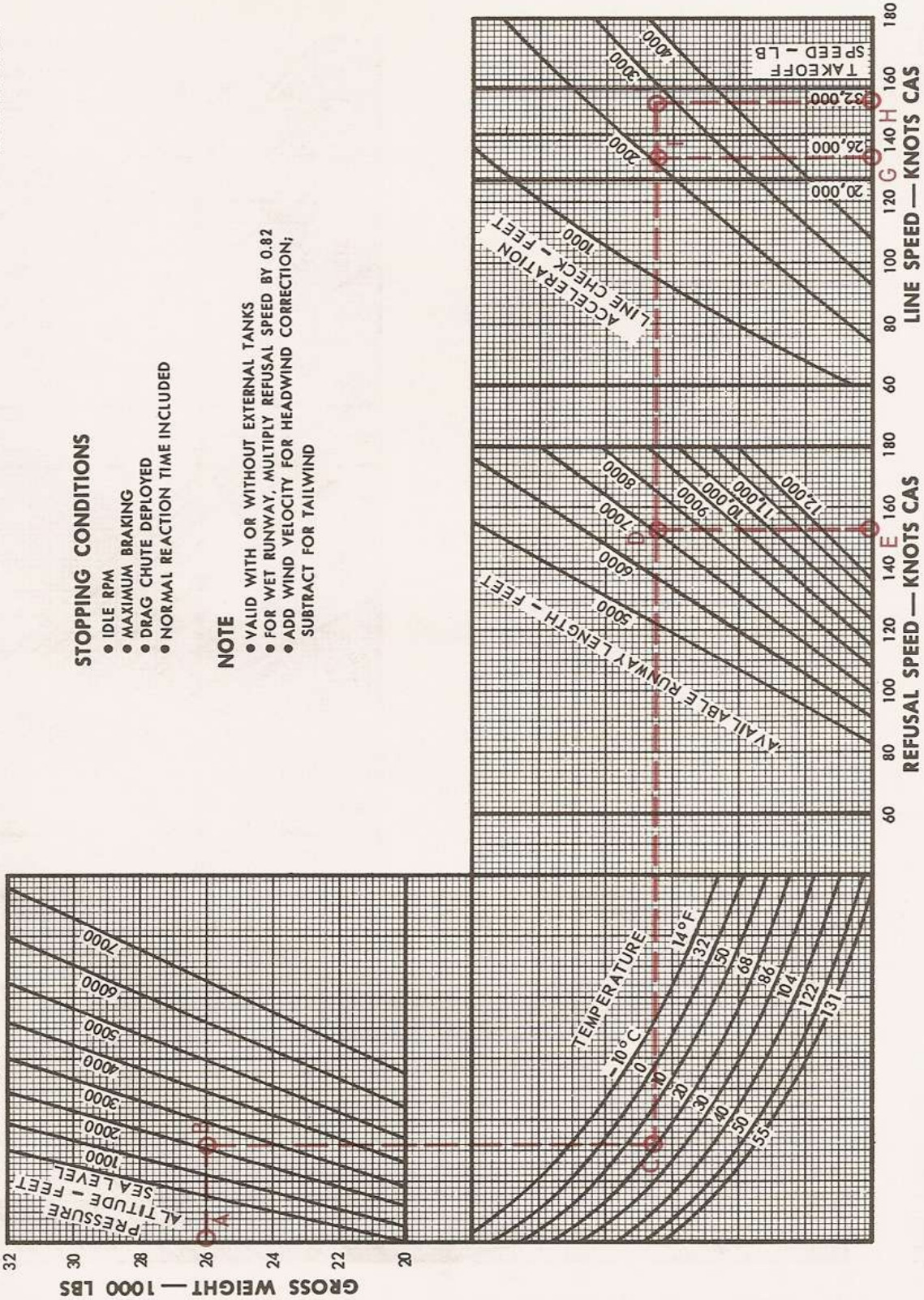
DRY, HARD SURFACE RUNWAY • ZERO WIND
MAXIMUM THRUST • CONFIGURATION: GEAR DOWN

ENGINE: J57-23
ALL FUEL GRADES

STOPPING CONDITIONS
- IDLE RPM
- MAXIMUM BRAKING
- DRAG CHUTE DEPLOYED
- NORMAL REACTION TIME INCLUDED

NOTE
- VALID WITH OR WITHOUT EXTERNAL TANKS
- FOR WET RUNWAY, MULTIPLY REFUSAL SPEED BY 0.82
- ADD WIND VELOCITY FOR HEADWIND CORRECTION; SUBTRACT FOR TAILWIND

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



22,058G

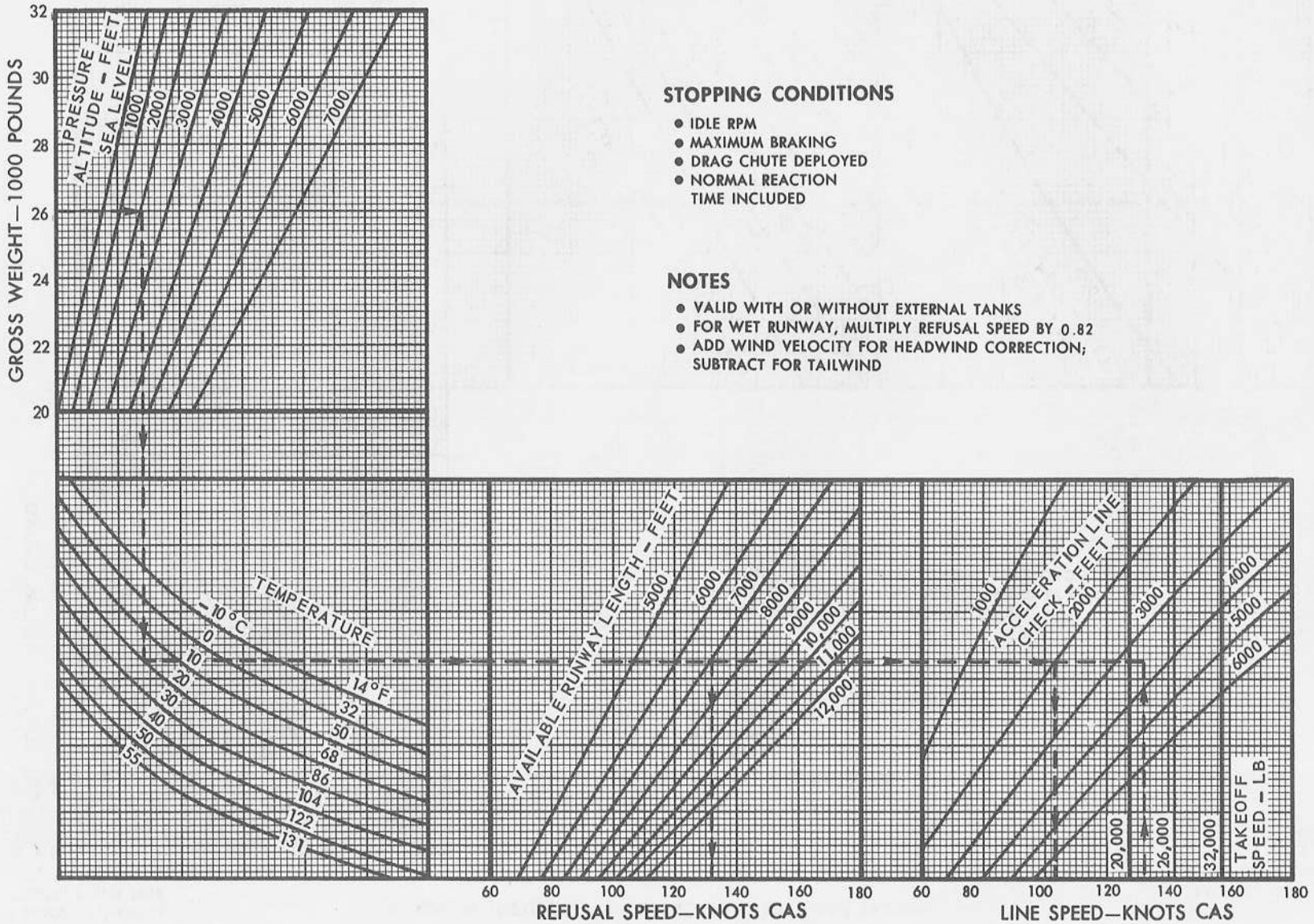
Figure A2-11

maximum refusal speed

MODEL: F/TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

DRY, HARD SURFACE RUNWAY • ZERO WIND • MILITARY THRUST
CONFIGURATION: GEAR DOWN

ENGINE: J57-23
FUEL GRADE: JP-4



STOPPING CONDITIONS

- IDLE RPM
- MAXIMUM BRAKING
- DRAG CHUTE DEPLOYED
- NORMAL REACTION TIME INCLUDED

NOTES

- VALID WITH OR WITHOUT EXTERNAL TANKS
- FOR WET RUNWAY, MULTIPLY REFUSAL SPEED BY 0.82
- ADD WIND VELOCITY FOR HEADWIND CORRECTION; SUBTRACT FOR TAILWIND

F TF

Figure A2-12

22,059G

Part 3 CLIMB

TABLE OF CONTENTS

Page

Climb Chart A3-2

CLIMB

Figures A3-1 through A3-8 are graphical presentations of the climb performance for military thrust and maximum thrust, with or without external tanks. These charts show time to climb and distance to climb as a function of gross weight. Gross weight guide lines represent the weight reduction as fuel is used in the climb. The recommended climb schedules are shown in table form on each chart, and are as follows:

F-102A

Military thrust: 330 KCAS to 0.89 indicated Mach
Maximum thrust: 410 KCAS to 0.94 indicated Mach

TF-102A

Military thrust: 335 KCAS to 0.79 indicated Mach
Maximum thrust: 365 KCAS to 0.90 indicated Mach

The above climb schedules allow for a constant attitude acceleration from takeoff speed to proper CAS value for maximum or military thrust; climb at appropriate CAS until reaching proper climb Mach number; then climb to altitude at constant Mach number. An intermediate step in the acceleration phase from takeoff to climb speed is included, namely 258 KCAS. Normal operational procedure is to use maximum thrust to this speed. When a military thrust climb is to be used, it is first required to obtain time, distance and fuel used to 258 KCAS from the maximum thrust climb chart. Then enter the military thrust climb chart at 258 KCAS to obtain the climb performance data. When for reasons of practicality or other operational reasons a military thrust takeoff is required, the performance data to 258 KCAS can be obtained from the military climb chart.

Note

The acceleration from takeoff to climb speed may be a climbing acceleration for terrain clearance. For good acceleration characteristics, the airplane attitude angle should not exceed 15° with maximum thrust and 10° with military thrust after accelerating in level flight to a speed at which the airplane attitude may be sustained without sinking.

Use

To obtain time, distance, and fuel to climb; enter the chart with the gross weight and altitude at the start of the climb. Follow parallel to the nearest gross weight guide line up to the altitude at end of the climb. Then read directly down to the gross weight scale for the gross weight at end of climb, to the left for distance in climb, and to the right for time to climb. The difference between the initial gross weight and the gross weight at end of climb is the fuel used. The figures obtained must then be corrected for temperature variation if a difference exists between actual and standard day conditions. Multiply each percent correction per degree (found in table on the chart) by the number of degrees variation from standard to obtain the percentage correction. Then multiply the percentage correction by the value for time, distance, or fuel to obtain the correction factor. Add or subtract (as applicable) the correction factor to the value for time, distance, or fuel.

Note

Add the correction factor when the temperature is above standard, and subtract the correction factor when temperature is below standard.

Sample Problem

Find the time, distance, and fuel used for a maximum thrust climb to an altitude of 45,000 feet from 258 KCAS at sea level. A temperature variation of −20°C from standard exists.

A. Enter figure A3-1 with initial gross weight—27,300 pounds.

B. Is initial altitude—Sea level at 258 KCAS.

C. Is initial distance—1.0 nautical miles.

D. Is initial time—.03 minutes.

E. Is final altitude—45,000 feet.

F. Is final gross weight—25,450 pounds.

G. Is final distance—42.7 nautical miles.

H. Is final time—5.3 minutes.

A-F. Total fuel used (standard day) (27,300 − 25,450) = 1850 pounds. Total fuel used (standard day − 20°C) = 1850 pounds minus (.016 × 20°C × 1850 lbs) = 1258 pounds.

G-C. Total distance traveled (standard day) (42.7 − 1.0) = 41.7 nautical miles. Total distance traveled (standard day − 20°C) = 41.7 nautical miles minus (.015 × 20°C × 41.7 n.mi) = 29.2 nautical miles.

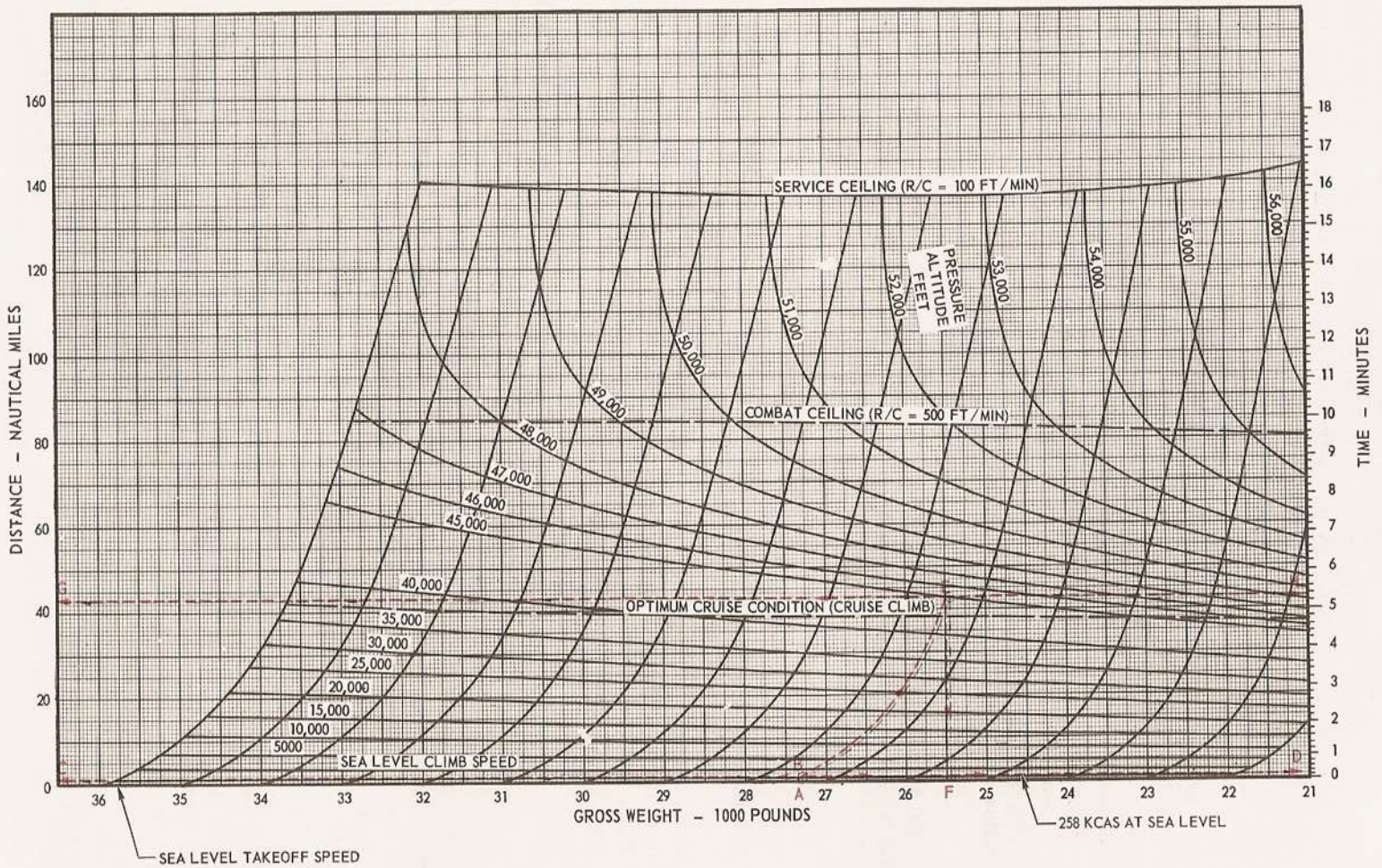
H-B. Total time to climb (standard day) (5.3 − .03) = 5.0 minutes. Total time to climb (standard day − 20°C) = 5.0 minutes minus (.015 × 20°C × 5.0 min.) = 3.5 minutes.

Climb - Maximum Thrust

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

CONFIGURATION: CLEAN STANDARD ATMOSPHERE

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



CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
55,000	205	0.94	537
50,000	231	0.94	537
45,000	259	0.94	537
40,000	291	0.94	537
35,000	326	0.94	540
30,000	363	0.94	551

CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
25,000	403	0.94	564
20,000	410	0.87	535
15,000	410	0.80	501
10,000	410	0.73	468
5,000	410	0.67	438
SL	410	0.62	410

PERCENT CORRECTION PER DEGREE TEMPERATURE VARIATION		
	°F	°C
TIME	0.8	1.5
DISTANCE	0.8	1.5
FUEL	0.9	1.6

NOTE

THE CORRECTION FACTOR IS ADDED WHEN TEMPERATURE IS ABOVE STANDARD, AND SUBTRACTED WHEN TEMPERATURE IS BELOW STANDARD.

Figure A3-1

22,011 I

F

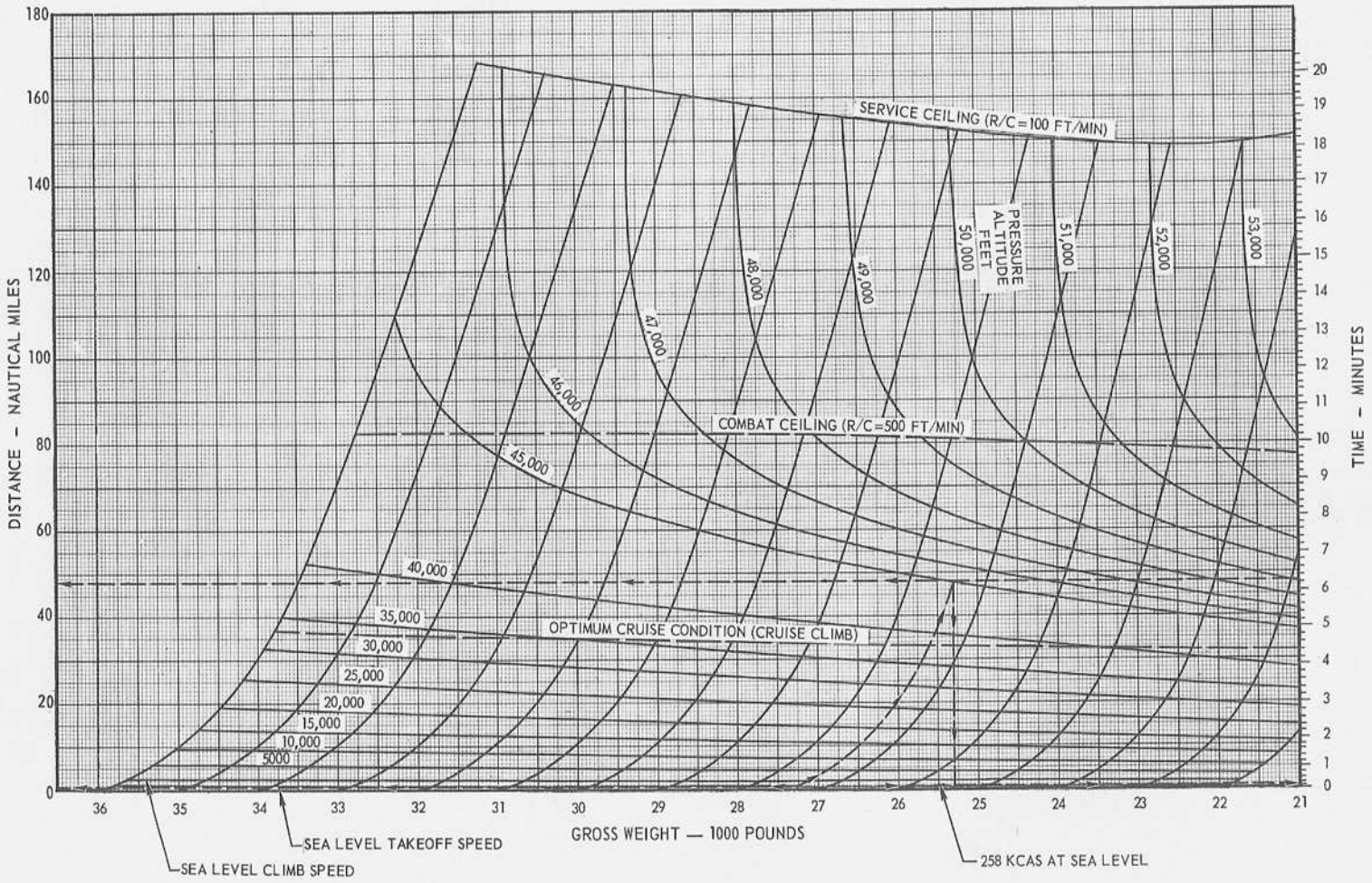
MODEL: TF-102A
DATE: 1 NOVEMBER 1959

climb - maximum thrust

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

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

DATA BASIS: FLIGHT TEST



CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
50,000	219	0.89	513
45,000	246	0.89	513
40,000	276	0.89	513
35,000	310	0.89	515
30,000	346	0.89	527
25,000	365	0.85	514

CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
20,000	365	0.78	478
15,000	365	0.71	446
10,000	365	0.65	416
5,000	365	0.60	389
SL	365	0.55	363

	PERCENT CORRECTION PER DEGREE TEMPERATURE VARIATION	
	°F	°C
TIME	0.6	1.2
DISTANCE	0.6	1.2
FUEL	0.6	1.0

NOTE

THE CORRECTION FACTOR IS ADDED WHEN TEMPERATURE IS ABOVE STANDARD, AND SUBTRACTED WHEN TEMPERATURE IS BELOW STANDARD.

22,185C

Figure A3-2

climb - maximum thrust

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

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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

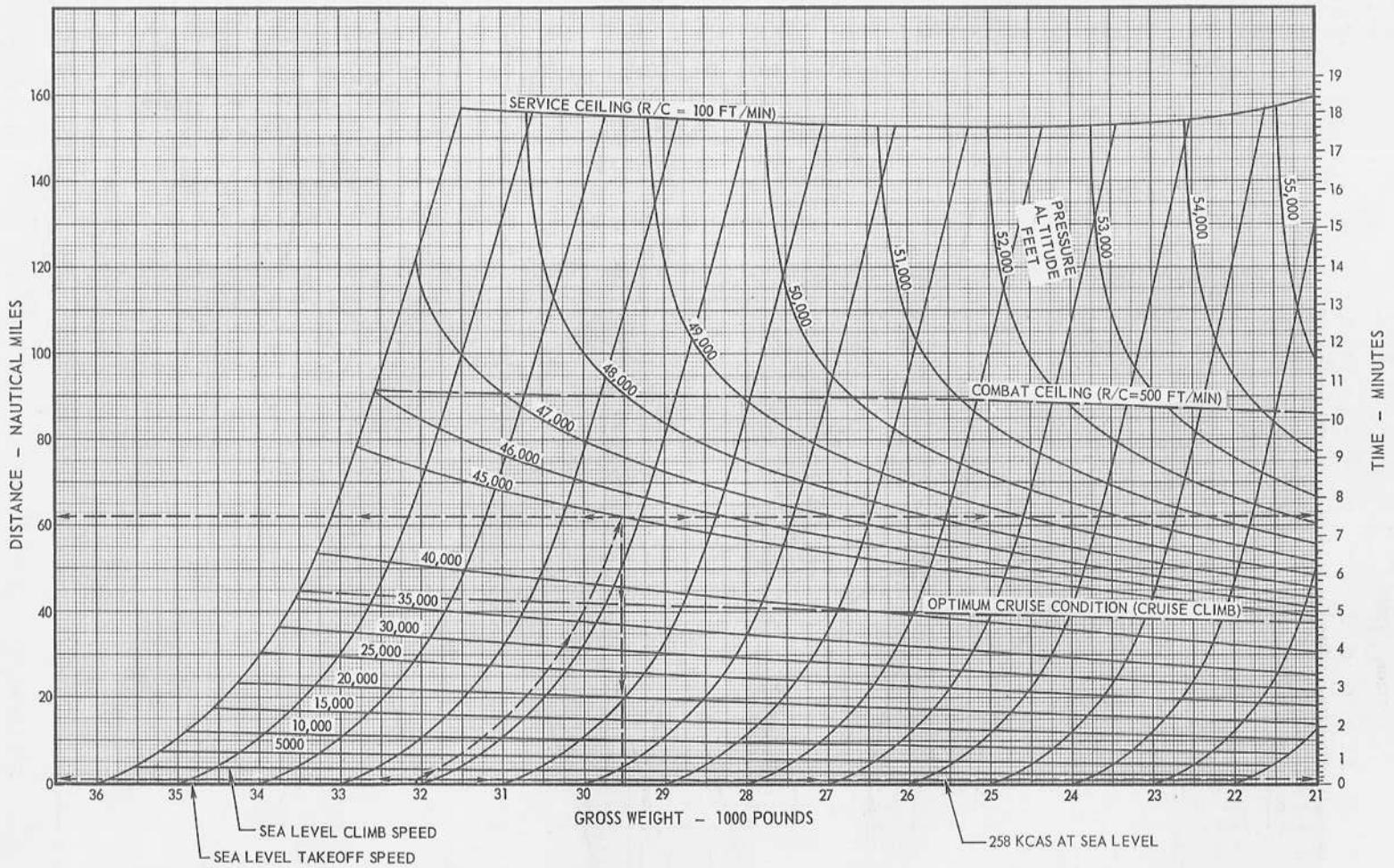


Figure A3-3

CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
55,000	205	0.94	537
50,000	231	0.94	537
45,000	259	0.94	537
40,000	291	0.94	537
35,000	326	0.94	540
30,000	363	0.94	551

CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
25,000	403	0.94	564
20,000	410	0.87	535
15,000	410	0.80	501
10,000	410	0.73	468
5,000	410	0.67	438
SL	410	0.62	410

PERCENT CORRECTION PER DEGREE TEMPERATURE VARIATION		
	°F	°C
TIME	0.6	1.2
DISTANCE	0.6	1.2
FUEL	0.6	1.1

NOTE

THE CORRECTION FACTOR IS ADDED WHEN TEMPERATURE IS ABOVE STANDARD, AND SUBTRACTED WHEN TEMPERATURE IS BELOW STANDARD.

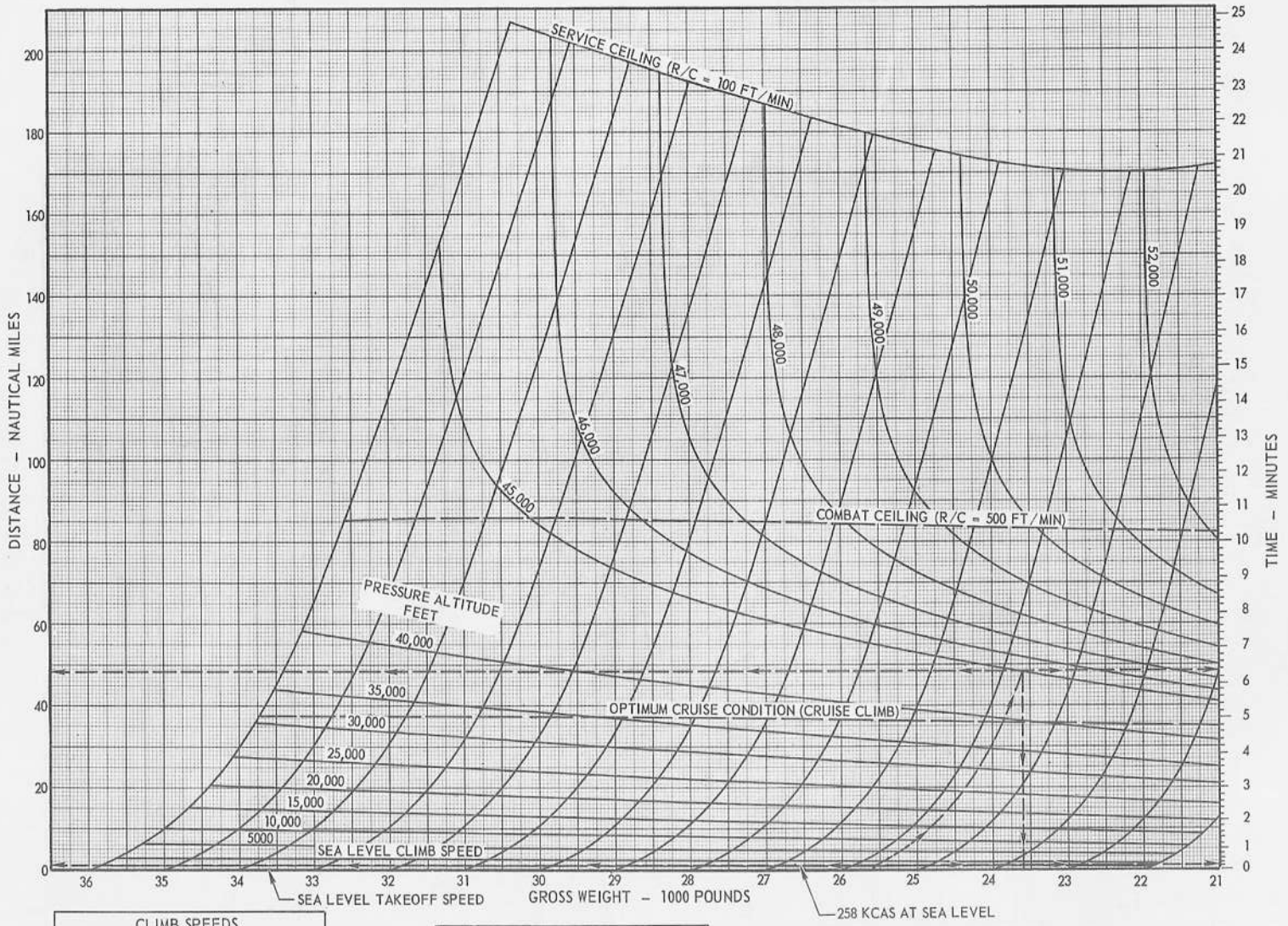
F

climb - maximum thrust

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
50,000	219	0.89	513
45,000	246	0.89	513
40,000	276	0.89	513
35,000	310	0.89	515
30,000	346	0.89	527
25,000	365	0.85	514

CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
20,000	365	0.78	478
15,000	365	0.71	446
10,000	365	0.65	416
5,000	365	0.60	389
SL	365	0.55	363

PERCENT CORRECTION PER DEGREE TEMPERATURE VARIATION		
	°F	°C
TIME	0.6	1.1
DISTANCE	0.6	1.2
FUEL	0.6	1.0

NOTE

THE CORRECTION FACTOR IS ADDED WHEN TEMPERATURE IS ABOVE STANDARD, AND SUBTRACTED WHEN TEMPERATURE IS BELOW STANDARD.

22,186C

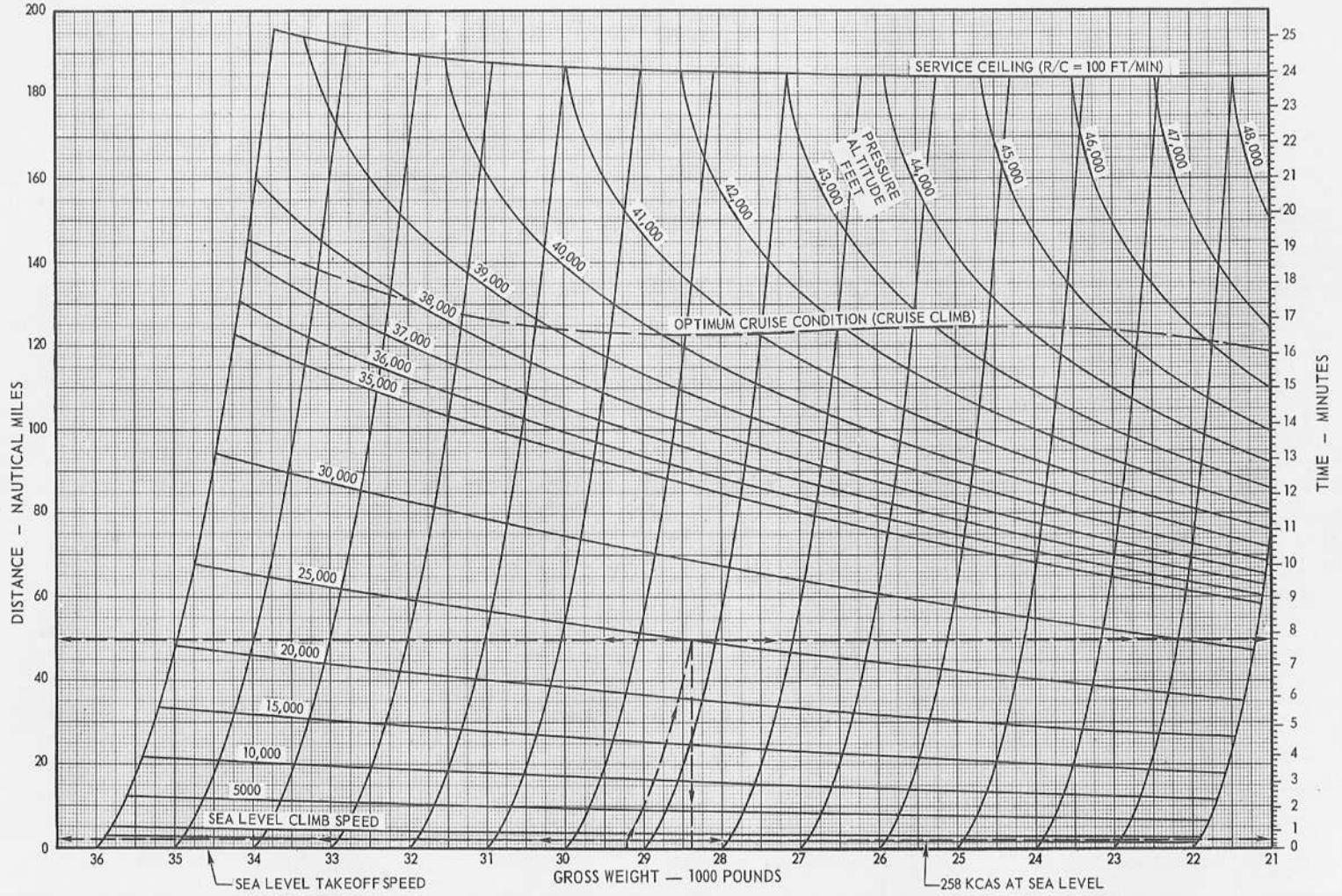
Figure A3-4

climb - military thrust

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

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

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



CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
50,000	217	0.89	509
45,000	243	0.89	509
40,000	273	0.89	509
35,000	306	0.89	511
30,000	330	0.86	507
25,000	330	0.78	471

CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
20,000	330	0.71	438
15,000	330	0.65	407
10,000	330	0.59	379
5,000	330	0.54	354
SL	330	0.50	330

	PERCENT CORRECTION PER DEGREE TEMPERATURE VARIATION	
	°F	°C
TIME	0.8	1.4
DISTANCE	0.9	1.6
FUEL	0.8	1.4

NOTE

THE CORRECTION FACTOR IS ADDED WHEN TEMPERATURE IS ABOVE STANDARD, AND SUBTRACTED WHEN TEMPERATURE IS BELOW STANDARD

F

Figure A3-5

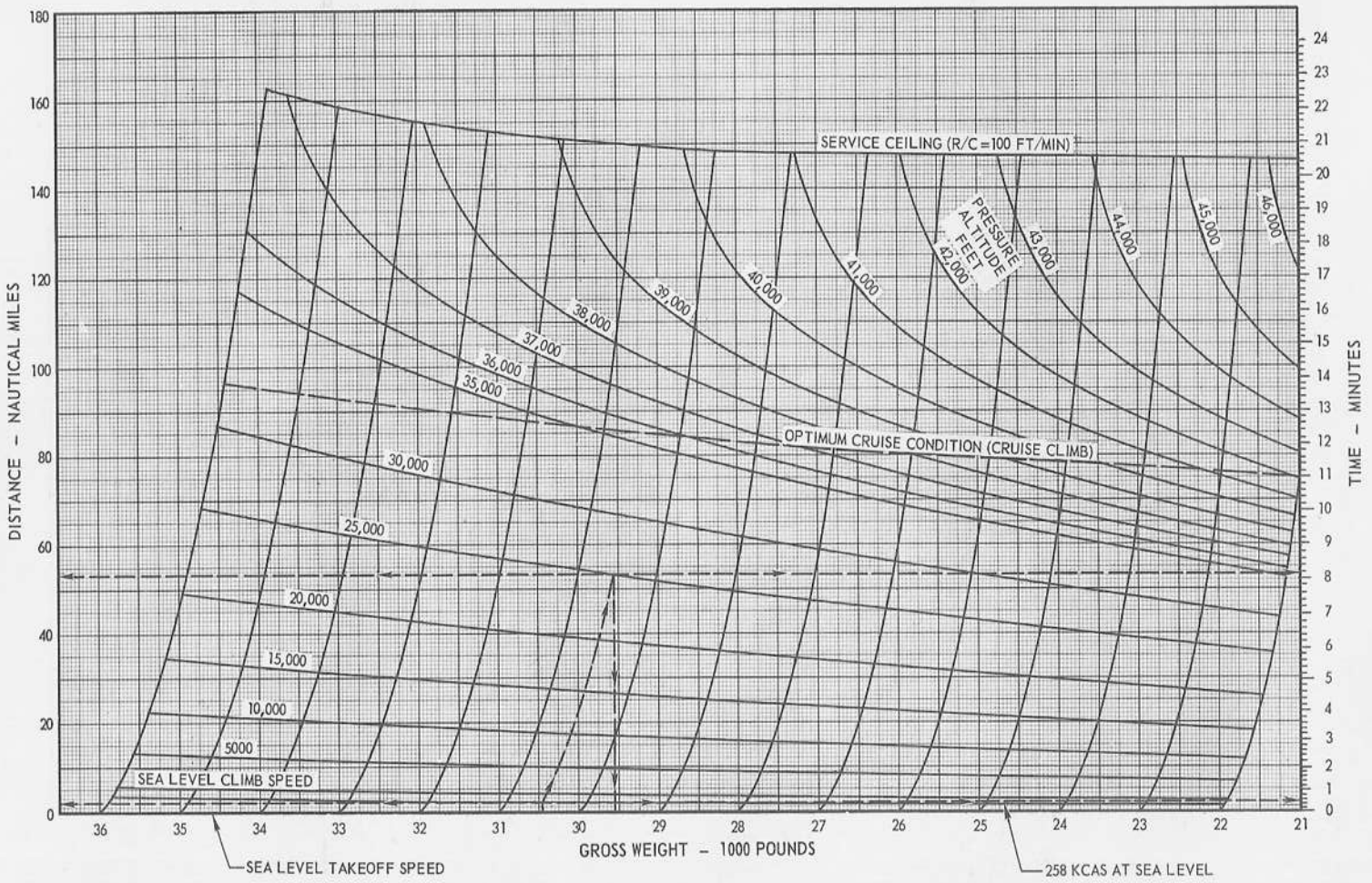
22,013H

climb-military thrust

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN STANDARD ATMOSPHERE

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



CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
50,000	189	0.78	450
45,000	212	0.78	450
40,000	238	0.78	450
35,000	268	0.78	452
30,000	300	0.78	463
25,000	333	0.78	473

CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
20,000	335	0.72	442
15,000	335	0.66	411
10,000	335	0.60	383
5,000	335	0.55	357
SL	335	0.50	334

	PERCENT CORRECTION PER DEGREE TEMPERATURE VARIATION	
	°F	°C
TIME	0.7	1.3
DISTANCE	0.8	1.4
FUEL	0.8	1.4

NOTE

THE CORRECTION FACTOR IS ADDED WHEN TEMPERATURE IS ABOVE STANDARD, AND SUBTRACTED WHEN TEMPERATURE IS BELOW STANDARD.

22,183C

Figure A3-6

F

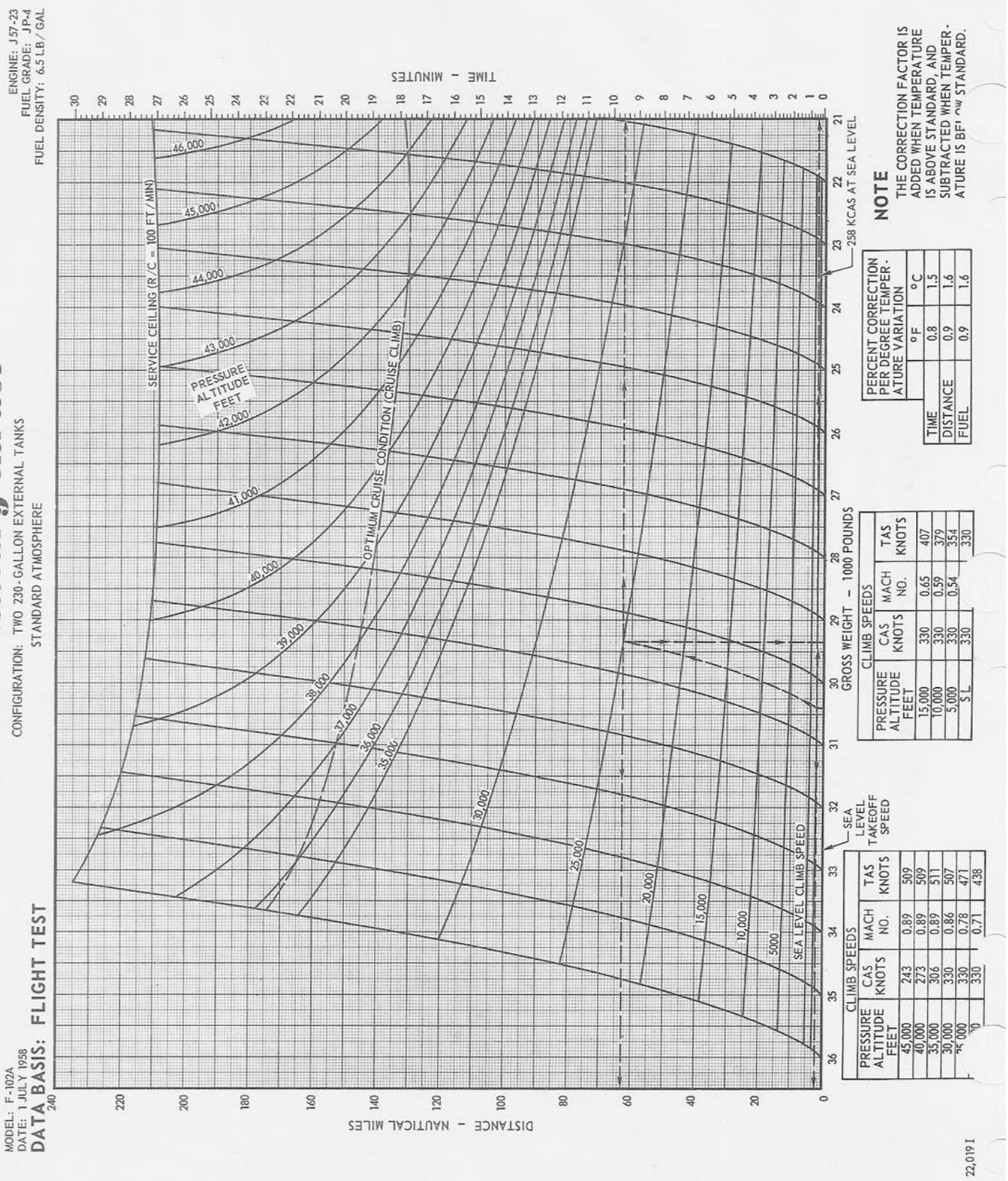

climb - military thrust
CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE
ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB / GAL
MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST
TIME - MINUTES
DISTANCE - NAUTICAL MILES
GROSS WEIGHT - 1000 POUNDS
SERVICE CEILING (R/C = 100 FT/MIN)
PRESSURE ALTITUDE FEET
OPTIMUM CRUISE CONDITION (CRUISE CLIMB)
SEA LEVEL CLIMB SPEED
258 KCAS AT SEA LEVEL
SEA LEVEL TAKEOFF SPEED
46,000
45,000
44,000
43,000
42,000
41,000
40,000
39,000
38,000
37,000
36,000
35,000
30,000
25,000
20,000
15,000
10,000
5000
CLIMB SPEEDS
PRESSURE ALTITUDE FEET | CAS KNOTS | MACH NO. | TAS KNOTS
45,000 | 243 | 0.89 | 509
40,000 | 273 | 0.89 | 509
35,000 | 306 | 0.89 | 511
30,000 | 330 | 0.86 | 507
25,000 | 330 | 0.78 | 471
SL | 330 | 0.71 | 438
CLIMB SPEEDS
PRESSURE ALTITUDE FEET | CAS KNOTS | MACH NO. | TAS KNOTS
15,000 | 330 | 0.65 | 407
10,000 | 330 | 0.59 | 379
5,000 | 330 | 0.54 | 354
SL | 330 | | 330
PERCENT CORRECTION PER DEGREE TEMPERATURE VARIATION
°F | °C
TIME | 0.8 | 1.5
DISTANCE | 0.9 | 1.6
FUEL | 0.9 | 1.6
NOTE
THE CORRECTION FACTOR IS ADDED WHEN TEMPERATURE IS ABOVE STANDARD, AND SUBTRACTED WHEN TEMPERATURE IS BELOW STANDARD.
22,019 I

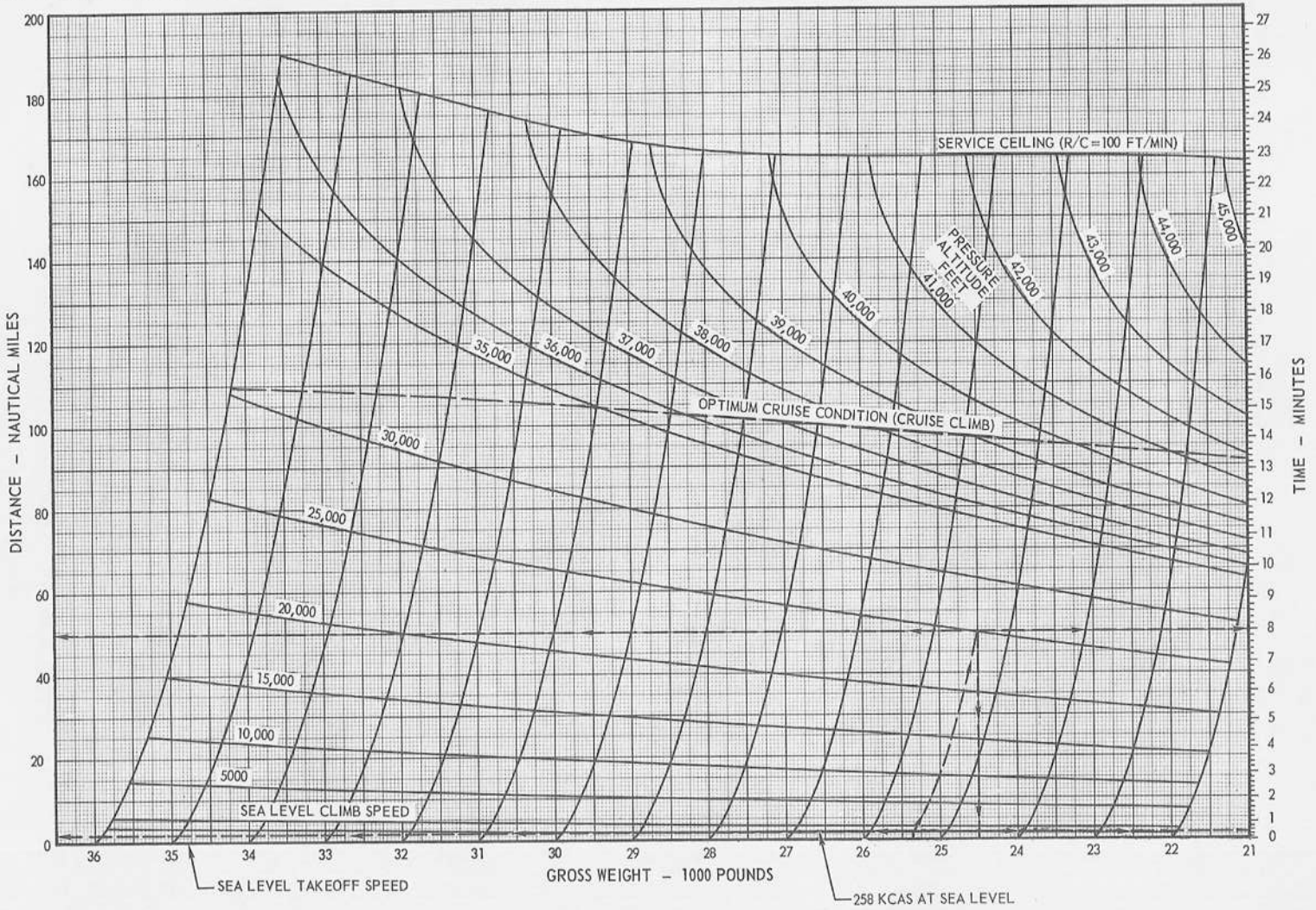
Figure A3-7

climb - military thrust

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
45,000	212	0.78	450
40,000	238	0.78	450
35,000	268	0.78	452
30,000	300	0.78	463
25,000	333	0.78	473

CLIMB SPEEDS			
PRESSURE ALTITUDE FEET	CAS KNOTS	MACH NO.	TAS KNOTS
20,000	335	0.72	442
15,000	335	0.66	411
10,000	335	0.60	383
5,000	335	0.55	357
SL	335	0.50	334

PERCENT CORRECTION PER DEGREE TEMPERATURE VARIATION		
	°F	°C
TIME	0.9	1.6
DISTANCE	1.0	1.8
FUEL	0.9	1.6

NOTE

THE CORRECTION FACTOR IS ADDED WHEN TEMPERATURE IS ABOVE STANDARD, AND SUBTRACTED WHEN TEMPERATURE IS BELOW STANDARD.

22,184C

Figure A3-8

Part 4 RANGE

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Nautical Miles Per 1000 Pounds of Fuel Chart	A4-15

MISSION PROFILE

Mission Profile Charts are shown for the clean airplane in figures A4-1 and A4-2 for the airplane carrying external fuel tanks in figures A4-3 and A4-4, and for the airplane with external fuel tanks dropped when empty in figures A4-5 and A4-6. These charts are basically a plot of altitude versus distance. They show the initial military thrust climb path to the optimum cruise climb condition with reference lines of time and fuel used for constant altitude cruise and for the optimum cruise climb. The time and fuel required to take off, climb, and cruise a given distance at any specified altitude is rapidly determined by a linear interpolation between reference lines. The fuel reference lines include takeoff allowances as noted but the time reference lines do not include any takeoff allowance. No allowance is made for landing reserves. The altitude for maximum range does not apply to short range flights. The cruise altitude for a short range mission is determined by the maximum range point along a constant fuel line in the vicinity of the desired range. The effect of wind may be closely approximated by computing the overall air distance required in cruise (neglecting the wind effect on climb since very little time is spent in this phase of the mission). The chart includes tabular data concerning speeds, fuel flows and engine power conditions for cruise at constant altitude and the cruise-climb. The climb-schedule is tabulated along the altitude scale. Takeoff weights are noted; the effect of weight change due to fuel used are included.

Use

The fuel and time required to fly a given air distance at a specified altitude are determined by entering the chart with the desired air distance, reading up to the specified altitude, and interpolating (if necessary) between reference lines for fuel and time. Given a specific fuel quantity, the reverse procedure may be used to determine the air distance. For a given fuel quantity the maximum available range is obtained by locating the fuel quantity available for the flight, total fuel quantity less desired fuel reserve for loiter, descent, and landing on the fuel reference lines (interpolating if necessary), then following the fuel reference line to the altitude at which the air distance is a maximum. The reverse procedure is used to find the minimum fuel required to fly a given air distance. To obtain the fuel and time required to fly a given ground distance with head or tail winds, correct the ground distance to an equivalent air distance and use this distance in the chart.

The recommended procedure for wind correction is as follows:

1. Compute average true airspeed (TAS) (Distance, time with zero wind conditions).
2. Apply wind to obtain ground speed (Subtract headwind; add tailwind).
3. Compute time with wind effect (Distance ÷ ground speed).
4. To obtain actual air distance (Multiply time with wind effect × TAS).
5. Enter profile chart at cruise altitude and corrected air distance – interpolate for fuel required.

Sample Problem

Find the minimum fuel and the time required to fly a distance of 250 nautical miles with 30-knot headwind. Takeoff gross weight is 28,150 pounds.

a. Enter figure A4-1 with the air distance—250 nautical miles.

b. Read up to the altitude at which the fuel reference lines indicate maximum range (cruise-climb). Fuel required (no wind) – 2995 pounds. Time—31.0 minutes.

c. Subtract headwind – fuel required with 30-knot headwind is 3100 pounds and time is 33.1 minutes.

To obtain this range, the mission profile could be flown with a military thrust climb to the optimum cruise altitude (approximately 38,000 feet) following takeoff.

INTERCEPT PROFILE

The intercept profile for the clean airplane, figures A4-7 and A4-8, is presented in a manner similar to the mission profiles. The major differences are: (1) climb with maximum thrust and (2) constant altitude cruise at military thrust. The intercept profile should be restricted to missions that require minimum time.

Use

The intercept profile chart is used in the same manner as the mission profile charts.

OPTIMUM RETURN PROFILE

The optimum return profiles are shown in figures A4-9 and A4-10 for the clean airplane, and figures A4-11 and A4-12 for airplanes carrying external fuel tanks. These profiles indicate the minimum fuel required to fly a given air distance; or conversely, the maximum distance that can be flown with a given amount of fuel remaining at any given altitude. The return flight path requires a military thrust climb from lower altitudes, to the optimum return altitude and either a cruise-climb or a

constant altitude cruise as required. For short range returns or low fuel quantities, the return is made at the initial altitude. The recommended return flight paths are indicated on the charts by different shaded areas and notes. Operating conditions for climb and cruise are shown in the tables along with the profile.

Note

Fuel required for loiter, descent, or landing is not included in these profiles.

Use

To use the charts, enter the respective chart at the desired return distance, read up to the initial airplane altitude and interpolate for the fuel required to return. The desired fuel reserves must be added to the fuel required. If the chart is entered with the fuel available for return at the initial airplane altitude, read down for the return distance. In this case, the fuel reserve must be subtracted from the total fuel available before entering the chart. Wind effects are computed in the same manner as discussed under the mission profiles.

Sample Problem

Find the fuel required to return a distance of 300 nautical miles under 30-knot wind conditions.

a. Enter figure A4-9 at the return distance—300 nautical miles.

b. Read up to the initial altitude—35,000 feet.

c. Read the fuel required for return (no wind)—1330 pounds. Fuel required with wind — 1270 pounds. Add required reserve fuel to the return fuel to get total fuel required.

The flight path would be as follows.

a. Climb with military thrust to cruise altitude 45,800 feet using schedule as indicated in the climb table.

b. Cruise-climb to destination—use cruise table.

c. Descend to base and land.

NAUTICAL MILES PER 1000 POUNDS OF FUEL

Charts of nautical miles per 1000 pounds of fuel are provided in figures A4-13 through A4-50 for the clean airplane and for the airplane with external tanks. These charts present the constant altitude cruise specifics as a function of Mach number at 5000-foot altitude intervals from sea level to 45,000 feet. Various gross weights covering the weight range of the airplane have been considered. Recommended cruise speeds for maximum range cruise and for maximum endurance are shown for the instantaneous weights. Engine fuel flow covering the full range of operation are included on the charts. Corrections to the nautical miles per pound of fuel under wind conditions may be made as indicated on the charts.

Note

Correction factors are included on figures A4-29 through A4-44 to provide emergency cruise information. These correction factors provide range and optimum cruise speeds when the armament bay doors are open and aft armament launchers extended. Reducing the suggested normal optimum cruise speed by 13% will provide the emergency cruise speed. Reduce the suggested normal miles per pound figure for a given altitude by 40% for airplanes with a clean configuration, and by 37% for airplanes with external tanks, to arrive at the more accurate mile-per-pound range information.

Use

To find the nautical miles per 1000 pounds of fuel for a given cruise speed and gross weight, enter the chart with the desired cruise speed, read up to the gross weight, and then to the left for the nautical miles per 1000 pounds of fuel. Fuel flow may be determined by interpolation. Engine thrust settings are determined by interpolation between lines of constant engine pressure ratio.

Sample Problem

Find the distance that can be flown with 1000 pounds of fuel at an altitude of 35,000 feet and at 0.90 Mach number. The airplane is in the clean configuration and the average gross weight is 26,000 pounds.

A. Enter figure A4-41 (Nautical Miles Per 1000 Pounds 35,000 Feet—Clean) with the cruise speed Mach 0.90.

B. Read up to the average gross weight — 26,000 pounds. At this point fuel flow and engine pressure ratio can be interpolated—2800 pounds per hour fuel flow and an engine pressure ratio setting of approximately 2.03.

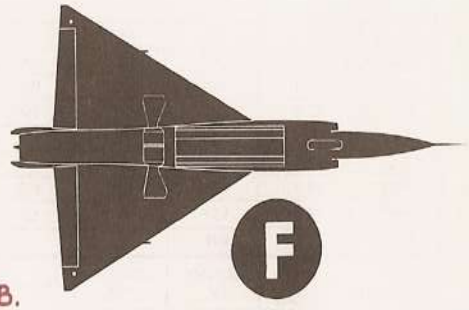
C. Read to the left for the nautical miles per 1000 pounds—185 nautical miles.

mission profile

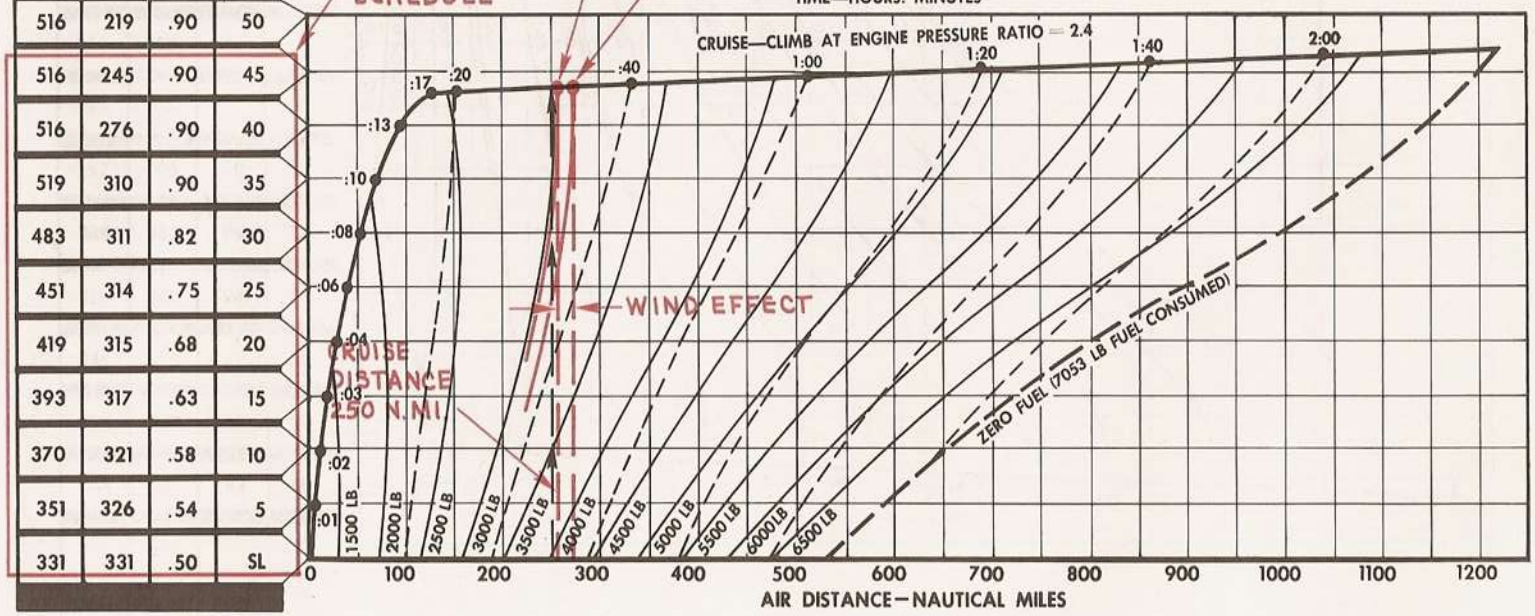
MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
TAKEOFF GROSS WEIGHT: 28,150 LBS.
STANDARD ATMOSPHERE

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



MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FEET
516	219	.90	50
516	245	.90	45
516	276	.90	40
519	310	.90	35
483	311	.82	30
451	314	.75	25
419	315	.68	20
393	317	.63	15
370	321	.58	10
351	326	.54	5
331	331	.50	SL



CRUISE SETTINGS

CRUISE CLEAN CONFIGURATION					
ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*
CRUISE-CLIMB	.92	259	530	2415	2.44
45,000	.91	250	520	2440	2.36
40,000	.90	278	518	2410	2.18
35,000	.83	284	480	2430	1.98
30,000	.74	279	437	2480	1.84
25,000	.67	280	405	2600	1.72
20,000	.62	282	378	2740	1.61
15,000	.56	284	352	2845	1.53
10,000	.51	283	326	2940	1.45
5,000	.47	285	307	3135	1.39
SL	.43	287	287	3300	1.34

*APPROXIMATE

TAKEOFF FUEL ALLOWANCE

NOTE
1. Fuel allowance for start, taxi, and takeoff (1130 lbs) included
2. No allowance or reserves made for loiter descent or landing
3. Use Military Thrust for climb (See Military Thrust Climb Chart for detail information)
4. Cruise at recommended Mach No.

- TIME TO CLIMB
- ZERO FUEL REMAINING
- FUEL CONSUMED
- CRUISE-CLIMB FLIGHT PATH
- TIME (START, TAXI AND TAKEOFF NOT INCLUDED)

22,047D

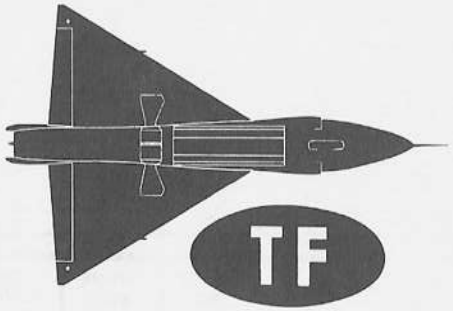
Figure A4-1

mission profile

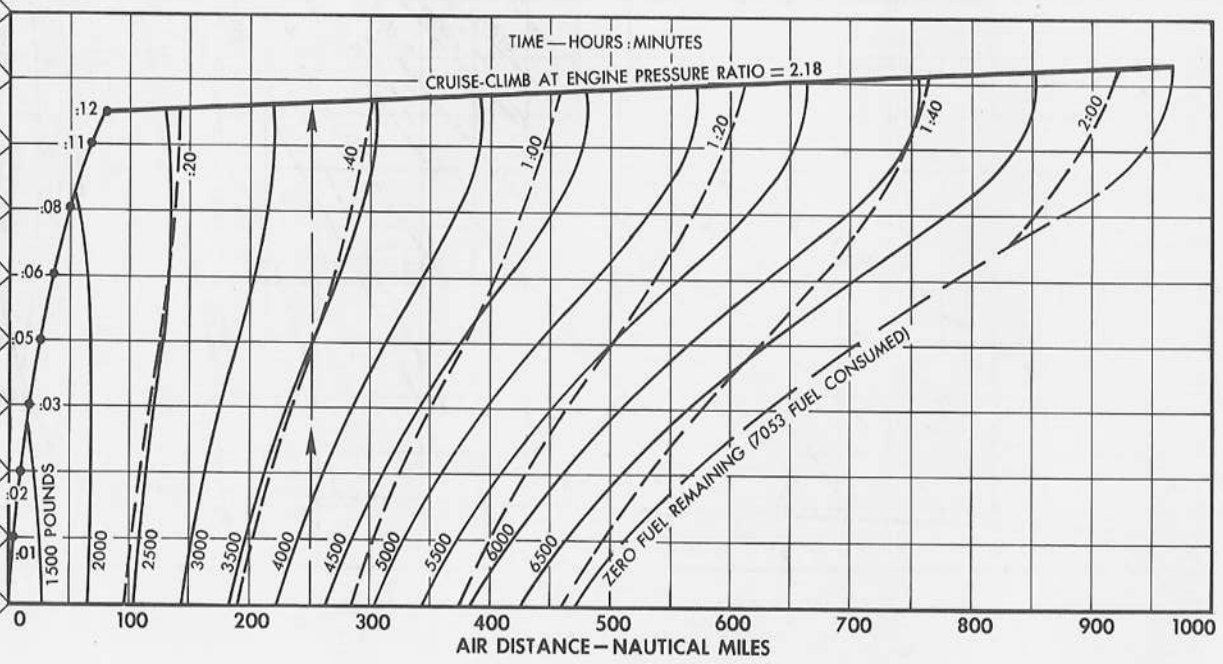
MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE
TAKEOFF GROSS WEIGHT= 28,978 LB

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



MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FT
474	224	.83	45
474	251	.83	40
474	280	.83	35
444	284	.75	30
417	290	.69	25
394	297	.64	20
375	302	.60	15
360	312	.57	10
348	323	.54	5
337	337	.51	SL



CRUISE CLEAN CONFIGURATION					
ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	ENGINE PRESSURE RATIO*	FUEL FLOW LB/HR*
CRUISE-CLIMB	.81	252	465	2.18	2690
35,000	.79	268	455	2.05	2640
30,000	.74	277	434	1.84	2760
25,000	.68	285	412	1.74	2950
20,000	.62	285	381	1.63	3130
15,000	.56	279	347	1.53	3240
10,000	.50	279	322	1.45	3350
5,000	.47	281	302	1.39	3500
SL	.43	283	283	1.34	3630

- • • • TIME TO CLIMB
- — — ZERO FUEL REMAINING
- ——— FUEL CONSUMED
- ——— CRUISE-CLIMB FLIGHT PATH
- — — — TIME (START, TAXI AND TAKEOFF NOT INCLUDED)

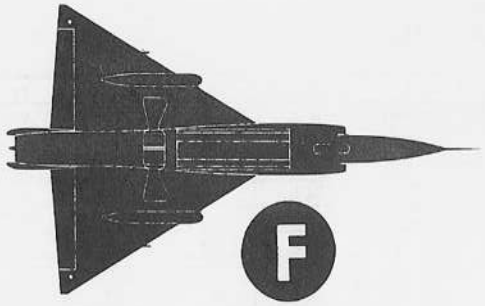
NOTE
1. Fuel allowance for start, taxi and takeoff 1130 lb. included.
2. No allowance or reserves made for loiter, descent or landing.
3. Use military thrust for climb. (See Military Thrust Climb Chart for detail information).
4. Cruise at recommended Mach No.

*APPROXIMATE

22,190B

Figure A4-2

mission profile

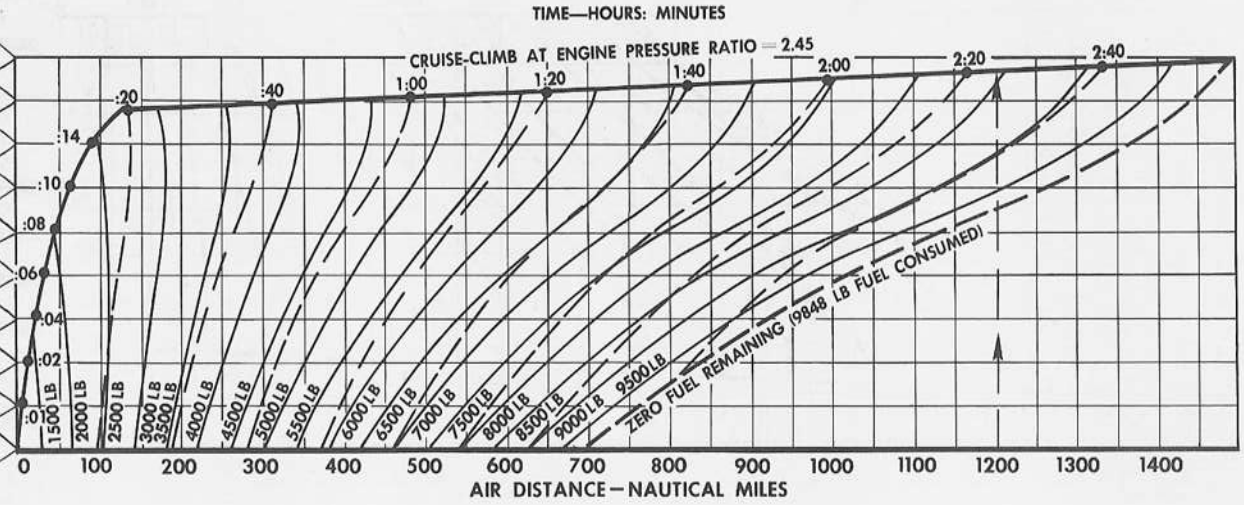


MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO – 230 GALLON EXTERNAL TANKS
TAKEOFF GROSS WEIGHT: 31,276 LBS.
STANDARD ATMOSPHERE

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

MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FEET
462	216	.81	45
462	244	.81	40
464	273	.81	35
427	273	.73	30
400	277	.67	25
376	281	.61	20
354	285	.57	15
337	292	.53	10
320	296	.49	5
304	304	.46	SL



CRUISE—TWO 230 GAL TANKS					
ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*
CRUISE-CLIMB	.89	270	510	2850	2.45
40,000	.89	273	510	2840	2.39
35,000	.83	283	477	2840	2.20
30,000	.72	272	427	2850	1.98
25,000	.65	272	392	2860	1.85
20,000	.59	272	365	2975	1.71
15,000	.54	270	335	3100	1.61
10,000	.49	270	312	3220	1.52
5,000	.45	271	291	3320	1.44
SL	.41	272	272	3440	1.38

*APPROXIMATE

- TIME TO CLIMB
- ZERO FUEL REMAINING
- FUEL CONSUMED
- CRUISE-CLIMB FLIGHT PATH
- TIME (START, TAXI AND TAKEOFF NOT INCLUDED)

NOTE
1. Fuel allowance for start, taxi, and take-off (1130 lbs) included.
2. No allowance or reserves made for loiter, descent or landing.
3. Use Military Thrust for climb (See Military Thrust Climb Chart for detail information).
4. Cruise at recommended Mach No.

22,048B

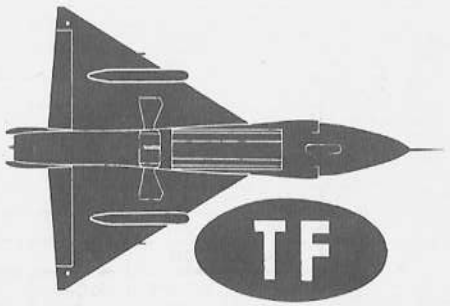
Figure A4-3

mission profile

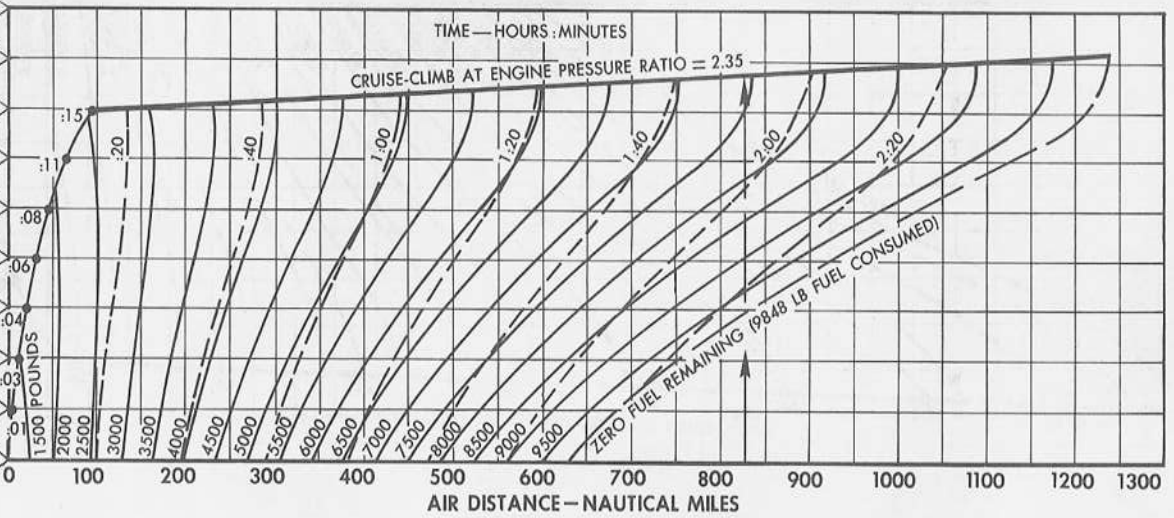
MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO—230 GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE
TAKEOFF GROSS WEIGHT= 32,104 LB

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



MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FT
465	217	.81	45
465	245	.81	40
465	274	.81	35
428	273	.73	30
399	277	.66	25
374	281	.61	20
354	284	.57	15
335	290	.53	10
320	297	.49	5
304	304	.46	SL



CRUISE—TWO 230 GALLON TANKS					
ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	ENGINE PRESSURE RATIO*	FUEL FLOW LB/HR
CRUISE CLIMB	.80	254	460	2.35	3130
35,000	.79	268	455	2.24	3080
30,000	.74	279	436	2.03	3210
25,000	.68	284	410	1.86	3360
20,000	.61	280	375	1.73	3490
15,000	.55	277	344	1.62	3600
10,000	.50	276	319	1.53	3700
5,000	.46	278	300	1.46	3800
SL	.43	281	281	1.39	3950

*APPROXIMATE

— — ZERO FUEL REMAINING
——— FUEL CONSUMED
▬▬▬ CRUISE-CLIMB FLIGHT PATH
– – – TIME (START, TAXI AND TAKEOFF NOT INCLUDED)
• • • • TIME TO CLIMB

NOTE
1. Fuel allowance for start, taxi and takeoff 1130 lb. included.
2. No allowance or reserves made for loiter, descent or landing.
3. Use military thrust for climb. (See Military Thrust Climb Chart for detail information.
4. Cruise at recommended Mach No.

Figure A4-4

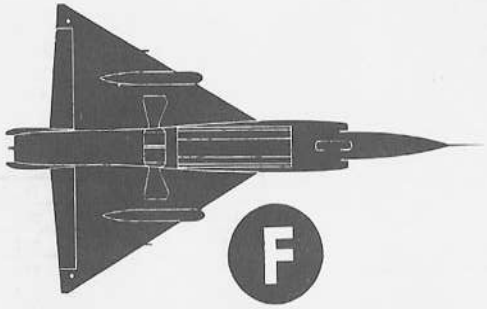
22,191B

mission profile

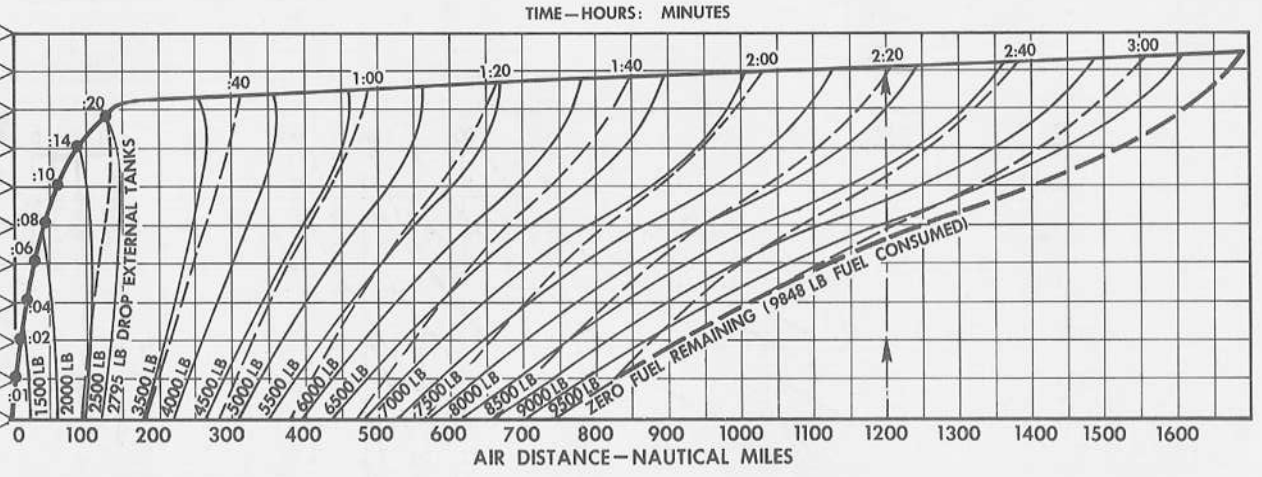
CONFIGURATION: TWO-230 GALLON EXTERNAL TANKS
(DROPPED WHEN EMPTY)
TAKEOFF GROSS WEIGHT: 31,276 LBS.
STANDARD ATMOSPHERE

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

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



MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FEET
462	193	.81	50
462	216	.81	45
462	244	.81	40
464	273	.81	35
427	273	.73	30
400	277	.67	25
376	281	.61	20
354	285	.57	15
337	292	.53	10
320	296	.49	5
304	304	.46	SL



CRUISE-CLEAN CONFIGURATION						CRUISE—TWO 230 GAL TANKS					
ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*	ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*
CRUISE-CLIMB	.92	259	530	2415	2.44	CRUISE-CLIMB	.89	270	510	2850	2.45
45,000	.91	250	520	2440	2.36						
40,000	.90	278	518	2410	2.18	40,000	.89	273	510	2840	2.39
35,000	.83	284	480	2430	1.98	35,000	.83	283	477	2840	2.20
30,000	.74	279	437	2480	1.84	30,000	.72	272	427	2850	1.98
25,000	.67	280	405	2600	1.72	25,000	.65	272	392	2860	1.85
20,000	.62	282	378	2740	1.61	20,000	.59	272	365	2975	1.71
15,000	.56	284	352	2845	1.53	15,000	.54	270	335	3100	1.61
10,000	.51	283	326	2940	1.45	10,000	.49	270	312	3220	1.52
5,000	.47	285	307	3135	1.39	5,000	.45	271	291	3320	1.44
SL	.43	287	287	3300	1.34	SL	.41	272	272	3440	1.38

*APPROXIMATE

22,049B

- TIME TO CLIMB
- ZERO FUEL REMAINING
- FUEL CONSUMED
- CRUISE-CLIMB FLIGHT PATH
- TIME (START, TAXI AND TAKEOFF NOT INCLUDED)

NOTE
1. Fuel allowance for start, taxi and takeoff 1130 lbs included.
2. No allowance or reserves made for loiter, descent or landing.
3. Use Military Thrust for climb (See Military Thrust Climb Chart for detail information).
4. Cruise at recommended Mach No.

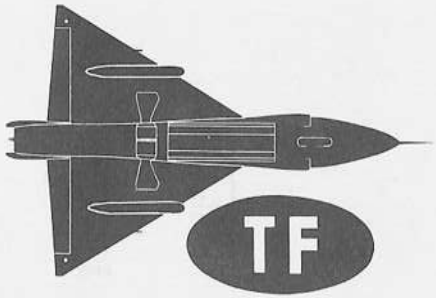
Figure A4-5

mission profile

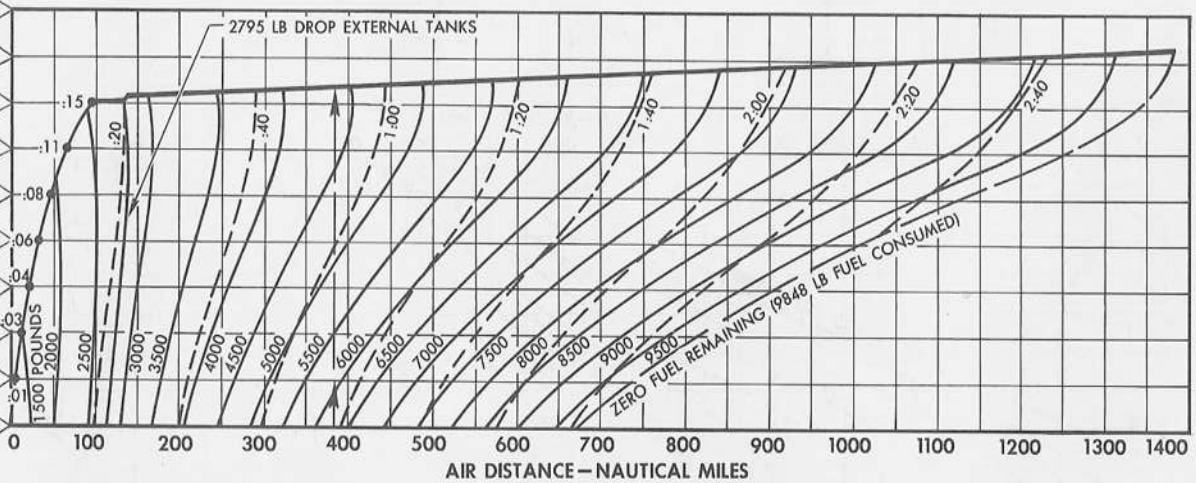
MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO—230 GALLON EXTERNAL TANKS
(DROPPED WHEN EMPTY)
STANDARD ATMOSPHERE
TAKEOFF GROSS WEIGHT= 32,104 LB

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



MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FT
465	217	.81	45
465	245	.81	40
465	274	.81	35
428	273	.73	30
399	277	.66	25
374	281	.61	20
354	284	.57	15
335	290	.53	10
320	297	.49	5
304	304	.46	SL



CRUISE CLEAN CONFIGURATION					
ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	ENGINE PRESSURE RATIO*	FUEL FLOW LB/HR*
CRUISE CLIMB	.81	252	465	2.18	2690
35,000	.79	268	455	2.05	2640
30,000	.74	277	434	1.84	2760
25,000	.68	285	412	1.74	2950
20,000	.62	285	381	1.63	3130
15,000	.56	279	347	1.53	3240
10,000	.50	279	322	1.45	3350
5,000	.47	281	302	1.39	3500
SL	.43	283	283	1.34	3630

CRUISE—TWO 230 GALLON TANKS					
ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	ENGINE PRESSURE RATIO*	FUEL FLOW LB/HR*
CRUISE CLIMB	.80	254	460	2.35	3130
35,000	.79	268	455	2.24	3080
30,000	.74	279	436	2.03	3210
25,000	.68	284	410	1.86	3360
20,000	.61	280	375	1.73	3490
15,000	.55	277	344	1.62	3600
10,000	.50	276	319	1.53	3700
5,000	.46	278	300	1.46	3800
SL	.43	281	281	1.39	3950

*APPROXIMATE

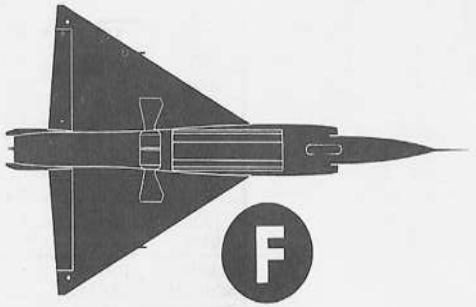
— — ZERO FUEL REMAINING
——— FUEL CONSUMED
▬▬▬ CRUISE-CLIMB FLIGHT PATH
— — — TIME (START, TAXI AND TAKEOFF NOT INCLUDED)
• • • • TIME TO CLIMB

NOTE
1. Fuel allowance for start, taxi and takeoff 1130 lb. included.
2. No allowance or reserves made for loiter, descent or landing.
3. Use military thrust for climb. (See Military Thrust Climb Chart for detail information.)
4. Cruise at recommended Mach No.

22,192B

Figure A4-6

intercept profile

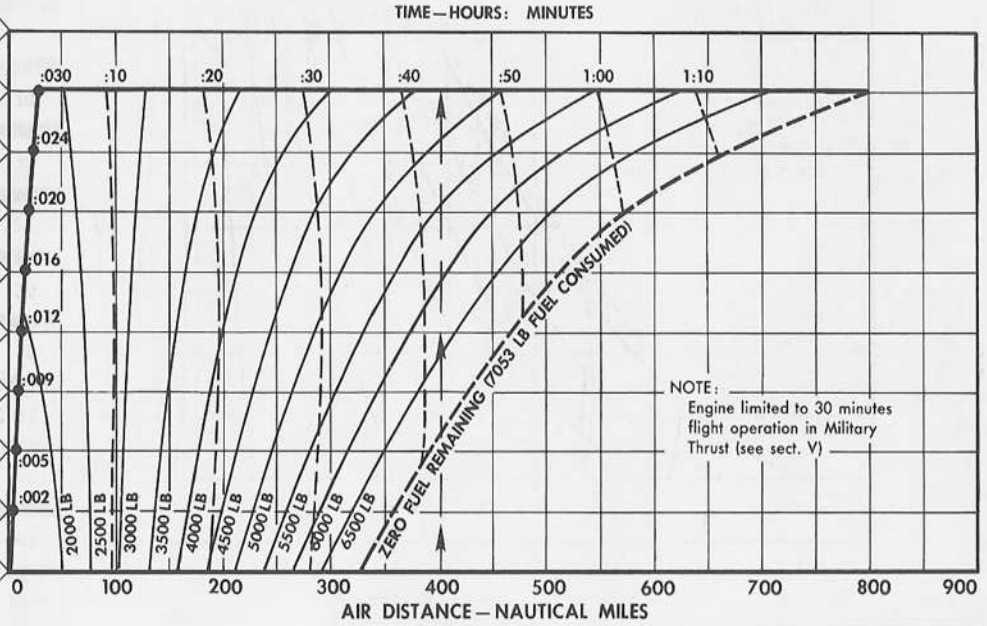


MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
TAKEOFF GROSS WEIGHT: 28,150 LBS.
STANDARD ATMOSPHERE

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

MAXIMUM THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FEET
533	256	.93	45
533	287	.93	40
536	322	.93	35
547	356	.93	30
558	396	.93	25
566	435	.92	20
576	476	.92	15
585	516	.92	10
593	558	.91	5
602	602	.91	SL



NOTE:
1. Fuel allowance for start, taxi and take-off 1130 lbs included.
2. No allowance or reserves made for loiter, descent or landing.
3. Use Military plus Afterburner Thrust for climb.
4. Cruise at Military Thrust.

● ● ● TIME TO CLIMB
— — — ZERO FUEL REMAINING
——— FUEL CONSUMED
— — — TIME (START, TAXI, AND TAKEOFF NOT INCLUDED)

CRUISE CLEAN					
ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	ENGINE PRESSURE RATIO*	FUEL FLOW LB/HR*
40,000	.95	297	547	2.52	3340
35,000	.97	334	559	2.52	4430
30,000	.96	371	567	2.35	5120
25,000	.96	411	576	2.24	5880
20,000	.95	450	584	2.09	6670
15,000	.94	486	588	1.98	7440
10,000	.91	514	582	1.87	8200
5,000	.88	538	571	1.80	9000
SL	.84	555	555	1.76	10,270

*APPROXIMATE

22,050B

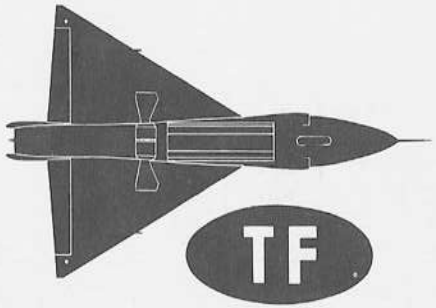
Figure A4-7

intercept profile

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE
TAKEOFF GROSS WEIGHT = 28,978 LB

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



MILITARY + AFTERBURNER THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FT
506	240	.88	45
506	270	.88	40
506	302	.88	35
515	334	.88	30
523	368	.87	25
531	406	.87	20
532	438	.85	15
529	465	.83	10
520	488	.80	5
502	502	.76	SL

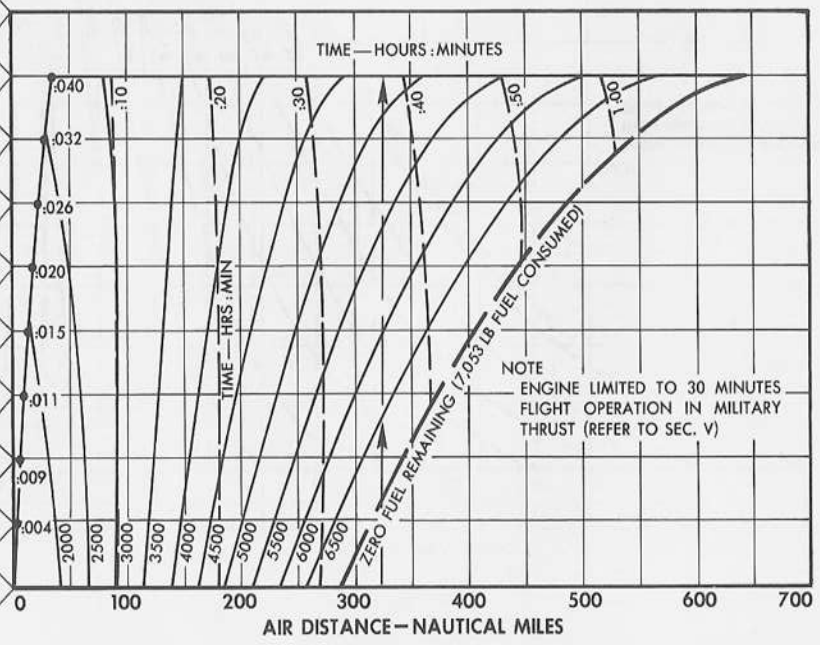


Figure A4-8

- •••• TIME TO CLIMB
- — — ZERO FUEL REMAINING
- ——— FUEL CONSUMED
- — — — TIME (START, TAXI AND TAKEOFF NOT INCLUDED)

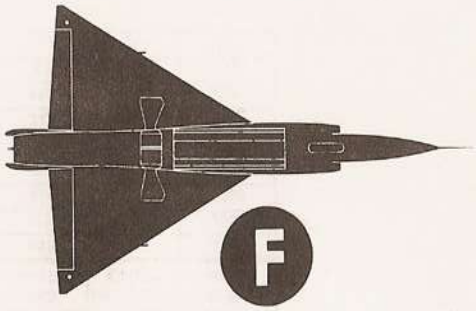
NOTE
1. Fuel allowance for start, taxi and takeoff 1130 lb. included.
2. No allowance or reserves made for loiter, descent or landing.
3. Use military plus afterburner thrust for climb.
4. Cruise at Military Thrust.

CRUISE CLEAN CONFIGURATION					
ALTITUDE FEET	MACH NO.	CAS* KNOTS	TAS* KNOTS	ENG PRESS. RATIO*	FUEL FLOW LB/HR*
40,000	.90	276	514	2.60	3630
35,000	.91	314	526	2.57	4670
30,000	.91	349	534	2.43	5390
25,000	.90	382	539	2.33	6250
20,000	.89	416	543	2.20	7040
15,000	.87	449	545	2.07	7980
10,000	.86	480	545	1.96	8940
5,000	.84	510	542	1.86	9920
SL	.81	536	536	1.79	11,120

*APPROXIMATE

22,193B

optimum return profile



MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
TAKEOFF GROSS WEIGHT: 28,150 LBS.
STANDARD ATMOSPHERE

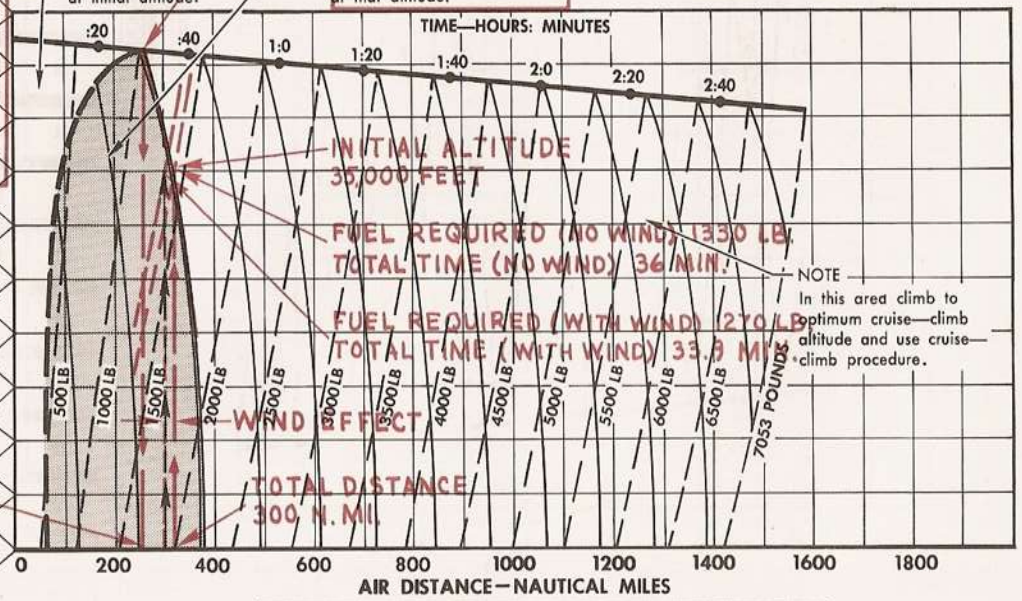
ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

CLIMB SCHEDULE
TIME TO CLIMB - 8.8 MIN. (FROM FIG. A 3-1)

CRUISE ALTITUDE
45,800 FEET

MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FEET
516	219	.90	50
516	245	.90	45
516	276	.90	40
519	310	.90	35
483	311	.82	30
451	314	.75	25
419	315	.68	20
393	317	.63	15
370	321	.58	10
351	326	.54	5
331	331	.50	SL

CRUISE DISTANCE 240 N.MI.



CRUISE TABLE
TRUE AIRSPEED FOR CRUISE AT 45,800 FT.
529 KNOTS

CRUISE-CLEAN CONFIGURATION					
ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*
CRUISE-CLIMB	.92	259	530	2415	2.44
45,000	.91	250	520	2440	2.36
40,000	.90	278	518	2410	2.18
35,000	.83	284	480	2430	1.98
30,000	.74	279	437	2480	1.84
25,000	.67	280	405	2600	1.72
20,000	.62	282	378	2740	1.61
15,000	.56	284	352	2845	1.53
10,000	.51	283	326	2940	1.45
5,000	.47	285	307	3135	1.39
SL	.43	287	287	3300	1.34

*APPROXIMATE

22,051D

- FUEL REQUIRED
- TIME AT CRUISING ALTITUDE
- LINE OF BEST RANGE FOR CONSTANT ALTITUDE
- CLIMB PATH GUIDE LINES
- LINE OF BEST RANGE FOR CRUISE-CLIMB ALTITUDE

NOTE:
(1) Fuel required at any point includes Military Thrust climb to cruise—climb altitude (if below that).
(2) No allowance made for loiter, descent or landing.
(3) Best cruise condition is determined by intersection of climb path guide line and lines of best range.
(4) Cruise at recommended Mach No.

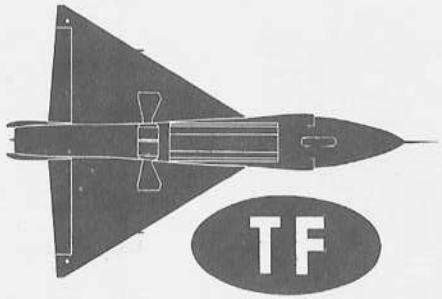
Figure A4-9

optimum return profile

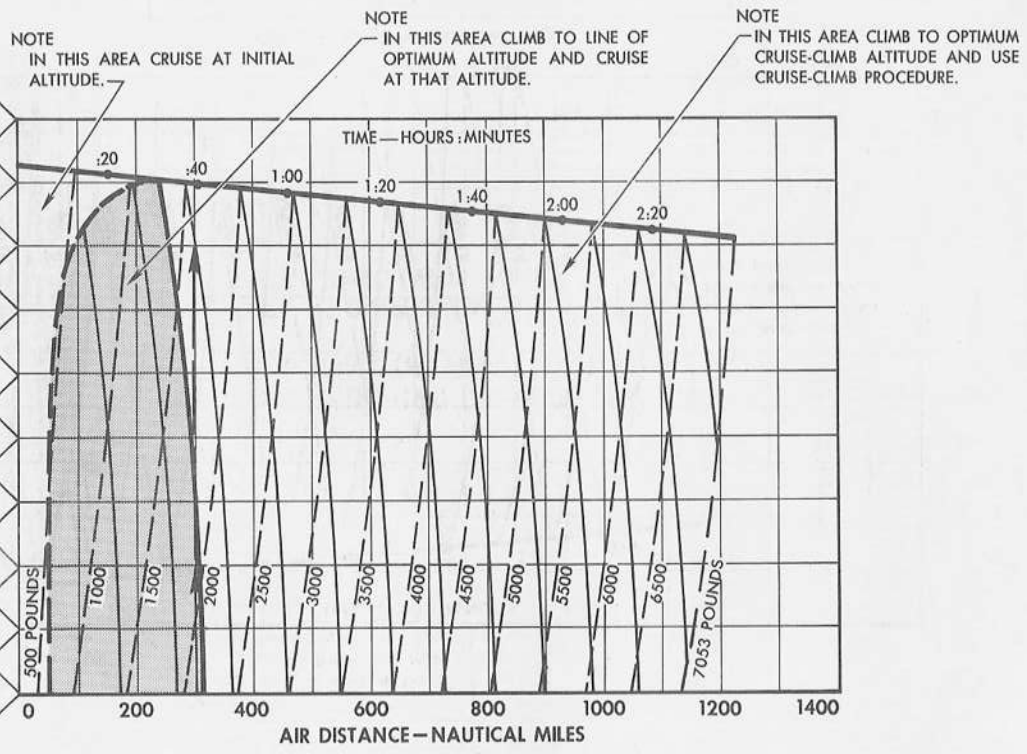
MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE
GROSS WEIGHT = 21,871 - 28,978 LB

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



MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FT
474	224	.83	45
474	251	.83	40
474	280	.83	35
444	284	.75	30
417	290	.69	25
394	297	.64	20
375	302	.60	15
360	312	.57	10
348	323	.54	5
337	337	.51	SL



- TIME AT CRUISING ALTITUDE
- LINE OF BEST RANGE FOR CONSTANT ALTITUDE
- CLIMB PATH GUIDE LINES
- LINE OF BEST RANGE FOR CRUISE-CLIMB ALTITUDE
- FUEL REQUIRED

NOTE
1. Fuel required at any point includes military thrust climb to cruise-climb altitude (if below that).
2. No allowance made for loiter, descent or landing.
3. Best cruise condition is determined by intersection of climb path guide line and lines of best range.
4. Cruise at recommended Mach No.

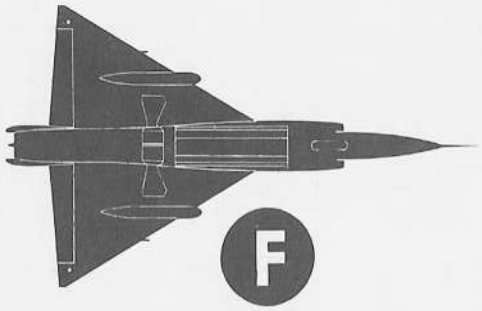
CRUISE CLEAN CONFIGURATION					
ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	ENG PRESS. RATIO*	FUEL FLOW LB/HR*
CRUISE-CLIMB	.81	252	465	2.18	2690
35,000	.79	268	455	2.05	2640
30,000	.74	277	434	1.84	2760
25,000	.68	285	412	1.74	2950
20,000	.62	285	381	1.63	3130
15,000	.56	279	347	1.53	3240
10,000	.50	279	322	1.45	3350
5,000	.47	281	302	1.39	3500
SL	.43	283	283	1.34	3630

*APPROXIMATE

22,194B

Figure A4-10

optimum return profile

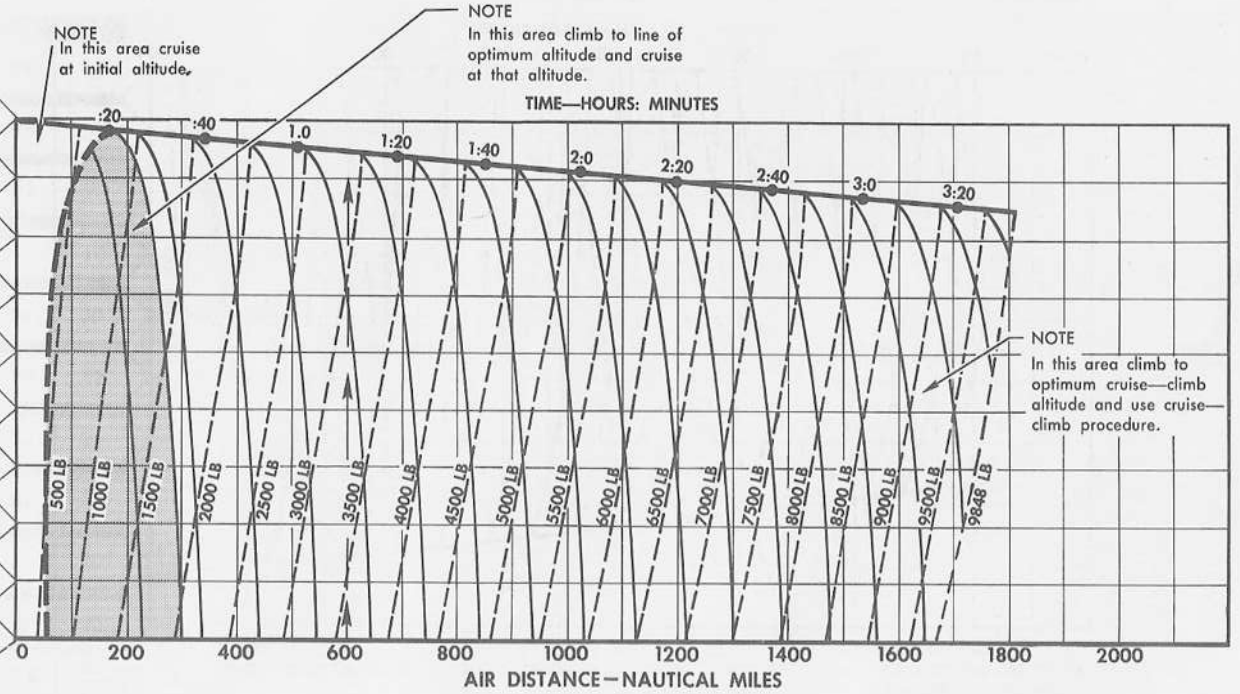


MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO-230 GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE
GROSS WEIGHT: 31,276 LBS.

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

MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FEET
462	216	.81	45
462	244	.81	40
464	273	.81	35
427	273	.73	30
400	277	.67	25
376	281	.61	20
354	285	.57	15
337	292	.53	10
320	296	.49	5
304	304	.46	SL



CRUISE—TWO 230 GAL TANKS					
ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*
CRUISE-CLIMB	.89	270	510	2850	2.45
40,000	.89	273	510	2840	2.39
35,000	.83	283	477	2840	2.20
30,000	.72	272	427	2850	1.98
25,000	.65	272	392	2860	1.85
20,000	.59	272	365	2975	1.71
15,000	.54	270	335	3100	1.61
10,000	.49	270	312	3220	1.52
5,000	.45	271	291	3320	1.44
SL	.41	272	272	3440	1.38

*APPROXIMATE

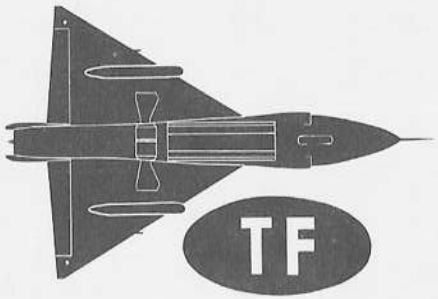
22,052B

- TIME AT CRUISING ALTITUDE
- LINE OF BEST RANGE FOR CONSTANT ALTITUDE
- CLIMB PATH GUIDE LINES
- LINE OF BEST RANGE FOR CRUISE-CLIMB ALTITUDE
- FUEL REQUIRED

NOTE
(1) Fuel required at any point includes Military Thrust climb to cruise—climb altitude (if below that).
(2) No allowance made for loiter, descent or landing.
(3) Best cruise condition is determined by intersection of climb path guide line and lines of best range.
(4) Cruise at recommended Mach No.

Figure A4-11

optimum return profile

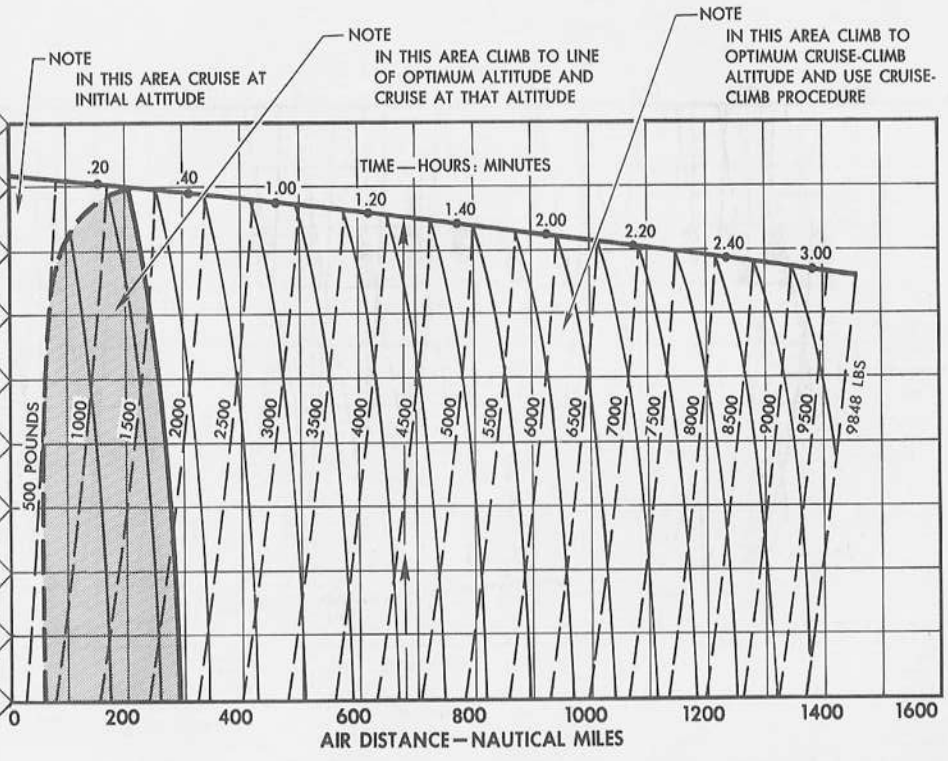


MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO—230 GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE
GROSS WEIGHT=22,202— 32,104 LBS

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

MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FT
465	217	.81	45
465	245	.81	40
465	274	.81	35
428	273	.73	30
399	277	.66	25
374	281	.61	20
354	284	.57	15
335	290	.53	10
320	297	.49	5
304	304	.46	SL



NOTE
1. Fuel required at any point includes military thrust climb to cruise-climb altitude (if below that).
2. No allowance made for loiter, descent or landing.
3. Best cruise condition is determined by intersection of climb path guide line and lines of best range.
4. Cruise at recommended Mach No.

— — — FUEL REQUIRED
— — LINE OF BEST RANGE FOR CONSTANT ALTITUDE
——— LINE OF BEST RANGE FOR CRUISE-CLIMB ALTITUDE
——— CLIMB PATH GUIDE LINES
22,195B • • • • TIME AT CRUISING ALTITUDE

CRUISE—TWO—230 GALLON TANKS					
ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	ENGINE PRESSURE RATIO*	FUEL FLOW LB/HR*
CRUISE-CLIMB	.80	254	460	2.35	3130
35,000	.79	268	455	2.24	3080
30,000	.74	279	436	2.03	3210
25,000	.68	284	410	1.86	3360
20,000	.61	280	375	1.73	3490
15,000	.55	277	344	1.62	3600
10,000	.50	276	319	1.53	3700
5,000	.46	278	300	1.46	3800
SL	.43	281	281	1.39	3950

*APPROXIMATE

Figure A4-12

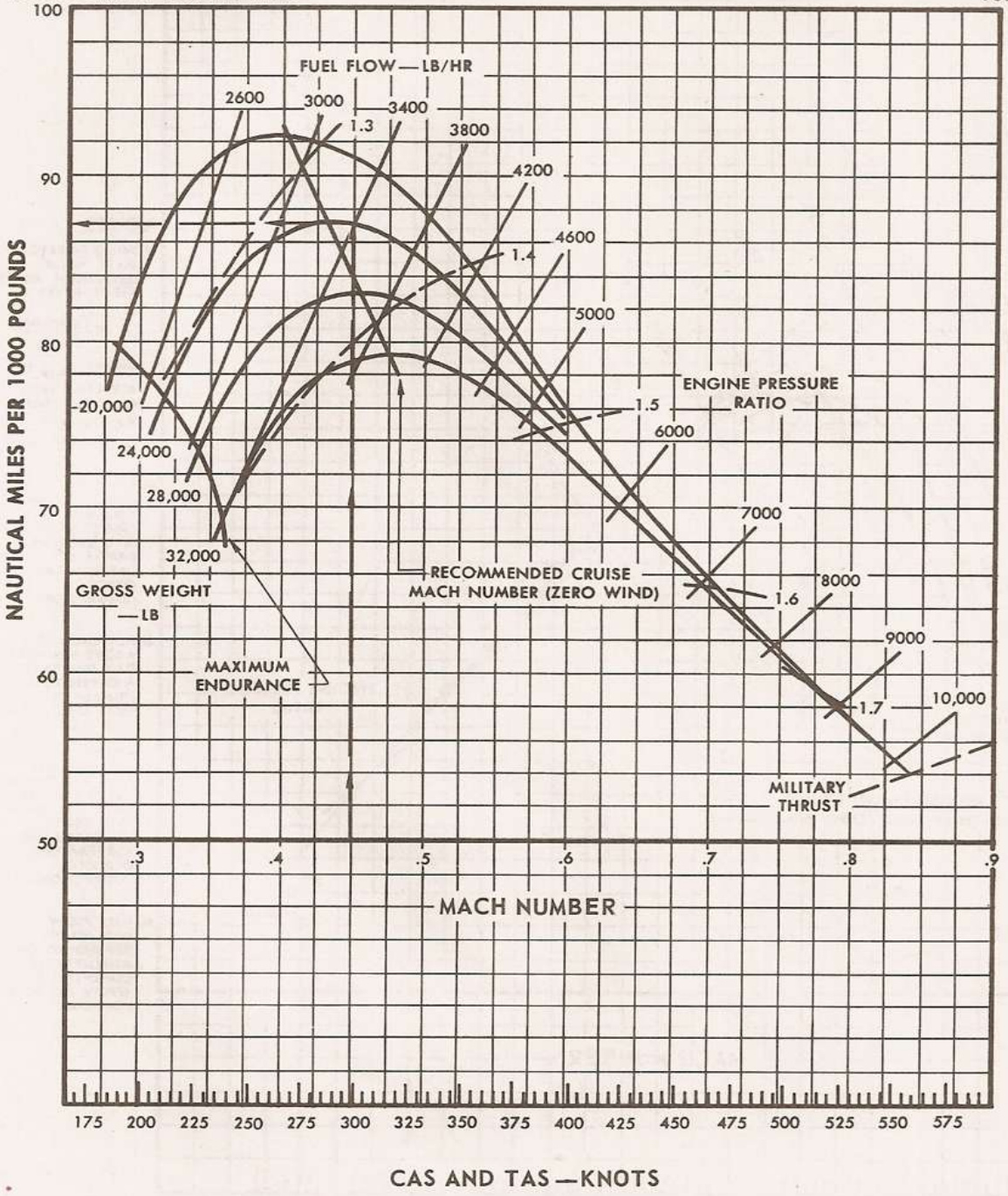
F

nautical miles per 1000 pounds — sea level

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE

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

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



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .02 AND DECREASE RANGE 16%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,029F

Figure A4-13

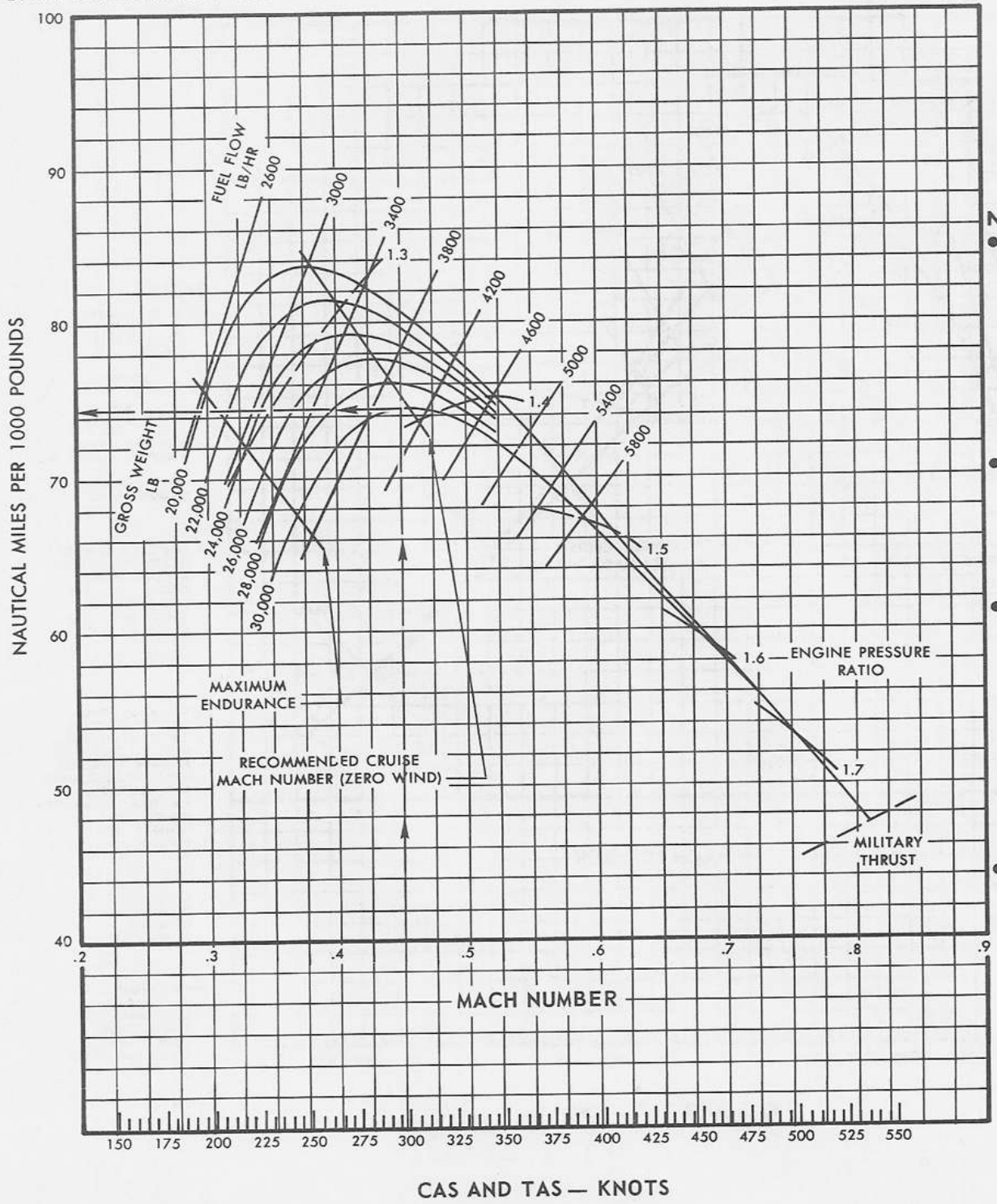
TF

nautical miles per 1000 pounds—sea level

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .025 AND DECREASE RANGE 17%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,196B

Figure A4-14

F

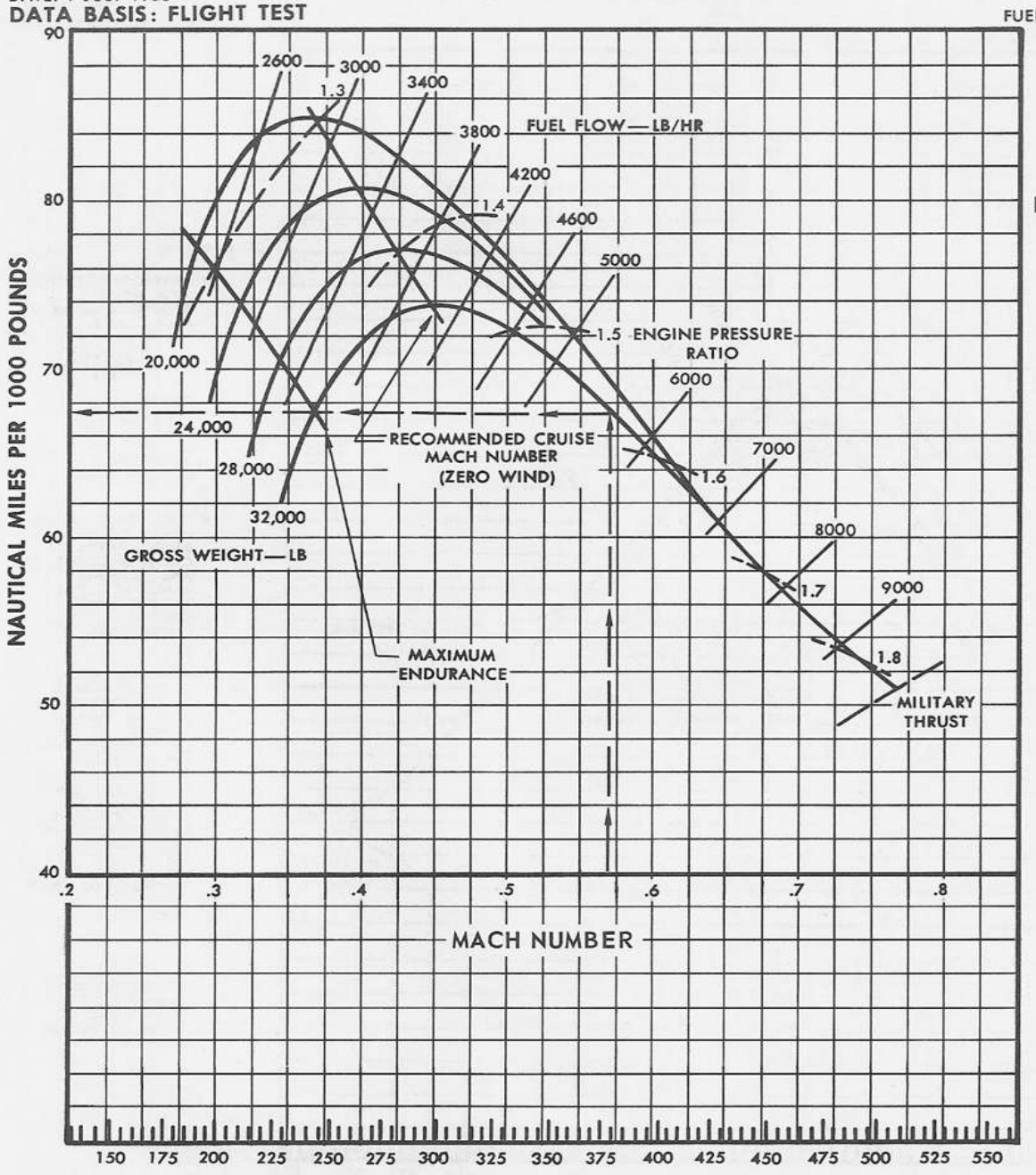
nautical miles per 1000 pounds - sea level

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS

STANDARD ATMOSPHERE

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

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



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 18%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22.038F

Figure A4-15

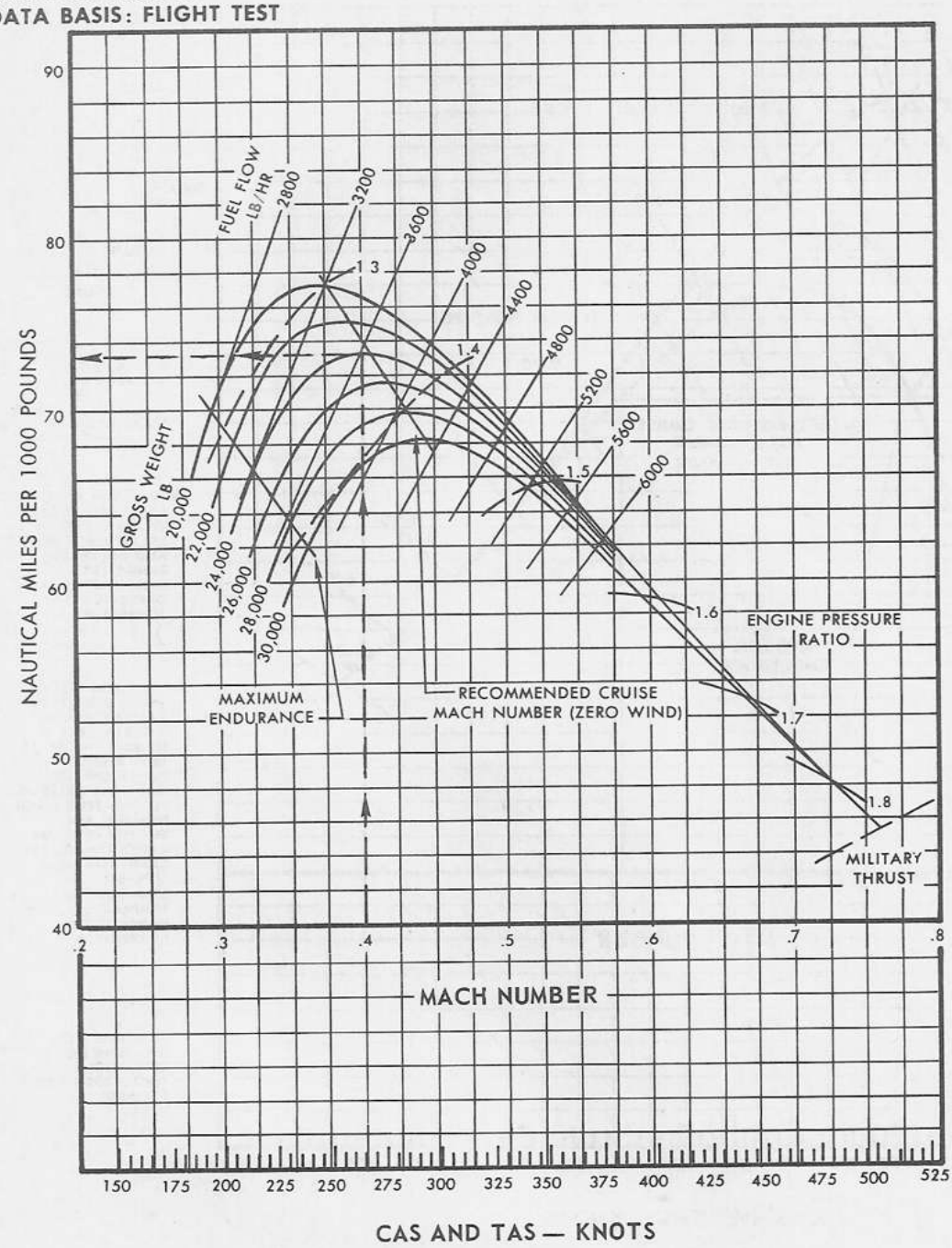
TF

nautical miles per 1000 pounds—sea level

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS • STANDARD ATMOSPHERE

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .025 AND DECREASE RANGE 17%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,197B

Figure A4-16

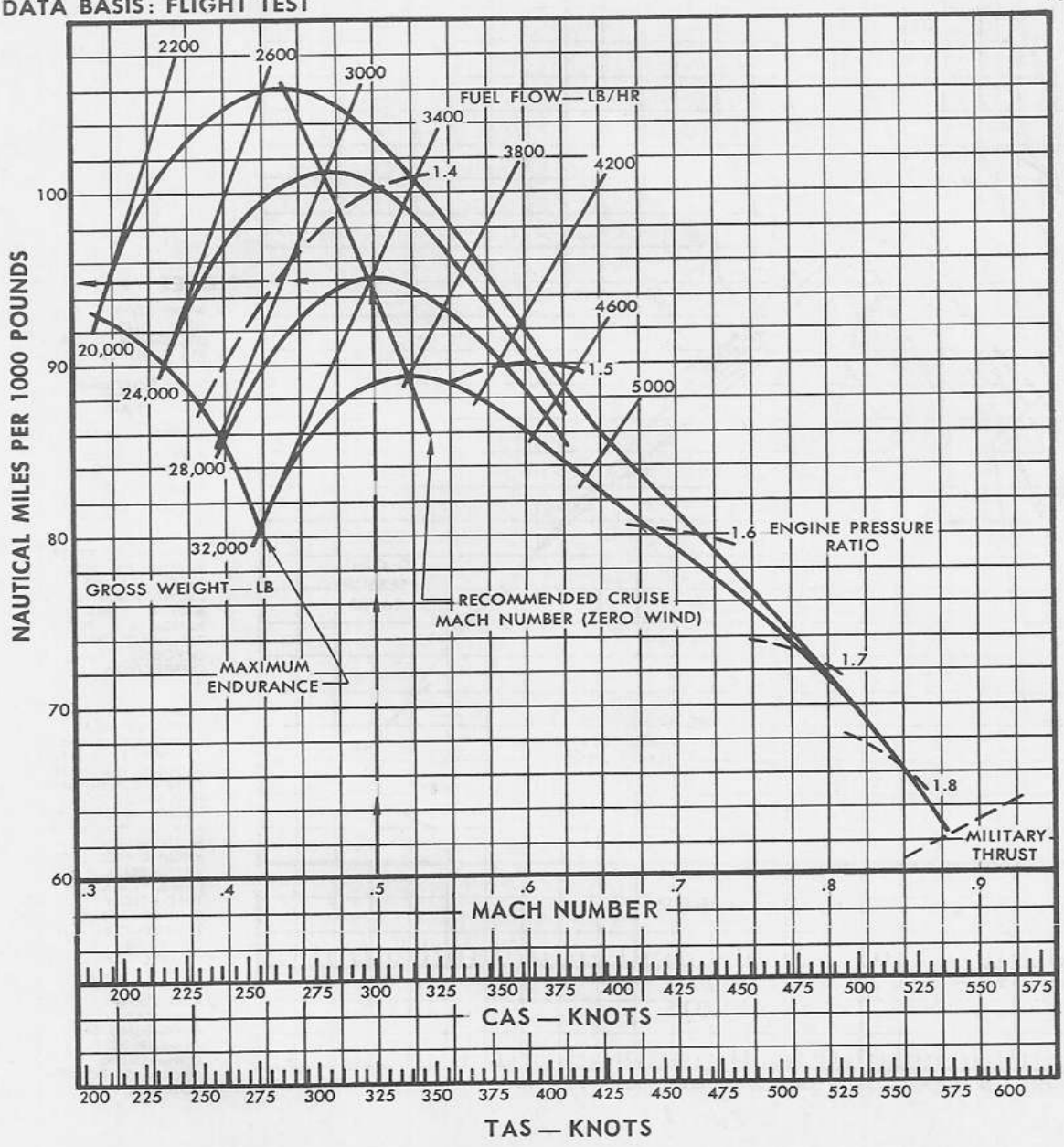
F

nautical miles per 1000 pounds — 5,000 feet

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

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



NOTE

- WIND CORRECTION TO N. MI/LB

  GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$

  WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.

- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 15% FOR TAILWIND APPLY OPPOSITE CORRECTION.

- FOR EACH 10° ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.

- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,030F

Figure A4-17

TF

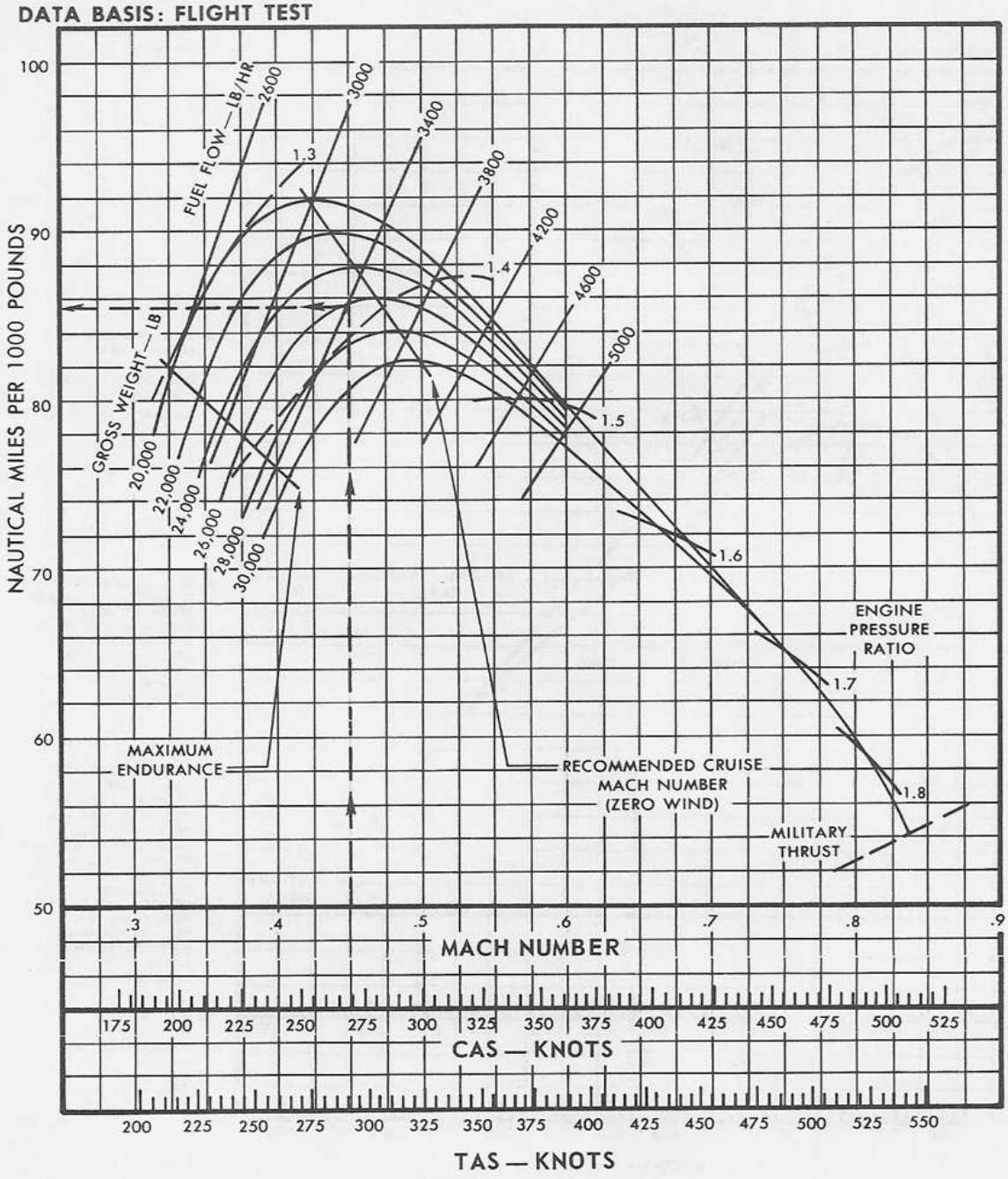
nautical miles per 1000 pounds—5000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

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

DATA BASIS: FLIGHT TEST



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .025 AND DECREASE RANGE 16%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,198B

Figure A4-18

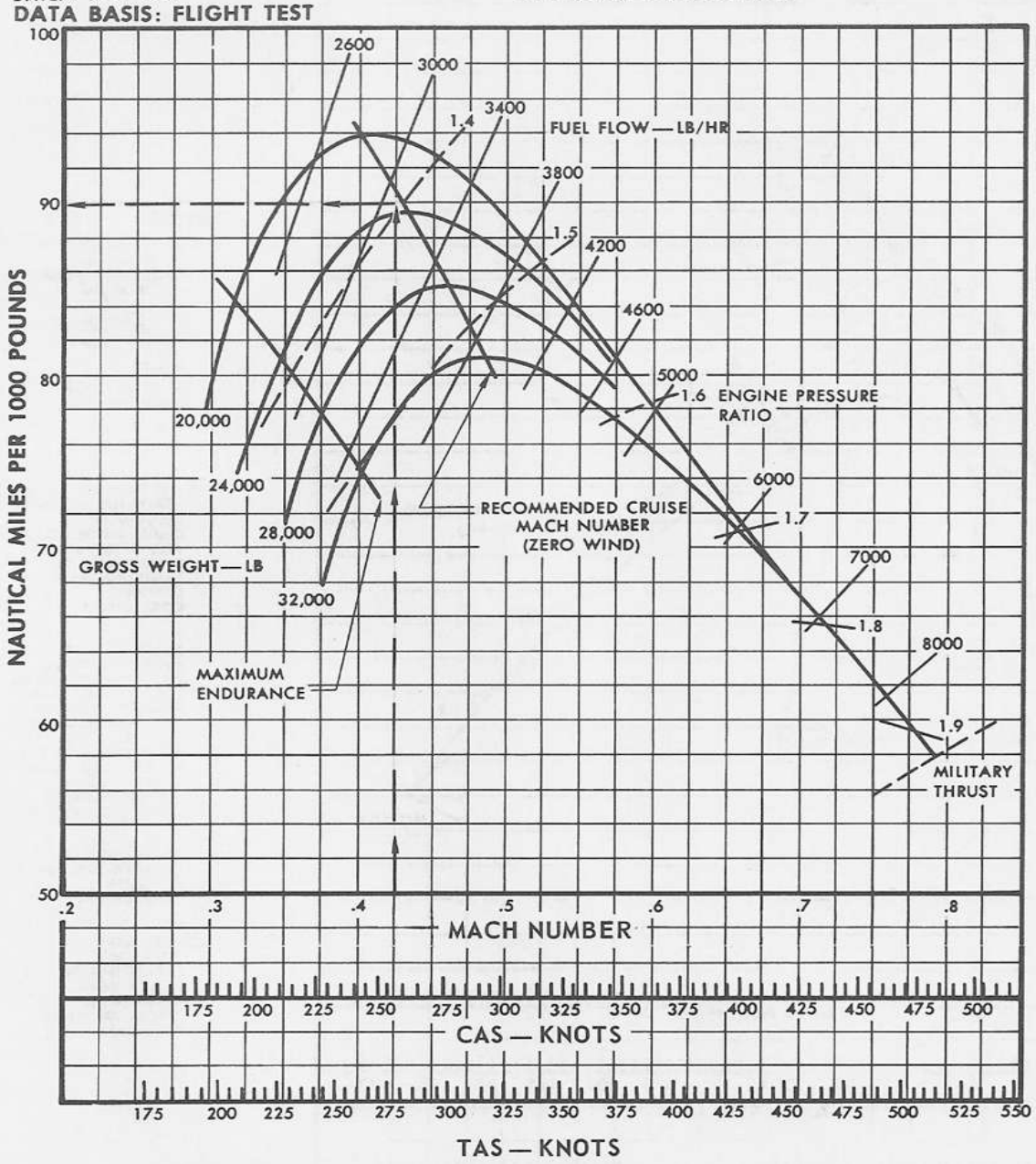
F

nautical miles per 1000 pounds — 5000 feet

CONFIGURATION: TWO 230 GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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

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



NOTE

- WIND CORRECTION TO N. MI/LB
  GROUND N. MI/LB = AIR N MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$
  WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 17%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS)

22,039F

Figure A4-19

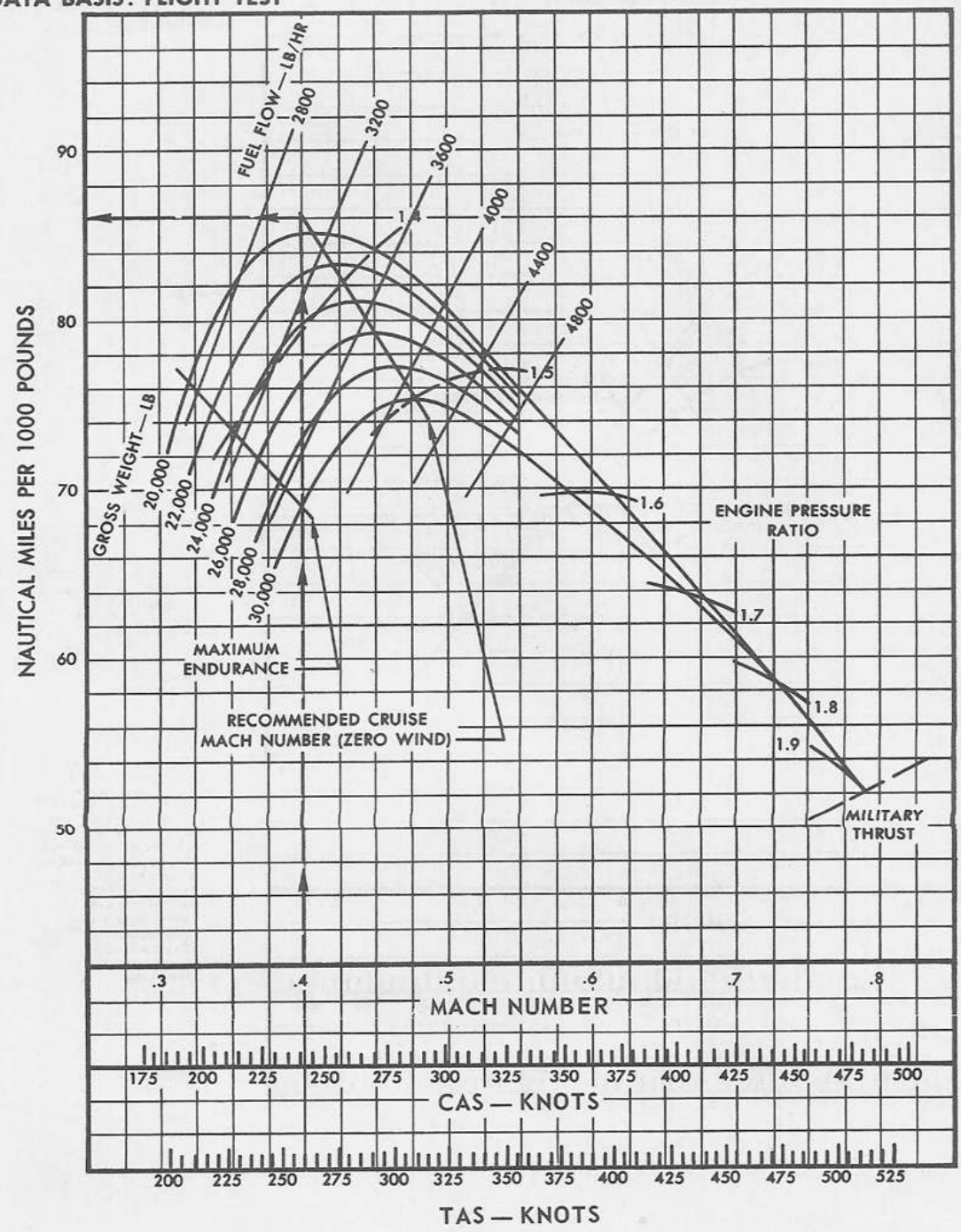
TF

nautical miles per 1000 pounds—5000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .025 AND DECREASE RANGE 16%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,199B

Figure A4-20

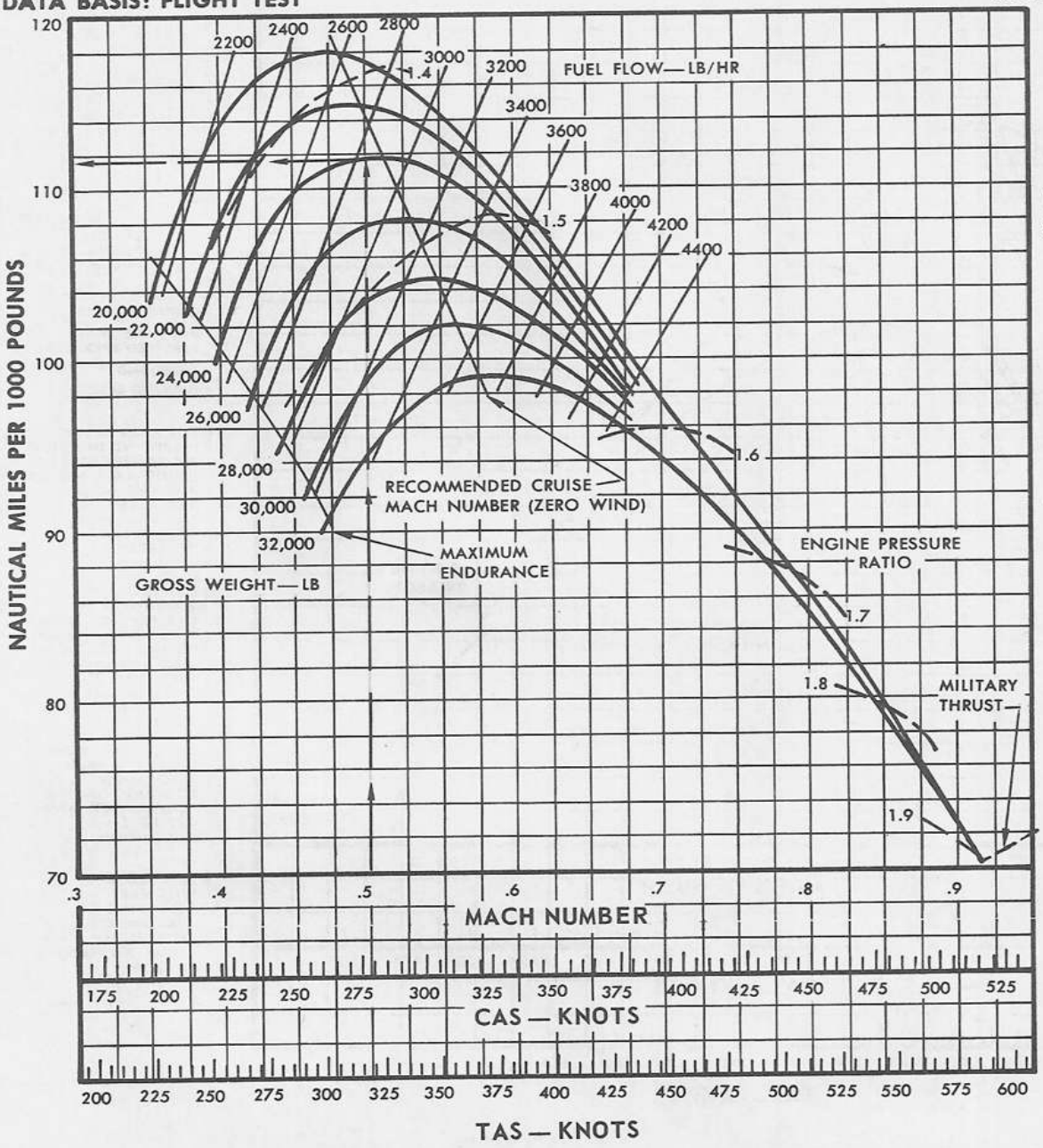
F

nautical miles per 1000 pounds — 10,000 feet

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

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



NOTE

- WIND CORRECTION TO N. MI/LB

  GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$

  WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.

- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .02 AND DECREASE RANGE 15% FOR TAILWIND APPLY OPPOSITE CORRECTION.

- FOR EACH 10° ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.

- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,031G

Figure A4-21

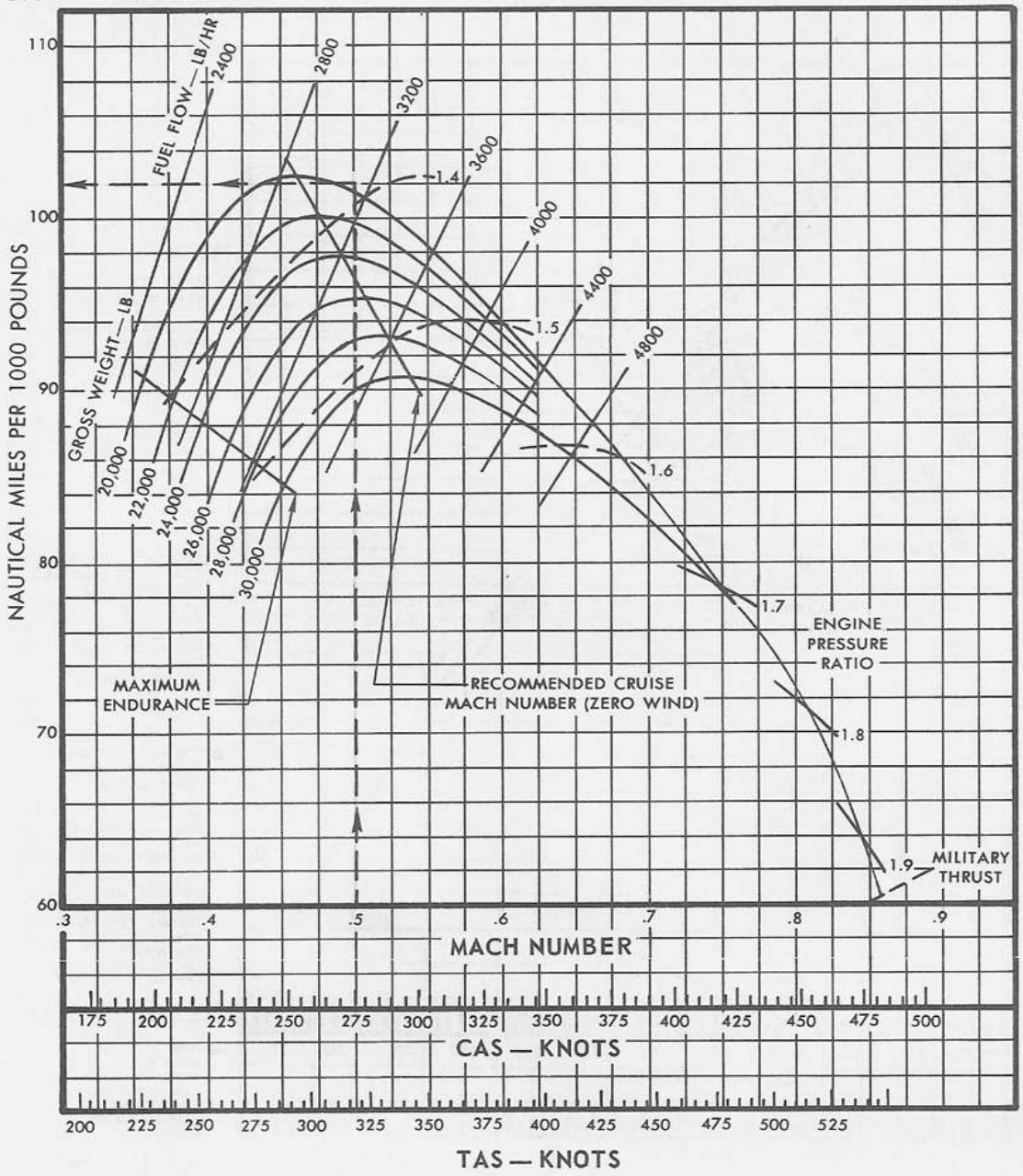
TF

nautical miles per 1000 pounds—10,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .025 AND DECREASE RANGE 15%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22.200B

Figure A4-22

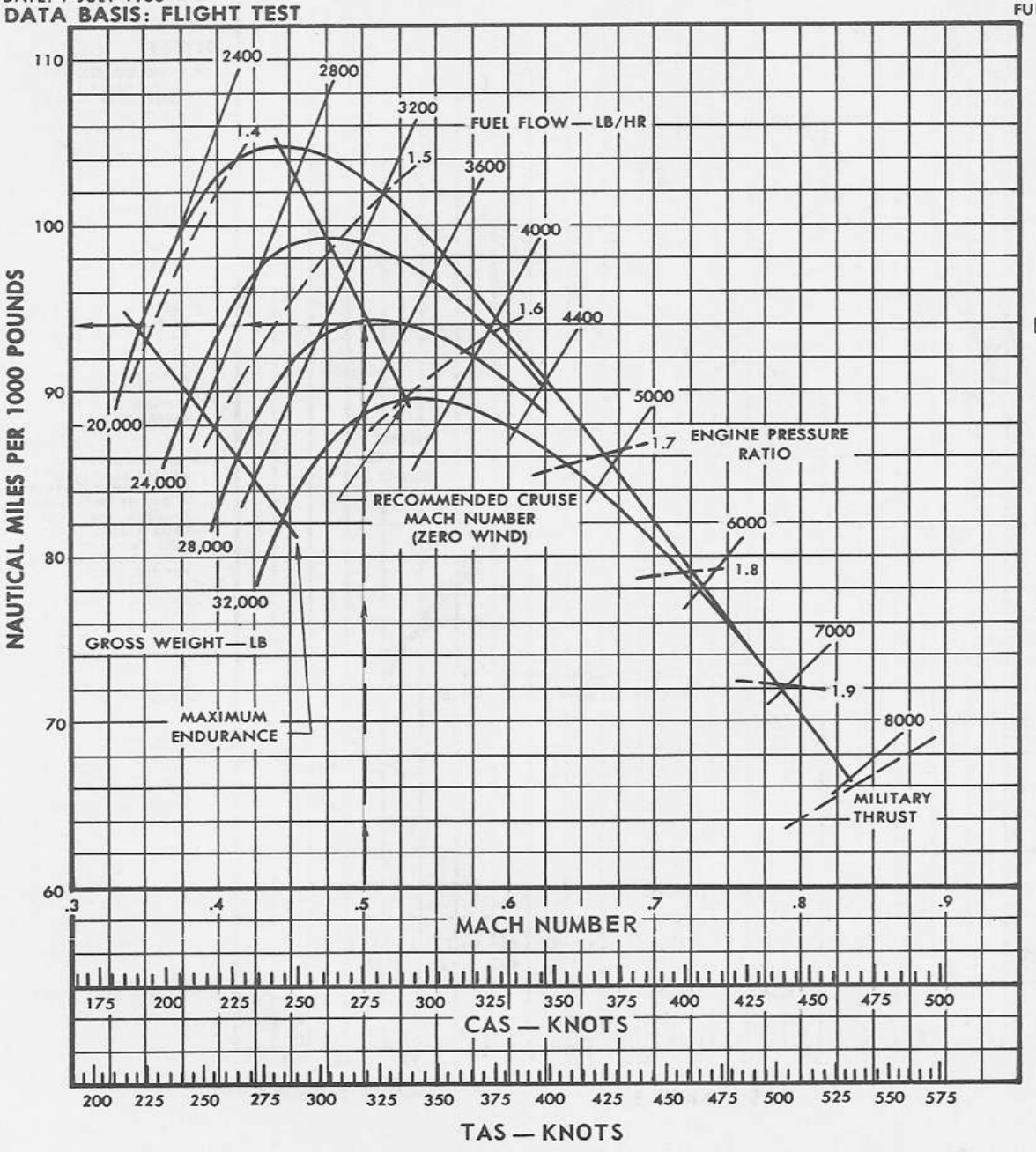
F

nautical miles per 1000 pounds — 10,000 feet

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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

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



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 16%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,040F

Figure A4-23

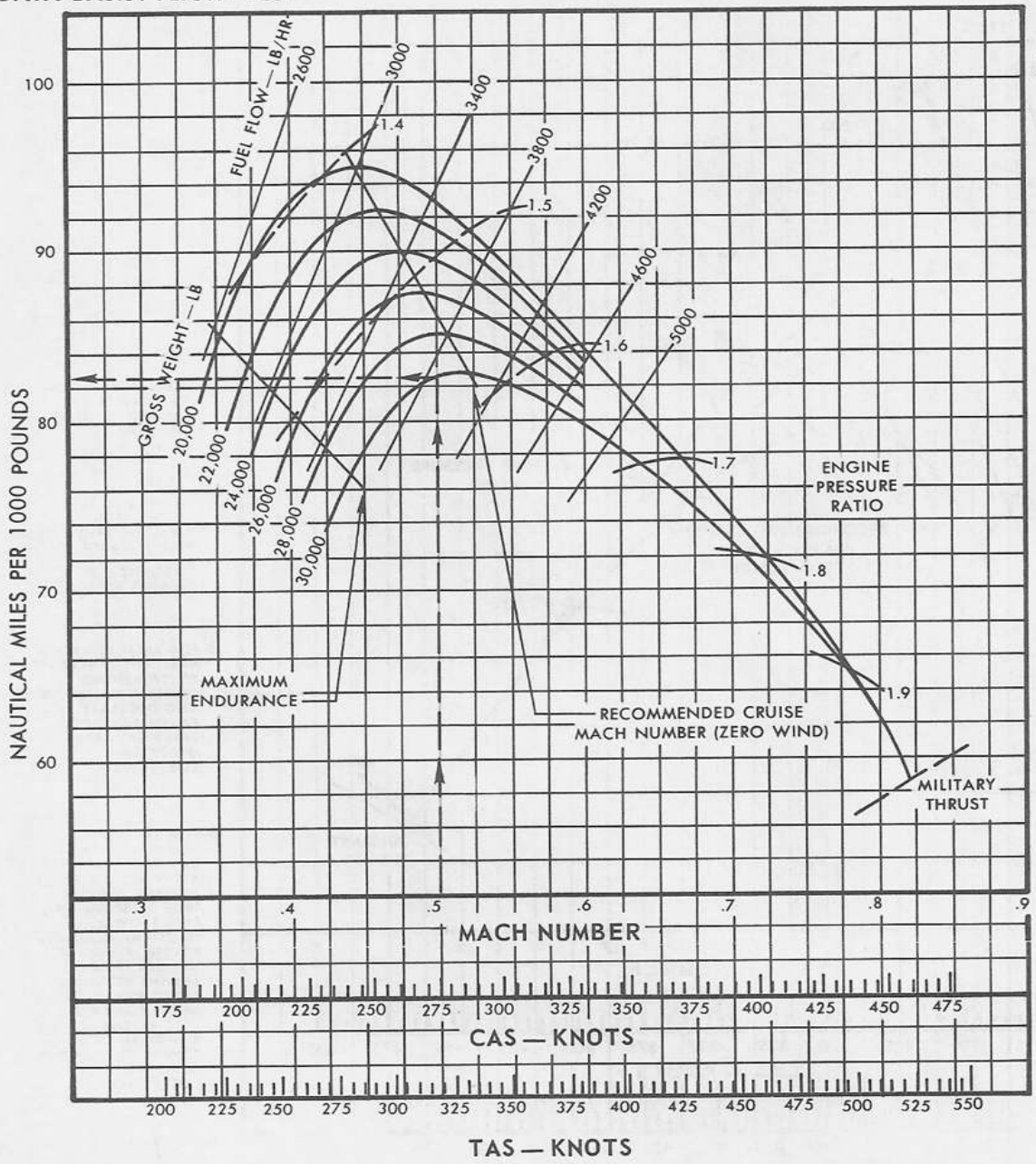
TF

nautical miles per 1000 pounds—10,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$
- WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED .020 CRUISE MACH NO. AND DECREASE RANGE 16%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,201B

Figure A4-24

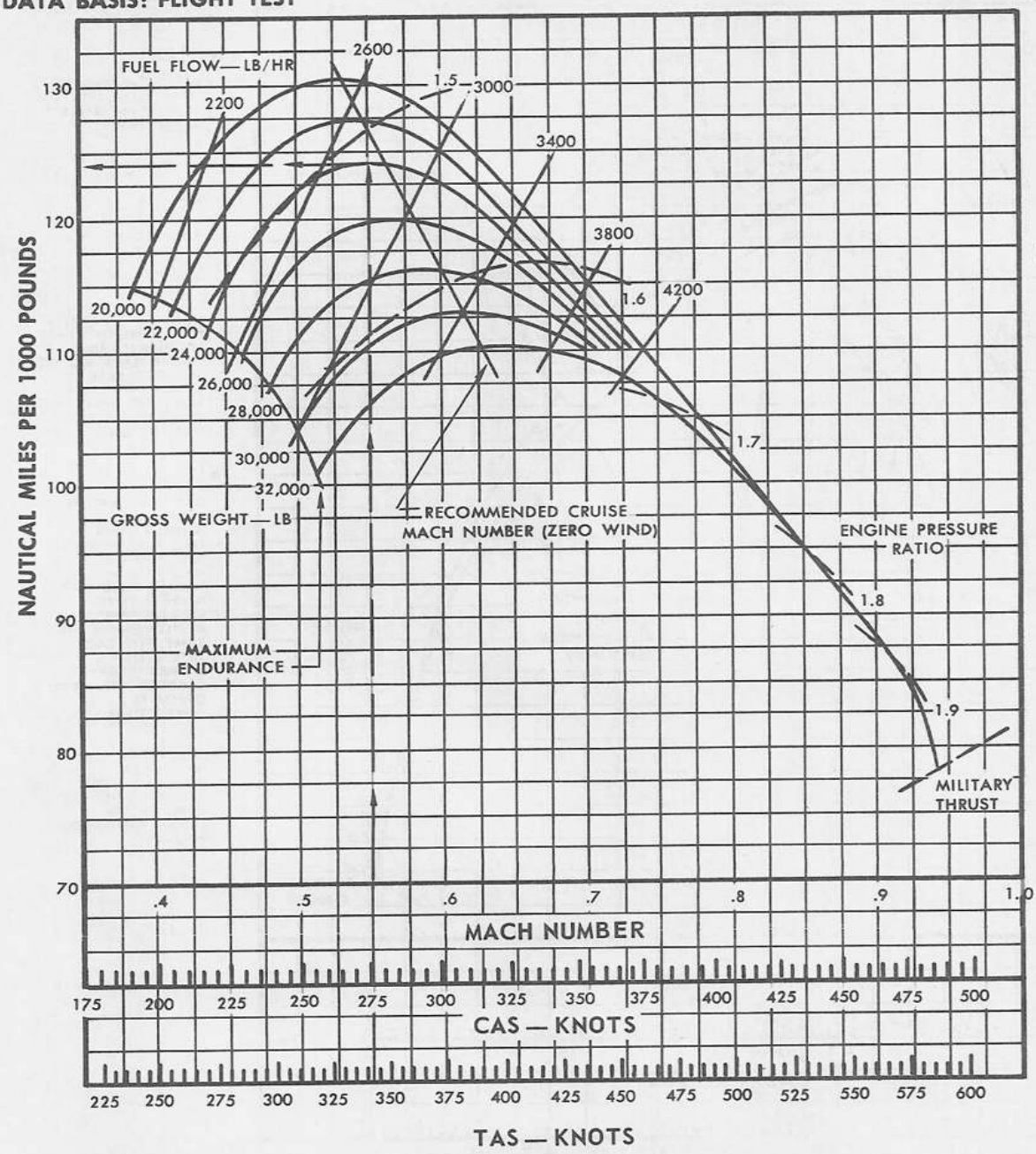
F

nautical miles per 1000 pounds — 15,000 feet

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

CONFIGURATION: CLEAN ● STANDARD ATMOSPHERE

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



NOTE

● WIND CORRECTION TO N. MI/LB

GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$

WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.

● FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .015 AND DECREASE RANGE 14% FOR TAILWIND APPLY OPPOSITE CORRECTION.

● FOR EACH 10° ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.

● FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,032G

Figure A4-25

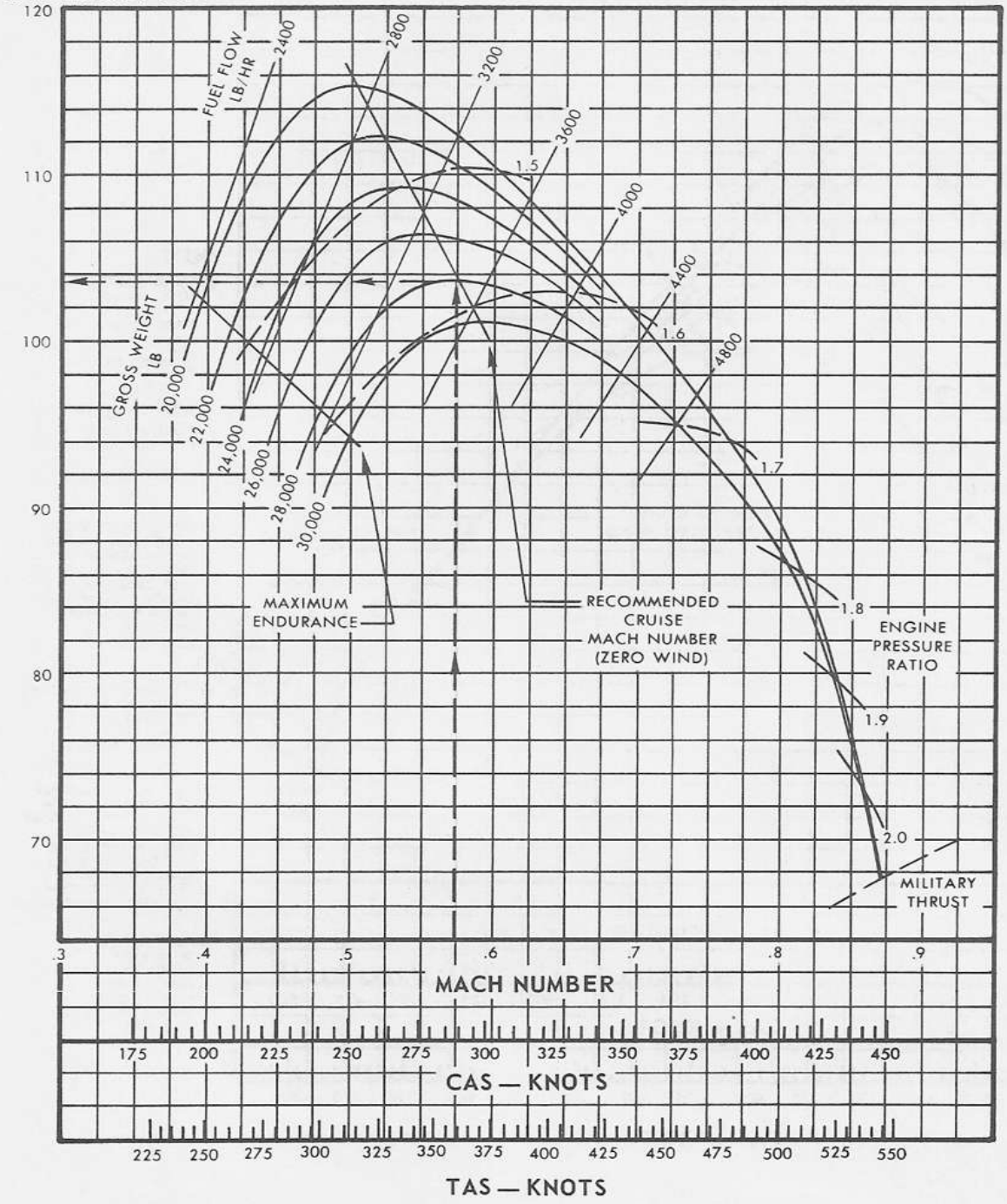
TF

nautical miles per 1000 pounds—15,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .020 AND DECREASE RANGE 14%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,202B

Figure A4-26

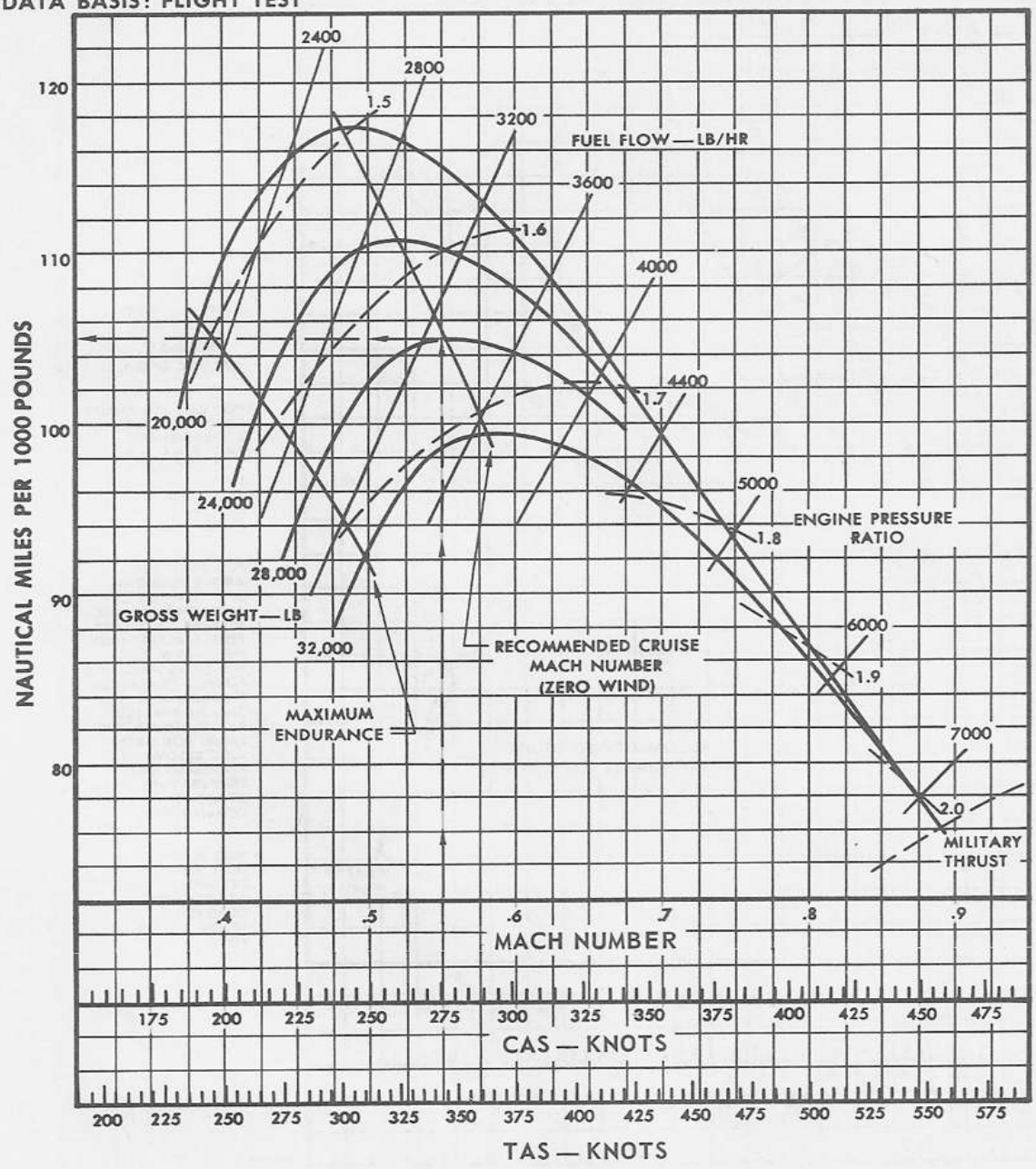
F

nautical miles per 1000 pounds—15,000 feet

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

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 15%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,041F

Figure A4-27

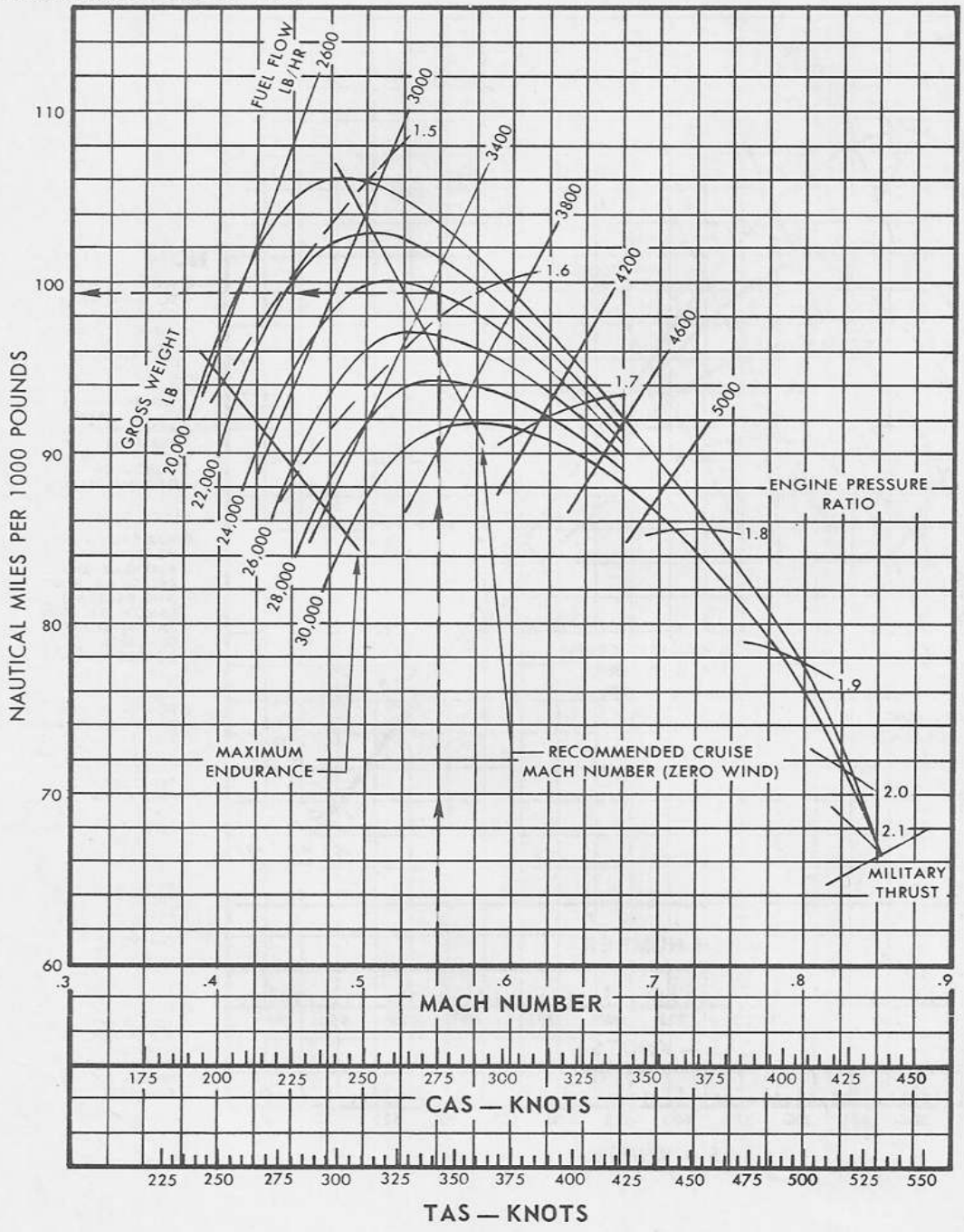
TF

nautical miles per 1000 pounds—15,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .020 AND DECREASE RANGE 14%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,203B

Figure A4-28

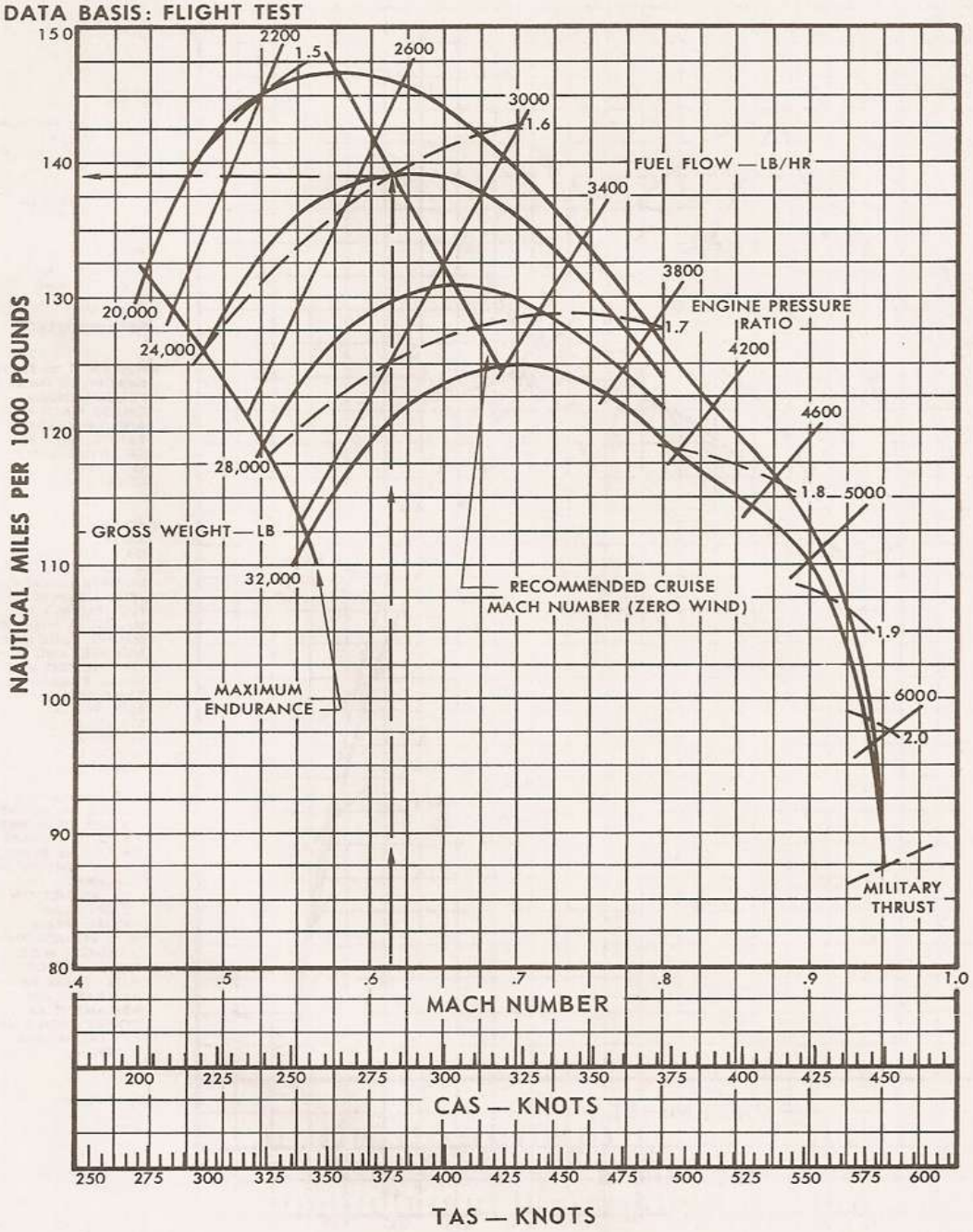
F

nautical miles per 1000 pounds — 20,000 feet

CONFIGURATION: CLEAN ● STANDARD ATMOSPHERE

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

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



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 11%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 40% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22.033G

Figure A4-29

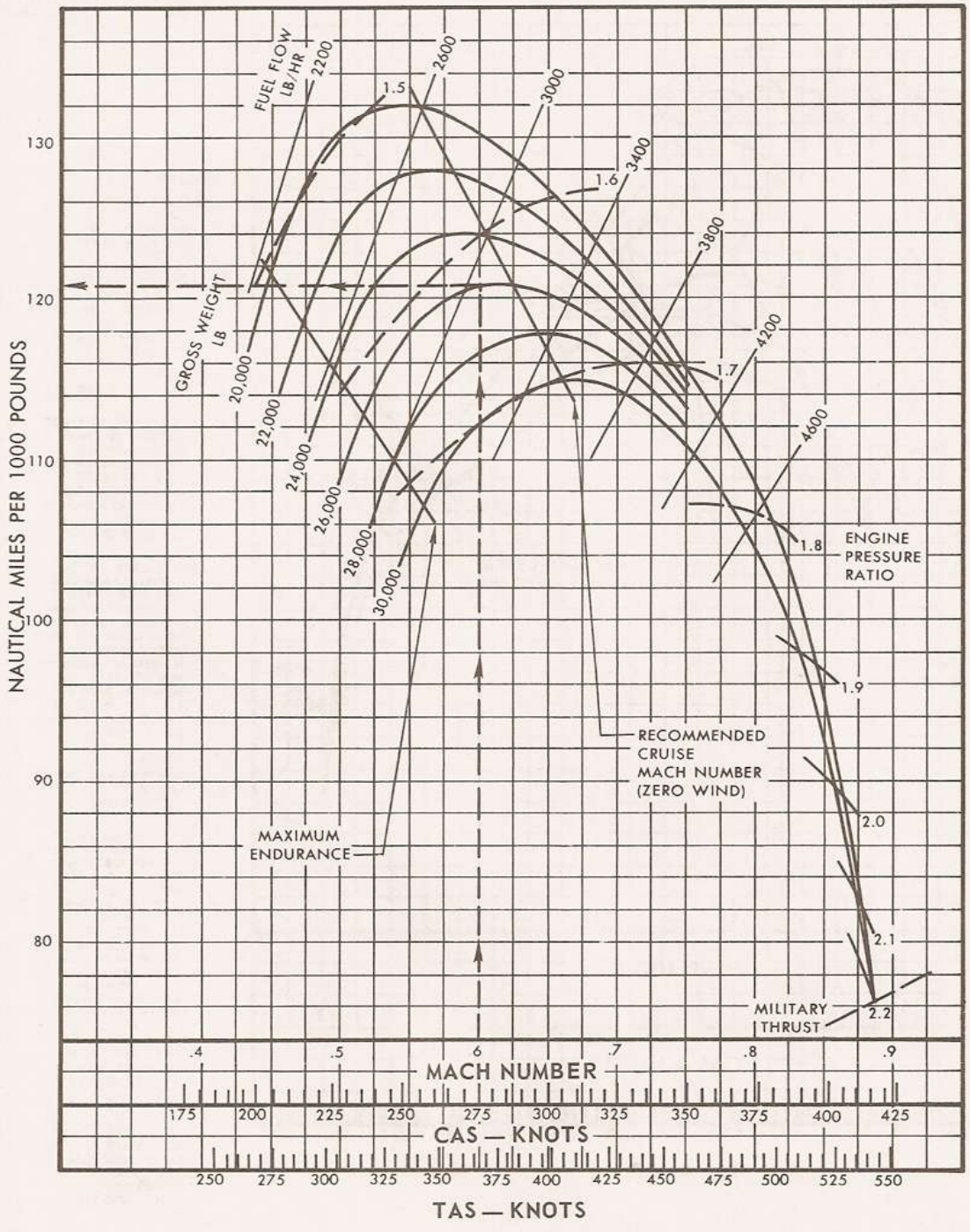
TF

nautical miles per 1000 pounds—20,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .020 AND DECREASE RANGE 13%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 40% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,204B

Figure A4-30

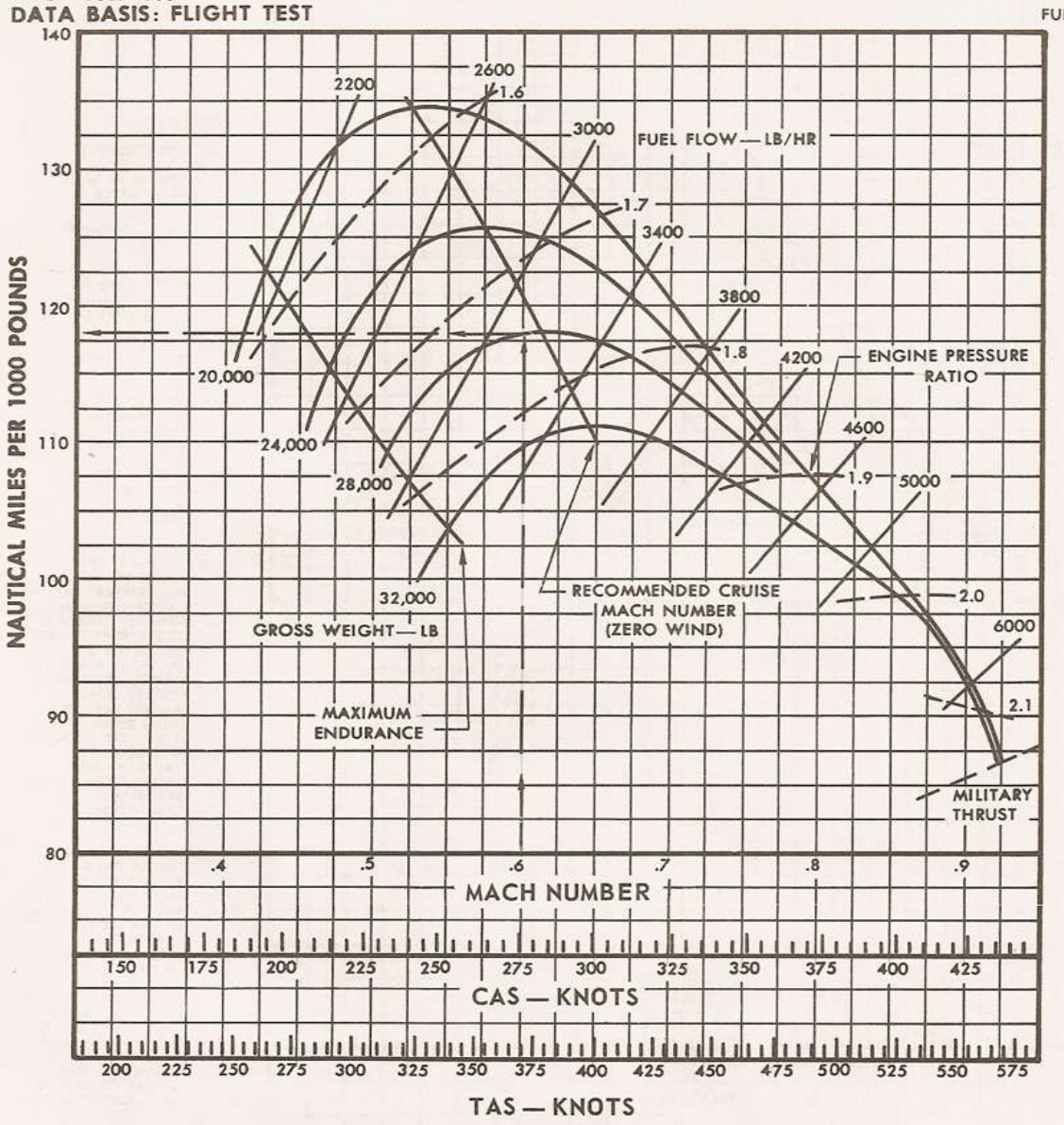
F

nautical miles per 1000 pounds—20,000 feet

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS • STANDARD ATMOSPHERE

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

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



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 14%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 37% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,042G

Figure A4-31

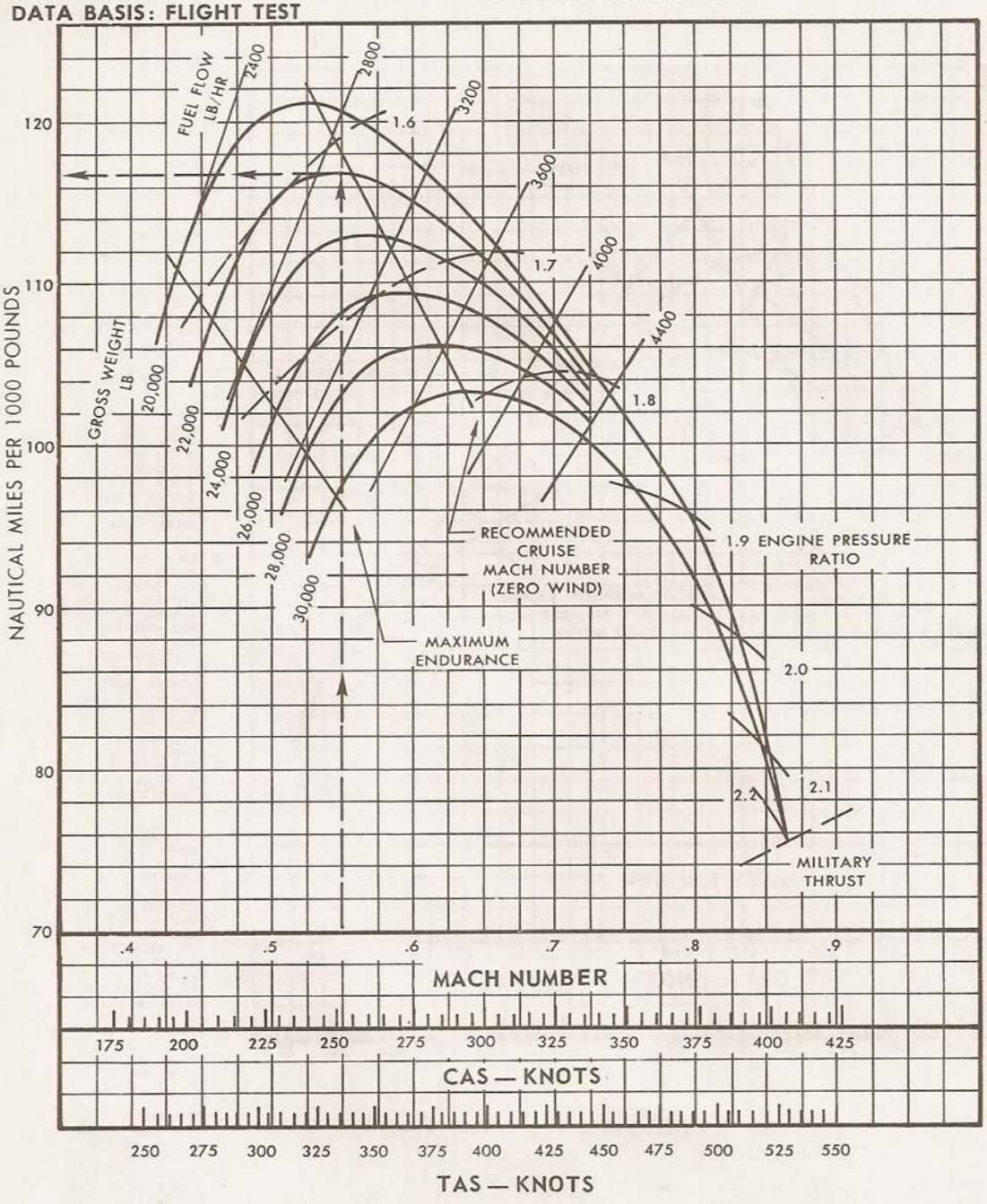
TF

nautical miles per 1000 pounds—20,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .020 AND DECREASE RANGE 13%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 39% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,205B

Figure A4-32

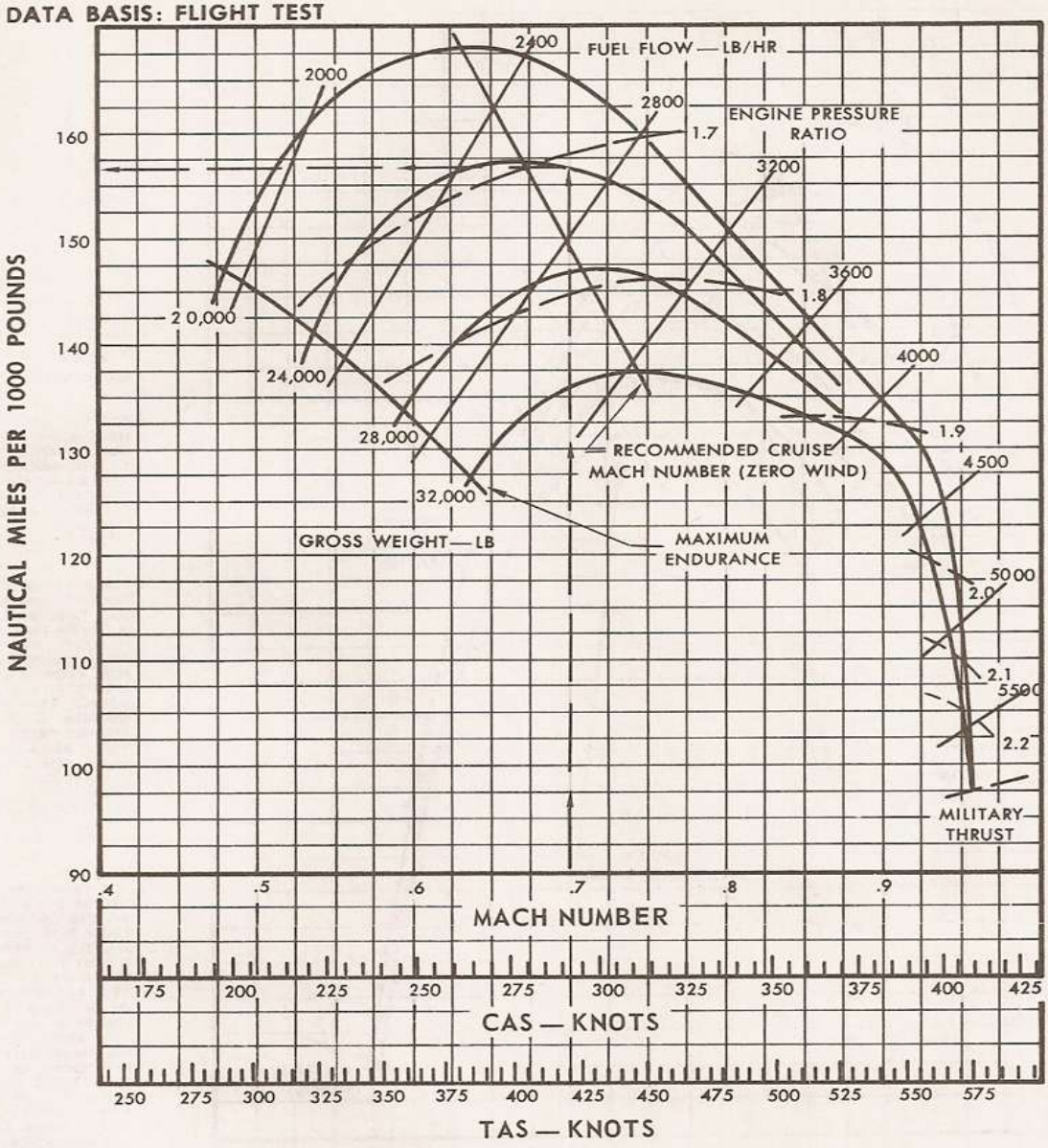
F

nautical miles per 1000 pounds—25,000 feet

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

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

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



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 12%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 40% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,034G

Figure A4-33

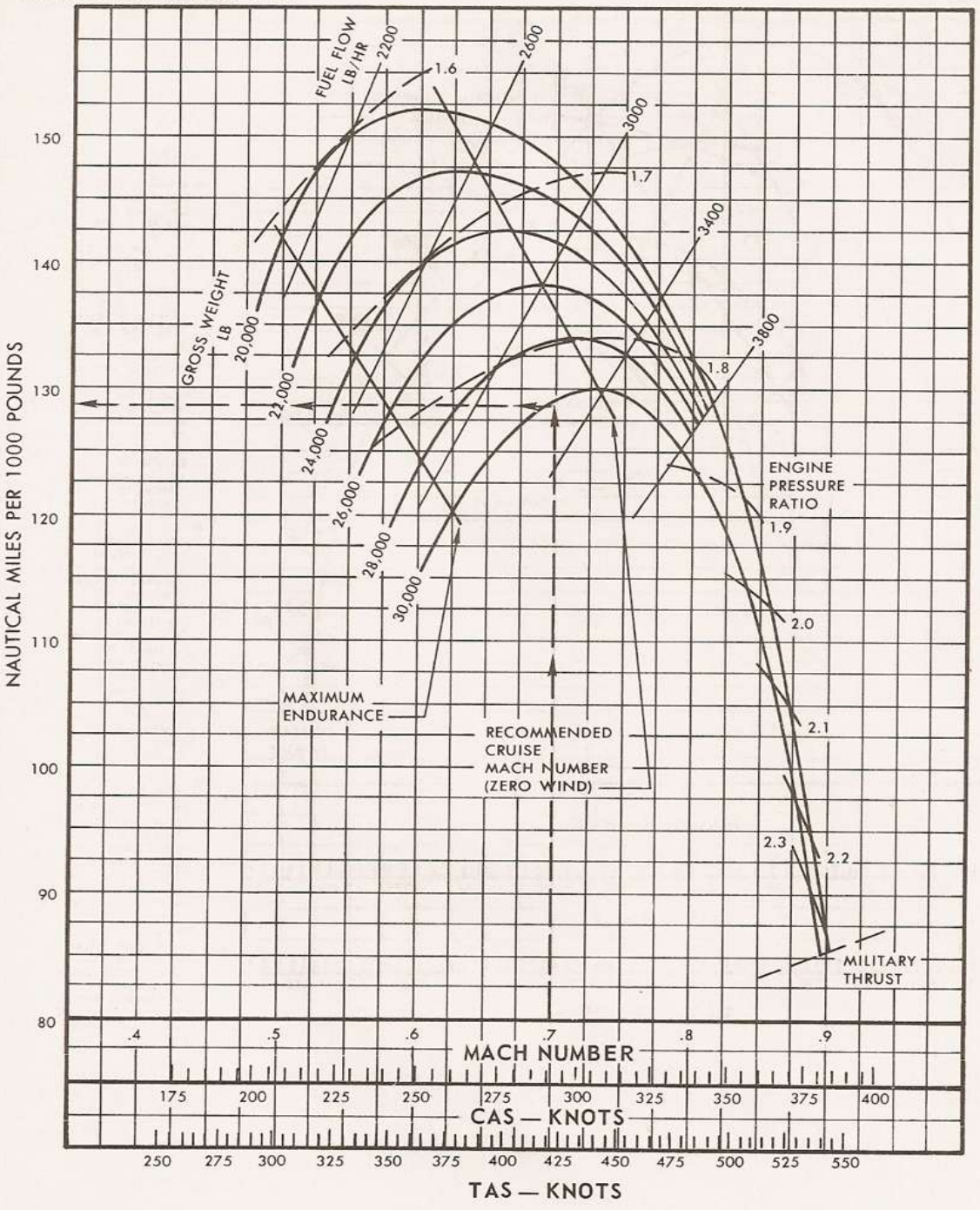
TF

nautical miles per 1000 pounds—25,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .015 AND DECREASE RANGE 12%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 40% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,206B

Figure A4-34

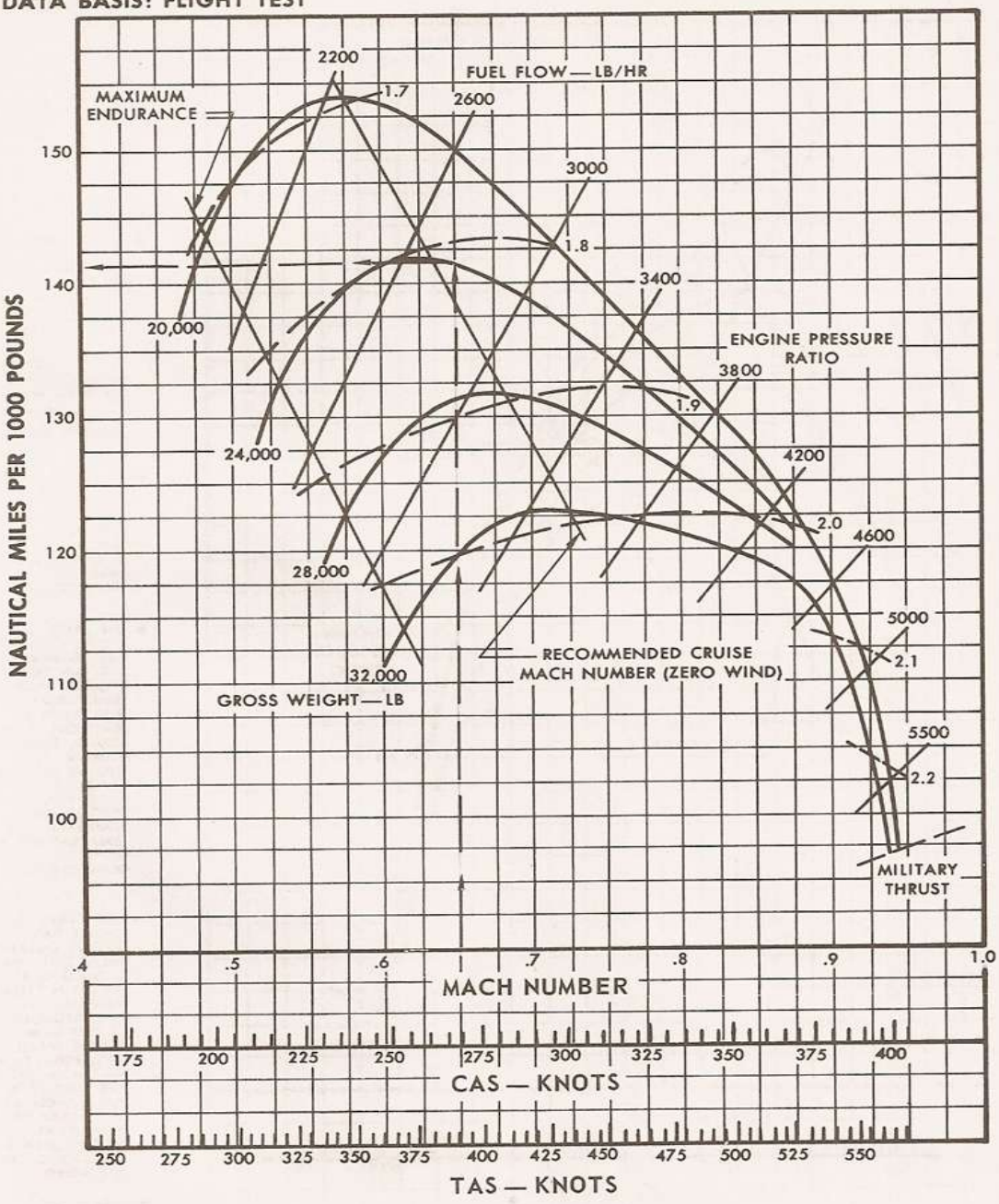
F

nautical miles per 1000 pounds — 25,000 feet

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

CONFIGURATION: TWO-230 GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 13%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 37% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,043G

Figure A4-35

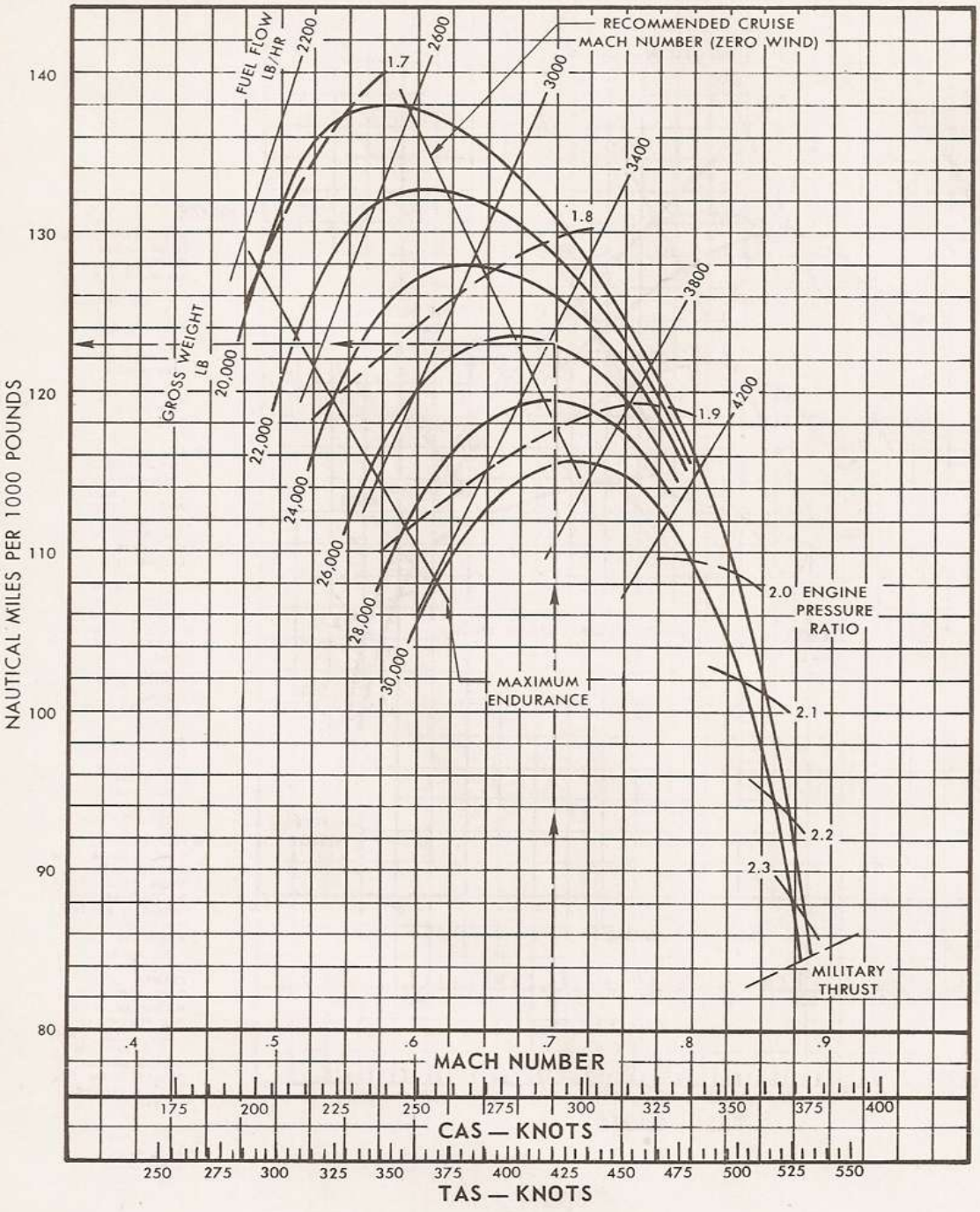
TF

nautical miles per 1000 pounds—25,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .020 AND DECREASE RANGE 12%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 39% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,207B

Figure A4-36

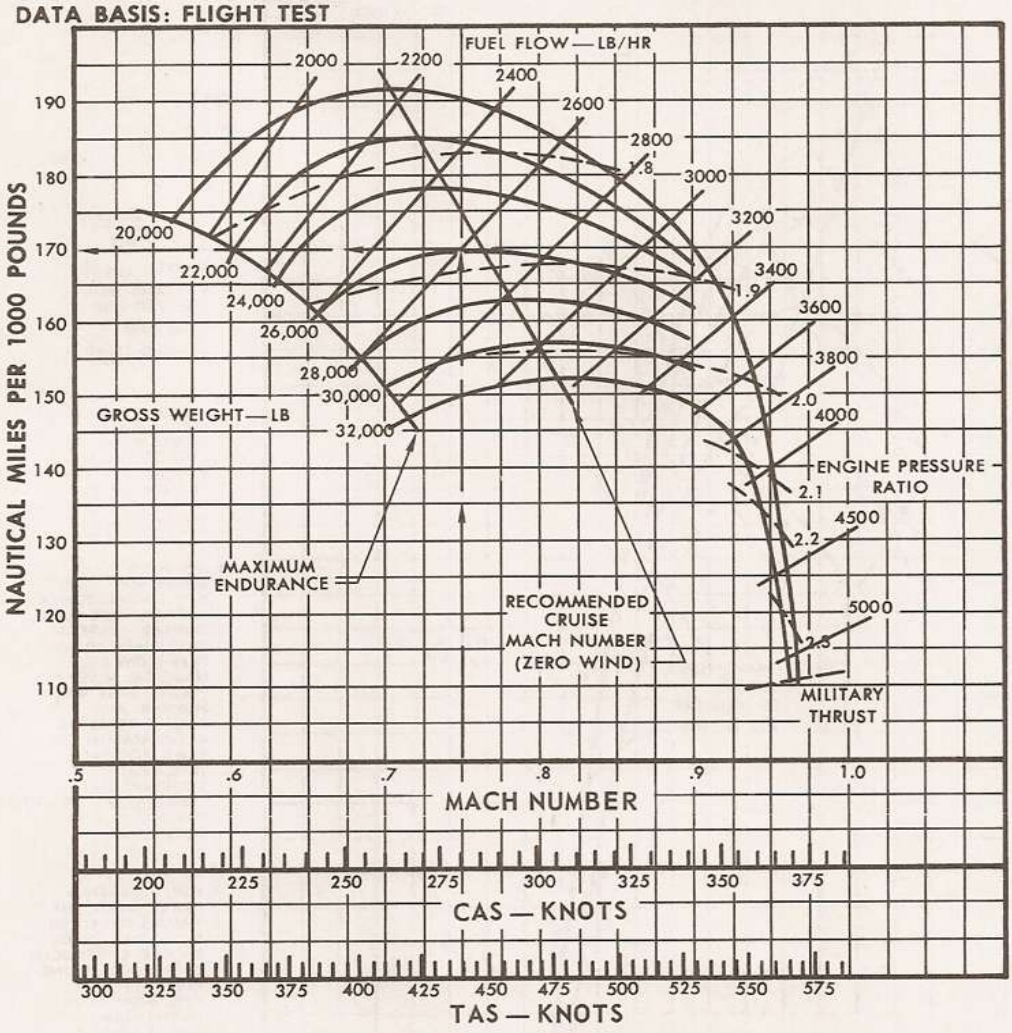
F

nautical miles per 1000 pounds — 30,000 feet

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

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

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



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 12%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 40% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,035G

Figure A4-37

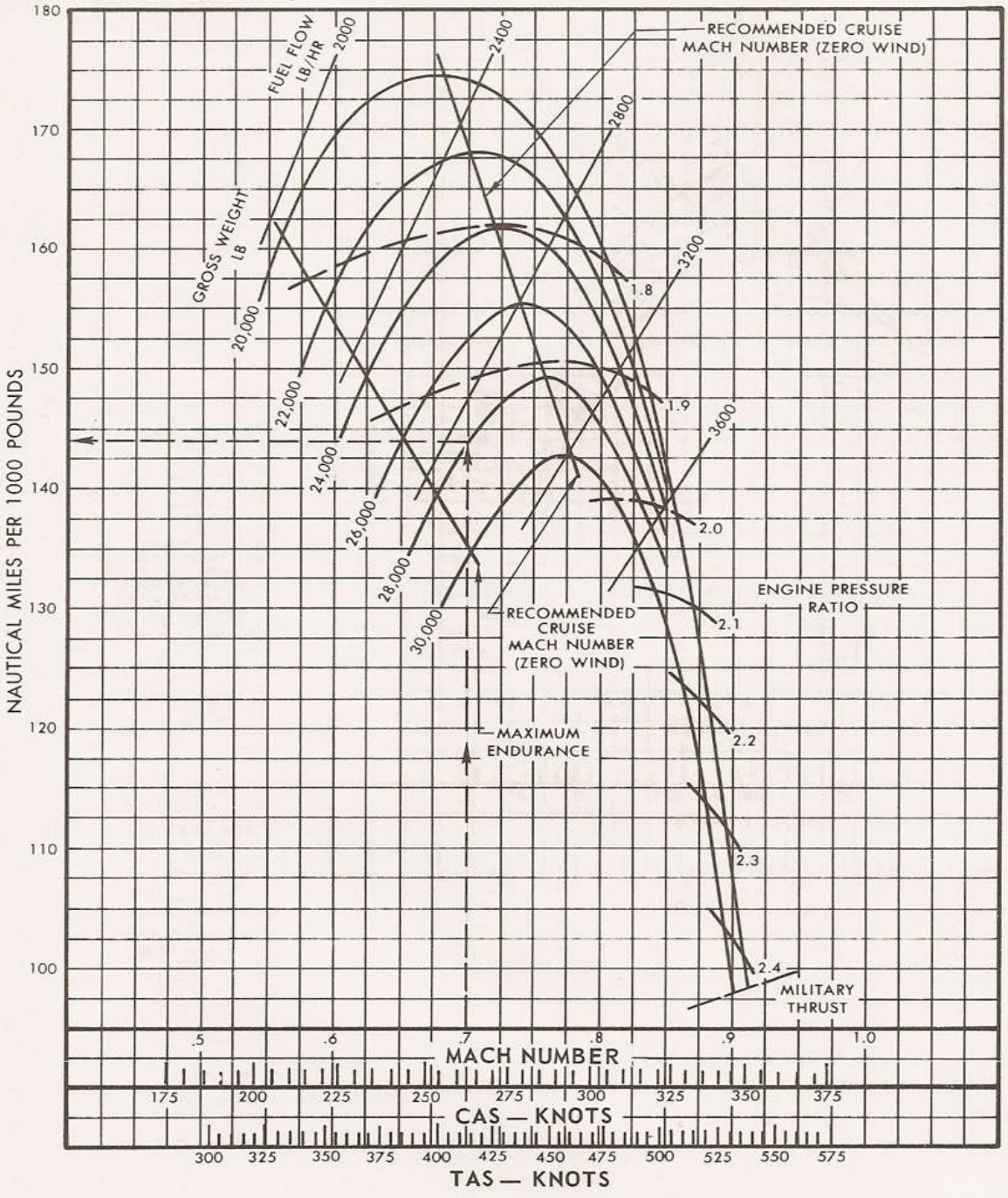
TF

nautical miles per 1000 pounds—30,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .015, AND DECREASE RANGE 11%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
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- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,208B

Figure A4-38

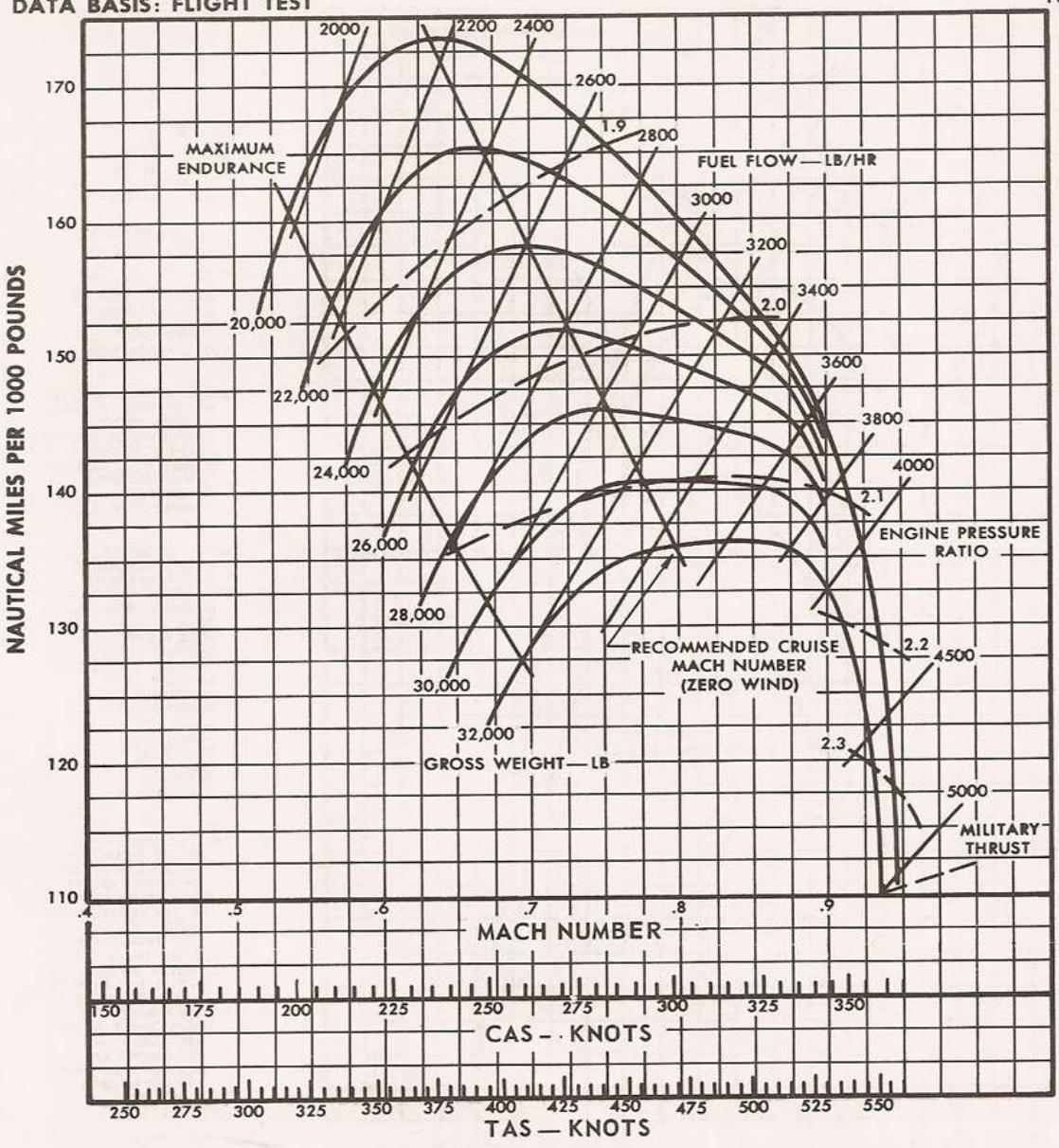
F

nautical miles per 1000 pounds—30,000 feet

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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

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



22,044G

NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. 01 AND DECREASE RANGE 12%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 37% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

Figure A4-39

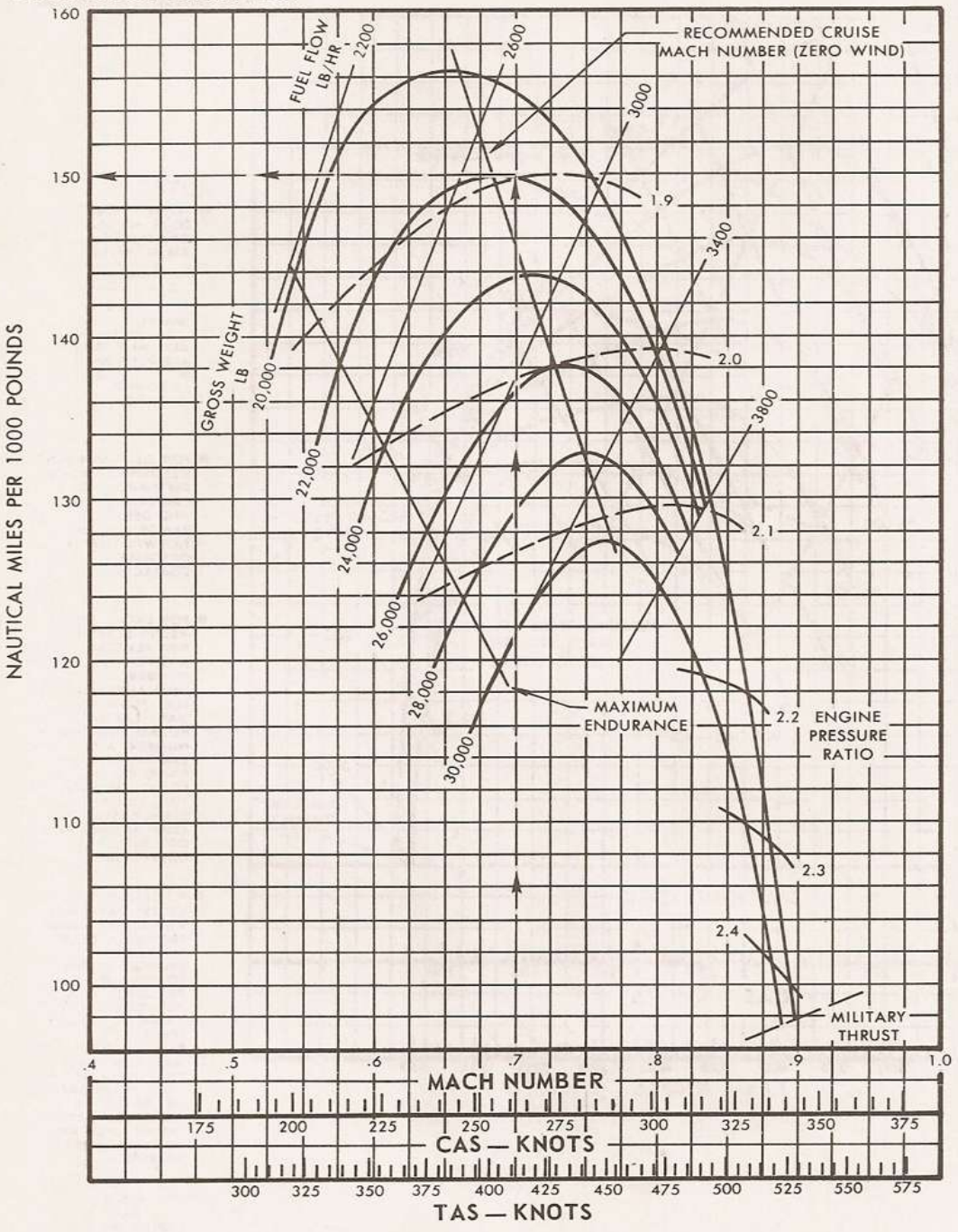
TF

nautical miles per 1000 pounds—30,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
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- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,209B

Figure A4-40

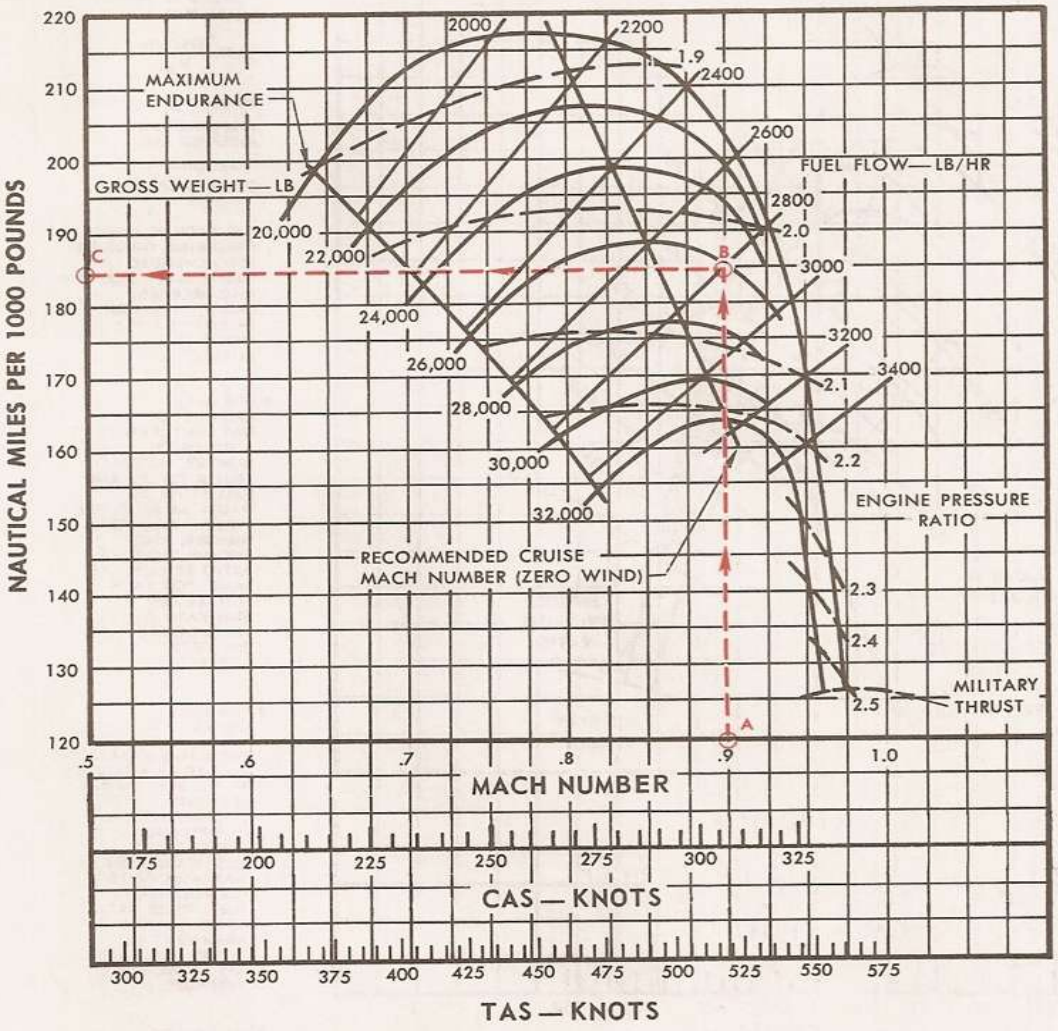
F

nautical miles per 1000 pounds — 35,000 feet

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

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



NOTE

- WIND CORRECTION TO N. MI/LB

  GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$

  WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.

- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 10% FOR TAILWIND APPLY OPPOSITE CORRECTION.

- FOR EACH 10° ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.

- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 40% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.

- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,036I

Figure A4-41

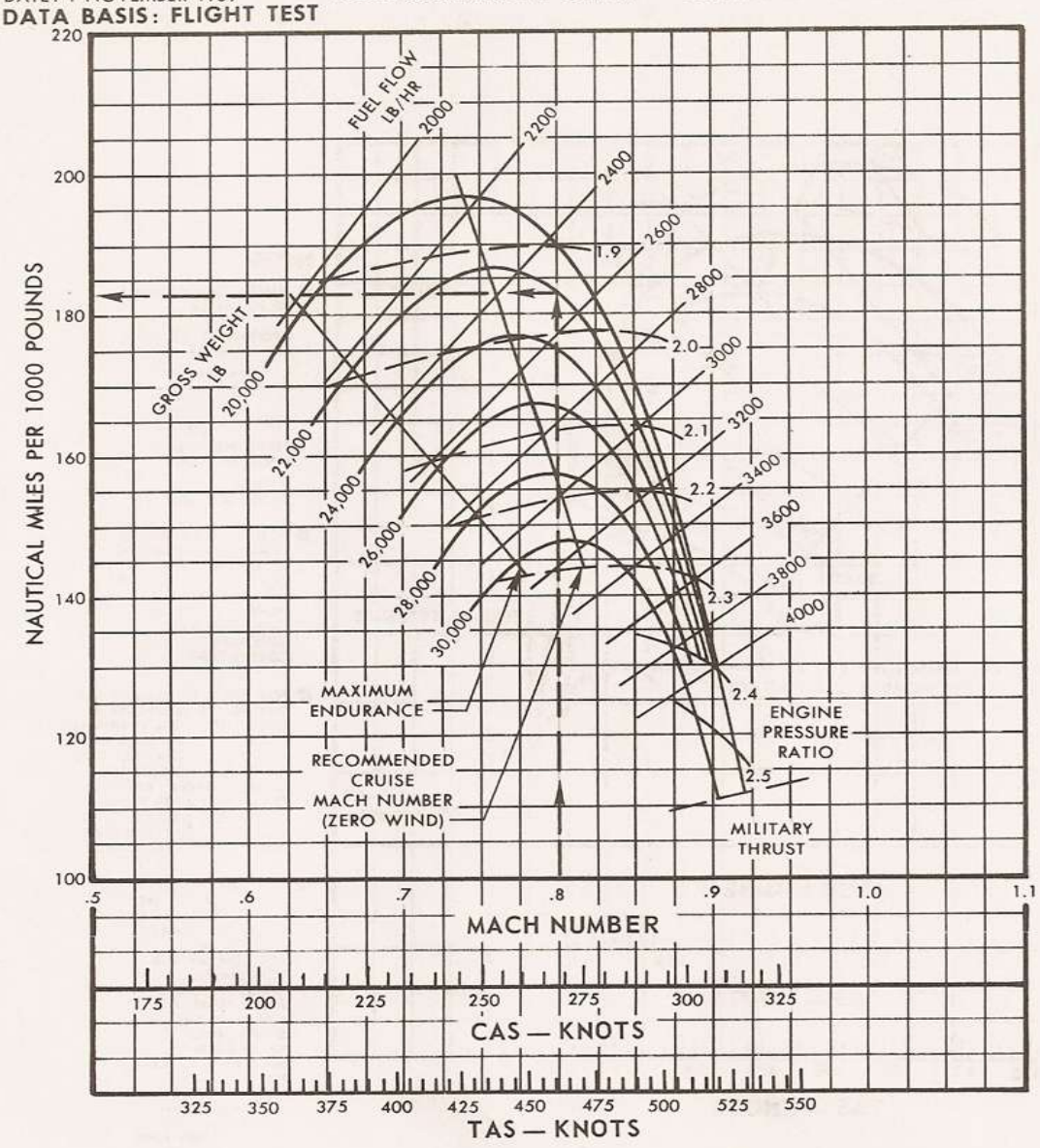
TF

nautical miles per 1000 pounds—35,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$

  WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.

- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .010 AND DECREASE RANGE 11%. FOR TAILWIND APPLY OPPOSITE CORRECTION.

- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.

- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 40% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.

- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS.

22,210B

Figure A4-42

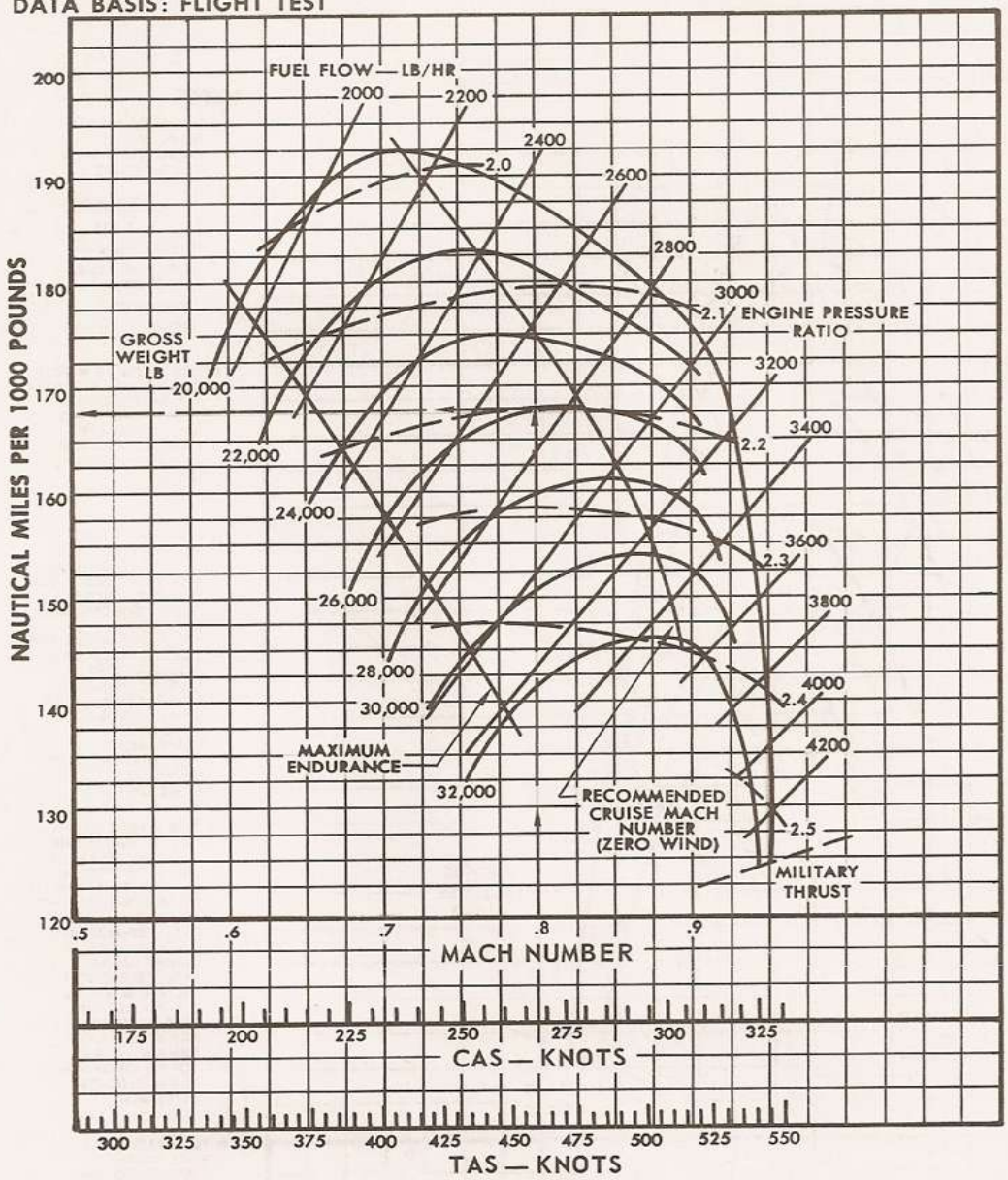
F

nautical miles per 1000 pounds — 35,000 feet

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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

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



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
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- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,045G

Figure A4-43

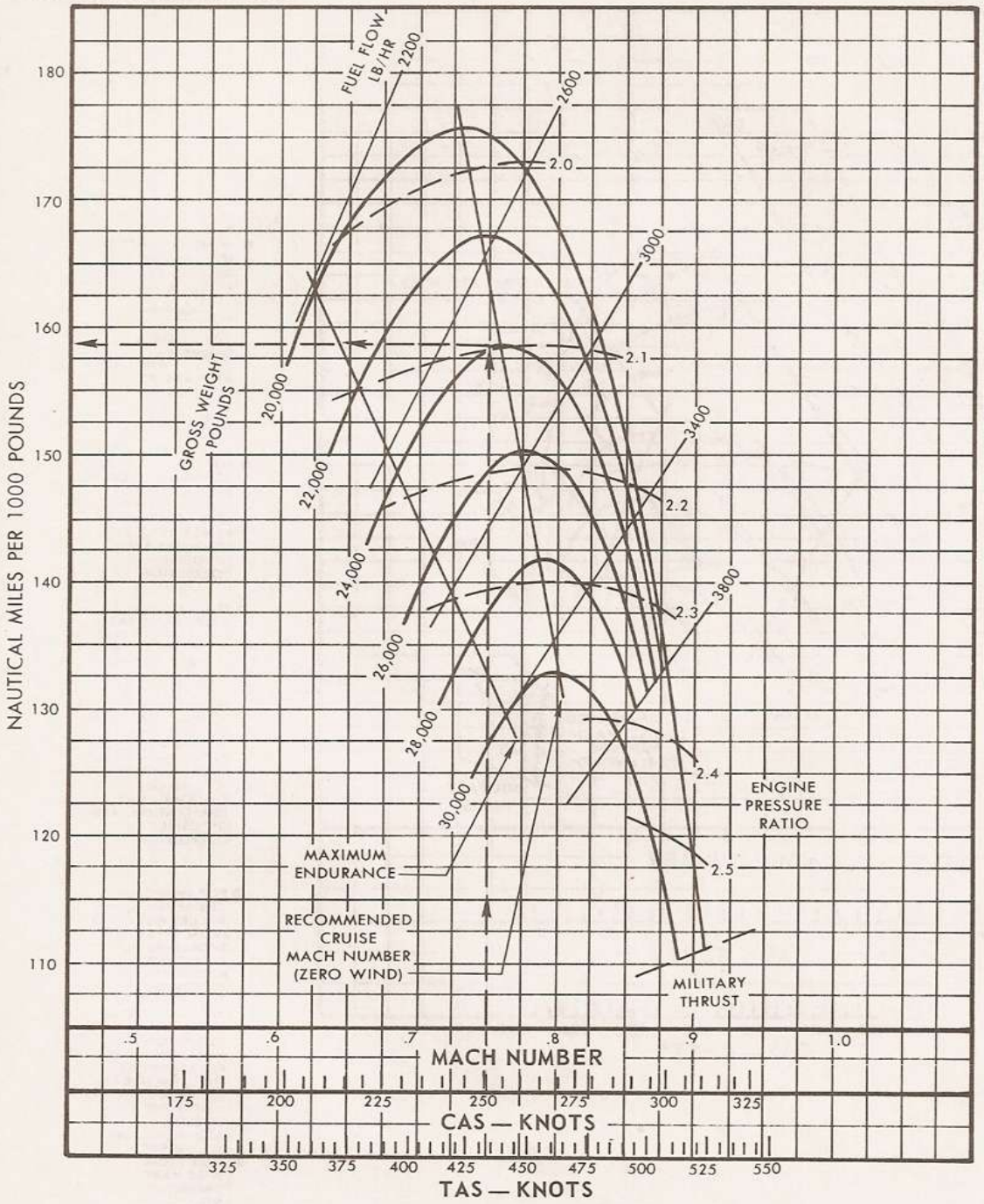
TF

nautical miles per 1000 pounds—35,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



22,211B

NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .015 AND DECREASE RANGE 11%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
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- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

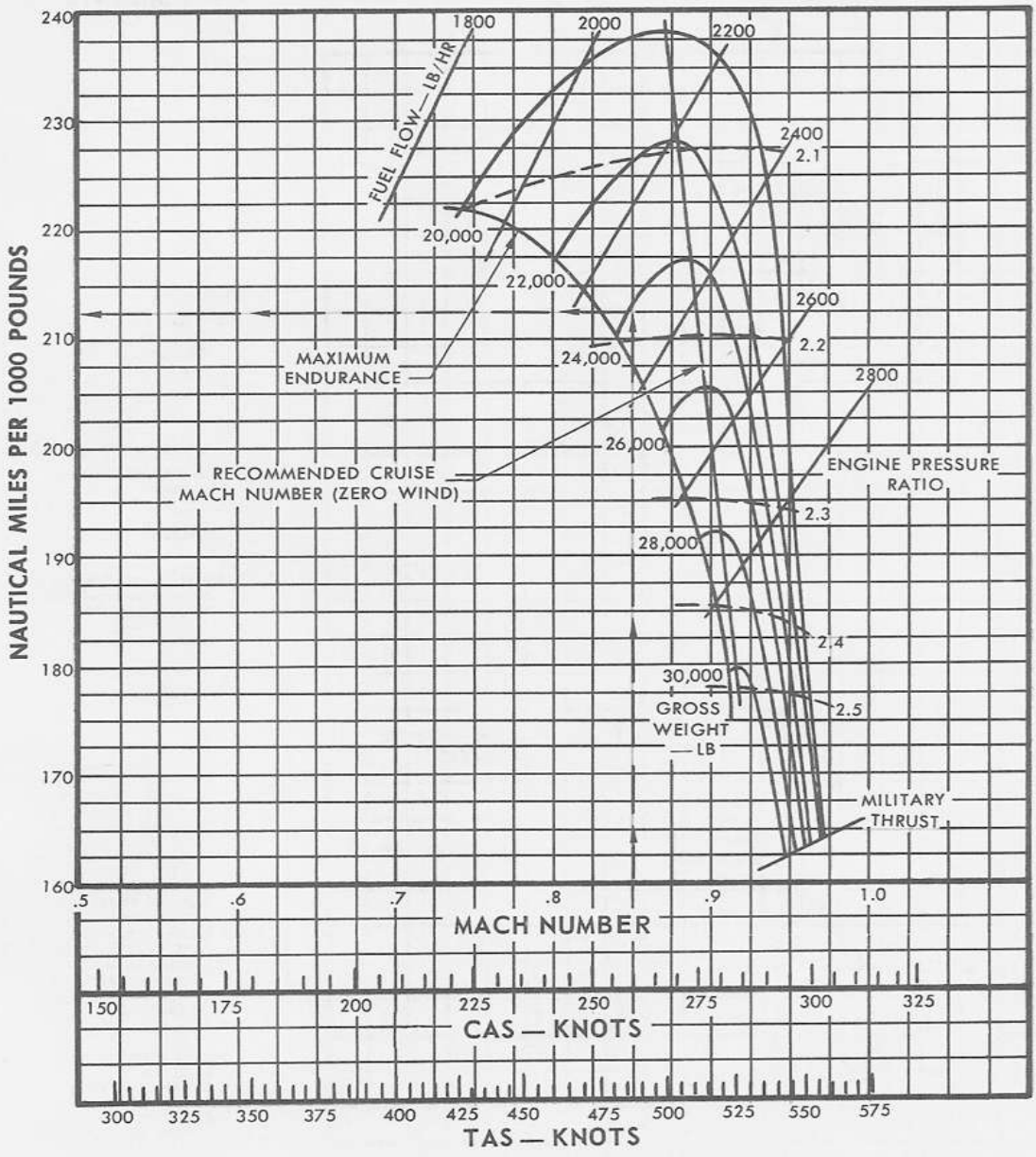
Figure A4-44

F

nautical miles per 1000 pounds — 40,000 feet

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST CONFIGURATION: CLEAN ● STANDARD ATMOSPHERE

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



NOTE

- WIND CORRECTION TO N. MI/LB

  GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$

  WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.

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- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,037G

Figure A4-45

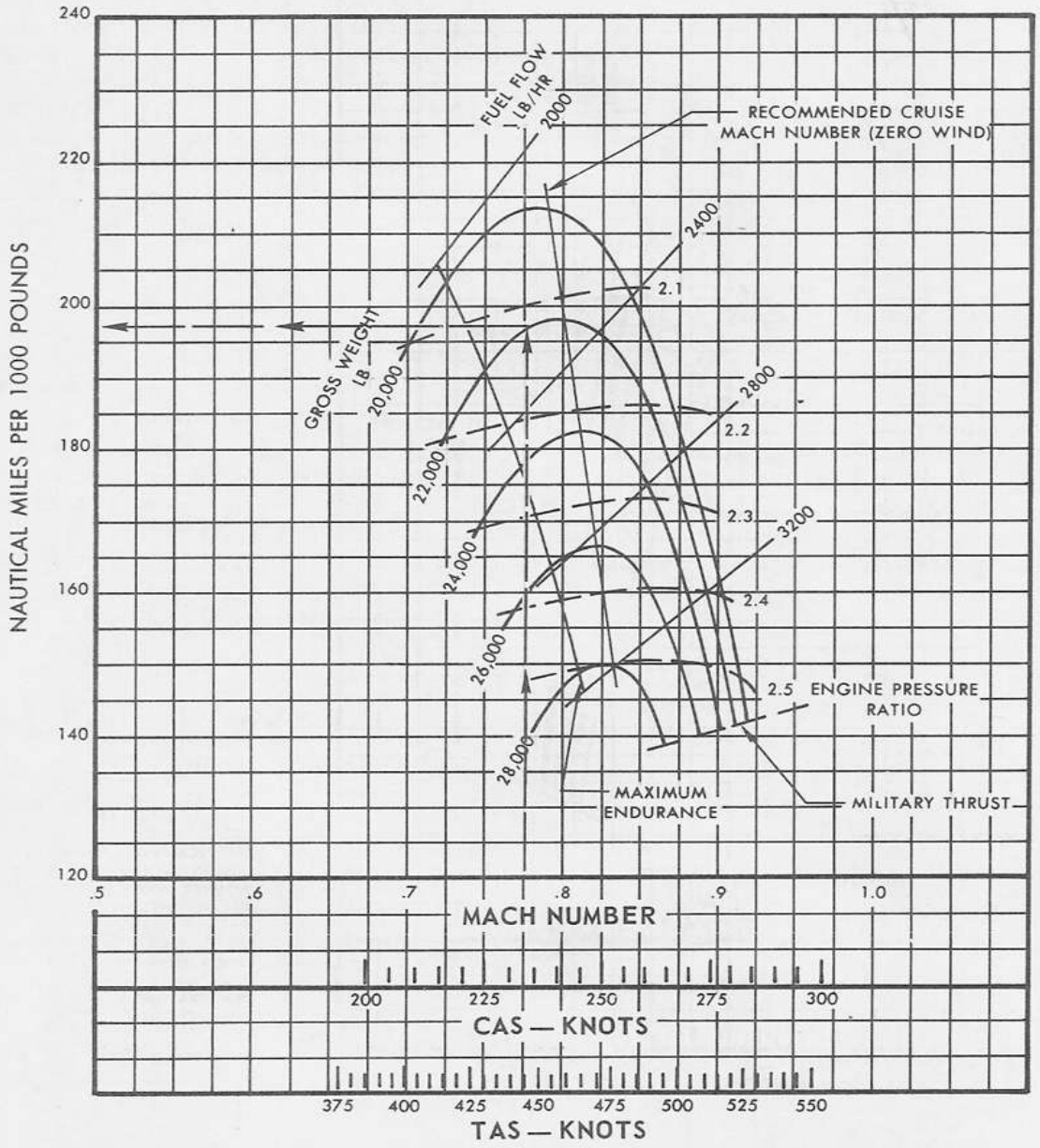
TF

nautical miles per 1000 pounds—40,000 feet

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
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- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,212B

Figure A4-46

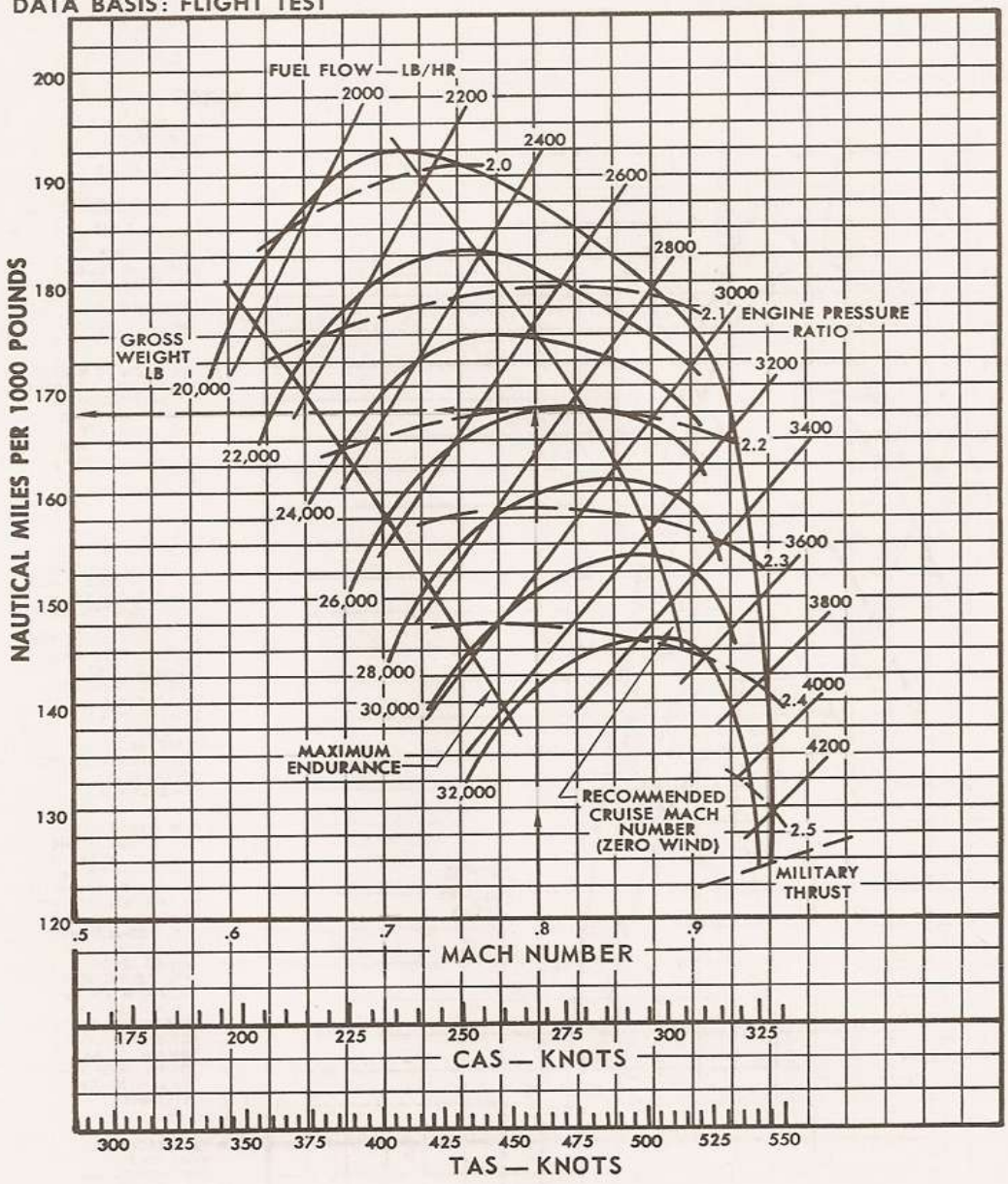
F

nautical miles per 1000 pounds — 35,000 feet

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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

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



22,045G

NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
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- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

Figure A4-43

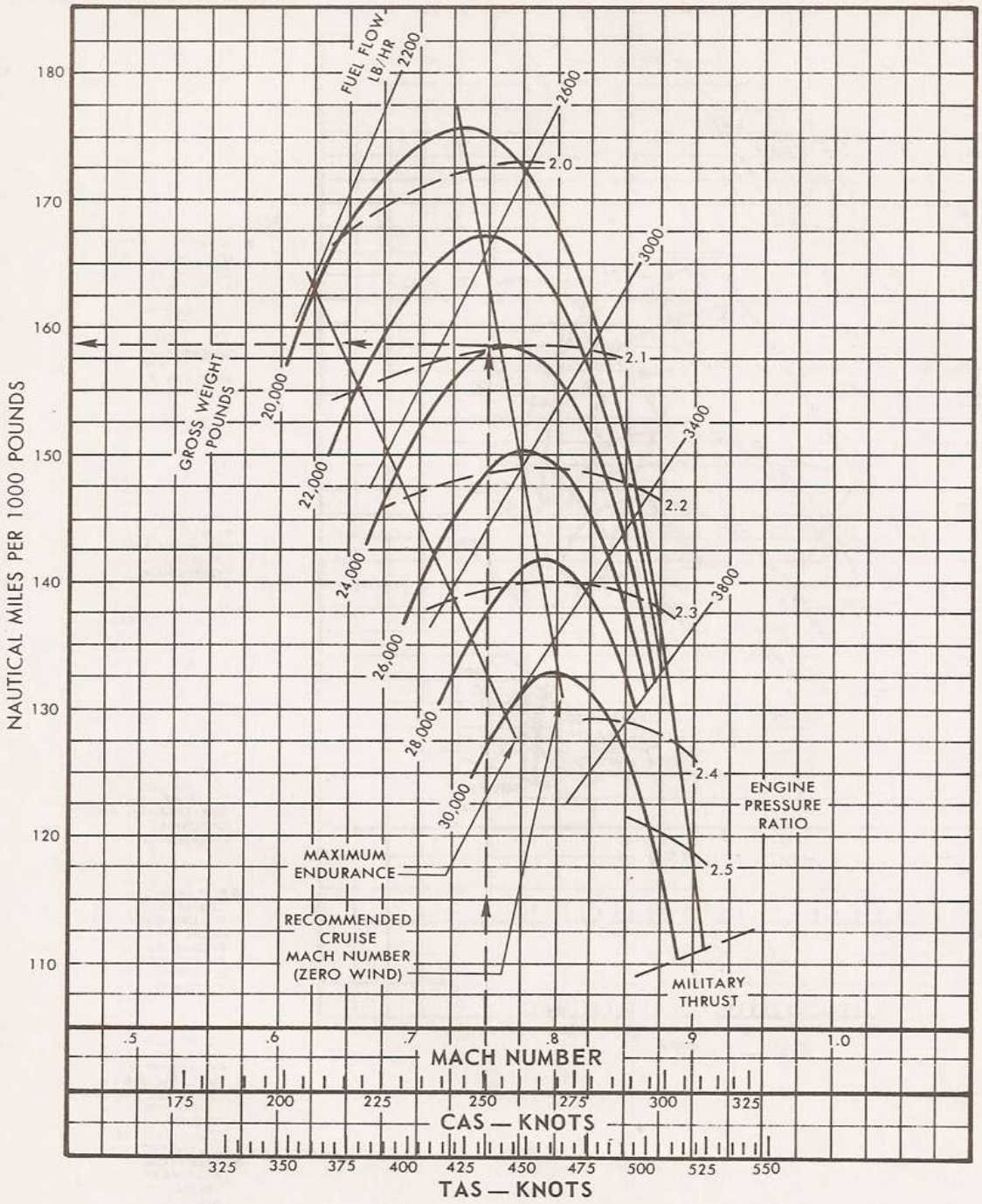
TF

nautical miles per 1000 pounds—35,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



22,211B

NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
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- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

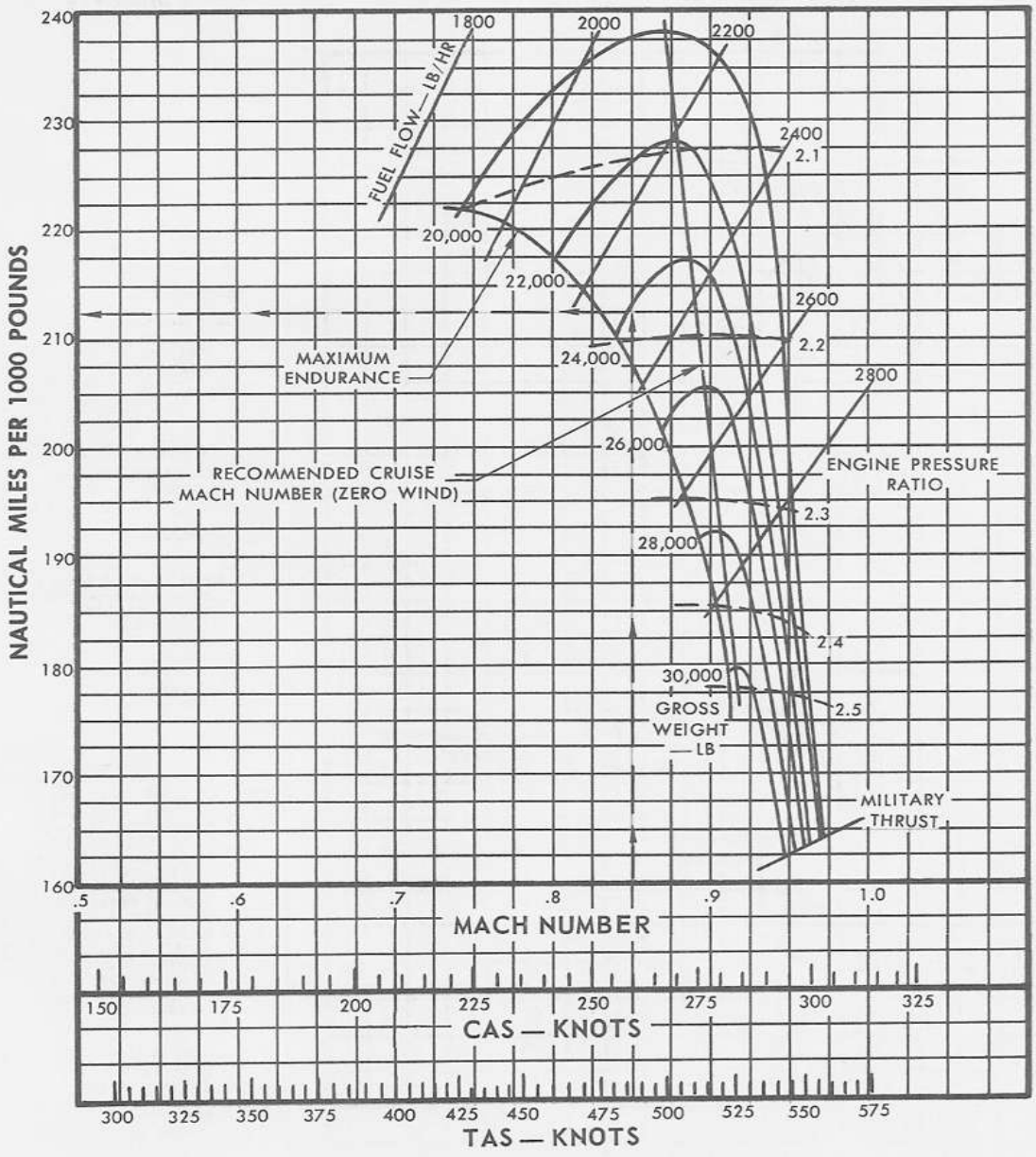
Figure A4-44

F

nautical miles per 1000 pounds — 40,000 feet

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST CONFIGURATION: CLEAN ● STANDARD ATMOSPHERE

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



NOTE

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- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,037G

Figure A4-45

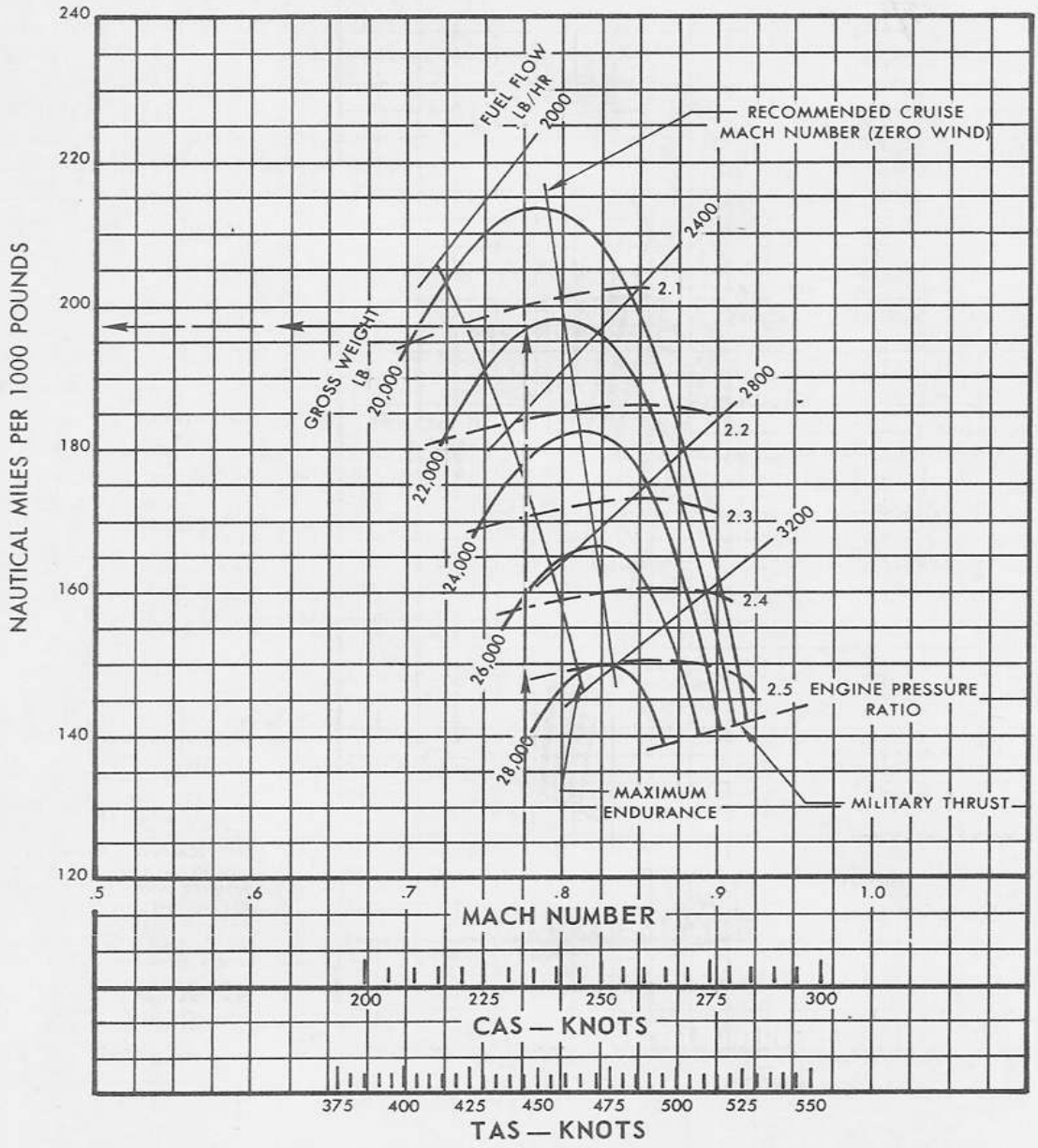
TF

nautical miles per 1000 pounds—40,000 feet

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
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- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,212B

Figure A4-46

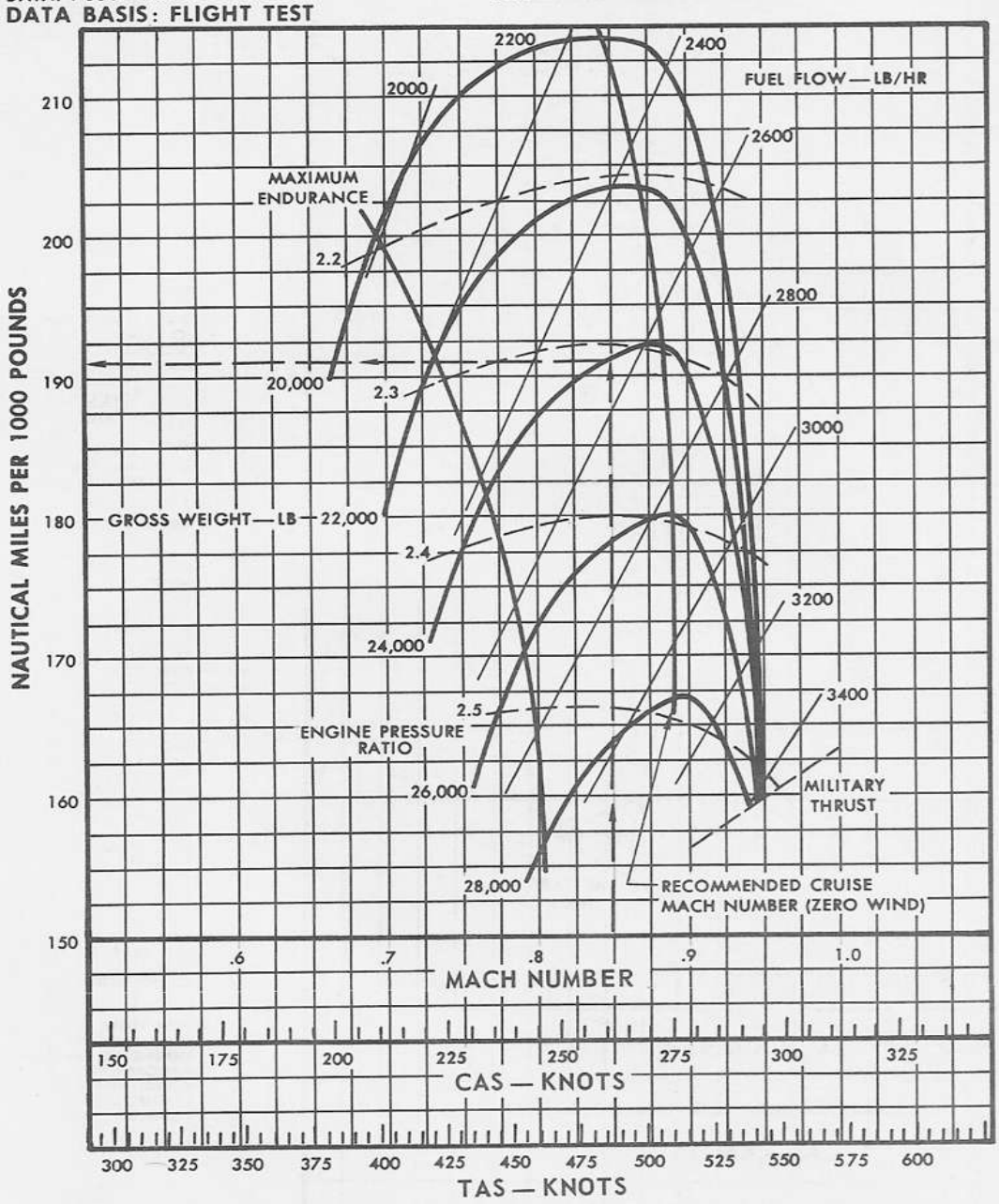
F

nautical miles per 1000 pounds — 40,000 feet

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

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 10%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,046F

Figure A4-47

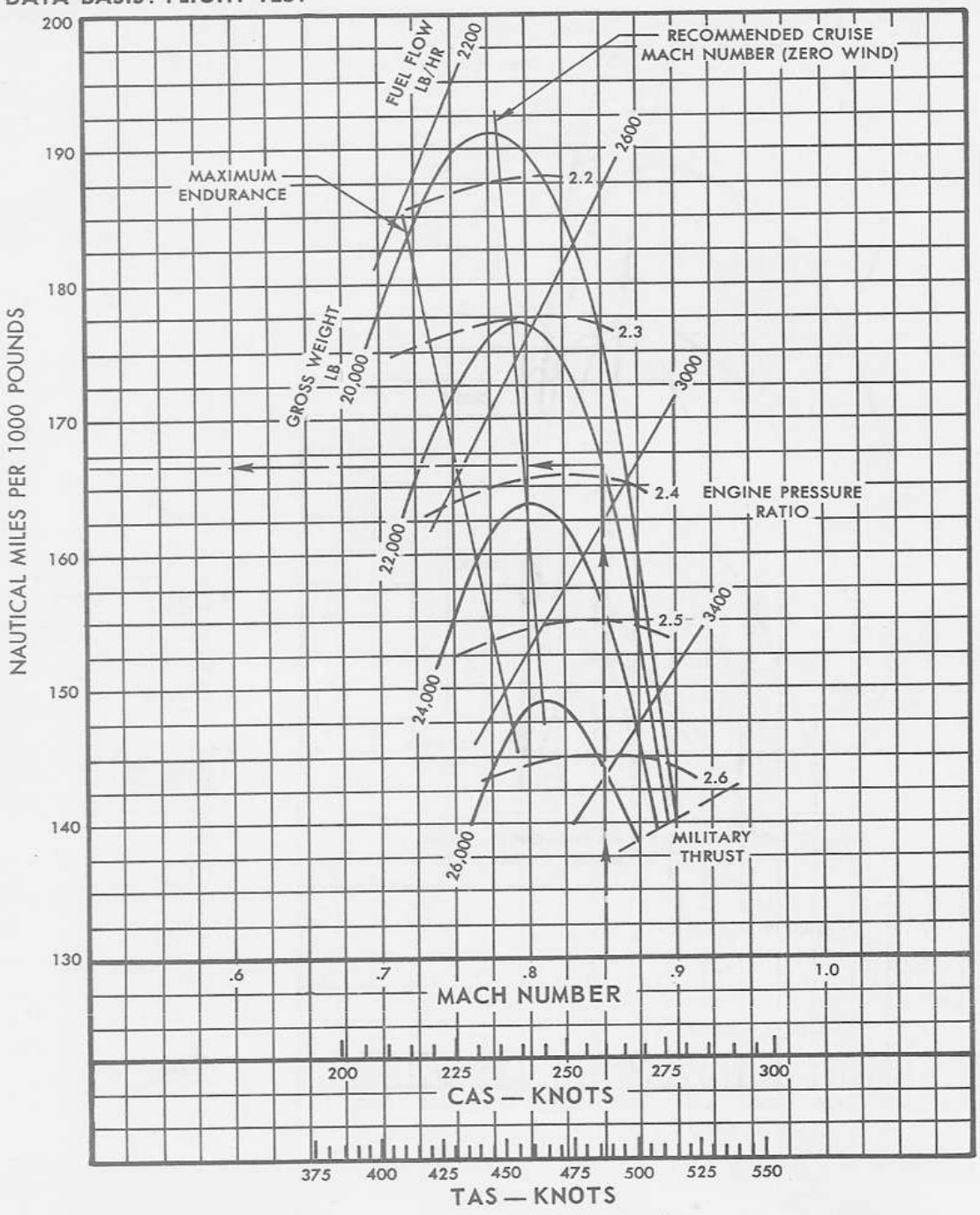
TF

nautical miles per 1000 pounds—40,000 feet

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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



NOTES

- WIND CORRECTION TO N. MI/LB: GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .010 AND DECREASE RANGE 11%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22.213B

Figure A4-48

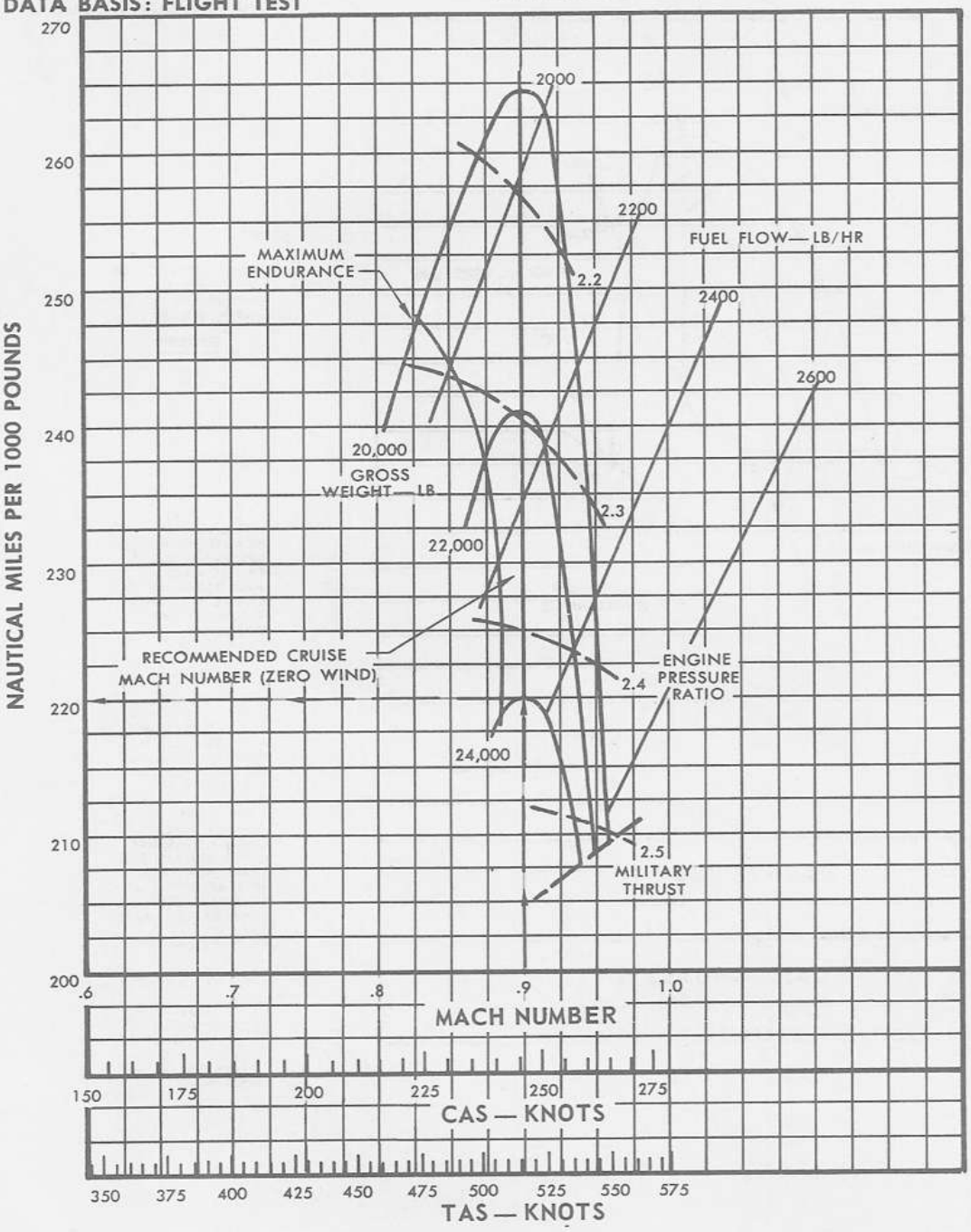
F

nautical miles per 1000 pounds — 45,000 feet

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

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE

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



NOTE

- WIND CORRECTION TO N. MI/LB

  GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$

  WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.

- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 10% FOR TAILWIND APPLY OPPOSITE CORRECTION.

- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.

- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

21,983D

Figure A4-49

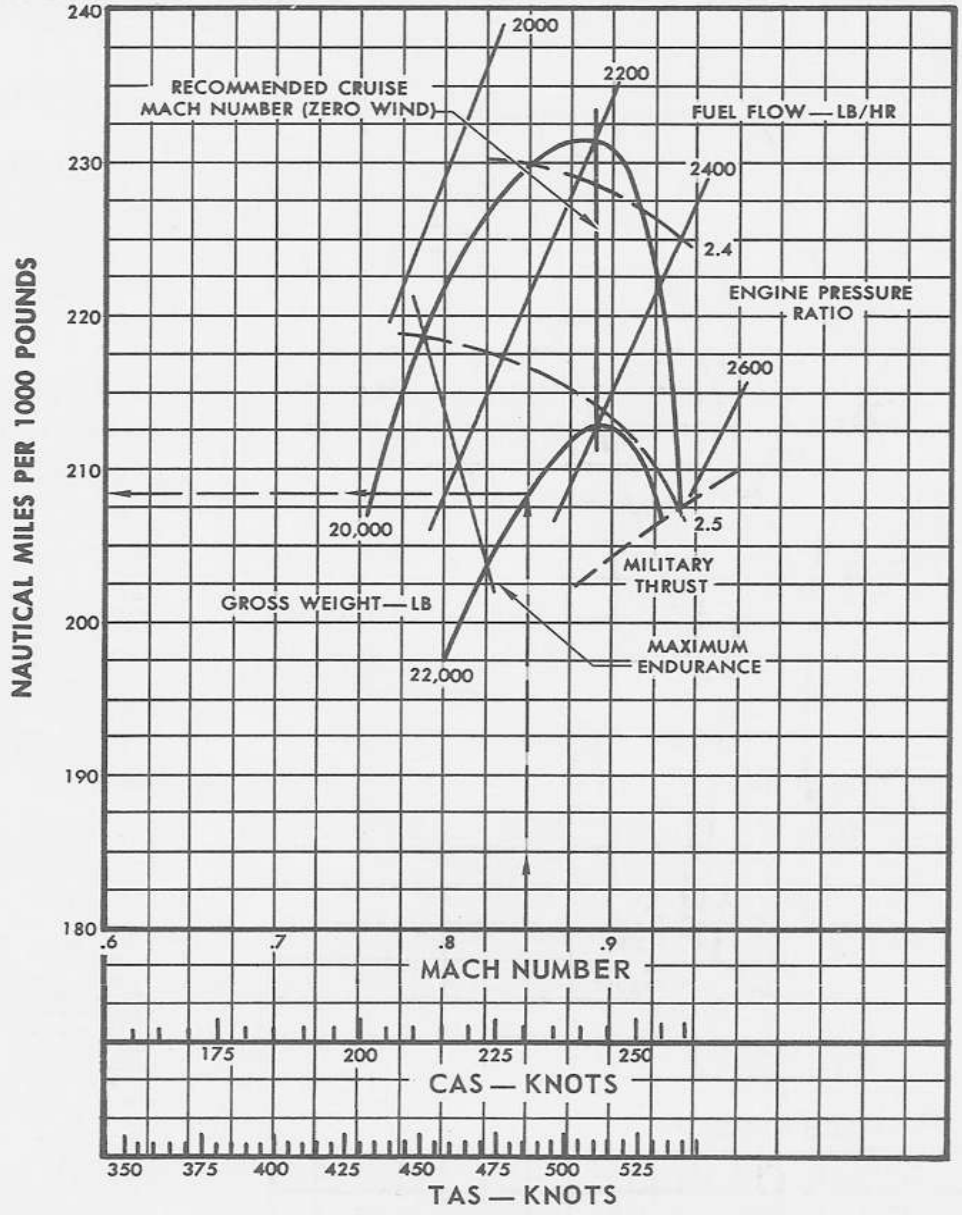
F

nautical miles per 1000 pounds — 45,000 feet

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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

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



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 10%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22,060D

Figure A4-50

Part 5 ENDURANCE

TABLE OF CONTENTS

	Page
Maximum Endurance Profile Chart	A5-2
Optimum Maximum Endurance Profile	A5-6
Combat Allowance Chart	A5-10

MAXIMUM ENDURANCE PROFILE

Figures A5-1 through A5-4 show the constant altitude endurance characteristics for the clean airplane and the airplane with external fuel tanks. These charts are plots of endurance time as a function of altitude for various amounts of fuel remaining. The recommended operating speeds are tabulated, at 5000-foot intervals, on the chart.

Use

Maximum endurance time available for a specified amount of fuel may be found by entering the chart at the endurance altitude, reading right to the desired fuel remaining, and then down to the maximum endurance time available. Conversely, fuel required for endurance for a specific time may be determined by entering the chart with the endurance time, reading up to the endurance altitude and interpolating if necessary for the fuel required.

Note

Fuel allowance for descent and landing has not been included.

Sample Problem

Find the maximum endurance time available with 5000 pounds of fuel remaining for endurance at 30,000 feet. The airplane is in its clean configuration:

A. Enter figure A5-1 at the endurance altitude—30,000 feet.

B. Proceed right to the fuel remaining—5000 pounds.

C. Read down for the maximum endurance time—2 hours and 17 minutes.

OPTIMUM MAXIMUM ENDURANCE PROFILE

The maximum possible endurance time available for a given fuel quantity at any initial altitude is shown in figures A5-5 and A5-6 for the clean airplane and in figures A5-7 and A5-8 for the airplane with external tanks. These charts are based on an optimum flight path from any starting altitude. The basic flight path consists of a military thrust climb or an idle thrust descent to the optimum endurance altitude (where required), a loiter at constant altitude, and a maximum range descent to sea level with idle thrust. Notes and guide lines on the charts indicate the proper flight path to be used. Tabular data include speeds for climb, descent, and loiter, and also for loiter fuel flows at 5000-foot altitude intervals.

Use

Mission planning with the optimum maximum endurance profile is the same as with the optimum profile except that cruise distance is replaced by loiter time, and the flight path utilizes an idle thrust descent.

Note

Fuel allowance for landing is not included.

COMBAT ALLOWANCE CHART

The Combat Allowance Chart, figures A5-9 and A5-10, shows the relationship between time and fuel consumed for military thrust high-speed and maximum thrust high-speed operation. Basically these charts show time as a function of altitude for various amounts of combat fuel available. Supplementary charts of fuel flow in pounds per minute as a function of altitude for the same speed and thrust conditions are also shown.

Use

To find the combat time available for a given amount of fuel, enter the chart at the combat altitude, read to the right to the allotted fuel, and then read down for the time available. For maximum or military thrust fuel flows, enter the chart at the desired altitude and proceed horizontally to the respective fuel flow curve and then down to get the fuel flow in pounds per minute.

Note

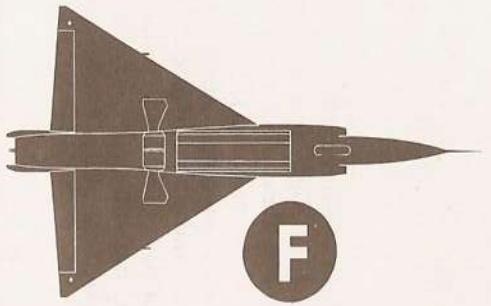
Does not include fuel to accelerate to combat Mach number.

maximum endurance profile

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE

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



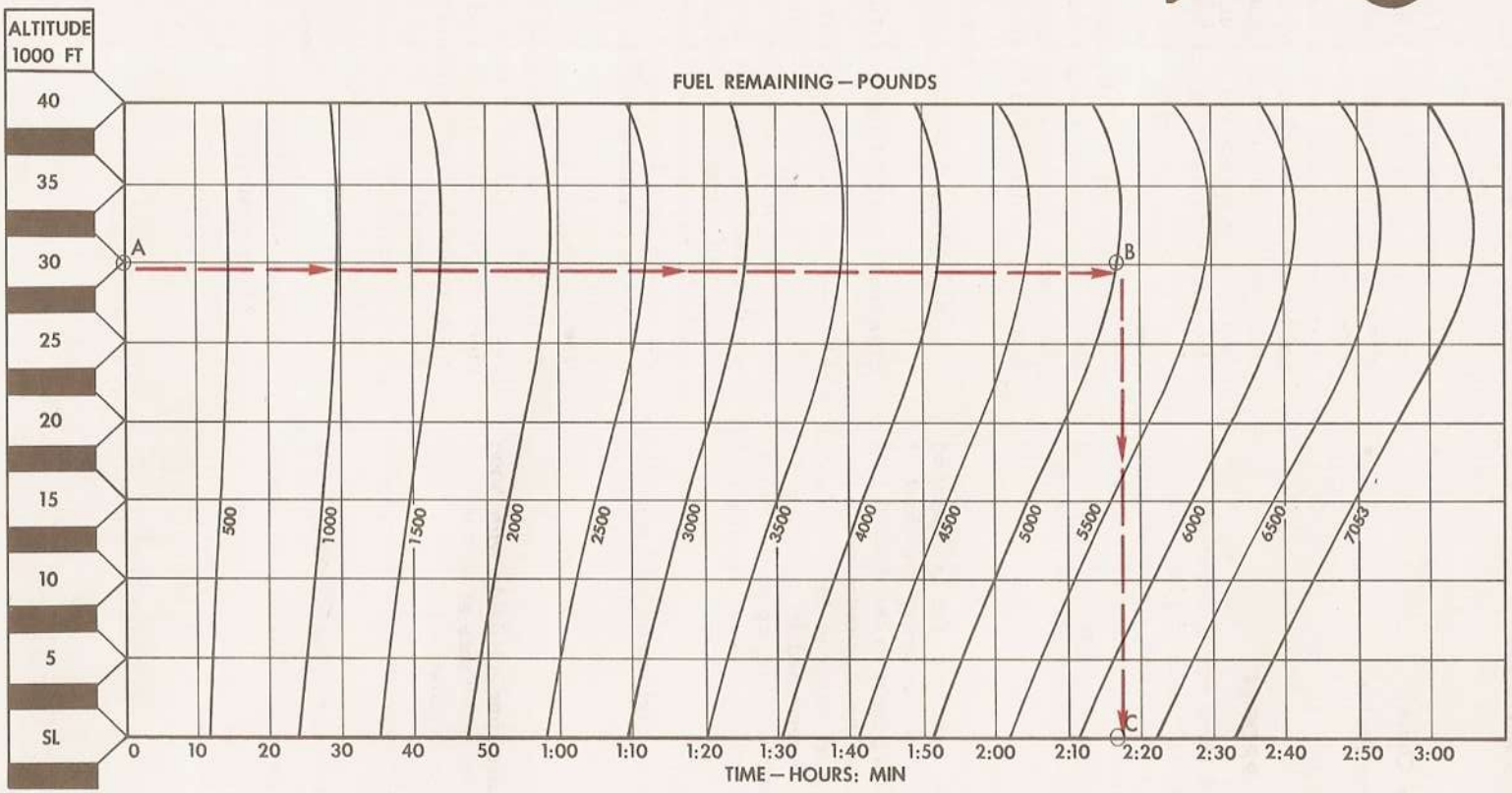


Figure A5-1

ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*
40,000	.85	259	485	2340
35,000	.72	240	412	2280
30,000	.64	238	375	2280
25,000	.54	225	327	2330
20,000	.49	224	301	2420
15,000	.44	223	278	2500
10,000	.40	222	257	2580
5,000	.37	220	237	2650
S.L.	.32	212	212	2780

*APPROXIMATE

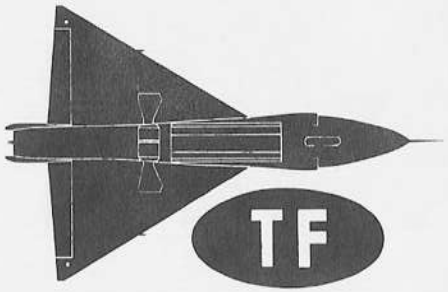
22,053D

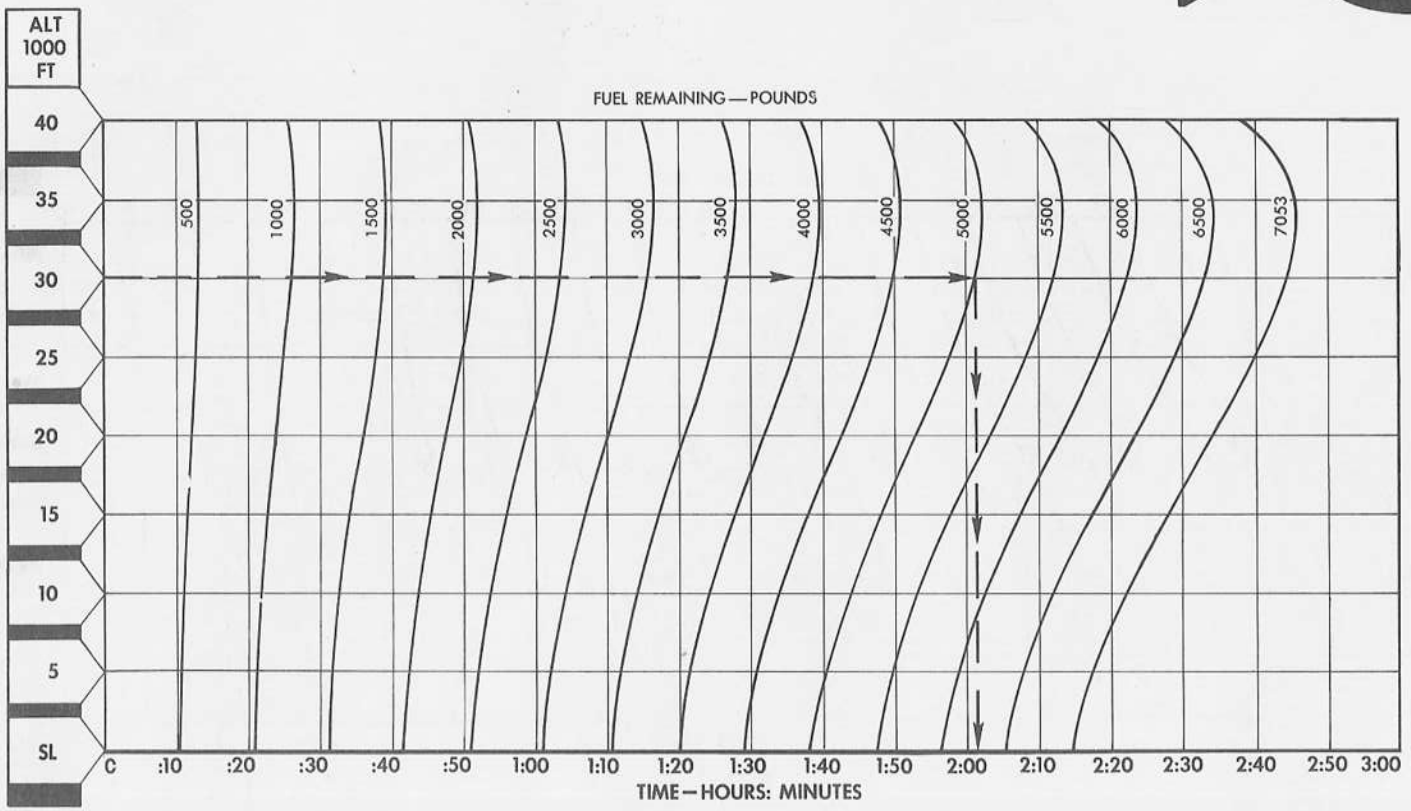
maximum endurance profile

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE

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





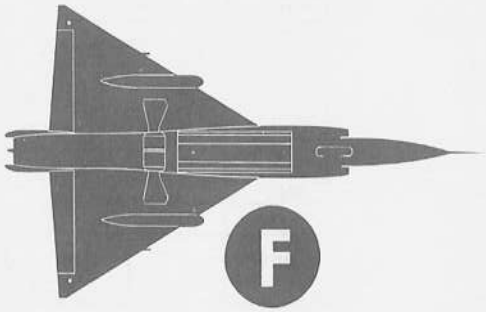
ALTITUDE FEET	MACH NO.	CAS KNOTS	TAS KNOTS	FUEL FLOW LB/HR
40,000	.79	238	452	2700
35,000	.72	240	412	2550
30,000	.64	241	380	2610
25,000	.57	237	345	2650
20,000	.51	233	314	2760
15,000	.45	228	285	2900
10,000	.41	226	262	3000
5,000	.37	225	242	3080
SL	.34	224	224	3160

Figure A5-2

22,215B

maximum endurance profile

MODEL: F-102A CONFIGURATION: TWO – 230 GALLON EXTERNAL TANKS ENGINE: J57-23
DATE: 15 OCTOBER 1958 STANDARD ATMOSPHERE FUEL GRADE: JP-4
DATA BASIS: FLIGHT TEST FUEL DENSITY: 6.5 LB/GAL



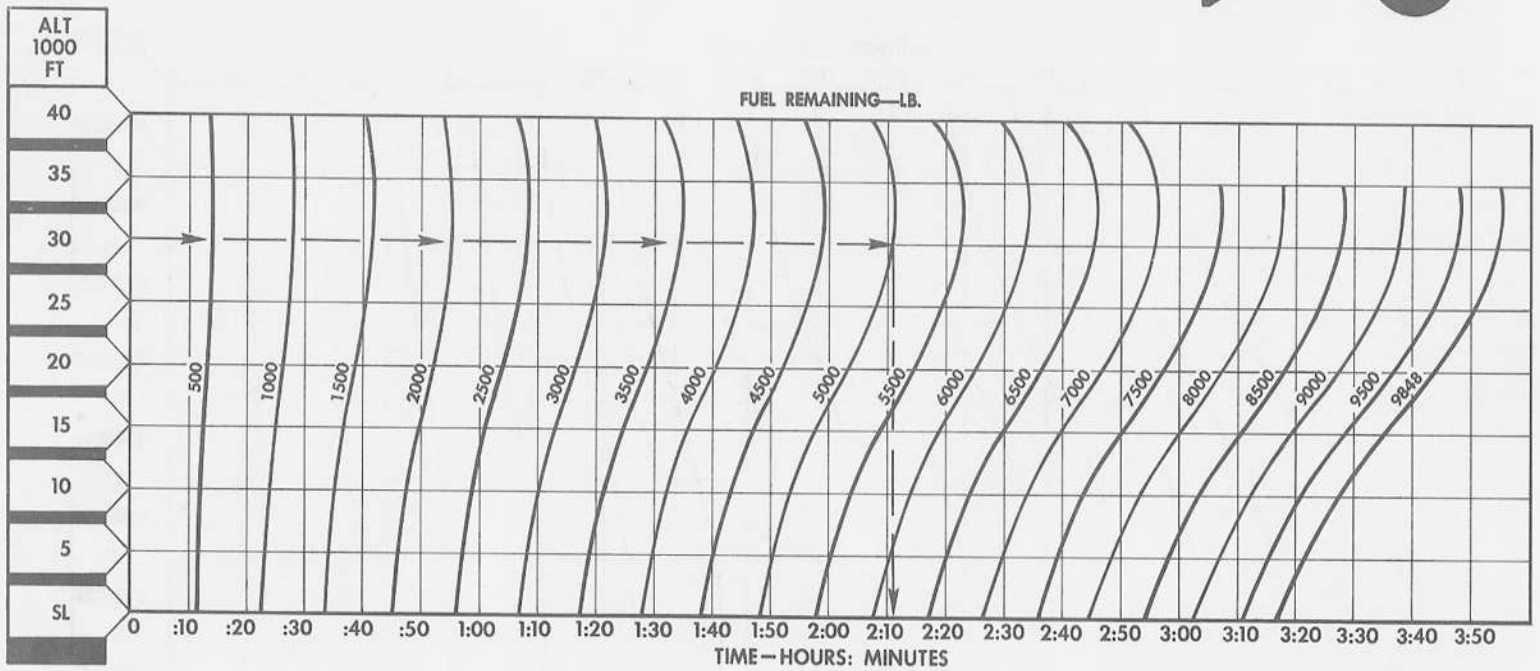
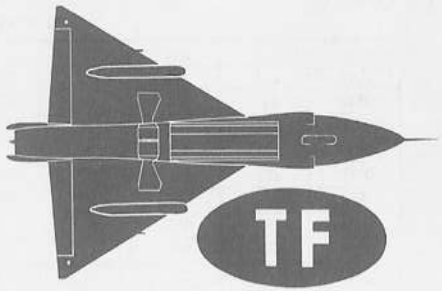


Figure A5-3

ALTITUDE FEET	MACH NO.	CAS KNOTS	TAS KNOTS	FUEL FLOW LB/HR
40,000	.79	240	455	2670
35,000	.70	235	404	2560
30,000	.62	233	368	2580
25,000	.55	229	333	2600
20,000	.49	224	302	2700
15,000	.44	223	278	2800
10,000	.40	218	253	2890
5,000	.36	217	234	2960
S.L.	.34	228	228	3040

22,054B

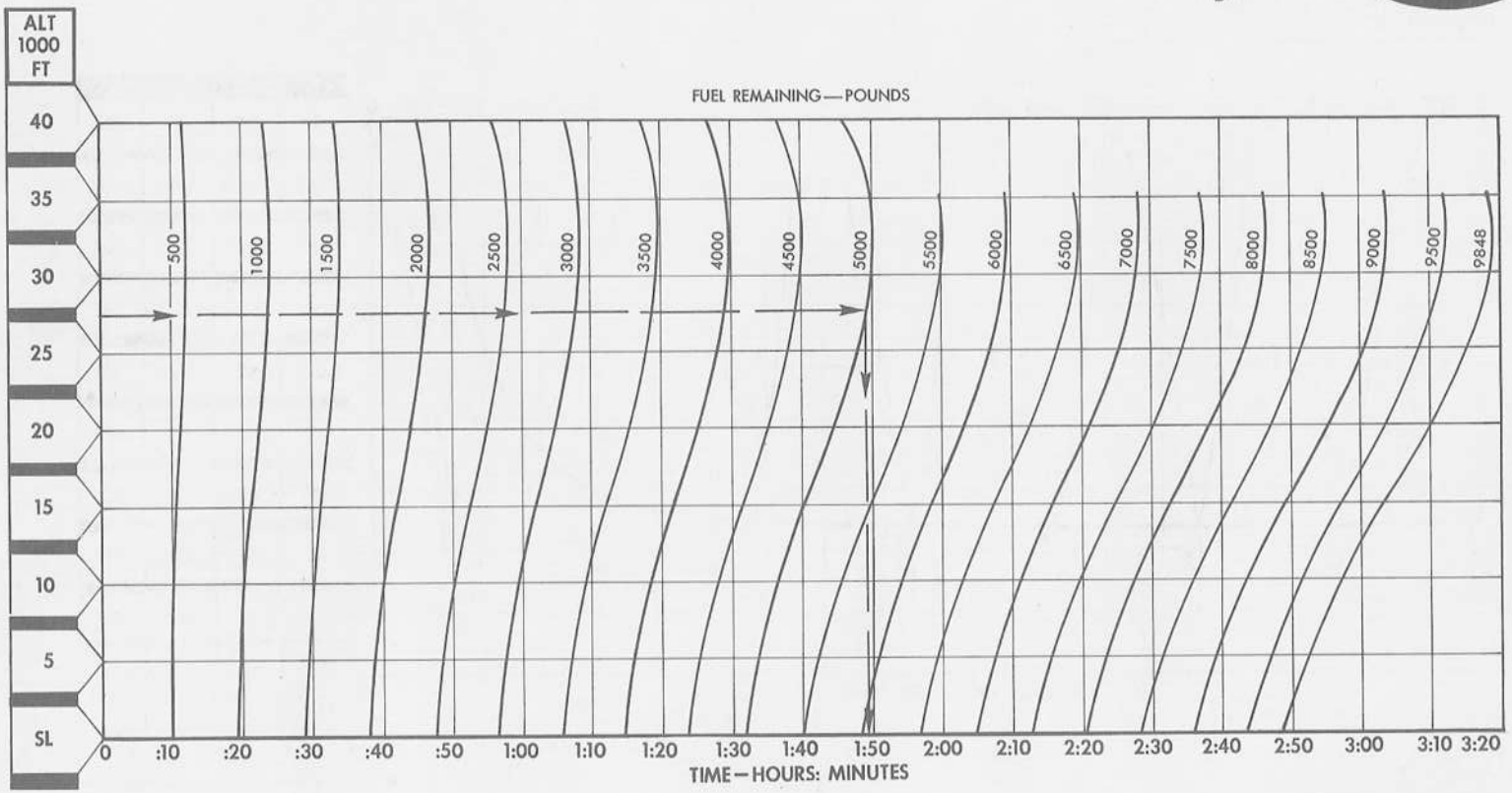
maximum endurance profile



MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO—230 GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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

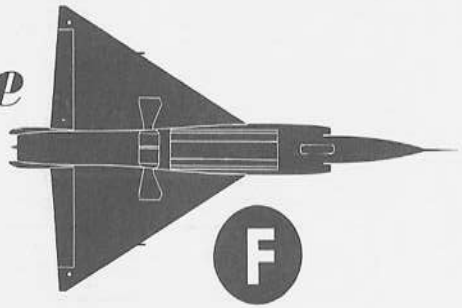


ALTITUDE FEET	MACH NO.	CAS KNOTS	TAS KNOTS	FUEL FLOW LB/HR
40,000	.80	242	459	3270
35,000	.73	244	418	3020
30,000	.65	243	383	3040
25,000	.58	239	348	3060
20,000	.51	235	316	3130
15,000	.46	231	288	3250
10,000	.42	229	265	3340
5,000	.38	228	245	3450
SL	.34	227	227	3540

22,216B

Figure A5-4

optimum maximum endurance profile

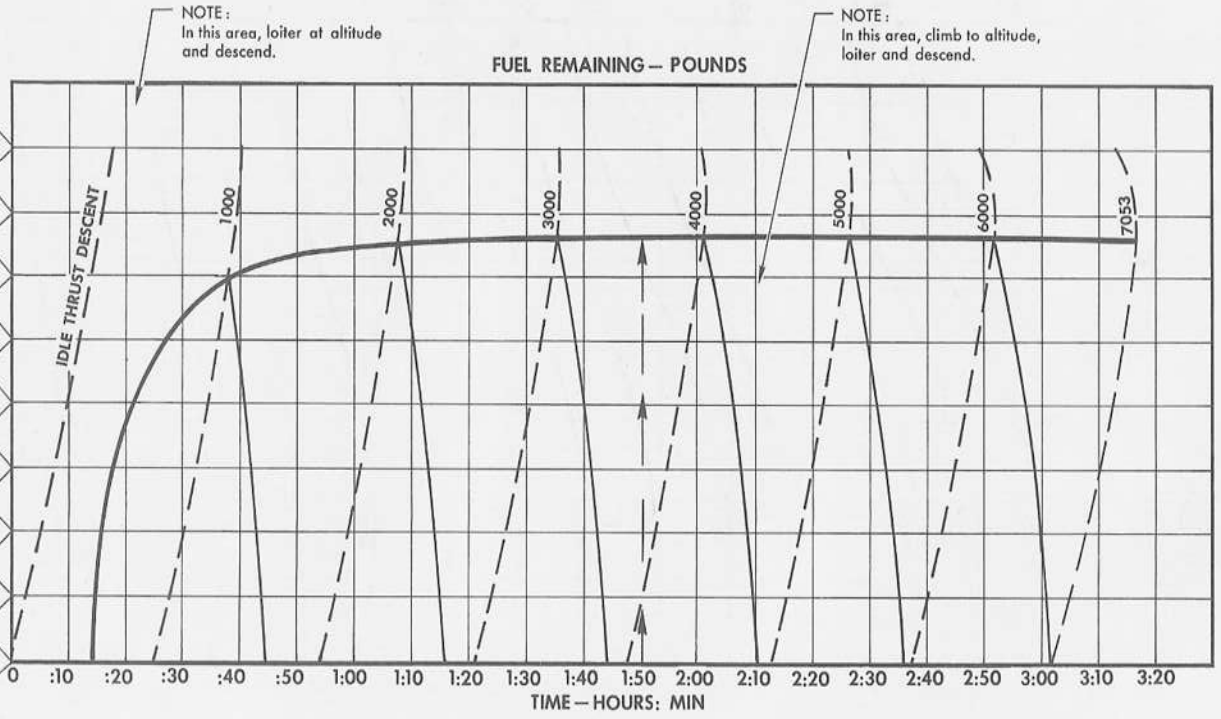


MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE
CLIMB-LOITER-DESCEND

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

MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FEET
516	276	.90	40
519	310	.90	35
483	311	.82	30
451	314	.75	25
419	315	.68	20
393	317	.63	15
370	321	.58	10
351	326	.54	5
331	331	.50	SL



ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*
40,000	.85	258	485	2340
35,000	.72	240	412	2280
30,000	.64	238	375	2280
25,000	.54	225	327	2330
20,000	.49	224	301	2420
15,000	.44	223	278	2500
10,000	.40	222	257	2580
5,000	.37	220	237	2650
S.L.	.32	212	212	2780

*APPROXIMATE

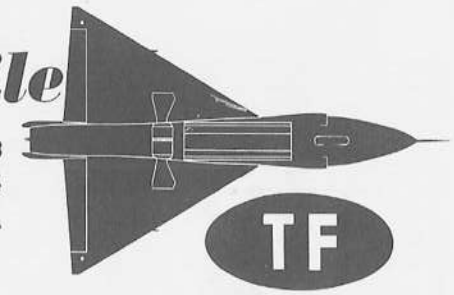
OPTIMUM LEVEL FLIGHT ENDURANCE ALTITUDE
CLIMB PATH GUIDE LINES
FUEL CONSUMED OR AVAILABLE

NOTE:
1. Fuel required at any point includes military thrust climb if required, to altitude, loiter and descend to sea level.
2. No allowance or reserve made for landing.

22,055B

Figure A5-5

optimum maximum endurance profile

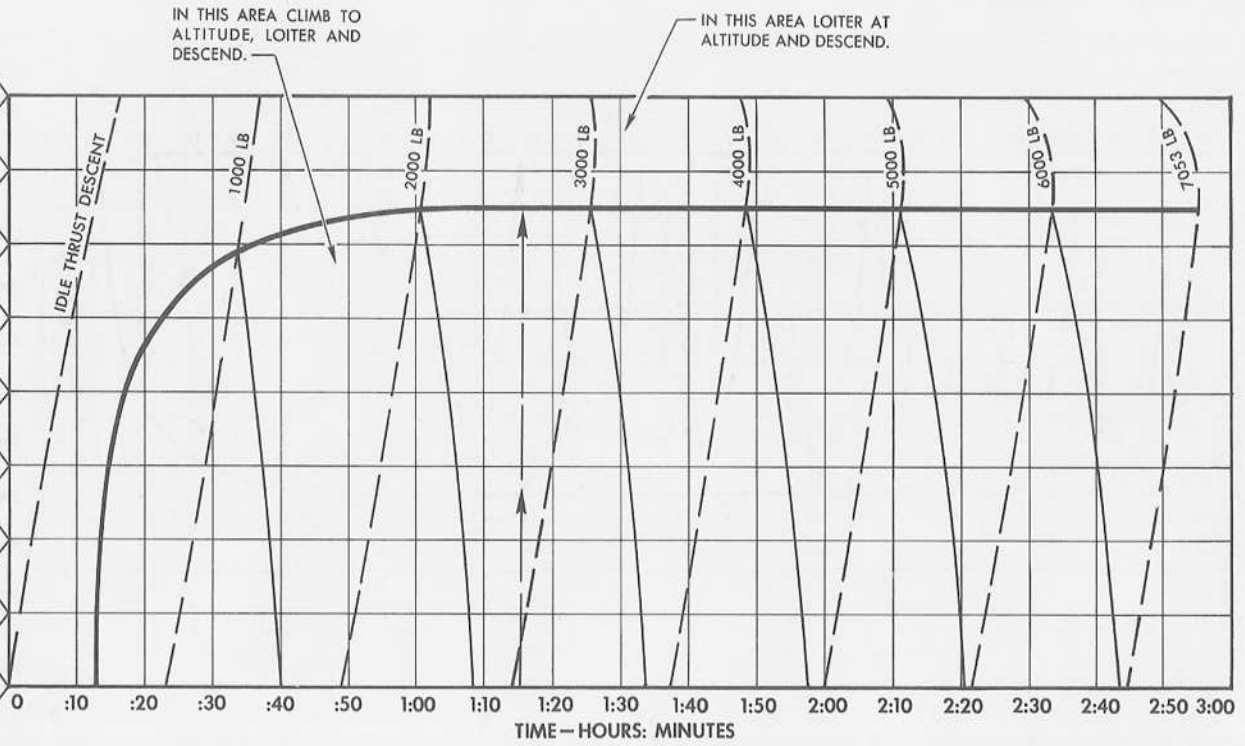


MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE
CLIMB—LOITER—DESCEND

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

MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FT
474	251	.83	40
474	280	.83	35
444	284	.75	30
417	290	.69	25
394	297	.64	20
375	302	.60	15
360	312	.57	10
348	323	.54	5
337	337	.51	SL



ALTITUDE FEET	MACH NO.	CAS KNOTS	TAS KNOTS	FUEL FLOW LB/HR
40,000	.79	238	452	2700
35,000	.72	240	412	2550
30,000	.64	241	380	2610
25,000	.57	237	345	2650
20,000	.51	233	314	2760
15,000	.45	228	285	2900
10,000	.41	226	262	3000
5,000	.37	225	242	3080
SL	.34	224	224	3160

OPTIMUM LEVEL FLIGHT ENDURANCE ALTITUDE
CLIMB PATH GUIDE LINES
FUEL CONSUMED OR AVAILABLE

NOTE
1. Fuel required at any point includes military thrust climb if required to altitude, loiter and descend to sea level.
2. No allowance or reserve made for landing.

22,217B

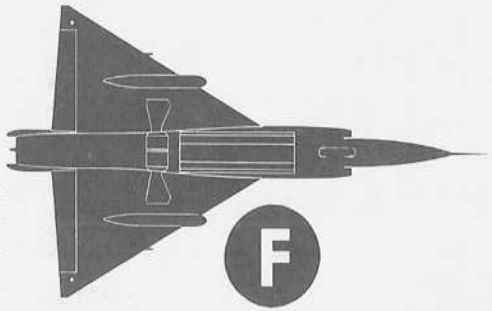
Figure A5-6

optimum maximum endurance profile

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO-230 GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE
CLIMB-LOITER-DESCEND

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



MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FEET
462	244	.81	40
464	273	.81	35
427	273	.73	30
400	277	.67	25
376	281	.61	20
354	285	.57	15
337	292	.53	10
320	296	.49	5
304	304	.46	SL

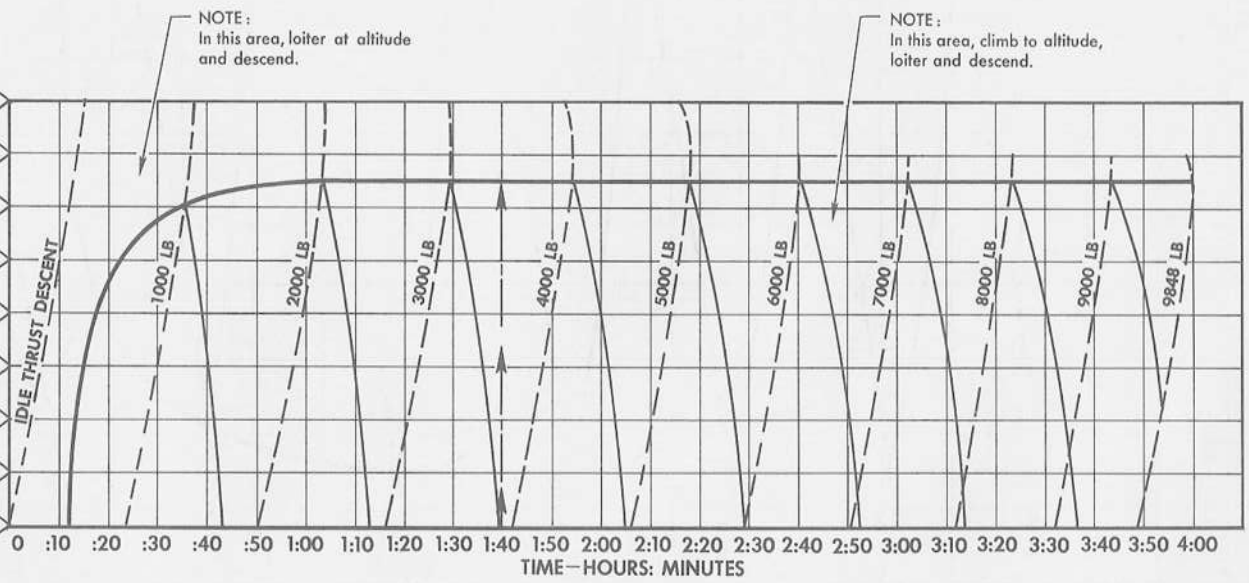


Figure A5-7

ALTITUDE FEET	MACH NO.*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*
40,000	.79	240	455	2670
35,000	.70	235	404	2560
30,000	.62	233	368	2580
25,000	.55	229	333	2600
20,000	.49	224	302	2700
15,000	.44	223	278	2800
10,000	.40	218	253	2890
5,000	.36	217	234	2960
SL	.34	228	228	3040

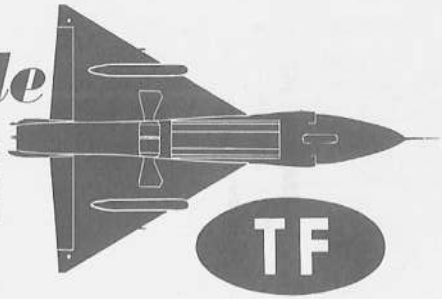
*APPROXIMATE

——— OPTIMUM LEVEL FLIGHT ENDURANCE ALTITUDE
——— CLIMB PATH GUIDE LINES
– – – – FUEL CONSUMED OR AVAILABLE

NOTE:
1. Fuel required at any point includes Military Thrust Climb (if required) to altitude, loiter and descend to sea level.
2. No allowance or reserve made for landing.

22,133B

optimum maximum endurance profile

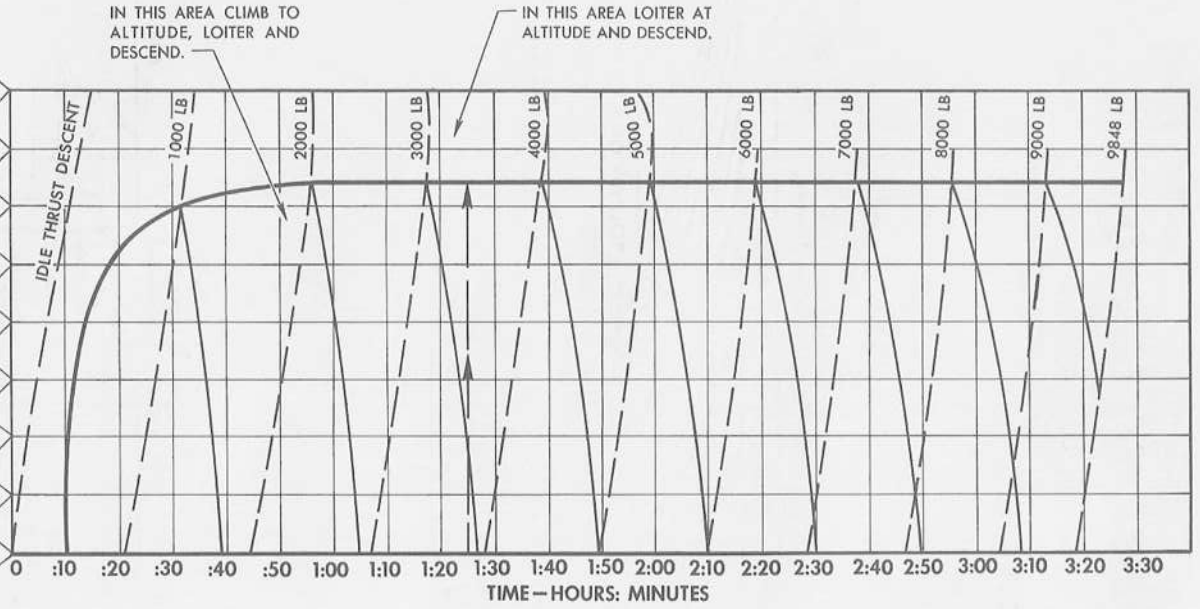


MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CLIMB—LOITER—DESCEND
CONFIGURATION: TWO—230 GALLON EXTERNAL TANKS
STANDARD ATMOSPHERE

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

MILITARY THRUST CLIMB			
TAS KNOTS	CAS KNOTS	MACH NO.	ALT 1000 FT.
465	245	.81	40
465	274	.81	35
428	273	.73	30
399	277	.66	25
374	281	.61	20
354	284	.57	15
335	290	.53	10
320	297	.49	5
304	304	.46	SL



ALTITUDE FEET	MACH NO.	CAS KNOTS	TAS KNOTS	FUEL FLOW LB/HR
40,000	.80	242	459	3270
35,000	.73	244	418	3020
30,000	.65	243	383	3040
25,000	.58	239	348	3060
20,000	.51	235	316	3130
15,000	.46	231	288	3250
10,000	.42	229	265	3340
5,000	.38	228	245	3450
SL	.34	227	227	3540

OPTIMUM LEVEL FLIGHT ENDURANCE ALTITUDE
CLIMB PATH GUIDE LINES
FUEL CONSUMED OR AVAILABLE

NOTE:
1. Fuel required at any point includes Military Thrust Climb (if required) to altitude, loiter and descend to sea level.
2. No allowance or reserve made for landing.

22,218B

Figure A5-8

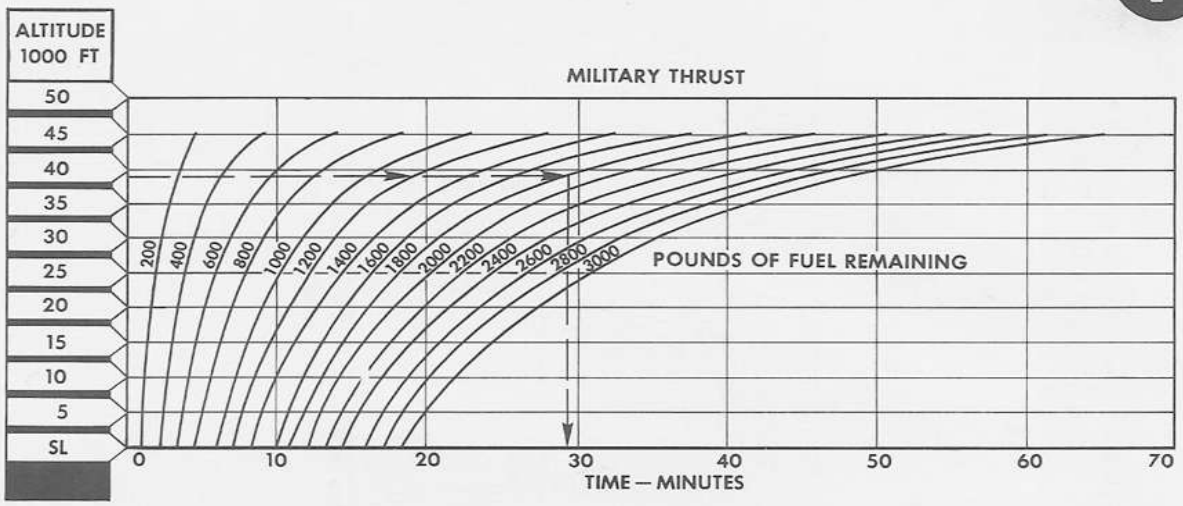
combat allowance chart

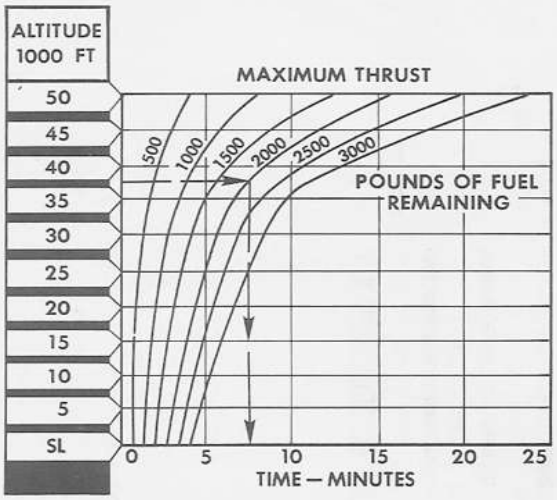
MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

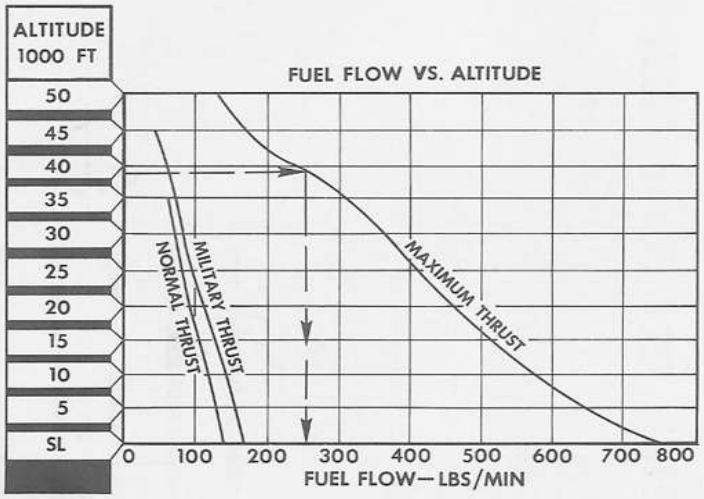
CONFIGURATION: CLEAN
STANDARD ATMOSPHERE

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

F



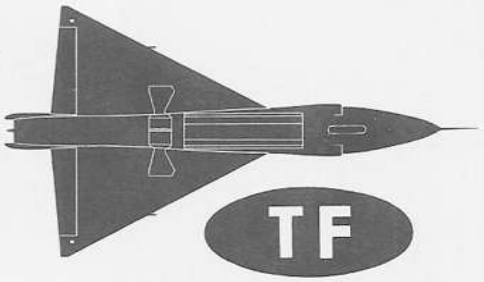




22,056

Figure A5-9

combat allowance chart

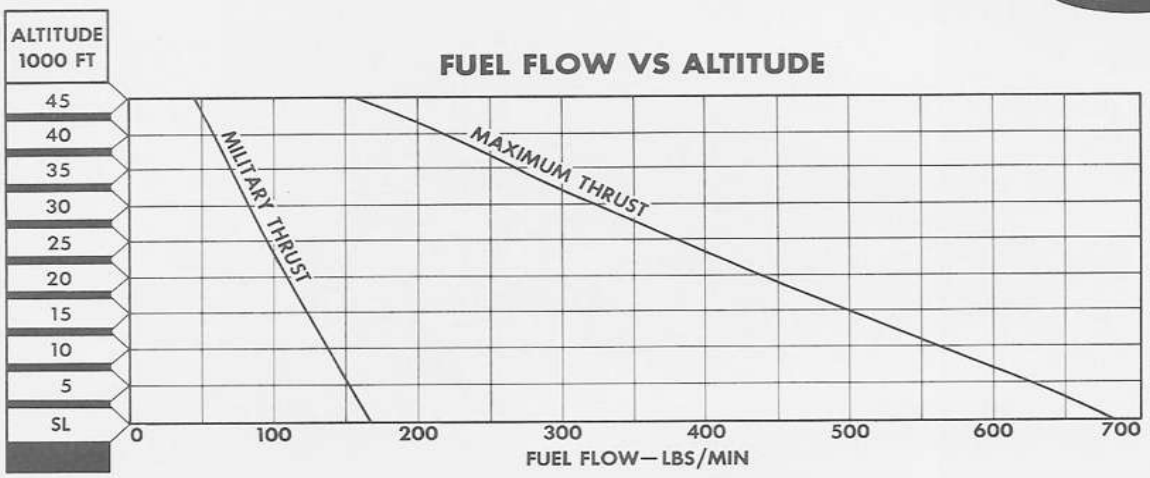


MODEL: TF-102A CONFIGURATION: CLEAN ENGINE: J57-23

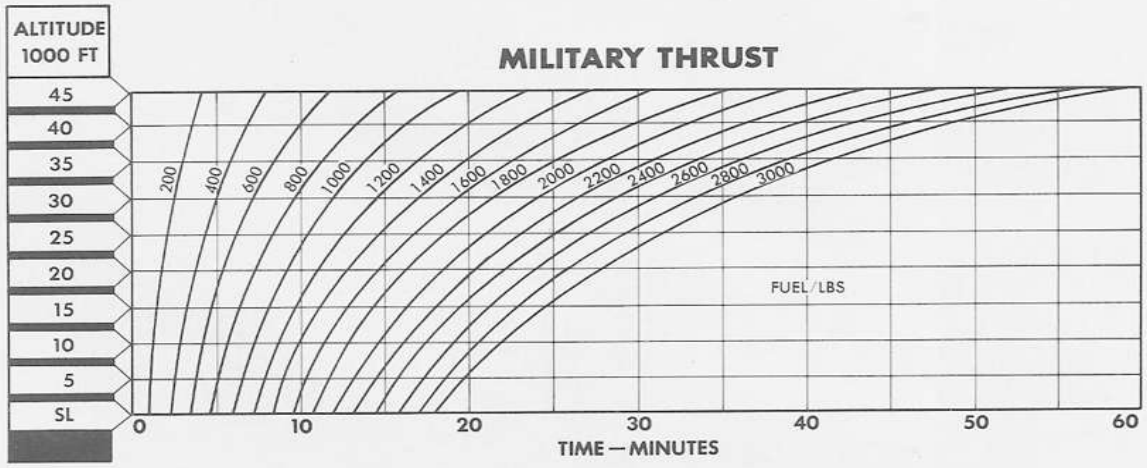
DATE: 1 NOVEMBER 1959 STANDARD ATMOSPHERE FUEL GRADE: JP-4

DATA BASIS: FLIGHT TEST FUEL DENSITY: 6.5 LB/GAL.

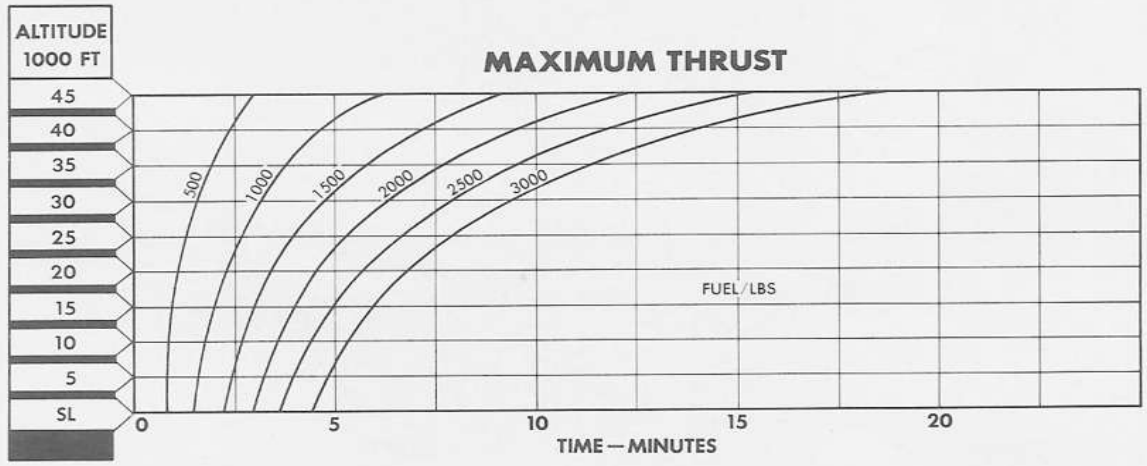
FUEL FLOW VS ALTITUDE



MILITARY THRUST



MAXIMUM THRUST



22,219

Figure A5-10

Part 6 DESCENT

TABLE OF CONTENTS

Page

Descent Chart A6-2

DESCENTS

Figures A6-1 through A6-9 graphically illustrate the descent characteristics. The data shown are fuel used, rate of descent, time to descend and distance traveled versus pressure altitude for various airplane configurations. Speed schedules are noted on the charts and in the remarks.

Use

Enter charts on the altitude scale and proceed right to the "recommended descent" or "maximum range descent" curves, whichever is applicable. Read down for required fuel rate of descent, time and distance traveled.

Sample Problem

Find the time and fuel required to descend from 43,700 feet clean configuration.

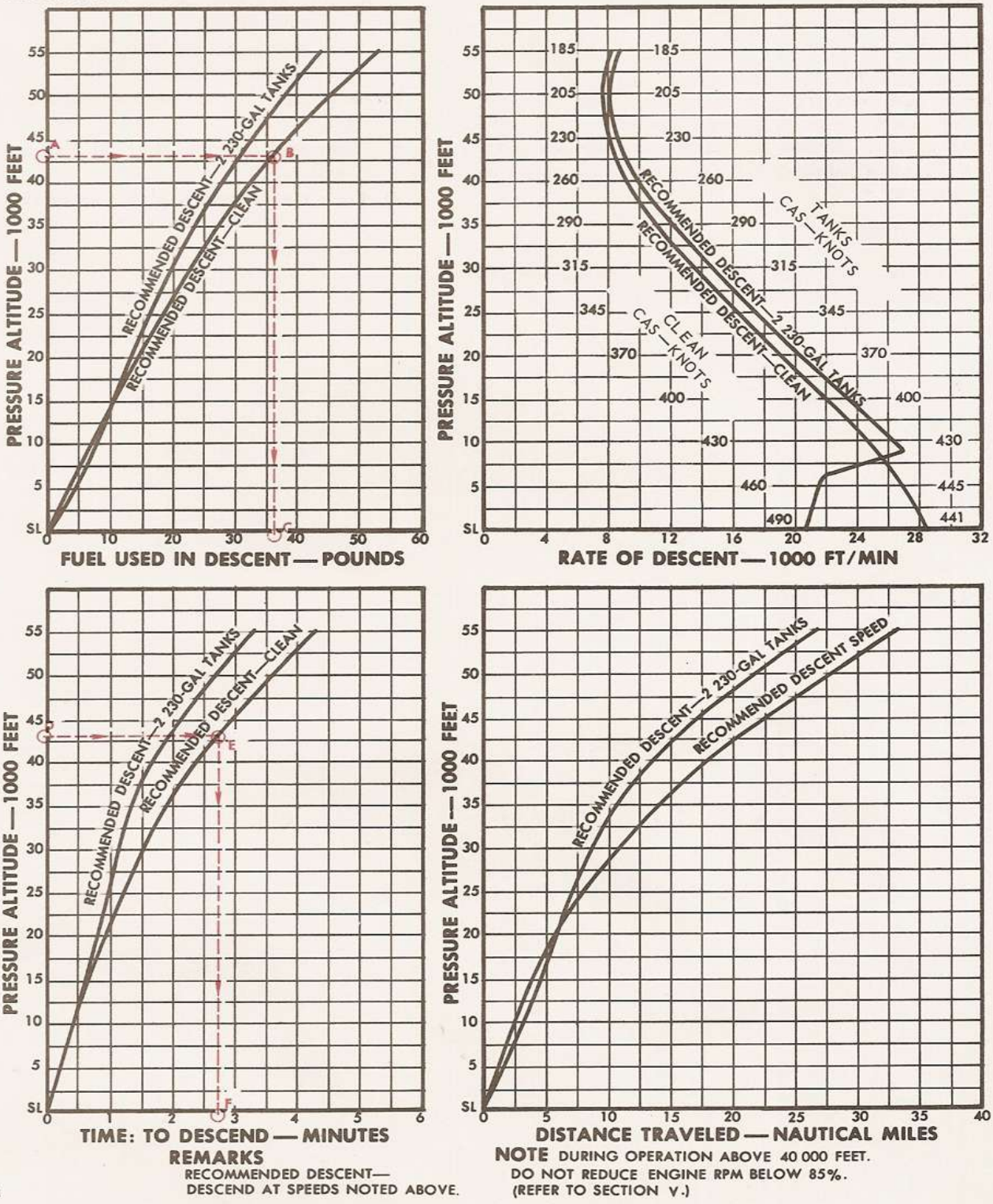
A. Enter the upper left block on figure A6-1 on the pressure altitude—43,700 feet.
B. Read right to the "recommended descent—clean" line.
C. Then read down to the fuel used in descent—37 pounds.
D. Enter the lower left block on figure A6-1 on the pressure altitude—43,700 feet.
E. Read right to the "recommended descent—clean" line.
F. Then read down to the time of descent—2.8 minutes.

F

descents

MODEL: F-102A STANDARD ATMOSPHERE ● GROSS WEIGHT: 22,000 LB
DATE: 15 OCTOBER 1958 IDLE THRUST ● SPEED BRAKES OPEN
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
ALL FUEL GRADES



REMARKS
RECOMMENDED DESCENT—
DESCEND AT SPEEDS NOTED ABOVE.

NOTE DURING OPERATION ABOVE 40 000 FEET.
DO NOT REDUCE ENGINE RPM BELOW 85%.
(REFER TO SECTION V.)

22,025H

Figure A6-1

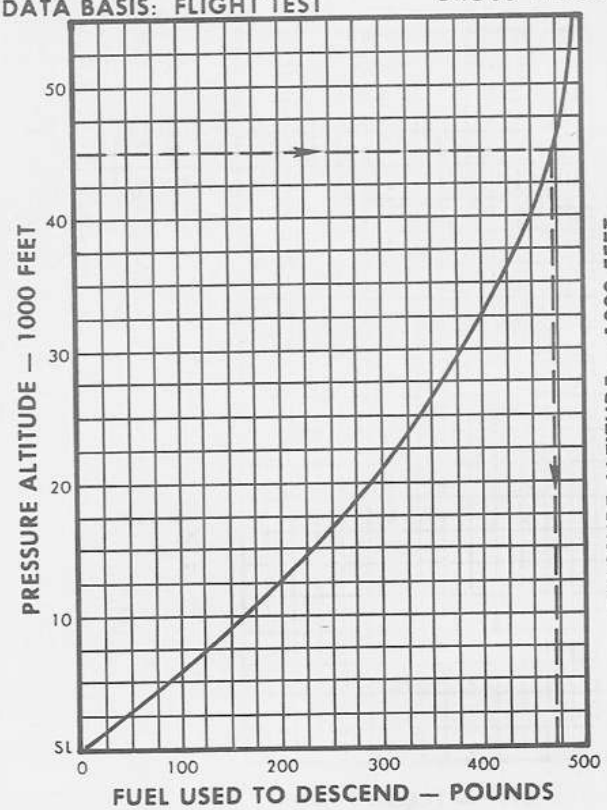
descents-recommended

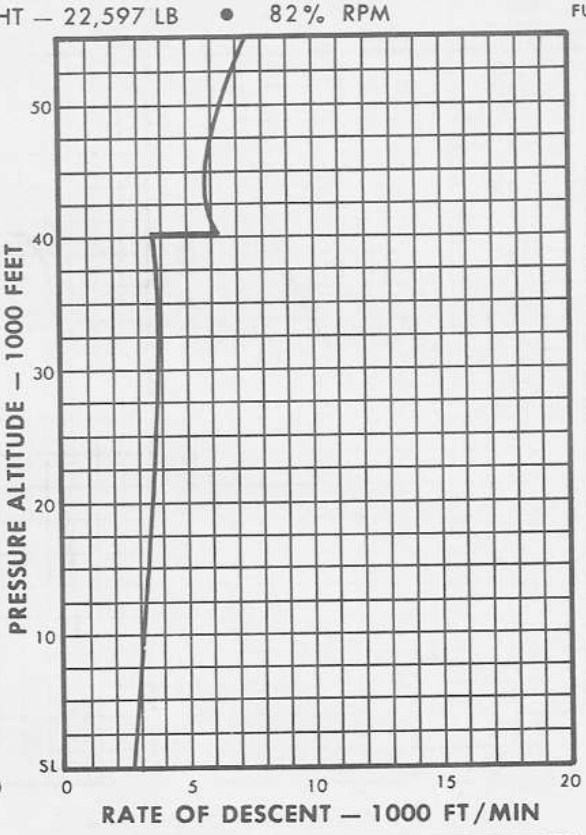
MODEL: F-102A
DATE: 15 OCTOBER 1958

CONFIGURATION: CLEAN • SPEED BRAKES OPEN • STANDARD ATMOSPHERE
GROSS WEIGHT — 22,597 LB • 82% RPM

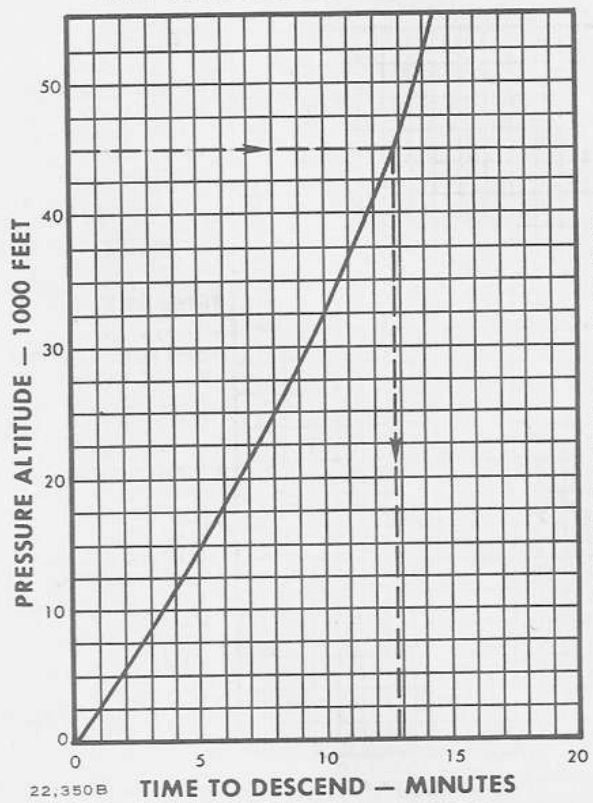
ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

DATA BASIS: FLIGHT TEST

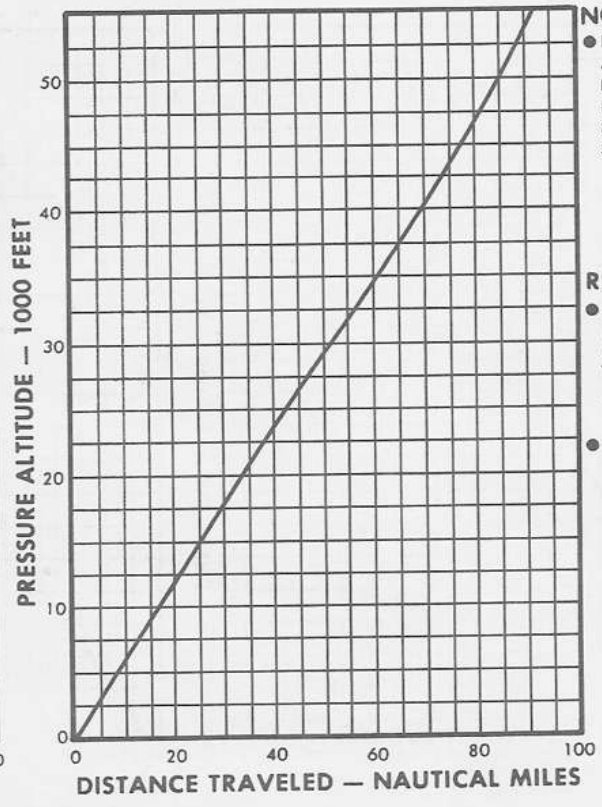




PRESS ALT 1000 FT	MACH NO.	CAS KNOTS
55	.90	195
50	.90	220
45	.90	246
40	.88	270
35	.79	270
30	.72	270
25	.65	270
20	.59	270
15	.54	270
10	.49	270
5	.45	270
SL	.41	270



22,350B



NOTE

- DURING OPERATION ABOVE 40,000 FEET, DO NOT REDUCE ENGINE RPM BELOW 85%. (REFER TO SECTION V).

REMARKS

- RECOMMENDED DESCENT — DESCEND AT 270 KCAS NOT TO EXCEED 0.90 MACH NO.
- THESE DATA ARE BASED ON 1500 POUNDS FUEL REMAINING AT ALTITUDE BUT ARE VALID FOR ALL FUEL REMAINING BECAUSE OF NEGLIGIBLE WEIGHT EFFECT.

Figure A6-2

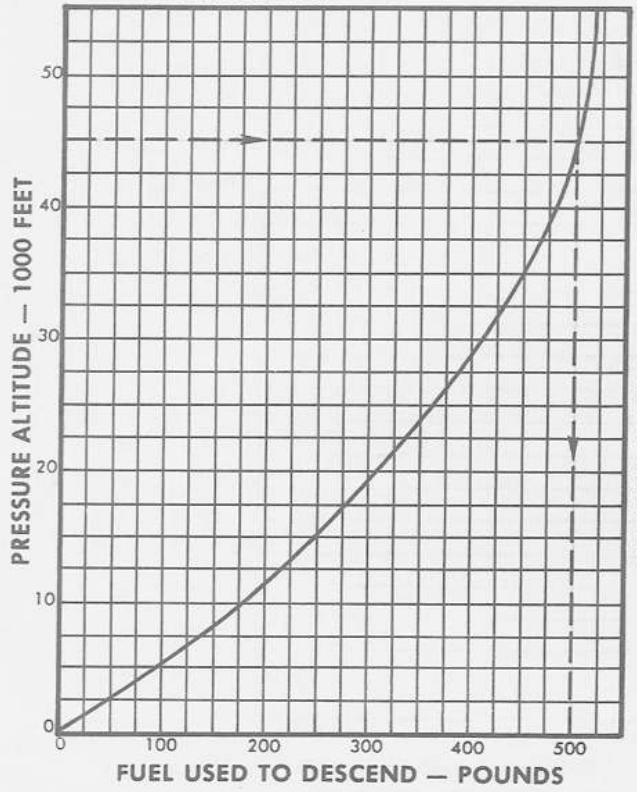
TF

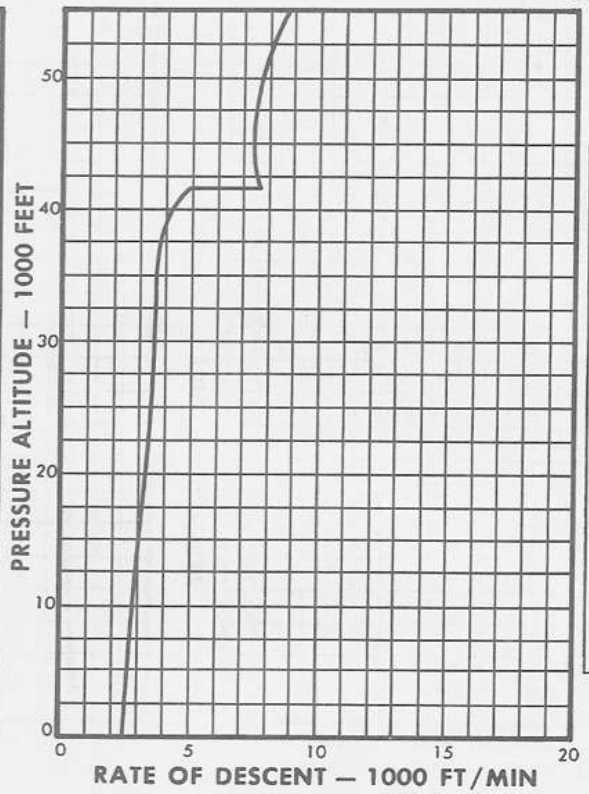
descents - recommended

MODEL: TF-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

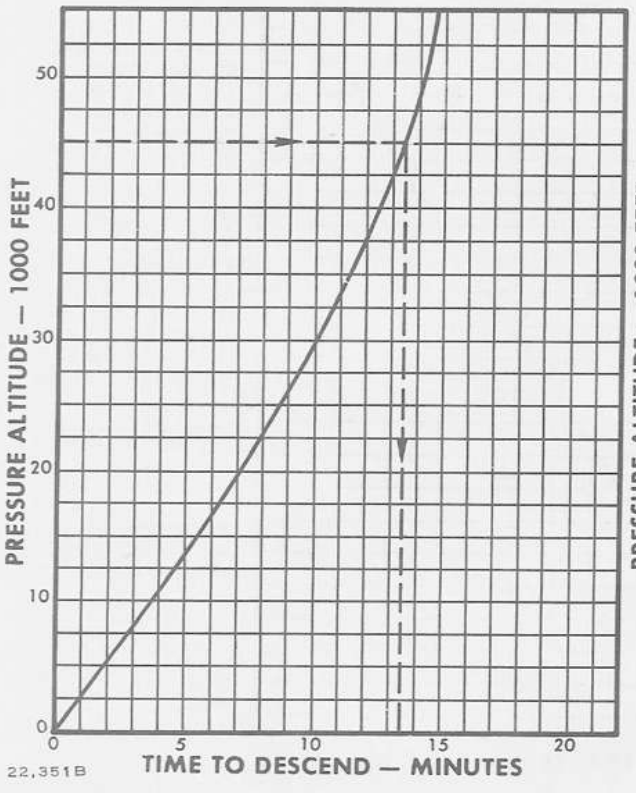
CONFIGURATION: CLEAN • SPEED BRAKES OPEN • STANDARD ATMOSPHERE
GROSS WEIGHT — 23,425 LB • 82% RPM

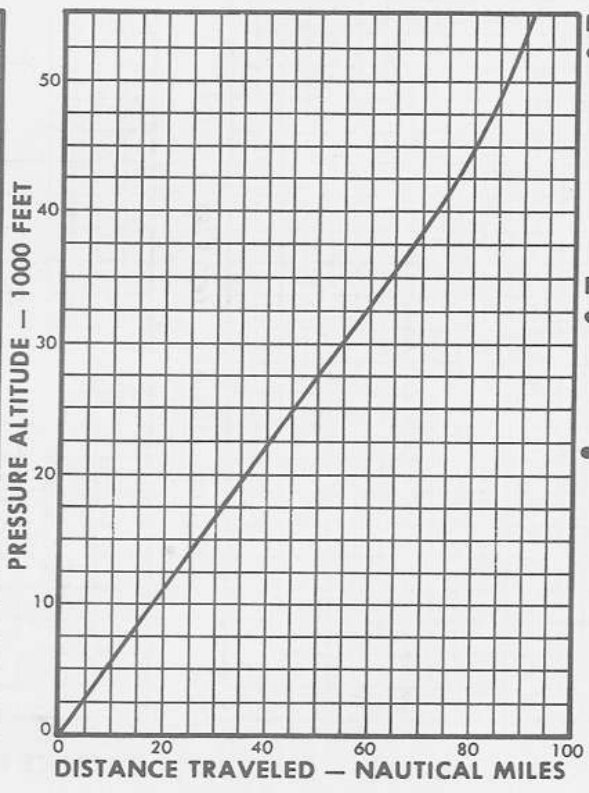
ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL





PRESS ALT 1000 FT	MACH NO.	CAS KNOTS
55	.90	195
50	.90	220
45	.90	246
40	.88	270
35	.79	270
30	.72	270
25	.65	270
20	.59	270
15	.54	270
10	.49	270
5	.45	270
SL	.41	270





22,351B

NOTE

- DURING OPERATION ABOVE 40,000 FEET, DO NOT REDUCE ENGINE RPM BELOW 85%. (REFER TO SECTION V).

REMARKS

- RECOMMENDED DESCENT — DESCEND AT 270 KCAS NOT TO EXCEED 0.90 MACH NO.
- THESE DATA ARE BASED ON 1500 POUNDS FUEL REMAINING AT ALTITUDE BUT ARE VALID FOR ALL FUEL REMAINING BECAUSE OF NEGLIGIBLE WEIGHT EFFECT.

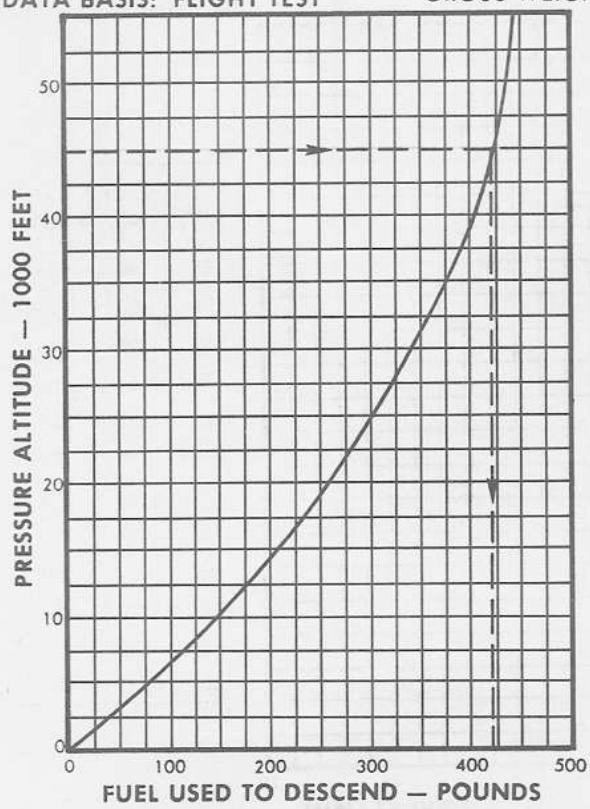
Figure A6-3

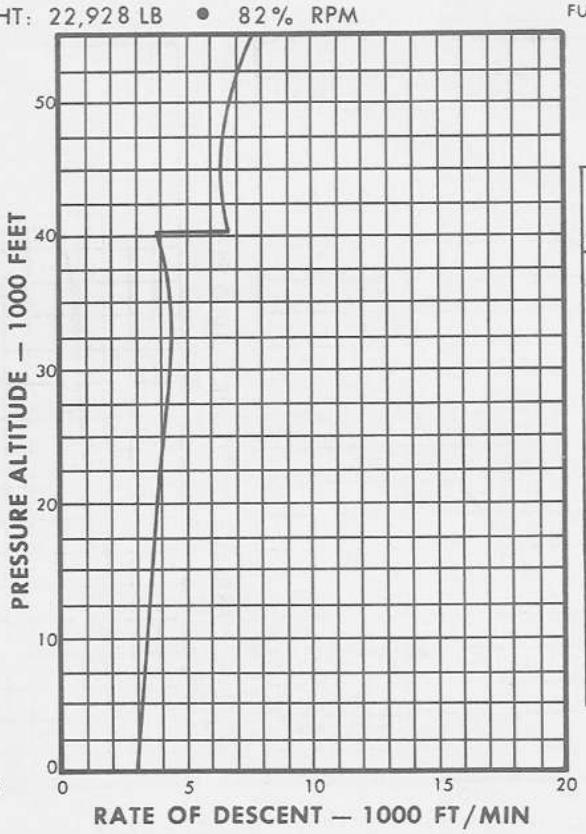
descents-recommended

CONFIGURATION: TWO 230-GAL EXTERNAL TANKS
SPEED BRAKES OPEN • STANDARD ATMOSPHERE
GROSS WEIGHT: 22,928 LB • 82% RPM

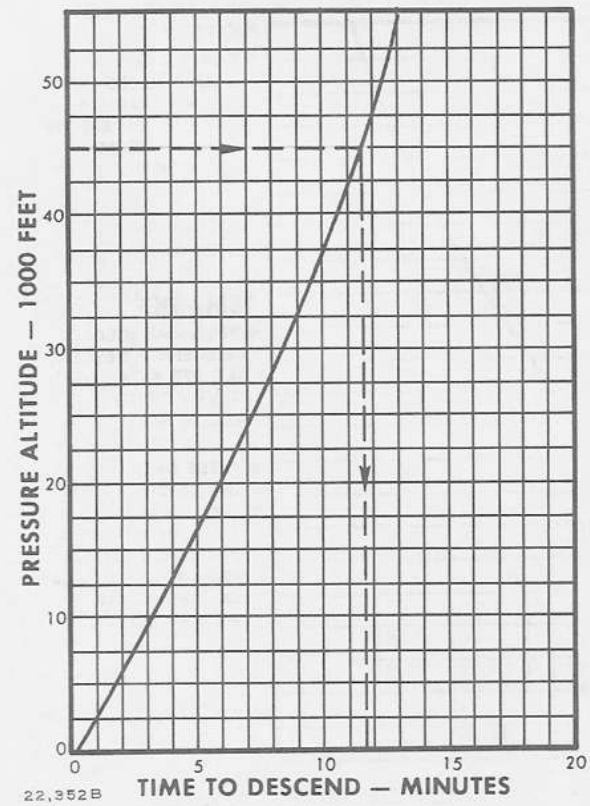
MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

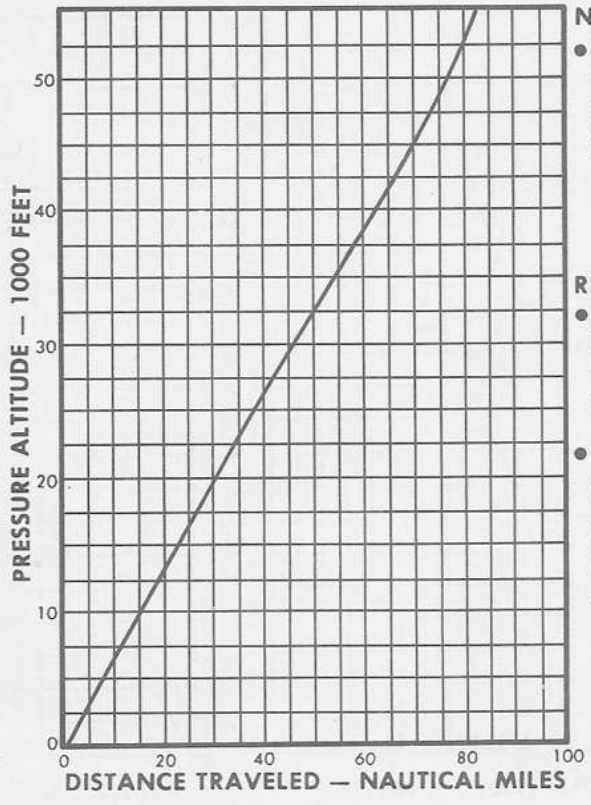
ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL





PRESS ALT 1000 FT	MACH NO.	CAS KNOTS
55	.90	195
50	.90	220
45	.90	246
40	.88	270
35	.79	270
30	.72	270
25	.65	270
20	.59	270
15	.54	270
10	.49	270
5	.45	270
SL	.41	270





NOTE

- DURING OPERATION ABOVE 40,000 FEET, DO NOT REDUCE ENGINE RPM BELOW 85%. (REFER TO SECTION V).

REMARKS

- RECOMMENDED DESCENT – DESCEND AT 270 KCAS NOT TO EXCEED 0.90 MACH NO.
- THESE DATA ARE BASED ON 1500 POUNDS FUEL REMAINING AT ALTITUDE BUT ARE VALID FOR ALL FUEL REMAINING BECAUSE OF NEGLIGIBLE WEIGHT EFFECT.

Figure A6-4

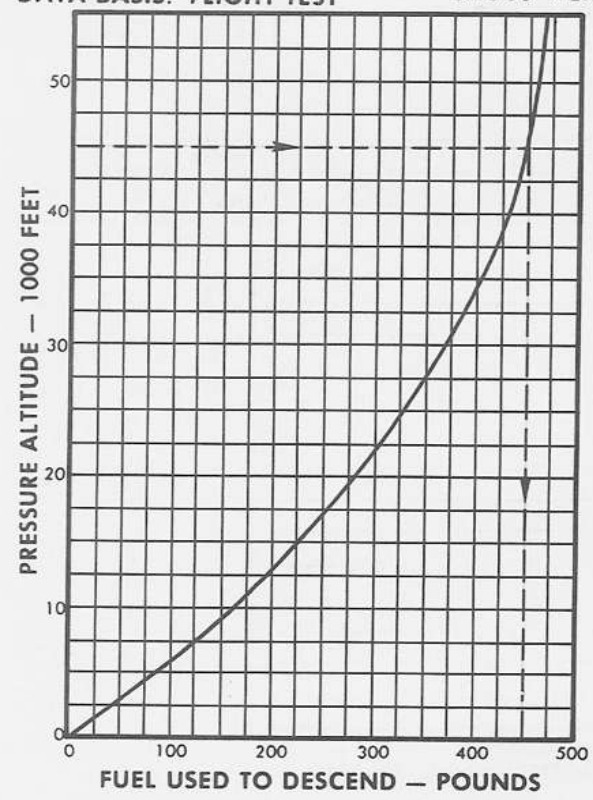
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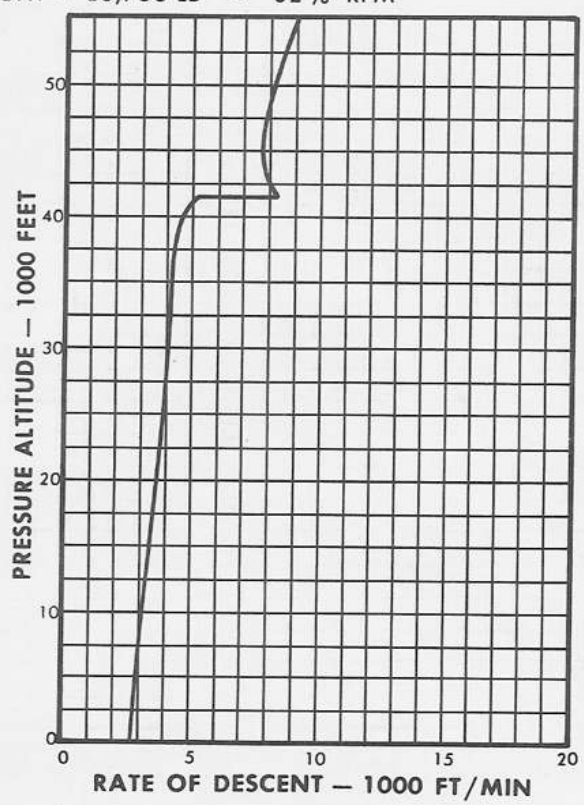
descents-recommended

MODEL: TF-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

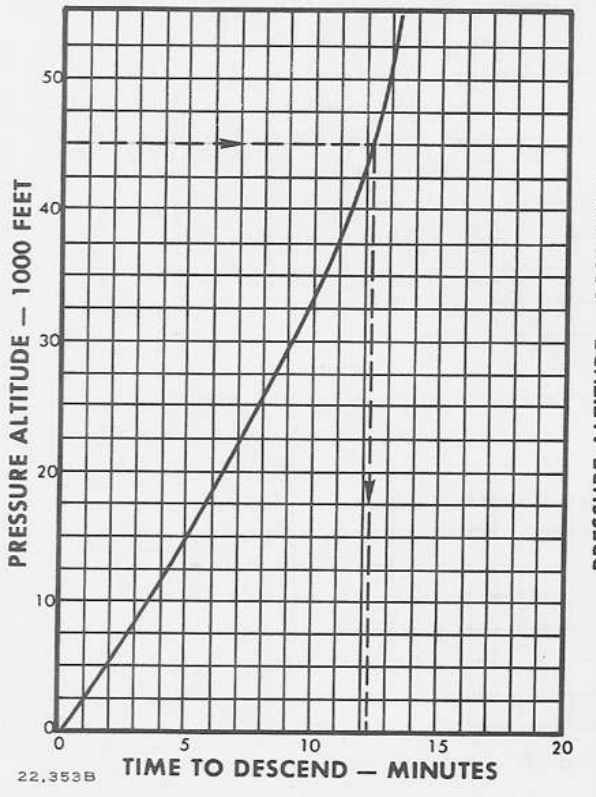
CONFIGURATION: TWO 230-GAL EXTERNAL TANKS
SPEED BRAKES OPEN • STANDARD ATMOSPHERE
GROSS WEIGHT — 23,756 LB • 82% RPM

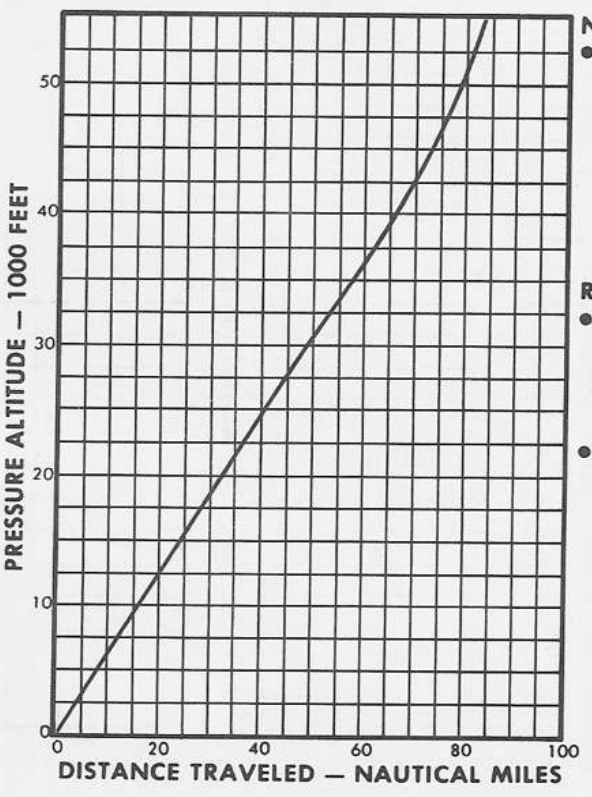
ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL





PRESS ALT 1000 FT	MACH NO.	CAS KNOTS
55	.90	195
50	.90	220
45	.90	246
40	.88	270
35	.79	270
30	.72	270
25	.65	270
20	.59	270
15	.54	270
10	.49	270
5	.45	270
SL	.41	270





NOTE

- DURING OPERATION ABOVE 40,000 FEET, DO NOT REDUCE ENGINE RPM BELOW 85%. (REFER TO SECTION V).

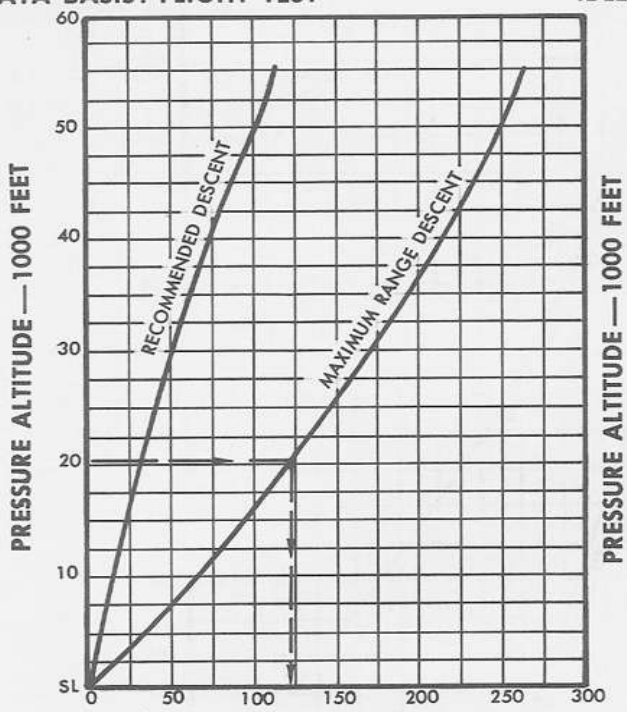
REMARKS

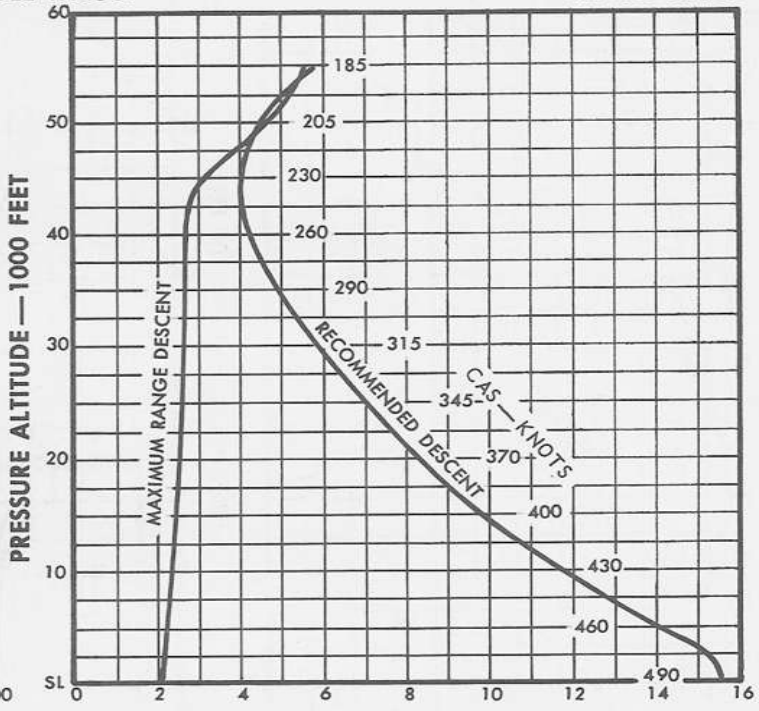
- RECOMMENDED DESCENT — DESCEND AT 270 KCAS NOT TO EXCEED 0.90 MACH NO.
- THESE DATA ARE BASED ON 1500 POUNDS FUEL REMAINING AT ALTITUDE BUT ARE VALID FOR ALL FUEL REMAINING BECAUSE OF NEGLIGIBLE WEIGHT EFFECT.

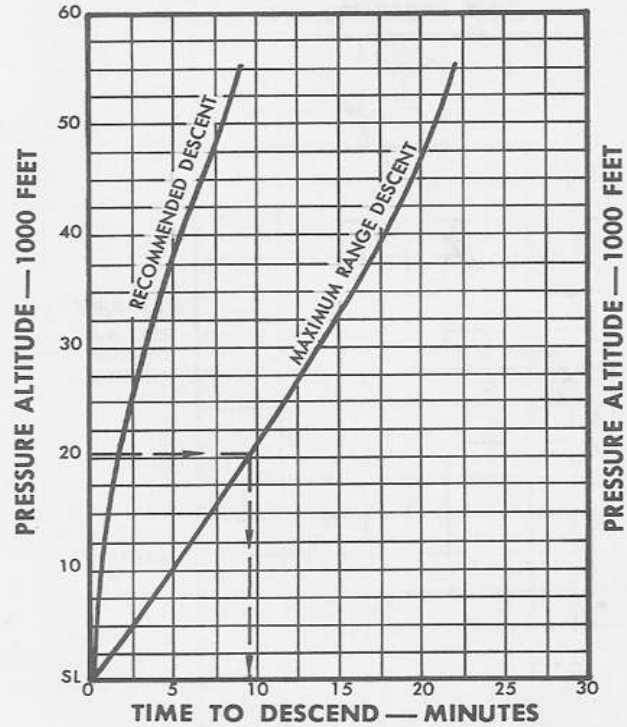
Figure A6-5

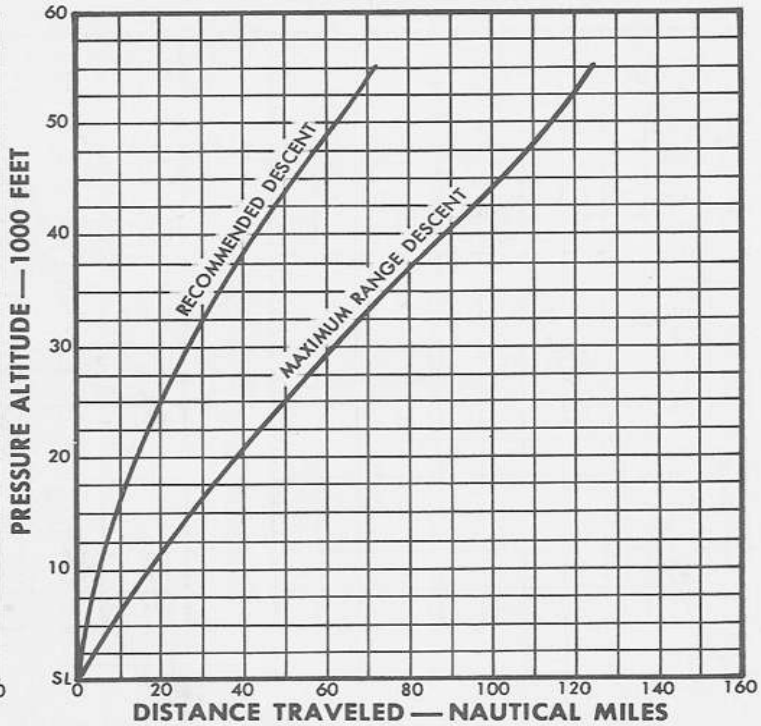
descents

MODEL: F-102A
DATE: 15 OCTOBER 1958
CONFIGURATION: CLEAN ● STANDARD ATMOSPHERE
GROSS WEIGHT: 22,000 LB ● SPEED BRAKES CLOSED
IDLE THRUST
ENGINE: J57-23
ALL FUEL GRADES
DATA BASIS: FLIGHT TEST









REMARKS

RECOMMENDED DESCENT—DESCEND AT SPEEDS NOTED ABOVE.
MAX RANGE DESCENT—DESCEND AT 215–235 KNOTS CAS; NOT TO EXCEED .90 MACH NO.

NOTE DURING OPERATION ABOVE 40,000 FT. DO NOT REDUCE ENGINE RPM BELOW 85%. (REFER TO SECTION V.)

22,023F

Figure A6-6

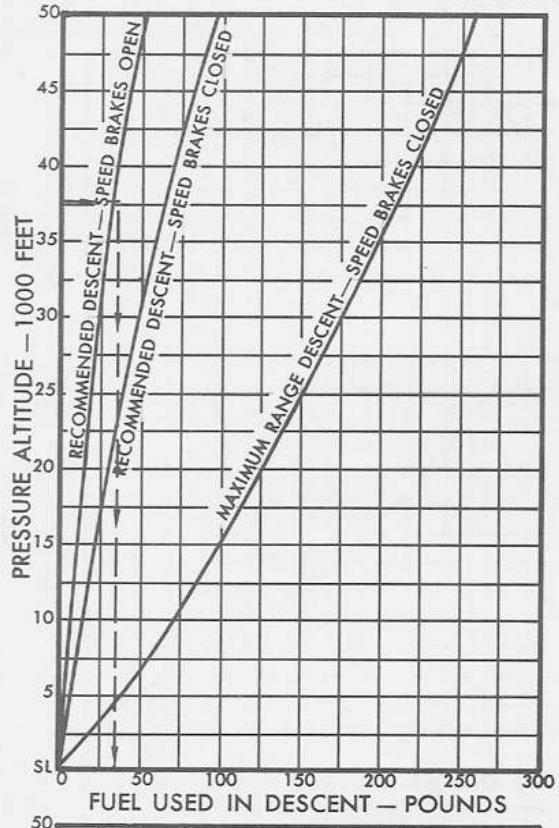
TF

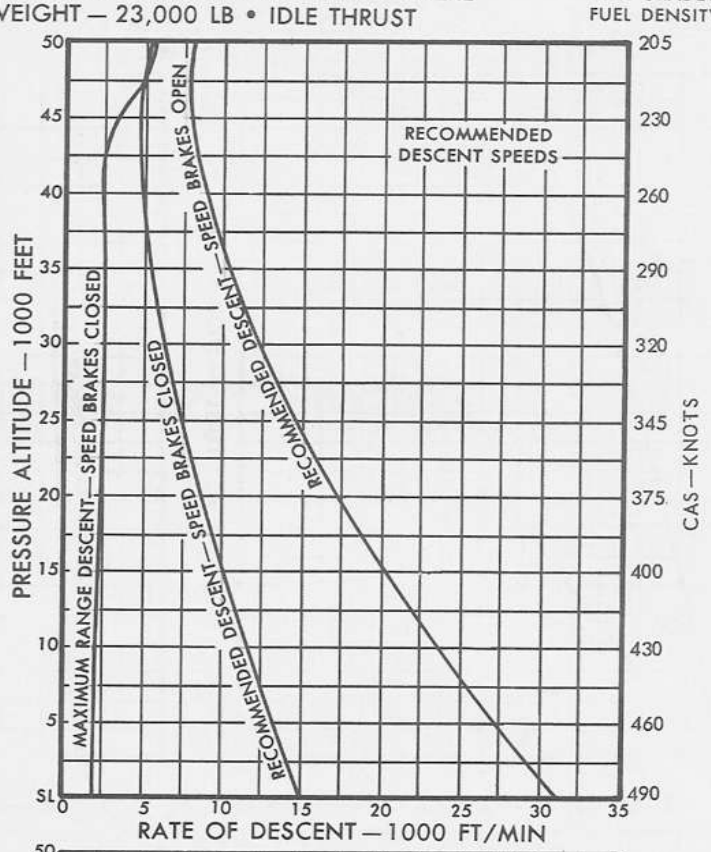
MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

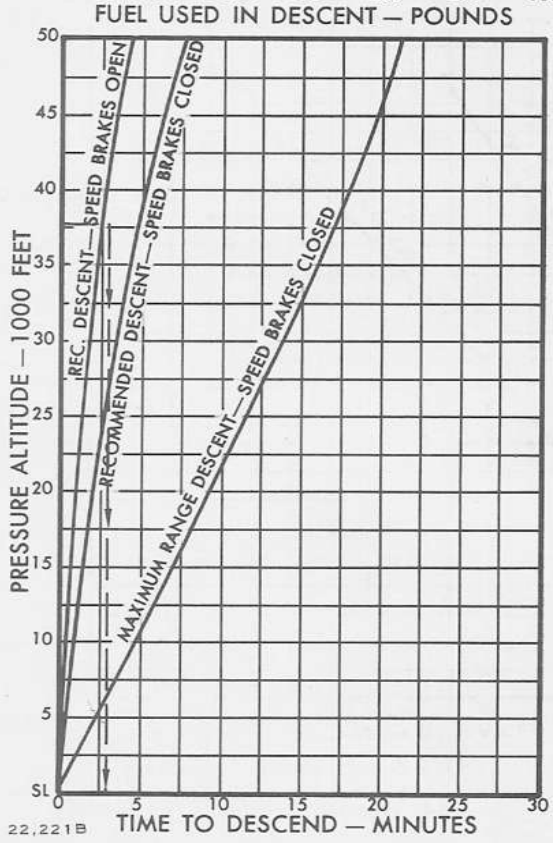
descents

CONFIGURATION: CLEAN • STANDARD ATMOSPHERE
GROSS WEIGHT — 23,000 LB • IDLE THRUST

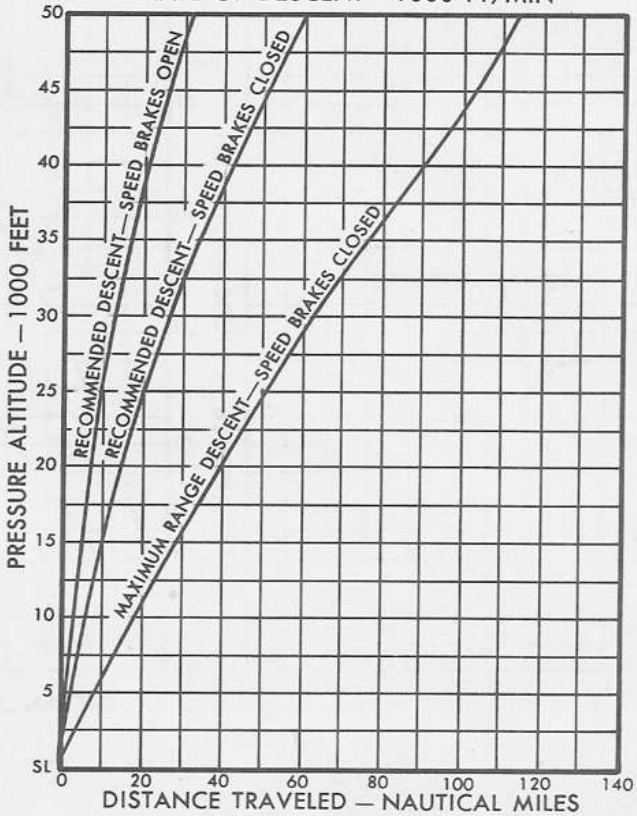
ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL







22,221B



NOTE

DURING OPERATION ABOVE 40,000 FEET, DO NOT REDUCE THROTTLE BELOW 85% RPM. (REFER TO SECTION V.)

REMARKS

RECOMMENDED DESCENT—DESCEND AT SPEEDS NOTED ABOVE.

MAXIMUM RANGE DESCENT—DESCEND AT 215–220 KNOTS CAS; NOT TO EXCEED .85 MACH NO.

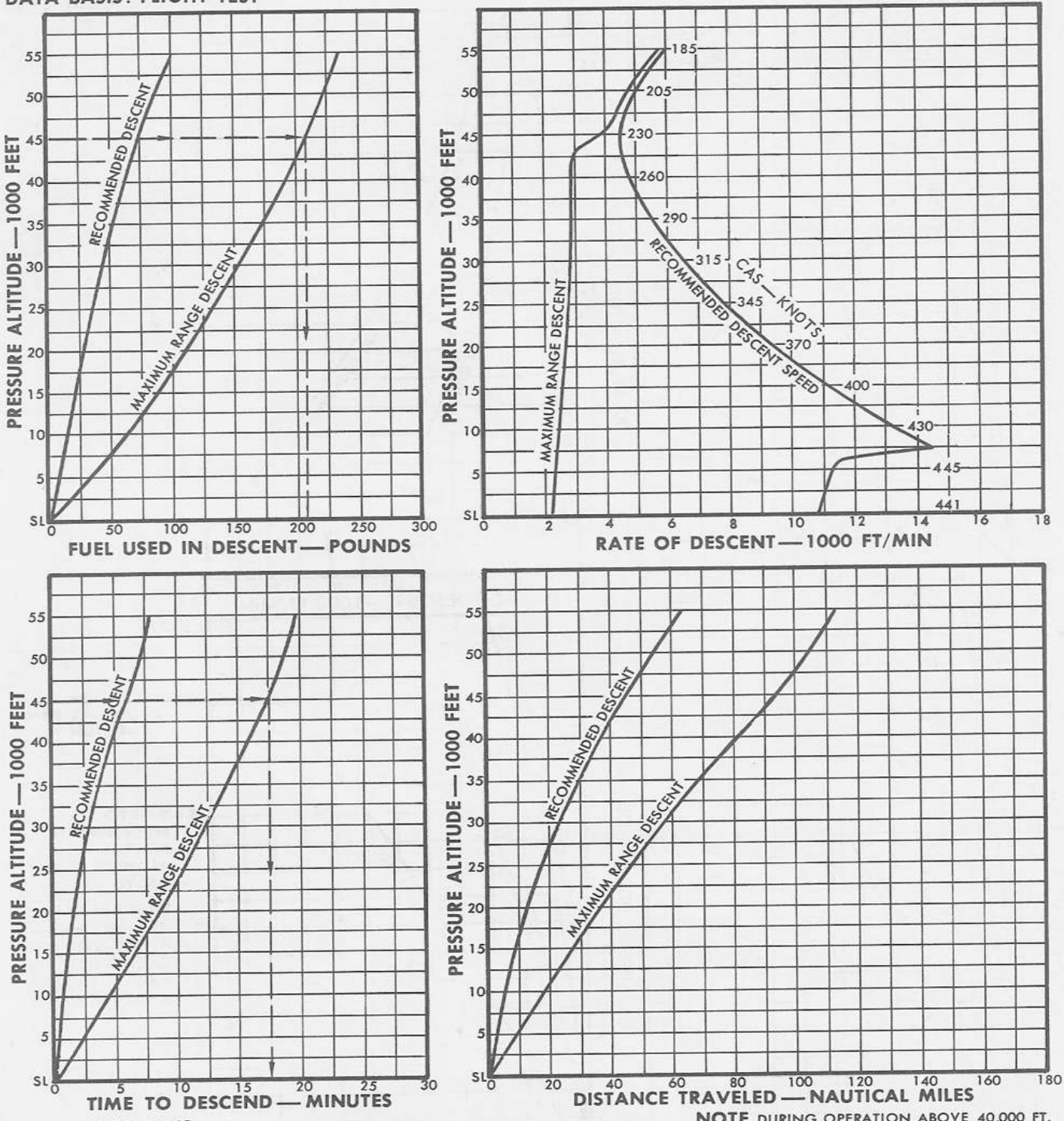
Figure A6-7

descents

CONFIGURATION: 2-230 GAL. TANKS ● GROSS WEIGHT: 22,000 LB
IDLE THRUST ● SPEED BRAKES CLOSED

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
ALL FUEL GRADES



REMARKS

RECOMMENDED DESCENT—DESCEND AT SPEEDS NOTED ABOVE.
MAXIMUM RANGE DESCENT—DESCEND AT 215–235 KNOTS CAS; NOT TO EXCEED .90 MACH NO.

NOTE DURING OPERATION ABOVE 40,000 FT. DO NOT REDUCE ENGINE RPM BELOW 85%. (REFER TO SECTION V.)

22.024G

Figure A6-8

TF

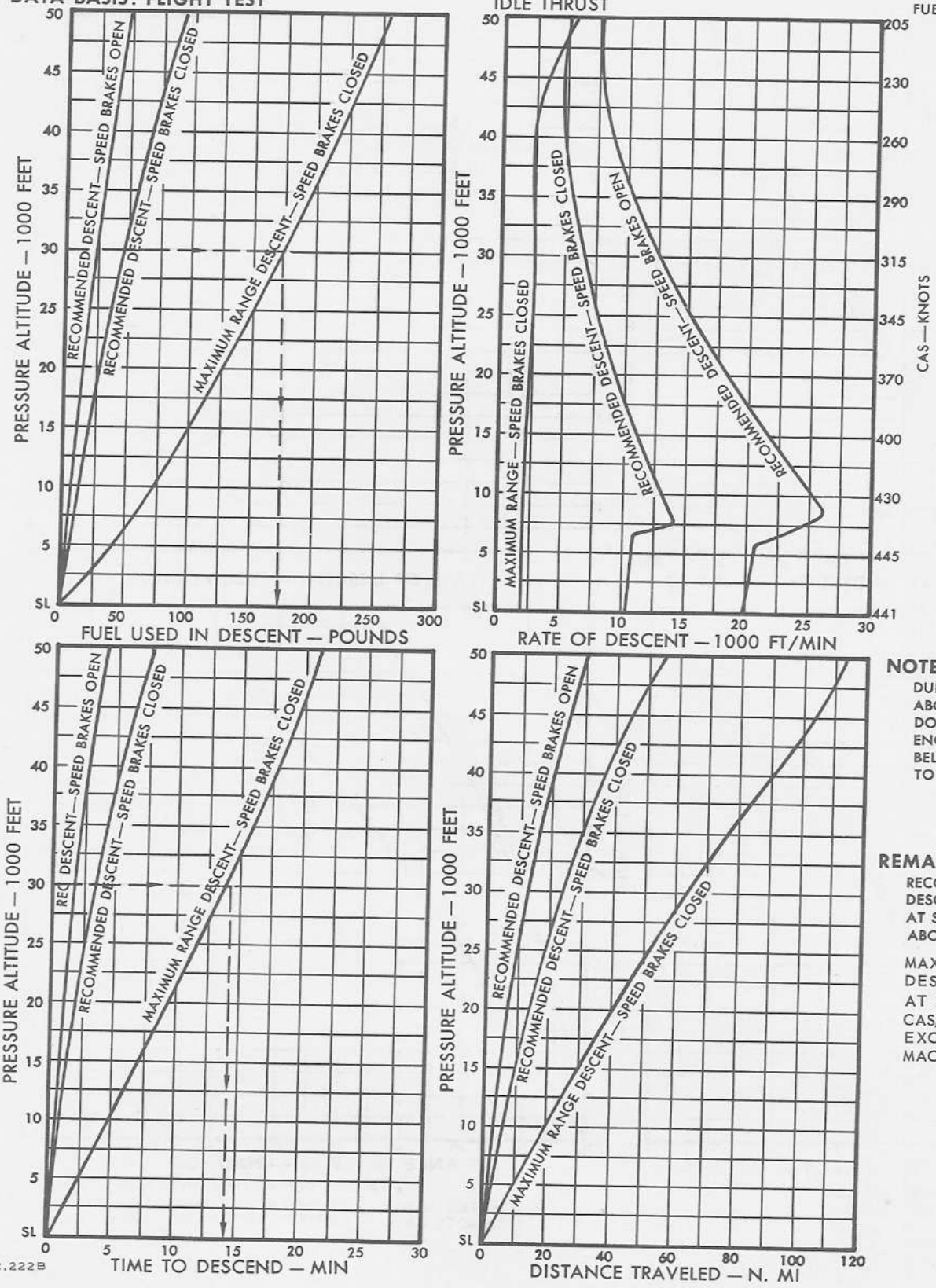
descents

MODEL: TF-102A
DATE: 1 NOVEMBER 1959

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
GROSS WEIGHT — 23,000 LB • STANDARD ATMOSPHERE
IDLE THRUST

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

DATA BASIS: FLIGHT TEST



NOTE

DURING OPERATION ABOVE 40,000 FEET, DO NOT REDUCE ENGINE RPM BELOW 85% (REFER TO SECTION V).

REMARKS

RECOMMENDED DESCENT— DESCEND AT SPEEDS NOTED ABOVE.

MAXIMUM RANGE DESCENT—DESCEND AT 215–220 KNOTS CAS; NOT TO EXCEED .85 MACH NO.

Figure A6-9

Part 7 APPROACH AND LANDING

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Page

Landing Distances Chart A7-2

LANDING DISTANCES

Landing distances are shown for two basic configurations: speed brakes open (figure A7-1 and figure A7-2), and speed brakes open with drag chute deployed at touchdown (figure A7-3 and figure A7-4). Either chart is valid for use with or without external tanks. As indicated on the sample problem, these charts account for the effects of ambient temperature, pressure altitude, gross weight and headwind for ground roll and total distance including the landing flare over a 50-foot obstacle.

Use

Enter the chart at a given ambient temperature, then read vertically up to intersect the field pressure altitude. Proceed horizontally to the right to the amount of fuel remaining, then read vertically down to the braking conditions base line and parallel lines to the braking condition. Read vertically down to zero headwind base line and parallel lines to the amount of headwind. Then read vertically down to zero increase in recommended touchdown speed base line and parallel lines to the increase in recommended touchdown speed. Read vertically down for ground roll and distance to clear a 50-foot obstacle.

Sample Problem

Given: Air temperature of 15°C, pressure altitude at sea level, and 300 pounds fuel remaining.

Determine: The ground roll and total distance to clear a 50-foot obstacle and land, under good braking conditions in a 15 knot headwind, when touching down 10 knots faster than recommended.

A. Enter figure A7-1 with the ambient air temperature—15°C or 59°F.

B. Read up to the pressure altitude—sea level.

C. Read horizontally to the right, for the fuel remaining—3000 pounds.

D. Read down to base line for all types of braking.

E. Follow parallel lines to good braking conditions.

F. Read down to zero headwind base line.

G. Follow parallel lines to a 15 knot headwind.

H. Read down to zero increase in recommended touchdown speed base line.

I. Follow parallel lines to 10 knots.

J. Read down for ground roll distance—3700 feet.

K. Read down for total distance to clear 50-foot obstacle—5200 feet.

CROSSWIND LANDINGS

The minimum touchdown speeds for a given crosswind condition are determined from the Takeoff and Landing Crosswind Chart, figure A2-2.

Use

The use of this chart is explained in part 2 titled "Takeoff and Landing Crosswind Chart."

Sample Problem

Given: A wind of 25 knots, gusts to 33 knots, 65° relative to runway heading.

Determine: The headwind and crosswind components and the minimum touchdown speed with 3000 pounds of fuel remaining.

A. Enter figure A2-2 on the wind angle line—65°.

B. Proceed to the steady wind velocity circle—25 knots.

C. Move horizontally to the left and read the headwind component—10.6 knots (used to obtain the landing distance from the landing charts).

D. Proceed along the wind angle line to the maximum gust velocity circle—33 knots.

E. Move vertically downward and read the crosswind component—30 knots.

F. Enter table with crosswind component and read the minimum touchdown speed—164 knots CAS.

Compute the gust factor (33 knots—25 knots) × ½ = 4 knots. The recommended touchdown speed with 3000 pounds of fuel remaining is 141 knots CAS (figure A7-1). Add the gust factor to the recommended touchdown speed—145 knots CAS. Since the value obtained in the crosswind chart is higher it should be used.

Note

If the touchdown speed is increased over the recommended touchdown speed the landing distance will be increased (see figures A7-1 and A7-3).

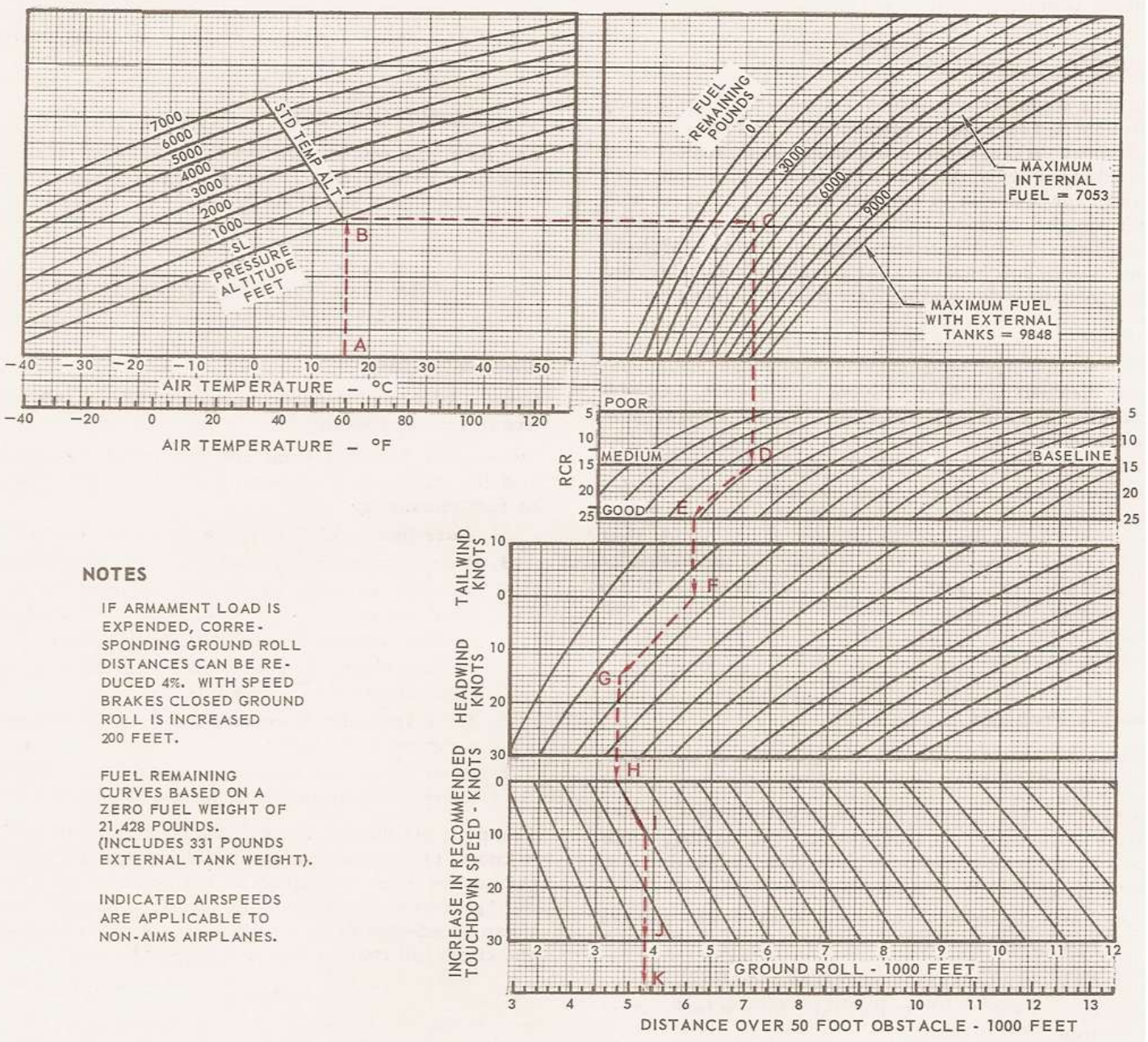
F

landing distances

CONFIGURATION: SPEED BRAKES OPEN
(DRAG CHUTE NOT DEPLOYED)
HARD SURFACE RUNWAY ARMAMENT IN

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

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



NOTES

IF ARMAMENT LOAD IS EXPENDED, CORRESPONDING GROUND ROLL DISTANCES CAN BE REDUCED 4%. WITH SPEED BRAKES CLOSED GROUND ROLL IS INCREASED 200 FEET.

FUEL REMAINING CURVES BASED ON A ZERO FUEL WEIGHT OF 21,428 POUNDS. (INCLUDES 331 POUNDS EXTERNAL TANK WEIGHT).

INDICATED AIRSPEEDS ARE APPLICABLE TO NON-AIMS AIRPLANES.

RECOMMENDED SPEEDS*						
FUEL REMAINING POUNDS	BASE		FINAL APPROACH		TOUCHDOWN	
	CAS	IAS	CAS	IAS	CAS	IAS
0 – 2500	180	175	170	165	130	125
2500 – 3500	185	180	175	170	135	130
3500 – 4500	190	185	180	175	140	135
4500 – 6500	195	190	185	180	145	140
Over 6500	200	195	190	185	150	145

22.026B

*ADD 5 KNOTS FOR FULL ARMAMENT LOAD.
ADD 5 KNOTS FOR TF WITH AIMS.

Figure A7-1

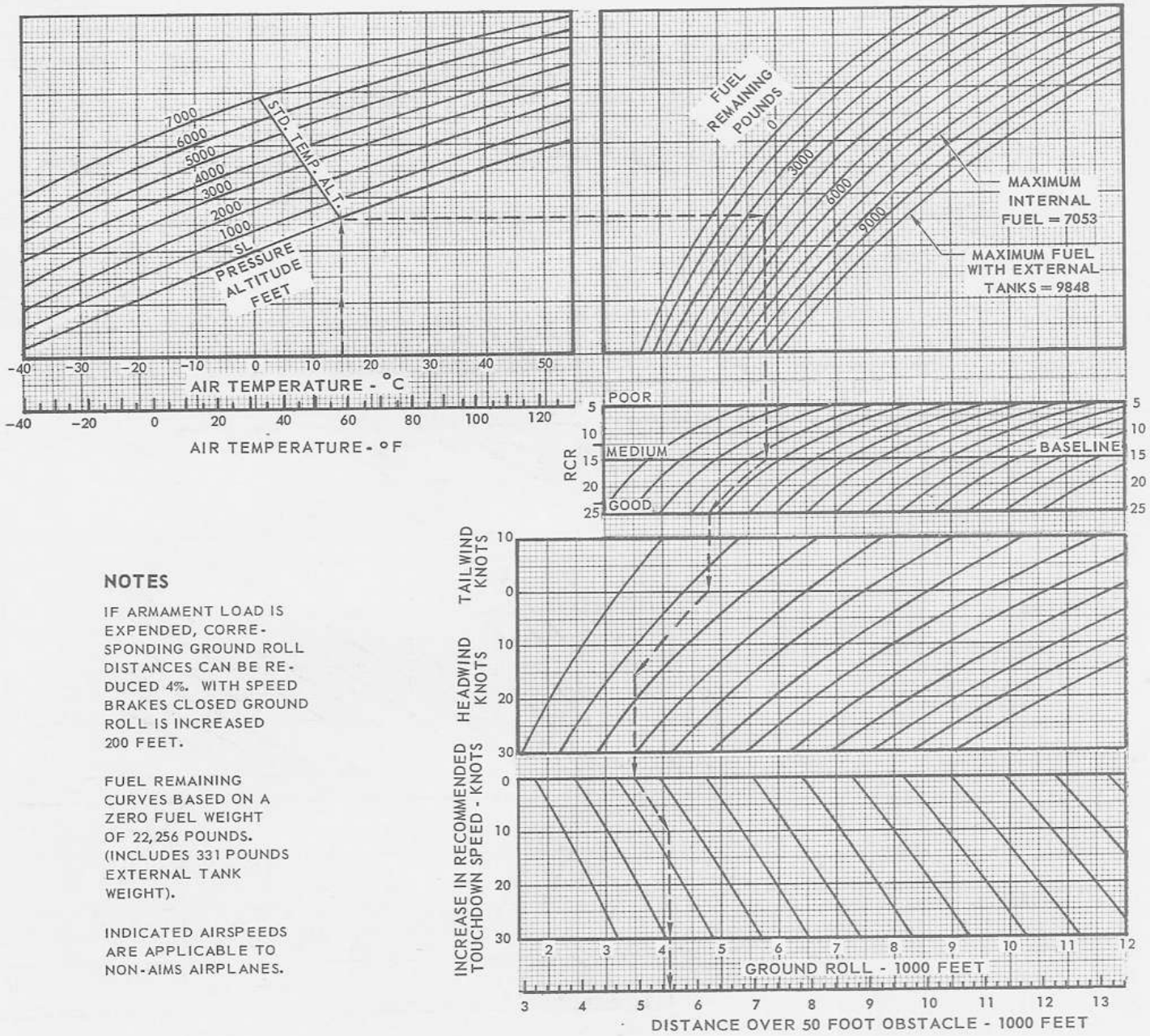
TF

landing distances

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: SPEED BRAKES OPEN
(DRAG CHUTE NOT DEPLOYED)

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



NOTES

IF ARMAMENT LOAD IS EXPENDED, CORRESPONDING GROUND ROLL DISTANCES CAN BE REDUCED 4%. WITH SPEED BRAKES CLOSED GROUND ROLL IS INCREASED 200 FEET.

FUEL REMAINING CURVES BASED ON A ZERO FUEL WEIGHT OF 22,256 POUNDS. (INCLUDES 331 POUNDS EXTERNAL TANK WEIGHT).

INDICATED AIRSPEEDS ARE APPLICABLE TO NON-AIMS AIRPLANES.

RECOMMENDED SPEEDS*						
FUEL REMAINING POUNDS	BASE		FINAL APPROACH		TOUCHDOWN	
	CAS	IAS	CAS	IAS	CAS	IAS
0 – 2500	180	175	170	165	130	125
2500 – 3500	185	180	175	170	135	130
3500 – 4500	190	185	180	175	140	135
4500 – 6500	195	190	185	180	145	140
Over 6500	200	195	190	185	150	145

*ADD 5 KNOTS FOR FULL ARMAMENT LOAD.
ADD 5 KNOTS FOR TF WITH AIMS.

22,225B

Figure A7-2

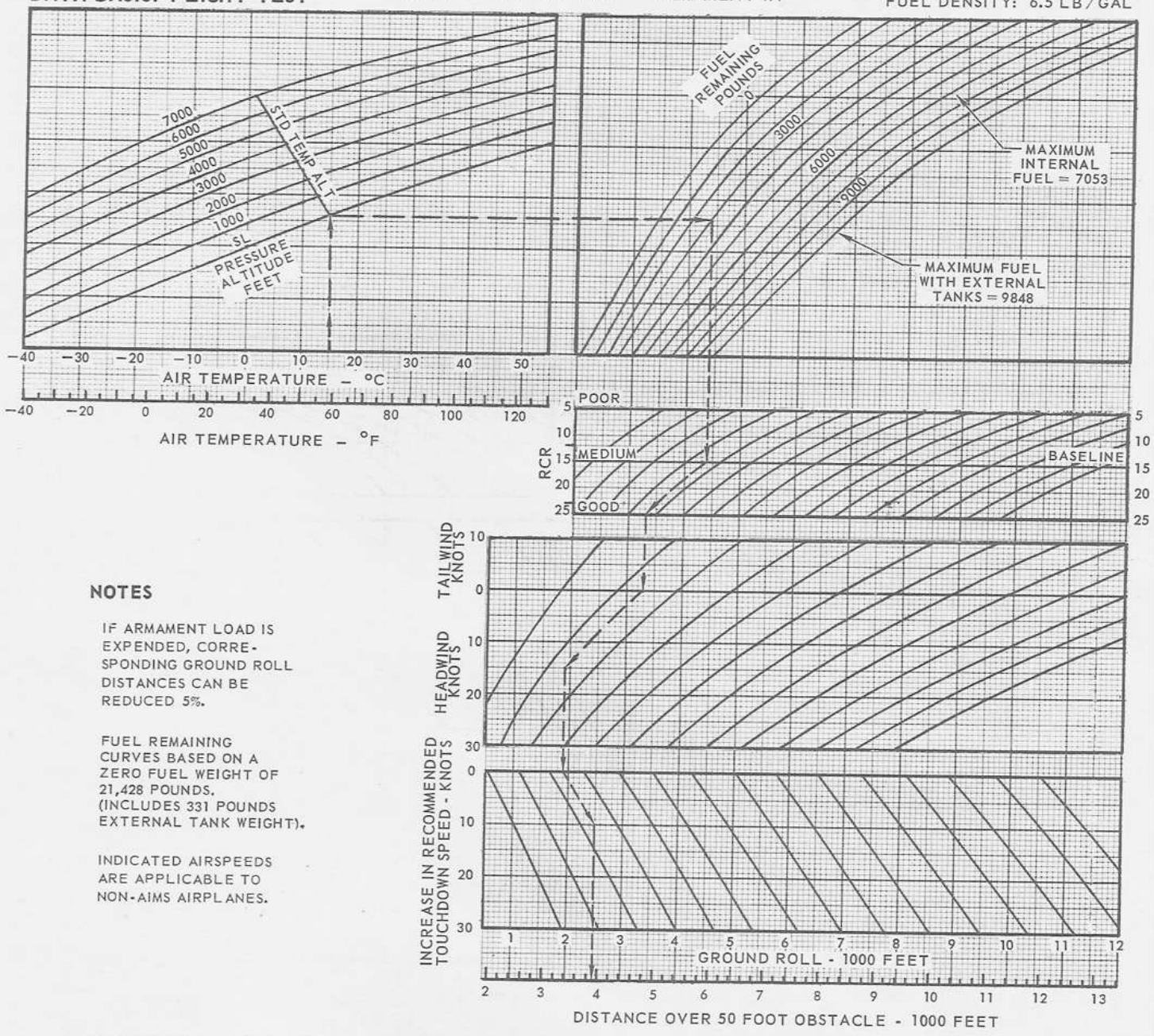
F

landing distances

CONFIGURATION: SPEED BRAKES OPEN
(DRAG CHUTE DEPLOYED)
HARD SURFACE RUNWAY ARMAMENT IN

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

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



NOTES

IF ARMAMENT LOAD IS EXPENDED, CORRESPONDING GROUND ROLL DISTANCES CAN BE REDUCED 5%.

FUEL REMAINING CURVES BASED ON A ZERO FUEL WEIGHT OF 21,428 POUNDS. (INCLUDES 331 POUNDS EXTERNAL TANK WEIGHT).

INDICATED AIRSPEEDS ARE APPLICABLE TO NON-AIMS AIRPLANES.

RECOMMENDED SPEEDS*						
FUEL REMAINING POUNDS	BASE		FINAL APPROACH		TOUCHDOWN	
	CAS	IAS	CAS	IAS	CAS	IAS
0 – 2500	180	175	170	165	130	125
2500 – 3500	185	180	175	170	135	130
3500 – 4500	190	185	180	175	140	135
4500 – 6500	195	190	185	180	145	140
Over 6500	200	195	190	185	150	145

22.028L

*ADD 5 KNOTS FOR FULL ARMAMENT LOAD.
ADD 5 KNOTS FOR TF WITH AIMS.

Figure A7-3

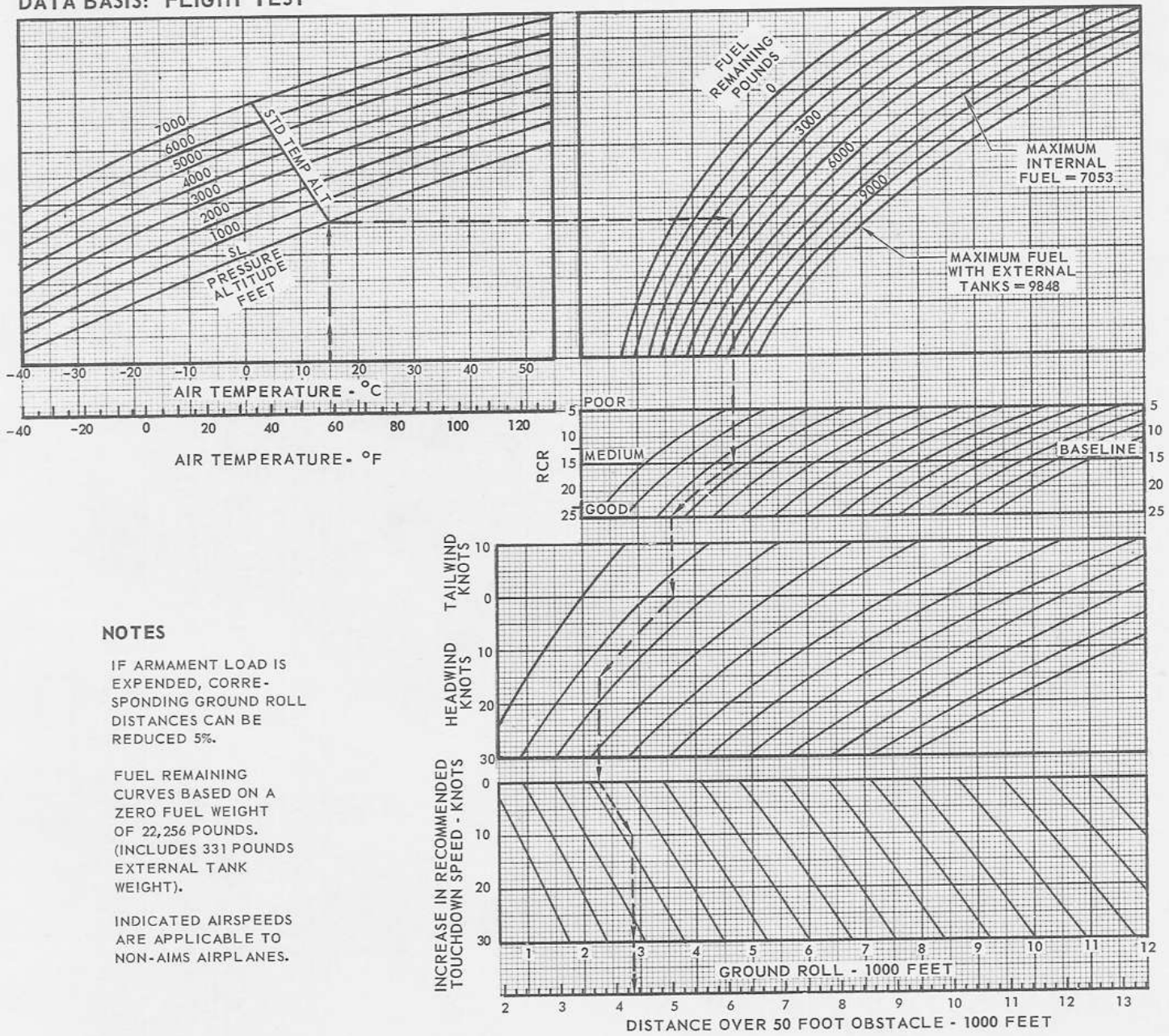
TF

landing distances

MODEL: TF-102A
DATE: 1 NOVEMBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: SPEED BRAKES OPEN
(DRAG CHUTE DEPLOYED)

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



NOTES

IF ARMAMENT LOAD IS EXPENDED, CORRESPONDING GROUND ROLL DISTANCES CAN BE REDUCED 5%.

FUEL REMAINING CURVES BASED ON A ZERO FUEL WEIGHT OF 22,256 POUNDS. (INCLUDES 331 POUNDS EXTERNAL TANK WEIGHT).

INDICATED AIRSPEEDS ARE APPLICABLE TO NON-AIMS AIRPLANES.

RECOMMENDED SPEEDS*						
FUEL REMAINING POUNDS	BASE		FINAL APPROACH		TOUCHDOWN	
	CAS	IAS	CAS	IAS	CAS	IAS
0 – 2500	180	175	170	165	130	125
2500 – 3500	185	180	175	170	135	130
3500 – 4500	190	185	180	175	140	135
4500 – 6500	195	190	185	180	145	140
Over 6500	200	195	190	185	150	145

*ADD 5 KNOTS FOR FULL ARMAMENT LOAD.
ADD 5 KNOTS FOR TF WITH AIMS.

22.226B

Figure A7-4

Part 8 ABBREVIATED CHECKLIST PERFORMANCE

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Scope of Checklist Performance Charts	A8-1
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Cruise Charts	A8-1
Optimum Return Tables	A8-2
Landing Charts	A8-3

SCOPE OF CHECKLIST PERFORMANCE CHARTS

The performance charts shown in the Pilot's Abbreviated Flight Crew Checklist, T.O. 1F-102A-1CL-1, replace the MB-8 computer. These charts provide inflight performance data and are based on the following assumptions:

1. Armament in.
2. Takeoff data based on full fuel load, 2000-foot marker for check speeds, and 8000-foot runway for refusal speeds.
3. 1000 pounds fuel allowance from engine start to climb speed.
4. On (F) airplanes cruise data based on 24,500 pounds gross weight for the clean configuration and 26,000 pounds gross weight for the tank configuration.
5. On (TF) airplanes cruise data based on 25,500 pounds gross weight for the clean configuration and 27,000 pounds gross weight for the tank configuration.
6. Cruise conditions: 290 KCAS, optimum range cruise speed, and Mach 0.90.
7. Saunter data based on 1500 pounds of fuel remaining.
8. Descent data based on speed brakes open 82% rpm, and 270 KCAS (not to exceed Mach 0.90).

The accuracy of these charts when used for weights or conditions other than those which the data is presented is explained in the discussion of each individual performance phase. Refer to other parts of the Appendix for data involving flight planning for weights or conditions other than those presented.

Note

The checklist has included indicated airspeeds (IAS) and indicated Mach (IMN) where appropriate to preclude pilots flying non-AIMS airplanes having to convert numerous figures from calibrated airspeed (CAS) and true Mach.

TAKEOFF CHARTS

Takeoff performance is presented for maximum and military thrust takeoffs. The charts are divided into clean and external tanks configurations with takeoff roll distance, takeoff check speed, and refusal speed information presented on each chart. It might be noted that for low temperatures and altitudes takeoff speed is reached before refusal speed, and in some instances, before check speed is reached.

TAKEOFF ROLL DISTANCE

Takeoff roll distances are based on the following fuel conditions:

CLEAN		
	Total Fuel	**Gross Weight**
F-102A	7053 pounds	28,150 pounds
TF-102A	7053 pounds	28,978 pounds

WITH EXTERNAL TANKS		
	Total Fuel	**Gross Weight**
F-102A	9848 pounds	31,276 pounds
TF-102A	9848 pounds	32,104 pounds

Distances are valid either with or without armament.

Pressure altitude reads across the top of the chart and ambient temperature reads down the side of the chart. Linear interpolations are valid for temperatures and altitudes between the ones presented. After the takeoff roll distances have been established for the proper temperature and pressure altitude, the wind correction can be made. The wind correction is obtained from the bottom of the chart. Wind corrections are shown for various zero wind takeoff roll distances. Interpolate for winds or roll distances other than those presented. Add or subtract the resulting wind corrections to the zero wind roll distances. These wind corrections are based on the headwind correction shown in Part 2 of the Appendix and are, in general, shorter distances than would be obtained if the tailwind correction were used.

TAKEOFF CHECK SPEED

Takeoff check speeds are valid for the 2000-foot marker only. The takeoff check speed is obtained from the chart exactly the same way the takeoff roll distance was obtained. The charts are corrected for wind by adding the headwind and subtracting the tailwind.

REFUSAL SPEED

Refusal speeds are valid for an 8000-foot runway only. The refusal speed is obtained in the same manner as takeoff roll distance. The charts are corrected for wind by adding the headwind and subtracting the tailwind.

CRUISE CHARTS

Cruise performance is presented for the clean and the external wing tank configuration. The charts show climb, cruise, saunter and descent information.

CLIMB

Climb data are presented for both military and maximum thrust. The climb schedules are as follows:

F-102A

Military thrust: 330 KCAS to 0.89 Mach
Maximum thrust: 410 KCAS to 0.94 Mach

TF-102A

Military thrust: 335 KCAS to 0.79 Mach
Maximum thrust: 365 KCAS to 0.90 Mach

Maximum thrust climbs in the checklist utilize maximum thrust to accelerate to reach climb speed. Military thrust climbs in the checklist utilize maximum thrust to accelerate to 258 KCAS, then military thrust to accelerate to climb speed. Time for climb is shown from brake release. Climb data are based on a full fuel load less 1000 pounds. 1000 pounds of fuel are assumed used from engine start to climb speed. The fuel remaining values on the charts reflect this 1000 pounds. To correct these fuel remaining values to allowances other than 1000 pounds, add 1000 pounds to the values, then subtract the preferred allowance. Climb data are based on standard temperatures and correction factors shown on the climb charts in Part 3 of the Appendix should be used for non-standard temperatures.

CRUISE

Cruise data are presented for three cruise conditions. They are:

1. 290 KCAS.
2. Optimum range cruise speed.
3. Mach 0.90 indicated.

At altitudes where afterburner operation is required to maintain level flight at 0.90 indicated Mach, the speed values given in the checklist correspond to high speed with military thrust. This condition may be noted to exist for the TF-102A clean and the F-102A with external tanks at low altitudes and for the TF-102A with external tanks at both low and high altitudes. The optimum range cruise speed values are underlined. Cruise data are based on one gross weight for the clean configuration (24,500 pounds for the F-102A and 25,500 pounds for the TF-102A) and one gross weight for the two 230 gallon external tank configurations (26,000 pounds for the F-102A and 27,000 for the TF-102A). Data are to be used for all gross weights. In a ferry mission the error in using one gross weight would cancel. An error of approximately five percent would be realized if these data were used for an outbound or an inbound phase only. The effect of wind can be corrected for in the same way as shown on the cruise chart in Part 4 of the Appendix. This correction is repeated here for convenience.

$$\text{(nautical miles/pounds [with wind])} = \text{(nautical miles/pounds [no wind])} \times \left(\frac{\text{GS}}{\text{TAS}}\right)$$

Cruise data are based on standard day temperatures. For temperatures other than standard, fuel flow and true airspeed are the only items presented that are not valid. For this reason, Mach or KCAS (KIAS) should be used to set up cruise rather than fuel flow. Standard day temperatures are presented under the altitude for reference use only.

SAUNTER (MAXIMUM ENDURANCE)

Saunter data are based on 3500 pounds of fuel remaining (armament in) and a speed of 260 KCAS. Saunter data are also valid from 5000 pounds to 1500 pounds of fuel remaining with armament in or from 6500 pounds to 3000 pounds of fuel remaining with armament out. The minutes per 1000 pound values decrease 15% and 20% respectively for the clean and tank full fuel conditions.

DESCENT

Descent data are presented for 270 KCAS (not to exceed Mach 0.90) with speed brakes open and 82% rpm. Descent data are valid for all gross weights.

OPTIMUM RETURN TABLES

Optimum return cruise information is provided in tabular form for airplanes with and without external tanks. The tables indicate the minimum fuel required to fly a given air distance; or, conversely, the maximum distance that can be flown with a given amount of fuel remaining at a given altitude. The following is an example of the use of the tables.

KNOWN

Airplane: F-102A with external tanks insta[illegible]

Initial altitude: 20,000 feet.

Fuel remaining: 1210 pounds.

PROBLEM

Find the maximum air distance that can be flown with and without climb to optimum cruise altitude.

Enter the appropriate table (F Optimum Return with External Tanks) at an initial altitude of 20,000 feet. Read across to determine that the fuel required to fly 200 nautical miles, including a climb to optimum altitude of 42,000 feet, is 1210 pounds. Therefore, 200 nautical miles is the maximum distance that can be flown with 1210 pounds of fuel remaining. If the initial altitude of 20,000 feet is held, the maximum distance by interpolation is approximately 147 nautical miles. The desired fuel reserves must be added to the fuel required because the tables are based on zero fuel remaining over the landing base.

LANDING CHARTS

Landing data are presented for the speed brakes open configuration and for the speed brakes open plus drag chute configuration. Landing data are based on arma-

ment in. To correct for armament out, reduce the corresponding distance by 5% with drag chute deployed or 4% without drag chute deployed. Landing charts are the same as the charts shown in Part 7 of the Appendix.

KNOWN
Pressure altitude.
Ambient temperature.
Fuel remaining.
Braking conditions.
Reported wind.
Increase in recommended touchdown speed.

Enter chart with ambient temperature and pressure altitude. Proceed horizontally to the right until an intersection with the fuel remaining line is accomplished. Move vertically down to intersect the braking condition base line. Parallel the nearest guide line until the proper braking condition is reached. Move vertically down and intersect the zero headwind line. Parallel the nearest guide line until the reported headwind is reached. Proceed vertically down until an intersection is made with the zero increase in recommended touchdown speed line. Parallel the nearest guide line until the proper increase in recommended touchdown speed is reached.

Part 9 MISSION PLANNING

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Mission Planning A9-1

MISSION PLANNING

The charts presented in Parts 1 through 8 provide performance necessary to plan many different types of missions. The following sample problem illustrates how the performance charts may be integrated to form a complete mission. The sample problem utilizes the F-102A for a military thrust climb to 40,000 feet; a 40,000 foot cruise at optimum Mach, dropping external tanks when empty; a maximum range descent to a combat altitude of 5000 feet; a combat allowance of two minutes at maximum thrust at 5000 feet; retain armament; return climb to 40,000 feet at military thrust; a 40,000-foot cruise at 0.94 Mach; a maximum range descent at idle thrust to base; and land with 1500 pounds of fuel remaining. The mechanics of mission planning can be learned by working the sample problem. This sample mission could be flown by the pilot to check the performance of his airplane, or the pilot may wish to construct a new mission that would possibly reflect the local airplane usage. The pilot may wish to climb at maximum thrust and cruise supersonic; and descend to 25,000 feet or 20,000 feet; have a combat allowance of three minutes, and climb back to cruise altitude at military thrust; return cruise at military thrust, etc. This mission could be constructed and plotted, then flown by the pilot; just as any mission the pilot could conceive, construct and plot can be flown. By using the charts in this manual to construct a mission, the pilot should become better acquainted with the performance of the F-102A and TF-102A airplanes. The pilot who knows the performance of his airplane is better equipped to carry out the over-all mission of his command.

Note

The following sample problem is an exercise in the use of performance charts. It is not intended to reflect actual or proposed missions employing this airplane.

NEED FOR MISSION PLANNING

Because of the many types of missions capable of being flown, operation of the airplane may cover a wide altitude and speed range. This wide range of performance variables makes pre-mission planning a necessity. The most important results of mission planning are:

1. Combat radius available for a given combat time and reserve fuel.

2. Reserves available as a function of combat radius for a given combat time.

3. Combat time available as a function of combat radius for a given reserve fuel.

4. Familiarization with charts used to obtain performance data.

The first step in obtaining these results is establishing a mission. A mission is defined here as a series of performance phases combined in a particular order. A simple example of a mission would be:

1. Takeoff.
2. Climb.
3. Combat.
4. Descend.
5. Land.

SAMPLE PROBLEM

Objective: To plan and plot a simple mission using the airplane gross weight as a function of distance flown.

Configuration: Clean airplane plus two 230-gallon external wing tanks. 175 knot tires installed.

Fuel loading: 7053 pounds or 1085 gallons internal. 2795 pounds or 430 gallons external.

Fully loaded gross weight: 31,276 pounds.

Zero fuel weight (armament in): 21,097 pounds.

Droppable weight (External tanks, pylons and external trapped fuel: 331 pounds.

Supplemental Data

1. Local weather at takeoff: 10°C ambient temperature, 3000-foot pressure altitude, 10-knot wind from 015°.

2. Takeoff runway: 0°, 9000-foot length. Fair braking conditions exist.

3. Enroute weather outbound: standard day conditions with a 40-knot headwind at cruise altitude.

4. Enroute weather inbound: standard day conditions with a 10-knot tailwind at cruise altitude.

5. Local weather at landing: 5°C ambient temperature, 3000-foot pressure altitude, 40-knot wind from 050°.

6. Landing Runway: 0°, 9300-foot length, good braking conditions (dry hard runway, good tires and brakes and a practiced braking technique).

Mission is assumed to be as follows:

1. Engine start, warm-up, taxi, maximum thrust takeoff and acceleration to climb speed.

2. Military thrust climb to 40,000 feet.

3. Outbound cruise at 40,000 feet and optimum cruise Mach number. Drop external tanks when empty.

4. Descend to combat altitude of 5000 feet using a maximum range descent (speed brakes closed).

5. Combat allowance: two minutes in maximum thrust at 5000 feet. Retain armament.

6. Climb back to 40,000 feet with military thrust.

7. Inbound cruise at 40,000 feet and optimum cruise Mach number.

8. Descend at maximum range descent speed and idle thrust to base.

9. Landing with 1500 pounds of fuel remaining.

Takeoff

Takeoff speed is obtained from figure A2-7. Enter this chart with an ambient temperature and pressure attitude. Proceed to the right until the correct fuel remaining line is intersected. Then read down for takeoff speed.

KNOWN:

Ambient temperature is 10°C.

Pressure altitude is 3000 feet.

Fuel remaining is maximum fuel with external tanks.

FROM CHART:

Takeoff speed is 152 KCAS.

Next, obtain the headwind component of the wind from figure A2-2 by entering this chart with wind and wind angle and read the headwind component on the left side of the chart and the crosswind component on the bottom of the chart. If the crosswind component is below 25 knots, use the takeoff speeds obtained from figure A2-7. If the crosswind component is 25 knots or greater, then compare the speeds from the minimum touchdown or nosewheel liftoff speed table on figure A2-2 with the takeoff speeds given in figure A2-7. Use whichever speed is greater.

KNOWN:

The wind is 10 knots.

The wind is 15° off the nose.

FROM CHART:

The headwind component is 9.6 knots.

The crosswind component is small, therefore the takeoff speed above is valid.

Check to see if the tire limit speed shown in figure A2-7 is exceeded.

KNOWN:

Ambient temperature is 10°C.

Pressure altitude is 3000 feet.

Takeoff speed is 152 KCAS.

FROM CHART:

Tire limit speed would not be exceeded unless there was a 14.5 knot tail wind.

166.5 (from figure A2-7)
−152.0 (takeoff speed)
14.5 knots

Takeoff distance can now be obtained from figure A2-3. Enter this chart with ambient temperature and pressure altitude. Proceed to the right to intersect with the correct fuel remaining line. Then read vertically down to obtain zero wind ground roll distance. Then parallel the nearest guide line to read the ground roll for the reported headwind. Then proceed vertically down until an intersection is made with proper base line for total distance. Then proceed horizontally to the left and read total distance over a 50-foot obstacle. These distances would be incorrect if the takeoff speed was obtained from the crosswind chart.

KNOWN:

Ambient temperature is 10°C.

Pressure altitude is 3000 feet.

Maximum fuel aboard with external tanks.

Headwind component of the reported wind is 9.6 knots.

FROM CHART:

Takeoff ground roll is 2975 feet.

Total takeoff distance over a 50-foot obstacle is 4300 feet.

Maximum refusal speed and distance can be obtained from figure A2-11. Enter the chart with takeoff gross weight and proceed horizontally to the pressure altitude line. From this intersection move down to intersect the proper ambient temperature line, then proceed horizontally to intersect the correct available runway length. From this intersection proceed vertically down and read refusal speed (zero wind). This is of course the maximum speed which the airplane attains, then has to abort, and still is able to stop in the remaining available distance. To correct for wind, add headwind and subtract tailwind. The refusal distance is obtained by continuing horizontally from the available runway line intersection to the right until an intersection with the refusal speed (corrected for wind) is accomplished. The check marker is the prior even 1000-foot marker. The line check speed is obtained by reading vertically down from the check marker intersection.

KNOWN:

Takeoff gross weight is 31,276 pounds.

Pressure altitude is 3000 feet.

Ambient temperature is 10°C.

Available runway is 9000 feet.

Headwind component of the reported wind is 9.6 knots.

FROM CHART:

Refusal speed for zero wind is 169 KCAS.

Refusal speed corrected for wind is 178.6 KCAS.

Line check speed at the 4000-foot marker is 168 KCAS.

The landing, approach and prior to flare speeds for a landing immediately after takeoff, are found from the table on the landing distance chart (figure A7-3) as a function of fuel remaining.

KNOWN:

Fuel remaining is maximum fuel with external tanks.

FROM CHART:

Approach speed is 195 KCAS.

Touchdown speed is 155 KCAS.

Landing distances for a landing immediately after takeoff are obtained from figure A7-1 for landings with speed brakes open without drag chute and figure A7-3 for landings with speed brakes open and drag chute deployed. Enter these charts with ambient temperature and pressure altitude. Then proceed horizontally to the right and intersect the fuel remaining line. From this intersection move down and read the zero wind ground roll for medium braking conditions. Then parallel the nearest guideline to the headwind component of the reported wind and read the ground roll distance with headwind. Then move down and read the total landing distance over a 50-foot obstacle.

KNOWN:

Ambient temperature is 10°C.

Pressure altitude is 3000 feet.

Maximum fuel with external tanks.

Medium braking conditions exist.

Headwind component of the reported wind is 9.6 knots.

No increase in recommended touchdown speed.

FROM CHARTS:

Ground roll distance with speed brakes open and drag chute deployed is 6475 feet.

Total distance over a 50-foot obstacle is 7850 feet.

Ground roll distance with speed brakes open is 7520 feet.

Total distance over a 50-foot obstacle is 8910 feet.

Takeoff fuel allowance for this mission was assumed to be 1000 pounds. This allowance is assumed to account for engine start and warm-up, taxi, maximum thrust takeoff and acceleration to climb speed. No distance is assumed made good, but three minutes are assumed to have elapsed from engine start. A plot of gross weight versus distance will be constructed to show graphically how the different phases fit together to establish a mission radius. Figure A9-1 depicts the initial fuel consumption of 1000 pounds. See figure A9-13 for an over-all picture of the mission.

Climb

The data for a military thrust climb with two 230 gallon external fuel tanks is presented on figure A3-7. Enter this chart with gross weight at the beginning of the climb and pressure altitude. Parallel the nearest climb guide line until an intersection of the final altitude is accomplished. From this intersection proceed vertically down and read gross weight upon reaching altitude. Fuel used is obtained by subtracting the final weight from the initial weight. Next, proceed horizontally from the final altitude to the left to read final distance and to the right to read final time. Also proceed horizontally to the left from the initial altitude and read initial distance and to the right to read initial time. To obtain the actual time and distance in the climb, subtract the initial values from the final values. See figure A9-2 for climb plots.

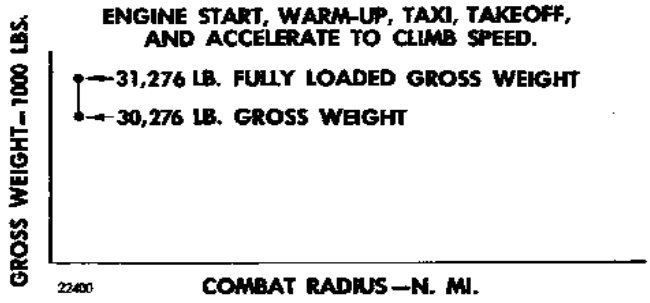


Figure A9-1

KNOWN:

Initial pressure altitude is 3000 feet.

Initial gross weight is 30,276 pounds.

Final altitude is 40,000 feet.

FROM CHART:

Final gross weight is 28,500 pounds.

Fuel used is 30,276 − 28,500 = 1776 pounds.

Initial distance is 8.0 nautical miles.

Final distance is 180.5 nautical miles.

Distance covered is 180.5 − 8.0 = 172.5 nautical miles.

Initial time is 1.7 minutes.

Final time is 23.6 minutes.

Time elapsed is 23.6 − 1.7 = 21.9 minutes.

Outbound Cruise

At this stage of mission planning the length of the cruise phase is not known. The recommended procedure is to construct a cruise line which will terminate at approximately 40% of fuel used. The outbound cruise must be divided into two phases. The first phase ends when the tanks are dropped, the second phase begins with the clean configuration and continues to approximately 40% of fuel used. A simple form (figure A9-3) can be used to retain the essential information.

Outbound Cruise With External Tanks

Data known: altitude 40,000 feet, configuration two external tanks, optimum cruise Mach number, headwind 40 knots. The data for a 40,000 foot cruise with external

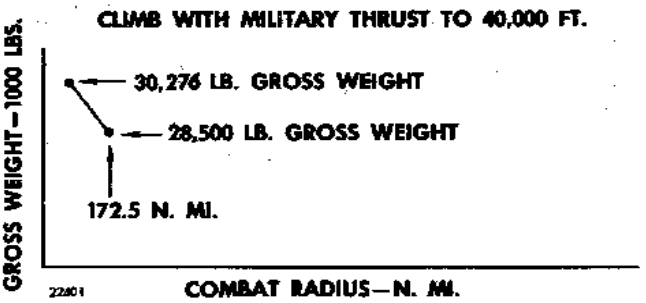


Figure A9-2

TABULAR CRUISE REFERENCE CHART

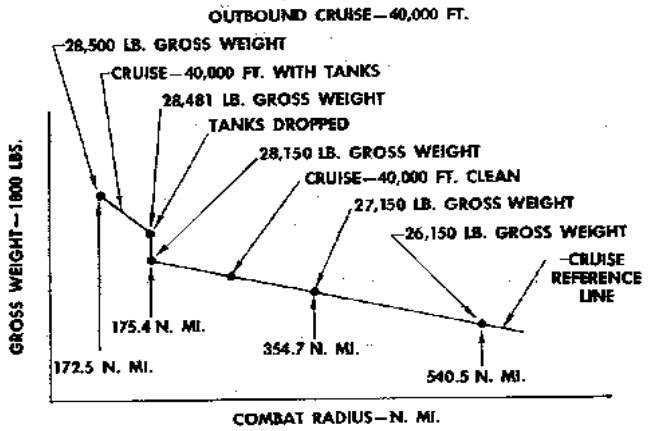
ALTITUDE: 40,000 FT.

CONFIGURATION: 2 EXTERNAL TANKS UNTIL EMPTY THEN CLEAN

WIND: 40 KT. HEADWIND • OPTIMUM CRUISE

ITEM	1ST PHASE				2ND PHASE			3RD PHASE		
	INITIAL	PHASE	FINAL		INITIAL	PHASE	FINAL	INITIAL	PHASE	FINAL
① GROSS WEIGHT (KNOWN)	28500	28491	28481		28150	27650	27150	27150	26650	26150
② FUEL USED (ASSUMED)	2776	19	2795		2795	1000	3795	3795	1000	4795
③ EPR (FROM CHART)		2.51				2.31			2.26	
④ CAS (FROM CHART)		275				280			278	
⑤ (N. MI./ 1000 LB) (FROM CHART)		164				194			201	
⑥ TAS (FROM CHART)		511				519			516	
⑦ FUEL FLOW (6 ÷ 5)		3120				2670			2562	
⑧ GS (6 ± WIND)		471				479			476	
⑨ TIME (2 ÷ 7)		.00609				3745			.3903	
⑩ DISTANCE (8 × 9)	172.5	2.9	175.4		175.4	179.3	354.7	354.7	186.8	540.5

TANKS DROPPED (331 LB)



22402

Figure A9-3

tanks is found on figure A4-47. The initial gross weight is 28,500 pounds. The initial fuel used is 2776 pounds. The external tank capacity is 2795 pounds, so the assumed total fuel used in this segment of cruise will be 19 pounds (2795 pounds – 2776 pounds). Subtracting the 19 pounds from the initial gross weight, leaves 28,481 pounds gross weight at the end of the first phase. This is an average gross weight of 28,491 pounds. Enter the chart and proceed along the recommended cruise Mach number (zero wind) line until an intersection with the average gross weight during cruise is reached. Interpolate and read EPR (2.51). Move horizontally to the left and read nautical miles per 1000 pounds (164). Return to the optimum cruise point and read vertically down to obtain Mach number (0.892), TAS (511 knots), and CAS (275 knots). Calculate GS (471 knots) by subtracting the headwind from the TAS. Calculate fuel flow (3120 pph) by dividing TAS by nautical miles per 1000 pounds. Then calculate time (.00609 hrs.) by dividing fuel used in the phase by fuel flow. Finally calculate the phase distance (2.9 n.m.) by multiplying GS by time. By adding this phase distance (2.9 n.m.) to the initial distance (172.5 n.m.), a total of 175.4 nautical miles is reached. At 175.4 nautical miles the external tanks are dropped. The weight of the external tanks and trapped fuel (331 lbs.) is subtracted from the final gross weight of the last segment (28,481 lbs.). See figure A9-3 for the plotting of this phase.

Outbound Cruise Clean Configuration

The initial gross weight for the remaining phase of cruise is then 28,150 pounds. The initial fuel used to this point is 2795 pounds. From this point (28,150 pounds gross weight and 175.4 nautical miles) on the mission planning chart, a 40,000-foot cruise line can be constructed. Assume 1000 pound increments of fuel used to cruise an unknown distance. The gross weight at this new point will then be 27,150 pounds, or an average gross weight of 27,650 pounds. Enter the chart (figure A4-45) along the recommended cruise Mach number (zero wind) line until an intersection with the average gross weight during cruise is reached. Proceed to extract the essential information from the chart as was done when calculating the 40,000-foot cruise, with tanks. A change in EPR (2.31) and fuel flow (2670 pph) will be noted. The essential information derived from the chart will be that the airplane has traveled 179.3 nautical miles during this phase with a total distance traveled of 354.7 nautical miles. A new point can now be plotted on the mission planning chart at 27,150 pounds gross weight and 354.7 nautical miles. To plot an accurate cruise reference line, one more point should be plotted. Assume another 1000 pounds of fuel used and calculate, as before, a new point on the mission planning chart. This point will be 26,150 pounds gross weight and at a distance of 540.5 nautical miles. The cruise reference line can now be completed; see figure A9-3.

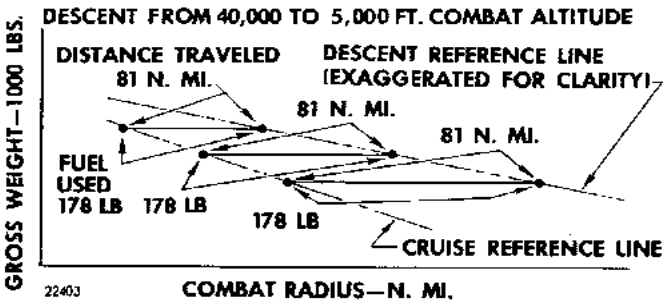


Figure A9-4

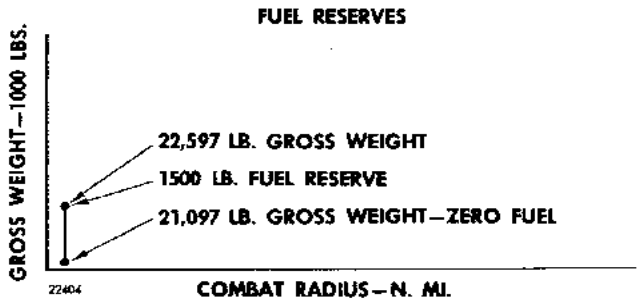


Figure A9-5

Descent

The descent data for a maximum range descent (with speed brakes closed) is shown on figure A6-6. Enter this chart with initial and final altitudes. Read initial and final times, distances and fuels used. Calculate time, distance and fuel used by subtracting the final values from the initial values. See figure A9-4.

KNOWN:

Maximum range descent (speed brakes closed).
Initial altitude is 40,000 feet.
Final altitude is 5000 feet.

FROM CHART:

Initial fuel is 212 pounds.
Final fuel is 34 pounds.
Fuel used is 212 34 = 178 pounds.
Initial distance is 89 nautical miles.
Final distance is 8 nautical miles.
Distance covered is 89 − 8 − 81 nautical miles.
Initial time is 17.8 minutes.
Final time is 2.7 minutes.
Time elapsed is 17.8 2.7 = 15.1 minutes.

These values can be used at any gross weight because gross weight has a negligible effect on descent performance. It would now be convenient for mission planning purposes to compute the descent to base, return cruise and climb phases of the mission.

Reserves

Prior to adding the descent to base phase to the mission plot, the reserves are added. In some instances these reserves may be calculated from (1) a performance chart (figure A5-1); (2) a certain percentage of initial fuel; (3) a given amount of time at a certain thrust setting and altitude, or, (4) a certain amount of fuel remaining as required from an operational standpoint. In this example the latter case is chosen, namely a flat 1500 pounds of fuel remaining. See figure A9-5.

Descend to Base

The data for a maximum range descent at idle thrust is found on figure A6-6. Enter this chart with initial and final pressure altitudes. Read initial and final times, distances and fuel used. Calculate time, distance, and fuel used by subtracting the final values from the initial values. See figure A9-6.

KNOWN:

Initial altitude is 40,000 feet.
Final altitude is 3000 feet.

FROM CHART:

Initial fuel is 212 pounds.
Final fuel is 22 pounds.
Fuel used is 212 − 22 = 190 pounds.
Initial distance is 89 nautical miles.
Final distance is 6 nautical miles.
Distance covered is 89 − 6 = 83 nautical miles.
Initial time is 17.8 minutes.
Final time is 1.8 minutes.
Time elapsed is 17.8 − 1.8 − 16 minutes.

RETURN CRUISE

As was the case with outbound cruise, the return cruise distance is unknown at this stage of mission planning. The recommended procedure is to construct a cruise phase to terminate at approximately 40% of the fuel remaining. A simple form (figure A9-7) can be used to retain essential information. Data known: altitude 40,000 feet, configuration clean, optimum cruise Mach number, tailwind 10 knots. Data for a 40,000-foot cruise in the clean configuration is found on figure A4-45. The gross weight prior to descent is 22,787 pounds at a distance of 83 nautical miles. Assume 1000 pounds of fuel is needed to cruise an unknown distance. The gross weight at this new point on the mission planning chart

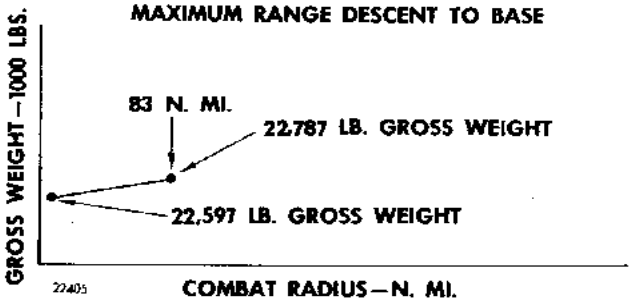


Figure A9-6

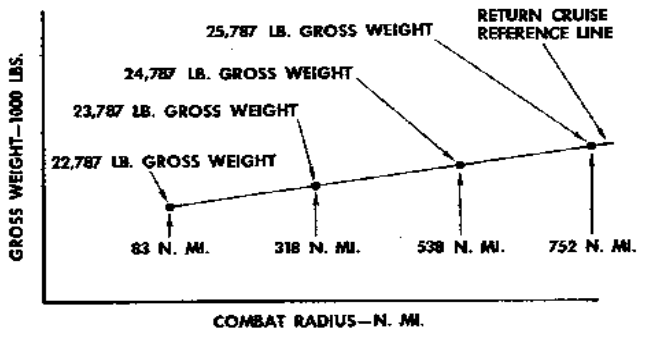
TABULAR CRUISE REFERENCE CHART

CONFIGURATION: CLEAN

ALTITUDE: 40,000 FT. • WIND: 10 KT. TAILWIND • OPTIMUM CRUISE

ITEM	1ST PHASE			2ND PHASE			3RD PHASE		
	INITIAL	PHASE	FINAL	INITIAL	PHASE	FINAL	INITIAL	PHASE	FINAL
① GROSS WEIGHT (KNOWN)	22787	23287	23787	23787	24287	24787	24787	25287	25787
② FUEL USED (ASSUMED)	1690	1000	2690	2690	1000	3690	3690	1000	4690
③ EPR (FROM CHART)		2.136			2.167			2.203	
④ CAS (FROM CHART)		273			274			275	
⑤ (N. MI./ 1000 LB) (FROM CHART)		221			215.8			209.5	
⑥ TAS (FROM CHART)		507			509			512	
⑦ FUEL FLOW (6 - 5)		2295			2360			2245	
⑧ GS (6 ± WIND)		517			519			522	
⑨ TIME (2 ÷ 7)		.436			.424			.409	
⑩ DISTANCE (8 × 9)	.93	225	318	318	220	538	538	214	752

RETURN CRUISE—40,000 FT. CLEAN



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Figure A9-7

will then be 23,787 pounds, or an average gross weight of 23,287 pounds. Enter the chart, figure A4-45 on the "recommended cruise Mach number (zero wind)" line and proceed to extract the essential information from the chart as was done when calculating the outbound cruise. Note the EPR (2.136) and the fuel flow (2295). The essential information derived from the chart will be that the airplane has traveled a distance of 225 nautical miles during this phase and is now 318 nautical miles out. A new point can now be plotted on the mission planning chart at 23,787 pounds gross weight and 318 nautical miles. To plot an accurate return cruise reference line, plot two more points. Assume another 1000 pounds of fuel is needed and calculate, as before, a new point on the mission planning chart. This point will be at 24,787 pounds gross weight and 538 nautical miles out. Assume another 1000 pounds of fuel is needed and plot another point. This point will be at 25,787 pounds gross weight and 752 nautical miles out. The return cruise reference line can now be completed. See figure A9-7.

Return Climb

At this stage of mission planning the initial or final conditions of the return climb are unknown. It is therefore recommended that a parameter of final gross weight upon reaching cruise altitude be assumed. The data for this climb can be found on figure A3-5. Enter this chart with final altitude and gross weight. At this intersection proceed horizontally to the right and read final time and horizontally to the left and read final distance. Parallel the nearest climb guide line to initial altitude. At this intersection proceed horizontally to the right and read initial time and horizontally to the left and read initial distance. To obtain time and distance, subtract the initial conditions from the final conditions. To obtain the fuel used, subtract the final gross weight from the initial gross weight. See figure A9-8.

Combat

At this stage of mission planning it is possible to calculate several alternate missions. One could calculate, (A) the maximum radius for any given combat time (primary example shown); (B) the available combat time for any combat radius (first alternate); or (C) the fuel available for additional reserves or any other operational use at a given radius and combat time (second alternate). The primary and first alternate missions will be discussed first since the return climb and return cruise guide lines have already been constructed. Fuel allowance for combat is obtained from figure A5-9. For primary mission "A" enter figure A5-9 with combat time and latitude. Interpolate between constant fuel lines for fuel used.

KNOWN:

Combat altitude is 5000 feet.

Combat thrust setting is maximum.

Combat time is two minutes.

ASSUMED

ITEM	UNITS			
1. FINAL GROSS WEIGHT	LB	25,000	24,000	23,000

KNOWN

ITEM	UNITS			
2. FINAL ALTITUDE	FT	40,000	40,000	40,000
3. INITIAL ALTITUDE	FT	5,000	5,000	5,000

FROM CHART

ITEM	UNITS			
4. INITIAL GROSS WEIGHT	LB	25,980	24,900	23,870
5. FUEL USED (4—1)	LB	980	900	870
6. FINAL DISTANCE	N. MI.	92.7	87.0	81.5
7. INITIAL DISTANCE	N. MI.	8.1	7.7	7.1
8. DISTANCE COVERED (6—7)	N. MI.	84.6	79.3	74.4
9. FINAL TIME	MIN.	13	12.3	11.6
10. INITIAL TIME	MIN.	1.7	1.6	1.5
11. TIME ELAPSED (9—10)	MIN.	11.3	10.7	10.1

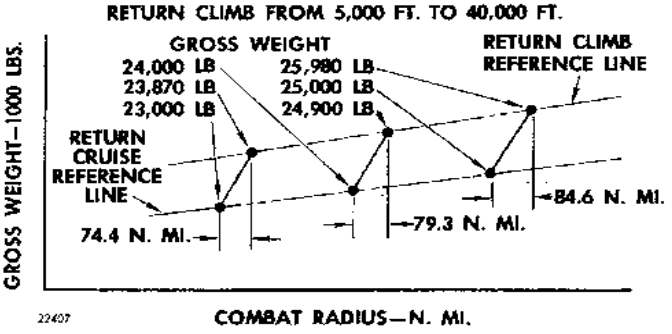


Figure A9-8

FROM CHART:

Combat fuel is 1333 pounds.

The resulting maximum combat radius is 515 nautical miles. See figure A9-9.

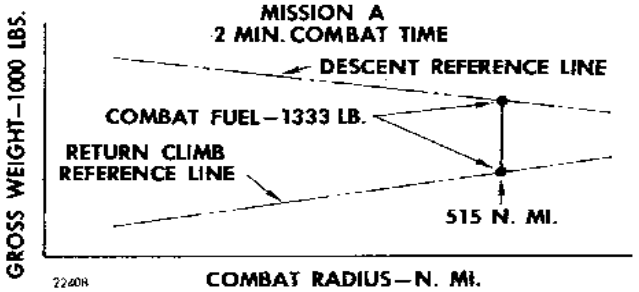


Figure A9-9

Consider alternate mission "B" (available combat time for a given radius) for a radius of 300 nautical miles. See figure A9-10.

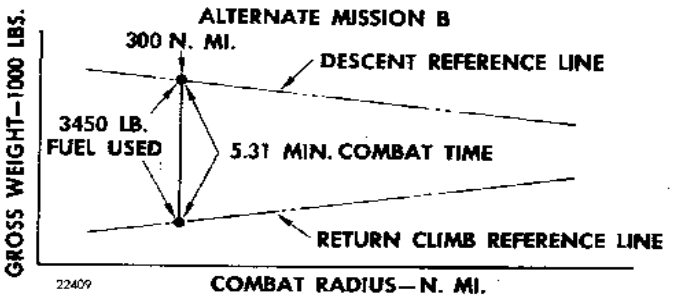


Figure A9-10

KNOWN:

Combat radius is 300 nautical miles.
Combat altitude is 5000 feet.
Combat fuel is 3450 pounds (from Mission Planning Chart, figure A9-12).

FROM CHART:

Combat fuel flow is 650 lb./min.
Combat time available will have to be calculated from the fuel flow since chart does not as high as the fuel available for combat.

$$\text{Combat time (minutes)} = \frac{\text{Combat fuel}}{\text{Fuel Flow}} = \frac{3450}{650} = 5.31 \text{ minutes}$$

In order to complete the alternate mission "C," it will be necessary to calculate the gross weight at the beginning of the return climb, an additional return climb from combat, a return cruise, and a descent to base. The gross weight at the beginning of the return climb is as follows:

KNOWN:

Combat radius is 300 nautical miles.
Combat time is 2 minutes.
Combat fuel is 1333 pounds (same as primary mission "A").

CALCULATED:

Gross weight at beginning of combat is 27,710 pounds. (See figure A9-12.)
Weight at end of combat or beginning of return climb is 27,710 1333 = 26,377 pounds.

The climb is obtained by using the data from figure A3-7.

KNOWN:

Initial pressure altitude is 5000 feet.
Initial gross weight is 26,377 pounds.
Final altitude is 40,000 feet.

FROM CHART:

Final gross weight is 25,360 pounds.
Fuel used is 1017 pounds.

Initial distance is 8.1 nautical miles.
Final distance is 94 nautical miles.
Distance covered is 94 − 8.1 − 85.9 nautical miles.
Initial time is 1.8 minutes.
Final time is 13.3 minutes.
Time elapsed is 13.3 − 1.8 = 11.5 minutes.

The return cruise portion for alternate mission "C" is obtained by a slightly different method than that used for the cruise out portion or the return cruise of the primary mission "A" and first alternate mission "B." A cruise reference line is determined by using the weight at end of climb, plus 500 pounds and the weight at end of climb minus 500 pounds. A table as shown in figure A9-11 can be used to retain essential information. The details of procedures for obtaining this information are omitted since they have been discussed in the return cruise portion of this section. The results of these calculations are shown in figure A9-12. It should be noted that the return cruise reference line on figure A9-8 and figure A9-12 is drawn all the way to zero combat radius in order to obtain a reference gross weight for the descent to base portion of this particular mission.

The distance and fuel used to descend for the alternate mission "C" are the same as have previously been calculated for missions "A" and "B."

KNOWN:

Fuel used is 190 pounds.
Distance covered is 83 nautical miles.
Time elapsed is 16 minutes.
Gross weight on return cruise reference line at 83 nautical miles − 24,750 pounds. (From figure A9-12.)

CALCULATED:

Gross weight at end of descent = gross weight at 83 nautical miles from base minus fuel used to descend = 24,750 − 83 = 24,667 pounds.

The additional reserves can now be calculated.

KNOWN:

Gross weight at end of descent = 24,667 pounds.
Gross weight at end of descent for normal 1500 pound reserves = 22,597 pounds.

CALCULATED:

Additional reserves = 24,667 − 22,597 = 2070 pounds.

See figure A9-13 for the completed mission planning chart.

Total Mission And Cruise Phase Time

If it is necessary to know total mission or cruise phase time, one could calculate them after the cruise distances are known. The example which follows is for primary mission "A," however the procedure can be used for other missions as well.

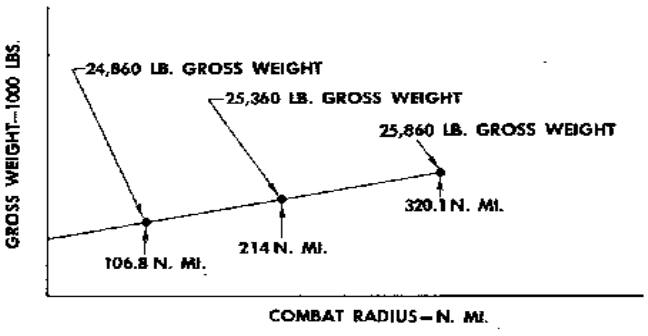
TABULAR CRUISE REFERENCE CHART

(ALTERNATE MISSION "C")

CONFIGURATION: CLEAN

WIND: 10 KT. TAILWIND • OPTIMUM CRUISE

ITEM	1ST PHASE				3RD PHASE		
	INITIAL	PHASE	FINAL		INITIAL	PHASE	FINAL
① GROSS WEIGHT (KNOWN)	25360	25610	25860		25360	25110	24860
② FUEL USED (ASSUMED)		500				-500	
③ EPR (FROM CHART)		2.215				2.195	
④ CAS (FROM CHART)		276				275	
⑤ N. MI/ 1000 LB (FROM CHART)		208				210.8	
⑥ TAS (FROM CHART)		512				511	
⑦ FUEL FLOW (6×5)		2460				2425	
⑧ GS (6 ± WIND)		522				521	
⑨ TIME (2÷7)		2035				206	
⑩ DISTANCE (8×9)	214	+106.1	320.1		214	-107.2	106.8



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Figure A9-11

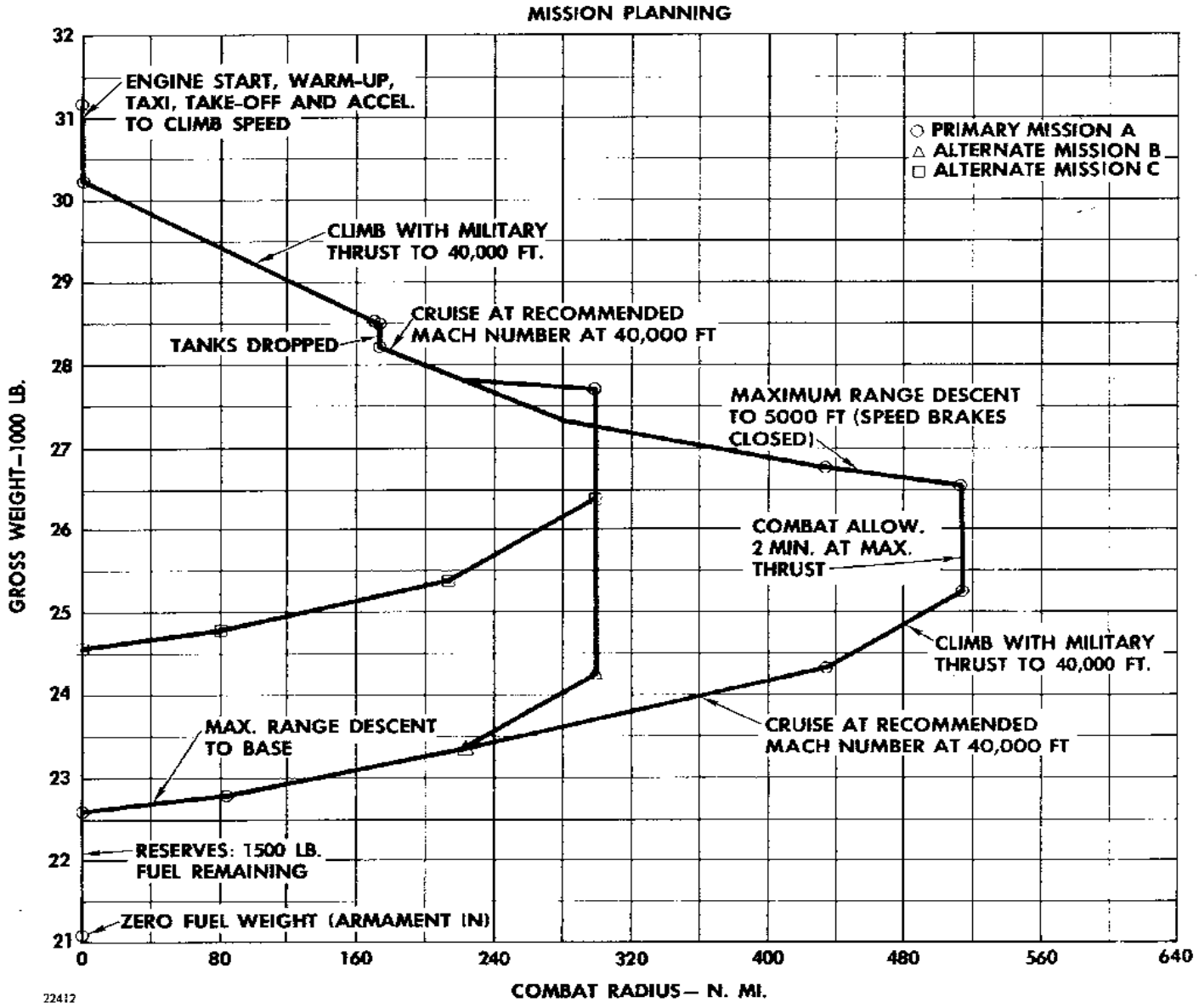


Figure A9-12

Cruise phase time can be obtained as follows:

Obtain the initial and final weights and distance traveled for a particular cruise phase. Using the average gross weight enter Chart A4-45 on the recommended cruise Mach number (zero wind) line at the average gross weight and read TAS. Determine the GS by using the appropriate wind speed, then use the following relation to compute time:

$$\text{Time (minutes)} = \frac{\text{Cruise Distance (n. mi.)}}{\text{GS (knots)}} \times 60$$

Total mission time can be calculated by summing the individual phases of the mission in this example.

Phase	Phase Time (Minutes)	Total Time (Minutes)
Engine start, warm-up, taxi, takeoff, and accelerate to climb speed.	3.0	3.0
Climb to 40,000 feet	21.9	24.9
Cruise outbound with tanks	.4	25.3
Cruise outbound clean	32.4	57.7
Descend to combat altitude	15.1	72.8
Combat at 5000 feet	2.0	74.8
Return climb to 40,000 feet	10.8	85.6
Return cruise	40.7	126.3
Descend to base	16.0	142.3

Landing

Landing speed and distances at the termination of the mission are obtained from figures A7-1 and A7-3. Figure A2-2 is also used to obtain the headwind component of the wind and to check whether the recommended touchdown speed needs to be increased due to crosswind. Consider the crosswind chart first. Enter this chart with wind and wind angle. At this intersection move horizontally to the left and read the headwind component of the wind. Also from the intersection of the wind and wind angle, move down to read the crosswind component. With the crosswind component, enter the chart below to find the minimum touchdown speed or nosewheel liftoff speed.

KNOWN:

Wind is 40 knots.

Wind angle from the nose is 50°.

FROM CHART:

Headwind component is 25.7 knots.

Crosswind component is 30.5 knots.

Minimum touchdown speed is 167 KCAS.

Descend to base & total mission time	16	142.3

The recommended landing approach and prior to flare speeds are found from the table on the landing distance charts as a function of fuel remaining.

KNOWN:

1500 pounds of fuel remaining.

FROM CHART:

Approach speed is 170 KCAS.

Touchdown speed is 130 KCAS.

These values will be valid only if the recommended touchdown speed is not increased due to a high crosswind. Since the touchdown speed has to be increased for the example shown the approach and prior to flare speeds shown are not valid. Enter the landing distance charts with ambient temperature and pressure altitude. Then proceed horizontally to the right until an intersection with the proper fuel remaining curve is accomplished. Next move vertically down to the braking condition base line (medium) and parallel the nearest guide line to the reported braking condition. Then continue down to the zero wind line and then parallel the nearest guide line until the headwind component of the reported wind is intersected. Now move vertically down till the zero increase in recommended touchdown speed line is intersected. Next, parallel the nearest guide line until an intersection with the proper increase in recommended touchdown speed is accomplished.

KNOWN:

Ambient temperature is 5°C.

Pressure altitude is 3000 feet.

Fuel remaining is 1500 pounds.

Braking conditions are "good."

Headwind component is 25.7 knots.

Increase in recommended touchdown speed is 165 − 135 = 30 knots.

FROM CHARTS:

Ground roll with speed brakes open and drag chute deployed is 2900 feet.

Ground roll with speed brakes open is 4175 feet.

Total distance with speed brakes open and drag chute deployed is 4340 feet.

Total distance with speed brakes open is 5590 feet.



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