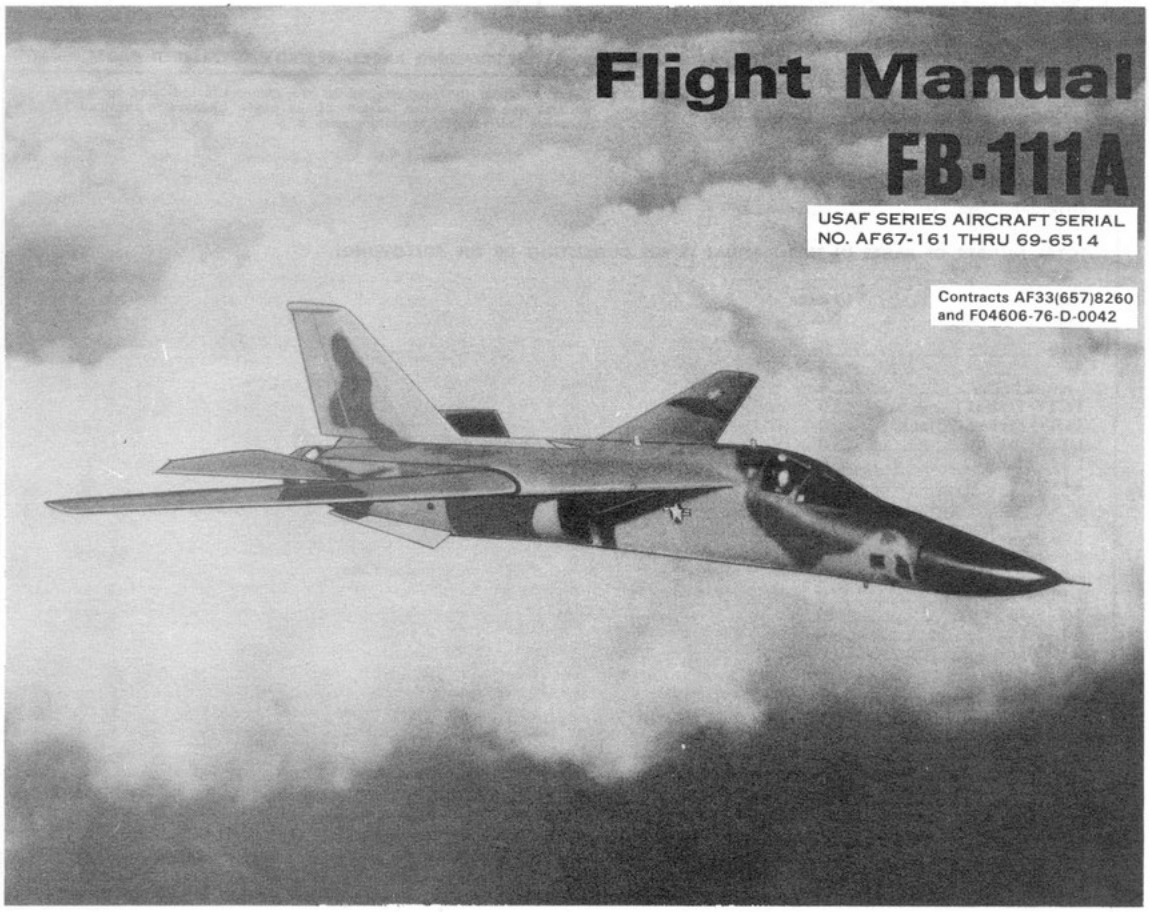
Flight Manual FB-111A

USAF SERIES AIRCRAFT SERIAL NO. AF67-161 THRU 69-6514

Contracts AF33(657)8260 and F04606-76-D-0042



THIS MANUAL IS INCOMPLETE WITHOUT T.O. 1F-111(B)A-1-1, T.O. 1F-111(B)A-1-2 AND T.O. 1F-111(B)A-1-3

This manual supersedes T.O. 1F-111(B)A-1 dated 26 July 1974 and Safety Supplements T.O. 1F-111(B)A-1SS-29, -33, -36 and -38; and Operational Supplements T.O. 1F-111(B)A-1S-32, -35, -40 and -44.

See Numerical Index and Requirement Table for Current Status of Safety and Operational Supplements

Commanders are responsible for bringing this publication to the attention of all Air Force personnel cleared for operation of subject aircraft.

Published under the authority of the Secretary of the Air Force

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25 FEBRUARY 1977

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1-1 - 1-193	0
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2-1 - 2-57	0
2-58 Blank	0
3-1 - 3-67	0
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4-1 - 4-18A	0
4-18B Blank	0
4-19 - 4-52	0
5-1 - 5-38	0
6-1 - 6-35	0
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CURRENT
FLIGHT CREW CHECKLIST
1F-111(B)A-1CL-1
25 FEB 1977

* Zero in this column indicates an original page.

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SCOPE.

This manual contains the necessary information for safe and efficient operation of the aircraft. These instructions provide you with a general knowledge of the aircraft, its characteristics, and specific normal and emergency operating procedures. Your experience is recognized, and therefore, basic flight principles are avoided. Instructions in this manual are for a crew inexperienced in the operation of this aircraft. This manual provides the best possible operating instructions under most circumstances. Multiple emergencies, adverse weather, terrain, etc., may require modification of the procedures.

PERMISSIBLE OPERATIONS.

The Flight Manual takes a "positive approach" and normally states only what you can do. Unusual operations or configurations are prohibited unless specifically covered herein. Clearance must be obtained before any questionable operation not specifically permitted in this manual is attempted.

HOW TO BE ASSURED OF HAVING LATEST DATA.

Refer to the Flight Manual and Supplement Status page (flyleaf) published with each Safety and Operational Supplement for a comprehensive listing of the current Flight Manuals, Flight Crew Checklists, Safety Supplements and Operational Supplements. T.O. 0-1-1-4 should be checked periodically to make sure you have current Flight Manuals, Checklists and Supplements.

ARRANGEMENT.

The manual is divided into seven fairly independent sections to simplify reading it straight through or using it as a reference manual.

Note

Performance data normally included in Appendix I is contained in T.O. 1F-111(B)A-1-1.

SAFETY SUPPLEMENTS.

Information involving safety will be promptly forwarded to you by Safety Supplement. Supplements covering loss of life will get to you within 48 hours by TWX, and those covering serious damage to equipment within 10 days by mail. The title page of the Flight Manual and the title block of each Safety Supplement should be checked to determine the effect they may have on existing Supplements. You must remain constantly aware of the status of all Supplements. Current Supplements must be complied with, but there is no point in restricting your operation by complying with a replaced or rescinded Supplement.

OPERATIONAL SUPPLEMENTS.

Information involving changes to operating procedures will be forwarded to you by Operational Supplements. The procedure for handling Operational Supplements is the same as for Safety Supplements.

CHECKLISTS.

The Flight Manual contains only amplified procedures. Abbreviated Checklists are issued as separate documents; see the back of the title page for the date of your latest Checklist. Line items in the Flight Manual and Checklists are identical with respect to arrangement and step number. Whenever a Safety Supplement affects the Checklist, write in the applicable change on the affected Checklist page. As soon as possible, a

new Checklist page incorporating the Supplement will be issued. This will keep handwritten entries of Safety Supplement information in your Checklist to a minimum.

HOW TO GET PERSONAL COPIES.

Each flight crew member is entitled to personal copies of the Flight Manual, Safety Supplements, Operational 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 appropriate Numerical Index and Requirement Table (NIRT). T.O. 00-5-1 and T.O. 00-5-2 give detailed information for properly ordering these publications. Make sure a system is established at your base to deliver these publications to the flight crews immediately upon receipt.

FLIGHT MANUAL BINDERS.

Looseleaf 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 Group 75, Office Supplies, Part 1). Check with your supply personnel for assistance in procuring these items.

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 it exists. 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 Form 847 through your Command Headquarters to:

Sacramento Air Logistics Center
McClellan AFB, California 95652
Attn: MMSRBD

SYSTEM LIMITS AND TOLERANCES.

In some cases, the limits and tolerances presented in the Flight Manual are not precisely identical to those presented in the system maintenance manuals. The numerical values in the Flight Manual are to be used as operating guides by aircrews.

AIRCRAFT DESIGNATION CODES.

Major differences between aircraft covered in this manual are designated by number symbols which appear in the text or on illustrations. Symbol designations for individual aircraft and groups of aircraft are as follows:

❸	67-161	㊵	68-268
❹	67-162	㊶	68-269
❺	67-163	㊷	68-270
❻	67-7192	㊸	68-271
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❾	67-7195	㊻	68-274
❿	67-7196	㊼	68-275
⓫	68-239	㊽	68-276
⓬	68-240	㊾	68-277
⓭	68-241	㊿	68-278
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⓱	68-245	(54)	68-282
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⓳	68-247	(56)	68-284
⓴	68-248	(57)	68-285
㉑	68-249	(58)	68-286
㉒	68-250	(59)	68-287
㉓	68-251	(60)	68-288
㉔	68-252	(61)	68-289
㉕	68-253	(62)	68-290
㉖	68-254	(63)	68-291
㉗	68-255	(64)	68-292
㉘	68-256	(65)	69-6503
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㉜	68-260	(69)	69-6507
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◗ "through" or "and on".

GLOSSARY

A/B	Afterburner
ADI	Attitude Director Indicator
AFRS	Auxiliary Flight Reference System
AILA	Airborne Instrument Low Approach
AIMS	Air Traffic Control Radar Beacon System/IFF(SIF)/MK12IFF/ System
AIR FF	Air Freefall
AIR RET	Air Retard
ALT CAL	Altitude Calibration
ALT HLD	Altitude Hold
ALTM	Altimeter
AMAC	Aircraft Monitor and Control
AME	Amplitude Modulation Equivalent
AMI	Airspeed-Mach Indicator
ANT	Antenna
ASTM	American Society for Testing Materials
ATF	Automatic Terrain Following
ATT	Attitude
AVVI	Altitude-Vertical Velocity Indicator
AYC	Adverse Yaw Compensation
BCN	Beacon
BDHI	Bearing Distance Heading Indicator
BNDTI	Bomb Nav Distance Time Indicator
CADC	Central Air Data Computer
CCM	Counter-Counter Measures
CCU	Computer Control Unit
C/D	Climb/Dive
CDS	Control and Display Set
CES	Code Enabling Switch
CIR	Circular
CKT	Circuit
CMD	Command
CMDS	Countermeasures Dispenser Set
COM	Common
COMP	Compass
CRS LINE	Course Line
CRS SEL NAV	Course Select Navigation
CSD	Constant-Speed Drive
CS	Converter Set
CSS	Coded Switch Set
CSSC	Coded Switch Set Controller
CTR	Constant Track Release
DCC	Digital Computer Complex
DEST	Destination
DG	Directional Gyro
DISP	Dispenser
DIV	Dive
EBL	Emergency Boom Latching
EPI	Elevator Position Indicator
EPR	Engine Pressure Ratio
EVF	Enter Visual Fix
FDC	Flight Director Computer
FLSC	Flexible Linear Shaped Charges
FSK	Frequency Shift Keying
FTC	Fast Time Constant
FXPT	Fixpoint
G/C	Gyrocompass
GM	Ground Mapping
GNC	General Navigation Computer
HOM	Homing
HSI	Horizontal Situation Indicator
ICU	Indicator Control Unit
IFIS	Integrated Flight Instrument System
INS	Inertial Navigation Set
IRT	Infrared Track
IRS	Infrared Search
IRRS	Infrared Receiver System
IRU	Inertial Reference Unit
ISC	Instrument System Coupler
LNCH	Launch
LSB	Lower Side Band
MFC	Manual Frequency Control
MR	Milliradian
MRT	Modulator-Receiver Transmitter
MSMA	Maximum Safe Mach Assembly
NDU	Navigation Display Unit
NTIK	Non-Tactical Instrumentation Kit
NWS/AR	Nose Wheel Steering/Air Refueling
ODS	Optical Display Sight
OMO	Omni Warning Open
OMS	Off, Monitor, Safe
OMT	Omni Warning Threat
OTL	Operational Test Launch
PCSS	Pitch Control Stick Steering
PP, PRES/POS	Present Position
PPI	Plan Position Indicator
RAD	Radiation
RCSS	Roll Control Stick Steering
RHAWS	Radar Homing and Warning System
SAF	Safe, Arm and Fuze
SAL	SRAM Alternate Launch
SIT	Situation Display

SLC	Side Lobe Cancellation
SMDC	Shielded Mild Detonating Cord
SMS	Stores Management System
SPC	Special Purpose Chaff
SPU	Systems Program Unit
SRAM	Short Range Attack Missile
STAB AUG	Stability Augmentation
STC	Sensitivity Time Control
TAL	Transfer Alignment Procedure
TBC	Trackbreaker Chaff
TDU	Threat Display Unit
TED	Trailing Edge Down
TEU	Trailing Edge Up
TFR	Terrain Following Radar
TG (Tg)	Time-to-Go
TIT	Turbine Inlet Temperature
TKR RV	Tanker Rendezvous
TTI	Total Temperature Indicator
USB	Upper Sideband
WDC	Weapons Delivery Computer

DEFINITIONS.

ASYMMETRICAL LOADING

Weapon/tank load on any pylon is not identical to the corresponding pylon on opposite wing

SYMMETRIC MANEUVER

A pull-out or push-over maneuver or steady bank

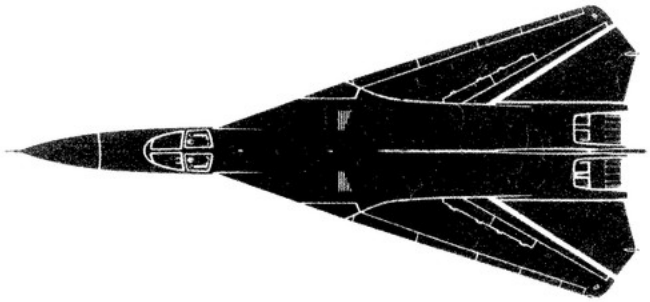
ASYMMETRIC MANEUVER

A rolling pull-out

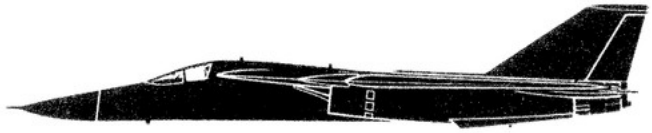
MINIMUM ENROUTE ALTITUDE (MEA)

The acronym MEA, as used in this flight manual, refers to the applicable Emergency Minimum Safe Altitude (EMSA), Safe Clearance Altitude (SCA), or IFR altitude for a given low level leg segment.

FB-111A









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AIRCRAFT RETROFIT TECHNICAL ORDER INFORMATION

This list includes the applicable TCTO numbers that have been issued up to the date of this publication. Those issued after that date will appear in the next Change/Revision. This is not a complete TCTO listing. Refer to the Basic Index (T.O. 0-1-1-4) for the complete listing of TCTO's which are applicable to this aircraft. Throughout this manual, the terms "Prior to T.O. —" and "After T.O. —" refer to aircraft prior to modificaton and aircraft after modification, respectively.

T.O. No.	*Short Title*	*System/Equipment Affected*
1F-111-946	Weapons Bay Overheat Detection	Wheel Well Overheat Detection System, Section I
1F-111-948	Modify Weapons Bay Door Ground Safety Lockpin Access Door	Armament System, Section I
1F-111-962	Repair of Wing Substructure	Load Factor Limits, Section V
1F-111-1066	Modify Bay Door Hinge Fairing Support	Airspeed Limitations, Section V
1F-111-1151	Add Bleed Duct Failure Indicating System	Engines, Air Conditioning and Pressurization Systems, Section I; Caution Lamp Analysis, Section III
1F-111-1162	Finger Tank Isolation	Fuel System, Section I
1F-111-1180	Flight Program Change (NCU Tape) (PIN-15)	Bombing Navigation System, Section I; Caution Lamp Analysis, Section III
1F-111(B)A-593	Modify Yaw and Roll Flight Control Computers, Feel and Trim Assy, and Control Stick	Autopilot System, Section I
1F-111(B)A-659	Modify Stores Control Panel to Prevent Inadvertent Release of Stores/Weapons	Electrical Power Supply System, Section I

SECTION I

DESCRIPTION & OPERATION

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THE AIRCRAFT.

The FB-111A is a two-place (side-by-side) long-range fighter bomber built by General Dynamics, Convair Aerospace Division, Fort Worth Operation. The aircraft is designed for all weather supersonic operation at both low and high altitudes, utilizing conventional or nuclear weapons. An automatic low altitude terrain following system enhances penetration capability. Power is provided by two TF-30 axial-flow, dual-compressor turbofan engines equipped with afterburners. The wings, equipped with leading edge slats and trailing edge flaps, may be varied in sweep, area, camber, and aspect ratio by the selection of any wing sweep angle between 16 and 72.5 degrees. This feature provides the aircraft with a highly versatile operating envelope. The empennage consists of a fixed vertical stabilizer with rudder for directional control and a horizontal stabilizer that is moved symmetrically for pitch control and asymmetrically for roll control. Stability augmentation incorporates triple redundant features which enhance system reliability. The tricycle-type forward retracting landing gear is hydraulically operated. The main landing gear consists of a single common trunnion upon which two wheels are singly mounted and contains but one extending/retracting/locking system, which ensures symmetrical main gear operation. Also, ground loads imposed upon the gear tend to extend the drag strut to the locked position. Stores are carried in a fuselage-enclosed weapons bay and externally on both pivoting and fixed wing-mounted pylons. The fuel system incorporates both inflight and single point ground refueling capabilities. See figure 1-1 aircraft general arrangement.

AIRCRAFT DIMENSIONS.

Length (overall including pitot-static boom) — 75 feet, 6.5 inches

Wing span (wings swept) — 34 feet

General Arrangement Diagram

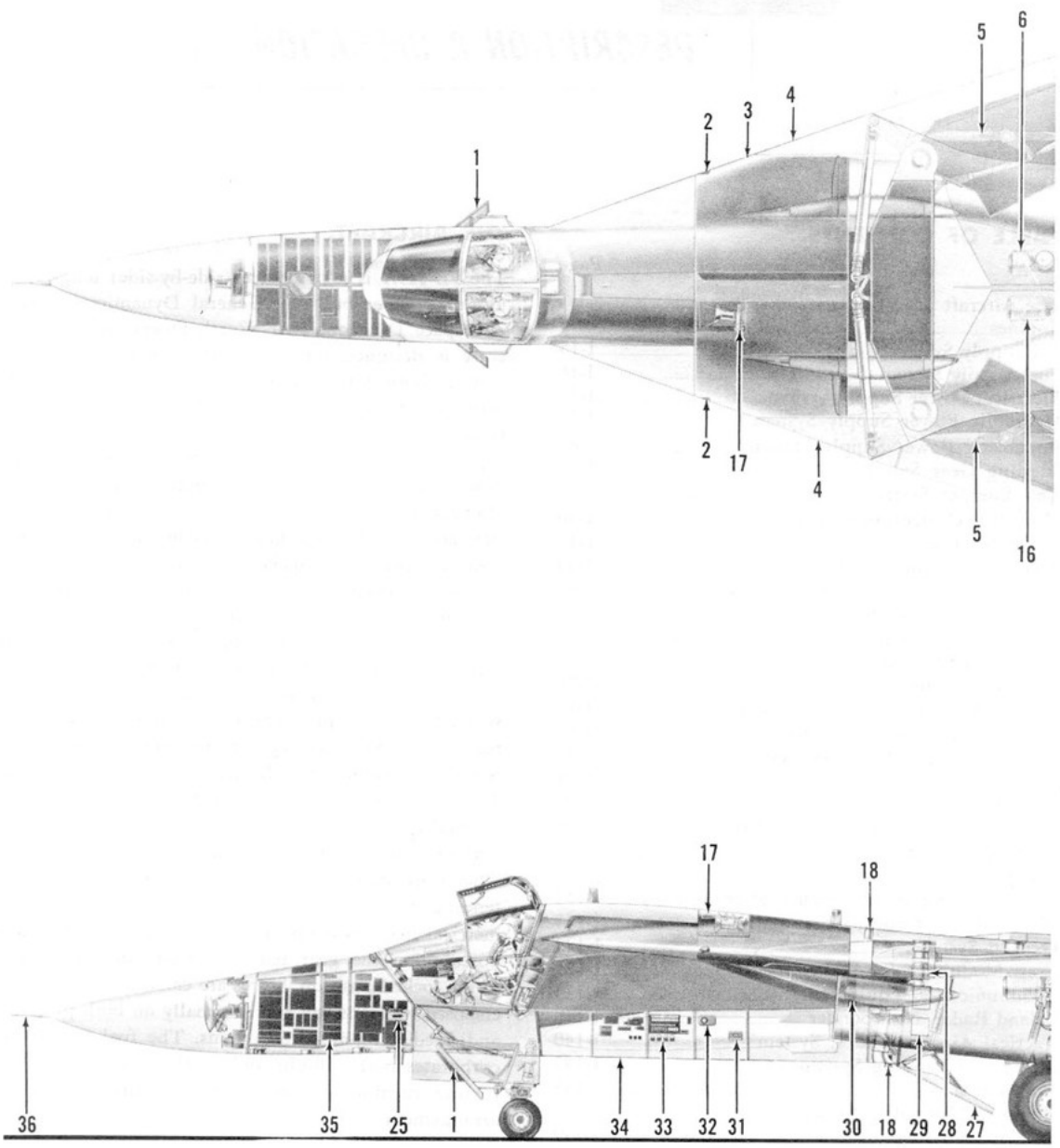
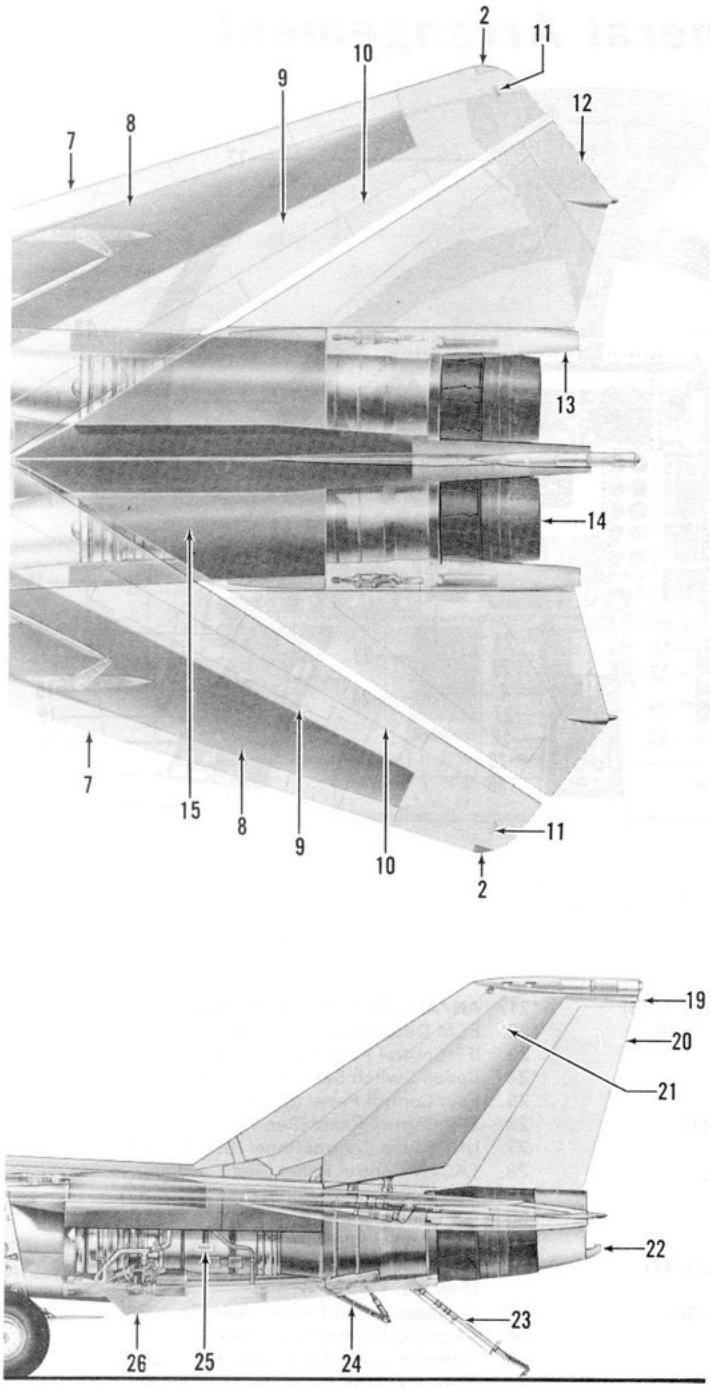


Figure 1-1. (Sheet 1)

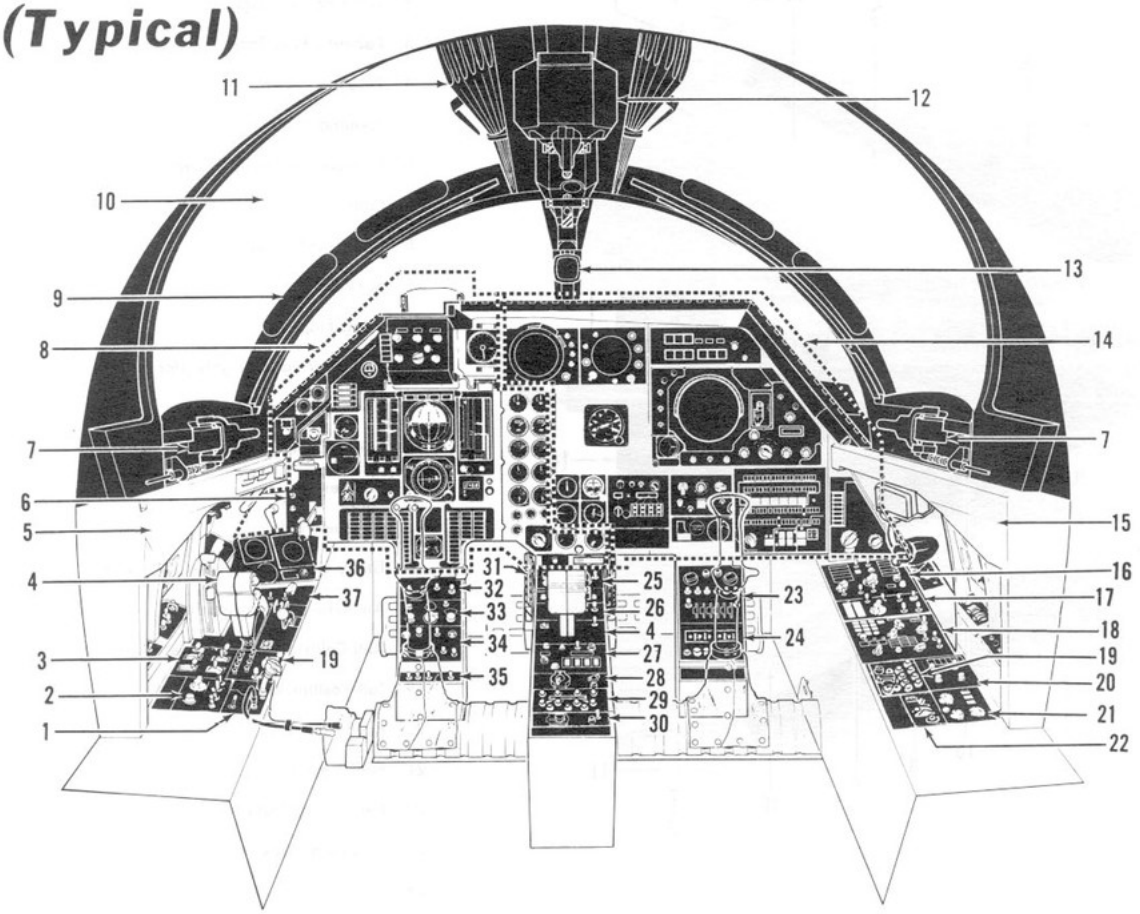


1. Entrance Ladder and Step.
2. Wing Position Lights.
3. Forward Fuel Tanks.
4. Rotating Glove.
5. Pivoting Pylons.
6. Primary Hydraulic System Reservoir.
7. Slats.
8. Wing Fuel Tanks.
9. Spoilers.
10. Wing Flaps.
11. Wing Formation Lights (Upr & Lwr).
12. Horizontal Stabilizer.
13. Speed Bumps.
14. Engines.
15. Aft Fuel Tank.
16. Utility Hydraulic System Reservoir.
17. Air Refueling Receptacle.
18. Anti-Collision Lights (Upper & Lower).
19. Tail Position Light.
20. Rudder.
21. Fuel Vent Tank.
22. Fuel Dump Outlet.
23. Arresting Hook.
24. Tail Bumper.
25. Fuselage Formation Lights (4).
26. Strake (2).
27. Forward Landing Gear Door/Speed Brake.
28. Air Conditioning System Cooling Air Intake.
29. Blow-In Doors.
30. Spike.
31. Fuel System Precheck Selector Panel.
32. Single Point Refueling Adapter Receptacle.
33. Aft Electronic Equipment Bay.
34. Weapons Bay.
35. Forward Electronic Equipment Bay.
36. Pitot Static Probe.

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Figure 1-1. (Sheet 2)

Crew Station General Arrangement (Typical)



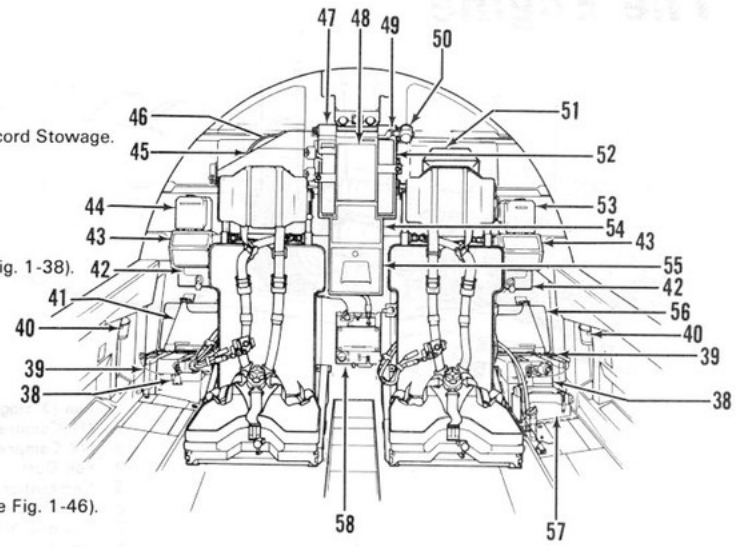
1. Left Station Oxygen Control Panel (See Fig. 1-46).
2. Air Conditioning Control Panel (See Fig. 1-40).
3. Autopilot Damper Panel (See Fig. 1-26).
4. Throttle Panels (See Fig. 1-5).
5. Left Sidewall (See Fig. 1-25).
6. Landing Gear Control Panel (See Fig. 1-16).
7. Internal Canopy Latch Handle (2).
8. Left Main Instrument Panel (See Fig. 1-6).
9. Mirrors (4).
10. Canopy Hatches (2).
11. Thermal Curtains (2).
12. Canopy Center Beam Assembly (See Fig. 1-47).
13. Magnetic Compass.
14. Right Main Instrument Panel (See Fig. 1-30).
15. Right Sidewall (See Fig. 1-42).
16. Store Control Panel (See Fig. 1-63).
17. SRAM Control Panel (See Fig. 1-64).
18. Computer Control Panel (See Fig. 1-62).
19. Communications Panel (See Fig. 1-54).
*20. CMDS Control Panel.
*21. AN/ALQ-94 ECM Control Panel.
*22. ECM Destruct Control Panel.
23. IFF Control Panel (See Fig. 1-58).
**24. Coded Switch Set Controller Panel.
25. Fuel Control Panel (See Fig. 1-9).
26. TFR Control Panel (See Fig. 1-76).
27. UHF Radio Control Panel (See Fig. 1-51).
28. TACAN Control Panel (See Fig. 1-56).
29. HF Radio Control Panel (See Fig. 1-52).
30. ILS Control Panel (See Fig. 1-57).
31. Ejection Handles (2).
32. Windshield Wash/Anti-Icing Control Panel (See Fig. 1-43).
33. Compass Control Panel (See Fig. 1-28).
34. Electrical Control Panel (See Fig. 1-11).
35. Antenna Select Panel (See Fig. 1-59).
36. Auxiliary Gage Panel (See Fig. 1-15).
37. Miscellaneous Switch Panel (See Fig. 1-73).

* See T.O. 1F-111(B)A-1-3.
** See Applicable Weapons Delivery Manual.

F0000000-F004 F

Figure 1-2. (Sheet 1)

38. Mission Data Box (2).
39. Relief Container (2).
40. Drinking Cups (2).
41. Weight and Balance T.O. and Flight Record Stowage.
42. Food and Liquid Stowage (2).
43. Air Diffusers (2).
44. Spare Bulb Stowage.
45. Quick Rescue Kit.
46. Letdown Chart Stowage.
47. Internal Lighting Fuse Panels (2) (See Fig. 1-38).
48. Lighting Control Panel (See Fig. 1-37).
49. Cabin Air Distribution Control Lever.
50. Utility Light.
51. Hood Stowage.
52. Letdown Chart Holder Stowage.
53. Safety Pin Stowage.
54. Circuit Breaker Panel (See Fig. 1-13).
55. Ground Check Panel (See Fig. 1-27).
56. Manual Stowage.
57. Arrival and Inflight IFR Chart Stowage.
58. Right Station Oxygen Control Panel (See Fig. 1-46).



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Figure 1-2. (Sheet 2)

Wing span (wings extended) — 70 feet
Height (to top of vertical tail) — 17 feet, 5 inches.

AIRCRAFT WEIGHT.

The aircraft operating weight is in the 49,000 pound class. The maximum gross weight varies with weapon configuration. For information regarding gross weights for various weapon configurations refer to Center-of-Gravity Envelope, Section VI. For specific aircraft weight, refer to the associated handbook of Weight and Balance Data, T.O. 1-1B-40.

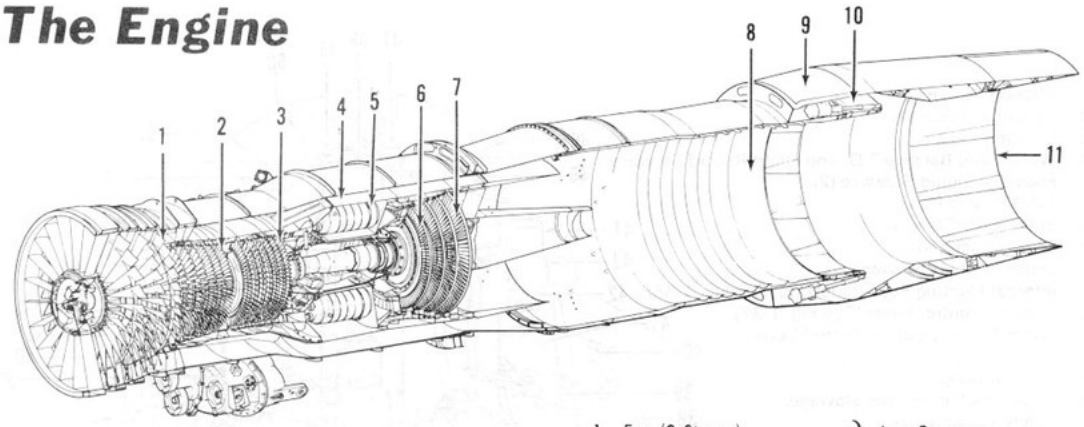
FLIGHT CREW.

The flight crew consists of a pilot and navigator seated side-by-side. The crew member assigned to the left crew station serves as pilot. The crew member assigned to the right crew station serves as navigator and operates the offensive and defensive equipment associated with the controls at that station.

ENGINES.

The aircraft is powered by two interchangeable Pratt and Whitney TF-30, sixteen-stage axial flow turbo-fan engines equipped with afterburners. See figure 1-3. The sea level, standard day uninstalled thrust rating of the engine is in the 12,000 pound class in military power and in the 20,000 pound class in afterburner. Provisions are made for starting the engines with an external pneumatic ground starter cart. Also, the left engine has the capability of being started without the aid of ground support equipment by means of a pyrotechnic cartridge. With either engine operating, the other engine can be started by using bleed air from the operating engine. Electrical power is supplied for the engine igniter plugs by an engine-driven alternator. An automatically controlled, movable spike is used in each inlet duct to control airflow to the engines. The engine air inlets are equipped with free floating blow-in doors which allow additional air to the engine as it is required. Splitter vanes are used at the front of the inlet ducts to remove the low energy air from the fuselage and the lower surface of the wing glove, thus preventing boundary layer air from disturbing engine inlet air. These features allow optimum engine performance throughout a wide range of aircraft operating conditions. Air from the inlet of each engine is routed through a single duct for both the basic engine section and the fan section. Three compressor stages provide initial pressurization of the

The Engine



1. Fan (3 Stages). } Low Pressure
2. N1 Compressor (6 Stages). } Compressor
3. N2 Compressor (7 Stages).
4. Fan Duct.
5. Combustion Chambers (8).
6. N2 Compressor Turbine (Single Stage).
7. Fan and N1 Compressor Turbine (3 Stages).
8. Afterburner Section.
9. Free Floating Blow-in Doors.
10. Variable Nozzle.
11. Tail Feathers.

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Figure 1-3.

air flowing into the engine and into the fan duct. The fan duct is a full annular duct which directs flow aft to join the engine airflow coming from the turbine discharge. The fan air develops a significant portion of total engine thrust. Engine air is compressed by 9 stages of the low pressure compressor (N1), of which three stages are the fan, and 7 stages of the high pressure compressor (N2). The air is then diffused into the combustion section, which contains 8 combustion chambers. The turbine section of the engine consists of a single-stage turbine to drive the high pressure compressor and a three-stage turbine to drive the low pressure compressor. The turbines are mechanically independent of each other. High pressure compressor speed is indicated by a tachometer. Speed of the low pressure compressor is not monitored except by an overspeed caution lamp. After leaving the turbine section of the engine, the air is joined with the fan air in the afterburner section. Bleed air from the engine compressors is used for cockpit and equipment bay air conditioning and pressurization, hydraulic system reservoir pressurization, fuel tank pressurization, hydraulic oil cooling, engine vortex destroyers, generator/CSD cooling, ground oil cooling, and windshield rain removal. Also, hot bleed air is used for engine inlet and guide vane anti-icing. (Refer to "Anti-Icing and Defog Systems", this section). Compressor bleed valves are installed on the engines to bleed air from the compressor during operation in certain flight regimes as an aid in the prevention of compressor stalls. The valves are controlled to automatically open at mach 1.75 ($\pm$0.10) and above.

ENGINE FUEL CONTROL SYSTEM.

Each engine fuel control system (figure 1-4) automatically provides optimum fuel flow for any throttle setting. This system responds to several engine operating parameters and makes it unnecessary to adjust the throttle in order to compensate for variations in inlet air temperature, altitude, or airspeed. The engine fuel system consists of a two-stage engine-driven fuel pump, fuel control unit, flowmeter, filter, pressurizing and dump valve, nozzles, and a fuel-oil heat exchanger. Fuel from the tanks is routed through the flowmeter to the centrifugal stage of the engine fuel pump, through a filter, and back to the gear stage of the pump. Bypass valves route fuel past the filter or first pump stage in event of failure of these components. The second pump stage delivers fully pressurized fuel to the fuel control unit, which provides metered fuel flow through the fuel-oil heat exchanger to the fuel pressurizing and

dump valve. This dual-function valve directs the fuel through the primary and secondary fuel manifolds to fuel nozzles which spray the fuel into the eight engine combustion chambers. When the fuel pressure drops during engine shutdown, the fuel pressurizing and dump valve automatically opens and drains the primary fuel manifold.

Fuel Control Unit.

The engine fuel control unit consists of a fuel metering system and a computing system which operates as a function of throttle setting, main combustion chamber pressure, high pressure rotor N2 speed, compressor inlet pressure, compressor inlet temperature, and flight mach number which is provided from the CADC.

Note

Malfunctions of the CADC are normally indicated by the CADC caution lamp. However, failures can occur which result in incorrect mach data from the CADC to the fuel control unit without an accompanying CADC caution lamp. The effect of a CADC mach failure on the fuel control unit can occur only when the landing gear handle is in the UP position and will manifest itself with a sudden reduction in engine thrust. This malfunction will also result in an abnormally high mach indication on the AMI.

The metering system selects the rate of fuel flow to be supplied to the engine in response to the throttle setting. However, metering sections are regulated by the fuel control computing system which monitors the various engine operating parameters. Fuel enters the fuel control through a filter provided with a spring-loaded bypass. Fuel metering is accomplished by maintaining a constant pressure across a variable valve area which is controlled by the computing system. The constant pressure is maintained by use of a pressure-regulating bypass valve. This valve consists of a servo-operated valve and a spring-loaded valve. Normally, the servo maintains constant valve regulation, but in the event of servo malfunction, the spring valve alone will provide adequate regulation. Deviations from the desired metering pressure are sensed in the valve regulating unit which varies the bypass flow area, thereby restoring the desired pressure by returning excess fuel to the pump inlet.

ENGINE AFTERBURNERS.

The afterburner (AB) augments engine thrust by injecting fuel into the engine exhaust stream in the afterburner section, where it is ignited by a hot streak ignition system. Operation is controlled by the throttle. When the throttle is moved forward within the afterburner range, the afterburner fuel control pressurizes the afterburner first fuel manifold, (zone 1) schedules light-off flow, and activates the variable nozzle system. This system senses a pressure change and controls the exit area of the afterburner exhaust nozzle. Six blow-in doors are located near the aft end of the afterburner. These doors open any time outside air pressure is greater than pressure inside the duct, allowing outside air to enter and thus increasing the total engine thrust. The trailing edge of the afterburner consists of free-floating leaves which reduce drag at the aft end of the engine by directing the exhaust gases into the slipstream with minimum turbulence.

Afterburner Fuel System.

The afterburner fuel system (figure 1-4) consists of the following major components: an exhaust nozzle pump, an afterburner fuel pump, an afterburner fuel control unit with integral exhaust nozzle control, and fuel spray rings. Fuel from the tanks flows through the flowmeter to the afterburner fuel pump. The exhaust nozzle pump is supplied fuel from the boost stage of the engine main fuel pump. The exhaust nozzle pump supplies fuel to the afterburner fuel control until a predetermined fuel flow rate is exceeded. At this flow rate, the afterburner fuel pump inlet is opened and begins to supply fuel to the afterburner fuel control unit. Fuel from the afterburner pump passes through a fuel-oil cooler before entering the afterburner fuel control unit. This unit includes a computer and a high pressure flow section. Fuel is then directed to the spray rings, where it is atomized and ignited in the afterburner combustion chamber. Five zones of afterburning can be selected through the afterburner fuel control unit, which schedules fuel to the spray rings in the various zones of the afterburner as a function of throttle setting. When the throttle is advanced for afterburner initiation and when high pressure compressor speed exceeds approximately 83 percent rpm, the afterburner initiation valve schedules light-off fuel flow until afterburner light-off occurs, as sensed by the exhaust nozzle control.

Afterburner Ignition.

The function of the afterburner ignition system is to provide a means of igniting fuel in the afterburner to initiate afterburner operation. With the advancement of the throttle into AB, the afterburner igniter valve releases an auxiliary squirt of fuel which is injected just aft of the fourth stage turbine; then zone 1 fuel flow begins. After zone 1 flow begins, initial afterburner ignition is provided by a hot streak ignition system. The igniter valve injects a slug (main squirt) of fuel into number 4 combustion chamber creating a local overrich mixture. This fuel is ignited by the combustion chamber fire and the rich mixture results in a longer flame that burns past the turbines to provide hot streak ignition for the auxiliary squirt, which in turn ignites zone 1. Completion of the main squirt into number 4 combustion chamber provides a signal for cessation of the auxiliary squirt. If afterburner operation is not achieved, the throttle must be retarded to MIL or below and readvanced into AB to repeat the above series of events required for afterburner ignition.

Engine Fuel Control System

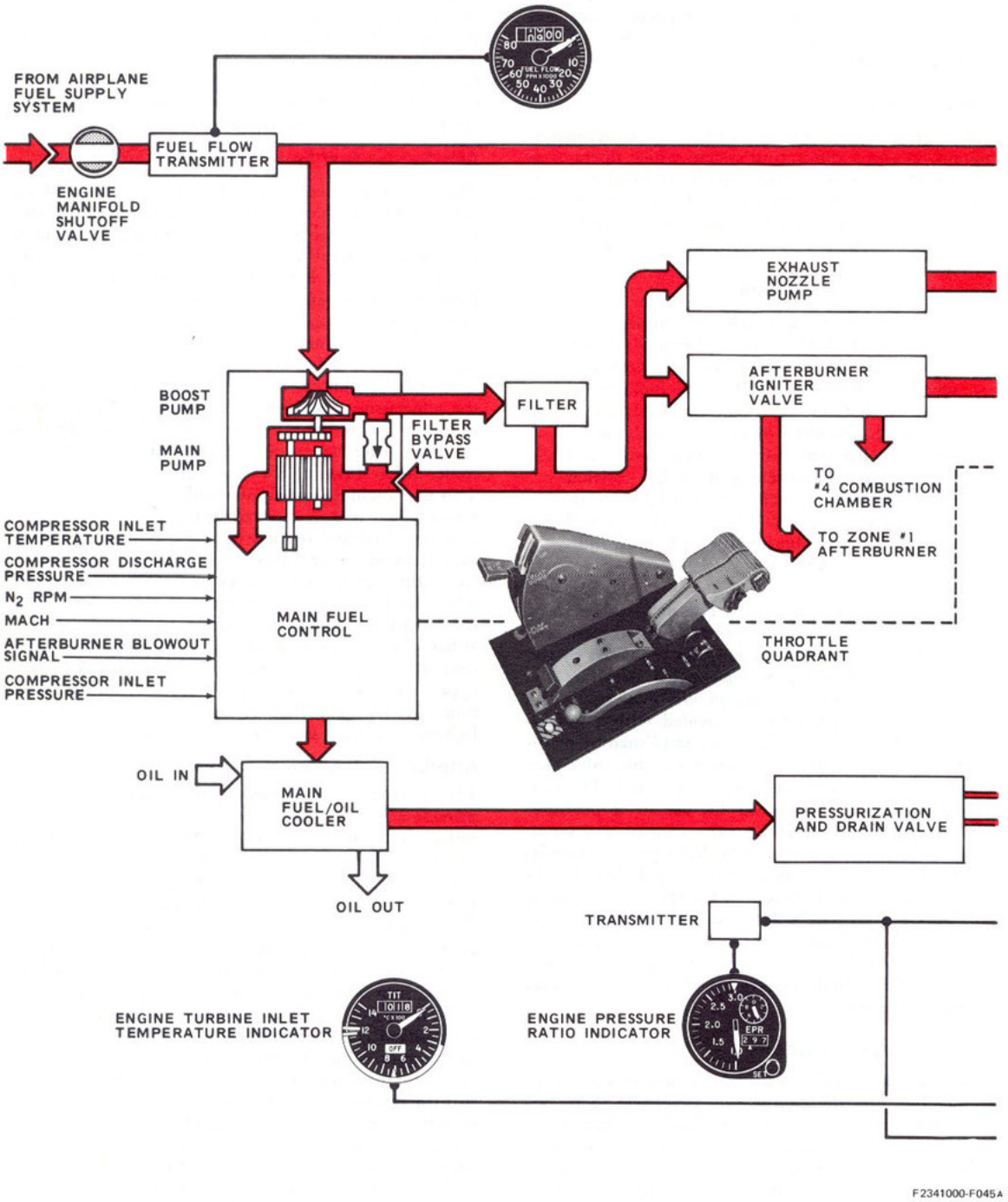


Figure 1-4. (Sheet 1)

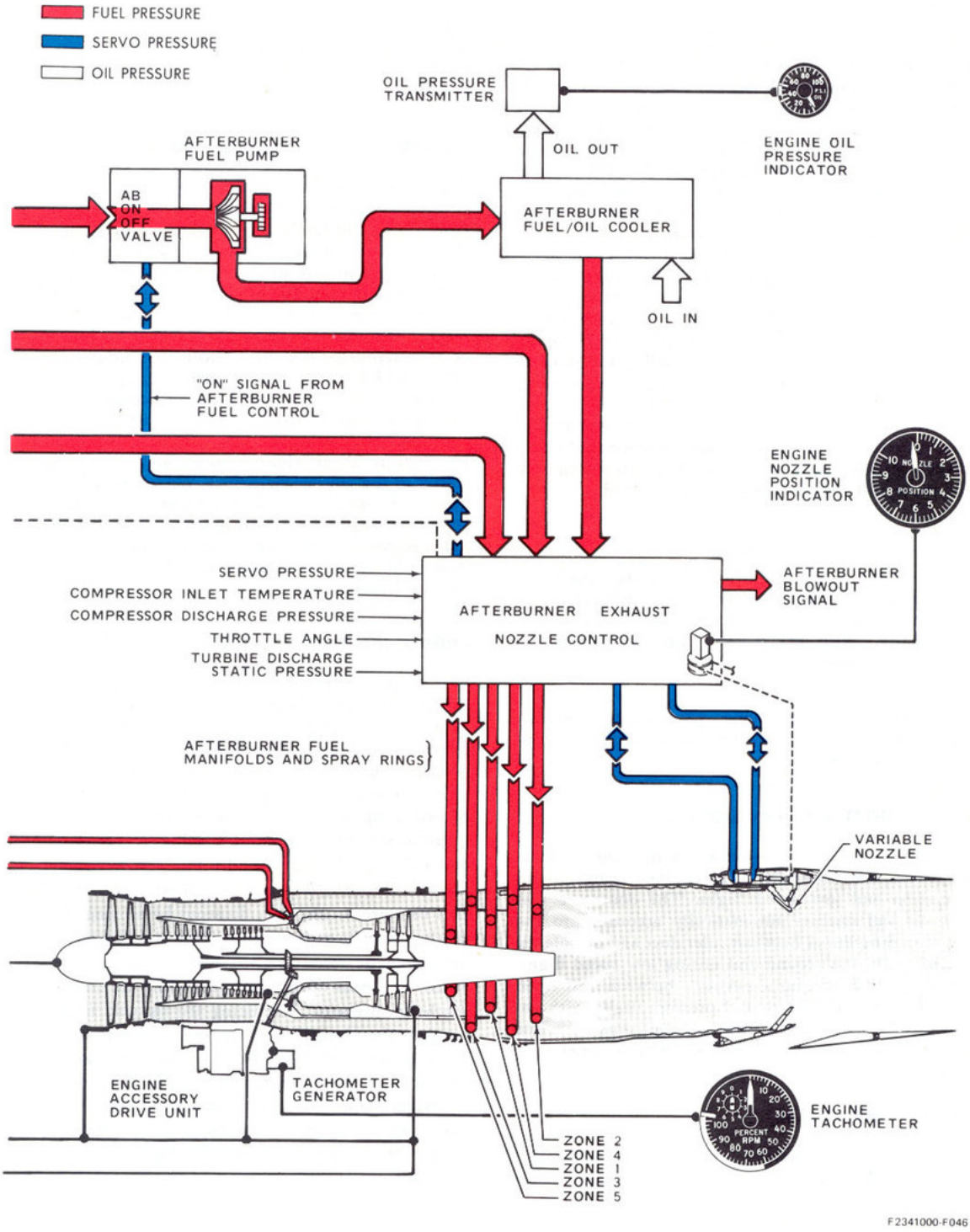


Figure 1-4. (Sheet 2)

ENGINE INLET SPIKES.

Engine inlet air velocity is regulated throughout the entire aircraft speed range by a movable spike located in the inlet of each engine. Each spike is a quarter circle, conical-shaped, variable diameter body that is independently movable forward and aft. The spikes are located in each air intake at the intersection of the wing lower surface and the fuselage boundary plate. Position and shape of the spikes are changed automatically to vary the inlet geometry and to control the inlet shock wave. Local air pressure changes due to variation in inlet local mach and diffuser exit mach number are measured by sensors in the spike control unit. Signals from the control unit operate hydraulic actuators powered by the utility hydraulic system to position the spike fore and aft and to adjust the spike cone angle by contracting and expanding the spike as required. When the aircraft is on the ground, and any time the aircraft speed is less than approximately mach 1.0, the spike will be full forward and fully retracted. In the event the system malfunctions, a one-shot pneumatic override system is provided to position and lock the spike full forward and fully contracted.

Note

Once the pneumatic override has been used, the hydraulic shuttle valves in the spike control system must be repositioned on the ground, with the utility hydraulic system depressurized.

An electronic anti-icing system prevents ice formation on the sensors. Refer to "Anti-Icing and Defog System", this section.

ENGINE INLET BLOW-IN DOORS.

The engine inlets are equipped with blow-in doors to provide additional air to the engine during ground operation and low speed flight. The doors are free floating and operate automatically according to the pressure differential between inlet duct and outside air pressure. On the ground and during low speed flight, the doors will assume a position between full open and closed as this differential pressure varies with engine demand. As speed is increased the doors will close and remain closed when ram effect increases inlet pressure to above outside air pressure.

VORTEX DESTROYERS.

The vortex destroyers serve to prevent the formation of vortexes below the inlet, thereby preventing foreign objects from being entrained in a vortex and sucked into the engine. When the weight of the aircraft is on the landing gear, a ground safety switch, located on the landing gear, automatically activates the vortex prevention air screen.

ENGINE VARIABLE EXHAUST NOZZLES.

The variable nozzle system incrementally opens and closes the engine exhaust nozzle for afterburner modulation. The control is a hydromechanical computing device that determines and sets the nozzle area required to maintain a desired turbine pressure ratio during afterburner operation. The nozzle position is scheduled by the throttle setting and governed by internal pressure ratio in the engine. The nozzle is closed for all ranges of nonafterburner operation except for ground engine idle, at which time it is positioned fully open for minimum thrust. If afterburner blowout occurs, the blowout signal valve is actuated, and the nozzle closes. In addition, the afterburner fuel selector valve closes off fuel flow to all afterburner zones, and a signal is directed to the engine main fuel control to reduce fuel flow to the main combustion chamber. When the nozzle has moved to the closed position, the blowout signal is removed. Afterburner operation can again be initiated; however, the throttle must first be moved to a nonafterburning position.

ENGINE IGNITION SYSTEM.

The engine ignition system provides combustion during the starting cycle and engine ignition in the event of a flameout. Each engine has a dual main ignition system including two ignition exciters, two igniter plugs, an ignition alternator, and an automatic restart switch. The alternator is engine driven and is capable of providing sufficient energy to both exciters of the ignition system for ground starting or for windmill starts during all flight conditions. The automatic restart circuit energizes the ignition system in the event of a combustion chamber flameout by sensing the rate of change of burner pressure. This is accomplished by an automatic restart switch which will remain activated for 15 to 60 seconds depending on compressor discharge pressure. Advancing the throttle from OFF actuates the throttle ignition switch for that engine. This action provides ignition when the engine start switch is in PNEU or CARTRIDGE. Ignition is cut off when the ground start switch returns to OFF. This occurs when the starter centrifugal cutout switch opens on the last engine to be started. Ingition is also cut off when the throttle is retarded to the OFF position.

ENGINE STARTING SYSTEM.

The left engine can be started by pyrotechnic cartridge, both engines can be started by external pneumatic pressure, and once either engine is running, the remaining engine can be started by pneumatic crossbleed from the operating engine. Electrical power required for starting can be obtained from either an external ground source or the aircraft battery. When starting the left engine with the cartridge, the cartridge is ignited by placing the ground start switch to CARTRIDGE and lifting the left throttle out of the OFF position. When starting the engines with a pneumatic source, either external or crossbleed, placing the ground start switch to PNEU and lifting the throttle out of the OFF position, opens the starter pressure shutoff valve on the engine being started and allows pneumatic pressure to drive the respective starter. After a pneumatic start, the ground start switch will return to OFF when the centrifugal cutout switch in the starter on the second engine started opens. This breaks the starter control circuit and allows the starter pressure shutoff valve to close, shutting off the pneumatic pressure. Two spare cartridges can be carried in the main landing gear wheel well.

ENGINE CONTROLS AND INDICATORS.

Throttles.

A set of throttles (7, figure 1-5) is provided for both crew members. The respective left and right throttles in each set are mechanically linked together. Each throttle provides thrust setting adjustment for its respective engine. Pneumatic power boost, from the cabin pressurization system, is provided to assist throttle movement. Throttle friction for both sets of throttles is controlled by means of the friction lever located adjacent to the left set. Moving the lever toward INCR increases throttle friction, and moving the lever toward DECR decreases the friction. Force required to move the throttles varies from 2 to 30 pounds, with pneumatic boost, depending on the position of the friction lever. In the event pneumatic boost is lost, the force required to move the throttles is from 10 to 40 pounds depending on the friction lever position. Both

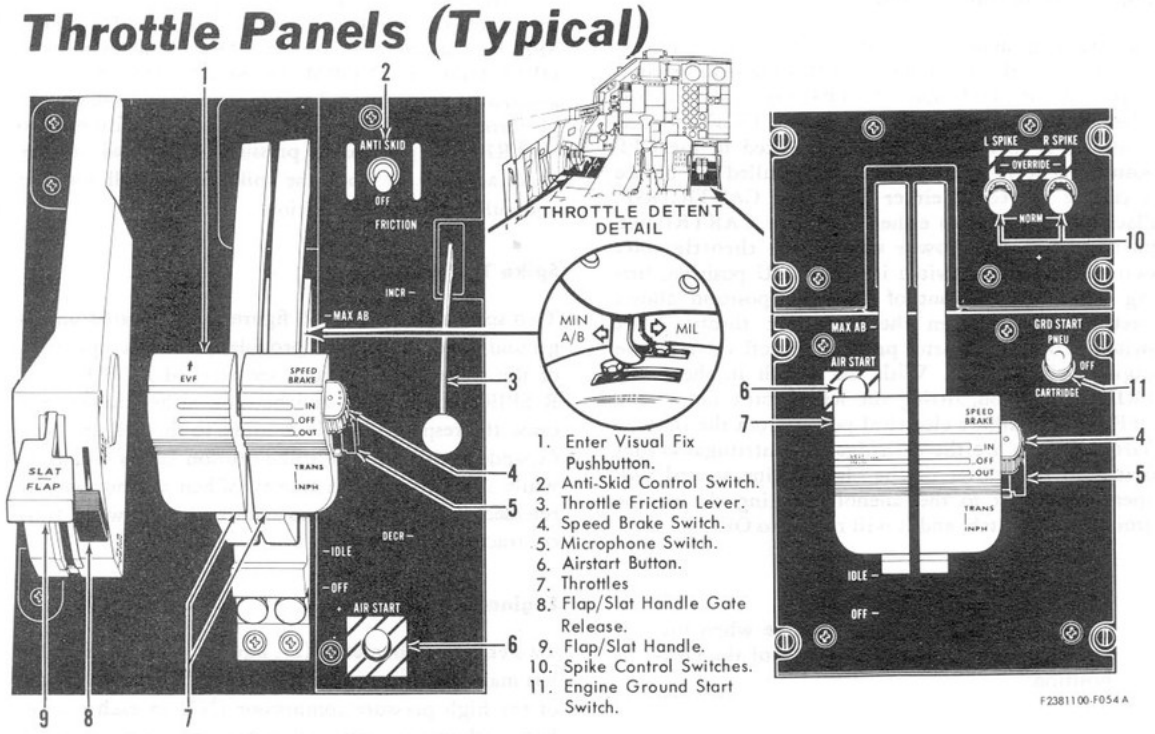


Figure 1-5.

sets of throttles have positions marked OFF, IDLE, MIL, and MAX AB. Only the left set of throttles can be raised to go into or from the OFF position. The center throttles cannot be used for engine starting or shutdown. When the left set of throttles is lifted to move them out of the OFF position, the throttle starter switches are actuated. If the ground start switch is in the CARTRIDGE position, lifting the left throttle will automatically fire the left engine starter cartridge. If the ground start switch is in the PNEU position, lifting either throttle will open the starter pneumatic pressure shutoff valve on the respective engine to allow starting by pneumatic pressure. Movement of either throttle forward of the OFF position activates the respective engine ignition system. A detent at the MIL position provides a means of readily selecting military power. A detent is also provided at the minimum AB position. To attain the minimum AB detent position the throttle must first be advanced into the afterburner range and then retarded until the detent is felt. Refer to figure 1-5 for a detail of both the MIL and MIN AB throttle positions. The left throttle of the left set includes an EVF (enter visual fix) button for updating the bomb nav system. The right throttle of each set includes a microphone switch and a speed brake switch.

Engine Ground Start Switch.

The engine ground start switch (11, figure 1-5), located on the center throttle panel, is a three-position switch marked PNEU, OFF, and CARTRIDGE. The switch is solenoid held in the PNEU and CARTRIDGE positions and is spring-loaded to and locked in the OFF position. The switch toggle must be pulled out before it can be moved to either PNEU or CARTRIDGE. Placing the switch to either PNEU or CARTRIDGE position supplies power to arm the throttle start switches. With the switch in the PNEU position, lifting either throttle out of the OFF position allows electrical power from the respective throttle start switch to open the starter pressure shutoff valve on the engine being started. With the switch in the CARTRIDGE position, lifting the left throttle out of the OFF position allows electrical power from the throttle start switch to fire the cartridge. A centrifugal cutout switch in the starter of the last engine started will open the circuit to the solenoid holding the engine ground start switch, and it will return to OFF.

Note

Nose wheel steering is inoperative when the engine ground start switch is out of the OFF position.

Air Start Buttons.

Two air start pushbuttons marked AIR START (6, figure 1-5), one located on each throttle panel, provide a means of obtaining ignition for air-starting the engines. When either button is momentarily depressed, ignition will be provided to both engines for approximately 55 seconds after the air start button is released.

Ground Ignition Cutoff Switch.

The ground ignition cutoff switch (10, figure 1-27), located on the ground check panel, is labeled GRD IGN and has two positions marked NORM and OFF. When the switch is in OFF, the engine electrical ignition system is deactivated. When the switch is in the NORM position, the ignition circuits are activated.

Mach Trim Test Switch.

The mach trim test switch (3, figure 1-27), located on the ground check panel, has two positions marked NORM and TEST. The switch is spring-loaded to the NORM position. The switch is provided for maintenance ground check of the engine mach lever on the fuel control unit.

Spike Control Switches.

Two spike control switches (10, figure 1-5), located on the center throttle panel, are labeled L SPIKE and R SPIKE. The switches are lever-lock switches marked OVERRIDE and NORM. In NORM, the spikes are automatically controlled to maintain maximum engine performance. When either switch is positioned to OVERRIDE, pneumatic pressure is applied to the spike actuator to move the spike to the full forward and fully contracted position.

Spike Test Buttons.

Two spike test buttons (2, figure 1-27), located on the ground check panel, are provided to check operation of the spikes. The buttons are marked L SPIKE and R SPIKE. Depressing and holding either button will cause the respective spike to move to the full aft, fully expanded position. The spike caution lamps will light while the spikes are in transit. When the buttons are released, the spikes will move to the full forward, fully contracted position.

Engine Tachometers.

Two engine tachometers (27, figure 1-6), located on the left main instrument panel, indicate the percent of rpm of the high pressure compressor (N2) in each engine. Each tachometer main dial is graduated from 0 to 100 percent rpm in increments of 2 percent; the subdial is graduated from 0 to 10 percent in increments of 1 percent.

Engine Overspeed Caution Lamp.

Two engine overspeed lamps are located on the main caution lamp panel (figure 1-36). When lighted, the letters L ENG OVERSPEED and R ENG OVERSPEED are visible. The lamps will light to indicate N-1 compressor overspeed. In addition, the lamps will light prior to engine start, provided there is electrical power on the aircraft and will go out prior to reaching idle rpm. The lamps operate on 28 volt dc electrical power from the essential dc bus.

Engine Fuel Flow Indicators.

Two engine fuel flow indicators (29, figure 1-6), located on the left main instrument panel, show fuel flow for each engine in pounds per hour. The indicators are calibrated from 0 to 80,000 pph in increments of 2000 pph. A digital readout of fuel flow to the nearest 100 pph is displayed on the face of the indicator.

Engine Nozzle Position Indicators.

Two engine nozzle position indicators (30, figure 1-6), located on the left main instrument panel, show nozzle position. The indicators are calibrated from 0 (closed) to 10 (fully open). The indicators use 115 volt ac power from the left main ac bus.

Engine Oil Pressure Indicators.

Two engine oil pressure indicators (32, figure 1-6), located on the left main instrument panel, indicate engine oil pressure in pounds per square inch. The indicators are calibrated from 0 to 100 psi in increments of 5 psi. The oil pressure indicating system operates on 26 volts ac which has been reduced by a transformer which has an input of 115 volts ac from the ac essential bus.

Engine Pressure Ratio Indicator.

Two engine pressure ratio (EPR) indicators (31, figure 1-6), located on the left main instrument panel, indicate the ratio of turbine discharge pressure to engine inlet pressure. The main dial of each indicator is calibrated from 1.0 to 3.0 in 0.1 increments. A smaller circular dial on the indicator face is calibrated in 0.01 increments for precise reading. A set button on each indicator moves an index pointer to set computed EPR. The precise EPR position of the reference pointer is displayed by a digital readout window on the indicator face. 115 volt ac power to the indicators is supplied from the essential ac bus.

Engine Turbine Inlet Temperature Indicators.

Two engine turbine inlet temperature (TIT) indicators (28, figure 1-6), located on the left main instrument panel, show turbine inlet temperature in degrees centigrade. The indicator dials are graduated from 0 to 1400 degrees in 50-degree increments. In addition, a digital readout of turbine inlet temperature in two-degree increments is displayed. Power to the TIT indicators is supplied from the 28 volt dc engine start bus; thus operation is normal with battery power. A flag marked OFF appears on the face of the indicator when power to the indicator is interrupted.

Spike Caution Lamps.

Two amber spike caution lamps, one for the spike in each engine inlet, are located on the main caution lamp panel (figure 1-36). When lighted, the letters L ENG SPIKE and R ENG SPIKE are visible. A spike caution lamp lights when the aircraft mach number is less than 0.35 and the respective spike is not full forward and fully contracted. When the spike control switch is placed to OVERRIDE, the spike caution lamp will light and remain on until the spike has reached the full forward and fully contracted position. During spike self-test the lamps will light until the spike has reached its full aft and full expanded position. The lamps operate on 28 volt dc electrical power from the essential dc bus.

Engine Oil Hot Caution Lamps.

The two engine oil hot caution lamps are located on the main caution lamp panel (figure 1-36). When the oil temperature of either engine exceeds 250 degrees F, the associated lamp will light. When lighted, the following letters will be visible in the lens of the respective lamp: L ENG OIL HOT and R ENG OIL HOT.

ENGINE FIRE DETECTION AND EXTINGUISHING SYSTEM.

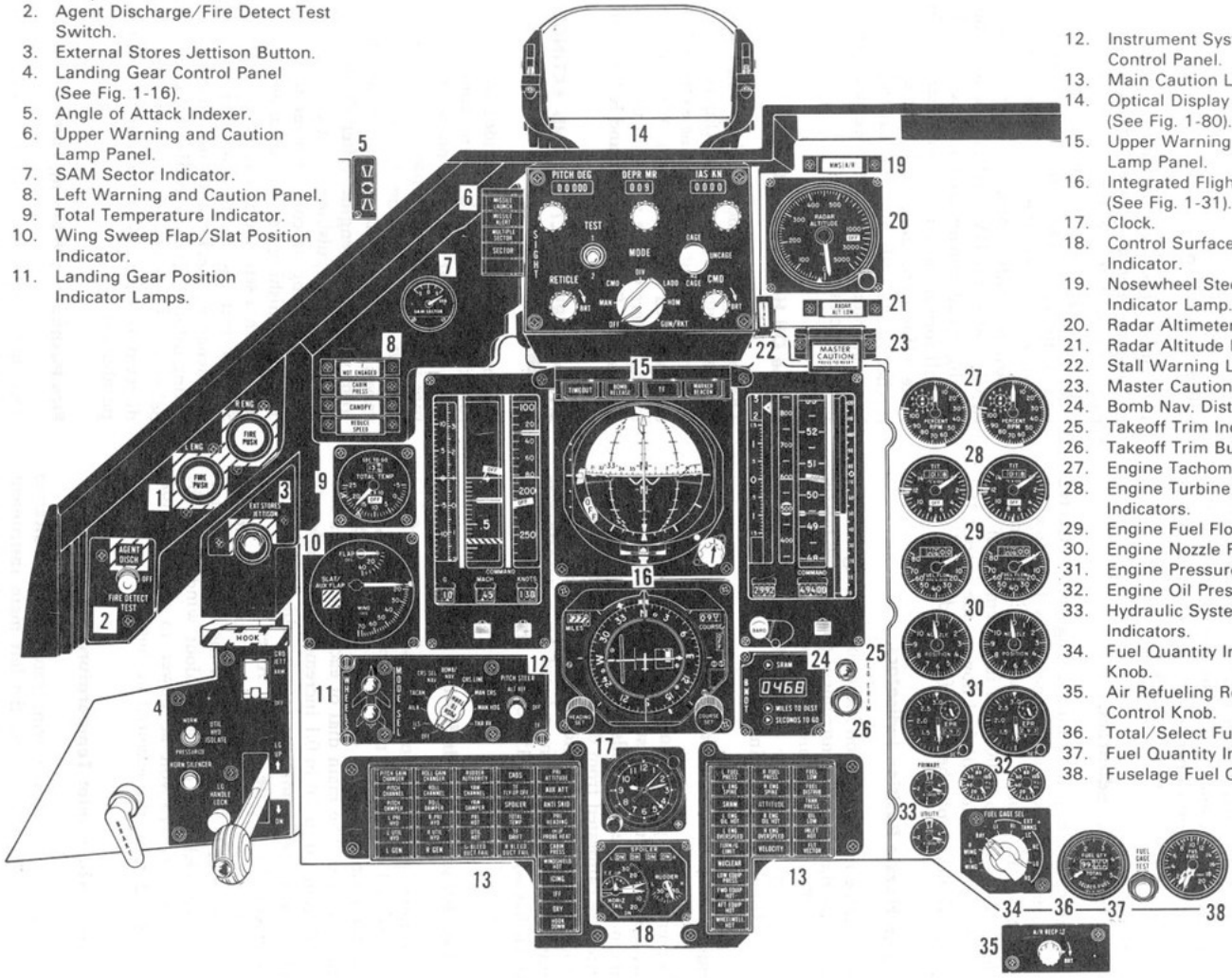
Engine fire detection is provided by sensing elements routed throughout each engine compartment. Should a fire or overheat condition occur, the rise in temperature is detected by the sensors which light the respective left or right engine fire warning lamp. Shutoff valves are provided to isolate fuel and hydraulic fluid from the affected engine. After the shutoff valves are closed, fire extinguishing agent can be discharged into the affected engine compartment to put out the fire. The extinguishing agent is contained in a single container with a separate discharge valve for each engine. Self-test features are incorporated in the system for maintenance checks and troubleshooting. The fire extinguishing agent is available for one engine only. Selection of the engine to which the agent is to be directed is made by depressing the appropriate fire pushbutton.

Fire Pushbutton Warning Lamps.

Two fire pushbutton warning lamps (1, figure 1-6), labeled L ENG and R ENG, are located on the left main instrument panel. Depressing either button will

Left Main Instrument Panel (Typical)

1. Engine Fire Pushbuttons Warning Lamps.
2. Agent Discharge/Fire Detect Test Switch.
3. External Stores Jettison Button.
4. Landing Gear Control Panel (See Fig. 1-16).
5. Angle of Attack Indexer.
6. Upper Warning and Caution Lamp Panel.
7. SAM Sector Indicator.
8. Left Warning and Caution Panel.
9. Total Temperature Indicator.
10. Wing Sweep Flap/Slat Position Indicator.
11. Landing Gear Position Indicator Lamps.



12. Instrument System Coupler Control Panel.
13. Main Caution Lamp Panel.
14. Optical Display Sight Set (See Fig. 1-80).
15. Upper Warning and Caution Lamp Panel.
16. Integrated Flight Instruments (See Fig. 1-31).
17. Clock.
18. Control Surface Position Indicator.
19. Nosewheel Steering/Air Refueling Indicator Lamp.
20. Radar Altimeter.
21. Radar Altitude Low Warning Lamp.
22. Stall Warning Lamp.
23. Master Caution Lamp.
24. Bomb Nav. Distance Time Indicator.
25. Takeoff Trim Indicator Lamp.
26. Takeoff Trim Button.
27. Engine Tachometers.
28. Engine Turbine Inlet Temperature Indicators.
29. Engine Fuel Flow Indicators.
30. Engine Nozzle Position Indicators.
31. Engine Pressure Ratio Indicators.
32. Engine Oil Pressure Indicators.
33. Hydraulic System Pressure Indicators.
34. Fuel Quantity Indicator Selector Knob.
35. Air Refueling Receptacle Lights Control Knob.
36. Total/Select Fuel Quantity Indicator.
37. Fuel Quantity Indicator Test Button.
38. Fuselage Fuel Quantity Indicator.

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Figure 1-6.

close the engine fuel shutoff valve and utility and primary hydraulic system shutoff valves to the respective engine and will arm the extinguishing agent discharge switch to the affected engine. Depressing the button again will open the fuel shutoff valve and disarm the fire extinguisher agent discharge valve; however, the hydraulic shutoff valves will remain closed. The buttons are covered by frangible covers to provide a visual indication when the buttons have been actuated.

WARNING

Caution must be exercised to prevent inadvertently depressing the wrong pushbutton and shutting down the good engine since the hydraulic shutoff valves cannot be reopened in-flight.

Agent Discharge/Fire Detect Test Switch.

The agent discharge/fire detect test switch (2, figure 1-6), located on the left main instrument panel, is a three-position lever-lock switch marked AGENT DISCH, OFF, and FIRE DETECT TEST. The switch is spring-loaded to the OFF position, and is locked out of the AGENT DISCH position to prevent inadvertent actuation. To move the switch to AGENT DISCH, it must be pulled out of the lock. Momentarily positioning the switch to the AGENT DISCH position will discharge fire extinguishing agent into the engine compartment of the engine selected after depressing the affected engine fire pushbutton warning lamp. Holding the switch to the FIRE DETECT TEST position will light both fire warning lamps, the WHEEL WELL HOT caution lamp and, after T.O. 1F-111-1151, both bleed duct fail caution lamps, if the engine fire detection system and wheel well overheat detection system are operational.

Note

The engine fire extinguishing agent will be depleted after one actuation.

Fire Detection System Ground Test Switches.

(Prior to T.O. 1F-111-1151)

Two fire detection system ground test switches, located on the ground check panel, are labeled R ENG and L ENG. The switches have three positions marked CONTROL BOX, NORM, and ELEMENT and are spring-loaded to the NORM position. The switches are used to ground check the system circuitry. Only one circuit can be checked at a time.

Fire Detection System Short Test Button.

(Prior to T.O. 1F-111-1151)

The fire detection system short test button, located on the ground check panel, is marked SHORT TEST. The button is provided to ground check the system short discriminating test circuit during maintenance or troubleshooting.

Fire Detection System Module Self-Test Button.

(After T.O. 1F-111-1151)

The fire detection system module self test button (1, figure 1-27), located on the ground check panel, is labeled MODULE SELF TEST. The button is used to ground check the fire detection system circuitry in the event the WHEEL WELL HOT caution lamp, either of the two fire pushbutton (FIRE PUSH) warning lamps, or either of the bleed duct fail caution lamps (figure 1-36) fails to light when the agent discharge/fire detect test switch is positioned to FIRE DETECT TEST.

ENGINE OPERATION.

Engine Acceleration.

Engine acceleration time is severely affected by the amount of compressor discharge air being bled from the engine and by outside temperature. The engine may require 15 seconds to accelerate from idle to military when air conditioning air is taken from that engine during ground operation. In flight this effect is minimized, but during final approach for landing, engine acceleration may require as much as 10 seconds to increase thrust from idle to military.

Afterburner Rumble.

During zone 4 and low zone 5 operation, the engine may exhibit afterburner fluctuations at 40 to 60 cps. This will be heard and felt as a low frequency beat-type vibration in the airframe and is referred to as afterburner rumble (rough combustion). Should this occur, change throttle settings to a point where it can not be detected.

OIL SUPPLY SYSTEM.

Each engine is equipped with an oil supply system which consists of an oil tank, a main supply pump, six scavenger pumps, a deoiler, two filters, an overboard breather pressurizing valve, a pressure valve, and three oil coolers (air-oil, fuel-oil, and afterburner fuel-oil). The air-oil cooler operates with engine bleed air. Oil is fed to the main oil supply pump from the oil tank. It is then pumped in series through the two filters, the air-oil cooler, fuel-oil cooler, and afterburner fuel-oil cooler. Oil flow through the fuel-oil coolers is controlled by temperature and pressure sensing bypass valves. The oil is then directed to the engine bearings and to the accessory gearbox. Scavenger pumps return the oil to the oil tank. Capacity of the tank is five gallons, four gallons of which are usable. For oil specification and servicing location, see figure 1-83.

ENGINE OIL QUANTITY INDICATOR.

The engine oil quantity indicator, located on the auxiliary gage panel (4, figure 1-15), is labeled L and R for the left and right engine, respectively. Each display is graduated from 0 to 16 in one quart increments. A pointer for each display provides an indication of the number of quarts of usable oil remaining in each oil tank. The indicator operates on 115 volt ac power from the left main ac bus.

Note

- The indicated oil quantity exhibits variations during normal operations. When a cold engine is started, the oil quantity indication may drop as much as five quarts at idle power settings. After an engine has warmed up, the oil quantity indications may vary as much as three quarts (increase or decrease) with varying power settings from IDLE to MIL.
- If the oil quantity indicating system for either engine malfunctions, that indicator will drive to below zero and the oil low caution lamp will be inoperative for that engine. The oil low caution lamp will, however, continue to monitor the oil quantity for the other engine. To confirm that the malfunction is in the oil quantity indicating system rather than an actual oil low condition, the oil low caution lamp may be checked by holding the oil/oxygen quantity indicator test switch, located on the auxiliary gage panel, to OIL QTY.
- The oil quantity indicator gives a valid reading only during engine run and up to 15 minutes after engine shutdown, provided electrical power is supplied by external power source or the generator of the other engine. This is due to the fact that oil in the tank seeps into the accessory gear box.

OIL/OXYGEN QUANTITY INDICATOR TEST SWITCH.

The oil/oxygen quantity indicator test switch (6, figure 1-15), located on the auxiliary gage panel, is used to check the oil quantity indicator. The switch has three positions marked OXY QTY, OIL QTY and is spring-loaded to the center unmarked OFF position. Holding the switch to the OIL QTY position drives the left indicator to 5 quarts, the right indicator to 5.7 quarts, and lights the oil low caution lamp. Refer to Oxygen System this section for the oxygen quantity indicator test functions of the switch.

OIL LOW CAUTION LAMP.

An oil low caution lamp (figure 1-36), located on the main caution lamp panel, lights any time the oil level in either the left or right engine oil supply tank drops to four (4) quarts usable oil remaining or when the test switch is used. When the lamp is lighted, the letters OIL LOW are visible.

FUEL SUPPLY SYSTEM.

The fuel supply system (figure 1-8) consists of a forward and aft integral fuselage tank, two integral wing tanks, an integral vent tank, and the associated fuel pumps, controls, and indicators. The fuel system employs twelve fuel pumps, of which six deliver fuel to the engines and six are used to transfer fuel from the wing tanks and weapons bay tanks to the fuselage tanks. Provisions are made for inflight refueling of the internal and external fuel tanks from a boom-type tanker aircraft. Single-point refueling is provided for ground servicing. All tanks are equipped with automatic refuel shutoff valves. Gravity refueling can be accomplished through filler caps in the wings and fuselage. For fuel tank capacities refer to figure 1-7.

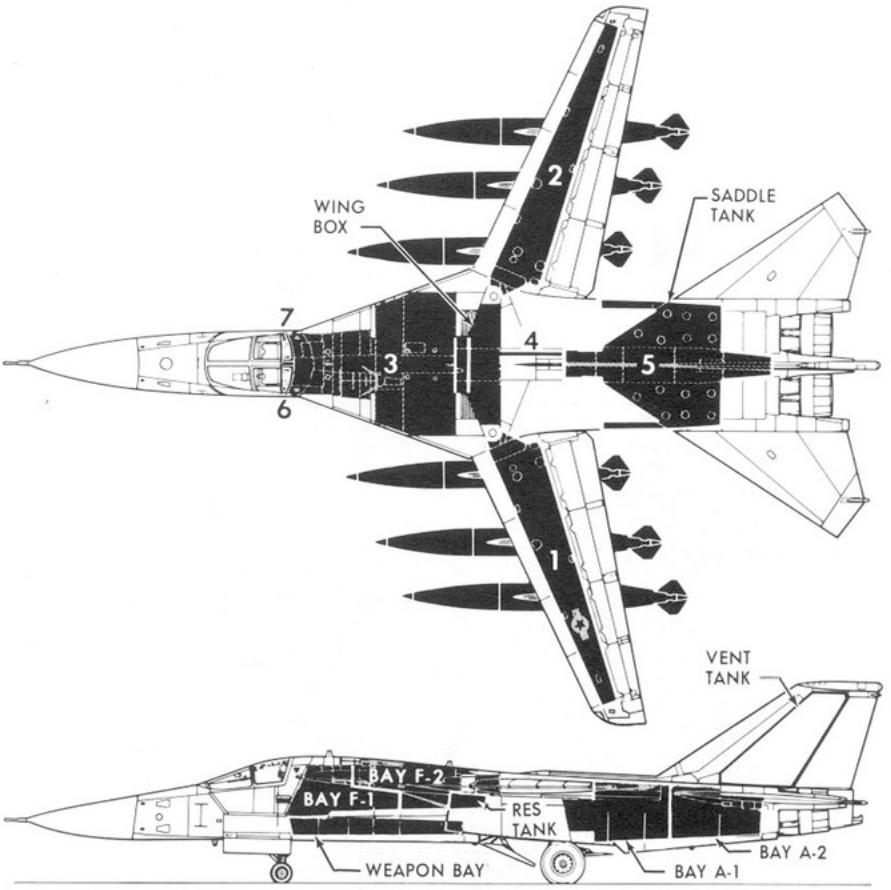
FUEL TANKS.

The fuel tanks consist of internal forward and aft fuselage tanks, left and right internal wing tanks, up to six detachable external wing tanks, two removable weapon bay tanks and an integral vent tank in the vertical stabilizer. See figure 1-7 for tank locations and capacities. The fuselage tanks are divided into compartments called bays. The forward fuselage tank is divided into bays F-1 and F-2 and a reservoir tank. The reservoir tank includes the fuel contained in the wing carry through box. Flapper valves allow fuel to flow from bay F-1 to bay F-2 and from bay F-2 to the reservoir tank. The reservoir tank reserves approximately 2552 pounds of fuel after all other fuel in the system has been used. A float switch in the reservoir tank provides a caution lamp indication when the fuel level in the reservoir tank drops below 2300 ±235 pounds. The aft tank is divided into bay A-1, incorporating two "saddle" tanks, and bay A-2. Interconnecting stand pipes provide fuel flow between bays A-1 and A-2, and ejector pumps transfer saddle tank fuel into bay A-1. All fuel in the wing external and internal tanks and weapons bay tanks must be transferred into the fuselage tanks before it can be used. All tanks are pressurized by cooled engine compressor bleed air to prevent fuel vaporization. The vent tank provides space for expansion of fuel in the system when all tanks are fully serviced. Booster pumps in the fuselage tanks are provided for engine feed, and transfer of fuel from the aft to forward tank. Transfer pumps in the internal wing tanks and weapon bay tanks transfer fuel into the fuselage. Air pressure is used to transfer fuel from the external wing tanks.

FUEL QUANTITY MEASURING SYSTEM.

The fuel quantity measuring system is a basic capacitance sensing type system. There are four independent indicating functions: forward, aft, select, and total.

Fuel Quantity and Tank Arrangement (Typical)



LOCATION		QUANTITY			
		USABLE FUEL		FULLY SERVICED	
		GALLONS	POUNDS	GALLONS	POUNDS
1	LEFT WING INTERNAL TANK	389.2	2,530	390.7	2,540
2	RIGHT WING INTERNAL TANK	389.2	2,530	390.7	2,540
3	FORWARD FUSELAGE TANK	2,808.3	18,254	2,822.9	18,349
4	FUEL LINES	28.3	184	53.4	347
5	AFT FUSELAGE TANK	1,423.7 1,378.7*	9,254 8,962*	1,425.8 1,380.8*	9,268 8,975*
6	LEFT WEAPONS BAY TANK	284.6	1,852	287.4	1,868
7	RIGHT WEAPONS BAY TANK	299.7	1,948	300.9	1,956
TOTAL		5,623.0 5,578.0*	36,552 36,260*	5,671.8 5,626.8*	36,868 36,575*

NOTES:

1. These are average figures based on single point refueling at normal ramp attitude. Weights based on JP-4 fuel at 6.5 pounds per gallon. (Std. Day Only)

2. Each external tank, when carried, will have the following capacities.

USABLE FUEL		FULLY SERVICED	
GALLONS	POUNDS	GALLONS	POUNDS
600.0	3,900	602.2	3,914

*After T.O. 1F-111-1162

F0000000-F009B

Figure 1-7.

Fuel Supply System

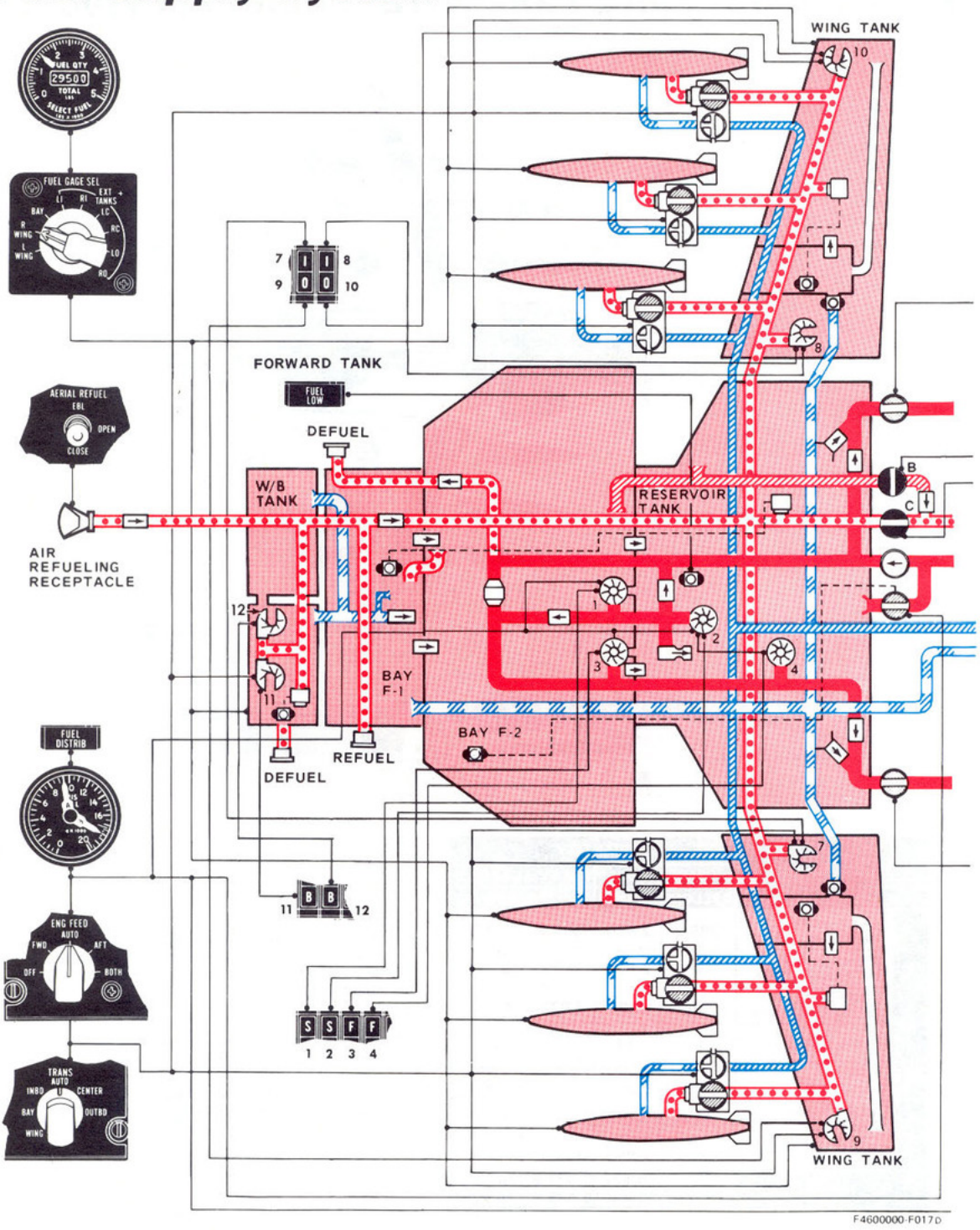


Figure 1-8. (Sheet 1)

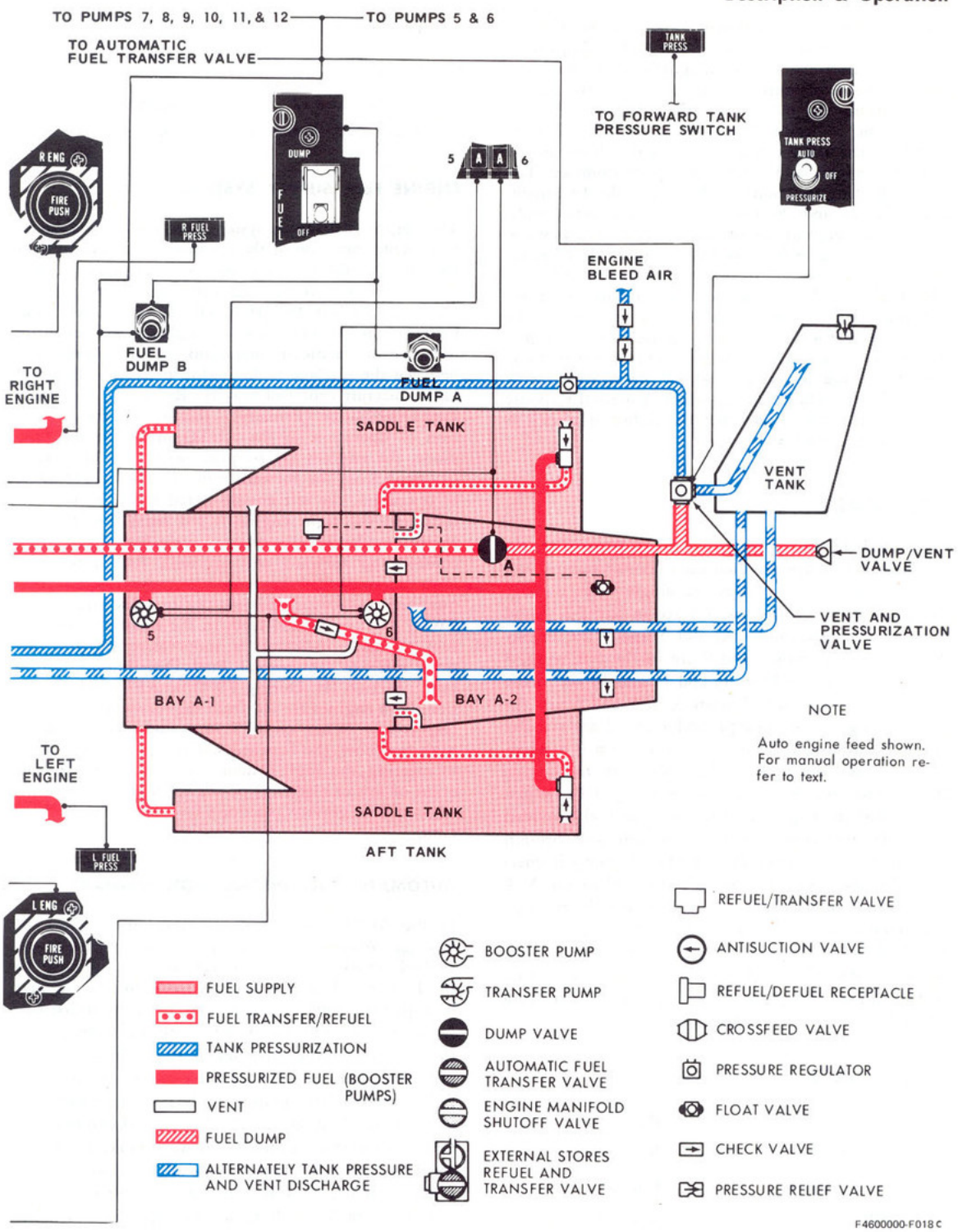


Figure 1-8. (Sheet 2)

Each consists of the following components: Capacitance sensors (tank units), located in each tank, provide a value proportional to fuel height/volume. An intermediate device measures the sum of tank units. A signal positions the indicator until a balance is established. The indicator consists of a servo, which responds to the amplifier; a gear train which drives the rebalance wiper; and a rebalance potentiometer. The four indicating functions are housed in the fuel quantity indicator and the total select fuel quantity indicator. The independence of the totalizer from possible errors occurring in the forward, aft, and select indicators is a design feature of this system and is made possible by use of dual sensor units in all tanks. Each indicating function has a capacitor that will always be covered with fuel until the respective tank is empty. An exception occurs in the select circuit in the aft tank. The fuel gage test circuit substitutes a fixed value which should indicate 2000 pounds. A normal response by the indicators verifies that the indicator circuit is functioning normally.

FUEL PUMPS.

There are 12 fuel pumps in the system. The six fuselage fuel pumps are dual inlet booster pumps, and the four wing fuel pumps are single inlet transfer pumps. Booster pumps 1 and 3 are in bay F-2, 2 and 4 are in the reservoir tank, and 5 and 6 are in bay A-1. Transfer pumps 7 and 9 are in the left wing, 8 and 10 are in the right wing and 11 and 12 are in the left weapons bay tank. Pumps 3, 4, 5 and 6 are the primary engine feed pumps, and 1 and 2 are standby engine feed pumps. Number 1 boost pump is a standby pump and operates continuously with the engine feed selector switch in any position except OFF. When not needed for engine fuel supply, the fuel provided by pump 1 is circulated into the reservoir tank through a pressure relief valve. The number 2 pump is energized by the pressure sensing switch when on AFT feed, BOTH or when on AUTO feed and the fuselage fuel quantity indicator indicates greater than approximately 8500 pounds differential between the forward and aft tanks. All pumps are controlled by 28 volt dc power and are energized by 115 volt ac power from the following electrical busses:

AC POWER.

Pumps	*Bus*
1, 3, 6, 9, 10 and 11	R. Main
4, 5, 7, 8 and 12	Essential
2 only	L. Main

DC POWER.

Pumps	*Bus*
4, 5, 7, 8 and 12	Essential
1, 2, 3, 6, 9, 10 and 11	Main

ENGINE FUEL SUPPLY SYSTEM.

The engine fuel supply system functions in five modes to provide fuel flow to the engines and a control over the fuel distribution between the fuselage tanks. The five modes as selected with the engine feed selector knob are: AUTO, BOTH, FWD (forward), AFT and OFF. In the AUTO mode the fuselage fuel quantity indicator automatically maintains the fuel distribution between the fuselage tanks within prescribed limits to assure aircraft center-of-gravity. Refer to "Automatic Fuel Distribution (Primary)," this section, for description of operation in the AUTO mode. In the OFF mode, the engines are supplied with fuel by gravity (suction) from the forward tank. In the BOTH mode of operation, the left engine is fed from the forward tank and the right engine is fed from the aft tank. In this mode there is no automatic fuel distribution control and forward and aft tank fuel differential must be controlled by monitoring the fuselage fuel quantity indicator and manually selecting either FWD, AFT, or BOTH feed. In the event the fuselage fuel quantity indicator is inoperative or malfunctions refer to "Abnormal Fuel Distribution/Indication," Section III. During FWD or AFT mode operation, both engines are fed from the forward or aft tank respectively. When on AFT feed, under conditions of high fuel flow, the forward standby pumps will assist in meeting the high demand on an aft tank. In the event of loss of electrical power to the fuel system the engines will gravity (suction) feed from the forward tank.

AUTOMATIC FUEL DISTRIBUTION (PRIMARY).

In the AUTO mode, fuel distribution between the fuselage tanks is automatically maintained within prescribed limits to assure aircraft center-of-gravity by the F (forward) and A (aft) pointers on the fuselage fuel quantity indicator. The engines are supplied fuel from the forward tank or both tanks depending upon the position of the switches in the indicator. If the differential in the forward tank is greater than approximately 8500 pounds, as is the case when the tanks are fully serviced and until all wing and external fuel has been transferred into the forward tank, the indicator will turn the aft tank pumps off, and feed both engines from the forward tank. As the differential between the tanks is decreased to approximately 8200

pounds, the indicator will detect the proper fuel distribution and feed the left engine from the forward tank and the right engine from the aft tank. When the differential between the tanks decreases to approximately 7900 pounds, the indicator will open an automatic transfer valve, to transfer fuel forward and regain the proper fuel distribution.

WARNING

The use of AUTO engine feed when the fuselage fuel quantity indicator is malfunctioning or inoperative could result in exceeding the center-of-gravity limits and loss of control of the aircraft.

Additionally, a mechanical float valve, located in the forward tank, allows the automatic transfer valve to open when the fuel level drops below 9000 pounds. If the engine feed selector knob is placed to a position that will cause the aft tank pumps to operate, fuel will be transferred forward to maintain the 9000 pound level until the aft tank is empty.

ALTERNATE FUEL DISTRIBUTION MONITORING SYSTEM.

An alternate fuel distribution monitoring system provides a means to indicate an abnormal aft fuel distribution between the forward and aft tanks independent of the fuel quantity indication system. The system includes four fuel level sensing units and a control unit. Two of the sensors are installed in the forward tank and two in the aft tank. One sensor in the forward tank is located at a fuel level of approximately 12,000 pounds, the other at about 9000 pounds. Likewise, the two in the aft tank are located at approximately 5300 pounds and 2500 pounds. When operating in OFF, FWD, AFT or BOTH and the forward tank fuel level drops below the 12,000 pound sensor, a signal will be provided to turn the fuel distribution caution lamp on if the 5300 pound sensor in the aft tank is covered. Likewise, when the fuel level in the forward tank decreases to a point below the 9000 pound level, the fuel distribution caution lamp will light if the aft tank level is above the 2500 pound sensor. A 12 second time delay is provided to eliminate fuel distribution signals due to fuel sloshing. When operating in AUTO, the alternate fuel distribution monitoring system is a backup to the normal system. If a malfunction occurs in the automatic fuel distribution control system that allows the actual fuel distribution to reach the above conditions, the alternate monitoring system will light the fuel distribution caution lamp and turn on the aft tank pumps if they were not operating.

FUEL TRANSFER.

In order to use the fuel in the external tanks, internal wing tanks, or weapons bay tanks, it must first be transferred to the fuselage tanks. Normally the tanks are emptied in the order of external, weapons bay, and then internal wing tanks. Fuel transfer is controlled by the transfer knob. The fuel level in the fuselage is maintained by float valves which open or close refuel valves to allow transfer into the fuselage tanks anytime they are not full. The refuel valves cannot be controlled from the cockpit. Transfer from any pair of external tanks, the weapons bay tank, or internal wing tanks can be manually selected. (Refer to "Fuel System Operation," this section.) When automatic transfer is selected, the transfer of fuel is automatically sequenced from the outboard, center, and inboard external tanks, the weapons bay tanks, and then the internal wing tanks, in that order.

Note

Both external tanks in a pair must be empty before auto transfer will commence from the next pair.

When the weapons bay tank runs dry, a one-minute delay will occur to assure complete scavenging of the tank before the wing tanks will transfer. Transfer from the weapons bay and internal wing tanks is effected by transfer pumps. Transfer from the external wing tank is accomplished by pressurizing the selected tanks with cooled engine compressor bleed air. When transferring from the weapons bay tank or wing tanks, the fuel pump low pressure indicator lamps should be used in conjunction with the fuel quantity indicator to determine when the particular tank is empty. The exact fuel quantity where the individual wing pump lamps light cannot be established accurately because it depends upon a large number of variables: attitude, wing sweep, roll angle, load factors, fuel temperature and density, weapon loading, wing deflection, etc. However, for level flight with the wing sweep forward, the outboard pumps normally run out of fuel and cause the outboard pump low pressure lamp to light before the inboard pump lights. If the wings are swept aft, the reverse is true.

FUEL PRESSURIZATION AND VENT SYSTEM.

Fuel system pressurization is provided to prevent loss of fuel from vaporization during flight. All tanks are pressurized by this system except the external tanks. Pressurization is provided by cooled engine compressor bleed air. The system functions in two modes: automatic and manual, as controlled by a fuel tank pressurization selector switch. In the automatic mode the tanks are pressurized when the landing gear handle is up. In this mode the system is automatically de-

pressurized when the refueling receptacle door is opened or when the gear handle is down. The tanks can also be pressurized by manually placing the fuel tank pressurization switch to PRESSURIZE in the event the automatic feature fails or if it is desired to pressurize the tanks with the air refueling receptacle door open or when the landing gear handle is down. The system maintains a pressure differential of 5 to 6 psi by means of a fuel tank vent and pressurization control valve. Should the pressure exceed 6 psi the valve will open and vent the excess air overboard through the dump/vent outlet at the rear of the fuselage. Except when transferring fuel from them, the external tanks are vented to atmospheric pressure through a vent port at the trailing edge of each pylon. Engine compressor bleed air is used to pressurize the external tanks for fuel transfer. The fuel transfer knob controls servo air to pressurize the external tanks. This is independent of the fuel tank pressurization system; however, engine bleed air must be available as selected by the air source selector.

FUEL DUMP SYSTEM.

The fuel dump system provides the capability of jettisoning fuel at a rate of 2300 pounds per minute. Fuel tank pressurization provides the force to jettison the fuel from the forward tank into the dump manifold and overboard through the vent/dump outlet at the aft end of the fuselage. This flow is controlled by motor operated dump valves A and B which receive power through circuit breakers located in the crew compartment. These two valves provide redundant shutoff capability for the dump system and valves are normally closed except during dumping operation. Dump valve B normally prevents fuel loss from the forward tank in the event of a broken refuel/dump line. Dump valve A normally prevents refuel and transfer flow from going overboard through the vent/dump outlet. In addition to dump valves A and B, dump valve C is provided. This valve is normally open but closes during dumping operation to prevent tank pressurization from flowing overboard through the dump line from the wings when the wing tanks are empty. Dump valve C receives power from dump A circuit breaker. The fuel dump system also utilizes the fuel transfer system to transfer fuel from the aft, bay and wing tanks to the forward tank. This is accomplished by relays which also receive power from dump B circuit breaker through the dump switch. When DUMP is selected, fuel immediately starts to transfer from the aft and wing tanks. Weapons bay tanks (if installed) will transfer before the wing tanks.

AIR REFUELING SYSTEM.

The air refueling system is capable of receiving fuel from a flying-boom type tanker aircraft. The system consists of a hydraulically actuated receptacle and slipway door, a signal amplifier, and the associated controls and indicators. Hydraulic pressure for operation of the receptacle and its latch mechanism is supplied by the utility hydraulic system. The receptacle is located on top of the fuselage, offset to the left and aft of the crew module. When the receptacle is extended, a mechanical linkage retracts the aft end of the slipway door into the fuselage, forming a slipway into the receptacle. When retracted, the slipway door is flush with the fuselage skin. The refueling receptacle is equipped with two lamps, one located on each side. As the receptacle extends, the lamps will light the receptacle and slipway area. During normal refueling operations, the refueling boom enters the receptacle and is automatically latched in place by a hydraulically actuated latching mechanism. When the boom is latched in place, fuel flows through the receptacle and the refuel/transfer fuel manifold lines into the fuel tanks at a rate of 5100 to 5800 pounds per minute. As the tanks are filled, float-operated valves automatically close the tank refueling valves, shutting off flow to the tanks. When the last refuel valve closes, an increase in the refuel line pressure is sensed by a pressure switch which automatically provides a signal to unlatch the boom from the receptacle. A disconnect signal can be manually initiated at any time during refueling by either receiver pilot or by the tanker boom operator. If a disconnect cannot be made by other methods, a brute force pullout can be safely accomplished. An emergency boom latch (EBL) capability is provided to latch the boom in place in the event the boom will not latch in the receptacle during normal operation. The EBL function also provides pneumatic power to extend the receptacle in the event utility hydraulic pressure is lost. Sufficient pneumatic pressure is available to operate the receptacle through two cycles (open and close) with four hookups during each cycle.

SINGLE-POINT REFUELING SYSTEM.

The single-point refueling system enables all aircraft fuel tanks to be pressure filled simultaneously from a single refueling receptacle. During ground refueling operations, fuel flows through the refueling receptacle and refueling manifold into the fuel tanks. As each tank fills, a float-operated valve automatically closes the refuel valve, stopping flow to the tank. The single-point refueling receptacle is located on the left side of the fuselage, forward of the engine air intake.

GRAVITY REFUELING.

Gravity refueling is accomplished through filler caps in the top of the wing and fuselage. There is one filler cap in each wing on the trailing edge near the fuselage. There are four filler caps in the fuselage: one each for F-1, F-2, A-1 and A-2 tanks. In addition to the filler cap located above the right saddle tank in bay A-1, a vent cap is provided above the left saddle tank. This cap must be removed to allow air to escape while the tank is being filled from the right side. To service the reservoir (trap) tank during gravity

refueling, the booster pumps in the foward tank must be operated at least two minutes. The external tanks can be refilled by transferring fuel from the wing tanks. The weapon bay tanks cannot be serviced by gravity refueling. For detailed gravity refueling, refer to "Strange Field," Section II.

FUEL SYSTEM CONTROLS AND INDICATORS.

Engine Feed Selector Knob.

The engine feed selector knob (6, figure 1-9), located on the fuel control panel, has five positions marked OFF, FWD, AUTO, AFT, and BOTH. When the knob is rotated to OFF, all fuel boost pumps are de-energized. Each knob position will energize the following pumps or place them on standby as indicated:

FWD —1, 2, 3 and 4 energized.

AUTO—Fwd & Aft fuselage fuel quantity indicator differential 8200 (approx.); 1, 3, 4, 5 & 6 energized; 2 on standby.

Fwd & Aft fuselage fuel quantity indicator differential approximately 7900 or less; 1, 3, 4, 5 & 6 energized, 2 on standby.

Fwd & Aft fuselage fuel quantity indicator differential approximately 8500 or greater; 1, 2, 3 & 4 energized.

AFT —1, 5 & 6 energized, 2 on standby.

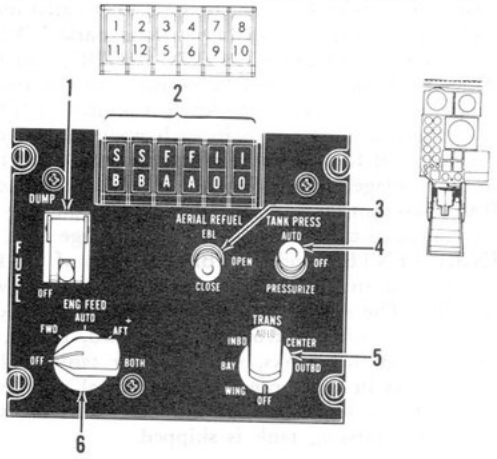
BOTH—1, 3, 4, 5 & 6 energized, 2 on standby.

When the knob is placed to FWD both engines are fed from the forward tank. When the knob is placed to the AUTO position the fuselage fuel quantity indicator, controlled by the F (forward) and A (aft) pointers on the instrument, automatically maintains the fuel distribution between the fuselage tanks within prescribed limits to assure aircraft center-of-gravity. The engines are supplied fuel from the forward tank or both tanks depending upon the position of the pointers. When the knob is placed to AFT, both engines are fed from the aft tank. However, when on AFT feed under conditions of high fuel flow, the forward standby pumps will assist in meeting the high demand on the aft tank. The standby pumps will also feed the engines from the forward tank should the aft tank run dry when on AFT feed.

WARNING

Do not use AFT feed when negative "g" operation is anticipated. Under negative "g" conditions only number 2 standby pump will be feeding the engines and engine flameout could result at MIL power or above.

Fuel Control Panel



1. Fuel Dump Switch.
2. Fuel Pump Low Pressure Indicator Lamps.
3. Air Refueling Switch.
4. Fuel Tank Pressurization Selector Switch.
5. Fuel Transfer Knob.
6. Engine Feed Selector Knob.

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Figure 1-9.

When the knob is placed to the BOTH position, the left engine is fed from the forward tank, and the right engine is fed from the aft tank. In this position, there is no automatic fuel distribution control, and forward and aft tank fuel differential must be controlled by monitoring the fuselage fuel quantity indicator and manually selecting either FWD, AFT, or BOTH feed.

Note

- The knob must be in either the AUTO or BOTH position to enable the functions of the air refueling switch.
- A failure of the auto transfer valve in the open position when the engine feed selector is in BOTH or AFT will cause all fuel in the aft tank to be transferred into the forward tank until full and may transfer into the wing tanks then through the vent tank and then overboard through the dump/vent valve. In the AUTO position, a failure will result in stopping and starting of the aft tank pumps to maintain proper distribution and will not result in overfilling of the forward tank.

Fuel Transfer Knob.

The fuel transfer knob (5, figure 1-9), located on the fuel control panel, has seven positions marked WING, BAY, INBD, AUTO, CENTER, OUTBD, and OFF. When the knob is in the OFF position, all fuel transfer functions are off. When the knob is rotated to WING, four transfer pumps, two in each wing tank, are energized, and fuel is transferred from the wing tanks to the fuselage tanks. When the knob is positioned to BAY, two transfer pumps in the left weapons bay tank are energized to transfer fuel to the fuselage tank. The INBD, CENTER, and OUTBD positions of the knob are for transferring fuel from external tanks when installed. The AUTO position automatically sequences the transfer of fuel from the outboard, center, and inboard external tanks, weapons bay tank, and the wing tanks in that order. If an external tank is not installed, the sequence of transfer remains the same, except the missing tank is skipped.

Note

- When all fuel has been transferred, as indicated by the fuel quantity indicator and fuel pump low pressure indicator lamps, the knob should be turned to OFF. This will prevent excessive fuel transfer pump wear, conserve electric power, and turn off the fuel pump indicator lamps.
- The fuel transfer knob must be in the OFF position to allow refueling the wing and weapons bay tanks. Common fuel manifold lines are utilized for both fuel transfer and refueling; therefore, if the transfer system is maintaining pressure in the manifold, the refueling valves in these tanks cannot open to allow refueling.

Fuel Dump Switch.

The fuel dump switch (1, figure 1-9), located on the fuel control panel, is marked DUMP and OFF. A guard holds the switch in the OFF position to prevent inadvertent actuation. The functions of this switch are explained under "Fuel Dump System," this section.

Fuel Tank Pressurization Selector Switch.

The fuel tank pressurization selector switch (4, figure 1-9), located on the fuel control panel, is a lever-lock switch marked AUTO, OFF, and PRESSURIZE. When the switch is positioned to AUTO, the fuel tanks are pressurized, except when the landing gear is down or the air refueling door is open. When the switch is placed to OFF, the pressurization airflow to the tanks is turned off and the tanks are vented. When the switch is placed to PRESSURIZE and pressurization air is available, fuel tank pressurization is maintained with the landing gear down or the refuel door open. Pressurization of the fuel tanks will not be provided when the air source selector knob is in OFF or EMER position.

Air Refueling Switch.

The air refueling switch (3, figure 1-9), located on the fuel control panel, is a lever-lock switch marked EBL, OPEN, and CLOSE. (Refer to "Air Refueling," this section, for operation of air refueling.)

Nose Wheel Steering/Air Refuel Buttons.

The nose wheel steering/air refuel buttons (3, figure 1-24), one located on each control stick grip are labeled NWS and A/R DISC. The air refueling functions of the buttons are activated when the aircraft is airborne to provide a means of manually disconnecting the air refueling boom. Depressing either button will interrupt power to the boom latching mechanism, causing it to unlatch. For a description of the NWS function of the buttons, refer to "Nose Wheel Steering System," this section.

Position Lights/Stores Refuel Battery Power Switch.

The position lights/stores refuel battery power switch (5, figure 1-27), located on the ground check panel, is marked POS LIGHTS, NORM, and STORES REFUEL. Placing the switch to the STORES REFUEL position will supply battery power directly from the battery bus to the external fuel tank valves for single-point ground refueling, regardless of the position of the battery switch. A float switch in the external tank will break the circuit and shut off the flow of fuel when the tank is full. This position will also prevent fuel transfer from any external tank regardless of fuel control panel switch positions. Placing the switch to NORM de-energizes the circuit. The switch is mechanically held in the NORM position when the ground check panel door is closed. For a description of the POS LIGHTS position of the switch, refer to "Lighting System," this section.

Fuel Quantity Indicator Test Button.

The fuel quantity indicator test button (37, figure 1-6), located on the left main instrument panel, is provided to self test the fuel quantity indicators. The button has the additional function of self-testing the alternate fuel distribution monitoring system. When the button is depressed, the fuel quantity indicators will simultaneously drive to the following indications:

1. Forward and aft tank pointers 2000 ($\pm$400).
2. Select tank pointer 2000 ($\pm$100).
3. Total fuel digital counter 2000 ($\pm$1250).

The fuel distribution caution lamp will light 10 to 12 seconds after the test button has been held depressed, to indicate that the alternate fuel distribution monitoring system is operative. When the button is released, the fuel quantity indicators will return to their original readings, and the fuel distribution caution lamp will go out in less than 15 seconds for all engine feed selector knob selections except AUTO. In the AUTO position, the lamp will remain on until the fuselage fuel quantity indicator pointers show a differential between the forward and aft tank greater than 7600 pounds. During the short period of time that the pointers show an abnormal fuel distribution (while they are returning to their original readings), the automatic fuel distribution control system will open the automatic fuel transfer valve and allow a small amount of fuel to be transferred from the aft to the forward tank.

Note

If fuel tank expansion space has been reduced due to fuel overfill or thermal expansion, some fuel venting may occur with the engine feed selector in AUTO, while the fuselage fuel quantity indicators are returning from the test indications. Fuel venting must cease prior to taxiing.

Fuel Quantity Indicator Selector Knob.

The fuel quantity indicator selector knob (34, figure 1-6), located on the left main instrument panel, has nine positions marked L WING, R WING, BAY (Weapons bay tank), LI (left inboard external tank), RI, LC (left center external tank), RC, LO (left outboard external tank), and RO. Placing the knob to the desired tank enables reading the amount of fuel remaining in that tank on the total/select fuel quantity indicator.

Note

Placing the fuel quantity indicator selector knob to an external or weapons bay tank position when there is no tank installed will result in a below zero indication.

Total/Select Fuel Quantity Indicator.

The total/select fuel quantity indicator (36, figure 1-6), located on the left main instrument panel, provides indications of total fuel in all tanks and fuel remaining in individual wing or external pylon tanks and the weapons bay tank. The indicator is graduated from 0 to 5 (times 1000 pounds) in increments of 100 pounds and has a pointer and a five-digit counter. The pointer will read the fuel remaining in the wing, bay, or external tank as selected by the fuel quantity indicator selector knob. The counter continuously reads the total fuel remaining in all tanks. Due to fuel quantity indicating system tolerance, it is possible to have a small amount of fuel remaining in the wing tanks when the select fuel indicator reads empty. The fuel pump low pressure indicator lamps for the wing transfer pumps provide the most positive indication that the wing tanks are completely empty. The select fuel quantity indicator circuit uses a single compensator sensor, located in the aft fuel tank, to correct for variations in fuel densities. If the aft tank is emptied while there is fuel in one or more of the wing or external tanks or the weapons bay tank, the uncovering of the compensator will cause the select gage indications to read erroneously high. The actual error will depend on the amount of fuel remaining in other tanks; however, a maximum error of 1000 pounds could exist.

Fuselage Fuel Quantity Indicator.

The fuselage fuel quantity indicator (38, figure 1-6), located on the left main instrument panel, provides indications of the amount of fuel in the forward and aft fuselage tanks. In addition, when operating in automatic engine feed, the indicator, through a series of internal switches controlled by the F (forward) and A (aft) pointers on the instrument, automatically maintains the fuel distribution between the fuselage tanks within prescribed limits to assure aircraft center-of-gravity. Refer to "Automatic Fuel Distribution (Primary)", this section, for a description of this function of the fuel quantity indicator.

WARNING

The use of AUTO engine feed when the fuselage fuel quantity indicator is malfunctioning or inoperative could result in exceeding the center-of-gravity limits and loss of control of the aircraft.

The indicator is graduated from 0 to 20 (times 1000 pounds) in 500 pound increments. The indicator has two pointers marked F (forward)and A (aft) for the forward and aft tanks. When operating in automatic engine feed, the A pointer will be maintained approximately 8200 pounds below the F pointer. In this position, the F pointer will be between two dot indices on the outer scale of the indicator. One dot indicates the point at which aft to forward transfer will occur, and the other indicates the point at which the aft tank pumps are shut off to maintain the 8200 pound differential. Two bar indices outboard of the dots indicate the point at which the fuel distribution caution lamp will light to indicate that the fuel differental between the forward and aft tanks is out of tolerance. The indices move with the A pointer and provide a ready reference of fuel differential when operating in manual engine feed.

Fuel Manifold Low Pressure Caution Lamps.

Two amber fuel manifold low pressure caution lamps (figure 1-36) are located on the main caution lamp panel. The letters R FUEL PRESS or L FUEL PRESS are visible when the respective lamp is lighted. The applicable lamp lights any time the fuel pressure in the right or left fuel manifold is less than 15.5 psia.

Fuel Low Caution Lamp.

The amber fuel low caution lamp (figure 1-36), located on the main caution panel, is controlled by a float switch in the reservoir tank. When the lamp is lighted, the letters FUEL LOW are visible indicating that the fuel level in the reservoir tank is less than 2300 (±235) pounds. With maximum gaging system tolerances, the "F" pointer may indicate as low as 1700 pounds or as high as 3000 pounds.

WARNING

Zero or negative "g" operation must be avoided whenever the fuel low caution lamp is lighted. The fuel system can supply fuel to the engines during zero or negative "g" operation for 10 seconds if the reservoir tank is initially full. There may be no negative "g" capability if the fuel low caution lamp is on, indicating that the reservoir tank is not full.

Note

- If boost pump 1 fails to provide fuel circulation through the reservoir from bay F-2, the small amount of air trapped in the top of the wing carry through box may expand, lowering the fuel level and causing the fuel low caution lamp to light. Engine fuel supply, other than for zero or negative "g", will not be jeopardized.
- During climb, with afterburners operating, the fuel low caution lamp may occasionally light. This is caused by the expansion of air at the top of the reservoir tank, allowing the fuel low float switch to actuate. This does not indicate a malfunction or constitute a hazardous condition for positive "g" flight. The lamp should go out after engine flow from the forward tank is reduced to less than 40,000 pounds per hour.

Fuel Pump Low Pressure Indicator Lamps.

Twelve fuel pump low pressure lamps (2, figure 1-9), one for each fuel pump, are located on the fuel control panel. When a fuel pump is energized, whether by automatic or manual tank selection, and the pump is not generating at least 3.5 (±0.5) psi, the lamp corresponding to the pump will light. The lamps are arranged in a double row, and the faces of the lamps are marked in pairs to correspond to each pump as follows:

S—Standby pumps 1 and 2

F—Forward fuselage tank pumps 3 and 4

I—Wing tank inboard transfer pumps 7 and 8

B—Weapons bay tank transfer pumps 11 and 12

A—Aft fuselage tank pumps 5 and 6

O—Wing tank outboard transfer pumps 9 and 10.

Fuel Tank Pressurization Caution Lamp.

The amber fuel tank pressurization lamp (figure 1-36), located on the main caution lamp panel, lights when fuel tank air pressure drops below approximately 3.5 (±0.5) psi during flight with the landing gear and the air refueling receptacle retracted. The lamp also lights any time the fuel tanks are pressurized and the landing gear or air refueling receptacle is extended. When the lamp lights, the letters TANK PRESS are visible.

Note

During descent, with the engines at idle, engine bleed air pressure is reduced, resulting in a lower airflow rate to the fuel tanks. At descent rates greater than 6000 feet per minute, it is possible for the fuel tank pressure to drop below 3.5 psi, causing the lamp to light. This is not an indication of a malfunction or hazardous condition.

Fuel Distribution Caution Lamp.

The amber fuel distribution caution lamp (figure 1-36), located on the main caution lamp panel, is provided to indicate an abnormal fuel distribution between the forward and aft tanks. The lamp has two signal input sources: (1) With the engine feed selector in AUTO, the automatic fuel distribution control system will light the lamp if the differential between the F and A pointers becomes less than 7600 pounds or greater than 10,000 pounds. (2) With the engine feed selector in any position, including OFF, the alternate fuel distribution monitoring system will light the lamp for aft abnormal fuel distribution conditions only. When the lamp is lighted, the letters FUEL DISTRIB are visible. For normal operation of the lamp during ground checkout, refer to "Fuel Quantity Indicator Test Button," this section.

Nose Wheel Steering/Air Refueling Indicator Lamp.

The nose wheel steering and air refueling indicator lamp (19, figure 1-6), located on the left main instrument panel, is labeled NWS A/R. For air refueling, the lamp indicates when the air refueling circuitry is set to receive the refueling boom. As the receptacle extends into place, the lamp will light. When the boom is latched in the receptacle, the lamp will go out. When the boom disconnects, the lamp will light again. When the air refueling switch is in the EBL position, the lamp indications are the same as when in normal operation, except the lamp will go out if the NWS A/R DISC button is depressed. The lamp will light when the NWS A/R DISC button is released if a disconnect has occurred. For a description of the NWS function of the lamp, refer to "Nose Wheel Steering System" this section.

Fuel Tank Pressure Gage.

The tank pressure gage, located adjacent to the single-point refueling receptacle, is provided to monitor tank pressure during ground refueling. The gage is graduated from 0 to 15 psi, in 0.5 psi increments.

FUEL SYSTEM OPERATION.

The fuel system can be operated in either an automatic or manual mode. The automatic mode is normally used since it requires a minimum amount of crew monitoring. Manual mode serves primarily as a backup in the event automatic operation malfunctions.

Normal (Automatic) Operation.

Normal system operation is accomplished with both the engine feed selector and fuel transfer knobs in AUTO. In this configuration, the following functions are automatically performed:

1. As fuselage fuel is used, fuel is transferred into the fuselage tanks from the external tanks, weapon bay tanks and internal wing tanks, in that sequence.
2. If all tanks were fully serviced at takeoff, both engines will be fed from the forward fuselage tank until external and internal wing tanks and weapon bay tank are expended and the fuel level in the forward tank is burned down to approximately 8200 pounds of fuel more than the aft tank. At this point the system will automatically switch to a split feed condition (feeding the right engine from the aft tank and the left engine from the forward tank) to maintain the differential thereby keeping the aircraft center-of-gravity within operational limits.
3. If the forward tank is burned down to approximately 7900 differential, the automatic transfer valve will open to allow fuel to be transferred from the aft tank to the forward tank. This will reestablish the 8200 pound differential.
4. If the aft tank is burned down to 8500 pounds differential, the aft tank pumps are turned off and both engines are fed from the forward tank until the 8200 pound differential is reestablished.

Manual Operation.

In the event that either automatic engine feed or automatic fuel transfer becomes inoperative, manual backup is available. During manual engine feed, the forward tank must be maintained at least 8000 pounds more than the aft tank by manual selection of either FWD or AFT feed to establish the proper differential. Once the differential has been established BOTH should be selected to maintain the differential. During manual transfer, the fuel transfer knob is positioned progressively to OUTBD, CENTER, INBD, BAY, and WING to empty the external wing, bay, and internal wing tanks, in that sequence. As each tank is selected for transfer, the corresponding fuel quantity indicator selector knob position should be selected to monitor the fuel level in the tank being emptied. It will be necessary to frequently switch the knob between the left and right external and internal wing tanks to monitor fuel transfer from these tanks.

Note

There should be a delay of approximately one minute after each tank(s) (external, bay or internal) indicates empty to insure all fuel is transferred before selecting the next tank.

Gravity Fuel Feed.

The engine driven fuel pumps will gravity (suction) feed the engines in the event of an electrical malfunction which prevents booster pump operation. In this condition, fuel will be used from the forward tank only. An anti-suction valve between the forward and aft tanks prevents suction feed from the aft tank to prevent the suction of air into the engine feed line in the event the aft tank is empty.

Fuel Dumping.

With the fuel dump switch in the OFF position, dump valves A and B are closed and C is open. When the switch is positioned to DUMP, the following events occur:

1. Dump valves A and B open and C closes.
2. The automatic transfer valve opens.
3. The fuel tanks pressurize (with the air source selector knob in any position other than OFF or EMER).

4. Booster pumps 5 and 6 in the aft tank transfer fuel to the forward tank (if in AUTO with more than 8,500 pounds differential, 6 only).
5. Transfer pumps 11 and 12 in the weapons bay tank, if installed, transfer fuel to the forward tank.
6. When the weapons bay tanks are empty, pumps 7, 8, 9 and 10 transfer fuel from the wing tanks to the forward tank.
7. Transfer from the external wing tanks, if selected, will continue at a reduced rate.

The fuel tanks will pressurize when the dump switch is in DUMP regardless of the position of the fuel tank pressurization selector switch, the landing gear handle, or the air refueling door, provided the air source selector knob is in a position other than OFF or EMER. Sufficient air is available to obtain the normal dump rate of 2300 pounds per minute when either engine RPM exceeds 85 percent. Tank pressurization forces fuel from the forward fuselage tank into the dump manifold and overboard through the vent/dump valve located on the aft centerbody. Fuel will be transferred from aft to forward tank at approximately 1750 pounds per minute if both aft tank pumps are operating or at 1100 pounds per minute if only one pump is operating. If external tank transfer is selected during a dumping operation, the rate of transfer from the external tanks is relatively slow; therefore, if required by operational considerations, these tanks should be jettisoned. All fuel except that in the reservoir tank (approximately 2552 pounds) can be dumped.

WARNING

To avoid the possibility of dumped fuel reentering the aircraft and causing a fire hazard, fuel dumping should be accomplished in coordinated flight at approximately 1 g with airspeeds no greater than 350 KIAS or mach 0.75, whichever is less.

Note

If dumping operation is necessary during afterburner operation, the fuel may ignite behind the aircraft. Other aircraft in the immediate vicinity should be advised to stay well clear during dumping operations.

To eliminate prolonged fuel dripping from the fuel dump outlet after dumping is discontinued, the fuel system may be momentarily depressurized to clear residual fuel from the fuel dump lines. (This will happen automatically when the landing gear is extended for landing.) During fuel dumping operations it should be noted that the automatic center-of-gravity control will not operate normally. If the engine feed selector knob is in AUTO during dumping, the No. 5 booster pump in the aft tank will shut off when the 8200 pound fuel differential is exceeded; the No. 6 booster pump will continue to run. Assuming that fuel is also being transferred from the wing tanks, the forward fuselage tank will remain full while the aft fuselage and wing tanks are emptying. This will cause the center-of-gravity to gradually shift forward and the 8200 pounds differential may not be maintained, causing the fuel distribution caution lamp to light. When the wing tanks are emptied, fuel from the forward fuselage tank will be dumped at a faster rate than that being transferred from the aft fuselage tank. This will cause the center-of-gravity to shift aft until the 8200 pound fuel differential is reestablished. From this point until the aft fuselage tank is empty, the No. 5 booster pump in the aft tank will cycle on and off to maintain the 8200 pound fuel differential. Although fuel is normally forced overboard by tank pressurization during fuel dumping, some dump capability still exists when tanks are not pressurized (air source selector knob in the OFF or EMER position). The fuel that is transferred to the forward tank will flow overboard, through the dump/vent outlet, at approximately the transfer rate, if the forward tank is nearly full. If the forward tank is not initially full, a portion of the fuel being transferred may partially fill the forward tank. After the tanks from which fuel is being transferred are empty, a portion of the fuel in the forward tank will flow overboard by gravity. The fuel flow rate from the forward tank will be approximately 500 pounds per minute when the tank is full, and will gradually decrease to zero. The quantity of fuel that can be dumped from the forward tank depends on the attitude of the aircraft, the higher the nose of the aircraft, the more fuel dumped. At level flight, the dump flow from the forward tank will cease at a fuel quantity in the forward tank of approximately 13,000 pounds. In order to obtain maximum fuel dump rate, without tank pressurization, the engine feed selector switch should be positioned to BOTH to prevent the automatic fuel distribution system from turning off the number 5 booster pump.

Air Refueling.

In order to open the receptacle the engine fuel feed selector must be selected to AUTO or BOTH, and the air refueling switch must be selected to OPEN or EBL. When the receptacle is open, the NWS A/R lamp will light to indicate the receptacle is open and the system is ready to accept the refueling probe.

Note

During ground operation when the air refueling door is open, the nose wheel steering/air refueling indicator lamp will light to indicate door position and nose wheel steering will function only while the NWS A/R DISC button is held depressed.

When the tanker/refueling probe is inserted into the receptacle, it is automatically latched in place and the NWS A/R lamp will go out to indicate when the latches have closed. Refueling is accomplished with the refuel switch selected to OPEN. In this position a disconnect signal can be provided from the tanker or from either crew member by use of the NWS A/R DISC button. In addition, when all tanks are full, fuel flow is interrupted by automatic closing of the refuel valves. A pressure switch will sense a rise in pressure in the refuel manifold and automatically provide a disconnect signal. Three seconds after the disconnect has occurred, the refuel system will automatically "reset" itself and light the NWS A/R lamp to indicate the system is again ready to receive a probe or that the receptacle should be closed. In addition, if the air refueling amplifier malfunctions, the EBL position on the air refuel switch will permit refueling. The procedure for EBL refueling is the same as the automatic procedure described above except a disconnect signal cannot be provided from any source other than the crew using the NWS A/R DISC button. When the button is depressed, the NWS A/R lamp will remain out until it is released. The NWS A/R lamp will light when the probe is out of the receptacle. If a malfunction of the hydraulic control solenoid has occurred that prevents opening of the receptacle, opening can then be accomplished by selecting EBL. This mode uses a separate solenoid to open the receptacle. Certain failures may require the air refuel circuit breaker to be reset after EBL is selected. In the event utility hydraulic power is not available, a back-up pneumatic system is provided. This system is energized by selecting EBL. After the receptacle has opened pneumatically and the boom is latched, the air refueling switch may be returned to the OPEN position to reinstate the automatic disconnect features of the system. Pneumatic power to operate the system will remain on until 5 seconds after the air refueling switch is placed to CLOSE. Sufficient pneumatic power is available to operate the receptacle through two cycles (open and close) with four hookups during each cycle. In the event of a failure that prevents a normal disconnect, a pressure relief valve is provided in the receptacle hydraulic latch actuator that will allow the probe to be pulled out by brute force if the boom tension exceeds 5000 pounds. Normal operating boom loads do not exceed 2300 pounds. Design loads for the receptacle and the tanker boom exceed 16,000 pounds. Lights are provided to illuminate the receptacle. The lights are turned on by a switch when the slipway door is open. The intensity of the lights is controlled, by the air refueling receptacle control knob. The knob should normally be at the mid-point of its control range when not in use. This assures that the lights are on at the beginning of night refueling but does not waste the service life of the bulbs during day refueling. Refer to T.O. 1-1C-1-1 for general air refueling procedures and T.O. 1-1C-1-21 for specific air refueling procedures for this aircraft.

ELECTRICAL POWER SUPPLY SYSTEM.

The electrical power supply system provides 115 volt, three-phase, 400-cycle ac power and 28 volt dc power. Two ac generator drive assemblies, one mounted on each engine, supply ac power. Two transformer rectifier units provide 28 volt dc power.

ALTERNATING CURRENT POWER SUPPLY SYSTEM.

AC power is supplied by two 60 kva generating systems. See figure 1-10. When operating, the systems are isolated from each other but are synchronized with respect to frequency. Each generator is driven by a constant-speed drive assembly which regulates generator frequency at 400 cycles per second. Voltage regulation and system protective functions are performed by generator control units. A frequency synchronizer controls the right constant-speed drive to keep its output speed in synchronization with the left constant-speed drive. There are three ac buses: a left main ac bus, a right main ac bus, and an essential ac bus. During normal operation, the right generator supplies power to the right main ac bus, and the left generator to the left main ac bus and the essential bus. Each main ac bus is normally isolated from the other. The system provides a bus-tie function automatically in the event a generator fails. If a fault or malfunction occurs, causing an undervoltage, overvoltage, underfrequency, or overfrequency condition to exist on a generating system, the associated ac generator control unit removes the generator from the bus. If a malfunction is corrected, the generator may be reconnected to the bus by properly positioning the generator switches. If a malfunction causing an excessive amount of heat occurs in the constant-speed drive (CSD) unit, a thermal device in the unit automatically decouples the drive from the engine. Once decoupled, the drive cannot be recoupled during flight. An emergency generator with 10 kva capacity provides electrical power to the essential ac bus in the event of failure of both main ac generators. The emergency generator is driven by a hydraulic motor which receives power from the utility hydraulic system. During engine cartridge start, the emergency generator will not operate with the engine ground start switch in the CARTRIDGE position.

AC Electrical Power Supply System

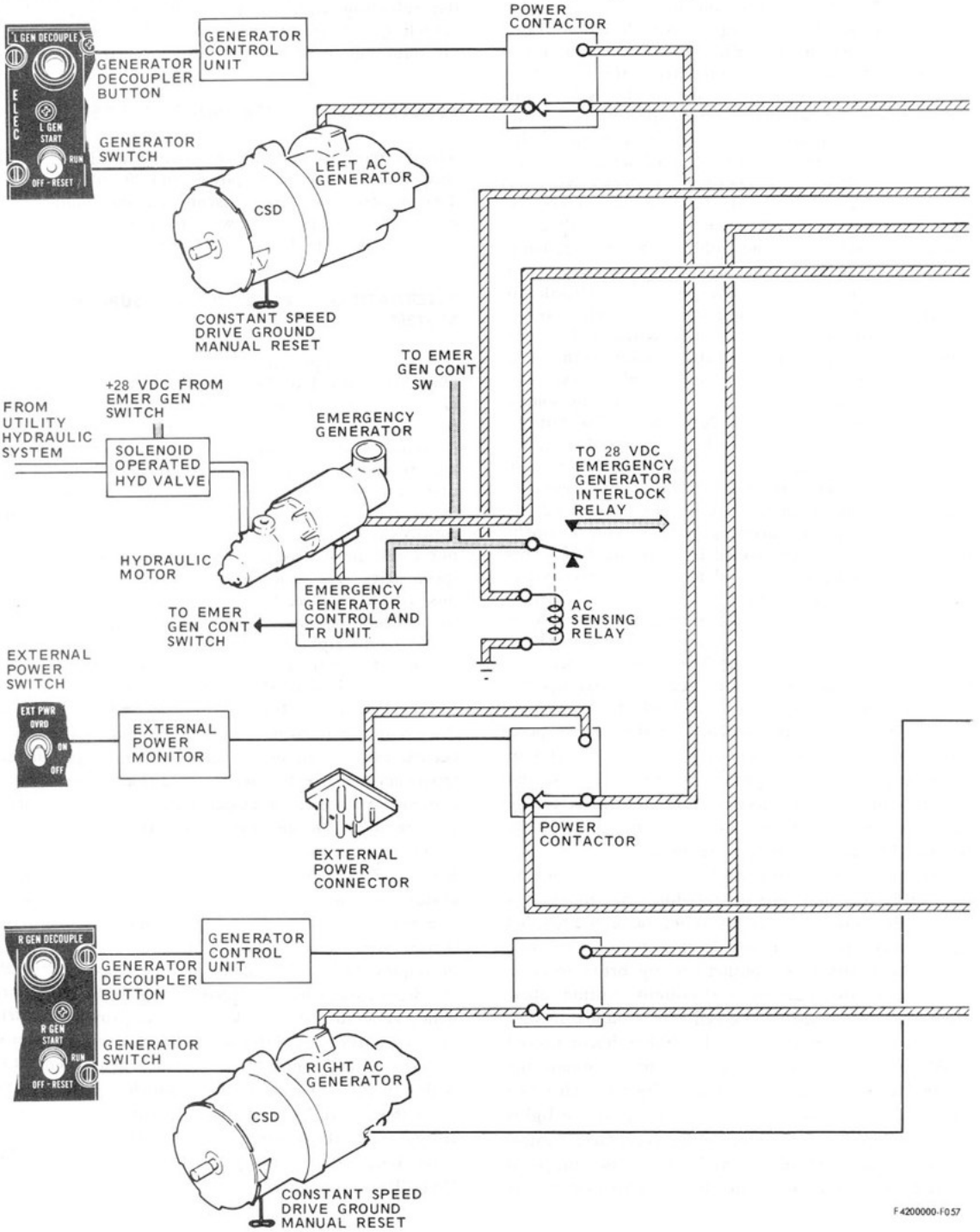


Figure 1-10. (Sheet 1)

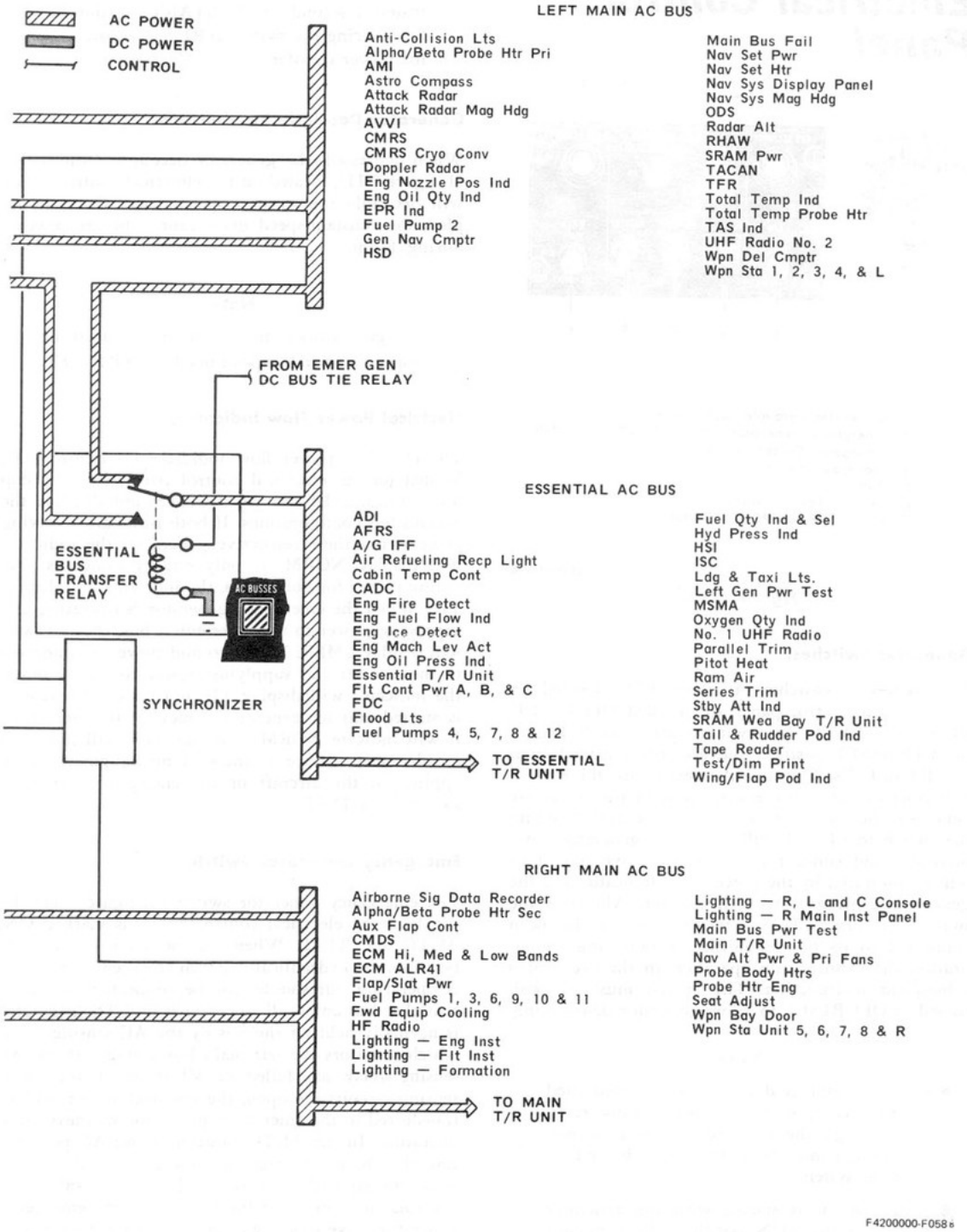
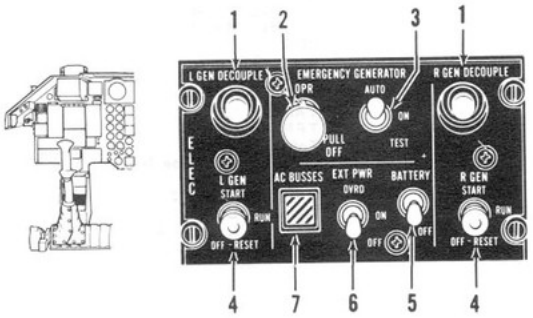


Figure 1-10. (Sheet 2)

Electrical Control Panel



1. Generator Decouple Pushbuttons.
2. Emergency Generator Indicator/Cutoff Pushbutton.
3. Emergency Generator Switch.
4. Generator Switches.
5. Battery Switch.
6. External Power Switch.
7. Electrical Power Flow Indicator.

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Figure 1-11.

Generator Switches.

Two generator switches (4, figure 1-11), located on the electrical control panel are marked OFF-RESET, RUN and START. The switches are lever locked in the OFF-RESET position and are spring-loaded from START to RUN. Placing either switch to OFF-RESET will reset the generator-control unit of the respective generator, but will not excite the generator. Holding the switch to START will excite the generator automatically and connect it to its respective bus. This will be indicated by the power flow indicator and the generator caution lamp will go out. Allowing the switch to return to RUN, after the generator has been connected to its bus, establishes a switching configuration that assures safe operation in the event of a subsequent malfunction. The switch must be positioned to OFF-RESET to allow generator de-coupling.

Note

- If a generator is deexcited while connected to the bus, it will not automatically reset, even though the fault condition is cleared. The switch must be held to OFF-RESET to reset the system.
- If the engine is started with the generator switch in the RUN position, the generator will be excited but not connected to the bus until the switch is placed to START.
- There should be a definite pause (approximately 1 second)in the START position before placing the switch to RUN to allow time for power transfer.

Generator Decouple Pushbuttons.

When depressed, the generator decouple pushbuttons (1, figure 1-11), located on the electrical control panel, will decouple the constant-speed drive. Once decoupled, the constant-speed drive cannot be reconnected during flight.

Note

The generators cannot be decoupled until the respective switch is positioned to OFF-RESET.

Electrical Power Flow Indicator.

The electrical power flow indicator (7, figure 1-11), located on the electrical control panel, is a flip-flop type indicator labeled AC BUSSES and displays the various bus configurations. If both buses are receiving power from their respective generator, the indicator will display NORM. If only one generator is providing power for both buses, the indicator will display TIE. When the emergency generator is operating and supplying power to the ac essential bus, the indicator will display EMER. When ground power is connected to the aircraft and supplying power to the ac buses, the indicator will display TIE until the right engine is started and its generator comes on the line; then it will indicate NORM. The indicator will display a crosshatched surface if there is no ac power being applied to the aircraft or the emergency generator switch is in TEST.

Emergency Generator Switch.

The emergency generator switch (3, figure 1-11), located on the electrical control panel, is marked ON, AUTO and TEST. When the switch is in the ON position, the hydraulically driven emergency generator is operating but should not be connected to the essential ac bus unless all ac power is lost. The generator is normally held off the bus by the AC sensing relay, which monitors the left main bus voltage. If the AC sensing relay has failed or all three of the phase sensing circuits are open, the essential ac bus will be transferred to the emergency generator whenever it is operating. In the AUTO position, if all AC power is lost, the DC bus voltage (main and essential) will be zero; consequently, the solenoid operated valve that controls the flow of hydraulic fluid to the emergency generator drive motor will open. The emergency generator will then automatically begin to operate. When its voltage and frequency are within prescribed limits,

the essential bus transfer relay will be energized, thereby transferring the essential ac bus to the emergency generator. When the switch is in the TEST position, the emergency generator is operating and the emergency generator indicator lamp lights, but the emergency generator is not connected to the essential ac bus if the left main ac bus is powered. If the ac sensing relay has failed, the emergency generator will be connected to the essential ac bus when TEST is selected.

WARNING

- Failure of the ac sensing relay will connect the emergency generator to the essential bus when TEST or ON is selected. This failure will be indicated if the flow indicator reads EMER when the emergency generator switch is placed to ON while an engine-driven generator is operative.
- In flight, avoid selecting the ON position of the emergency generator switch. To do so places an unnecessary drain on the hydraulic system and does not prevent a fast erection cycle of the AFRS if emergency generator power is required.

The TEST position also opens the dc bus-tie contactor to provide a method of checking operation of the two transformer rectifier (T/R) units. If the main and essential dc buses remain energized, both T/R units are operating and the electrical power flow indicator will display a crosshatch. When the emergency generator switch is placed in TEST, the essential T/R unit has failed if the landing gear position lamps or the angle-of-attack indexers are out. If the main T/R unit has failed, the ODS reticle will not be lighted and the instrument test will be inoperative.

Emergency Generator Indicator/Cutoff Pushbutton.

The emergency generator indicator/cutoff pushbutton (2, figure 1-11), located on the electrical control panel, is marked OPR (operate) and PULL OFF. When the button is depressed, the emergency generator will come on the line and supply power to the aircraft systems whenever both engine-driven generators fail. Should this occur, a green indicator lamp in the button will light. When the emergency generator is supplying power, pulling the button out will de-excite the emergency generator and shut off its power output.

External Power Switch.

The external power switch (6, figure 1-11), located on the electrical control panel, is marked OFF, ON and OVRD. In the OFF position, external power cannot be supplied to the aircraft ac buses. In the ON position with neither engine operating, external power supplies all aircraft power. With the left engine operating, the left main ac generator will supply total aircraft electrical load regardless of whether external power is connected or disconnected. With only the right engine operating, the right main ac generator feeds the right main ac bus and external power feeds the left main ac and essential buses. A power monitor measures external power voltage, frequency, and phase sequence. Should any one of these parameters be out of tolerance, the monitor prevents application of external power. When the switch is in the OVRD position, the monitor circuit is bypassed, allowing external power to be applied to the aircraft buses, even though the voltage, frequency, or (in some cases) phase may be out of tolerance.

CAUTION

The OVRD position should not be used unless required by an emergency.

Generator Caution Lamps.

Two amber generator caution lamps (figure 1-36) are located on the main caution lamp panel. Either lamp lights when the respective generator is disconnected from the ac bus. When lighted, the letters L GEN are visible in the left lamp and R GEN in the right lamp.

DIRECT CURRENT POWER SUPPLY SYSTEM.

DC electrical power is provided by two 28 volt dc transformer-rectifier (T/R) units and a 24 volt battery. See figure 1-12. There are four dc buses: a main dc bus, an essential dc bus, a battery bus, and an engine start bus. The essential dc bus is divided into two separate buses, one located in the forward equipment bay and one located in the crew module on the aft bulkhead (figure 1-12). The essential buses are electrically connected. During normal operation, the main dc bus section receives power from the main T/R unit, which is connected to the right main ac bus. The essential dc bus, engine start bus and crew compartment power distribution panel receive power from the essential T/R unit, which is connected to the essential ac bus.

DC Electrical Power Supply System

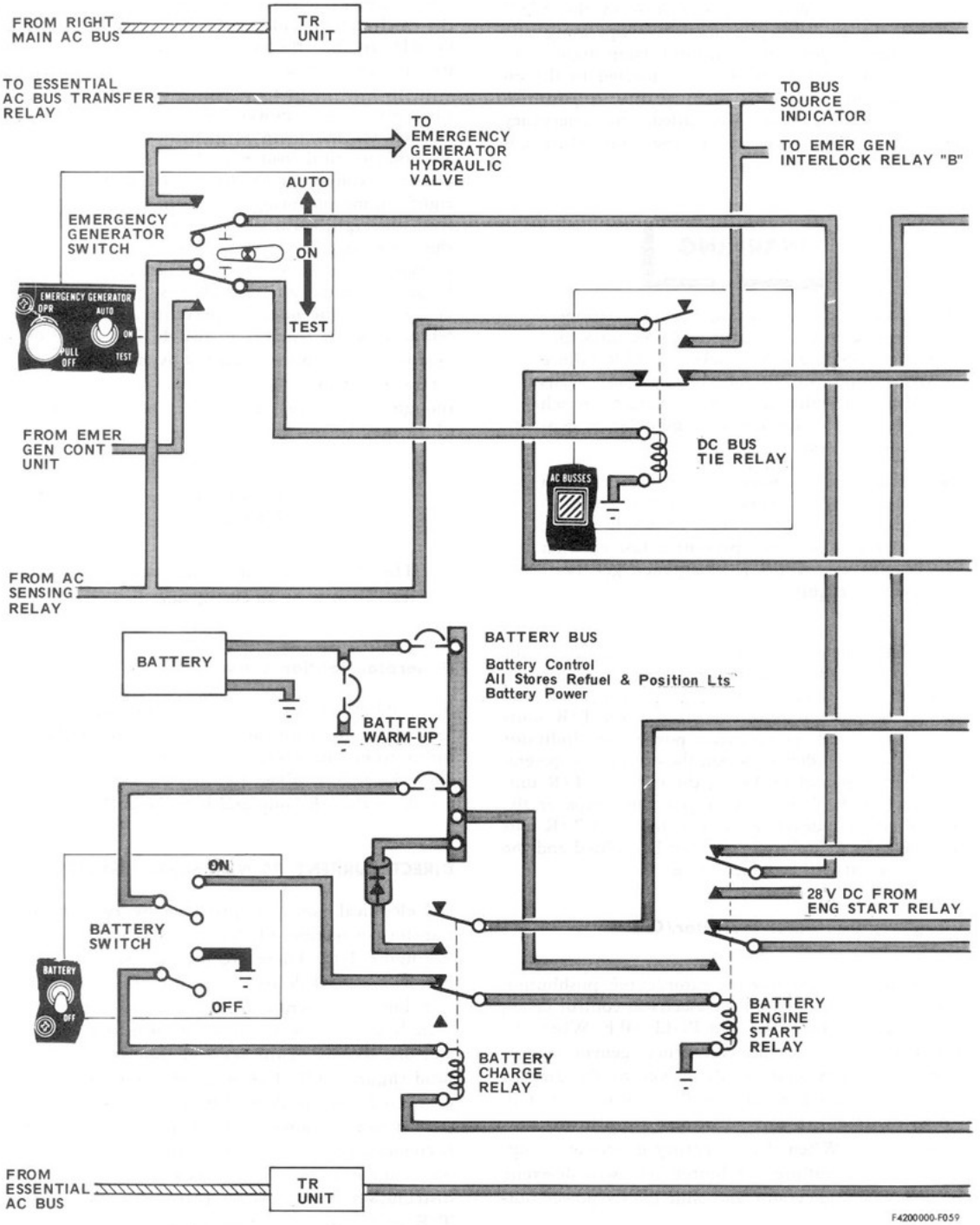


Figure 1-12. (Sheet 1)

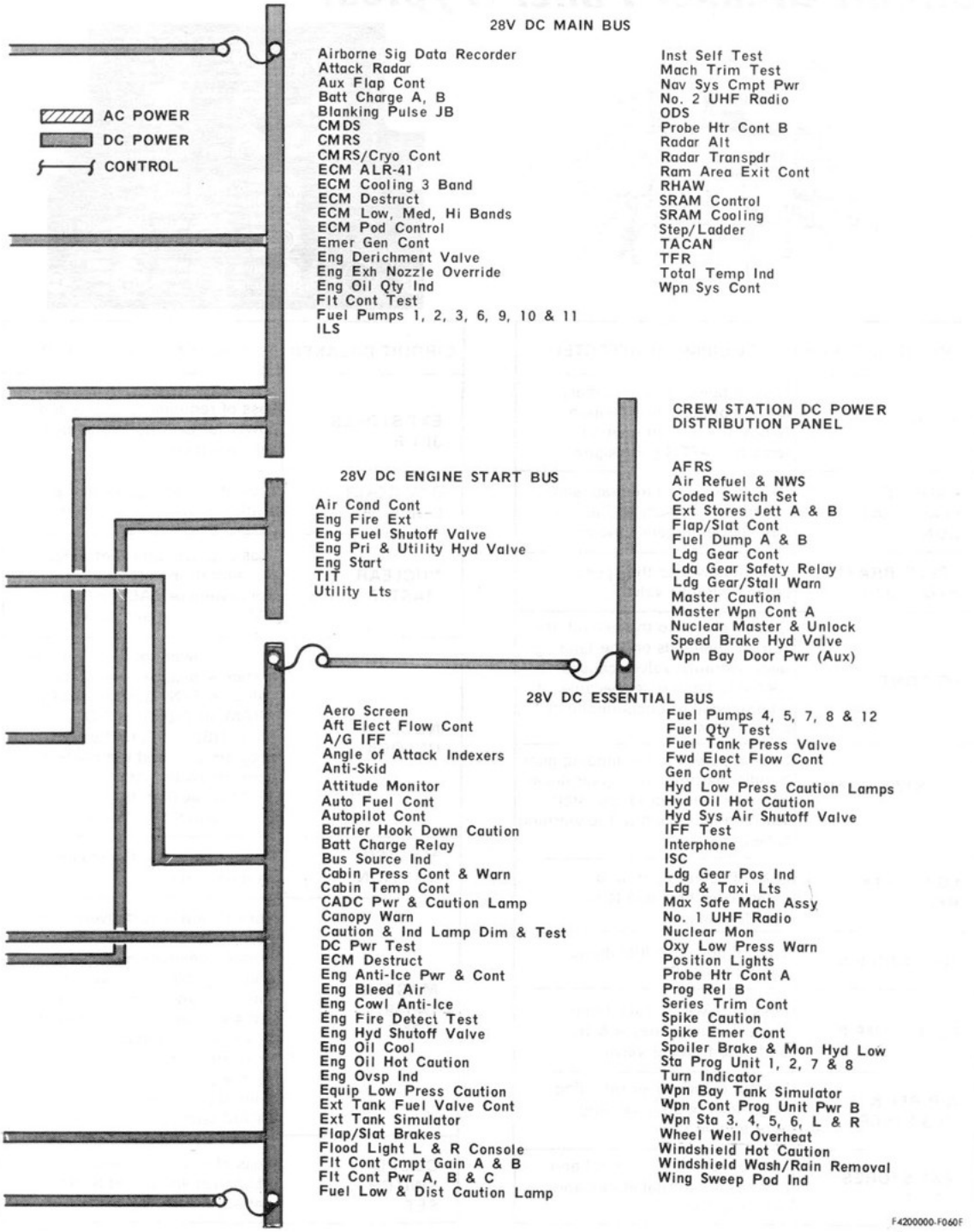
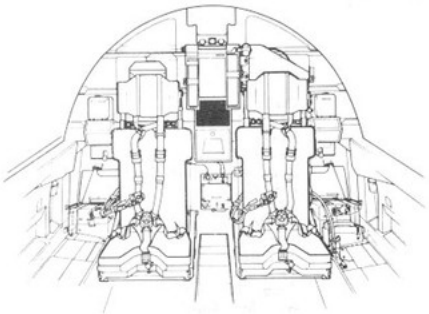
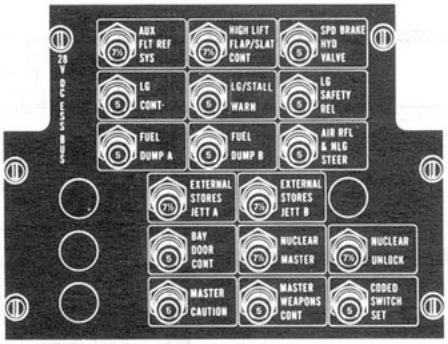


Figure 1-12. (Sheet 2)

Circuit Breaker Panel (Typical)





CIRCUIT BREAKER	EQUIPMENT AFFECTED
AFRS	Loss of power to the primary att and primary hdg caution lamps, the aux att caution lamp and AFRS good signal.
HIGH LIFT FLAP/SLAT CONT	Loss of power to the flap/slat asymmetry system and to the flap/slat emergency motor.
SPEED BRAKE HYD VALVE	Loss of power to the speed brake hydraulic valve.
LG CONT	Loss of power to the extend and retract solenoids on the landing gear hydraulic valve; causes SRAM to latch reset because of commanded or uncommanded power removal.
LG/STALL WARN	Loss of power to the landing gear handle warning lamp, gear down and lock indicator lamps, stall warning lamp, and to the warning tone generator.
LG SAFETY REL	Loss of power to ground safety switches, and to gear handle release.
FUEL DUMP A	Loss of power to fuel dump valves A and C.
FUEL DUMP B	Loss of power to fuel dump valve B, dump relay A & B, and auto transfer valve.
AIR RFL & NLG STEER	Loss of power to air refueling receptacle or nose landing gear steering.
EXT STORES JET A	Loss of redundant control and power for external stores and pylons jettison.

CIRCUIT BREAKER	EQUIPMENT AFFECTED
EXT STORES JET B	Loss of redundant control and power for external stores and pylons jettison.
BAY DOOR CONT	Loss of control power for operation of weapon bay door.
NUCLEAR MASTER	Loss of power and control for the aircraft monitor and control system (AMAC) and for SAF prearming and safing.
NUCLEAR UNLOCK	Loss of power source for nuclear station select, nuclear station select to B/N, station select to SRAM, and (prior to T.O. 1F-111(B)A-659) to the central program unit and the nuclear consent switch; after T.O. 1F-111(B)A-659, to the conventional SPU.
MASTER CAUTION	Loss of power to the master caution lamp.
MASTER WPNS CONT	Loss of power to delivery mode selector knob, weapon release button, weapons mode selector knob, master power switch missile launch and selective jettison capability, and essential bus redundant power to central program unit, release program unit and landing gear up input to central program unit causes SRAM latch reset.
CODED SWITCH SET	Loss of nuclear weapon enabling capability and SRAM safe-arm-fuse capability.

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Figure 1-13.

A bus-tie contactor connects the essential dc bus to the main dc bus during normal operation. Normally the outputs of the two T/R units supply the total dc load in parallel.

Battery Switch.

The battery switch (5, figure 1-11) is located on the electrical control panel. The two-position switch is marked OFF and BATTERY. Positioning the switch to BATTERY connects the engine start bus to the battery. If the essential dc bus is energized, the battery is connected to the main dc bus through the battery charger circuit and the engine start bus is connected to the essential dc bus. When the battery switch is positioned to OFF, the engine start bus is connected to the essential dc bus, and the battery charger circuit is disconnected from the main dc bus.

HYDRAULIC POWER SUPPLY SYSTEM.

Hydraulic power is supplied by two independent, parallel hydraulic systems designated as the primary and utility systems (figure 1-14). Both systems operate simultaneously to supply hydraulic power for the flight controls and wing sweep. If one system should fail, the other is capable of supplying sufficient reduced power for wing sweep and flight control operation. The primary hydraulic system supplies hydraulic power solely for the operation of the wing sweep and flight control systems. In addition to supplying wing sweep and flight control hydraulic power, the utility system supplies power for operation of:

- Nose wheel steering
- Landing gear
- Wheel brakes
- Speed brake
- Flaps/slats
- Rotating glove
- Weapons bay doors
- Tail bumper
- Emergency electrical generator
- Air inlet (spike) control
- Air refueling system
- Rudder authority

Hydraulic pressure for each system is supplied by two engine-driven pumps. To assure hydraulic pressure in the event of single engine failure, each engine drives one utility and one primary pump. Pressurized accumulators are installed in the system to supplement engine-driven pump delivery during heavy demand. Each system has a piston-type reservoir for hydraulic fluid storage that also acts as a surge damper for return line pressures. These reservoirs are pressurized with nitrogen to insure critical pump inlet pressure for all operating conditions. Hydraulic pressure of each system is displayed on the left main instrument panel. Low pressure caution lamps for each of the four pumps are displayed on the caution light panel. An isolation unit incorporated into the system reserves utility pressure for flight control and wing sweep only, in the event of primary system failure. It also performs a second function of isolating hydraulic pressure after takeoff from those systems normally only associated with takeoff and landing.

HYDRAULIC PUMPS.

Four variable delivery pumps are used: one primary pump and one utility pump are driven by each engine. (See figure 1-14.)

HYDRAULIC ACCUMULATORS.

Eight accumulators are provided, five in the utility hydraulic system and three in the primary hydraulic system. Each system has two accumulators for the horizontal stabilizer and one for the damper servos. The utility system has two accumulators for the wheel brake system. See figure 1-83 for servicing data.

HYDRAULIC FLUID RESERVOIRS.

Both utility and primary hydraulic reservoirs are floating piston, pneumatic-oil separated types, using pneumatic pressure on one side of the piston to maintain hydraulic pressure on the other. Pneumatic pressure is supplied from pneumatic storage reservoirs located on the forward end of each hydraulic reservoir and, as an alternate source, from the engine bleed air system. A pressure-operated hydraulic relief valve prevents overpressurization by venting excess fluid overboard. Steady-state fluid flow is passed through the reservoir to maintain reservoir warmth and to remove air from the fluid. During high flow rates, the fluid is bypassed around the reservoir and cooler loop directly to the pumps by means of the suction bypass valve. A bypass-type filter is located up-stream of the reservoir. The reservoir also acts as a surge damper for return line impulse pressures. See figure 1-83 for servicing data.

HYDRAULIC COOLING SYSTEM.

Cooling is provided by an air-to-hydraulic heat exchanger and a fuel-hydraulic heat exchanger in each hydraulic system. The controls are arranged so that the cooling medium is air only at low speeds, fuel and air at intermediate speeds, and fuel only at high speeds.

Hydraulic Power Supply System (Utility)

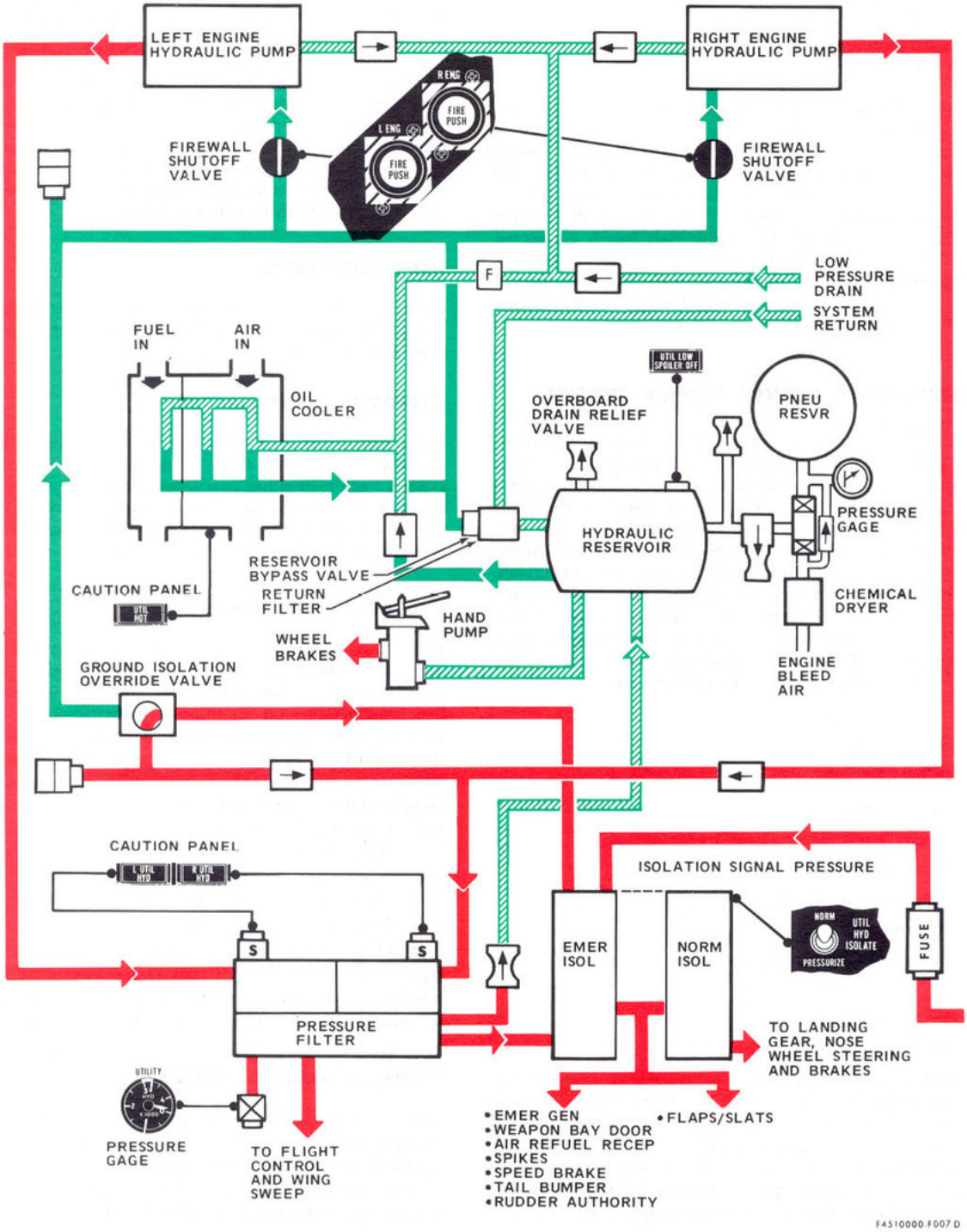


Figure 1-14. (Sheet 1)

Hydraulic Power Supply System (Primary)

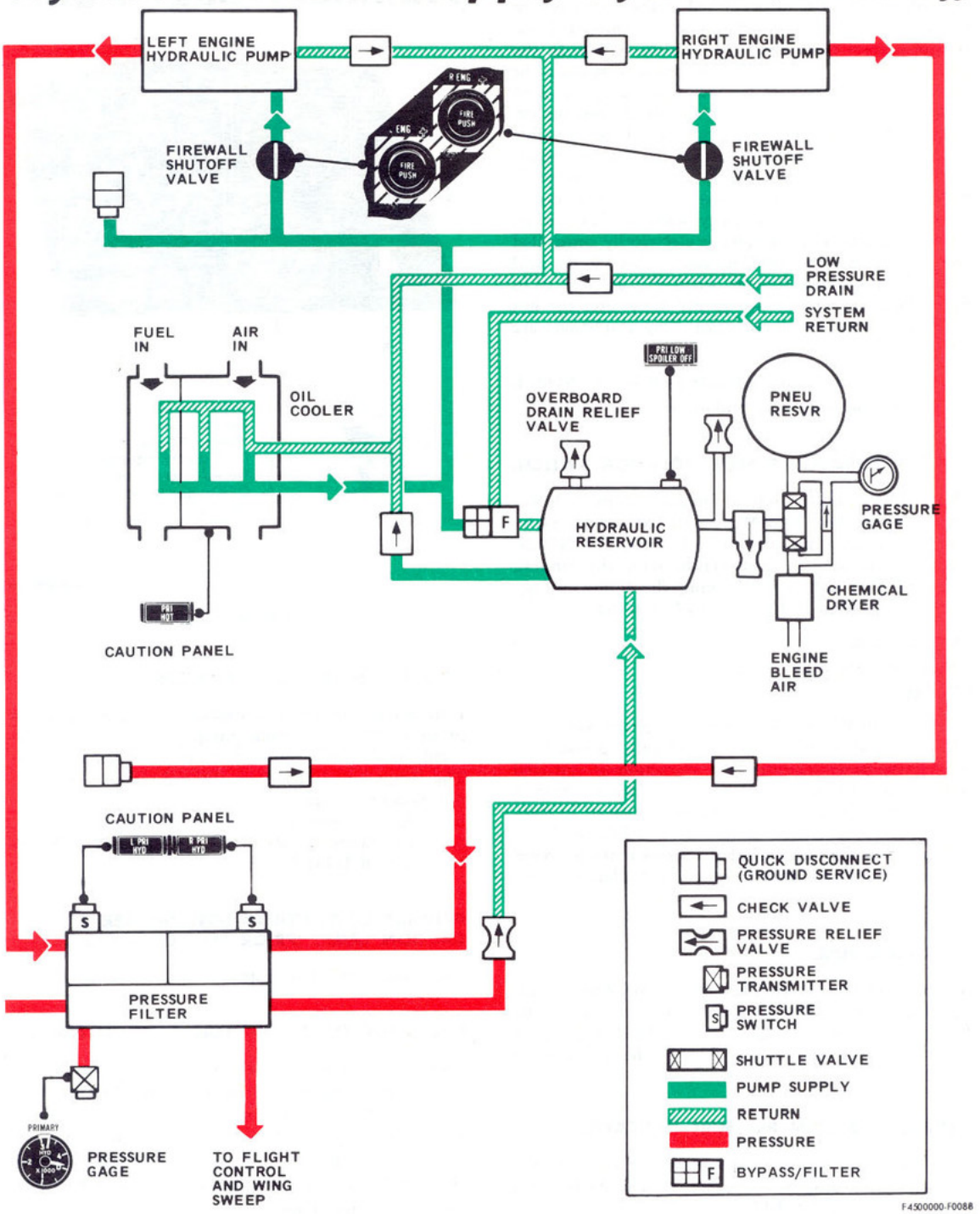


Figure 1-14. (Sheet 2)

HYDRAULIC ISOLATION VALVE.

An isolation valve is incorporated in the utility system to automatically provide emergency and normal isolation of certain functions of the utility system. In the event of loss of pressure in the primary system, the valve will automatically go into emergency isolation at approximately 500 psi, and cut off all systems except flight controls and wing sweep. It is not possible to manually override the emergency isolation feature. If the primary system pressure increases to approximately 1200 psi, the utility system will automatically be brought out of the emergency isolation. The normal isolation function of the valve is electrically controlled and will isolate the landing gear, wheel brake and nose wheel steering systems when the aircraft is in flight. Isolation takes place immediately after the last of all the following three controlling conditions are satisfied:

1. The utility hydraulic isolation switch is in NORM.
2. The landing gear is up and locked.

UTILITY HYDRAULIC SYSTEM ISOLATION SWITCH.

The utility hydraulic system isolation switch (7, figure 1-16), marked NORM and PRESSURIZE, is located on the landing gear control panel. The NORM position functions in conjunction with the landing gear and flaps and slats, allowing the following systems to be isolated from the utility system:

- Landing gear
- Nosewheel steering
- Brakes

During normal operation, positioning the switch to PRESSURIZE supplies utility hydraulic pressure to these systems. If landing gear retraction is initiated with the speed brake switch in any position other than IN, the speed brake may remain fully open or in trail. Placing the hydraulic isolation switch to PRESSURIZE will override the function of the speed brake switches and cause the speed brake to function as a landing gear door and retract.

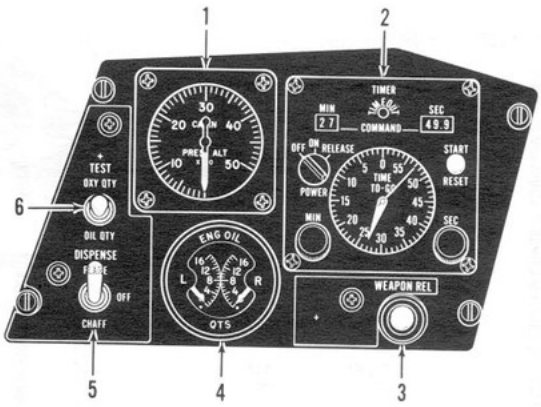
HYDRAULIC FUSE.

A hydraulic fuse is installed on the primary system (sensing) pressure line to system isolation valve. The fuse prevents loss of primary system fluid in event of line rupture by shutting off flow at the downstream port of the fuse.

HYDRAULIC SYSTEM PRESSURE INDICATORS.

Two 0-4000 psi pressure indicators, one each for the primary and utility systems, are located on the left main instrument panel (33, figure 1-6).

Auxiliary Gage Panel





1. Cabin Altitude Indicator.
2. Bombing Timer.
3. Weapon Release Button.
4. Oil Quantity Indicator.
5. Flares/Chaff Dispense Switch.
6. Oil/Oxygen Quantity Indicator Test Switch.

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Figure 1-15.

LOW PRESSURE CAUTION LAMPS.

Four amber low pressure caution lamps, energized by pressure switches in each pump pressure line, are located on the main caution lamp panel (figure 1-36). The lamps will light when the individual pump output pressure drops to approximately 400 psi. When lighted, the following letters will be visible in the respective lamp; L PRI HYD, L UTIL HYD, R PRI HYD, and R UTIL HYD.

PRIMARY LOW, UTILITY LOW SPOILER OFF CAUTION LAMP. (PRIOR TO T.O. 1F-111-1151)

These lamps are deactivated.

HYDRAULIC FLUID OVERHEAT CAUTION LAMPS.

Two hydraulic fluid overheat caution lamps, one for each system, are located on the main caution lamp panel (figure 1-36). A lamp lights when the hydraulic fluid temperature of the associated system exceeds 240 $\pm$ 10 degrees F (115 $\pm$ 6 degrees C). When lighted, the PRI HOT and UTIL HOT letters will be visible in the respective lamp.

PNEUMATIC POWER SUPPLY SYSTEMS.

There are eight independent pneumatic power supply systems as follows:

1. Hydraulic reservoir pressurization system
2. Landing gear emergency extend system
3. Inflight refueling emergency system
4. Spike emergency extend system
5. Overwing fairing system
6. Wheel brakes parking/emergency system
7. Arresting hook extend system
8. Tail bumper system.

Two pneumatic reservoirs, one for each hydraulic system reservoir, provide pneumatic pressure for hydraulic system operation. Pressure for emergency extension of the landing gear is provided by a pneumatic reservoir located in the main landing gear wheel well. Pneumatic pressure for emergency operation of the air refueling doors, receptacle, and boom latch is provided by a pneumatic reservoir located in the main landing gear wheel well. Each spike is provided with a separate pneumatic reservoir located in the main landing gear wheel well. The overwing fairing actuators are provided pneumatic pressure to hold the fairing firmly against the aircraft structure regardless of the wing sweep. Each wheel brake circuit is provided with a pneumatic air bottle for parking and emergency braking. The arresting hook and tail bumper pneumatic systems are located in the tail cone between the engines. For a functional description of each pneumatic system, refer to the associated system description, this section. For servicing information on the pneumatic systems, see figure 1-83.

LANDING GEAR SYSTEM.

The landing gear is tricycle-type, forward retracting, and hydraulically operated. The main landing gear consists of a single common trunnion with two wheels, insuring symmetrical main landing gear operation. Thermal pressure relief plugs in the main landing gear wheels relieve tire pressure in the event of maximum performance braking. The nose landing gear has two dual-mounted wheels. The landing gear system is normally powered by the utility hydraulic system. A pneumatic system is provided as an alternate (emergency) means of extending the gear in the event the normal system fails.

MAIN GEAR.

Three hydraulic actuators are provided for operation of the main landing gear. One actuator retracts the main landing gear. Two actuators, one for an uplock and one for a downlock, are provided to lock the landing gear in the retracted or extended position. There are two main landing gear doors. The aft door is mechanically linked to the main landing gear and opens and closes with movement of the gear. The forward door, which also serves as the speed brake, is hydraulically operated. A mechanical connection between the main landing gear and the speed brake selector valve causes the main landing gear door to open and close in the proper sequence during landing gear operation. Ground safety switches, located on the lateral trunnion beam, prevent normal gear retraction while the aircraft is on the ground. See figure 1-17 for location of the ground safety switches.

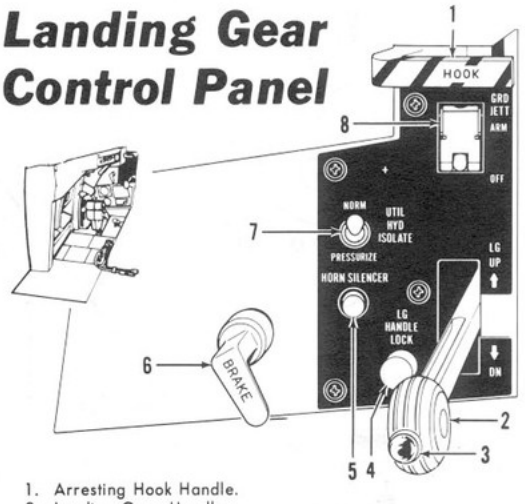
NOSE GEAR.

Three hydraulic actuators are provided for operation of the nose landing gear and nose wheel well doors. A single-acting actuator retracts the nose landing gear. An uplock actuator locks the nose landing gear in the retracted position and also, through linkages, opens and closes the two nose wheel well doors. A downlock actuator locks the nose landing gear drag strut when the nose landing gear is extended.

LANDING GEAR CONTROLS AND INDICATORS.

Ground Safety (Squat) Switches.

The main landing gear safety (squat) switches are located on the main landing gear lateral beams as shown on figure 1-17 and provide an electrical signal that



1. Arresting Hook Handle.
2. Landing Gear Handle.
3. Landing Gear Warning Lamp
4. Landing Gear Handle Lock Release Button.
5. Landing Gear Warning Horn Silencer Button.
6. Auxiliary/Parking Brake Handle.
7. Utility Hydraulic System Isolation Switch.
*8. External Stores Ground Jettison Switch.

*See Applicable Weapons Delivery Manual.

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Figure 1-16.

Ground Safety Locks and Safety Pins

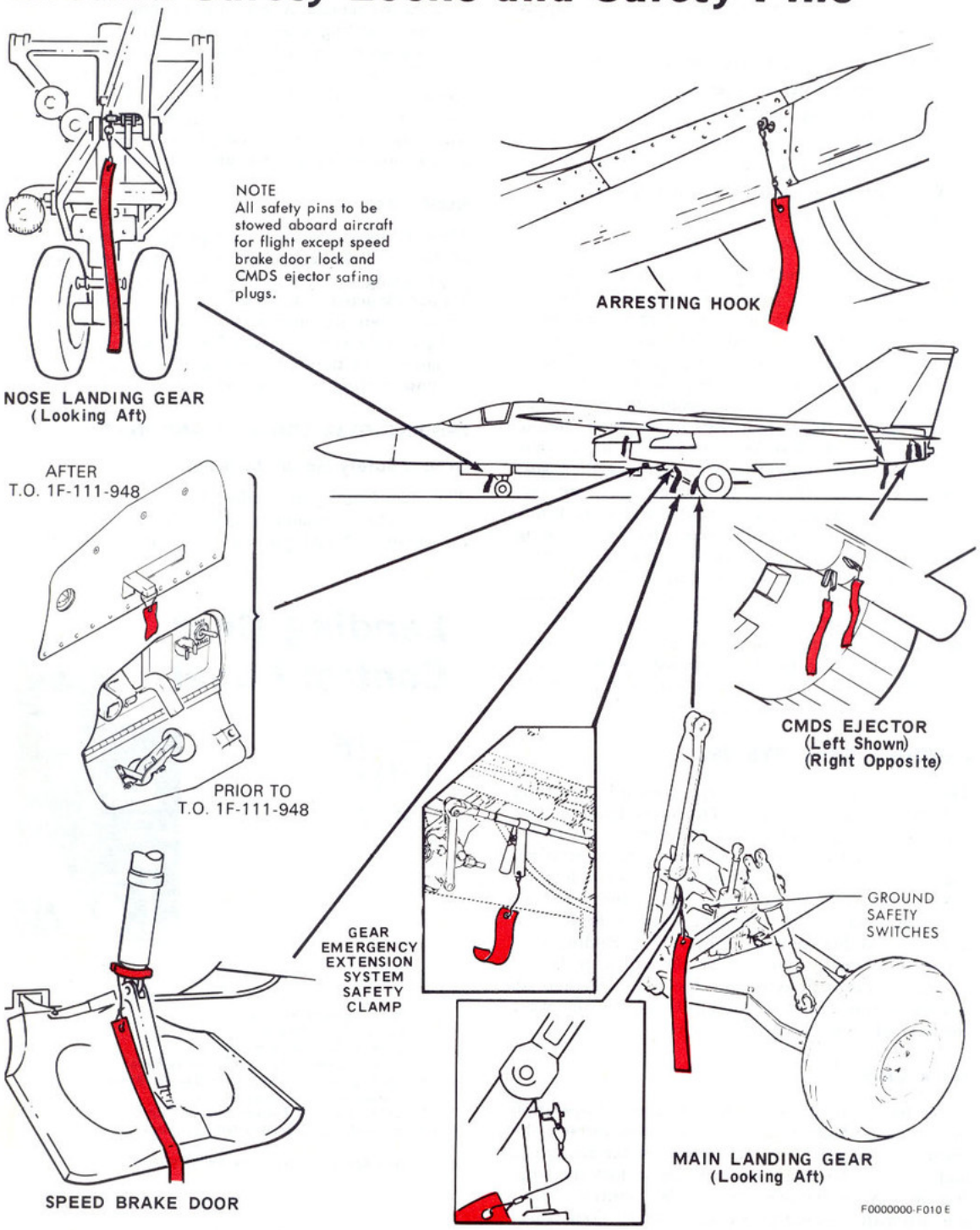


Figure 1-17. (Sheet 1)

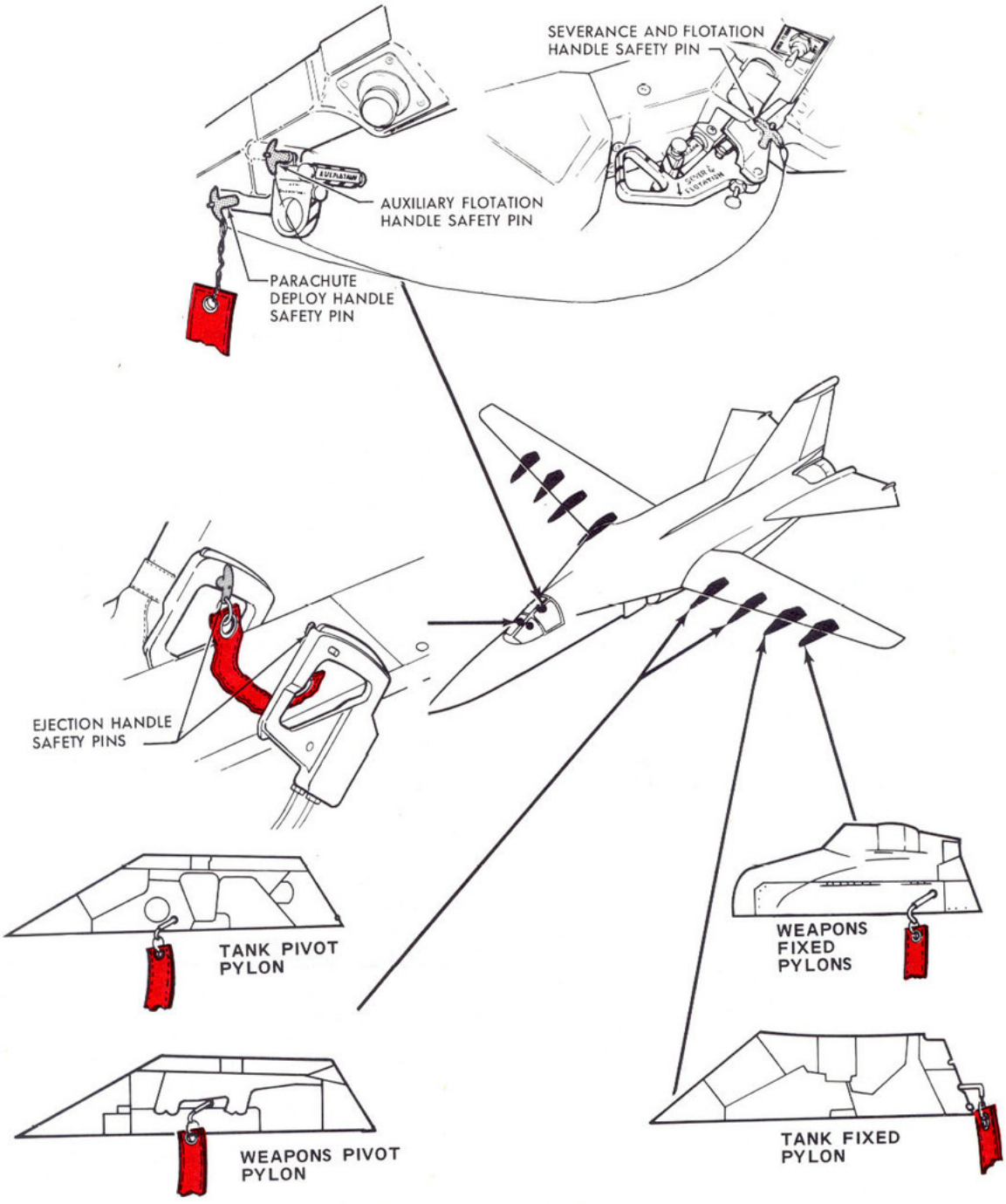


Figure 1-17. (Sheet 2)

will affect the operation of certain aircraft systems. While the weight of the aircraft is on the wheels, the squat switches prevent activation of the following in-flight systems.

1. Speed brake.
2. Adverse yaw compensation.
3. Artificial stall warning.
4. Landing gear handle lock release.
5. Secondary alpha/beta probe heater (with pitot/probe switch in OFF/SEC).
6. Anti-skid touchdown control.
7. External stores jettison button (with external stores ground jettison switch in OFF)

The switches will permit the activation of the following systems when the aircraft is on the ground and de-activate them when the aircraft becomes airborne.

1. Nose wheel steering.
2. Ground roll spoilers.
3. Ground to pilot interphone.
4. Ejector air hydraulic cooling.
5. Flight control ground test will remain armed.
6. Vortex destroyers.
7. Engine nozzle opening at idle.

Landing Gear Handle.

The landing gear handle (2, figure 1-16), located on the landing gear control panel, has two positions marked UP and DN. A landing gear warning lamp is located in the landing gear handle.

Gear Up

When the handle is moved to the UP position, an electrical signal actuates a solenoid-powered landing gear control valve, sending hydraulic pressure to the nose gear downlock actuator, nose gear retract actuator, nose gear uplock door actuator, the speed brake door actuator, and the brake control valve. Hydraulic pressure is metered to one brake circuit to stop main gear wheel rotation. The nose gear unlocks and retracts. When it is almost retracted, it mechanically triggers the nose gear uplatch which then locks the gear up and closes and locks the doors. As the nose wheel doors close, snubbers mounted on the doors engage the tires to stop nose wheel rotation. The speed brake door actuator extends the main gear forward door. When the door is sufficiently open to allow the main gear to retract, a linkage from the door opens a valve which sends hydraulic pressure to the main gear downlock actuator, main gear retract actuator, and the uplock actuator. The gear then unlocks and retracts. When it is almost retracted, it mechanically triggers the uplatch which locks the gear up and also actuates a valve to close the speed brake door.

Note

If the warning lamp on the gear handle remains lighted following gear retraction, the landing gear is not fully up and locked or the speed brake is open. For landing gear retraction malfunction procedures, refer to Section III.

Gear Down

When the handle is moved to the DN position, an electrical signal actuates a solenoid-powered valve, sending hydraulic pressure to the nose gear uplock actuator, nose gear downlock actuator, and the speed brake door actuator. The nose gear uplock actuator unlocks and drives the nose gear doors open and locked, at which time the nose gear is allowed to free fall (extend) against the snubbing of its retract actuator. When the gear is almost extended, the downlock actuator drives it fully extended and locked. The speed brake door actuator opens the door until the door clears the main gear. A linkage then actuates a valve to pressurize the main gear uplock actuator and downlock actuator. The uplock opens, allowing the gear to free fall (extend) against the damping of its retract actuator. When the gear is extended, the downlock actuates. This causes the speed brake door actuator to position the door in the partially retracted (trail) position. The landing gear handle is locked in the DN position by a spring-loaded electrical solenoid when the weight of the aircraft is on the landing gear. Landing gear safety switches control 28 volt dc power to the solenoid. The weight of the aircraft compresses the shock strut and opens the safety switches, which breaks the circuit to the solenoid. When the solenoid is de-energized, the solenoid extends a mechanical lock, holding the landing gear handle in the DN position. Removing the weight from the landing gear closes the safety switches on the landing gear and energizes the solenoid. The energized solenoid retracts the lock and frees the landing gear handle.

Landing Gear Handle Lock Release Button. The landing gear handle lock release button is located on the landing gear control panel (4, figure 1-16). The button must be depressed to release the landing gear handle from the UP position to lower the gear. Normally, it is not necessary to depress the button when retracting the gear since the gear handle is locked in the down position by a solenoid which will release the handle as the weight of the aircraft comes off the gear on takeoff. Should the solenoid malfunction, depressing the button will release the handle to allow gear retraction.

WARNING

Any time it is necessary to depress the landing gear handle lock release button to move the handle to the UP position, the crewmember should immediately suspect a malfunction of the landing gear ground safety switches or a tripped LG SAFETY REL circuit breaker. If a malfunction of the landing gear safety switches is suspected, refer to "Landing Gear Ground Safety (Squat) Switch Failure", Section III.

Landing Gear Emergency Release Handle.

The landing gear emergency release handle (10, figure 1-30), located on the right main instrument panel, is provided to extend the landing gear in the event the normal hydraulic system fails. When the handle is pulled, pneumatic pressure is directed to simultaneously open the speed brake door and unlock the nose and main gear uplocks. The gear will free fall to the extended position; then pneumatic pressure will actuate the nose and main gear downlocks and retract the speed brake door to the trail position. The speed brake door may fail to retract to the trail position. This will be indicated by the landing gear handle warning lamp remaining on (with the landing gear handle down) after the gear is extended and locked. Should this occur, pushing the handle back in will relieve the pressure in the system and allow the air load to push the speed brake door to the trail position.

CAUTION

If the landing gear emergency release handle has been pushed in, the weight of the speed brake door and reduced air loads during the landing roll may allow the speed brake door to extend and drag the runway.

Note

Once the gear has been extended by the emergency method, it cannot be retracted.

Landing Gear Warning Horn.

The landing gear warning horn provides an intermittent audible tone in the headsets when an unsafe landing gear situation exists. The horn sounds intermittently when the nose and main landing gear are not down and locked and/or the speed brake door is not in trail and all of the following conditions exist:

- Indicated airspeed is below 160 (±12) knots.
- The aircraft is below an altitude of 10,000 (±350) feet.
- One or both throttles are below full military power.

The warning horn also provides a steady audible tone in the headsets as a stall warning indication. Refer to "Artificial Stall Warning System," this section.

The malfunction and indicator lamp test button located on the lighting control panel may be used to test the landing gear warning horn. The warning horn may be silenced by depressing the horn silencer button adjacent to the landing gear handle (5, figure 1-16).

Note

The stall warning lamp will flash and the rudder pedal shaker will operate as long as the horn silencer button is held depressed.

Landing Gear Position Indicator Lamps.

A plan view silhouette of the aircraft having two green indicator lamps is located on the left main instrument panel (11, figure 1-6). The lamps are positioned to represent the nose and main landing gear. When the landing gear is down and locked, the lamps are lighted. In-transit positions of the landing gear and unsafe landing gear conditions are indicated by lighting of the red warning lamp in the landing gear handle knob. A safe up-and-locked landing gear condition is indicated when both the green indicator lamps and the red warning lamp are off.

TAIL BUMPER SYSTEM.

The tail bumper protects the control surfaces, engines, and portions of the airframe from damage that might occur if the tail inadvertently contacts the ground during ground handling. The tail bumper also provides limited protection during overrotation on takeoff and during landings. In flight, the tail bumper is held in the fully retracted position by hydraulic pressure in the tail bumper lift cylinder. The hydraulic pressure is ported to the tail bumper lift cylinder from the speed brake control valve. When the landing gear is extended and the speed brake returns to trail position, the lift cylinder pressure is relieved and the tail bumper is extended by the pneumatic action of the tail bumper dashpot. The dashpot, which functions as the impact shock absorber, has its own separate reservoir that is charged with compressed nitrogen. Retraction of the landing gear allows hydraulic pressure to again be ported to the tail bumper lift cylinder to retract the bumper and hold it in this position.

Note

The tail bumper may contact the ground if the pitch angle exceeds approximately 11 degrees when the main landing gear is in contact with the runway.

NOSE WHEEL STEERING SYSTEM.

Nose wheel steering provides aircraft directional control during taxiing, takeoff and landing. The system is electrically engaged, hydraulically powered and controlled by the rudder pedals. The nose wheels are positioned by a hydraulic actuator controlled by a mechanical rotary servo valve. Rudder pedal movement at either crew station is transmitted to the steering valve by mechanical linkage which includes a cam device on the valve input shaft. The cam device provides a gradually increasing ratio between steering angle and rudder pedal displacement. A relatively larger pedal displacement is required to obtain an increment of steering angle near the neutral rudder pedal position than is required near the full rudder pedal position. Utility hydraulic system pressure supplied to the steering servo valve is controlled by a solenoid operated shutoff valve and a pressure regulator. When steering is engaged, the energized solenoid valve applies full system pressure to achieve a high level steering torque. When steering is disengaged, the pressure regulator supplies approximately 10 percent system pressure for a low level steering torque used to center the nose wheels during retraction. Steering input linkage motion occurring during nose gear retraction automatically centers the nose wheels with up to 50 percent rudder pedal displacement.

CAUTION

If a misaligned/malfunctioning steering system is evident, do not take off unless required and do not retract the landing gear. Nose gear steering alignment can be checked by disengaging the nose wheel steering while taxiing on a level surface. If a steering transient is observed on re-engagement, a misalignment/malfunction is indicated.

Maximum rudder pedal deflection steers the nose wheels 40 degrees either side of center with resultant aircraft turning radius as shown on figure 2-3. Nose wheel shimmy damping is accomplished by restricting hydraulic flow within the steering valve. If the flaps and slats are retracted, the flight control system switch must be in the T.O. & LAND position or the rudder authority switch must be in the FULL position to allow sufficient rudder pedal movement for full steering authority. The nose wheel steering system is equipped with a limit switch mounted on the nose landing gear shock strut adapter. When nose wheel steering exceeds approximately 40° from center, the switch opens an electrical circuit to the control valve and automatically prevents controlled steering through the rudder pedals. The NWS/AR lamp will go out whenever controlled steering range is exceeded. When nose wheels are returned to normal steering range (0° to 40°), controlled steering automatically reengages. Power for engaging steering is furnished from the essential dc bus.

Note

- Nose wheel steering will not be available if the landing gear is extended using the emergency release handle.
- With the air refuel switch in OPEN and the engine feed selector knob in AUTO or BOTH, nose wheel steering will be operative only while the nose wheel steering/air refuel button is depressed.
- With the engine ground start switch in any position except OFF, nose wheel steering will be operative only while the nose wheel steering/air refuel button is depressed.

NOSE WHEEL STEERING/AIR REFUEL BUTTONS.

A nose wheel steering/air refuel button (3, figure 1-24), is located on each control stick grip. The buttons are labeled NWS and A/R DISC. With the weight of the aircraft on the main landing gear, depressing either button actuates a holding relay to engage the system. The button can then be released, and the system will remain engaged until the button is again depressed and released to open the relay and disengage the system.

Note

- When the nose wheel steering button is depressed and released to disengage the system, a three second delay is initiated. The system may be re-engaged during this period but the holding relay will not be energized.
- During ground operation when the air refueling door is open, the nose wheel steering/air refueling indicator lamp will light to indicate door position and nose wheel steering will function only while the NWS A/R DISC button is held depressed.

The button receives 28 volt dc power from the essential bus. For a description of the A/R DISC function of the buttons refer to "Fuel Supply System", this section.

NOSE WHEEL STEERING/AIR REFUELING INDICATOR LAMP.

A green nose wheel steering/air refueling indicator lamp labeled NWS A/R is located on the left main instrument panel (19, figure 1-6). The lamp will light when the nose wheel steering system is energized or the A/R door is open. For a description of the A/R DISC function of the lamp, refer to "Fuel Supply System", this section. The lamp receives power from the 28 volt dc essential bus.

BRAKE SYSTEM.

Each main landing gear wheel is equipped with a hydraulically operated multiple disc brake. Pressure for operation of the brakes is supplied by the utility hydraulic system for normal operation and by two hydraulic accumulators for power-off braking. Anti-skid control, automatic braking during landing gear retraction, and an auxiliary brake are provided. Normal brake operation is controlled by conventional brake pedals, each mechanically connected to brake metering valves. The brake hydraulic system is a dual-normal type, separated into two circuits. Each circuit operates independently of the other. One circuit operates one-half the pressure pistons on each brake; the other circuit operates the other half. During normal operation of the brakes, pressure is metered to the brakes from both hydraulic circuits in proportion to applied force on the brake pedals. Full braking effectiveness is achieved with approximately 60 percent of full brake pedal travel. If one hydraulic circuit becomes inoperative, the brake system can provide sufficiently increased pressure to the operative circuit for 90 percent of normal braking effectiveness. This is accomplished by application of greater than normal brake pedal travel and slightly higher pedal force. The dual-normal type brake hydraulic system provides emergency brake operation automatically. Two hydraulic accumulators are provided in the system to supply brake system pressure for normal power off braking due to failure or isolation of the utility hydraulic system. Each accumulator is precharged and supplies pressure to only one of the individual brake circuits. Fully charged accumulators will provide 10-14 full-pressure brake applications or one full-pressure brake application with 32 anti-skid cycles. A priority valve, which limits the quantity of fluid that can be displaced from the brake accumulator through the brake metering valves by actuating the brake pedals, is included in each hydraulic circuit. If the brake accumulators are not replenished as fluid is displaced by repetitive brake applications or by anti-skid cycling, the priority valves will close when accumulator pressure has been reduced to approximately 1100 (±100) psi. At this pressure all normal braking will be lost and the pedals will be fully depressed.

Note

Brake pedals should not be actuated inflight when utility hydraulic pressure is isolated from the brake system, as emergency brake accumulators will be depleted. If the utility hydraulic system subsequently fails or becomes isolated, pedal braking may not be available on landing. If brake pedals are actuated inflight during normal utility hydraulic system isolation, placing the utility hydraulic system isolation switch to PRESSURIZE for several seconds will replenish brake accumulators.

After the priority valves close, the remaining fluid can be utilized only by pulling the auxiliary brake handle; however, this will lock the brakes. No braking action can be achieved by actuating the brake pedals.

AUTOMATIC BRAKING SYSTEM.

The automatic braking system functions to stop wheel rotation after takeoff. When the aircraft has become airborne, moving the gear handle to the UP position will provide hydraulic pressure through the brake control valve at approximately 750 psi to one circuit of the brake system. The hydraulic pressure is applied to one half the pressure pistons in each wheel to stop wheel rotation. The pressure is relieved after the brakes are isolated from the hydraulic system.

CAUTION

Make certain that the aircraft is definitely airborne before retracting landing gear. Premature movement of the landing gear handle to UP may result in blowing out both main gear tires due to the action of the automatic braking system.

ANTI-SKID SYSTEM.

The anti-skid touchdown control system provides the following functions:

- Anti-skid touchdown control.
- Proportional skid control.
- Lock wheel skid control.
- Anti-skid failure detection.

Anti-skid touchdown control prevents the brakes from being applied when the weight of the aircraft is off the landing gear and the speed of both wheels is below 20 MPH. Proportional skid control operates throughout the aircraft ground speed range by utilizing wheel deceleration to reduce brake pressure in proportion to a skid tendency. Locked wheel skid control is activated above 20 MPH and causes either brake to be fully released if proportional skid control does not prevent a skid from occurring. Locked wheel skid control would function, for example, should a brake seize or if a wheel is unable to spin-up due to hydroplaning.

Note

The failure detection circuit will automatically return the brake system to manual control in the event of an anti-skid malfunction or essential dc electrical failure. However, a malfunction in the failure detect circuit may not revert the brake system to manual control; if loss of braking occurs during the landing roll, the anti-skid system should be turned off to assure deactivation of the anti-skid system.

Anti-Skid Control Switch.

The anti-skid control switch (2, figure 1-5) is located on the left throttle panel and labeled ANTI-SKID. The switch has two positions, one marked OFF and an unmarked ON (up) position. Placing the switch to ON will provide anti-skid control during normal braking. With the switch in OFF, anti-skid control will not be available and brake pressure will be in direct response to pedal pressure.

Anti-Skid Caution Lamp.

A caution lamp labeled ANTI-SKID is located on the main caution lamp panel (figure 1-36). The lamp will light when the anti-skid switch is in the ON position and a malfunction has caused the anti-skid system to become de-energized or when the landing gear is down and the anti-skid switch is not in the ON position.

Note

When the lamp is lighted, anti-skid control is not available and braking will be in direct response to pedal pressure.

AUXILIARY PARKING BRAKE HANDLE.

The auxiliary parking brake handle (6, figure 1-16), labeled BRAKE, is located on the landing gear control panel. When the handle is pulled out, a mechanical linkage opens a selector valve which admits pressure from the hydraulic accumulators directly into the brake lines downstream of the brake control valve. The primary function of the auxiliary parking brake handle is to apply the brakes while the aircraft is parked. The auxiliary parking brake handle can be used to set the brakes for engine run-up. A secondary function of the auxiliary parking brake is to serve as a supplemental emergency brake in the event that accumulator pressure is reduced sufficiently to cause the priority valves to close and prevent normal brake application by pedal actuation. Brake pressure cannot be metered by the auxiliary parking brake handle. The total accumulator pressure is ported directly to the brake cylinders, bypassing the metering valves and the anti-skid valves. Therefore, the auxiliary parking brake handle should not be pulled while the aircraft is in motion except when braking cannot be achieved by pedal actuation.

WARNING

If normal brake accumulator pressure is available, pulling the auxiliary parking brake handle while the aircraft is moving may cause the wheels to lock, resulting in tire skidding or blowout. Tire blowout while moving may result in fire.

Note

If brake accumulator pressure has been reduced following a hydraulic system failure, pulling the auxiliary parking brake handle while the aircraft is moving may lock the brakes on a wet or icy surface; on a dry surface, the brakes should not lock.

BRAKE HYDRAULIC HAND PUMP.

A hydraulic hand pump, located in the main landing gear wheel well, is provided to replenish brake accumulator pressure during ground handling operation.

AIRCRAFT ARRESTING SYSTEM.

The arresting hook system provides for emergency arrestment of the aircraft. The system consists of an arresting hook, arresting hook dashpot, a dashpot air bottle, an uplock latch, arresting hook controls, a pressure gage, and an air filler valve. Except for the controls, the arresting hook components are located in the lower aft end of the fuselage tail cone.

ARRESTING HOOK HANDLE.

The arresting hook handle (1, figure 1-16), located on the landing gear control panel, is connected to a low friction push-pull type mechanism contained in a flexible metal housing. The handle is labeled HOOK on diagonal stripes. The mechanism provides a direct mechanical linkage from the handle to the arresting hook uplatch mechanism in the tail cone. The arresting hook is released by grasping the handle and pulling aft. The total travel of the handle from retract to extend position is approximately 4 inches. Approximately 1 second is required for the arresting hook to extend. The hook must be raised manually to its stowed position.

ARRESTING HOOK CAUTION LAMP.

A hook caution lamp, labeled HOOK DOWN, is located on the main caution lamp panel (figure 1-36). The caution lamp lights to indicate hook down position only.

AERODYNAMIC DECELERATION EQUIPMENT.

SPEED BRAKE.

The speed brake, which also serves as the main landing gear forward door, is provided as an aid to deceleration during flight. The speed brake is hydraulically operated and may be used as a speed brake only when the landing gear is up and locked. For operation of the speed brake as a landing gear door refer to "Landing Gear System," this section.

Speed Brake Switches.

A three-position speed brake switch (4, figure 1-5), marked IN, OFF and OUT is located on the right throttle of each crew station. The switches are thumb actuated and slide forward (IN) and aft (OUT). The left crew station switch is detented in all positions. The right crew station switch is spring-loaded to OFF from both the IN and OUT positions and will override the left crew station switch. When the right switch is released to OFF, the speed brakes will move to the position selected by the detented left crew station switch. When both switches are positioned to OFF, the speed brake is hydraulically locked in its present position. To maintain a constant load on the landing gear door and to insure minimum drag, the left crew station switch must remain in the IN detent, except during speed brake operation. If the speed brake switch is positioned to OUT or OFF and the landing gear is retracted, the speed brake may remain fully open or retract only to the trail position. The speed brake will retract when the speed brake switch is moved to the IN detent.

GROUND ROLL SPOILERS.

For deceleration during ground roll, the ground roll spoilers are used symmetrically to destroy lift and increase wheel brake effectiveness. In this case, they are armed with the ground roll spoiler switch and activated by the throttle position switches and landing gear squat switches. Ground roll spoiler operation is not affected by the position of the flight control computer power switches.

Ground Roll Spoiler Switch.

The ground roll spoiler switch (8, figure 1-25), located on the left sidewall, has positions BRAKE and OFF. If the weight of the aircraft is on the landing gear and both throttles are in IDLE, positioning this switch to BRAKE will cause the ground roll spoilers to extend. Under the same conditions, placing the switch to OFF will retract the spoilers. With the spoiler switch positioned to BRAKE, if the aircraft weight is removed from the landing gear or if either throttle is advanced out of IDLE, the spoilers will automatically retract.

WING FLAPS AND SLATS.

MAIN WING FLAPS.

The main (wing) flaps are full span, multisection Fowler-type flaps. Each wing flap is divided into six sections. The five outer sections, designated as the main flaps, are mechanically connected and operate as one unit. The main flaps are powered by a single hydraulic motor which is connected to a gearbox located in the fuselage section. The hydraulic motor and gearbox assembly drive a torque shaft which is connected through gearboxes to mechanical actuators attached to the flaps. An electric motor mounted on this same gearbox provides an emergency mode of operation in the event of either hydraulic system failure. A mechanical interlock control prevents the wing from be-

ing swept aft of the 26 degree position with the main flaps extended. The mechanical interlock locks the flap and slat handle in the UP position when either the wing sweep angle is greater than 26½ degrees or when the wing sweep handle is at a position greater than 26½ degrees. Asymmetrical flap travel is monitored by an asymmetry device. When the asymmetry device senses more than 3 degrees asymmetrical flap travel, a signal is sent to close the flap drive control valve and to engage torque shaft brakes which stop travel of the flaps and slats. Once the flap drive control valve has been closed and the torque shaft brakes are engaged by this method, the flaps cannot be extended or retracted by either the normal or emergency mode. Integral with each main flap section is a mechanically controlled vane. As the flap extends downward, the vane is positioned by a mechanical linkage to provide the proper airflow through the space between the flap leading edge and the spoiler trailing edge. The main flap hydraulic motor receives pressure from the utility hydraulic system.

AUXILIARY FLAPS.

The inboard section, designated as the auxiliary flap, operates independently from the main flaps by electrical actuators. The auxiliary flap actuators are disabled by electrical interlocks when either the wing sweep angle switch senses more than 16 degrees wing sweep or when the wing sweep handle is at a position greater than 16 degrees. A wing sweep mechanical interlock control prevents the wing from being swept aft of the 26 degree position with the auxiliary flaps up and the main flaps extended. There is no mechanical connection between the auxiliary flaps since there is no necessity to prevent asymmetrical operation. The auxiliary flap actuators receive 115 volt ac power from the right main ac bus.

WING SLATS.

Each wing is equipped with a leading edge slat. Each slat is divided into five sections which are connected and operate as one unit. The slats operate in conjunction with the main flaps and are connected to the main flap drive assembly by flexible drive shafts. On the extend cycle, the slats will extend to the full down position before the main flaps start to extend. On the retract cycle, the flaps will fully retract before the slats start to retract.

Note

If an asymmetrical slat condition occurs, the aircraft will enter a roll in the direction of the extended slat. The initial movement of the flaps will cause the slats and flaps to lock.

ROTATING GLOVES.

The outboard edges of the wing gloves, adjacent to the wing inboard leading edges, are equipped with movable surfaces to allow full forward movement of the inboard slats. These surfaces are called rotating gloves (4, figure 1-1). A door forms the lower surface of each rotating glove. Each rotating glove and its associated door are operated by a mechanical actuator and linkage which is connected to the slat drive flexible shaft. When the slats are extended, the rotating gloves automatically rotate (leading edge down and trailing edge up), and the doors open to allow full extension of the slats.

Flap/Slat Handle.

The flap/slat handle (9, figure 1-5), located on the left throttle panel, has three positions marked UP, SLAT DOWN, and FLAP DOWN. A manually operated gate (8, figure 1-5), located between the SLATS DOWN and FLAP DOWN areas, must be released before the handle can be moved from one area to another.

Note

Allow the flaps to fully retract or the slats to extend completely before releasing the gate. This will expedite reversing flap/slat travel if an asymmetrical condition occurs.

When the handle is moved from UP to any position in the SLAT DOWN area, a mechanical linkage opens the flap drive control valve, directing hydraulic pressure to the flap drive motor. The flap drive assembly rotates the flexible shafts connected to the slat drive mechanism to position the rotating glove and to extend the slats to a position corresponding to handle position. Moving the handle down to the gate will cause the slats to fully extend. When the gate is released and the handle is moved into the FLAP DOWN area, the flap drive assembly will rotate the flexible shafts connected to the main flap actuators, extending the main flaps to a position corresponding to handle position. The flaps can be set at an infinite number of positions between full up and full down. The flap and slat drive assembly is so designed that it will not extend the flaps until the slats are fully extended. Two detent positions are provided in flap handle travel to aid in selecting the 15-degree and 25-degree flap positions. When the handle is moved down to the 25-degree detent, a contact closes and supplies electrical power to the auxiliary flap actuators, causing the auxiliary flaps to extend to 25 degrees. When the handle is moved beyond the 25-degree detent, the main flaps extend, but the auxiliary flaps remain at 25 degrees. The full FLAP DOWN position of the handle provides 34 degrees of flap extension. The retract cycle is the opposite of the extension cycle. When the handle is moved from the full FLAP DOWN position

to the UP position, the flaps retract first, followed by the slats. It should be noted that at no time will the flaps extend unless the slats are fully extended, nor will the slats retract until the flaps are fully retracted, regardless of the position of the flap and slat handle. When the handle is moved up to any position less than the 25-degree detent, the auxiliary flaps will fully retract. Normal extension or retraction of the flaps and slats takes approximately 12 seconds.

Note

The handle controls only the auxiliary flaps when the flap/slat system selector switch is in the EMER position.

Flap/Slat System Selector Switch.

The flap/slat system selector switch (14, figure 1-25), located on the left sidewall, has two positions marked EMER and NORM. When the switch is in the NORM position, the flaps and slats are actuated normally by use of the flap handle. When the switch is in EMER, the flaps and slats may be extended or retracted electrically by holding the emergency flap and slat switch to EXTEND or RETRACT, as applicable, and the flap drive control valve is closed, disabling the flap drive motor. The EMER position is used in the event of hydraulic system failure.

Emergency Flap and Slat Switch.

The emergency flap and slat switch (15, figure 1-25), located on the left sidewall, has positions marked EXTEND and RETRACT and is spring-loaded to the center unmarked OFF position. The switch is provided as an emergency method of operating the main flaps and slats in the event of a utility hydraulic system failure. Operation of the flaps and slats using this switch is identical to that when using the flap and slat handle except that electrical power is used to operate the flap drive motor instead of hydraulic power.

Note

Emergency flap extension or retraction takes approximately 60 seconds at 180 KIAS. This time will vary with airspeed.

Flap and Slat Position Indicators.

The flap and slat position indicators are a part of the wing sweep, flap/slat position indicator (10, figure 1-6), located on the left main instrument panel. The main flap position indicator provides flap position in degrees. The slat/auxiliary flap indication is a window which provides the following indications:

- UP—Slats and auxiliary flaps retracted.
- SLAT DN—Slats down and auxiliary flaps retracted.
- BOTH DN—Slats and auxiliary flaps full down.
- Crosshatch—Power off or slats or auxiliary flaps in transit.

WING SWEEP SYSTEM.

The variable sweep wings are moved to and held in position by two hydraulic, motor-driven, linear actuators. The actuators are mechanically interconnected to insure positive synchronization (figure 1-18). The left actuator is furnished power by the primary hydraulic system, and the right actuator is furnished power by the utility hydraulic system. In the event of failure of either hydraulic system, the remaining system, by utilizing the load transfer capability of the mechanical interconnect, will still provide wing actuation. However, actuation under this condition will be at a reduced rate. Wing position is controlled by an input signal from the wing sweep handle. The maximum rate at which the wings extend or retract is controlled by flow-limiting devices in the hydraulic lines. Directional reversal, due to aerodynamic loads, is prevented by the nonreversing threads in the actuator. A mechanical interlock prevents the wing sweep handle from being moved past the 26½ degree position when either the flap and slat handle is out of the UP position or the main flaps are out of the fully retracted position. The wing sweep handle is locked in the 16 degree position by a solenoid operated latch whenever the auxiliary flaps are out of the zero position. Should total electrical power be removed, the solenoid-operated latch would be inoperative and wing position would lock at 16 degrees if the wing sweep control handle were positioned fully forward.

CAUTION

Do not attempt to sweep the wings aft of the 16 degree position with the auxiliary flaps extended. Should the solenoid-operated latch malfunction, the auxiliary flaps will contact the fuselage when the wings are swept aft of 16 degrees.

WING SWEEP HANDLE.

The wing sweep handle (12, figure 1-25) is shaped like a pistol grip and is spring-loaded to a stowed position under the canopy sill on the left side of the crew module. Teeth in the top of the handle lock it to serrations in the handle support when it is stowed, to prevent inadvertent movement. To adjust wing sweep, the handle must be rotated to the vertical position to unlock it; then it can be moved forward or aft as necessary. The handle is mechanically linked to the wing sweep control valve. The handle is pulled aft to sweep the wings aft and pushed forward to

Wing Sweep and Pylon System

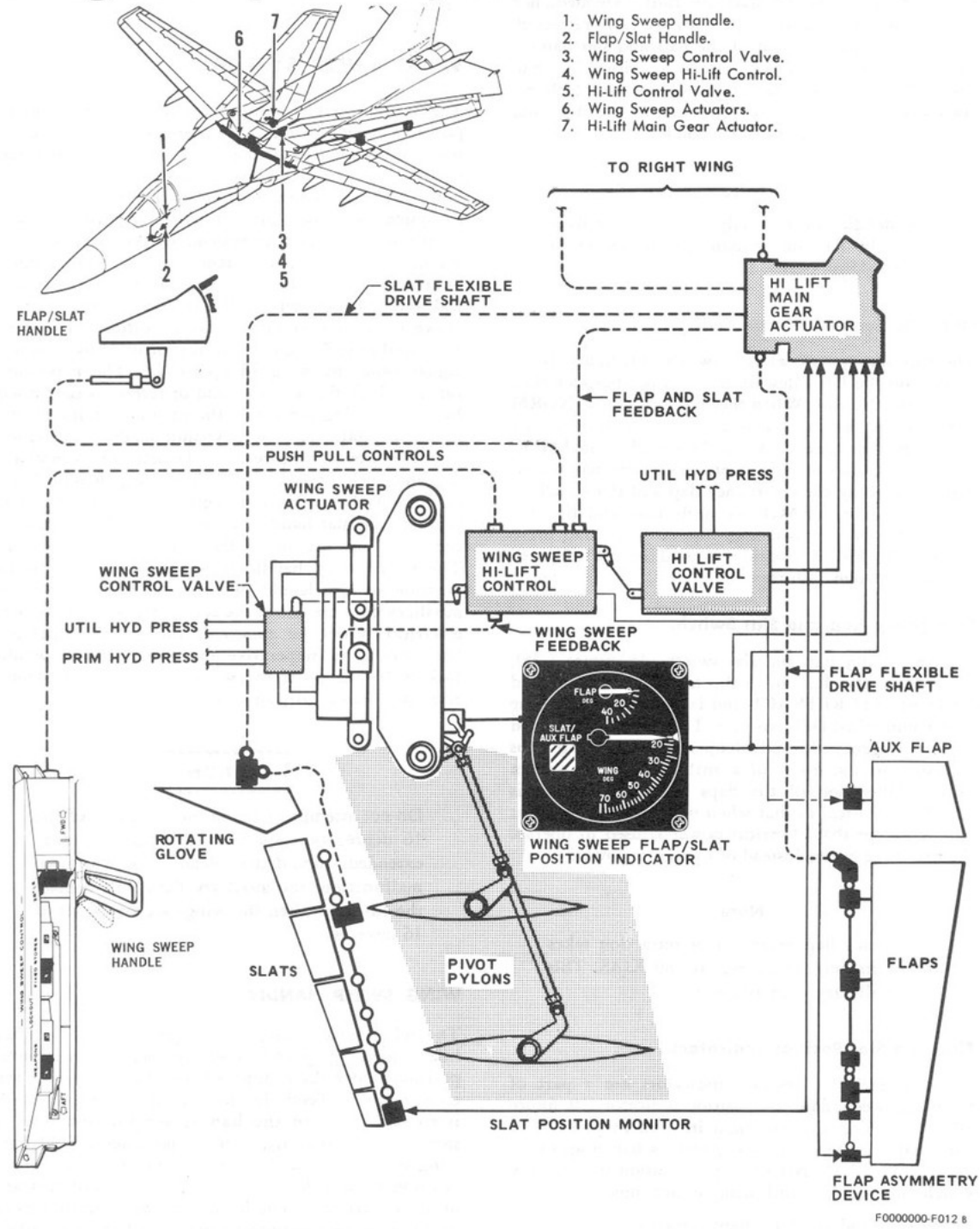


Figure 1-18.

sweep the wings forward. As the handle is moved, an index marker on the wing sweep position indicator follows the handle position to assist in selecting the desired wing sweep position.

WING SWEEP HANDLE LOCKOUT CONTROLS.

Two wing sweep handle lockout controls (10, figure 1-25), one labeled FIXED STORES and the other labeled WEAPONS, are located just above and aft of the wing sweep control handle. When either control is moved forward, the word ON is visible, and a latch extends which prevents aft movement of the wing sweep handle past the latch. When either control is moved aft, the word OFF is visible and the latch retracts. The fixed stores lockout control, when ON, prevents the wing sweep handle from being moved aft past the 26 degree position. This is the sweep angle at which the fixed pylons and stores are in a streamlined configuration. The weapons lockout control restricts aft movement of the wing sweep handle to 54 degrees. This is the wing sweep angle past which certain weapons on the inboard pivot pylons would strike the fuselage. The wing sweep handle lockout controls restrict aft movement of the wing sweep handle only. Forward motion is unrestricted between 72.5 and 26 degrees.

WING SWEEP HANDLE 26 DEGREE FORWARD GATE.

A wing sweep handle 26 degree forward gate (11, figure 1-25), located above the wing sweep handle, is provided to stop forward motion of the wing sweep handle at 26 degrees. The gate is thumb-actuated and is spring-loaded to the latched position. Depressing the gate will retract a latch, allowing the wing sweep handle to be moved forward past the 26 degree position.

WING SWEEP POSITION INDICATOR.

The wing sweep position indicator (10, figure 1-6) is a part of the wing sweep, flap/slat position indicator located on the left main instrument panel. The indicator displays the wing position in degrees and is graduated in 2 degree increments from 16 to 72 degrees. An index mark at 26 degrees provides a reference for selecting this position. Selected wing sweep angle is indicated by a reference index marker on the outside of the scale. The actual angle of wing sweep is monitored by a transmitter which mechanically follows the change in wing position and converts this information to an electrical signal which drives the wing sweep indicator pointer. The indicator receives power from the 28 vdc essential bus.

FLIGHT CONTROL SYSTEM.

The primary aircraft control surfaces consist of a rudder, spoilers on each wing and movable horizontal stabilizers. Pitch attitude of the aircraft is controlled by symmetrical deflection of the horizontal stabilizer surfaces. Roll attitude is controlled by asymmetrical deflection of the horizontal stabilizer surfaces; and when the wing sweep angle is less than 45 degrees, roll control is aided by action of two spoilers on top of each wing. Yaw control of the aircraft is accomplished by deflection of a rudder surface located on the trailing edge of the vertical stabilizer. Hydraulic servo actuators are used to produce control surface movement. The control stick at each crew station is mechanically and electrically interconnected with the flight control system. The right stick may be removed for various mission requirements. This must be accomplished while the aircraft is on the ground. When the right stick has been removed, an electrical plug is inserted in place of the stick to maintain electrical continuity. The two sets of rudder pedals are mechanically linked together. The stability augmentation system employs redundant sensors, electronic circuitry and electro-hydraulic dampers. The three damper actuators, the horizontal stabilizer actuators, and the rudder actuator are supplied by both primary and utility hydraulic systems and can operate on either system should one system fail. The pitch and roll damper response (gain) is varied by a self-adaptive system as flight conditions change. Command augmentation, through the pitch and roll dampers, augments the pilot inputs to provide a near constant relationship between control force and aircraft response throughout the operational envelope. Automatic failure detection and rejection, as well as self-test features, are provided in the pitch, roll, and yaw stability augmentation systems. Should electrical power be absent from one or more of the redundant computers, the applicable channel caution lamp will light. Power to all computers is controlled from the three computer power switches located on the ground check panel. The pitch and roll damper systems accept inputs from the CADC and the navigation system to provide pitch and roll autopilot modes. The flight control system functions in conjunction with the terrain following radar (TFR) through the pitch damper to maintain the aircraft at a preselected altitude above the terrain.

PITCH CHANNEL.

Mechanical Linkage.

Manual control of the aircraft in pitch is achieved by fore and aft movement of the control stick. This movement is transmitted along the pitch channel push-pull tubes and bellcranks to the left and right horizontal stabilizer actuator control valves. These control valves control the flow of hydraulic fluid to the actuators, thus causing the horizontal stabilizers to move symmetrically. Figure 1-19 is a simplified schematic of the mechanical flight control linkage. Control stick centering and feel forces are provided by the pitch feel spring.

Parallel Trim Actuator.

The parallel trim actuator will cause displacement of the horizontal stabilizers as well as the control stick, unless the stick is manually restrained.

Series Trim Actuator.

The series trim actuator output is added downstream of the pitch feel spring. Displacement of this actuator will cause the horizontal stabilizers to displace, but it will not normally cause displacement of the control stick. Likewise, displacement of the pitch damper servo will cause the horizontal stabilizers to displace but will not normally cause displacement of the control stick.

Control Stick Movement.

Control stick displacement can occur due to series trim actuator or pitch damper displacements if, for some reason, their action can not cause displacement of the pitch command output rod shown in figure 1-19. (Refer to "Stick Talk Back", this section.) Control stick stops are provided in the crew module to limit stick motion.

Pitch Command Limits.

Pitch commands to the pitch/roll mixer are limited by the pitch mixer stops. The following table summarizes these stop limits and the limits of the other components just described. In all cases, the values listed are in degrees of horizontal stabilizer deflection assuming all the other inputs are zero.

INPUT	NOSE UP LIMIT	NOSE DOWN LIMIT
Control stick stop	22 degrees	14 degrees
Pitch mixer stop	25 degrees	15 degrees
Parallel trim actuator	10 degrees	8 degrees
Series trim actuator	10 degrees	4 degrees
Pitch damper servo	13 degrees	13 degrees

With both trim actuators at zero and the pitch damper at zero and OFF, total stick travel from neutral to full aft is approximately 6½ inches, and from neutral to full forward is approximately 4 inches. The force required to move the stick from neutral to full aft ranges from the initial breakout force of 1.7 pounds to 65 pounds. The force required to move the stick from neutral to full forward ranges from 1.7 pounds to 55 pounds. When takeoff trim is set, the parallel trim actuator drives to a zero degree surface command, and the series trim actuator will drive to position both horizontal stabilizers to 3.8 degrees trailing edge up. With the pitch damper on, control stick displacement or parallel trim actuator displacement will cause the pitch damper to displace in response to signals from the pitch stick transducer. (Refer to "Pitch Command Augmentation," this section.)

Power Supply.

Power to the pitch damper servo, series trim actuator, parallel trim actuator, and stick transducer is controlled from the three computer power switches on the ground check panel. When these switches are turned OFF, the pitch damper actuator is hydraulically driven to its zero position and the series and parallel trim actuators will stop. If hydraulic pressure is not available, the position of the pitch damper servo is indeterminate.

Pitch/Roll Mixing.

Combined pitch and roll movements of the control stick are transmitted by the linkage of their respective channel to the pitch/roll mixer assembly. In this assembly the pitch/roll commands are summed mechanically and converted into left and right horizontal stabilizer actuator command signals. (See figure 1-19.)

Mixer Outputs.

The output of the pitch roll mixer is transmitted by two push-pull tubes causing displacement of control valves on the left and right horizontal stabilizer actuator. When the control valves are displaced from neutral, hydraulic fluid is ported to the respective actuator which results in horizontal stabilizer displacement. Deflection of the left and right horizontal stabilizers is limited by the horizontal stabilizer actuator ram stroke. The nominal nose up limit is 30 degrees and the nose down limit is 15 degrees.

Pitch Trim System.

Pitch trim can be effected from one of three inputs: paralled pitch trim, series pitch trim, and auxiliary pitch trim through the pitch damper servo. Parallel pitch trim allows the pilot to change the stick neutral point and is the prime trim mode in the takeoff and landing configuration. Also it may be used as a supplemental trim when the flight control system is in normal. Series pitch trim is provided to carry the elevator required for trim and serves to minimize

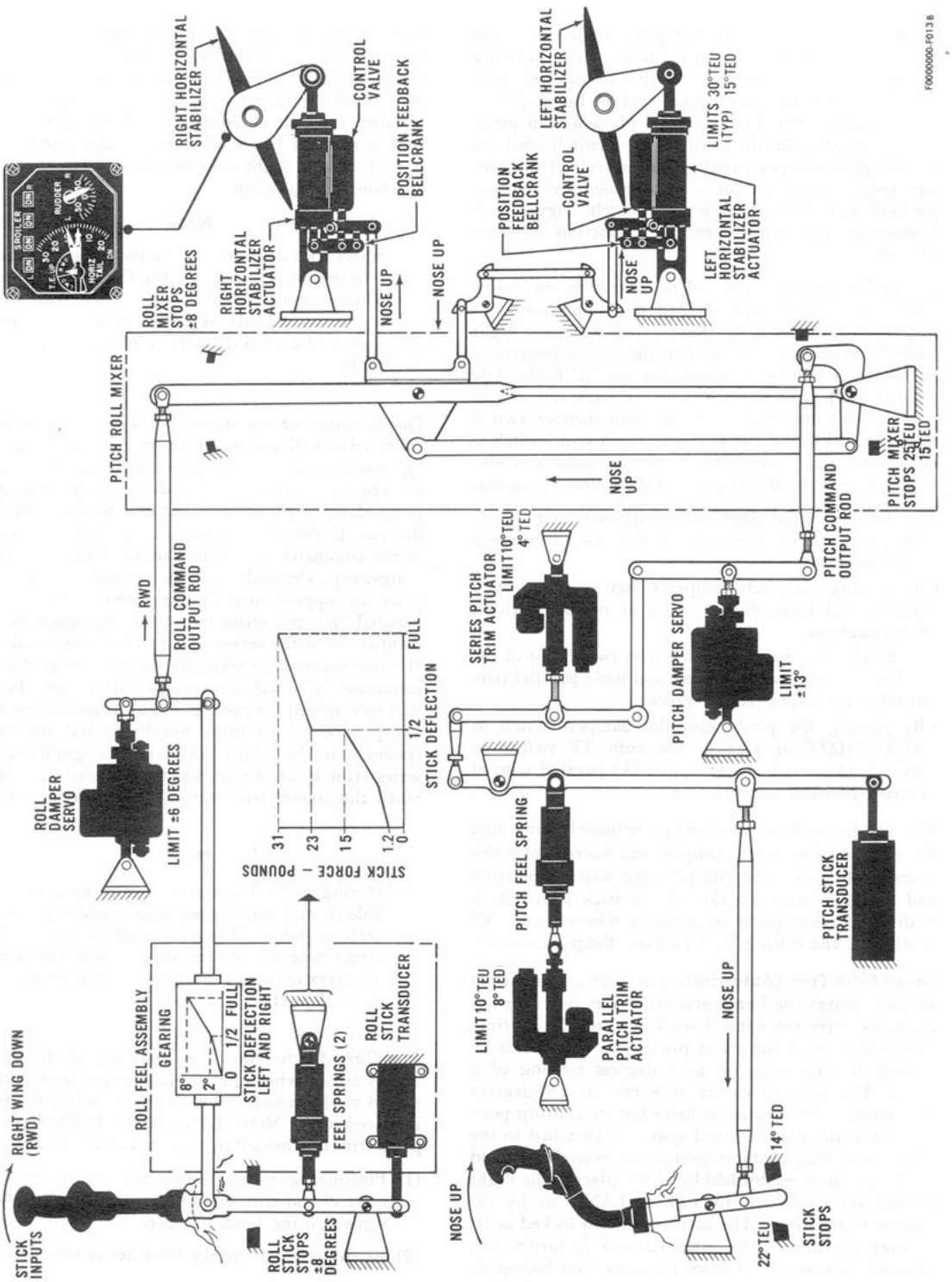

Pitch and Roll Mechanical Schematic
F0000000-F013 B
SPOILER
RUDDER
T.E. UP
HORIZ TAIL
RIGHT HORIZONTAL STABILIZER
CONTROL VALVE
POSITION FEEDBACK BELLCRANK
RIGHT HORIZONTAL STABILIZER ACTUATOR
ROLL MIXER STOPS ±8 DEGREES
NOSE UP
LEFT HORIZONTAL STABILIZER
LIMITS 30°TEU (TYP) 15°TED
POSITION FEEDBACK BELLCRANK
CONTROL VALVE
LEFT HORIZONTAL STABILIZER ACTUATOR
PITCH ROLL MIXER
PITCH MIXER STOPS 25°TEU 15°TED
PITCH COMMAND OUTPUT ROD
RWD
ROLL COMMAND OUTPUT ROD
SERIES PITCH TRIM ACTUATOR
LIMIT 10°TEU 4° TED
PITCH DAMPER SERVO
LIMIT ±13°
ROLL DAMPER SERVO
LIMIT ±6 DEGREES
STICK FORCE – POUNDS
31
23
15
1.2
0
1/2 STICK DEFLECTION
FULL
PITCH FEEL SPRING
PITCH STICK TRANSDUCER
LIMIT 10°TEU 8°TED
PARALLEL PITCH TRIM ACTUATOR
ROLL FEEL ASSEMBLY
GEARING
8°
2°
0 1/2 FULL
STICK DEFLECTION LEFT AND RIGHT
FEEL SPRINGS (2)
ROLL STICK TRANSDUCER
RIGHT WING DOWN (RWD)
STICK INPUTS
ROLL STICK STOPS ±8 DEGREES
NOSE UP
14° TED
22°TEU
STICK STOPS

Figure 1-19.

steady state errors during autopilot modes and auto TF operation. An auxiliary pitch trim system, utilizing the series trim actuator and pitch computer, is provided as a backup trim system and to be used if a trim failure occurs. The operation of each trim input is determined by (a) the auxiliary trim switch position, (b) the pitch damper switch position, and (c) the control system configuration as determined by the slat position and the control system switch. Figure 1-20 summarizes the trim system as a function of these variables.

Parallel Pitch Trim. Parallel pitch trim is the prime trim mode in the takeoff and landing configuration. It may also be used as supplemental trim when the flight control system is in normal. During normal operation, parallel trim will be at neutral for one "g" flight. The parallel pitch trim actuator is driven from the control stick(s) trim button only if the pitch damper switch is in DAMPER and the auxiliary pitch trim switch is in STICK. The parallel pitch trim actuator can also be commanded to drive by one of the following means:

- By depressing the takeoff trim button which centers the parallel trim actuators at the takeoff position (zero degrees).
- By turning the pitch damper switch to OFF which centers and locks parallel trim at takeoff position (zero degrees).
- By moving the auxiliary pitch trim switch out of the STICK position which centers and locks parallel trim at takeoff position (zero degrees).
- By placing the pitch autopilot/damper switch to AUTOPILOT or placing the auto TF switch to AUTO TF, which centers and locks parallel trim at takeoff position (zero degrees).

The parallel trim actuator, when trimmed, will cause the control stick, pitch damper, and horizontal stabilizers to displace. The control stick will be centered and the pitch trim function of the stick trim button is disabled when pitch autopilot is selected, auto TF is selected, and during TFR fail safe flyup maneuvers.

Series Pitch Trim (Autotrim). The series pitch trim actuator drives the horizontal stabilizers but does not normally drive the control stick or pitch feel spring. The authority of the series pitch trim actuator is 10 degrees trailing edge up to 4 degrees trailing edge down. The actuator drives at a rate of 1.4 degrees per second. The actuator is locked at its existing position when the flight control system is switched to the takeoff and land configuration. (The takeoff and land configuration is established by either placing the flight control system switch to T.O. & LAND or by extension of the slats.) The actuator is also locked at its existing position if the pitch damper is turned off, but can then be driven from the stick trim button(s). While the flight control system is in its normal flight configuration, the pitch damper acts to maintain the aircraft normal acceleration ("g") at a value proportional to stick position, then the series trim drives in response to a pitch damper position signal to replace the pitch damper output. When the pitch damper is at zero (nulled), the series trim will stop driving. This is called the null mode of operation (autotrim). The null mode will be in effect unless the pitch damper is OFF or the flight control system is in the takeoff and land configuration.

Note

Series trim does not relieve the stick pressure required to maintain a given flight condition. Thus (assuming that parallel pitch trim has not been added) if stick pressure is released, the aircraft will return to one "g" flight.

Displacement of the control stick by force or by parallel trim will command an increase or decrease in "g" condition as initially provided by the pitch damper. The pitch series trim is then commanded by the pitch damper position transducer to null the pitch damper. If the elevator required to hold the aircraft at the command "g" varies due to changes in power, wingsweep, airspeed, etc., the damper will again displace to oppose aircraft variations from the commanded "g", and series trim will once again null the damper. Thus the series trim provides the steady state elevator required to maintain the aircraft at the stick-command "g" level. Because of this action, the stick will be centered for one "g" flight, regardless of wingsweep, airspeed, or gross weight (unless the control system is in the takeoff and land configuration). The series trim is driven to 3.8 degrees trailing edge up when the takeoff trim button is held depressed.

Note

During ground operation, series trim is not able to null the damper since there is no aircraft response. This will result in trim drift either nose up or nose down while the slats are retracted and the control system switch is in NORMAL.

Auxiliary Pitch Trim. The auxiliary pitch trim system is armed when the auxiliary pitch trim switch is placed to the center (OFF) position. When the switch is placed to the NOSE UP or NOSE DOWN position, pitch trim is provided by one or both of the following:

(1) Positioning an auxiliary pitch trim integrator in the feel and trim assembly, which sends command signals to the pitch damper.

(2) By directly driving the pitch series trim actuator.

The net effect is a change to the aircraft pitch trim. Figure 1-20 defines whether (1), (2), or both is used as a function of the flight control systems configuration.

Flight Control System Configuration vs Pitch Trim Operation

PITCH TRIM DURING MANUAL CONTROL MODES

FLIGHT CONTROL SYSTEM CONFIGURATION			*PITCH TRIM OPERATION*
Auxiliary Pitch Trim Switch	*Autopilot Damper Switch*	**Flight Configuration*	
STICK	DAMPER	Clean	1. Stick trim drives parallel pitch trim. 2. Series trim actuator in damper null mode.
STICK	DAMPER	Takeoff & Land	1. Stick trim drives parallel pitch trim. 2. Series trim locked at existing position.
STICK	OFF	Clean or Takeoff & Land	1. Parallel pitch trim actuator centers and locks at zero. 2. Stick trim drives series pitch trim actuator.
OFF	OFF	Clean or Takeoff & Land	1. Parallel pitch trim actuator centers and locks at zero. 2. Series trim actuator may be driven from the stick trim switch or the auxiliary pitch trim switch.
OFF	DAMPER	Clean	1. Parallel pitch trim actuator centers and locks at zero. 2. Series pitch trim actuator in damper null mode. 3. Auxiliary pitch trim switch drives damper.
OFF	DAMPER	Takeoff & Land	1. Parallel pitch trim actuator centers and locks at zero. 2. Auxiliary pitch trim switch drives damper and also drives the series pitch trim actuator.

*Flight Configuration:

Clean: Flight control system switch—NORM and slats up

T.O. & L: (1)Flight control system switch—NORM and slats-down or
(2) Flight control system switch—T.O. & LAND

PITCH TRIM DURING AUTOMATIC CONTROL MODES

MODE	*PITCH TRIM OPERATION*
AUTO TF	1. Stick trim deactivated and parallel pitch trim actuator centers. 2. Auxiliary pitch trim switch must not be used in this mode. 3. Series pitch trim actuator is driven by Auto TF climb/dive error.
PITCH AUTOPILOT MODES	1. Stick trim deactivated and parallel pitch trim actuator centers. 2. Auxiliary pitch trim switch must not be used in these modes. 3. Series pitch trim is driven by pitch autopilot error signal.

Figure 1-20.

WARNING

Improper operation will result if the auxiliary pitch trim switch is operated during auto TF or pitch autopilot modes.

The authority of the auxiliary pitch trim integrator command to the pitch damper will be a function of the pitch computer gain.

Stability Augmentation.

Pitch stability augmentation is provided through the redundant pitch damper system to provide aircraft damping and to improve the handling characteristics of the aircraft. (Refer to "Pitch Channel and Pitch Damper Redundancy," this section.) Figure 1-21 shows the basic mechanical system and the pitch damper system. The pitch computer provides damping signals to the pitch damper servo in response to normal acceleration and pitch rate feedback signals from the accelerometers and gyros. The stick position transducer signals are zero when the control stick is at neutral, thus providing a zero command signal to the pitch computer. Series trim will provide the steady state elevator to maintain the aircraft in trim; the normal accelerometer and pitch rate gyro signals will provide damping, and the aircraft will be in one "g" flight.

Command Augmentation.

The effectiveness of any control surface varies with the flight conditions. At low speed and high altitude, several degrees of elevator are required to command a one "g" maneuver; while at high speed and low altitude, it may take less than a degree. With the pitch damper off, stick force and surface movement are directly related to stick motion; thus heavy stick forces will be required at low speed, and light forces will be required during high speed, low altitude flight. With the pitch damper on, the stability augmentation feature, sensing pitch rate and normal acceleration, may oppose the initial aircraft response. However, when the stick is moved to command "g's", the stick input to the elevator is augmented by the pitch damper so that for practically any flight condition (high speed or low speed, at all altitudes) "g" response will be about the same for a given stick force. The command augmentation feature accomplishes this by adding elevator, if it is needed, to get the aircraft started on its pitch change, then as it approaches the commanded "g's," the damper and series trim will move the elevator so that aircraft response zeros out on the commanded "g." At low airspeed where control effectiveness is low, the damper will work to give the higher elevator deflection needed to reach commanded "g's."

Note

"G" command augmentation is not available when the flight control system is in the takeoff and land configuration.

As airspeed increases, less deflection is required and the system works the damper to result in less deflection for the same stick movement. Under steady state conditions, the series trim is driven by the out-of-neutral damper signal. This has two simultaneous effects: the series trim action provides the final input to achieve the commanded response and also nulls the damper, allowing full authority to be restored. Basically the command augmentation system changes the elevator deflection so that the aircraft response to stick inputs is the same regardless of airspeed or altitude.

WARNING

The command augmentation feature of the flight control system will attempt to maintain the stick-command level of pitch rate, "g" force, and roll rate independent of airspeed variations. For instance, during flight conditions where airspeed is decreasing, the horizontal stabilizer will be commanded to increase angle-of-attack, without additional pilot input, in an attempt to maintain the commanded level of pitch rate, roll rate, and "g" force. Failure to monitor and control angle-of-attack within limits can result in inadvertent rapid departure from controlled flight.

Automatic Pitch Control.

The pitch computer can also accept inputs from either the autopilot pitch submodes or the TFR computer to provide automatic pitch control through the pitch damper. Refer to figure 1-21. Interlocks are provided to prevent incompatible mode selection. Refer to "Autopilot System" and "Terrain Following Radar" this section.

Pitch Gain Control.

The gain of the command signal sent to the pitch damper servo is automatically varied as flight conditions change. This is accomplished by continuously monitoring the pitch rate gyro and normal acceleration signals to determine if the gain should be either increased or decreased. Since the system modifies its

Pitch Channel Electrical Schematic

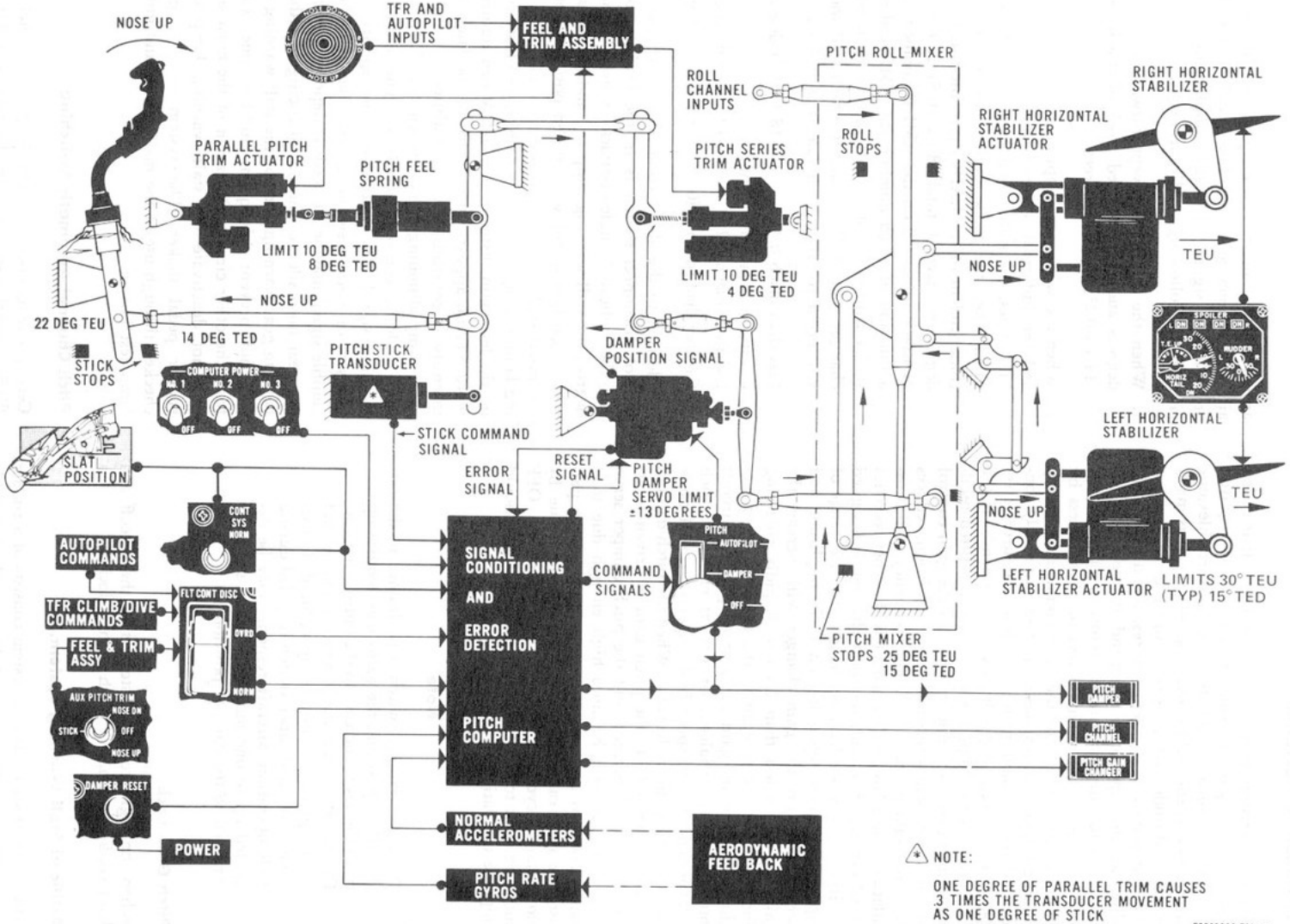


Figure 1-21.

gain as a function of its own performance, it is called a self-adaptive gain system. In general, the required gain varies inversely with dynamic pressure. Elevator effectiveness is affected by many variables such as mach number, altitude, wing sweep angle, gross weight, center-of-gravity, and external stores. The damper contributions in response to command augmentation inputs are a function of gain; therefore, a gain increase will compensate for reduced elevator effectiveness by giving more elevator for the same command, thereby holding the aircraft response and damping nearly constant. If the available gain is too low, the aircraft response will appear sluggish to stick commands. If the gain becomes too high, a small amplitude pitch oscillation may exist for a few cycles until the gain control circuit, which senses this oscillatory condition, reduces the gain to the proper value. The frequency of this oscillation (the adaptive frequency) will be between 1.4 to 3.0 cycles per second; the gain changer is designed to either increase or decrease the gain for this range of input frequencies depending on their amplitude and persistence. Also, the gain changer will increase the gain for inputs less than 1.4 cps. Rapidly changing flight conditions can result in the computed gain lagging the optimum gain for a short period. Aircraft motion due to turbulence or aircraft vibration, such as experienced with speed brake operation, will cause the adaptive gains to decrease. When the pitch damper switch is turned OFF, the pitch gain is driven to its minimum of 12 percent, and the pitch damper servo centers. If the gain becomes high enough due to a malfunction to cause the adaptive frequency to persist, the gain can be reset to its minimum value by momentarily cycling the pitch damper switch OFF and then back to the DAMPER position. This should stop the oscillation.

Note

When the damper switch is positioned to the DAMPER position, the gains may require up to 2 minutes to increase to the optimum value. During this time the aircraft response and damping may be noticeably reduced. Consequently if the damper switch is cycled during TFR operation, aircraft response may be degraded up to one minute after the switch is returned to the damper position.

Pitch Gain Lock.

When the flight control system is in the takeoff and land configuration, the pitch gain is locked.

Artificial Stall Warning System.

The artificial stall warning system consists of a rudder pedal shaker, a stall warning lamp, and an audible stall warning signal. The system is automatically armed by the landing gear squat switch when the aircraft becomes airborne. The pedal shaker, lamp, and audible signal all occur simultaneously when either of the following conditions exists:

1. When the wings are swept forward of 50 (±2) degrees and the indicated angle-of-attack exceeds 14 (+0.25, −0.75) degrees.
2. When the wings are swept aft of 50 (±2) degrees and the indicated angle-of-attack is greater than 14 degrees, the stall warning system will be activated when the probe angle-of-attack, in degrees, (independent of the CADC) plus the pitch rate, in degrees per second, total 18 (±1). Since the angle-of-attack presented on the AMI is compensated as a function of mach number, the AMI reading for stall warning activation will vary as mach number changes. The AMI reading at which stall warning will occur for zero pitch rate is as follows:

Less than mach 0.30	18 (±1.6) degrees
Greater than mach 0.45 but less than mach 1.25	19.7 (±1.6) degrees
Greater than mach 1.40	18.8 (±1.6) degrees

In addition to the above conditions, when the flight control disconnect switch is in the OVRD position and the indicated angle-of-attack is greater than 14 degrees the stall warning lamp and audible signal will occur regardless of the wing sweep position, however the pedal shaker will be inoperative. The stall warning lamp (22, figure 1-6) is a flashing red lamp located on the left main instrument panel. When lighted, the word STALL appears on the face of the lamp. Lamp intensity is controlled by the malfunction and indicator lamp dimming switch when the flight instrument lighting control knob is on. The audible stall warning signal is a continuous tone applied to the headsets of both crew members. The stall warning audible signal may be silenced by depressing the landing gear horn silencer button. Silencing of either the landing gear warning horn or the stall warning signal will not prevent subsequent audible tone warning from the other circuit. Operation of the horn silencer will not deactivate the stall warning lamp or the rudder pedal shaker. The system may be ground checked through use of the malfunction and indicator lamp test button.

Pitch Channel Automatic Switching.

Certain changes occur within the pitch channel in the flight control system during the takeoff and landing phase. Automatic switching is accomplished by the

slat position and the weight on gear signals. The following describes the configuration for four different conditions:

1. Taxi—Weight on gear, slats down, pitch damper switch—DAMPER.
   a. Series trim is locked at its existing position but may be set to 3.8 degrees trailing edge up by the takeoff trim button.
   b. Parallel trim can be driven from the stick trim button(s) if auxiliary pitch trim switch is in STICK. When the stick trim button is depressed NOSE UP, the parallel trim actuator will drive the horizontal stabilizers, the control stick, the stick transducer, and the pitch damper in the nose up direction. The opposite will occur for nose down trim.
   c. Pitch gains are locked.
   d. The stall warning device is not armed.
   e. Normal accelerometer signals are locked out.
2. Lift off—Slats down, weight off gear, pitch damper switch — DAMPER, auxiliary trim switch — STICK. Takeoff trim set prior to takeoff.
   a. Series trim is locked at 3.8 degrees trailing edge up.
   b. Parallel trim can be driven from the control stick trim button(s) to trim the aircraft as required.
   c. The stall warning device is armed.
   d. Normal accelerometer signals are locked out.
3. Cruise—Slats retracted, gear up, pitch damper switch — DAMPER.
   a. Series trim actuator is in the null mode.
   b. Pitch gain is in its adaptive mode and automatically increases or decreases the gain of the pitch damper command signal as flight conditions change.
   c. Parallel trim should be used to trim the aircraft to 1 "g" flight. The stick will be at neutral when this is accomplished.
   d. The stall warning device is armed.
   e. Normal accelerometer signals are active.
4. Slat Extension Prior to Landing
   a. Series trim actuator locks at its existing position.
   b. Parallel trim can be used to reestablish trimmed flight.
   c. The stall warning device is armed.
   d. Normal accelerometer signals are locked out.

Pitch Channel and Pitch Damper Redundancy.

All stability augmentation signals enter the pitch computer where they are converted into three command signals. Computer inputs are supplied from three separate sources to three identical pitch computer boards. Each board separately computes a pitch gain and a pitch command signal. The three separately computed command signals are inputs to signal selectors 1, 2, and 3. Each signal selector rejects the highest signal and the lowest signal. The output of each selector is the remaining signal, or the middle value signal. Each board has a signal comparator circuit, which compares that board's input signal with the selector's output signal. Should these signals be significantly different, the pitch channel caution lamp will light. For an initial failure, which causes a pitch channel caution lamp to light, the operation of the pitch damper system will be unaffected. However, a subsequent failure of another branch could result in (a) no change in operation, (b) zero pitch damper commands, or (c) a hard over pitch damper. The operation depends on the nature of the first and subsequent failure(s). If the first failure resulted in a zero command from the affected board, normal operation can be continued and the pitch channel caution lamp can be reset. If the lamp cannot be reset, this means that the first failure is a hard over failure, and a second failure in the same direction will cause a hard over damper. For this reason, it is recommended that flight conditions be changed to observe the damper off envelope and then turn the pitch damper OFF. When the pitch damper is turned OFF, the pitch damper caution lamp will light.

Signal Selection.

The output of each signal selector will be the same for a single malfunction. Downstream of the signal selectors are three servo amplifiers, each of which receives its signal selector's output and in-turn sends a command signal to its separate servo valve within the pitch damper servo. When the pitch damper switch is turned OFF, the amplifier currents to the damper servo go to zero and the damper is hydraulically centered.

Damper Hydraulic Logic.

The pitch damper has two active valves and one model valve. The average of the command signals from the two active valves hydraulically control the movement of the damper output rod. A third servo valve controls a model servo and does not control the damper rod output. The position of the damper output rod and the position of the model servo are compared to detect malfunctions. Should a malfunction exist, the damper output rod position will not agree with the model servo position and the pitch damper caution lamp will light. Hydraulic logic within the pitch damper servo will identify the discrepant servo valve command. If the failure is due to one of the two active valves controlling the damper output rod, a vote will occur and the discrepant valve is hydraulically shut off. Control of the output rod is then dependent upon the command from the remaining active servo valve. A transient will

be felt in the aircraft when this vote occurs. The discrepant valve can be placed into operation by depressing the damper reset button. Should the valve still have a discrepant output command, another vote will occur and the valve will again be shut off. Should the discrepant valve be the model valve, the pitch damper caution lamp will light, but the pitch damper rod will not be affected, and a transient will not be felt.

Hydraulic Servos.

The two active servo valves which control the damper output rod are supplied from separate hydraulic systems. In the event of a single hydraulic system failure, the valve controlled by the failed hydraulic system will be hydraulically shut off, and the pitch damper caution lamp will light. Normal damper operation will continue by using the remaining active servo valve and hydraulic system. In this case, the damper reset button has no effect unless hydraulic pressure returns to normal.

Pitch Gain Changer Redundancy.

Within the pitch computer, three separate circuits compute the required gain. These three outputs are processed through their gain selectors in the same manner described above for the signal selectors. The output of the gain selectors is the middle value gain. The gain of each damper command signal is varied according to that computed by the middle value gain control circuit. Should a discrepancy exist between the three separate gain calculations, a pitch gain changer caution lamp will light, indicating failure of one of the three circuits. Operation will be unaffected until a second failure occurs. If the gain changer light can not be reset, a second failure could cause the total gain to go to minimum, maximum, or be unaffected. For this reason, a decrease in airspeed is recommended.

ROLL CHANNEL.

Mechanical Linkage.

Lateral movement of the control stick is transmitted to the pitch roll mixer assembly by a system of push-pull rods and bellcranks. The pitch roll mixer adds the roll commands to the pitch commands and sends summed commands to the left and right horizontal stabilizer control valves and actuators. Figure 1-22 shows a simplified mechanical schematic of the roll channel linkage and damper system. Stick centering and stick feel forces are provided by the roll feel assembly. In this assembly, two feel springs are provided. When compressed, the low gradient feel spring provides nonlinear stick forces until its limit is reached at one-half stick travel. One-half stick displacement commands a 2-degree roll displacement through the pitch/roll mixer to each of the horizontal stabilizers with a stick force detent being encountered at this point. Approximately eight additional pounds must be exerted on the control stick before additional travel can be achieved. The high gradient feel spring breakout creates this force detent and provides the stick force gradient until maximum stick deflection is achieved. Maximum stick deflection commands 8 degrees of roll displacement through the pitch/roll mixer to each of the horizontal stabilizers. With the roll damper off, the available stick deflection is limited by stick stops located within the cockpit to ±8 degrees of mechanical command. Figure 1-22 shows the gearing, and approximate stick force provided by the roll feel assembly. Stick breakout force is approximately 1.2 pounds. A 15-pound force is required to reach the detent; 31 pounds will give maximum stick deflection. The roll damper servo also provides roll control inputs to the pitch roll mixer. The roll damper is a redundant electrohydraulic servo actuator with an authority of ± 6 degrees. The roll damper servo is used to provide roll trim, stability augmentation, and command augmentation to the horizontal stabilizers. The roll damper servo is identical to the pitch and yaw damper servos.

Power Supply.

The electrical power to the roll damper system is provided through the three computer power switches located on the ground check panel. Electrical power to the spoiler system is provided through the computer power switches, numbers 1 and 2. When these switches are OFF, spoiler operation for roll control is not available. When the roll damper is off, the roll damper servo actuator is hydraulically driven to its neutral position. When hydraulic pressure is not available, the position of the roll damper servo is indeterminate. A stick position transducer, located in the roll feel assembly sends nonlinear command signals to the roll damper and to the spoiler actuators.

Spoiler Operation.

When the wings are forward of 45 degrees, roll control is aided by action of two spoilers on the top of each wing. Each spoiler surface is actuated by a hydraulic servo actuator. The outboard pair of spoiler actuators has extension pressure supplied by the utility hydraulic system and has lock down pressure supplied by the primary hydraulic system. The inboard pair of spoiler actuators receives extension pressure from the primary hydraulic system and lock down pressure from the utility hydraulic system. Lateral movement of the control stick causes the stick position transducers to generate electrical command signals which are sent through the feel and trim and the wing sweep sensor assembly to the spoiler actuators. There is no mechanical linkage between the stick and the spoiler. Both commanded spoilers extend to a maximum of 45 degrees at the stick force detent. The spoiler extension versus stick displacement is nonlinear.

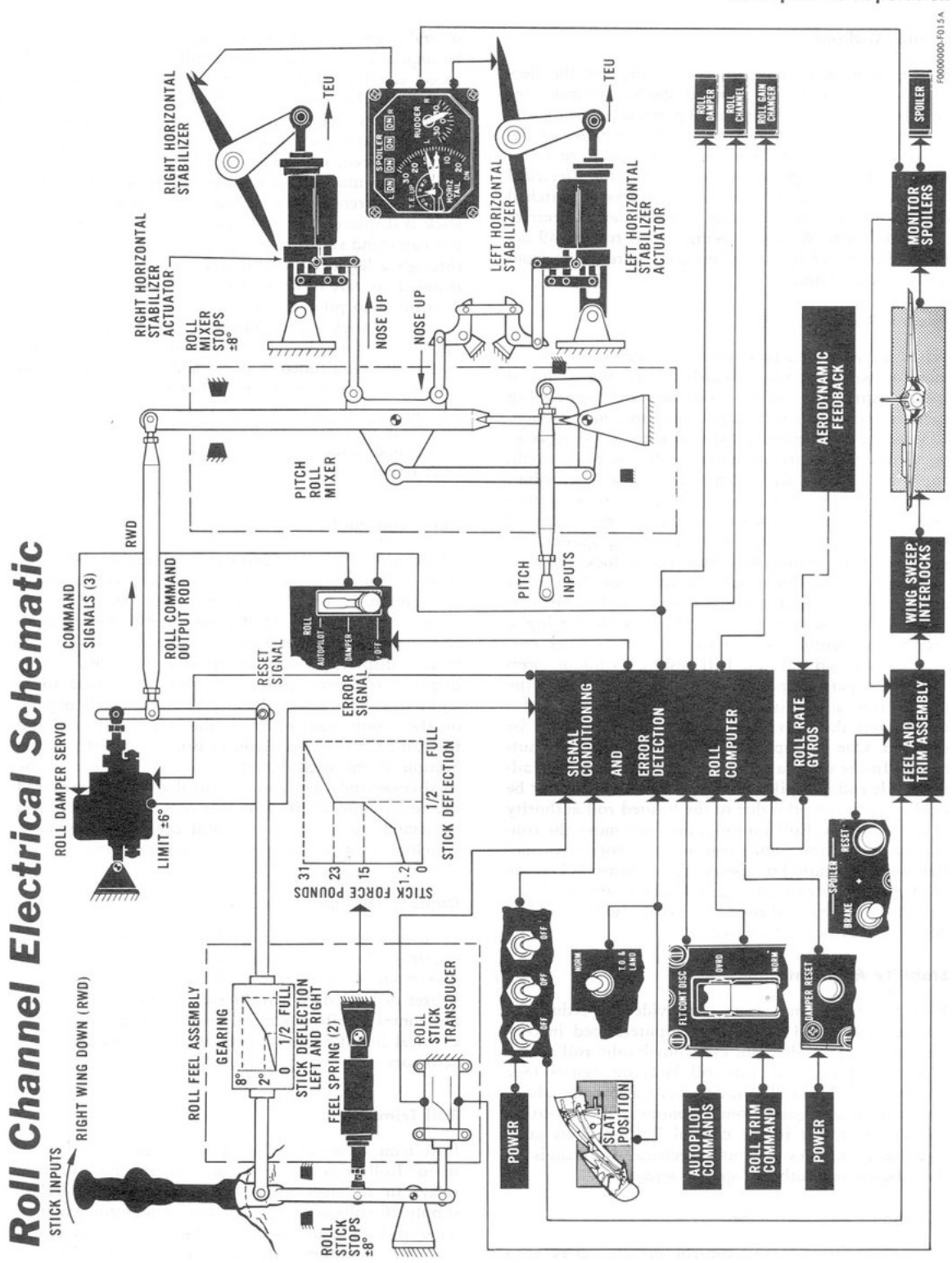

Roll Channel Electrical Schematic
STICK INPUTS
RIGHT WING DOWN (RWD)
ROLL FEEL ASSEMBLY
GEARING
8°
2°
0 1/2 FULL
STICK DEFLECTION LEFT AND RIGHT
FEEL SPRING (2)
ROLL STICK TRANSDUCER
ROLL STICK STOPS ±8°
ROLL DAMPER SERVO
LIMIT ±6°
ROLL COMMAND OUTPUT ROD
RWD
COMMAND SIGNALS (3)
RESET SIGNAL
ERROR SIGNAL
ROLL AUTOPILOT DAMPER OFF
STICK FORCE POUNDS
31
23
15
1.2
0
1/2
FULL
STICK DEFLECTION
PITCH ROLL MIXER
PITCH INPUTS
ROLL MIXER STOPS ±8°
NOSE UP
NOSE UP
RIGHT HORIZONTAL STABILIZER ACTUATOR
RIGHT HORIZONTAL STABILIZER
TEU
LEFT HORIZONTAL STABILIZER
LEFT HORIZONTAL STABILIZER ACTUATOR
TEU
SPOILER ON ON ON
RUDDER
HORIZ TAIL
ROLL DAMPER
ROLL CHANNEL
ROLL GAIN CHANGER
SPOILER
MONITOR SPOILERS
AERODYNAMIC FEEDBACK
WING SWEEP INTERLOCKS
SIGNAL CONDITIONING AND ERROR DETECTION ROLL COMPUTER
ROLL RATE GYROS
FEEL AND TRIM ASSEMBLY
SPOILER RESET
BRAKE
POWER
NORM
T.O. & LAND
SLAT POSITION
FLT CONT DISC
GYRO
NORM
AUTOPILOT COMMANDS
ROLL TRIM COMMAND
DAMPER RESET
POWER
F0000000-F015A

Figure 1-22.

Spoiler Lockout.

When the wing sweep angle is at 45 degrees, the electrical commands to the inboard spoiler actuators are switched out by the wing sweep sensor, causing the inboard spoilers to retract and lock down. At 47 degrees wing sweep, primary hydraulic pressure is removed from all spoiler actuators, and the electrical command signal to the outboard spoilers is switched out by the wing sweep sensor, causing them to retract and lock down. When the wing sweep reaches 49 degrees, the utility hydraulic pressure is removed from all spoiler actuators.

Spoiler Monitor.

When the wings are forward of 45 degrees, the spoiler monitor will lock the inboard pair or the outboard spoiler pair down should both left and right wing spoilers extend above 15 degrees due to a malfunction. If a spoiler inadvertently extends without being commanded and the aircraft starts a roll, the pilot would apply an opposite stick command to maintain wings level. Extension of spoilers on the opposite wing will cause the monitor, through a voting process, to cut off hydraulic pressure to the malfunctioning spoiler and its mate. This action will retract and lock the pair of spoilers in the down position and cause the spoiler caution lamp to light and result in reduced roll power. The spoiler monitor may be reset by depressing a spoiler reset button. This will cause the spoiler caution lamp to go out and will restore hydraulic pressure to the pair of spoilers that is locked down. If the malfunction still exists, the faulty spoiler will again extend and the previous sequence of events will be repeated. One attempt to reset a faulty spoiler is sufficient. In the event a spoiler extends because of a failure while roll autopilot is engaged, the wings must be held level by the pilot due to the limited roll authority of the autopilot. Roll autopilot does not move the control stick, and the pilot's control stick corrective motion will be required to operate the monitor. When the pilot moves the control stick to hold wings level, the monitor will vote and the failed spoiler will be locked down as previously described.

Stability Augmentation.

Roll stability augmentation is provided by redundant roll rate gyros and electronic computers used in conjunction with a redundant electrohydraulic roll damper servo to provide aircraft roll damping signals. (See figure 1-22.) The roll damper servo also responds to stick command signals from a nonlinear stick position transducer located in the control linkage. This same transducer supplies separate electrical commands to the inboard and outboard spoiler actuators.

Command Augmentation.

The effectiveness of any control surface varies with the flight conditions. At low speed and high altitude, several degrees of differential horizontal stabilizer may be required to achieve a given roll rate; while at high speed and low altitude, it may take less than a degree. Also, the roll rate may be significantly increased when spoilers are operable. With the roll damper off, horizontal stabilizer displacement is proportional to stick displacement. With the roll damper on, the stability augmentation input causes roll damper commands to oppose aircraft roll rate. However, when the control stick is displaced, the roll damper also receives an input command signal from the stick position transducer through a lag circuit. This signal represents the commanded aircraft response and reduces the roll damper opposition to pilot initiated maneuvers and augments the pilot's stick input. The steady state roll damper displacement will be proportional to the difference between the commanded response and the actual aircraft response and is directly proportional to the roll adaptive gain. The horizontal stabilizer surface displacement due to control stick inputs will then vary with flight conditions so that variations in the resulting aircraft response will be minimized.

Roll Commands.

The control stick transducer output reaches the maximum at the stick force detent and represents a roll rate command of 160 degrees per second. The roll damper authority is 12 degrees of differential horizontal stabilizer deflection; i.e., for a left roll the left surface displaces 6 degrees up and the right surface displaces 6 degrees down; the opposite occurs for a right roll. The actual damper deflection will depend on the commanded roll rate, the actual roll rate, and the roll adaptive gain. Refer to Roll Gain Control, this Section. If the stick deflection exceeds the detent, the total command (mechanical plus damper) may exceed the roll command limit. If this occurs, the excess roll command from the damper will cause stick talkback which may appear as pitch or roll stick movements.

Damper Off Operation.

When the roll damper is OFF, full stick deflection requires a force of 31 pounds. This will command the maximum of 16 degrees of differential horizontal stabilizer deflection. When the roll damper is turned off, horizontal stabilizer control reverts to the direct mechanical linkage command to the horizontal stabilizer actuators.

Roll Trim.

Roll trim is accomplished through the roll damper servo. Roll trim command signals operate roll trim relays in the feel and trim assembly. These relays supply 26 volts ac to the roll trim integrator which is a motor driving an electrical transducer. This electrical transducer supplies a roll rate command signal to the roll computer which causes the roll damper servo to

position the horizontal stabilizer. Since the output of the roll damper servo is in series with the roll channel linkage, the control stick does not move as trim is applied. Roll trim is controlled by trim buttons located on each control stick. Approximately eight seconds is required to insert the maximum roll trim command. Roll trim is inoperative when the flight control disconnect switch is placed to OVRD or the roll damper switch is positioned to OFF. Under these conditions, any previous roll trim inputs will be removed. The pilot can either hold stick force, or hold the wings level with yaw trim and accept the accompanying side slip when roll trim is not available.

Roll Gain Control.

The gain of the command signal sent to the roll damper servo is automatically varied as flight conditions change. This is accomplished by continuously monitoring the roll rate gyro signals to determine if the gain should be either increased or decreased. Since the system modifies its gain as a function of its own performance, it is called a self-adaptive gain system. In general, the required gain varies inversely with dynamic pressure. A gain increase will compensate for reduced horizontal stabilizer effectiveness. If the gain is too low, the aircraft will appear sluggish to lateral stick commands. If the gain becomes too high, a small amplitude roll oscillation may exist for a few cycles until the gain control circuit, which senses this oscillatory condition, reduces the gain to the proper value. The frequency of this oscillation (the adaptive frequency) will be between 1.4 to 3.0 cycles per second. The gain changer is designed to either increase or decrease the gain for this range of frequencies depending on the amplitude and persistence. The gain will be increased for frequencies of less than 1.4 cps. Rapidly changing flight conditions can result in the computed gain lagging the optimum gain for a short period. Aircraft motion due to turbulence or aircraft vibration, such as experienced with speed brake operation, will cause the adaptive gains to decrease. When the roll damper switch is turned OFF, the roll gain is driven to its minimum value of 20 percent, and the roll damper servo centers. If the gain becomes high enough, due to a malfunction, to cause the adaptive frequency to persist, resulting in a small amplitude roll oscillation, the gain can be reset to its minimum value by momentarily cycling the roll damper switch OFF and then back to the DAMPER position. This should stop the oscillation. When the damper is turned on, the gains may require up to 2 minutes to increase to the optimum value. During this time the aircraft response and damping may be noticeably reduced. When in the takeoff and land configuration, the roll gain is locked at maximum.

Roll Channel and Roll Damper Redundancy.

The stick position transducer, roll rate gyros, electronic computers, gain control circuits, and the roll damper servo are redundant to the same extent as described under "Pitch Channel and Pitch Damper Redundancy," this section. An electronic malfunction causes either the roll channel or roll gain changer caution lamps to light. A damper malfunction will cause the roll damper caution lamp to light. The roll damper servo actuator is identical to the pitch and yaw damper servo actuator. Refer to "Spoiler Operation," this section, for the redundancy features incorporated into the spoiler system.

YAW CHANNEL.

Manual control of the aircraft in the yaw axis is achieved by left or right pedal motion on either of two sets of rudder pedals. This movement is transmitted by cables, push-pull tubes, and bell cranks to the rudder control valve and rudder actuator. The control valve, located on the rudder actuator, controls the flow of hydraulic fluid from both the primary and utility hydraulic systems to the rudder actuator. Figure 1-23 is a simplified yaw channel mechanical and electrical schematic. Rudder pedal breakout (12 pounds) and centering is provided by the yaw feel spring. Rudder feel forces (80 pounds for available pedal travel) are provided by the yaw feel spring and the yaw variable feel actuator. The yaw variable feel actuator provides two rudder authorities: full pedal travel of approximately 2.5 inches and ± 30 degrees of rudder deflection or limited authority of approximately 1-inch pedal travel and ± 11.25 degrees of rudder deflection. The available authority depends on the slat position, the flight control system switch position, and the rudder authority switch position. A rudder authority caution lamp will light when available authority does not agree with the authority programmed by the slat position and the flight control system switch position.

Yaw Damping.

Aircraft damping is provided by the yaw damper servo. The yaw damper command and the yaw trim actuator command are summed with the rudder pedal command in the yaw feel assembly and then applied by mechanical linkage to the control valve of the rudder actuator. The authority of the yaw damper servo is ± 15 degrees.

Rudder Limits.

Pedal stops are provided in the cockpit to limit rudder pedal travel. These stops allow ± 30.75 degrees of rudder command. The total command to the rudder control valve is limited by the aft stop, which is set at 30 degrees (nominal).

Pedal Shaker.

A pedal shaker is attached to the pilot's left rudder pedal to provide stall warning. Refer to "Artificial Stall Warning System," this section.

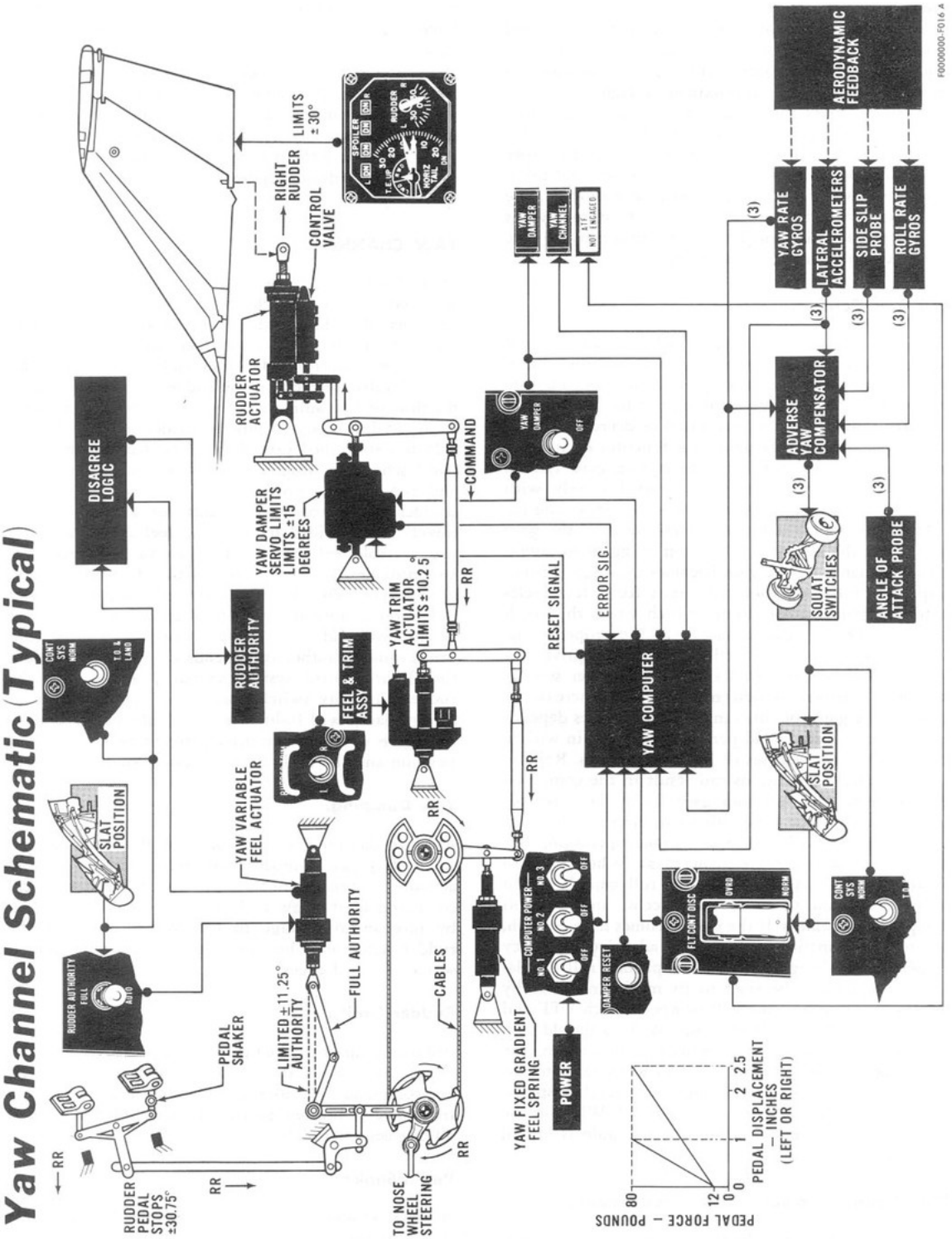

Yaw Channel Schematic (Typical)
F0000000-FD16-A
AERODYNAMIC FEEDBACK
YAW RATE GYROS
LATERAL ACCELEROMETERS
SIDE SLIP PROBE
ROLL RATE GYROS
(3)
YAW DAMPER
YAW CHANNEL
ATT NOT ENGAGED
LIMITS ± 30°
RIGHT RUDDER
CONTROL VALVE
RUDDER ACTUATOR
SPOILER
RUDDER
DISAGREE LOGIC
ADVERSE YAW COMPENSATOR
COMMAND
YAW DAMPER SERVO LIMITS ±15 DEGREES
YAW TRIM ACTUATOR LIMITS ±10.25°
RR
RESET SIGNAL
ERROR SIG.
SQUAT SWITCHES
ANGLE OF ATTACK PROBE
CONT SYS NORM
T.O. & LAND
RUDDER AUTHORITY
FEEL & TRIM ASSY
YAW COMPUTER
SLAT POSITION
YAW VARIABLE FEEL ACTUATOR
RUDDER AUTHORITY FULL AUTO
COMPUTER POWER — NO. 1 NO. 2 NO. 3 OFF
FLT CONT DISC
GYRO
NORM
DAMPER RESET
PEDAL SHAKER
LIMITED ±11.25° AUTHORITY
FULL AUTHORITY
CABLES
YAW FIXED GRADIENT FEEL SPRING
POWER
RUDDER PEDAL STOPS ±30.75°
TO NOSE WHEEL STEERING
PEDAL FORCE — POUNDS
80
12
0
0
1
2
2.5
PEDAL DISPLACEMENT — INCHES (LEFT OR RIGHT)

Figure 1-23.

Power Supply.

Power to the yaw damper servo is controlled from the computer power switches on the ground check panel.

Damper Off Operation.

The yaw damper is controlled from the yaw damper switch located on the autopilot/damper panel. When the yaw damper switch is turned to OFF, the yaw damper servo will hydraulically center, and the yaw damper caution lamp will light. When the yaw damper switch is returned to the DAMPER position, the yaw damper caution lamp will go out, and yaw damping will be immediately provided. If hydraulic pressure is not available, the position of the yaw damper servo is indeterminate.

Yaw Trim.

Yaw trim is accomplished by an electrically driven actuator which mechanically positions the rudder linkage. Since the yaw trim actuator is downstream of the yaw feel spring, there is no movement of the rudder pedals as trim is applied. Yaw trim is controlled by a rudder trim switch, located on the auxiliary flight control panel. Trim authority is ±10.25 degrees of rudder and the trim rate is 2.3 degrees per second.

Yaw Stability Augmentation.

Yaw stability augmentation is provided through a fixed gain redundant yaw damper system to provide continuous aircraft damping. As in the case of the pitch and roll damper system, automatic failure detection and rejection is provided. Refer to "Yaw Channel and Yaw Damper Redundancy," this section. Lateral acceleration and yaw rate signals are used as inputs to the yaw damping system. Yaw rate input is decreased as a function of time (washed out) to prevent the damper from opposing steady state pilot commands.

Adverse Yaw Compensation (AYC).

AYC is provided to improve turn coordination in the takeoff and landing configurations. This function is provided automatically whenever the slats are extended 70% or more. Turn coordination is improved by augmenting the directional stability with a sideslip to rudder feedback. This latter signal produces rudder in the direction of the roll and proportional to the roll rate. The roll rate feedback is gain adjusted as a function of angle-of-attack. AYC may also be obtained by placing the flight control system switch to the T.O. & LAND position. This should only be done below 295 KIAS or mach 0.62, whichever is less. The main gear squat switch deactivates AYC when weight is on the main gear. AYC can be switched out by placing the flight control disconnect switch in the OVRD position.

WARNING

Attempting abrupt rolling maneuvers or bank angles in excess of 60 degrees with the flight control system in the takeoff and land configuration by extension of the slats or by placing the flight control system switch in T.O. & LAND can result in loss of control of the aircraft.

Yaw Channel Automatic Switching.

The yaw damper system uses fixed gains; however, certain switching configurations occur as shown below for a typical takeoff and land cycle.

1. Pre-taxi—Flight control system switch — NORM, slats are retracted, weight on gear, yaw damper switch—DAMPER.
   a. Stability augmentation signal inputs are lateral accelerometer signals and washed out yaw rate gyro signals.

   Note

   If the aircraft is not on level ground, the rudder may be displaced slightly due to the lateral accelerometer input into the yaw damper system.

   b. AYC is inactive.
   c. Rudder authority is limited, and full nose wheel steering is not available.
2. Taxi—Flight control system switch—NORM, slats extended, weight on gear, yaw damper switch — DAMPER.
   a. Stability augmentation inputs are fixed gain yaw rate gyro inputs, and lateral accelerometer inputs. Yaw damper displacement due to unlevel runways may be larger in this mode, compared to the retracted slat configuration because the yaw rate signal is not washed out.
   b. AYC is inactive.
   c. Rudder authority is full, and full nose wheel steering is available.
3. Lift off—Same as taxi, except weight is not on the gear.
   a. Stability augmentation inputs are lateral accelerometer signals and washed out yaw rate signals.

b. AYC inputs are accepted by the yaw damper system to improve turn coordination (AYC inputs are sideslip and adjusted roll rate inputs).
c. Rudder authority is full.

4. Slat Retraction—Gear up, yaw damper switch — DAMPER, Flight control system switch—NORM.
   a. Stability augmentation inputs (yaw channel) are lateral acceleration and washed out yaw rate signals.
   b. AYC signals are switched out when slat extension is less than 70 percent.
   c. Rudder authority becomes limited when slat extension is less than 2 percent.

Yaw Channel and Yaw Damper Redundancy.

The yaw damping system utilizes redundant inputs for computing the required damping signals and uses signal selectors in the same manner as described under "Pitch Channel and Pitch Damper Redundancy," this Section. The yaw damper servo is a redundant unit and is identical to the pitch damper servo. Electronic failure causes the yaw channel caution lamp to light. A yaw damper malfunction will cause the yaw damper caution lamp to light.

Adverse Yaw Compensation (AYC) Failures.

When weight is off the main gear and the slats are extended or the flight control system switch is in T.O. & LAND, the yaw damper system receives additional inputs to provide improved turn coordination. These additional inputs are referred to as AYC. A major input to AYC is the sideslip probe. The probe is located along the aircraft centerline just forward of the nose gear. Two slots in the probe sense air pressure. A difference in pressure causes the probe to align itself to the air flow. Should one or both slots become plugged, or should the probe not be free to rotate, erroneous sideslip information will be fed to the AYC circuits which can cause significant rudder inputs during the take off or landing phase of flight. An identical probe is mounted on the left hand forward equipment bay door to sense angle of attack. For this reason, "Exterior Inspection" in Section II requires a check of both probes for cleanliness and freedom of motion.

AYC failures may result from a sideslip probe problem or electronic component failures. An erroneous sideslip probe position may cause up to 15 degrees of rudder displacement when AYC is operable (airborne and slats extended or airborne and the flight control system switch in T.O. & LAND). The electronics for AYC consists of three identical branches. Should one branch fail hard over, it would not be detected during ground operation because of the landing gear ground safety (squat) switch. At lift off, the yaw channel caution lamp would light, but no unwanted rudder motion would occur. Should an attempt be made to reset the light while the slats were extended, it would not reset. The lamp could be reset after the slats were retracted. When the slats were extended for landing, the yaw channel lamp would again light, and it could not be reset until after weight was on the gear or the slats were retracted.

WARNING

At lift off and/or airborne slat extension, it is possible to experience full AYC authority rudder deflection (15 degrees) with redundant circuit failures. No caution lamps will light until the rudder deflection occurs. AYC can be terminated by placing the flight control disconnect switch to OVRD.

Small rudder transients, which may occur at lift off and upon slat extension prior to landing, result from improper alignment of the sideslip probe housing. Since AYC is not provided when weight is on the main gear, AYC cannot be adequately checked during the pilot's pretaxi checks. Rudder displacement and a yaw channel light may occur during the stall warning system check in Section II. This is normal and is due to removal of one of the two weight-on-gear signals to AYC during this check.

Procedures for countering an unscheduled rudder input due to AYC are outlined under "Unscheduled Roll/Yaw Maneuver," Section III.

FLIGHT CONTROL SYSTEM OPERATION CHARACTERISTICS.

Pitch Trimming During Maneuvers.

When parallel pitch trim is used to trim-in a pitch command of one degree, the output of the pitch stick transducer will be only 30 percent of the value that would have been present had the stick been held in the same position. Thus the magnitude of the command signal to the pitch damper will be less when trim is used than when force is held for a given stick position. This means that for a constant altitude bank, the control stick must be trimmed to a farther aft position than would be required if force is used to hold the same maneuver.

Roll Inputs.

During ground operation, lateral stick inputs with the roll damper on will exhibit the following characteristics which are normal.

- The control stick cannot be held past the force detent. When the control stick is returned to neutral after a hard over deflection, the horizontal stabilizers will immediately change from 8 degrees of roll to 6 degrees, and after a small delay will return to zero roll. This delay is normal and is caused by the saturation of the roll damper system from the stick transducer.
- Rapid lateral stick motions with the roll damper on may result in a small bump being felt in the control stick and a momentary reduction in the affected spoiler position. This effect will be more pronounced at low engine rpm and is caused by rate limiting of the system.
- When roll trim is applied, the roll damper responds and moves the horizontal stabilizers. If the trim command switch is held several seconds, it will saturate the roll damper. When roll trim is then driven in the opposite direction, by either of the stick trim switches or by depressing takeoff trim, the stabilizers may take several seconds to begin to respond. This delay is normal, and represents the time required to bring the trim input below the roll damper saturation limits.

Stick Talk Back.

A condition known as stick talk back may be experienced whenever the pitch/roll mixer output to both or either horizontal stabilizer actuators is unable to respond to or lags the pitch/roll mixer input from damper servo or trim commands. Refer to figure 1-19. When the pitch and roll dampers are on, the stick plus damper command may exceed the mixer limit as set by the mixer linkage stops. The damper response may also lag the stick command for large/rapid control stick inputs. Once the mixer authority is reached, additional damper or trim input will result in feedback to the control stick. This can be felt during ground operation by making large/rapid nose down control stick inputs. Control stick talk back can also be experienced whenever the rate of pitch or roll command into the mixer assembly exceeds the rate at which the horizontal stabilizer actuators can respond. The output rods from the pitch/roll mixer assembly control the left and right horizontal stabilizer actuators (refer to figure 1-19). Each actuator has its own control valve and feedback bellcrank. A nose up command displaces the control valve input rod aft; this ports fluid from both primary and utility hydraulic systems to each actuator and drives the horizontal stabilizers. When the feedback bellcrank has been repositioned by the actuator to a point where the control valve input is again at a null, the horizontal stabilizer actuator will stop. The stroke of each control valve is limited within the valve housing. This limit may be reached whenever the rate demand of the horizontal stabilizer actuator is greater than the maximum capability of the actuator. Should the limit of the valve stroke be reached, the mixer output rod will momentarily stop and will not respond to further pitch or roll inputs until the commanded rate can be satisfied by the actuator. During this period, a momentary control stick pulse will be felt. This may be noticed during ground operations ("Surface Motion Check") with programmed step inputs to the pitch and roll dampers or with rapid lateral stick inputs. The horizontal stabilizer actuators are rate limited to 36 degrees per second.

PRE-TAXI CHECKS.

Control System Movement Check.

The control stick and rudder pedals are checked for freedom of movement while all dampers are off. During this check the pitch, roll, and yaw damper caution lamps should be on. The pitch, roll, and yaw channel lamps should not come on during this check. The mechanical linkage is checked for freedom and authority. If takeoff trim is first set, full stick deflection in pitch should cause a horizontal stabilizer indication of 10 degrees trailing edge down and 25 degrees trailing edge up. Roll stick displacement should cause a difference between left and right stabilizers of 4 degrees at detent and 16 degrees at full lateral stick displacement. Rudder control will be either full or limited, depending on the slat position and rudder authority switch position. Fore and aft motion of the control stick with the pitch damper on will result in stick talk back, which is normal. Lateral stick motion with the roll damper on may result in stick talk back, dependent upon the rate of application. The control stick cannot be held past the detent for this configuration. If lateral command inputs are made while maintaining large pitch inputs on the stick, stick talk back can be expected in the pitch direction.

Stability Augmentation Test Operation.

The purpose of the surface motion check is to ensure that all three damper systems will respond to three equal input signals by operating the appropriate damper servo without any malfunction. The purpose of the surface motion and lights check is to ensure that the error detecting system will detect the loss of one of the redundant branches in each channel and light the appropriate caution lamps. With all dampers on, all caution lamps out, flight control configuration normal, after takeoff trim is set, the stick is trimmed nose up momentarily to establish a nose up command to the pitch series trim actuator. Pitch series trim action is checked by verifying that both stabilizers drive trailing edge up until the trim authority limit is reached. After the stabilizers stop driving, the auto TF switch is placed to the AUTO TF position. This action causes the control stick to center, and the reference not engage caution lamp and TF flyup off caution lamp to light. This switch action also effectively prevents the

series trim actuator from driving while the surface motion test is in progress; however, stabilizer drifting may occur when this test is not in progress.

Surface Motion Check.

The master test button is depressed and the stability augmentation switch is held in the SURFACE MOTION position. This action causes all gyros in the pitch, roll, and yaw channels together with the accelerometers in the pitch and yaw channels to displace and send equal signals to the three computers. Unless a malfunction is present, all flight control system caution lamps will remain out; and the pitch, roll, and yaw dampers will displace as required in Normal Procedures, Section II. When the test switches are released, no caution lamps should light.

Surface Motion and Lights Check.

When the master test switch is held and the stability augmentation switch is placed to the SURFACE MOTION and LIGHTS position, two out of three gyros in the pitch, roll, and yaw channels are displaced but no accelerometers are torqued. This sends only two rate signals to each computer. The absence of the third branch is detected and all three channel caution lamps light. In addition, one branch of each servo is also failed and the three damper caution lamps light. Further, in the pitch and roll axes, one branch of the gain changer is failed to minimum, and this error is detected, causing the pitch gain changer and roll gain changer caution lamps to light. Thus eight lamps light and all three dampers displace as required in Normal Procedures, Section II. The damper reset button is used to reset all lamps.

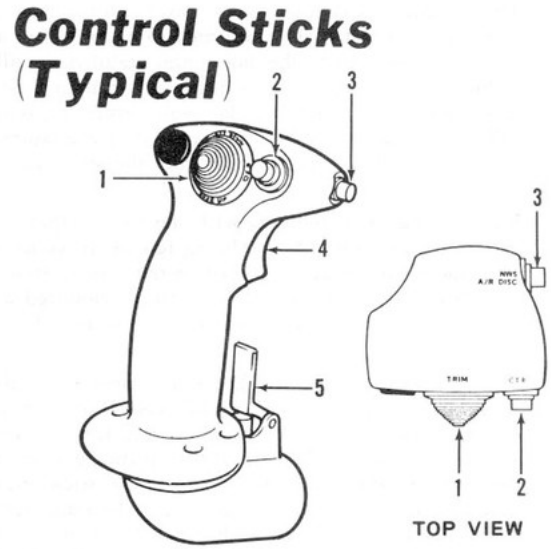
Stick Movement During Ground Checks.

During the stability augmentation checks, a rapid displacement of the control stick will occur. This is a normal condition. The stick should not be manually constrained during these checks since this will impose unnecessary loads on the stick-damper mechanical linkage.

FLIGHT CONTROL SYSTEM CONTROLS AND INDICATORS.

Control Sticks.

The two control sticks, one located at each crewmember's station, are mechanically interconnected. Prior to T.O. 1F-111(B)A-593, each stick grip contains a trim button, a reference engage button, a nose wheel steering/air refueling button, a gun trigger, and an autopilot release lever. After T.O. 1F-111(B)A-593, each stick grip (figure 1-24) contains a trim button, a constant track release button, a nose wheel steering/air refueling button, a gun trigger,



1. Trim Button.
2. Constant Track Release Button.
3. Nose Wheel Steering/Air Refuel Button.
4. Gun Trigger.
5. Autopilot Release/Pitch Control Stick Steering Lever.

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Figure 1-24.

and an autopilot release/pitch control stick steering lever. The control sticks also serve as a means of actuating the crew module bilge/flotation bag inflation pump. (Refer to "Crew Module Escape System," this section.) The right stick may be removed for various mission requirements. This must be accomplished while the aircraft is on the ground. When the right stick has been removed, an electrical plug is inserted in place of the stick to maintain electrical continuity.

Rudder Pedals.

Rudder control is provided by two sets of rudder pedals, one set located at each crewmember's station. The two sets of rudder pedals are mechanically interconnected, and in addition to controlling the rudder, each pedal operates the respective wheel brake in the conventional manner.

Stick Trim Button.

A trim button (1, figure 1-24), on each control stick grip, marked LWD, RWD, NOSE UP, NOSE DOWN, is spring-loaded to the center unmarked OFF position.

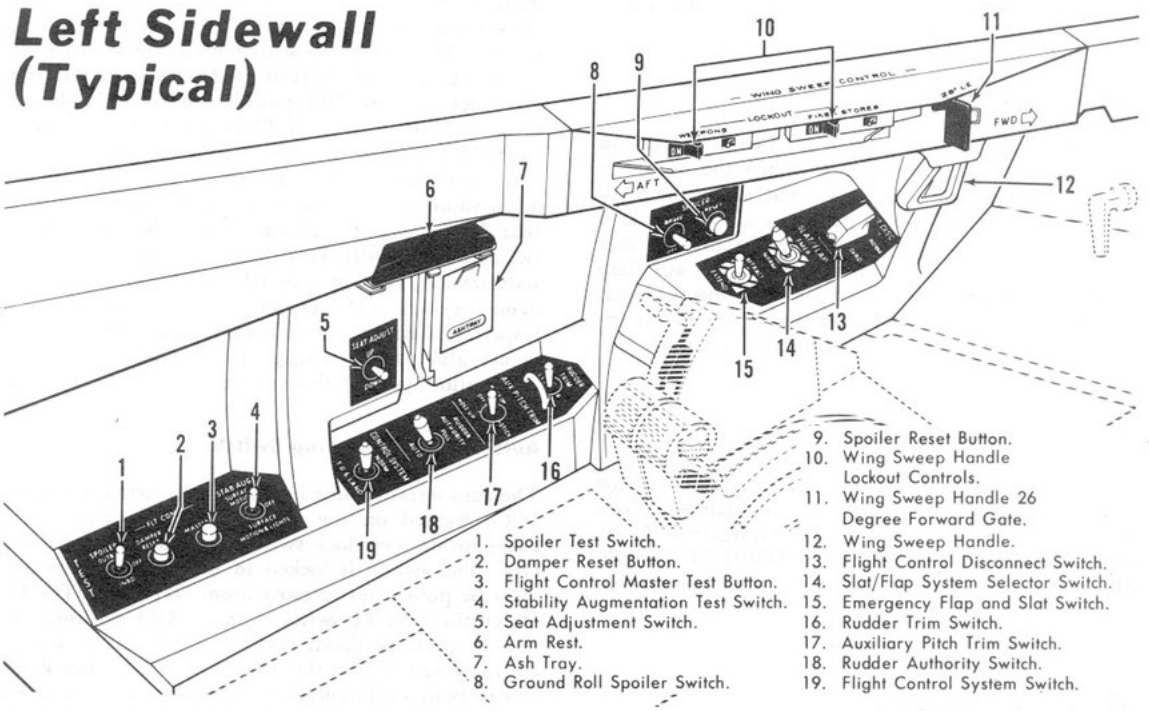


Figure 1-25.

Moving the button to NOSE UP or NOSE DOWN causes the pitch trim actuator to position the horizontal stabilizer surfaces symmetrically with trailing edge either up or down. Refer to "Pitch Channel," this section, for various trim authorities. Moving the button to LWD or RWD causes the roll damper servo to position the horizontal stabilizer surfaces asymmetrically as selected. The left stick trim button can always override the right stick button control. Approximately 8 seconds is required to trim the aircraft to its limit from neutral trim. When the roll damper is turned off, normal roll trim is inoperable and previous trim inputs are cancelled. The aircraft can be trimmed to the wings level position by use of rudder trim.

Auxiliary Pitch Trim Switch.

An auxiliary pitch trim switch (17, figure 1-25), marked STICK, NOSE DN, NOSE UP, and OFF, is located on the left sidewall. When the switch is in the STICK position, pitch trim can be commanded by the trim buttons on the control sticks. When the auxiliary pitch trim switch is placed to the center (OFF) position, the stick trim buttons are deactivated in pitch and the auxiliary pitch trim system is armed. When the switch is placed to the NOSE UP or NOSE DN position, pitch trim is provided by one of the following:

1. Positioning an auxiliary pitch trim integrator, which sends command signals to the pitch damper.
2. Directly driving the pitch series trim actuator.
3. Both of the above.

With the switch in the OFF, NOSE DN, or NOSE UP position, pitch trim inputs from the control sticks are inoperative. Roll trim is not affected by the switch position.

Rudder Trim Switch.

A rudder trim switch (16, figure 1-25), located on the left sidewall, is marked L and R and is spring-loaded to the center unmarked OFF position. Holding

the switch in L or R causes the rudder trim actuator to drive the rudder in the selected direction until the switch is released to OFF or a maximum deflection of 10.25 degrees is reached.

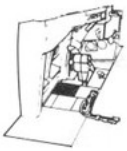
Takeoff Trim Button.

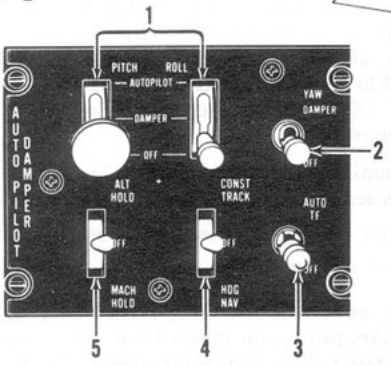
The takeoff trim button (26, figure 1-6), is located on the left main instrument panel. When the button is depressed, the takeoff trim relay is energized; the pitch parallel trim and yaw trim actuators are driven to 0 degrees; the roll trim integrator is synchronized so that the output to the roll damper is zero; the auxiliary pitch trim integrator is driven to a null; and the pitch trim series drives the horizontal stabilizers to 3.8 degrees trailing edge up. The button also functions during normal airborne operation.

Autopilot/Damper Switches.

Three switches, one each for the pitch, roll, and yaw channels, are located on the autopilot/damper panel. The pitch and roll damper switches (1, figure 1-26) are three position switches marked AUTOPILOT, DAMPER, and OFF. These switches are solenoid-held in the AUTOPILOT position, are spring-loaded to the DAMPER position, and are lever-locked in the OFF position. The switch lever of the pitch autopilot/damper switch is enlarged so that it can be readily identified by feel. The yaw damper switch (2, figure 1-26) is a two-position switch marked DAMPER and OFF. It is lever-locked in the OFF position and is spring-loaded to the DAMPER position. Placing any of the switches to DAMPER turns the respective damper on. The pitch and roll channels come on with the automatic gain at a low value and then begin setting the correct gain for that flight condition. Placing either the pitch or roll switch to AUTOPILOT will engage autopilot attitude stabilization. Placing a switch to OFF disengages the damper system of the respective channel and causes the respective damper caution lamp to light. These switches are also used to engage the autopilot. (Refer to "Autopilot System," this section.)

Autopilot/ Damper Panel (Typical)





1. Pitch and Roll Autopilot/Damper Switches.
2. Yaw Damper Switch.
3. Auto Terrain Following Switch.
4. Constant Track/Heading Nav Mode Selector Switch.
5. Altitude Hold/Mach Hold Selector Switch.

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Figure 1-26.

Auto Terrain Following Switch.

The auto terrain following (auto TF) switch (3, figure 1-26), located on the autopilot/damper panel, is a two-position lever-lock switch marked AUTO TF and OFF. The switch is locked in the OFF position and must be pulled out to move from OFF to AUTO TF. When the auto TF switch is in the OFF position, the aircraft must be flown manually to hold the terrain clearance selected by the TFR terrain clearance knob. In this switch configuration, the reference not engaged caution lamp (ATF not engaged lamp after T.O. 1F-111(B)A-593) will remain on. When the switch is placed to the AUTO TF position and either TFR channel mode selector knob is in the TF position, climb/dive signals from the TFR will control the pitch damper and the series trim actuator. With the switch in AUTO TF the reference not engaged caution lamp (ATF not engaged lamp after T.O. 1F-111(B)A-593) will go out if the auto TF mode is engaged. When AUTO TF is selected, at least one TFR channel must be in the TF mode or the TF fly-up off caution lamp and the reference not engaged caution lamp (ATF not engaged lamp after T.O. 1F-111(B)A-593) will be lighted. For additional information on the auto TF switch, refer to "Terrain Following Radar," this section.

Damper Reset Button.

The damper reset button (2, figure 1-25), located on the left sidewall, is labeled DAMPER RESET. When the button is depressed, the pitch, roll, and yaw damper caution lamps and their respective channel caution lamps on the main caution lamp panel will go out. Also the dampers and their respective electronic channels will be simultaneously reset to accept inputs for logic voting. If a malfunction is present at the time the reset button is released, the appropriate caution lamps will light. The button may also be used to reset the pitch and roll gain changer lamps.

Rudder Authority Switch.

The rudder authority switch (18, figure 1-25), located on the left sidewall, is marked FULL and AUTO. With the switch in the AUTO position, full rudder authority of 30 degrees either side of center is available when the control system switch is in the T.O. & LAND position or NORM position and the slats are extended. With the switch in AUTO and the control system switch in NORM with the slats retracted, only limited rudder authority is available. When the switch is in the FULL position, full rudder authority is available regardless of the slat position.

Flight Control System Switch.

The flight control system switch (19, figure 1-25), located on the left sidewall, is marked T.O. & LAND and NORM. The T.O. & LAND position is used in the event of a system malfunction. The NORM position is used at all other times, and the position of the slats determines the configuration of the flight control system. When the slats are extended beyond 70 degrees the following takeoff and landing functions are automatically provided: the yaw variable feel actuator moves to the full authority position, the TFR signals are locked out, the pitch and roll computer gains are locked, and the pitch series trim actuator is locked at its present position.

Note

The 70 percent slat switches are located on both the right and left slats. Both switches are utilized in the flap/slat asymmetry monitoring circuit. The right switch provides an input to the feel and trim assembly for flight control system switch functions.

If weight is not on the gear, AYC is activated. When the slats are retracted, the following inflight (clean configuration) functions are provided: the pitch damper can respond to TFR signals (if present), the pitch and roll gains are automatically determined by the flight control computers as flight conditions change, the pitch series trim actuator is unlocked and operates in the null mode, and the yaw variable feel actuator moves to the limited authority position. A warning system is provided to indicate that the flight control system is not in the takeoff and landing configuration while airborne when the landing gear handle is in the DN position. In this event, both the pitch gain changer and roll gain changer lamps will light. Extension of the slats, or placing the flight control system switch to T.O. & LAND will place the control system in the takeoff and land configuration. The lamps will go out unless a malfunction is present or unless the flight control disconnect switch is in OVRD. Failure of the two lamps to go out is caused by the override action of the flight control disconnect switch on the adverse yaw compensation and pedal shaker systems. Use of the flight control system switch in this case will not cause the lamps to go out even though all other switching has taken place.

WARNING

Attempting abrupt rolling maneuvers or bank angles is excess of 60 degrees, with the flight control system in the takeoff and land configuration by extension of the slats or by placing the control system switch in T.O. & LAND, can result in loss of control of the aircraft.

Flight Control Disconnect Switch.

The flight control disconnect switch (13, figure 1-25), located on the left sidewall, is marked NORM and OVRD (override). A guard covers the switch in the NORM position to prevent inadvertent actuation. Placing the switch to OVRD removes the following:

1. Pitch and roll autopilot commands.
2. Roll trim commands.
3. Auxiliary pitch trim inputs to the pitch damper.
4. TFR climb/dive commands to the pitch damper.
5. AYC and pedal shaker commands.

When in the takeoff and land configuration, the OVRD position lights the pitch and roll gain changer lamps. Also, the reference not engaged caution lamp (ATF not engaged caution lamp after T.O. 1F-111(B)A-593) will light when the switch is placed to OVRD. Placing the flight control disconnect switch in the OVRD position will cause the stall warning lamp and audible stall warning signal to be activated any time the indicated angle of attack exceeds 14 degrees, regardless of wing sweep position. Functions not affected by the flight control disconnect switch are:

1. Stability augmentation.
2. Pitch series trim while the auxiliary pitch trim is in stick.
3. TFR fail safe flyup maneuver.
4. TFR climb/dive commands to the pitch series trim actuator.

Flight Control Master Test Button.

The flight control master test button (3, figure 1-25), located on the left sidewall, provides power to the following:

1. Flight control test switches and buttons on the ground check panel.

2. CADC test switch on the ground check panel.
3. Stability augmentation test switch on the auxiliary flight control panel.
4. Spoiler test switch on the flight control switch panel.

Depressing the button applies power to the test switches and buttons. In addition, series trim is unlocked when the button is depressed. When the button is released, these switches and buttons are inoperable.

Accelerometer Test Button.

The accelerometer test button (12, figure 1-27) is located on the ground check panel. When depressed, in conjunction with the flight control master test button, all of the accelerometers in the pitch and yaw axes are torqued causing the rudder and horizontal stabilizers to displace. During this check, the rudder moves left and the horizontal stabilizers move trailing edge up due to the action of the yaw and pitch dampers. The yaw and pitch damper and channel caution lamps (4) should not light for this check.

Stability Augmentation Test Switch.

The stability augmentation test switch (4, figure 1-25), located on the left sidewall, is marked SURFACE MOTION and SURFACE MOTION & LIGHTS with an unmarked center OFF position. With the flight control system in the normal configuration, this switch, when used in conjunction with the flight control master test button, provides a means of ground checking the stability augmentation system.

Spoiler Reset Button.

The spoiler reset button (9, figure 1-25), located on the left sidewall, is a momentary pushbutton labeled SPOILER RESET. The button is provided to reset the spoiler monitor in the event that a malfunction has caused a pair of spoilers to be voted out and locked down. If a pair of spoilers has been locked down and the spoiler caution lamp is lighted, depressing the spoiler reset button will cause the caution lamp to go out and the spoiler circuitry to be reset to accept signals from the spoiler transducers. If the malfunction still exists, the faulty spoiler will again extend and the corrective control stick motion will cause the spoilers to again lock down and the spoiler caution lamp to light.

Spoiler Test Switch.

The spoiler test switch (1, figure 1-25), located on the left sidewall, is marked OUTBD, OFF, and INBD. The switch is spring-loaded to OFF and is used in conjunction with the flight control master test button to ground check the operation of the spoilers. With the switch in OUTBD, depressing the master test button will cause the outboard pair of spoilers to extend momentarily and then lock down and the spoiler caution lamp to light. Depressing the spoiler reset button will return the spoilers to operation and cause the spoiler caution lamp to go out. The INBD position of the switch is used to make the same check of the inboard spoilers. If the spoiler switch is moved from OUTBD to INBD (or vice versa) before the reset button is depressed, the first pair of spoilers locked down will be returned to operation; however, the caution lamp will remain on. This will invalidate the inboard spoiler check.

Computer Power Switches.

The computer power switches (16, figure 1-27) labeled NO. 1, NO. 2, and NO. 3 are located on the ground check panel. Each power switch controls the power to one branch of the pitch, roll, and yaw computers, and to other electrical flight control components. AC electrical power is supplied through the three computer power switches from the essential bus. When the switches are ON, and the electrical power is on the aircraft, 400-cycle electrical power is available to the entire flight control system. When these switches are OFF, the various trim functions, dampers, and spoilers can no longer be operated; and only the basic mechanical hydraulic system is available to operate the flight control surfaces. The switches must be in the ON position in order for the door on the ground check panel to be closed.

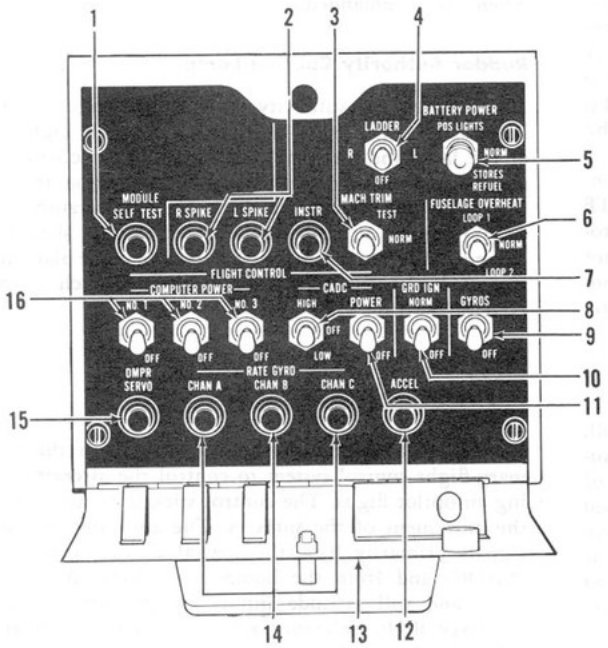
Damper Servo Button.

The damper servo button (15, figure 1-27), labeled DMPR SERVO, is located on the ground check panel. When the damper servo, rate gyro channel B and channel C buttons, and flight control test master switch are depressed and held, the electrical power to valve No. 1 on each damper servo is interrupted. Electrical command signals from each computer, cause each damper servo to vote hydraulically. This causes the pitch, roll, and yaw damper and channel caution lamps (6) to light.

Rate Gyro Test Buttons.

The rate gyro test buttons (CHAN A, CHAN B, and CHAN C) (14, figure 1-27) are located on the ground check panel. When two or more of the buttons are depressed in conjunction with the flight control master test button, the respective rate gyros are torqued, resulting in a predetermined displacement of the primary flight control surfaces. Depressing the CHAN A button causes the "A" gyros to be torqued in the pitch, roll, and yaw channels. Depressing the CHAN B and CHAN C buttons causes their respective gyros to be torqued. When a single CHAN button and the master test button are depressed, all three CHANNEL lamps will light. The control surface motion, if any, will be less than 2 degrees.

Ground Check Panel





1. Engine Fire Detection System Module Self-Test Button.
2. Spike Test Buttons.
3. Mach Trim Test Switch.
4. Entrance Ladder Switch.
5. Position Lights/Stores Refuel Battery Power Switch.
6. Fuselage Overheat Test Switch.
7. Instrument Test Button.
8. CADC Test Switch.
9. AFRS Power Switch.
10. Ground Ignition Cutoff Switch.
11. CADC Power Switch.
12. Accelerometer Test Button.
13. Ground Check Panel Access Door.
14. Rate Gyro Test Button (3).
15. Damper Servo Button.
16. Computer Power Switches (3).

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Figure 1-27.

Control Surface Position Indicator.

The control surface position indicator (18, figure 1-6), located on the left main instrument panel, is composed of three separate sets of indicators which provide indications of the positions of the spoilers, rudder, and horizontal tails (horizontal stabilizers). The position of the spoilers is indicated on four flip-flop type indicators, two for the left and two for the right spoilers. When the spoilers are retracted, the letters DN appear in each indicator. As the spoilers extend, the indicators become blank. Rudder position is indicated by a pointer on a scale, with a range of 30 degrees (L) left or (R) right of zero, graduated in 5 degree increments. The position of the horizontal stabilizer is indicated by two pointers, marked L and R. The indicator scale is 30 degrees up and 20 degrees down and is graduated in 2 degree increments. An index mark mounted on the axis of the left pointer provides indications of left or right wing down (LWD or RWD) against a scale mounted on the axis of the right pointer. In this manner, asymmetric stabilizer position indications also provide left or right wing down indications.

Takeoff Trim Indicator Lamp.

A takeoff trim indicator lamp (25, figure 1-6), located on the left main instrument panel, is provided to indicate when the horizontal stabilizer and rudder are in the proper trim position for takeoff and the auxiliary pitch trim integrator is zeroed. When the takeoff trim button is depressed and all affected surfaces reach their proper position, the lamp lights. When the takeoff trim button is released, the lamp goes out.

Pitch, Roll, and Yaw Channel Caution Lamps.

Three caution lamps, one each for the pitch, roll, and yaw channels, are located on the main caution lamp panel (figure 1-36). Lighting of any one of the lamps indicates that a malfunction has been sensed

in the computer of the respective pitch, roll, or yaw channel. Since the electronics in each channel is triple-redundant, lighting of one of these caution lamps indicates that one of the three sets of electronics is in error (passive first failure) and does not indicate a complete failure; however, it does indicate that system redundancy has been lost. (Refer to "Pitch Channel and Pitch Damper Redundancy," this Section.) A power supply failure in the yaw computer will cause a non-resettable yaw channel caution lamp to light. Such a failure can cause loss of response to TF climb/dive signals, loss of the TF flyup capability, loss of the reference not engaged caution lamp (ATF not engaged caution lamp after T.O. 1F-111(B)A-593), TF flyup off caution lamps, and loss of pitch or roll autopilot capability. For these reasons, TF operation is not recommended and autopilot operation should be monitored closely if a non-resettable yaw channel caution lamp is lighted.

Pitch, Roll, and Yaw Damper Caution Lamps.

Three caution lamps, one each for the pitch, roll, and yaw dampers, are located on the main caution lamp panel (figure 1-36). Lighting of any one of the lamps indicates that a malfunction has been sensed in its respective damper. Since each damper has two active valves and a model valve, lighting of one of the caution lamps does not indicate a complete damper failure; however, it does indicate that system redundancy has been lost.

Spoiler Caution Lamp.

The spoiler caution lamp, located on the main caution lamp panel (figure 1-36), is provided to indicate when a malfunction in the spoiler circuitry has occurred, causing a symmetric pair of spoilers to be locked down. The lamp is also used in conjunction with the spoiler test switch when ground checking spoiler operation. (Refer to "Spoiler Test Switch," this section.)

Roll and Pitch Gain Changer Caution Lamps.

Two gain changer caution lamps, one each for the pitch and roll gain changer, are located on the main caution lamp panel (figure 1-36). Lighting of either of these lamps indicates that a portion of the triple-redundant gain setting circuit in the respective channel is in error. Depressing the damper reset button will reset the lamp for a temporary error. Since the gain changer circuitry in each channel is triple redundant, lighting of one of these caution lamps indicates that one of these three sets of electronics is in error but does not indicate a complete failure; however, it does indicate that system redundancy is lost. Lighting of both lamps simultaneously will normally indicate a disagreement between the position of the control system switch and slats and the configuration of the flight control system. This will occur when the landing gear handle is in the DN position and the flight control system is not in the takeoff and land configuration. This will also occur when the landing gear handle is in the UP position and the flight control system is in the takeoff and land configuration when not commanded.

Rudder Authority Caution Lamp.

An amber rudder authority caution lamp (figure 1-36) is located on the main caution lamp panel. Lighting of the lamp indicates the rudder authority actuator is not in the position commanded by slat position and/or the flight control system switch. The rudder authority should switch to full authority when the slats are extended and to limited authority when the slats are retracted with the flight control system switch in the NORM position.

AUTOPILOT SYSTEM.

The autopilot operates in conjunction with the primary flight control system to control the aircraft during autopilot flight. The control stick does not follow the movement of the surfaces. The autopilot receives signals primarily from the central air data computer (CADC) and from the bomb nav system. Auxiliary pitch and roll attitude inputs are provided by the auxiliary flight reference system (AFRS). Pitch and roll attitude signals are obtained from the bomb nav or the AFRS system. To maintain stabilized controlled flight, these input signals are accepted by the flight control computers, which compute the command signals to the pitch and/or roll dampers and to the pitch and/or roll trim integrator during all autopilot modes. The CADC supplies a delta altitude or delta mach signal when altitude or mach hold modes are engaged. The ground track error and the heading navigation error signals are supplied by the bomb nav system.

Note

After T.O. 1F-111(B)A-593, magnetic heading information from the AFRS is used as a heading reference during the roll stabilization mode.

The autopilot system and the signals which are supplied to the autopilot are essentially non-redundant. However, the CADC has a failure indication capability which will light the CADS lamp should a malfunction occur in the CADC. In like manner, the bomb nav system has a failure malfunction detection system which will light the primary attitude and primary heading caution lamps when certain failures occur.

Note

- Caution should be exercised in using the autopilot when a non-resettable yaw channel caution lamp is lighted since power may be lost to the pitch and/or roll autopilot circuits.
- Caution should be exercised in operating the autopilot if malfunctions exist affecting the attitude information on the ADI or altitude or altitude rate information displayed on the AVVI or, after T.O. 1F-111(B)A-593, AFRS magnetic heading.

Normally the autopilot receives attitude information from the bomb nav system. However, in the event of a bomb nav system malfunction, or when the instrument select switch is placed to AUX, the attitude reference source is switched to the AFRS. Attitude signals from either system are used as references by the pitch and/or roll damper systems to hold the aircraft at the reference attitude existing at the time of autopilot engagement. When the autopilot is not being used, the pitch and roll attitude signals are continuously synchronized in the flight control computer so that at the time of engagement of pitch or roll attitude stabilization (or, after T.O. 1F-111(B)A-593, the constant heading mode), the respective synchronized signal is zero. If for some reason this signal is not zero at the time attitude stabilization is selected, the mode will not engage. Also, if the attitude limits are exceeded, the mode will not engage. The nominal attitude limits are ±30 degrees in pitch and ±60 degrees in roll. Should these limits be exceeded in one or both channels, attitude stabilization will not engage in that channel until its attitude angle is reduced. In addition, the roll channel cannot be engaged if the yaw damper is OFF. The flight control disconnect switch must be in NORM to achieve proper autopilot operation in either the pitch or roll autopilot modes.

AUTOPILOT MODES.

The pitch autopilot modes are attitude stabilization, mach hold, and altitude hold. The roll autopilot modes are attitude stabilization, constant track and heading navigation. After T.O. 1F-111(B)A-593, attitude stabilization includes magnetic heading hold. Incompatible autopilot mode selection is prevented by circuit interlocks. Pitch attitude stabilization is in effect when the pitch autopilot is engaged unless either the mach or altitude submode is selected. Attitude stabilization will hold the aircraft on the reference pitch attitude. When the roll autopilot is engaged, roll attitude is in effect. However, after T.O. 1F-111(B)A-593, magnetic heading is also held when engagement occurs with bank angle less than 7.0 (±2) degrees.

After T.O. 1F-111(B)A-593, if the roll autopilot is engaged when the roll attitude is greater than 7.0 (±2) degrees, the roll synchronizer will lock, and the aircraft will maintain the roll attitude existing at the time of engagement. If the roll attitude is less than 7.0 (±2) degrees, the constant heading and roll synchronizer signal will cause the aircraft to roll to a wings level attitude at a rate not to exceed 16 degrees per second. If constant track is engaged and a new track is desired, which requires less than 7.0 (±2) degrees roll attitude, the constant track release (CTR) button (2, figure 1-24), on the control stick, must be held in the depressed position until the aircraft is manually maneuvered to the new track, and then released. In roll, the aircraft may be manually maneuvered at any time, by use of roll control stick steering, without disengaging the roll attitude stabilization and constant heading or heading navigation mode. When in constant track, the CTR button must be used for changing the track, if bank angles of 7.0 (±2) degrees or less are used. In pitch, the aircraft may be manually maneuvered, at any time, by depressing the autopilot release/pitch control stick steering lever (5, figure 1-24) on the control stick to the first detent, which is the pitch control stick steering (PCSS) position, without disengaging the pitch attitude stabilization, mach hold, or altitude hold modes. The new references will be that which exists when the autopilot release/pitch control stick steering lever is released, provided the pitch atitude does not exceed ±30 degrees.

Attitude Stabilization Mode (After T.O. 1F-111(B)A-593).

The pitch attitude stabilization mode and/or the roll attitude and constant heading stabilization modes, for roll attitudes less than 7.0 (±2) degrees, are the initial control modes, and are established when the pitch and/or roll autopilot/damper switches (1, figure 1-26) are positioned to AUTOPILOT. Pitch and/or roll attitude signals are received by the yaw computer, which in turn supplies signals to the pitch and/or roll computers. The pitch and/or roll computers commands the pitch and/or roll dampers to hold the aircraft at the reference attitude, which existed at the time of autopilot engagement. The constant heading mode will be engaged when the roll autopilot/damper switch is positioned to AUTOPILOT, if the roll attitude is less than 7.0 (±2) degrees and the control stick is centered. If the roll autopilot/damper switch is positioned to AUTOPILOT when the roll attitude is greater than 7.0 (±2) degrees and less than 60 degrees, the roll synchronizer will lock and the roll attitude existing at the time of autopilot engagement will be maintained. If, under these conditions, the constant track mode is engaged, the aircraft will roll wings level, and the roll damper will control the aircraft according to the new reference. During constant track

mode of operation, to change the track, at roll angles of less than 7.0 (±2) degrees, the CTR button on the control stick must be depressed and then released when the new track is established. If during constant track mode of operation, roll angles of less than 7.0 (±2) degrees are introduced into the system by roll control stick steering, the aircraft will return to the original track when the control stick is centered. If roll angles greater than 7.0 (±2) degrees are introduced into the system by roll control stick steering, and then the control stick steering disengaged, by centering the control stick, the aircraft will roll wings level and the new track is established at the time the control stick is centered. However, if control stick steering remains engaged by holding the control stick on the opposite side of neutral and not centered until the roll angle has become less than 7.0 (±2) degrees, the aircraft will roll wings level, but the new track will be established at the time the aircraft rolled back through 7.0 (±2) degrees bank angle and not at the time the stick is centered. If the heading navigation mode of roll autopilot is selected in lieu of constant track, the aircraft will roll to the bank angle required to capture the course computed by the bomb nav system, then roll out wings level. If pitch autopilot is engaged at this time, the original pitch attitude will continue to be controlled by the pitch attitude stabilization signal, which will remain unchanged. If the mach hold or altitude hold mode of pitch autopilot is engaged, the mach or altitude references will remain unchanged. Anytime the constant track or heading navigation mode of roll autopilot is disengaged, the roll autopilot will revert to the attitude stabilization and constant heading mode. Either crewmember may at any time change either the pitch and/or roll autopilot reference by using the appropriate control stick steering. The flight control disconnect switch must be positioned to NORM prior to engaging the pitch and/or roll autopilot system.

Mach Hold Mode.

The mach hold mode maintains constant mach. In this mode, mach is controlled by aircraft pitch attitude through operation of the horizontal stabilizer surfaces. Upon engagement of this mode, a mach reference is set up in the CADC. Any deviation of mach from this reference results in an error signal from the CADC. If mach increases above the reference, the resulting mach error signal will command a nose up attitude through the pitch damper and pitch series trim, causing the aircraft to return to the referenced mach number. An opposite command is used for a decrease in mach.

Altitude Hold Mode.

The altitude hold mode automatically maintains constant pressure altitude. Upon engagement of this mode, an altitude reference is established in the CADC. Any deviation in aircraft altitude results in an altitude error from the CADC. If the aircraft altitude increases above the reference, the resulting altitude error signal will command a nose down attitude through the pitch damper and pitch series trim until the desired altitude is obtained. An opposite command is given for a decrease in altitude.

Constant Track Mode.

The constant track mode maintains the aircraft on a constant ground track. When this mode is engaged, the existing ground track is sensed in the bomb nav system and is set up as a mode reference. Any deviation from this reference by the aircraft results in an error signal being sent from the bomb nav system to the roll computer. The roll computer, in turn, sends a command to the roll damper, correcting the deviation.

Heading Navigation Mode.

When operating in the heading navigation mode, the autopilot receives signals from the bomb nav system to steer in accordance with the steering mode set by the instrument system coupler (ISC) mode selector knob. These modes are CRS SEL NAV, BOMB/NAV, CRS LINE, and MAN CRS. This steering signal is utilized by the roll damper system to steer the aircraft. The heading navigation position should not be used if the ISC mode selector knob is in a position other than those above.

Control Stick Steering.

When any autopilot mode is engaged, including basic attitude stabilization, autopilot operation can be disengaged by use of control stick steering. Prior to T.O. 1F-111(B)A-593, control stick steering is activated in the pitch channel by applying a force greater than 1.7 pounds in a forward or aft direction to the control stick. After T.O. 1F-111(B)A-593, pitch control stick steering is activated by depressing the autopilot release/pitch control stick steering lever to the first detent. Control stick steering is activated in the roll channel by applying a force of 1.3 pounds laterally to the control stick. Termination of control stick steering is accomplished by the reverse action used to engage it. When control stick steering is activated, the pilot can manually maneuver the aircraft as desired. When control stick steering is terminated, the autopilot will re-engage provided the attitude limits are not exceeded. Prior to T.O. 1F-111(B)A-593, attitude stabilization only will be engaged when control stick steering is terminated; submodes must be re-engaged if previously selected. After T.O. 1F-111(B)A-593, the selected modes will be automatically re-engaged.

AUTOPILOT CONTROLS AND INDICATORS.

Computer Power Switches.

Three computer power switches (16, figure 1-27), located on the ground check panel, control electrical

power for autopilot and certain flight control system operations. (Refer to "Flight Control System," this section.) The autopilot is normally ready to engage after the power switches are placed in the ON position and the stabilization platform of the bomb nav system is properly erected.

Autopilot/Damper Switches.

Two autopilot/damper switches (1, figure 1-26), one each for pitch and roll, are located on the autopilot/damper panel and have three positions marked AUTOPILOT, DAMPER, and OFF. These switches are solenoid-held by 28 volt dc power in the AUTOPILOT position, are spring-loaded to the DAMPER position, and are lever-locked in the OFF position. The switch lever of the pitch autopilot/damper switch is enlarged so that it can be readily identified by feel. The switches operate independently of each other. When the switches are in the AUTOPILOT position, attitude stabilization is engaged and the aircraft will maintain constant attitude. When the switches are moved to DAMPER or OFF, all other mode switches will move to OFF and the aircraft will then revert to pilot-controlled flight. If in AUTOPILOT, the pitch autopilot/damper switch will return to DAMPER when either TFR channel mode selector knob is positioned to TF. After T.O. 1F-111(B)A-593, when airborne, the autopilot/damper switches will return to the DAMPER position when the slats are extended or the flight control switch is placed to T.O. & LAND. For additional information on these switches, as related to control of the roll and pitch dampers, refer to "Flight Control System," this section.

Constant Track/Heading Nav Mode Selector Switch.

The constant track/heading nav mode selector switch (4, figure 1-26), located on the autopilot/damper panel, is a three-position switch marked CONST TRACK, OFF, and HDG NAV. The switch is solenoid held by 28 volt dc power to CONST TRACK or HDG NAV and is spring-loaded to OFF. The switch will not latch in the CONST TRACK or HDG NAV positions unless the roll autopilot/damper switch is in the AUTOPILOT position. When the switch is positioned to CONST TRACK, the aircraft will be held to within one degree of the reference ground track. When the switch is positioned to HDG NAV, the aircraft will be held to within one degree of the course to the destination set on the HSI. If, while operating in CONST TRACK or HDG NAV position, 28 volt dc power to the holding solenoid is lost, the switch will return to the OFF position. No caution lamp will light for this malfunction. When the switch is positioned to OFF, the constant track/heading nav operation will be discontinued.

Altitude Hold/Mach Hold Selector Switch.

The altitude hold/mach hold selector switch (5, figure 1-26), located on the autopilot/damper panel, is a three-position switch marked ALT HOLD, OFF, and MACH HOLD. The switch is solenoid held by 28 volt dc power to ALT HOLD or MACH HOLD and is spring-loaded to OFF. The switch will not latch in either ALT HOLD or MACH HOLD position if the pitch autopilot/damper switch is not in AUTOPILOT position. When the switch is in the ALT HOLD position, the pitch autopilot will control the aircraft to maintain the altitude present at the time of mode engagement. When the switch is positioned to MACH HOLD, the pitch autopilot will control the aircraft to maintain the mach number present at the time of mode engagement. Selection of the TF position on the TFR channel mode selector knobs is incompatible with altitude hold or mach hold mode and will cause the switch to move to the OFF position. If, while operating in MACH HOLD or ALT HOLD positions, 28 volt dc power to the holding relay is lost, the switch will return to the OFF position. No caution lamp will light for this malfunction.

Reference Engage Buttons.

Prior to T.O. 1F-111(B)A-593, a reference engage button, marked REF ENGAGE, is located on each control stick grip. When any autopilot mode other than attitude stabilization is selected, one of the reference engage buttons must be momentarily depressed before the mode will engage. Either button may be used to engage the autopilot.

Constant Track Release Button.

After T.O. 1F-111(B)A-593, a constant track release (CTR) button (2, figure 1-24), marked CTR, is located on each control stick grip. When any computer derived track submode is selected and being flown, the ground track reference can be changed or disengaged by depressing the CTR button and manually flying the aircraft. If constant track is being used, the track that the aircraft is on when the button is released will be the track flown, if the bank angle is less than 7.0 ($\pm$2) degrees. If heading nav is being flown with any of the associated ISC modes, depressing the CTR button will allow manual heading control. When the desired ground track is established, the CTR button is released and the current track becomes the new reference for the constant track submode.

Autopilot Release Lever.

Prior to T.O. 1F-111(B)A-593, the autopilot release lever, located at the base of the stick grip, permits either crew member to disengage all functions of the autopilot without removing his hand from the stick.

Depressing the lever will return the autopilot/damper switches to DAMPER. This disengages all autopilot functions and places the aircraft under pilot control. When the aircraft is being flown on TFR, the TFR commands to the flight controls can be interrupted by depressing and holding the autopilot release lever. If in auto TF operation, the reference not engaged lamp will light. The commands will reappear when the lever is released unless a mode other than TF is selected.

Autopilot Release/Pitch Control Stick Steering Lever.

After T.O. 1F-111(B)A-593, the autopilot release/pitch control stick steering lever (5, figure 1-24), located at the base of each stick drip permits either crewmember to disengage all functions of the autopilot by depressing the lever full travel to the second detent without removing his hand from the stick. Second detent activation returns the autopilot/damper switches to DAMPER and the autopilot submode switches to off. The first detent position of the lever allows pitch control stick steering (PCSS) with the pitch attitude stabilization mode engaged and also with mach hold or altitude hold submodes engaged without disengagement of any autopilot switches. At this detent, the aircraft can be maneuvered to a new reference, which is established when the lever is released. Reengagement of the selected pitch modes is accomplished immediately upon release of the lever out of the first detent. When the aircraft is being flown on TFR, the TFR commands to the flight controls can be interrupted by depressing the lever to and holding it in the first detent. If in auto TF operation, the ATF not engaged lamp will light. The commands will reappear when the lever is released out of the first detent.

Reference Not Engaged Caution Lamp.

Prior to T.O. 1F-111(B)A-593, the reference not engaged caution lamp, located on the left main instrument panel, will light under the following conditions:

1. The autopilot/damper switches are in the AUTOPILOT position and control stick steering is being used.
2. Any autopilot mode (altitude hold, mach hold, constant track, or heading nav) is selected, and the reference engage button has not been depressed.

Note

The use of control stick steering in the axis of the autopilot mode that has been engaged will result in the mode being disengaged. The lamp will light and remain on until the reference engage button is depressed again.

3. Either TFR channel mode selector knob is in the TF position and the auto TF switch is OFF.
4. The auto TF switch is in AUTO TF and neither TFR channel mode selector knob is in the TF position.
5. Autopilot release lever is depressed during auto TF operation.

Note

For TF modes, the reference not engaged lamp will light when the TF mode on the TFR is selected and the auto TF mode is not controlling the aircraft.

6. The flight control disconnect switch is placed to the OVRD (override) position.

The letters REF NOT ENGAGED are visible in the face of the lamp when lighted.

AUTOPILOT PRINCIPLES OF OPERATION. (PRIOR TO T.O. 1F-111(B)A-593)

The autopilot modes are selected by positioning the respective mode switch on the autopilot/damper panel. The mode switches are magnetically held to the selected mode position and may be turned off by manually repositioning the switches to OFF or by momentarily depressing the autopilot release lever on either control stick. If a mode other than basic attitude stabilization is selected, the reference engage button on either control stick must be momentarily depressed to engage the selected mode(s). The reference not engaged caution lamp will light whenever a selected autopilot mode is not controlling. When the roll autopilot is engaged and the roll attitude is less than 3.5 ($\pm$2) degrees, the aircraft will roll to wings level; if the roll angle is greater than 3.5 ($\pm$2) degrees the aircraft will maintain the roll attitude existing at the time of engagement. During autopilot operation on the AFRS attitude source, the AFRS gravitational erection system may slowly roll the aircraft; therefore, either constant track or heading nav submode must be engaged to maintain the aircraft in a wings level attitude. Also, to insure a reduction in AFRS gyro roll erection, bank angles of at least 10 degrees should be used when turning. If the constant track mode is selected, the roll damper will control the aircraft according to the new reference. However, when the constant track mode is discontinued, the autopilot will revert back to attitude stabilization and will maintain the attitude that existed at the time of disengagement. For example, if the roll autopilot is engaged with the aircraft in a twenty-degree bank, this bank angle will be held. If the constant track mode is then engaged, the aircraft

will respond by returning to the reference ground track. If the constant track mode is subsequently discontinued, the autopilot will revert back to attitude stabilization.

Roll Submodes.

If a submode of the roll autopilot is desired, it may be selected by positioning the roll autopilot/damper switch to AUTOPILOT and by positioning the constant track/heading nav mode selector switch to either position. If CONST TRACK is selected, the aircraft should be flown until the desired ground track is reached, and then engaged by depressing the reference engage button on either control stick. If HDG NAV is selected, depress the reference engage button on either control stick, and the aircraft will fly the computed course to destination. When stabilized, the autopilot will hold the aircraft course within ±1 degree of the steering error command received from the bomb nav system. When the instrument system coupler mode selector knob is in the CRS SEL NAV, BOMB/NAV, CRS LINE or MAN CRS position, proper steering signals are available for autopilot operation. Depressing the autopilot release lever on either control stick will return the constant track/heading nav selector switch to OFF and the roll autopilot/damper switch to DAMPER.

Heading Navigation Mode Characteristics.

When the heading navigation or constant track mode is engaged, bank angle oscillations and heading overshoots may occur. When engaging with a large heading error, the bank angles may exceed 45 degrees.

Pitch Stabilization.

Pitch autopilot (altitude hold) may be engaged by placing the pitch autopilot/damper switch to AUTOPILOT. The mode will not engage if pitch attitude exceeds ±30 degrees. A new pitch attitude reference may be established by using control stick steering. The mode may be disengaged in the same manner as roll autopilot.

Altitude Hold Characteristics.

If the altitude hold mode is engaged while at a stabilized altitude, the flight control system will hold the reference altitude within ±60 feet unless engine power is changed, the wings are swept, or the speed brake is used. The altitude hold mode can be engaged up to 2000 fpm rate of climb or dive. The autopilot will cause the aircraft to appear to stabilize at an altitude slightly above or below the reference altitude. The aircraft will then slowly return to within ±60 feet of the reference. Changes in engine power, wing sweep, or speed brake operation while this mode is engaged will initially cause an altitude standoff, followed by a slow return toward the altitude reference.

Mach Hold Characteristics.

If the mach hold mode is engaged while at a stabilized flight condition, the flight control system will hold the reference mach number within ±0.01 mach under stabilized flight conditions. Changes in engine power or aircraft configuration will initially cause a corresponding change in mach number, followed by a slow return toward the reference mach number.

WARNING

Pitch autopilot operation must not be attempted while the slats are extended or if the flight control system switch is in T.O. & LAND. To do so will result in improper operation and may cause abrupt transients when the autopilot is disengaged.

The control stick will be centered and the pitch functions of the stick trim button will be inoperative when pitch autopilot/damper switch is placed to AUTOPILOT.

AUTOPILOT PRINCIPLES OF OPERATION. (AFTER T.O. 1F-111(B)A-593)

The autopilot system is a pilot relief system which allows the aircraft to carry out automatically certain manually-selected modes of operation.

Mode Selection.

The autopilot modes are selected by positioning the respective mode switch on the AUTOPILOT/DAMPER panel. The mode switches are magnetically-held to the selected mode and may be deactivated by manually repositioning the switches or by momentarily depressing the autopilot release/pitch control stick steering lever on either control stick to the second detent.

Autopilot Inputs.

Normally, the autopilot receives attitude and heading signals from the bomb nav system. However, the crewmembers may use the AFRS for attitude and heading reference. The flight instrument reference select switch on the miscellaneous switch panel selects the attitude reference source. Positioning the flight instrument reference select switch to PRI selects the bomb nav system attitude source. Positioning this switch to AUX selects the AFRS attitude source. Positioning the pitch autopilot/damper switch to AUTOPILOT engages the pitch attitude stabilization mode, and positioning the roll autopilot/damper switch to AUTOPILOT engages the roll attitude stabilization and

constant heading mode. A magnetic heading error signal is continuously being supplied from the instrument system coupler to the feel and trim assembly. If roll autopilot is engaged when the roll angle is less than 7.0 (±2) degrees, the aircraft will roll back wings level and this heading error signal will cause the aircraft to maintain the heading established at the time of autopilot engagement. If roll autopilot is engaged when the roll angle is greater than 7.0 (±2) degrees, this heading error signal is synchronized to zero and the aircraft will maintain the roll attitude existing at the time of autopilot engagement. A heading error signal from the bomb nav system is available to the roll autopilot anytime the computer switches on the COMPUTER CONTROL panel are positioned to GNC and WDC, respectively. Positioning the constant track/heading nav mode selector switch on the AUTOPILOT/DAMPER panel to CONST TRACK maintains the aircraft on a constant ground track. Positioning the switch to HDG NAV supplies the bomb nav system with a 28-volt dc discrete signal. The bomb nav system then supplies a steering error signal to the autopilot computer to steer the aircraft to any selected destination. The CADC supplies a change of altitude or mach when ALT HOLD or MACH HOLD on the AUTOPILOT/DAMPER panel is selected.

Preengagement Synchronization.

When autopilot is not engaged, the pitch and roll attitude signals from the bomb nav system or the AFRS are continuously synchronized in the yaw computer, and the magnetic heading error signals from the instrument system coupler is continuously synchronized in the feel and trim assembly. This is done so that, at the time of engagement of pitch and/or roll attitude stabilization, the respective synchronized signal will be zero. If, for some reason, this signal is not zero at the time attitude stabilization is selected, the mode will not engage. Also, if the attitude limits are exceeded, the mode will not engage.

Autopilot Control.

The autopilot system controls the aircraft in the pitch and roll axes. The pitch autopilot mode consists of the primary attitude stabilization mode and two submodes: mach hold and attitude hold. The roll autopilot mode consists of the roll attitude stabilization and constant heading and two submodes, constant track and heading navigation.

Attitude stabilization for autopilot operation is supplied by the stabilized platform unit in the INS or the displacement gyroscope of the AFRS. Pitch and roll attitude stabilization is supplied from the stabilized platform unit when the stabilized platform unit is properly erected and furnishing a 28-volt dc platform good signal to the ISC. With the flight instrument reference select switch on the miscellaneous switch panel positioned to PRI, pitch and roll attitude inputs originate in the stabilized platform unit. At this time, the PRI ATT/HDG caution lamp is off, indicating that the stabilized platform unit attitude signal is good. Failure of the stabilized platform unit automatically turns on the PRI ATT/HDG caution lamp and switches the pitch and roll attitude signals from the stabilized platform unit to the AFRS.

Pitch Autopilot Control. The pitch attitude signal flow is from the stabilized platform unit or AFRS to the yaw computer, where the error signal is developed. The pitch attitude error is maintained at a null prior to engagement of the pitch autopilot. In the pitch autopilot mode, any deviation in the pitch attitude about the engaged attitude will result in an error signal. The error signal goes to the pitch computer where the signal may be summed with the mach error or altitude error signal. The resulting error signal is divided, one path going to the feel and trim assembly, where it is amplified and routed to the pitch series trim actuator (which serves as an error integrator); the other path going to the pitch damper, which may be deactivated by placing the flight control disconnect switch to NORM.

To engage the pitch autopilot, the following six conditions must be satisfied:

1. Neither TFR channel selector knob to TF.
2. Interlock conditions satisfied: pitch angle within ±30 degrees and the pitch attitude synchronizer error less than one-half degree.
3. Autopilot release/pitch control stick steering lever on each control stick not depressed to second detent.
4. With weight on the gear, the LG SAFETY REL circuit breaker closed. With weight off the gear, the flight control system switch (19, figure 1-25) positioned to NORM and the slats retracted.
5. Auto terrain following switch must be in OFF.
6. Pitch control stick steering (PCSS) must not be engaged (autopilot release lever not depressed to first detent).

When the first four of the foregoing conditions are met, power is applied to the pitch autopilot/damper switch. When all of the foregoing conditions are met and the pitch autopilot/damper switch is positioned to AUTOPILOT, power is applied to the pitch switch holding coil, to the ALT HOLD/MACH HOLD switch, to the feel and trim assembly, and through the feel and trim assembly to the yaw computer as the pitch autopilot engage signal. In the feel and trim assembly, power is applied to the pitch autopilot and pitch control stick steering (PCSS) circuits. The PCSS circuit has an output to the yaw computer, provided the autopilot release/pitch control stick steering lever is not depressed to the first detent (PCSS position). When the autopilot release/pitch

control stick steering lever is depressed to first detent, the PCSS circuit output to the yaw computer is zero and the pitch autopilot signals are interrupted and synchronized to zero. The pitch autopilot engage and the PCSS signals are applied to the pitch autopilot logic circuits of the yaw computer. Two interlock conditions must be satisfied before the pitch autopilot logic circuits will permit the autopilot to be engaged: (1) the pitch attitude must be within ±30 degrees and (2) the pitch attitude synchronizer error signal must be less than one-half degree. With the interlocks satisfied, the pitch autopilot logic enables and interlocks the altitude hold, mach hold, pitch static synchronizer drive, and fade logic (in the pitch computer); it also provides ground logic in the feel and trim assembly. The ground logic circuits apply the autopilot attitude error signal to a network which sums this signal with the output of a rate generator (driven by the pitch series trim actuator). This summed signal drives the pitch series trim actuator which acts as an integrator.

Pitch Submodes. Two pitch submodes are available: altitude hold and mach hold. With the pitch autopilot/damper switch (1, figure 1-26) in AUTOPILOT, power is applied to the altitude hold/mach hold selector switch (5, figure 1-26). In ALT HOLD or MACH HOLD, power is applied to the switch holding coil and the engage signal is applied to altitude hold or mach hold circuits in the yaw computer and to the altitude or mach clutch in the CADC. The altitude hold or mach hold logic in the yaw computer requires two additional inputs. These two inputs are the pitch autopilot logic signal and a CADC logic input to the pitch static synchronizer master counter. Application of these two inputs causes the synchronizer to go to the washout mode and prevents disengagement, should the pitch attitude exceed ±30 degrees. The CADC logic provides a washout time constant change as a function of mach number. For mach less than 1.1, the washout time constant is 6.25 seconds, and for mach greater than 1.1, it is 12.5 seconds. The washout pitch attitude signal is applied to the pitch computer. The purpose of this washout signal is to provide aircraft damping. In addition, an up-attitude signal is computed as a function of bank angle and summed with the washout pitch attitude signal in the pitch computer. This up-attitude signal prevents loss of altitude during turns. The mach or altitude clutch logic signal engages a synchro, located in the CADC, at the appropriate reference. This synchro develops an error signal proportional to the deviation from the reference pressure altitude or mach number. The static switch in the pitch computer allows the change in altitude or mach signal to be summed with the washout pitch attitude signal. This summed error signal is sent to the pitch damper servo and to the feel and trim assembly where it is amplified and used to drive the pitch series trim actuator as an error integrator to reduce standoff errors. When the autopilot release/pitch control stick steering lever on the control stick is depressed to the first detent (PCSS position), the pitch control stick steering disengages the pitch autopilot logic, which in turn deactivates the altitude or mach hold logic. As a result, the pitch static synchronizer is returned to the synchronizing mode, the appropriate clutch is deenergized, and the static switch is turned off. When the REL autopilot release/pitch control stick steering lever is released and the logic is again correct, the mach or altitude hold mode will be reactivated. Depressing the autopilot release/pitch control stick steering lever to the second detent will break the 28-volt dc power to the pitch autopilot/damper switch. As a result the pitch autopilot/damper switch will return to DAMPER and the altitude hold/mach hold selector switch will return to OFF, disengaging the pitch autopilot.

Altitude Hold Characteristics. If the altitude hold submode is engaged while the aircraft is at stabilized altitude, the flight control system will hold this steady state altitude to within the following bounds.

1. ±30 feet up to 30,000 feet.
2. ±0.1 percent from 30,000 to 55,000 feet.
3. Varies linearly from ±0.1 percent at 55,000 feet to ±0.2 percent at 80,000 feet.

If the altitude hold submode is engaged while the aircraft is in a climb or dive at rates up to 2000 feet per minute, the aircraft will settle to within ±30 feet of the steady state altitude within 25 seconds per mach numbers between 0.50 and 0.90, and 50 seconds per mach for mach numbers above 1.10. The altitude hold submode should not be used in the transonic region (mach 0.90 to 1.10). Changes in engine power or roll angle while this mode is engaged will initially cause an altitude standoff, followed by a return toward the altitude reference.

Mach Hold Characteristics. If the mach hold submode is engaged while in a stabilized flight condition, the flight control system will hold the steady state mach number to within ±0.01 mach, unless engine power is changed. Changes in engine power will initially cause a corresponding change in mach number, followed by a return toward the referenced mach number. The mach hold submode should not be used in the transonic region (mach 0.90 to 1.10).

Roll Autopilot Control.

The roll attitude signal flow is from the stabilized platform unit or AFRS to the roll static synchronous drive in the yaw computer, where the roll attitude

error signal is originated. The roll static synchronizer drive maintains the roll attitude error at a null prior to engagement of roll autopilot. In constant track or heading navigation, it functions as a heading repeater. If the roll attitude is less than 7.0 (±2) degrees when the roll autopilot is engaged, constant heading is also engaged. The constant heading error signal, obtained from the instrument system coupler, is synchronized by the feel and trim assembly. It will synchronize the heading error to zero under any of the following conditions:

a. Prior to engagement of roll autopilot.
b. Bank angles greater than 7.0 (±2) degrees.
c. Using roll control stick steering for bank angles greater than 7.0 (±2) degrees.
d. Constant track engaged.
e. Heading navigation engaged.

If the roll angle is less than 7.0 (±2) degrees when roll autopilot is engaged, the constant heading and roll synchronizer will cause the aircraft to roll wings level and maintain the heading reference. If the roll angle is greater than 7.0 (±2) degrees when roll autopilot is engaged, the roll static synchronizer will lock and maintain the aircraft at the roll attitude (bank angle) existing at the time of engagement. The roll attitude error signal from the roll static synchronizer goes to the roll computer where the error signal is split into two paths. One path supplies an error signal to the feel and trim assembly. The other path is split into three redundant signals for application to the roll damper servo circuit.

Attitude stabilization and constant heading is the basic mode of roll autopilot and must be engaged prior to engagement of constant track or heading navigation submodes. The attitude stabilization and constant heading mode are engaged when the roll autopilot/damper switch (1, figure 1-26) is positioned to AUTOPILOT. In order to engage roll autopilot, the following conditions must be satisfied:

a. Yaw damper switch (2, figure 1-26) in DAMPER.
b. Pitch attitude less than ±30 degrees.
c. Roll attitude less than 60 degrees.
d. Roll attitude synchronized within ½ degree.
e. Roll control stick steering (RCSS) not engaged.
f. With weight on the gear, the LG SAFETY REL circuit breaker closed. With weight off gear, the flight control system switch (19, figure 1-25) positioned to NORM and the slats retracted.
g. Autopilot release/pitch control stick steering lever on each stick not depressed to second detent.
h. Interlock conditions satisfied.

When the last three of the foregoing conditions are satisfied, power is applied to the roll autopilot/damper switch. When all of the foregoing conditions are met and the roll autopilot/damper switch is positioned to AUTOPILOT, power is applied to the switch holding coil, to the constant track/heading nax mode selector switch (4, figure 1-26), and through the feel and trim assembly to the yaw computer as the roll autopilot engage signal. In the feel and trim assembly, power is applied to the roll autopilot relay and the roll control stick steering (RCSS) relay. The RCSS circuits have a dc output to the yaw computer when the transducer in the roll feel assembly is at neutral. When the stick is out of neutral, the voltage output from the roll transducer results in a zero-volt output from the RCSS circuit; long as this condition exist, the roll autopilot is not controlling the aircraft. The roll autopilot engage signal and the RCSS signal are applied to the roll autopilot logic circuits of the yaw computer. The other inputs to the roll autopilot logic provide the interlocks for autopilot engagement. Four interlock conditions must be satisfied before the roll autopilot logic circuits will permit the autopilot to be engaged.

1. Pitch attitude within ±30 degrees.
2. Roll attitude error less than one-half degree.
3. Yaw damper switch in DAMPER.
4. Roll attitude less than 60 degrees.

When all interlocks are satisfied, the roll autopilot logic signal enables and interlocks the constant track, heading navigation, roll static synchronizer drive, fade logic (in the roll computer), and provides the ground logic for the roll autopilot relay in the feel and trim assembly. This relay switches the roll attitude error signal into a network where the signal is summed with the roll signal converter rate generator output and is applied to the converter motor. The converter integrates the signal and sends it back to the roll computer, where the integrated signal is summed with the roll attitude signal. The resulting error signal is used to position the roll damper servo.

Roll Submodes. Two roll submodes are available: constant track and heading navigation. A roll autopilot submode may be selected by positioning the roll autopilot/damper switch to AUTOPILOT and by positioning the constant track/heading nav mode selector switch to the desired position. If the constant track reference is to be changed using bank angles of less than 7.0 (±2) degrees, the constant track release (CTR) button on either control stick must be depressed and held until the new track has been established and then released. When using bank angles greater than 7.0 (±2) degrees, RCSS must be used and the new track will be the track established when the control stick is centered. However, if the aircraft is being rolled back wings level using RCSS, the new track will be the track established at the time the aircraft passes back through the 7.0 (±2) degree bank angle and not when the control stick is centered. If the heading navigation mode is selected, the air-

craft will fly a course dependent upon the instrument system coupler (ISC) mode selection. In all the modes except MAN CRS (manual course), the aircraft will fly the computed course to destination. When stabilized, the autopilot will hold the aircraft course within ±1 degree of the steering error received from the bomb nav system. The computed course may be either great circle or constant ground track. When MAN CRS is selected on the ISC, the aircraft will fly to and maintain the course set in the HSI course selector window. Depressing the autopilot release/pitch control stick steering lever on either control stick to the second detent will return the constant track/heading nav mode selector switch to OFF and the roll autopilot/damper switch to DAMPER.

When the constant track/heading nav mode selector switch on the autopilot/damper panel is positioned to CONST TRACK (with the roll autopilot/damper switch in AUTOPILOT), power is applied to the switch holding coil, the constant track discrete logic in the converter set, and to the constant track logic in the yaw computer. This action establishes the ground track references in the converter set and allows it to be applied to the roll static synchronizer input circuitry. The roll static synchronizer drive serves as a repeater servo with a bank angle rate limit of 16 degrees per second. The roll static synchronizer drive sums the constant track and roll attitude error signal. This combined signal is applied to the roll computer in the same manner as the roll attitude stabilization signal. Heading navigation operation is the same as constant track, except that the signal from the bomb nav system is proportional to the error between the present ground track and the computed course from the present position to the selected position, unless the ISC switch is in MAN CRS. When the control stick is displaced from neutral, the RCSS disengages the roll autopilot logic which deactivates the attitude stabilization and constant heading or heading navigation circuits. RCSS will not deactivate the constant track circuit unless bank angles greater than 7.0 (±2) degrees are used. For bank angles less than 7.0 (±2) degrees, the CTR button on either control stick must be depressed. When the control stick is returned to neutral, the RCSS logic is correct for attitude stabilization and constant heading or heading navigation. Depressing the autopilot release/pitch control stick steering lever will interrupt power to autopilot/damper switch. As a result, the roll autopilot/damper switch will return to DAMPER and the constant track/heading nav mode selector switch will return to OFF, disengaging the autopilot.

When any autopilot mode is engaged, including basic attitude stabilization, the reference controlling the aircraft can be disengaged by use of control stick steering. Pitch control stick steering is activated by depressing the autopilot release/pitch control stick steering lever to the first detent. Roll control stick steering is activated for attitude stabilization and constant heading, heading navigation, and for constant track (bank angles greater than 7.0 (±2) degrees) by applying a light force laterally to the control stick. To change the constant track reference using bank angles less than 7.0 (±2) degrees, the CTR button on either control stick must be depressed and held until the new track is established. When the appropriate control stick steering function is released the affected autopilot modes will automatically re-engage, provided the limits are not exceeded. The norminal attitude limits are ±30 degrees in pitch and 60 degrees in roll. Actual engage limits will usually be lower than the limits outlined above. Should the engage limits be exceeded in either channel, attitude stabilization will not re-engage in that channel until the attitude angle is reduced to less than the limit. In addition, the roll channel cannot be engaged if either the pitch attitude is greater than ±30 degrees or the yaw damper is off.

Note

Auto terrain following is not a mode of autopilot operation, therefore control stick steering will not disengage the auto terrain following mode.

AUTOPILOT PROCEDURES.

Engaging the Autopilot.

1. ADI—Check for normal indications.
2. Attain a safe altitude and trim the aircraft to the desired attitude.

WARNING

The roll autopilot authority may be insufficient to hold the wings level with large asymmetric wing loading. In this condition the aircraft may roll off the referenced heading and altitude without disengaging the submodes.

Note

- Do not use the autopilot in the mach or altitude hold mode during operation in the transonic flight region between 0.90 and 1.10 mach. To do so may result in rapid pitch inputs.
- Autopilot operation cannot be engaged at attitudes exceeding ± 30 degrees in pitch and ± 60 degrees in roll.

3. Pitch, roll, autopilot/damper, and yaw damper switches—DAMPER.
   Check that all dampers are operating properly.
4. Flight instrument reference select knob—As applicable.
   If attitude signals from the bomb nav system are reliable, place the switch to PRI. If the AFRS is utilized for attitude information place the switch to AUX.
5. Roll and pitch autopilot/damper switches—AUTOPILOT.
   The control stick will be centered and the pitch functions of the stick trim button will be inoperative when pitch autopilot/damper switch is placed to AUTOPILOT.
6. Reference not engaged caution lamp—Out. (Prior to T.O. 1F-111(B)A-593)
   Check that the reference not engaged caution lamp goes out with no force applied to the control stick.
7. Engage either constant track or heading nav submode, if required. (Prior to T.O. 1F-111(B)A-593)

Note

Prior to T.O. 1F-111(B)A-593, if the AFRS is being utilized for attitude signals, a heading sub-mode must be engaged to maintain the aircraft in a wings level attitude.

Selecting the Autopilot Control Modes.

After the autopilot is initially engaged in attitude stabilization, the pilot may select a single control mode or a combination of compatible modes by means of the mode switches on the autopilot/damper panel. A mode affecting the pitch channel (mach hold or altitude hold) may be selected simultaneously with a mode affecting the roll channel (constant track or heading nav). However, two modes in the same channel cannot be selected simultaneously.

Note

The flight director system (FDC) provides no steering commands to the autopilot, therefore autopilot steering commands are not available for ILS, TACAN, MAN HDG or TKR RV modes of ISC operation.

The following procedures are for selecting each control mode after attitude stabilization has been engaged.

Selecting Altitude Hold or Mach Hold Modes. Manually maneuver the aircraft to the desired mach, altitude, or heading.

1. Altitude hold/mach hold selector switch—Select desired mode.
2. Reference engage button—Depress. (Prior to T.O. 1F-111(B)A-593.)
   If it is desired to change the reference speed, altitude, or heading, it will be necessary to disengage the respective mode while the change is made. After T.O. 1F-111(B)A-593, if it is desired to change the reference speed or mach, it is not necessary to disengage the respective mode. The change can be accomplished by holding the autopilot release/pitch control stick steering lever depressed to the first detent until the new desired airspeed or mach conditions have been reached. Upon release of the lever, the mach conditions existing at that time will be maintained provided the aircraft was in acceptable engagement limits.

Selecting Constant Track Mode. Prior to T.O. 1F-111(B)A-593, select constant track as follows:

1. Constant track/heading nav mode selector switch —CONST TRACK.
2. Reference engage button—Depress.

Selecting Constant Track Mode. After T.O. 1F-111(B)A-593, select constant track as follows:

1. Constant track/heading nav mode selector switch —CONST TRACK.

Note

The ground track reference can be changed without disengaging the constant track/heading nav mode selector switch by depressing the CTR button on either control stick and manually flying to establish the desired track.

Selecting Heading Navigation Mode. Before T.O. 1F-111(B)A-593, select heading navigation mode as follows:

1. Instrument system coupler mode selector switch —Select CRS SEL NAV, BOMB/NAV, CRS LINE, or MAN CRS position.
2. Reference engage button—Depress.

Selecting Heading Navigation Mode. After T.O. 1F-111(B)A-593, select heading navigation mode as follows:

1. Instrument system coupler mode selector switch— Select CRS SEL NAV, BOMB/NAV, CRS LINE, or MAN CRS position.

Note

- If roll control stick steering is used to change heading in the heading navigation mode, the aircraft will steer to the previous commanded heading when the control stick is centered unless the autopilot or the heading navigation submode is disengaged.
- If the CTR button is used to change heading in the heading navigation mode, the aircraft will return to the required heading for the ISC mode selected when the CTR button is released unless the autopilot or the heading navigation submode is disengaged.

Disengaging the Autopilot.

To disengage all autopilot functions and place the aircraft under pilot control, either depress the autopilot release lever (autopilot release/pitch control stick steering lever to second detent after T.O. 1F-111(B)A-593) or place the pitch and roll autopilot/damper switches to DAMPER. In either case, all the submode switches will move to OFF.

CENTRAL AIR DATA COMPUTER SYSTEM (CADC).

The aircraft is equipped with a central air data computer system which provides aerodynamic information to various control systems. The system consists of an electromechanical computer which processes raw data from the angle of attack probe, pitot-static probe, and a temperature sensor probe. The computer utilizes the following raw data: indicated static pressure, pitot pressure, total temperature, and indicated angle of attack. When this data reaches the computer, it is transformed into electrical signal outputs through an arrangement of transducers, mechanical linkage, and servo repeaters. The CADC is equipped with a failure monitoring system. Should a computing function fail, a CADS caution lamp on the main caution lamp panel will light. If a computing function should fail, which affects the pressure altitude or indicated airspeed displays on the integrated flight instrument system, a warning flag will appear on the associated instrument. In addition, mach, angle of attack, and pressure altitude data good signals are supplied from the CADC monitor system to be used as failure monitor interlocks for the terrain following radar, the flight director computer system, and the bomb nav system. Listed below are the various aircraft systems served by the air data computer system, followed in parentheses by the computer outputs which go to the system:

1. Altitude-vertical velocity indicator (pressure altitude and vertical velocity)
2. Airspeed mach indicator (mach number, indicated airspeed, and true wing angle of attack)
3. Maximum safe mach assembly (pressure altitude, mach number, and true air temperature)
4. Flight control system (incremental mach number and incremental LOG static pressure)
5. Engine fuel control unit (mach number)
6. Spike caution lamp (mach number)
7. Bomb nav system (pressure altitude, true airspeed, static pressure, true air temperature, true body angle-of-attack and CADC data good)
8. True airspeed indicator (true airspeed)
9. Optical display sight (indicated airspeed)
10. Terrain following radar (true body angle-of-attack, true airspeed, and angle-of-attack failure monitor)
11. Angle-of-attack indexer (true wing angle-of-attack)
12. Environmental control (indicated airspeed and true air temperature)
13. Flight director (incremental pressure altitude and pressure altitude failure monitor)
14. Landing gear warning (indicated airspeed and pressure altitude)
15. Marker beacon (pressure altitude)
16. IFF (pressure altitude).
17. Stall warning system (true wing angle-of-attack).

CADC POWER SWITCH.

The CADC power switch (11, figure 1-27), with positions POWER and OFF, is located on the ground check panel. When the switch is in OFF, no aircraft power is supplied to the CADC or the maximum safe mach assembly. Also, the CADS caution lamp will light, and the OFF warning flags in the airspeed indicator and altimeter will appear. When placed in the POWER position, power is supplied to the CADC and the maximum safe mach assembly.

CADC TEST SWITCH.

The CADC test switch (8, figure 1-27), with positions HIGH, OFF, and LOW, is located on the ground check panel. The switch is spring-loaded to the OFF position. When the switch is positioned to either HIGH or LOW a set of pre-selected test input values, high or low respectively, of mach, altitude and angle-of-attack are fed into the CADC for use with the various systems being checked. Normally the switch is used by the flight crew during functional or acceptance flight grounds checks or by maintenance for troubleshooting. The HIGH position is also used to ground check the total temperature probe heater. CADC tests per-

formed while the INS is in the align mode may result in an INS fault legend upon going to the NAV mode. Tests performed in NAV will cause system altitude errors.

CADC CAUTION LAMP.

The CADC caution lamp (figure 1-36), located on the main caution lamp panel, will light to indicate malfunctions in the central air data computer. The lamp will also light when the CADC power switch is in the OFF position when power is on the aircraft. When lighted, the letters CADS are visible.

MAXIMUM SAFE MACH ASSEMBLY.

The maximum safe mach assembly (MSMA) receives mach number, pressure altitude, and true freestream air temperature signals from the central air data computer (CADC) and wing sweep position from the wing sweep sensor. The MSMA commands the maximum safe mach (MSM) bar on the airspeed-mach indicator (AMI) and the reduce speed warning lamp. The MSMA computes the maximum continuous safe mach (design speed) of the aircraft for wing sweeps between 16 degrees and 49 degrees based upon pressure altitude, wing sweep, and temperature. The actual aircraft mach number is compared with the computed safe mach in the MSMA which provides a signal to light the reduce speed warning lamp when the aircraft reaches this allowable design speed. For wing sweeps greater than 49 degrees, the airspeed limits presented in Section V must be observed since the MSMA is designed to compute a limitation less restrictive than those presented in Section V. A power failure to the MSMA will cause the CADC caution lamp to light.

CAUTION

The MSMA computes maximum safe mach (design speed) for wing sweeps between 16 degrees and 49 degrees based upon functions of pressure altitude and wing sweep. For wing sweeps greater than 49 degrees the maximum airspeed limits presented in Section V must be observed. Any flight speed or temperature restrictions which are more restrictive than the clean aircraft design values are not considered in the MSMA and consequently, are not displayed on the AMI. This is true for temporary or permanent limitations.

AUXILIARY FLIGHT REFERENCE SYSTEM (AFRS).

The auxiliary flight reference system (AFRS) provides standby or backup attitude and directional information. The system consists of a directional and vertical gyro platform, compass control panel, remote compass transmitter (flux valve), and a control amplifier. Changes in aircraft attitude are detected by the vertical gyro and electrically transmitted to the standby attitude indicator at all times and to the attitude director indicator (ADI) whenever the flight instrument reference select switch is in the AUX position or in event of malfunction of the bomb nav system. The directional gyro and compass transmitter provide heading information to the bearing-distance-heading indicator (BDHI) at all times and to the ADI and horizontal situation indicator (HSI) whenever the flight instrument reference select switch is in the AUX position, in event of malfunction of the bomb nav system or when the instrument system coupler mode selector knob is in the TACAN or TKR RV positions. The vertical gyro is unlimited in roll but is limited to ± 82 degrees in pitch. The directional gyro is attitude stabilized by the vertical gyro. The AFRS compass provides three modes of operation: SLAVED, DG (directional gyro), and COMP (compass). The SLAVED mode provides gyro stabilized magnetic heading from the remote compass transmitter. This mode is designed for use at latitudes up to 70 degrees. At higher latitudes the horizontal component of the earth's magnetic field becomes too weak to provide reliable heading information and the DG mode should be used. In the DG mode, the remote compass transmitter is disconunected from the system, and the directional gyro operates as a free gyro to provide directional reference. Free gyro drift of the directional gyro will not exceed ± 1 degree per hour. In the DG mode, apparent drift of the directional gyro due to earth's rotation is corrected. The COMP mode provides magnetic heading directly from the remote compass transmitter without gyro stabilization. This mode of operation should only be used when the AFRS gyros are suspected to be unreliable. Unreliable AFRS attitude and gyro fast erection are indicated by the auxiliary attitude (AUX ATT) caution lamp, the OFF flag on the standby attitude indicator, and the OFF flag on the ADI if it is receiving attitude information from the AFRS.

WARNING

Momentary power interruptions, such as electrical bus transfer, may cause the AFRS gyro to revert to automatic fast erection. If this occurs, gyro fast erection will be indicated as described above for the duration of the two-minute fast erection cycle.

The ADI OFF flag condition will also cause the ATTITUDE caution lamp to light if the ADI is receiving attitude information from the AFRS.

Note

The ADI OFF flag may not come into view if the data becomes unreliable while in the AFRS mode because of internal current drain in the attitude monitor system.

FLIGHT INSTRUMENT REFERENCE SELECT SWITCH.

The flight instrument reference select switch (1, figure 1-73), located on the miscellaneous switch panel, has two positions marked PRI and AUX. Placing the switch to the PRI (primary) position supplies pitch, roll and heading information from the bomb nav system to the following subsystems:

1. Autopilot
2. Attitude Director Indicator
3. Horizontal Situation Indicator
4. Flight Director Computer
5. Terrain Following Radar
6. Optical Display Sight
7. Attack Radar
8. Doppler Radar
9. Astro Tracker.

Placing the switch to the AUX (auxiliary) position supplies pitch, roll and heading information from the AFRS to all the above subsystems.

Note

If there is a difference between primary and auxiliary headings, verify DCC magnetic variation. If the magnetic variation is incorrect, the primary heading displayed to the pilot is in error. If the magnetic variation is correct, the auxiliary heading is in error. When the auxiliary heading is in error and it is selected, the TACAN magnetic bearing on the HSI/BDHI is correct but the relative bearing is in error. With an auxiliary heading error when primary heading is selected, the TACAN magnetic bearing and the relative bearing are incorrect. The CDI, bank steering bar, and autopilot steering corrections are usually valid with primary magnetic heading error on the ADI and HSI.

AUXILIARY FLIGHT REFERENCE SYSTEM POWER SWITCH.

The AFRS power switch (9, figure 1-27), located on the ground check panel, has positions GYROS and OFF. Placing the switch to GYROS supplies power to the AFRS, the BDHI, and the standby attitude indicator. Placing the switch to OFF de-energizes these components.

AFRS GYRO FAST ERECT BUTTON.

The AFRS gyro fast erect button (6, figure 1-73), located on the miscellaneous switch panel, is labeled ATT GYRO FAST ERECT. During initial turn-on of the system (initial erection), the gyro will automatically erect at the fast rate. If re-erection is required due to the gyro erecting to a false vertical or the pitch limits of the gyro being exceeded, fast erection may be accomplished by depressing and holding the fast erect button until the attitude indicators return to normal.

Note

To prevent erecting the gyro to a false reference, do not depress the button except when the aircraft is in a level attitude.

During initial erection or when the fast erect button is depressed, the AUX ATT and ATTITUDE lamps on the main caution lamp panel will light, the OFF flag on the standby attitude indicator will come into view, and if the flight instrument reference select switch is in the AUX position, the OFF flag on the ADI will come into view.

Note

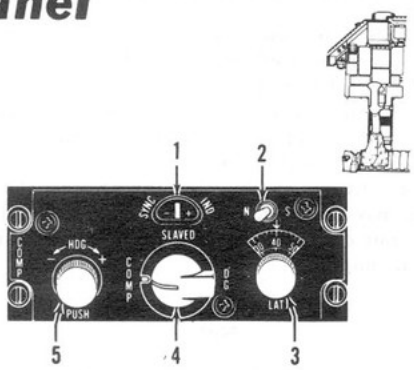
The ADI OFF flag may not come into view if the data becomes unreliable while in the AFRS mode because of internal current drain in the attitude monitor system.

During initial erection or whenever the fast erect button is depressed, the displacement gyroscope erects at a rate approximately 12 degrees per minute.

COMPASS MODE SELECTOR KNOB.

The compass mode selector knob (4, figure 1-28), located on the compass control panel, is used to select the mode of operation of the AFRS compass. The knob has three positions marked SLAVED, COMP, and DG. When the SLAVED mode is selected, gyro-stabilized magnetic heading from the remote compass transmitter is provided. In the DG mode, the remote compass transmitter information is removed from the system, and the system operates as a free gyro indicating an arbitrary gyro heading. In the COMP mode, the compass heading is obtained directly from the remote compass transmitter without stabilization by the directional gyro and is used in event of an attitude malfunction of the auxiliary flight reference system.

Compass Control Panel



1. Synchronization Indicator.
2. Hemisphere Selector Switch.
3. Latitude Correction Knob.
4. Compass Mode Selector Knob.
5. Heading Set Knob.

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Figure 1-28.

Note

When moving the knob from the SLAVED position to COMP, the compass cards on the HSI and BDHI and the attitude sphere of the ADI will rotate off the heading and immediately return. This is normal. When moving the knob from COMP back to the SLAVED position, the compass cards of the HSI and BDHI and the attitude sphere of the ADI will rotate off the heading but will not return until the heading set knob is depressed and held to null the synchronization indicator.

LATITUDE CORRECTION KNOB.

The latitude correction knob (3, figure 1-28), located on the compass control panel, is marked with latitudes from 0 degrees to 90 degrees. Setting the knob to the latitude at which the flight is being made determines the rate of gyro drift correction when operating in DG mode and improves heading accuracy when operating in SLAVED mode.

HEADING SET KNOB.

The heading set knob (5, figure 1-28), located on the compass control panel, provides a means of rapidly synchronizing the AFRS directional gyro with the remote compass transmitter when operating in the SLAVED mode, and to set in desired heading on the BDHI when operating in the DG mode. When the compass is operated in the SLAVED mode, fast synchronization is accomplished by depressing and holding the knob depressed until the synchronization indicator on the compass control panel becomes centered. When the compass is operated in the DG mode, system heading is changed by depressing and turning the knob to the right to increase the heading and left to decrease the heading. The rate of heading change is determined by the amount the knob is turned. When the compass is operated in the COMP mode, the system continuously tracks the remote compass transmitter and it is not necessary to use the knob.

HEMISPHERE SELECTOR SWITCH.

The hemisphere selector switch (2, figure 1-28), located on the compass control panel, has two positions marked N (North) and S (South). The switches must be positioned to the correct hemisphere in which the aircraft is operating to provide the proper balance of the earth's rate correction.

SYNCHRONIZATION INDICATOR.

The synchronization indicator (1, figure 1-28), located on the compass control panel, indicates whether or not the AFRS gyro and remote compass are synchronized. In the SLAVED mode, the pointer will normally fluctuate slightly when the compass set is synchronized with the gyro. Should the compass get out of synchronization, the pointer will deflect toward either the plus or minus sign on the face of the indicator. The landing set knob must be depressed and held until the pointer is centered to synchronize the system. The indicator is de-activated when operating in the DG or COMP modes.

AUXILIARY ATTITUDE (AUX ATT) CAUTION LAMP.

The AUX ATT caution lamp (figure 1-36) located on the main caution lamp panel will light for the following conditions:

- Initial erection.
- Loss of electrical power to the AFRS.
- Detected attitude malfunction.
- Manual fast erection.

When lighted, the letters AUX ATT are visible. If the lamp lights for a malfunction (other than erection) the standby attitude indicator will be unreliable. Also the ADI will be unreliable if it has been switched to the AFRS attitude source.

AUXILIARY FLIGHT REFERENCE SYSTEM OPERATION.

Placing the AFRS power switch to GYROS applies power to the AFRS. The AFRS gyro will initially erect at the fast erection rate of 12 degrees per minute minimum for the first two minutes of operation. After the initial erection cycle, the gyro erection rate is reduced to 5 (±1) degrees per minute. If subsequent re-erection of the gyro is required, manual fast erection may be accomplished by depressing the ATT GYRO FAST ERECT button. Aircraft acceleration/deceleration reduces the gyro pitch erection rate to one-fourth of the normal erection rate. Since pitch erection is not completely removed, pitch errors may develop in the AFRS after prolonged aircraft longitudinal acceleration. Therefore, the vertical velocity indicator, altimeter, and angle-of-attack indicator should be cross-checked during aircraft acceleration to insure proper aircraft pitch attitude. The gyro roll erection rate is reduced to one-fourth of the normal erection rate whenever the aircraft bank angle exceeds 8.5 degrees.

Note

When operating on AFRS as the primary source of attitude information, roll into turns at roll rates greater than one degree per second, and maintain at least 10 degrees bank angle in turns to insure that the gyro roll erection rate is reduced. If the roll erection rate is not reduced during turns, the gyro will erect to a false vertical and erroneous aircraft roll attitude will be displayed.

Whenever the ADI is receiving attitude information from the bomb nav system, AFRS roll attitude errors resulting from erroneous gyro roll erection will result in disagreement between roll indications on the ADI and standby attitude indicator. If both the ADI and standby attitude indicator are receiving AFRS attitude information, the turn rate pointer on the ADI and the HSI compass card should be monitored in order that erroneous roll indications may be detected. If the compass is operating in the SLAVED mode, the remote compass transmitter magnetic heading signals are disconnected whenever gyro roll or pitch erection is reduced to prevent the system from synchronizing to erroneous headings caused by an unlevel heading sensor. Disengagement of the remote compass transmitter is indicated by an inactive synchronization indicator on the compass control panel. Initial synchronization of gyro heading when operating in the SLAVED compass mode, or positioning of the heading indicators to a known heading when operating in the DG compass mode, is rapidly accomplished by the heading set knob on the compass control panel. Subsequent resynchronization may be required if the pitch limits of the gyros are exceeded. In the SLAVED mode, the system will resynchronize automatically at 1.5 (±0.5) degrees per minute if manual rapid synchronization is not accomplished.

PITOT-STATIC SYSTEMS.

The aircraft is equipped with two independent pitot-static systems. The primary system provides pitot and static pressures to the CADC only from the pitot-static probe on the nose radome. The secondary system utilizes a probe mounted on the right side of the fuselage to provide pitot and static pressure to the standby airspeed indicator, static to the standby altimeter and vertical velocity indicator and pitot pressure to the crew module "g" actuated selector. Both probes are equipped with heating elements for anti-icing. (Refer to "Anti-icing and Defog Systems," this section.) For pitot-static system instrument error and difference between primary and standby instruments, see figure 1-29. It will be noted that a relatively large difference exists between the primary and secondary instrument readings on this figure. This is because the standby instruments are provided with uncorrected data from the pitot-static system, while the primary instruments are provided with data from the CADC which compensates for pitot-static system errors.

INSTRUMENTS.

The instruments consist of the total temperature indicator (TTI), true airspeed (TAS), standby instruments and the integrated flight instrument system (IFIS).

TOTAL TEMPERATURE INDICATOR.

The total temperature indicator (9, figure 1-6), located on the left main instrument panel, provides indications of aerodynamic heating. The indicator is an electrical-resistance type instrument that uses a remote temperature sensing probe, an amplifier, and a motor to position the indicator pointer. The temperature sensing probe is equipped with a heating element for anti-icing. (Refer to "Anti-Icing and Defog System", this section.) The face of the indicator is graduated in 10 degree increments from —50 degrees to +250 degrees centigrade, with a critical temperature index mark of 153 degrees and a maximum temperature index mark of at 214 degrees. A digital readout counter in the face of the indicator, marked SEC TO GO, indicates the time remaining for operation in the critical temperature range between 153 and 214 degrees. The indicator functions in conjunction with the total temperature caution lamp and the reduce speed warning lamp to provide the following indications: (1) When the critical temperature of 153 is reached the counter will start to drive down from 300 seconds toward zero and the total temperature caution lamp will light. (2) The counter will continue to drive until; (a) it reaches zero, (b) the temperature is reduced below 153 degrees or (c) the maximum temperature index of 214

Allowable Differences Between Primary and Standby Instruments

Date: 19 May 1972

AIRSPEED DIFFERENCE TOLERANCES

Airspeed— Knots	*Altitude — Feet*				
	Sea Level	*10,000*	*20,000*	*30,000*	*40,000*
100	+ 6 — 12	+ 7 — 11	+ 7 — 11	+ 8 — 10	+ 8 — 10
200	+ 10 — 13	+ 11 — 12	+ 11 — 11	+ 11 — 11	+ 11 — 11
300	+ 16 — 16	+ 16 — 16	+ 16 — 16	+ 15 — 17	+ 11 — 21
400	+ 21 — 21	+ 21 — 22	+ 19 — 23	Note 4	+ 20 — 22
500	+ 25 — 27	+ 23 — 29	Note 4	+ 25 — 27	+ 24 — 29
600	+ 21 — 31	Note 4	+ 25 — 28	+ 23 — 29	+ 22 — 30

ALTITUDE DIFFERENCE TOLERANCES

Airspeed— Knots	*Altitude — Feet*				
	Sea Level	*10,000*	*20,000*	*30,000*	*40,000*
100	+ 70 — 130	+ 155 — 205	+ 245 — 305	+ 370 — 420	+ 570 — 610
200	+ 90 — 90	+ 170 — 170	+ 265 — 265	+ 385 — 385	+ 580 — 580
300	+ 90 — 90	+ 170 — 170	+ 265 — 265	+ 305 — 465	— 50 — 1210
400	+ 70 — 110	+ 145 — 195	+ 75 — 140	Note 4	+ 500 — 660
500	0 — 180	— 90 — 430	Note 4	+ 225 — 545	+ 180 — 980
600	— 285 — 465	Note 4	+ 105 — 425	— 90 — 860	+ 540 — 620

NOTES:
1. Enter tables at primary instrument readings to obtain upper and lower limits on differences.
2. Subtract primary reading from standby reading to obtain difference (may be negative).
3. Do not interpolate across heavy lines.
4. Check not recommended at this condition.
5. Difference limits include indicator and CADC tolerances, and standby instrument position error.
6. Primary and standby altimeter set to 29.92 for the checks.

Figure 1-29.

degrees is reached. (3) When the maximum temperature index is reached or when the counter drives to zero, the reduced speed warning lamp will light, and the total temperature caution lamp will go out. (4) The counter will reverse and drive back to 300 seconds anytime the temperature falls below 153 degrees. (a) If the reduce speed warning lamp was on when the counter reversed, it will go out. (b) If the total temperature caution lamp was out when the counter reversed, it will light and remain on until the counter has driven back to 300 seconds. An OFF flag will appear in the face of the indicator when power is removed from the instrument or when the amplifier output signal varies from the temperature probe input signal by 10 to 12 degrees centigrade.

Total Temperature Caution Lamp.

The total temperature caution lamp, located on the left main caution panel (figure 1-36), functions in conjunction with the total temperature indicator and reduce speed warning lamp to provide an indication that the aircraft is being operated in the critical temperature range between 153 degrees C and 214 degrees C. Refer to total temperature indicator this section for a description of lamp indications. When lighted, the words TOTAL TEMP are visible in the face of the lamp.

Reduce Speed Warning Lamp.

The reduce speed warning lamp, located on the left main instrument panel (figure 1-36), functions in conjunction with the total temperature indicator to indicate that the aircraft has flown for at least 300 seconds in the critical temperature range of from 153 to 214 degrees centigrade or that the maximum temperature index of 214 degrees has been reached or exceeded. Refer to Total Temperature Indicator this section for a description of the lamp functions in conjunction with the indicator. When lighted, the words REDUCE SPEED are visible in red on the face of the lamp. The lamp also functions in conjunction with the maximum safe mach assembly. Refer to "Maximum Safe Mach Assembly", this section.

TRUE AIRSPEED INDICATOR.

The true airspeed indicator (15, figure 1-30), located on the right main instrument panel, provides a digital readout of true airspeed. The instrument displays true airspeed on a servo-driven 4-digit counter within the range of 40-1750 knots. The indicator is operated by electrical signals from the CADC.

STANDBY INSTRUMENTS.

The standby instruments include the airspeed indicator, altimeter, vertical velocity indicator, magnetic compass, attitude indicator and bearing distance heading indicator (BDHI). These instruments provide backup indications in the event of failure of the integrated flight instrument system. Positon error must be applied to the airspeed and altimeter reading to obtain correct readings. Refer to "Appendix I, Part I".

Airspeed Indicator.

The airspeed indicator (3, figure 1-30), located on the right main instrument panel, is operated by pitot and static pressures direct from the secondary pitot-static system. The instrument is graduated from 0.6 to 8 times 100 knots.

Altimeter.

The altimeter (7, figure 1-30), located on the right main instrument panel, is a barometric type which operates on static pressure direct from the secondary pitot-static system. A barometric pressure set knob located on the left corner of the instrument provides a means of adjusting the barometric scale on the instrument.

WARNING

- Do not push in on the set knob when setting barometric pressure, as disengagement of the gear train between the indicating pointers and the barometric scale may occur, resulting in erroneous altimeter readings.
- Aircrews should not attempt to calibrate the standby altimeter to correct for an out-of-tolerance condition.

Vertical Velocity Indicator.

The vertical velocity indicator (4, figure 1-30), located on the right main instrument panel, provides rate of climb and descent information. The instrument operates on static pressure from the secondary pitot-static system.

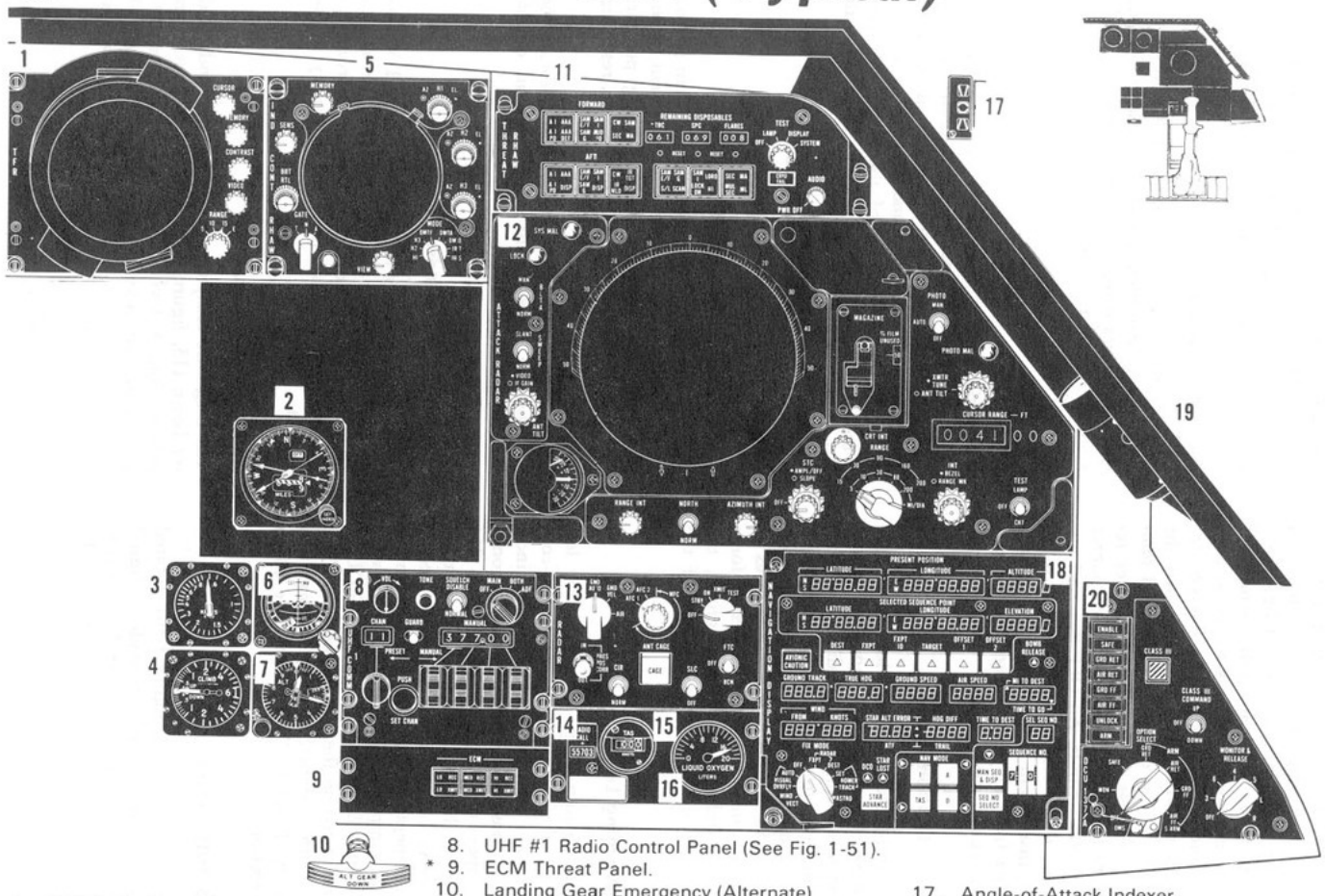
Magnetic Compass.

The magnetic compass, located on the windshield center beam (13, figure 1-2), provides magnetic heading information. A deviation correction card for the compass is located below the center of the glare shield. Heading, as read on the compass, should be accurate within ± 5 degrees.

Attitude Indicator.

The attitude indicator (6, figure 1-30), located on the right main instrument panel, provides backup attitude information in the event of malfunction or failure of the attitude director indicator (ADI). The indicator displays pitch and roll information on an attitude sphere in relation to a miniature aircraft. Pitch and

Right Main Instrument Panel (Typical)



1. TFR Radar Scope Panel (See Fig. 1-77).
2. Bearing-Distance-Heading Indicator.
3. Standby Airspeed Indicator.
4. Vertical Velocity Indicator.
* 5. RHAW Scope Panel.
6. Standby Altitude Indicator.
7. Standby Altimeter.
8. UHF #1 Radio Control Panel (See Fig. 1-51).
* 9. ECM Threat Panel.
10. Landing Gear Emergency (Alternate) Release Handle.
*11. RHAW Threat Panel.
12. Attack Radar Scope Panel (See Fig. 1-72).
13. Attack Radar Control Panel (See Fig. 1-70).
14. Radio Call Panel.
15. True Airspeed Indicator.
16. Oxygen Quantity Indicator.
17. Angle-of-Attack Indexer.
18. Navigation Display Panel (See Fig. 1-62).
19. Utility Light.
**20. Nuclear Weapons Control Panel.

* See T.O. 1F-111(B)A-1-3.
** See Weapons Delivery Manual.

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Figure 1-30.

roll signals are received from the AFRS. An attitude monitor system is provided to compare pitch and roll signals to the ADI with those to the standby attitude indicator. Refer to "Attitude Monitor System," this section. In the event of power failure or an AFRS malfunction, an OFF warning flag will appear on the lower left face of the indicator. A pitch trim knob on the lower right side of the instrument is provided to adjust the attitude sphere to the proper pitch attitude.

WARNING

The attitude warning flag will not appear with a slight electrical power reduction or failure of other components within the system. Failure of certain components can result in erroneous or complete loss of pitch and bank presentations without a visible flag.

Bearing-Distance-Heading Indicator.

Aircraft 63 ⏵ 52 are equipped with a bearing-distance and heading indicator (BDHI) (2, figure 1-30) which is located on the right main instrument panel. The instrument is a remote-type heading indicator with a rotating compass card. TACAN bearing information is displayed by means of pointers. A synchro-driven range indicator is provided which receives signals from the TACAN set. Range of the distance display is 0-999 nautical miles. A red and black striped range warning flag partially obscures the range indicator when distance-to-station signals are too weak or there is a loss of lock-on to TACAN distance signals. Magnetic heading of the aircraft is shown by the index at the top of the instrument and the compass card. A pointer designated as No. 2 is servo-driven and receives signals from a TACAN coupler. Bearing information is read from the compass card under the pointer tip. The No. 2 pointer is positioned concurrent with the No. 1 pointer and rotates with it. The number 1 pointer is deactivated, therefore it will not be visible. The BDHI receives heading information from the auxiliary flight reference system (AFRS). The set index knob, located on the lower right side of the indicator, is used to set the heading index to a desired magnetic heading. Once set, the index rotates with the compass card. A flag marked OFF will appear in the window when the BDHI is not energized or when power is not available to the compass card.

INTEGRATED FLIGHT INSTRUMENT SYSTEM.

The integrated flight instrument system takes outputs from the following systems and integrates them into usable displays on the integrated flight instruments.

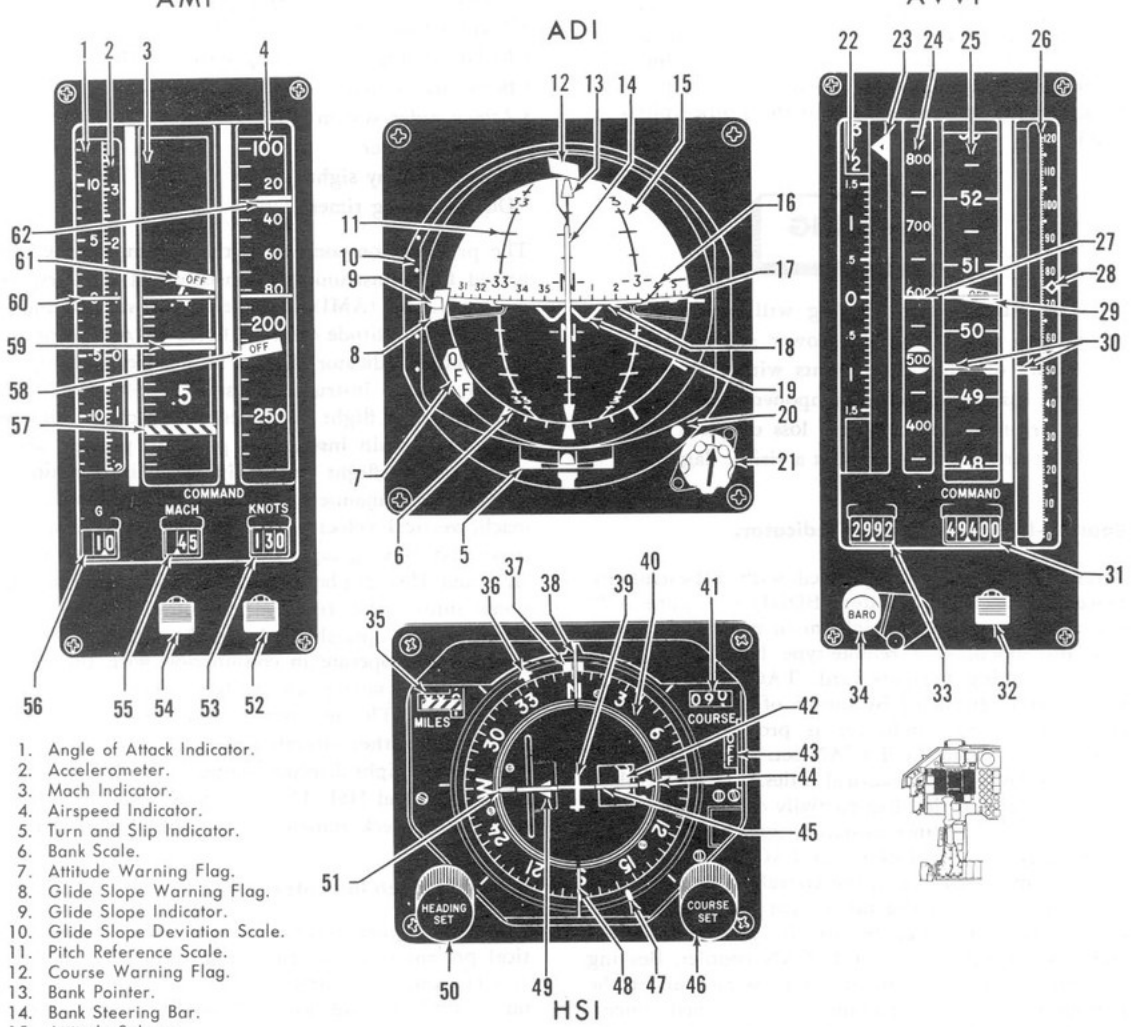
- Central air data computer (CADC)
- Auxiliary flight reference system (AFRS)
- Instrument landing system (ILS)
- Tactical air navigation system (TACAN)
- Terrain following radar (TFR)
- Radar homing and warning system (RHAWS)
- Bomb nav system
- Attack radar system
- Radar altimeter
- Optical display sight (ODS)
- Dual bombing timer (DBT)

The primary components of the system are the integrated flight instruments, consisting of the airspeed-mach indicator (AMI); altitude-vertical velocity indicator (AVVI); attitude director indicator (ADI); horizontal situation indicator (HSI);a flight director computer (FDC); and an instrument system coupler (ISC). The four integrated flight instruments are grouped together on the left main instrument panel to provide actual and command flight and navigational information in a clear concise manner. Altitude, airspeed, acceleration, mach, vertical velocity, and angle of attack are displayed on moving tapes on the AMI and AVVI. The ADI and HSI display attitude, heading, and navigational information from various other systems in the aircraft. The optical display sight (ODS) command steering bars operate in conjunction with the system to provide the same pitch and bank steering commands as the ADI. The instrument system coupler accepts inputs from other aircraft systems and channels them through the flight director computer for display on the ADI, ODS, and HSI. The system incorporates self-test features to check reliability and isolate malfunctions.

Airspeed Mach Indicator (AMI).

The AMI (figure 1-31) provides remote reading vertical presentations of true wing angle of attack "g" acceleration, mach airspeed and maximum safe mach on vertical moving scales. Readout windows below each moving scale present digital values for "g" acceleration, mach, and airspeed. Slewing switches for setting reference mach and airspeed markers are located on the bottom of the indicator. The slewing switches (54 and 52, figure 1-31) provide a manual means of setting command mach and command airspeed markers. The value set is displayed in the command readout window above the respective slewing switch. The command airspeed slewing switch has a side detent position. This allows the command airspeed readout window to display a continuous digital airspeed in knots. When in the detent position the command feature is lost, as the command marker remains at the fixed index line. The command mach slewing switch does not have a detent position. Signals for operation of the various scales are provided from

Integrated Flight Instruments



1. Angle of Attack Indicator.
2. Accelerometer.
3. Mach Indicator.
4. Airspeed Indicator.
5. Turn and Slip Indicator.
6. Bank Scale.
7. Attitude Warning Flag.
8. Glide Slope Warning Flag.
9. Glide Slope Indicator.
10. Glide Slope Deviation Scale.
11. Pitch Reference Scale.
12. Course Warning Flag.
13. Bank Pointer.
14. Bank Steering Bar.
15. Attitude Sphere.
16. Heading Reference Scale.
17. Horizon Bar.
18. Pitch Steering Bar.
19. Miniature Aircraft.
20. Pitch Trim Index.
21. Pitch Trim Knob.
22. Vertical Velocity Indicator.
23. Vertical Velocity Index.
24. Vernier Altimeter.
25. Coarse Altimeter.
26. Gross Altimeter.
27. Fixed Index Line.
28. Target Altitude Marker. (Inoperative)
29. Altitude Warning Flag.
30. Command Marker.
31. Command Altitude Readout Window.
32. Command Altitude Slewing Switch.
33. Barometric Pressure Readout Window.
34. Barometric Pressure Set Knob.
35. Tacan Range Indicator and Warning Flag.
36. Tacan Bearing Pointer.
37. Heading Marker.
38. Upper Lubber Line.
39. Aircraft Symbol.
40. Compass Card.
41. Course Selector Window.
42. Course Warning Flag.
43. Power Off Warning Flag.
44. Course Arrow Head.
45. Course Deviation Indicator.
46. Course Set Knob.
47. Bearing Pointer Tail.
48. Lower Lubber Line.
49. To-From Indicator
50. Heading Set Knob.
51. Course Arrow Tail.
52. Command Airspeed Slewing Switch.
53. Command Airspeed Readout Window.
54. Command Mach Slewing Switch.
55. Command Mach Readout Window.
56. Acceleration Readout Window.
57. Maximum Allowable Mach Marker.
58. Airspeed Warning Flag.
59. Command Mach Marker.
60. Fixed Index Lines.
61. Power Off Warning Flag.
62. Command Airspeed Marker.

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Figure 1-31.

the central air data computer, maximum safe mach assembly, and remote accelerometer. In the event of power failure, OFF warning flags will appear across the mach and airspeed scales. The (OFF) airspeed warning flag will appear in the event of a malfunction or failure in the airspeed section of the AMI or CADC. The circuit breaker for the airspeed mach indicator is located on the left main ac bus. Presentations on the face of the indicator are from left to right.

Note

The airspeed indicated on the airspeed mach indicator has been calibrated for pitot-static system errors by the CADC and, therefore, is actually KCAS (knots calibrated airspeed). However, this airspeed is referred to as KIAS (knots indicated airspeed) throughout this manual, since it is read directly from the instruments.

Angle of Attack Indicating System.

The angle-of-attack indicating system is composed of a conical probe transmitter located on the left side of the aircraft fuselage, an angle-of-attack correction cam in the central air data computer (CADC), two angle-of-attack indexers and a vertical scale read-out on the airspeed mach indicator. In flight, the angle-of-attack probe generates an indicated angle-of-attack signal which is sent to the CADC and corrected to true angle-of-attack. This correction is necessary since the indicated angle-of-attack contains position errors due to aircraft configuration as well as speed. The position error correction made in the CADC is accomplished as a function of mach number only, even though the actual position error is also a function of flap and slat configuration. Since flap and slat information is not supplied to the CADC, the correction cam is mechanized such that between mach 0.45 and 0.30 the position error correction is changed linearly from the flaps and slats retracted value to the flap and slat extended value. Between these two mach numbers the flaps and slats would be extended and the takeoff and landing configuration fully selected by 0.3 mach. If the aircraft is decelerated below mach 0.45 without extending the flaps and slats or is accelerated past 0.3 with the flaps and slats extended, the position error correction applied by the CADC will be in error and the angle-of-attack indicator will no longer read the true aircraft angle-of-attack. For the flaps and slats retracted case, the angle of attack indictor will read lower than true angle of attack below mach 0.45. The error will increase linearly from 0 error of mach 0.45 (approximately 300 KIAS at S.L.) to 1.7 degrees at 0.30 (approximately 200 KIAS at S.L.). Below mach 0.3 the error remains constant at 1.7 degrees. For the flaps and slats extended case, the angle-of-attack indicator will read higher than true angle-of-attack above mach 0.30. The error will increase linearly from 0 error at mach 0.30 to 1.7 degrees at mach 0.45. Above mach 0.45, the error remains constant at 1.7 degrees. Since the angle-of-attack indexers are commanded by the same signal from the CADC as the indicator, the on-speed lamp will be lighted when the tape reads 10 degrees even though the true angle-of-attack may not be 10 degrees. Anti-icing is provided to the angle-of-attack probe. The heating elements receive power from the main ac bus and are controlled by the pitot/probe heat switch and the squat switch on the main gear.

Note

Since the angle-of-attack indicator and indexers are commanded by the CADC, these instruments will be inoperative if the CADC is not operating.

Angle of Attack Indicator. The angle of attack indicator, located on the airspeed-mach indicator, indicates in degrees the angular position of the wing chord in relation to the aircraft flight path and is primarily used for approach speed monitoring. The vertically moving tape displays angle of attack from minus 10 degrees to plus 25 degrees. The angle of attack indicator is operated by means of synchro signal received from the central air data computer.

Angle of Attack Indexers. Two angle of attack indexers (5, figure 1-6 and 17, figure 1-30) are provided, one located on either side of the glare shield. Each indexer consists of 3 lamps arranged vertically. The low speed symbol, the top V-shaped red lamp, lights when the angle of attack exceeds 10.5 degrees. The on speed symbol, the center donut-shaped green lamp, lights between 11.0 and 9.0 degrees. The high speed symbol, the bottom inverted V-shaped amber lamp, will light when the angle of attack is less than 9.5 degrees. The indexer lamps function only when the landing gear is in the down position. A dimming rheostat, located on the side of the indexer, controls the intensity of the lamps.

Accelerometer. The accelerometer, located adjacent to the angle of attack indicator, provides normal "g" (load factor) information. The "g" forces being sustained by the aircraft are continuously shown by the acceleration scale read against a fixed index line. The tape scale is graduated from minus 4 to plus 10 "g". The presentation on the digital readout is from 0.0 to 9.0 "g". During negative acceleration, a shutter will appear across the acceleration readout window. The accelerometer and readout window are actuated by electrical signals from the remote accelerometer.

Note

During abrupt pitching maneuvers, the "g" build-up may exceed the 2 "g" per second maximum speed of the accelerometer tape. If this occurs, the accelerometer indicator readings will be less than actual aircraft acceleration levels.

Mach Indicator. The mach scale in the center of the airspeed-mach indicator indicates true mach number, which is shown on a moving scale and is read against the fixed index. The scale is calibrated in hundredths and shows numbers in the tenths from 0.4 through 3.5. At speeds below mach 0.4, the scale will continue to read 0.4. The moving scale is operated by electrical signals from the CADC. A command mach marker and command mach readout window indicate manually selected command mach. The command mach marker remains at the top or bottom of the display column until the selected command mach comes into view on the mach scale, at which time it will synchronize and move with the scale. The selected true mach is numerically displayed in the command mach readout window at all times. Command mach setting is controlled manually by the command mach slewing switch under the command mach readout window. When selecting a command mach number, slewing speed is proportional to the amount the slewing switch is displaced from its normal center position. The maximum allowable mach is indicated by a diagonally striped maximum allowable mach marker which normally rests at the bottom of the mach scale. When maximum allowable speed is approached, the marker will climb toward the fixed index line. The maximum allowable mach marker will show on the scale depending on the aircraft wing sweep position, pressure altitude, and true temperature. The maximum allowable mach marker is operated by an electrical signal from the maximum safe mach assembly (MSMA).

Airspeed Indicator. The airspeed scale on the right column of the airspeed-mach indicator indicates airspeed on a moving scale read against a fixed index. The scale is calibrated in 10 knot increments and displays numerals at each 20 knot intervals from 100 to 200 knots and each 50 knot interval from 200 through 1000 knots. At speeds below 50 knots, the scale will continue to read 50. The airspeed scale is operated by electrical signals from the CADC. If there is a detected instrument failure or airspeed signal failure within the CADC, the IAS monitoring flag marked OFF will appear across the airspeed scale. A command airspeed marker and command airspeed readout window below the scale indicates selected command airspeed. Command airspeed setting is controlled by the command airspeed slewing switch under the command airspeed readout window. When selecting a command airspeed, slewing speed is proportional to the amount the slewing switch is displaced up or down from the center position. Once the command airspeed is set into the command airspeed readout window, the command airspeed marker remains at the top or bottom of the display column until the selected command airspeed comes into view on the moving scale, at which time it will synchronize and move with the reading on the scale. This will be the same reading as shown in the readout window.

Note

If the slewing switch is moved to the detented position on the right, the command airspeed marker will align with the fixed index and continuous digital presentation of the airspeed will then be displayed in the readout window.

Altitude-Vertical Velocity Indicator (AVVI).

The AVVI (figure 1-31) provides remote reading presentations of altitude and vertical velocity on vertical moving scales. Readout windows across the bottom of the indicator present digital readout of barometric pressure and command altitude. A barometric pressure set knob and command altitude slewing switch are also located on the bottom of the indicator. Signals for operation of the moving scales, markers, and readouts are provided from the CADC. A spring-loaded OFF warning flag will appear across the face of the coarse altitude scale in the event of malfunction or power failure to the indicator. The barometric pressure reading is set by a knob marked BARO, located on the lower left corner of the indicator, and is numerically displayed in the barometric pressure readout window above the knob.

WARNING

A mechanical failure within the altitude-vertical velocity indicator may not cause the flag to appear, even though the indicator reading is unreliable. If a failure is suspected, rely on the standby altimeter using the position error shown in Appendix I, Part I. The radar altimeter also may be used since it provides an absolute indication of distance above the terrain at altitudes below 5000 feet.

Presentations on the face of the indicator are from left to right as follows:

Vertical Velocity Indicator. The vertical velocity indicator is located on the left side of the altitude-vertical velocity indicator. The instrument indicates climb or

dive velocities from 0 to 1500 feet per minute by means of a moving index pointer to the right of a vertical fixed scale. The scale is graduated in increments of one hundred feet from 0 to 1.5 thousands. When the vertical velocity exceeds this scale, the pointer index will move to the top or bottom of the instrument to a readout window where a moving scale, graduated in thousands of feet from 2 to 40 thousand feet per minute, will indicate the rate of climb or descent. The instrument receives information from the CADC.

Vernier Altimeter. The altitude scales in the center of the altitude-vertical velocity indicator indicate aircraft pressure altitude which is read on the altitude scale against a fixed index line. The vernier scale is calibrated in 50 foot graduations and indicates each hundred foot level from 0 to 1000 feet. The coarse scale is calibrated in 500 foot graduations and indicates each thousand foot level from — 1000 through + 120,000 feet. Both the vernier and coarse scales are operated by electrical signals from the CADC. A command altitude marker and the command altitude readout window below the scale indicate manually selected command altitude. The command altitude numerals are controlled manually by the command altitude slewing switch under the command altitude readout window. When selecting a command altitude, slewing speed of the command marker and readout window numerals is proportional to the amount the slewing switch is displaced from center. The command altitude marker remains at the top or bottom of the display column until the selected command altitude comes into view on the altitude scale, at which time it will synchronize and move with the scale. The selected command altitude is numerically shown in the altitude readout window at all times.

Gross Altimeter. The gross altimeter, located on the right side of the altitude-vertical velocity indicator, is a thermometer-type altitude index which shows aircraft altitude against a gross altitude scale. It is operated by electrical signals from the CADC. The gross altitude scale is calibrated in thousands of feet and numerically indicates 10,000 foot levels from 0 to 120,000 feet. Command altitude is indicated by a double line command altitude marker and is simultaneously shown and operated in conjunction with the command altitude marker on the vernier altimeter.

Attitude Director Indicator (ADI).

The ADI (figure 1-31) is a remote indicating instrument which displays attitude, heading, turn and slip, glide slope deviation, and bank and pitch steering information. The indicator includes an attitude sphere, turn and slip indicator, pitch and bank steering bars, miniature aircraft, glide slope indicator, warning flags, and a pitch trim knob. The attitude sphere displays pitch, bank, and heading in relation to the miniature aircraft. The pitch reference of the attitude sphere to the miniature aircraft may be adjusted with the pitch trim knob. The turn and slip indicator, located in the bottom of the ADI, receives turn signals directly from a remotely located rate-of-turn transmitter and is designed for a 4-minute turn. Pitch and bank steering commands from other systems are processed by the instrument system coupler and routed through the flight director computer to the pitch and bank steering bars and glide slope deviation indicator. Refer to "Instrument System Coupler Mode Selector Knob" and "Instrument System Coupler Pitch Steering Mode Switch," this section, for ADI indications during various modes of operation. Attitude data to the ADI is received directly from either the bomb nav system inertial reference unit (IRU) or the auxiliary flight reference system (AFRS) depending on the position of the flight instrument reference select switch. Normal operation of the ADI is with this switch in the PRI position which provides the instrument with signals from the IRU. An off warning flag indicates loss of power to the ADI when data is being supplied from the bomb nav system. When data is being supplied from the AFRS, the warning flag indicates loss of power to the ADI or that the data is unreliable. The attitude monitor system will also light the attitude caution lamp in the event of power failure, or if data is unreliable in the AFRS mode.

Note

The ADI OFF flag may not come into view if the data becomes unreliable while in the AFRS mode because of internal current drain in the attitude monitor system.

It is possible to have failures within the ADI, AFRS or IRU that can result in erroneous or complete loss of attitude reference without the presence of a warning flag or caution lamp indication. Other indications such as unrealistic or rapid changes in winds, ground speeds, or position, excessive radar cursor drift or unusual radar video uniformity, sudden attitude changes while on autopilot, or frequent flyups while on TF may also indicate a possible erroneous attitude reference. When abnormal disagreement between the ADI and standby attitude indicator is encountered without a warning flag or caution lamp indication, the aircraft should be returned to level flight using basic flight instruments. Do not assume either indicator is reliable until the aircraft is straight and level and one of the indicators is determined to be accurate. If the above checks have determined a malfunctioning IRU, the ADI source should be switched to the AFRS. A continuing attitude discrepancy indicates an ADI malfunction; therefore, the standby attitude indicator should be used. An attitude monitor system is pro-

Instrument Warning Flag Analysis

INSTRUMENT	*WARNING FLAG*	*FLAG CONDITION*	*DISPLAY VALIDITY*	*RECOMMENDED ACTION*
Airspeed Mach Indicator	Power warning flag (Mach tape)	In View	All displays not reliable.	Use standby airspeed indicator.
	Airspeed warning flag (Airspeed tape)	In View (CADS) caution lamp out	Airspeed display not reliable.	Use standby airspeed indicator.
		In View (CADS) caution lamp on	Only acceleration display reliable.	Use standby airspeed indicator.
		Out of View (CADS) caution lamp on	Only airspeed and normal acceleration displays reliable.	Use airspeed and altitude for mach number determination.
Altitude Vertical Velocity Indicator	Altitude Warning Flag	In View	All displays not reliable.	Use standby altimeter and standby vertical velocity indicator.
Attitude Director Indicator	Attitude (OFF) Warning Flag	In View	Only turn and slip reliable.	Use standby attitude indicator.
	Course Warning	In View	Bank steering bar not reliable.	Use HSI course deviation indicator.
	Glide Slope Warning Flag	In View	Glide slope indicator not reliable. If TF submode engaged, pitch steering bar not reliable.	Use other landing mode or system.
Horizontal Situation Indicator	Power (OFF) Warning Flag	In View	All displays not reliable. Also, ADI bank steering bar not reliable if selected ISC mode requires manually setting HSI heading or course.	Use ADI heading and BDHI TACAN bearing and distance.
	Range Warning Flag	In View	Range indicator not reliable.	Use bearing-distance-heading indicator.
	Course Warning Flag	In View	Course deviation indicator not reliable.	Use ADI bank steering bar.
Bearing-Distance-Heading-Indicator	Power (OFF) Warning Flag	In View	All displays not reliable.	Use HSI.
	Range Warning Flag	In View	Range indicator not reliable.	Use HSI.
Standby Attitude Indicator	Power (OFF) Warning Flag	In View	Displays not reliable.	Use ADI.

Figure 1-32.

vided to compare pitch and roll signals to the ADI with those to the standby attitude indicator. Refer to "Attitude Monitor System," this section.

WARNING

Frequent cross checks between the ADI, the standby attitude indicator and other basic flight instruments should be made to detect possible malfunctions. Failure to detect a malfunction and take corrective action could result in a flight attitude from which the aircraft cannot be recovered.

The ADI operates on 115 volt ac power from the essential ac bus.

Horizontal Situation Indicator (HSI).

The HSI (figure 1-31) is a remote indicating instrument which displays course, heading, distance, and bearing information. The indicator includes a compass card, course and heading set knobs, course arrow, to-from indicator, lubber lines, bearing pointer, course deviation indicator and scale, range indicator and course selector windows, warning flags, and an aircraft symbol. The compass card is servo-driven and receives magnetic heading signals directly from either the bomb nav system or auxiliary flight reference system. Aircraft heading or its reciprocal are read under an upper and lower lubber line. The aircraft symbol is fixed and is oriented to the nose of the aircraft. A heading set knob is provided to set a heading marker to the desired heading. Once it is set, the marker rotates with the compass card. A course set knob is provided to set the course arrow and digits in the course selector window to the desired course. Once set, the arrow will rotate with the compass card. The shaft of the course arrow provides course deviation indications. The reciprocal course may be read off the tail of the arrow. An unreliable course signal or loss of the course signal to the indicator will cause a warning flag to appear in the upper center of the indicator. The bearing and distance to TACAN stations are displayed by the bearing pointer and range indicator window. Loss of the TACAN signal or an unreliable signal will cause a range warning flag to appear in the range indicator window. Loss of power to the HSI will cause an OFF warning flag to appear on the right side of the instrument. (Refer to "Instrument System Coupler Mode Selector Knob", this section, for HSI indications during various modes of operation.)

Instrument System Coupler (ISC) Pitch Steering Mode Switch.

The ISC pitch steering mode switch (12, figure 1-6), located on the instrument system coupler control panel, is a three-position switch marked ALT REF (altitude reference), OFF, and TF (terrain following). The switch is solenoid held in either the ALT REF or TF position, when used with a compatible position of the ISC mode selector knob. When the switch is placed in the ALT REF position, pitch steering commands, referenced to the pressure altitude at the time the switch is engaged, will be displayed on the pitch steering bars on the attitude director indicator (ADI) and optical display sight (ODS). The ALT REF position is compatible with all positions of the ISC mode selector knob except TKR RV; however, when making an ILS or AILA approach, the switch will automatically return to OFF when the glide slope is intercepted. The switch will not hold in the ALT REF position if either TFR channel is in the TF mode and is operating normally. When the switch is placed to the TF position, pitch steering commands referenced to the altitude setting of the terrain following radar will be displayed on the pitch steering bars on the ADI and ODS. The TF position is compatible with all positions of the ISC mode selector knob except ILS, AILA, and TKR RV. However, with the knob in CRS SEL NAV, BOMB/NAV and CRS LINE position, the switch will return to OFF when a weapon delivery pull-up signal is generated.

Note

Altitude reference submode limits are ±500 feet from the reference pressure altitude. If the set limits are exceeded, the reference altitude will change by the amount that the altitude limits are exceeded.

Instrument System Coupler (ISC) Mode Selector Knob.

The ISC mode selector knob (12, figure 1-6), located on the ISC control panel, has eleven positions. Ten positions of the knob are activated and are marked OFF, ILS, AILA, TACAN, CRS SEL NAV, BOMB/NAV, CRS LINE, MAN CRS, MAN HDG, and TKR RV. An unmarked position is provided for the installation of new equipment. The knob must be depressed to change positions. For ISC modes versus ADI and HSI indications refer to figures 1-33 and 1-34. For ADI and HSI steering indication limits refer to figure 1-35. The knob positions provide the following functions:

ISC Mode Selector Knob Positions vs ADI Indications

ISC MODE SELECTOR KNOB POSITIONS

	OFF	*ILS*	*AILA*	*TACAN*	*CRS SEL NAV*	*BOMB/ NAV*	*CRS LINE*	*MAN CRS*	*MAN HDG*	*TKR/ RV*
Bank Steering Bar	Out of View	Steer to Loc	Com steer fr B/N to set crs	Steer to TACAN crs	Com steer fr B/N to set crs	Com steer to dest	Com steer to crs line	Com steer to selected crs	Hdg fr HSI and FDC	Target steer fr attack radar
Pitch Steering Bar	Out of View	In view in alt ref and when G/S beam inter	Out of view unless TF or ALT REF is selected							Target elev fr attack radar
Heading Reference Scale	Mag hdg from B/N or mag hdg from AFRS if pri hdg caution lamp is lighted, or mode selector knob is in TACAN or TKR RV.									
Attitude Sphere	Pitch and roll from B/N system when selected, or from the AFRS when PRI ATTITUDE caution lamp is lighted, or AUX is selected.									
Glide Slope Indicator	Out of View	G/S fr G/S rec	Dev fr set G/S fr B/N	—Out of View—						
Course Warning Flag	Out of View	Out of view when loc adeq sig stren	Out of view loc valid fr B/N	Out of view when TACAN adeq sig stren	Out of view when lateral steering is valid fr B/N				Out of view	Out of view
Glide Slope Warning Flag	Out of View	Out of view when G/S adeq sig stren	Out of view when dev fr G/S is valid	—Out of View—						
Attitude Warning Flag	Normal condition—Out of view. Abnormal condition—In view, disregard ADI and use stby att ind for attitude and ODS for pitch and azimuth steering.									
ISC Pitch Steer Sw	TF or ALT REF	ALT REF prior to G/S (Note 1)			TF or ALT REF					Not Useable

Note 1: ALT REF is useable prior to intercepting G/S. When G/S is intercepted the ALT REF is automatically disengaged by the FDC in ILS or B/N in AILA mode.

Figure 1-33.

ISC Mode Selector Knob Positions vs HSI Indications

ISC MODE SELECTOR KNOB POSITIONS

	OFF	*ILS*	*AILA*	*TACAN*	*CRS SEL NAV*	*BOMB/ NAV*	*CRS LINE*	*MAN CRS*	*MAN HDG*	*TKR RV*
Course Set Knob	Not Used	Used to set ldg approach crs in crs set window		Used to set TACAN crs	Used to set desired crs	Not Used	Not Used	Used to set des crs	Not Used	
Heading Set Knob	—Not Used—								Used to set desired mag hdg	Not Used
Compass Card	Heading fr B/N sys when sel, or mag hdg fr the AFRS when pri hdg caution light is lighted or AUX is sel, except TACAN or TKR RV modes which uses AFRS mag hdg only. (Note 1)									
Course Select Window & Course Arrow	Mag grd track fr B/N	Ldg approach crs man set by the crs set knob		TACAN crs man set	Desired crs man set	Cur grd trk	Cur grd trk	Fix at last set pos	Ground track from B/N	
Course Deviation Indicator	Not Used Ctr	Loc dev fr loc rec	Loc dev fr B/N	TACAN dev fr sel crs	Lateral dev fr B/N				Not Used—Center	
Heading Marker	Indicates computed course from B/N.								Mag hdg man set	Bearing to tkr fr attack radar
Course Deviation Bar Flag	Out of view			Out of view by TACAN	Out of view by B/N				Out of view by FDC	
Power Off Warning Flag	Normal Condition—Out of view. Abnormal Condition—When in view, disregard HSI and ADI in MAN/HDG mode and use BDHI or ODS.									
Range Ind and Warning Flag	Indicates distance to TACAN station.									
Bearing Pointer (TACAN)	Indicates bearing to TACAN station.									
To—From Indicator (TACAN)	Out of view—Not Used			To or fr TACAN sta	Out of view—Not Used					
Note 1: Primary mag heading display will be true heading plus or minus mag variation.										

Figure 1-34.

In the OFF position, the steering bars and OFF flags are biased out-of-view on the ADI and ODS leaving attitude and heading displays for use during GCA and other similar type operations.

The ILS (instrument landing system) position provides the capability of flying ILS approaches to runways equipped with localizer and glide slope transmitters. Localizer steering commands are displayed by the bank steering bars on the ADI and ODS and course deviation information is displayed on the course deviation indicator of the horizontal situation indicator (HSI). Pitch steering commands will be displayed on the pitch steering bars on the ADI and ODS if the pitch steering mode switch is in the ALT REF position. When the glide slope beam is intercepted, the pitch steering mode switch, if on, will return to OFF and glide slope steering commands will then be displayed on the pitch steering bars on the ADI and ODS and glide slope deviation will be displayed on the ADI glide slope deviation indicator and on the left deviation indicator on the ODS.

Note

- Once the glide slope is intercepted, a glide slope deviation of more than two dots as measured on the glide slope deviation scale will cause the pitch steering bar on the ADI and ODS to drive out of view and remain out of view until the aircraft is repositioned back on the glide slope.
- Also at glide slope intercept the bank steering bar reference is switched from normal (25 degrees) to approach reference (15 degrees). Refer to figure 1-35.

With the radar altimeter operating and set for a minimum altitude penetration, the pitch steering bars on the ADI and ODS will indicate a flyup command and the radar altitude low warning lamp will light when the aircraft penetrates the set altitude. If a pull-up is then initiated, the flyup command will be terminated and the warning lamp will go out when the aircraft is above the altitude set on the radar altimeter. The pitch steering bar commands will be regained once the glide slope indicator is recentered or by placing the pitch steering mode switch to ALT REF when level off altitude is reached. In the event an ILS approach or AILA is begun from above 5000 feet absolute altitude, the radar altitude low warning lamp will light, and the pitch steering bars on the ADI and ODS will momentarily indicate a flyup command when the aircraft descends through 5000 feet. In this case, the flyup command can be terminated, and the lamp will go out by taking the above action.

The AILA (airborne instrument low approach) position provides the capability of making instrument letdowns and approaches to runways not equipped with ground based letdown systems. The bomb nav system, in conjunction with the attack radar, is used to correct the present position longitude and latitude and will furnish simulated localizer and glide slope information to provide the same indications on the ADI, ODS, and HSI as when using the ILS position.

Note

During AILA approaches, when the bomb nav system is furnishing simulated localizer, the pitch steering bar will be stowed if the deviation from the simulated localizer exceeds ± 2.5 degrees or if glide slope deviation exceeds ± 0.9 degrees (2 dot glideslope deviation equals 0.7 degrees).

The TACAN (tactical air navigation) position provides the capability of making instrument approaches and flying a selected course to or from a TACAN station. The course arrow and the course selector window are set to the desired course to be flown using the course set knob. Course steering commands are displayed on the bank steering bars on the ADI and ODS and course deviation information is displayed on the course deviation indicator and bearing pointer on the HSI. Distance from the TACAN station is displayed in the range indicator window on the HSI. The bearing pointer will indicate the magnetic bearing to the station.

Note

The flight director computer is limited to a 40 degree bank angle.

The CRS SEL NAV (course select navigation) position provides the capability of approaching a selected destination along a selected course other than the most direct route. This provides the capability of avoiding weather, obstacles, and enemy areas. To commence the course select navigation procedure, depress the destination selected sequence point pushbutton, select CRS SEL NAV on the instrument system coupler and set the selected course in the HSI course selector window. This establishes a course signal to the bomb nav system where a course error signal is developed. When the selected course is set in the HSI, a right or left steering signal is generated for the ADI and ODS bank steering bars. This signal depends on (1) aircraft position in relation to selected course, and (2) aircraft ground track in relation to the ground track required to make good on approach flight path. The HSI course deviation bar will be displaced two dots until within

HSI and ADI Steering Limits

	ILS		AILA		TACAN	CRS SEL NAV	BOMB NAV	CRS LINE	MAN CRS	MAN HDG	TKR RV	ALT REF Sub Mode
	Norm.	Appr.	Norm.	Appr.								
2 Dot Displ of (HSI) CRS DEV IND	2.5°	2.5°	2.5°	2.5°	8° to 10°	10°	Nav 10° Bomb 5.0°	10°	10°	Not Used	Not Used	Not Used
2 Dot Displ of ADI G/S DEV IND	0.7°	0.7°	0.7°	0.7°	Not Used							
ADI Bank Steering Pointer Roll Limit	25°	15°	30°	15°	40°	30°	Nav 30° Bomb 42°	30°	30°	25°	60°	
ADI Pitch Steering Pointer Pitch Limit	20°	20°	20°	20°	Not Used						20°	20°

Specified limits are established by the flight director computer and the applicable sub-system supplying the signals and are not affected by the ODS.

Figure 1-35.

10 degrees of the selected course. To intercept the selected course at a predetermined position it is necessary to maintain the bank steering bars centered. When the aircraft is on the selected course the heading marker and course arrowhead will be aligned ± drift correction.

The BOMB/NAV (bombing navigation) position provides steering information from the bomb nav system to steer the aircraft to a weapon release point or to a selected destination depending on a position of the bomb nav function select knob. When the bomb nav function select knob is in the BOMB RADAR or NAV position, course steering commands are displayed on the bank steering bars on the ADI and ODS, and course deviation is displayed on the course deviation indicator on the HSI. The course set knob and the heading set knob are inoperative in the BOMB/NAV mode. The course arrow and course selector window display current magnetic ground track from the bomb nav system.

The CRS LINE (course line steering) position provides steering commands to steer the aircraft along the computed great circle course between the past destination and the current destination. If the aircraft deviates from this computed great circle course, this mode will provide steering commands to return to course by the shortest route.

The MAN CRS (manual course) position provides the capability of flying a manually selected course instead of a bomb nav system computed course. This position can be utilized to fly a constant course while taking a fix, changing destination, or working a navigation problem. The desired course is set in the course selector window of the HSI. The selected course is compared with actual course by the bomb nav system, and an error signal is provided to display course steering commands on the bank steering bars on the ADI and ODS and course deviation information on the course deviation indicator on the HSI.

The MAN HDG (manual heading) position provides the capability of flying any desired heading when use of the bomb nav system is impractical or inefficient or when the system is inoperative. The heading marker on the HSI is set to the desired heading on the compass card by using the heading set knob. Turn the aircraft to center the bank steering bars on the ADI and ODS. Any deviation from this heading will generate a steering command on the bank steering bars on the ADI and ODS. If the bomb nav system is inoperative, the course set knob should be used to set the desired heading in the course selector window. This will provide a digital readout of the heading and align the course arrow with the heading marker to reduce the possibility of heading confusion.

The TKR RV (tanker rendezvous) position provides the steering capability to a tanker for air refueling rendezvous. In this mode the HSI heading marker is driven by a bearing signal from the attack radar which also supplies a signal to indicate the necessary steering commands on the bank steering bars on the ADI and ODS to steer the aircraft to the tanker.

Note

The bank steering bars on the ODS and ADI will oscillate after the attack radar locks on the tanker signal. This is normal and is due to the automatic angle tracking of the radar. The amplitude of the oscillation will increase as the range is decreased; however, the frequency of the oscillation will remain constant.

The pitch steering bars on the ADI and ODS will be activated and indicate the necessary pitch steering correction (aircraft angle of attack plus radar antenna tilt angle) to rendezvous with the tanker.

Instrument Test Button.

The instrument test button (7, figure 1-27), located on the ground check panel, is provided for ground checking and troubleshooting of the integrated flight instruments, the instrument system coupler, the attitude monitor system, and the total temperature indicator. Depressing and holding the button will provide a set of predetermined indications on the above instruments. Test indications on the ADI and HSI will be compatible with the normal indications expected for each mode selected by the ISC mode selector knob. Tests selected with the button are completely independent of the CADC.

ATTITUDE MONITOR SYSTEM.

An attitude monitor system is provided. This system consists of a monitor unit installed behind the main instrument panel and an attitude caution lamp on the main caution lamp panel (figure 1-36). The attitude monitor system provides an indication of the reliability of the ADI, and the aircraft pitch and roll data signals to the ADI and the standby attitude indicator. With power on the aircraft, the attitude monitor system continuously compares these signals and also monitors the voltage to the ADI attitude warning flag. If either the pitch or roll signals to the ADI differ from those to the standby attitude indicator by more than 7 ($\pm$1.4) degrees or if the ADI attitude warning flag comes into view, a caution lamp will light. When the lamp is lighted, the word ATTITUDE is visible. The instrument test button is used to test the function of the attitude monitor unit. When the button is depressed, the ADI attitude warning flag will come into view and the attitude caution lamp will light.

Note

When operating on emergency generator power, the attitude monitor system may not be fully operable and the attitude lamp may not light; however, the ADI attitude warning flag and the attitude lamp, if lighted, should be considered valid.

WARNING, CAUTION, AND INDICATOR LAMPS.

In order to keep instrument surveillance to a minimum, warning, caution, and indicator lamps are located throughout the cockpit. All of these lamps except the master caution lamp are described under their respective systems. For location of the lamps throughout the cockpit, see figure 1-36.

MASTER CAUTION LAMP.

The master caution lamp (23, figure 1-6), located on the left main instrument panel, will light after a 3 second delay, to alert the crew that a malfunction exists when any of the individual caution lamps on the caution lamp panel lights to indicate a malfunction. The lamp will remain lighted as long as an individual caution lamp is on; however, it should be reset as soon as possible by depressing the face of the lamp so that other caution lamps can be monitored, should additional malfunctions occur. The intensity of the lamp can be adjusted with the malfunction and indicator lamp dimming switch. The lamp can be checked by depressing the malfunction and indicator lamp test button.

Malfunction and Indicator Lamp Dimming Switch.

The malfunction and indicator lamp dimming switch (2, figure 1-37), located on the lighting control panel, is a three-position switch marked BRT (bright) and DIM and is spring-loaded to an unmarked center position. The switch controls the light intensity, either bright or dim, of all the warning, caution, and indicator lamps in the cockpit except the TFR channel failure caution lamps, the IFF reply and test lamps, and the NWS A/R indicator lamp.

Note

All lamps which are controlled by this switch are automatically set to bright under the following conditions:

- The internal lighting control knob (FLT INST) is off.
- Aircraft power is turned off.

Malfunction and Indicator Lamp Test Button.

The malfunction and indicator lamp test button (3, figure 1-37), located on the lighting control panel, is provided to check the landing gear warning horn and all warning, caution and indicator lamps in the cockpit except the following:

- TFR Channel Failure Caution Lamps
- IFF Reply and Test Lamps
- IRRS Ready/Test Indicator Lamp
- Engine Fire Pushbutton Warning Lamps
- All lamps on the attack radar scope, navigation, ECM and RHAW panels.

The malfunction and indicator lamp test button is also used to ground check the stall warning system.

Note

During ground checks, with the flight control system switch in T.O. & LAND or with the slats down, depressing the malfunction and indicator lamp test button may cause the rudder to deflect due to an AYC input. In addition, when the test button is released, the YAW CHANNEL caution lamp may remain lighted. This lamp can be reset with the damper reset button.

LIGHTING SYSTEM.

EXTERIOR LIGHTING.

The exterior lights include position lights, formation lights, anti-collision/fuselage lights, air refueling lights, landing lights and a taxi light. The position lights consist of green lights in the right glove and wing tip, red lights in the left glove and wing tip and a white tail light. The wing tip position lights will light when the wing sweep angle is between 16 and 30 degrees. When the wings are swept aft of 30 degrees the wing tip light will go out and the glove light will light. The reverse will occur as the wings are swept forward. The formation lights consist of a set of two lights, located on the upper and lower surfaces of each wing tip, and four lights located forward and aft of each side of the fuselage. The lights in the wing tips correspond to the color of the left and right position lights. The fuselage lights are amber. Two anti-collision/fuselage lights, one located on top and one located on the bottom of the fuselage, serve as white fuselage lights when retracted and flashing red anti-collision lights when extended. Two air refueling lights mounted in the air refueling receptacle are provided for night refueling operations. A limit switch on the air refueling receptacle door provides power to the receptacle light control knob when the door is open. Two landing lights and a taxi light are located on the nose landing gear. A switch on the nose gear down lock will turn the lights off if they are on when the gear is retracted.

Position Light Switches.

Three position light switches (4, figure 1-37), are located on the lighting control panel. Two switches, labeled WING and TAIL, have three positions, marked BRT (bright), OFF and DIM, for selecting the desired intensity of the position lights. The third switch is a two position switch marked FLASH and STEADY to control the operation of the position lights. Placing the switch to FLASH causes the position lights to flash.

Position Lights/Stores Refuel Battery Power Switch.

The position lights/stores refuel battery power switch (5, figure 1-27), located on the ground check panel, has three positions marked POS LIGHTS, NORM and STORES REFUEL. Placing the switch to the POS LIGHTS position will supply battery power to the position lights for added safety during ground handling. Placing the switch to NORM connects these circuits to the essential dc bus. The switch is held in the NORM position when the ground check panel door is closed. For a description of the STORES REFUEL position of the switch refer to the Fuel Supply System, this section.

Formation Lights Switch.

The formation lights switch (5, figure 1-37), located on the lighting control panel, provides selection of the desired intensity of the lights. The switch is marked BRT (bright), OFF and DIM.

Anti-Collision Lights Switch.

The anti-collision lights switch (7, figure 1-37), is located on the lighting control panel. The switch is labeled ANTI-COLLISION and has one position marked OFF and an unmarked ON position. Placing the switch to ON causes the anti-collision lights to light, extend and rotate. Placing the switch to OFF causes the lights to retract, go out and stop rotating.

Fuselage Lights Switch.

The fuselage lights switch (6, figure 1-37), is located on the lighting control panel. The switch is labeled FUSELAGE and has a position marked OFF and an unmarked ON position. Placing the switch to ON, lights a white light in the top and bottom of the fuselage.

Warning, Caution and Indicator Lamps (Typ)

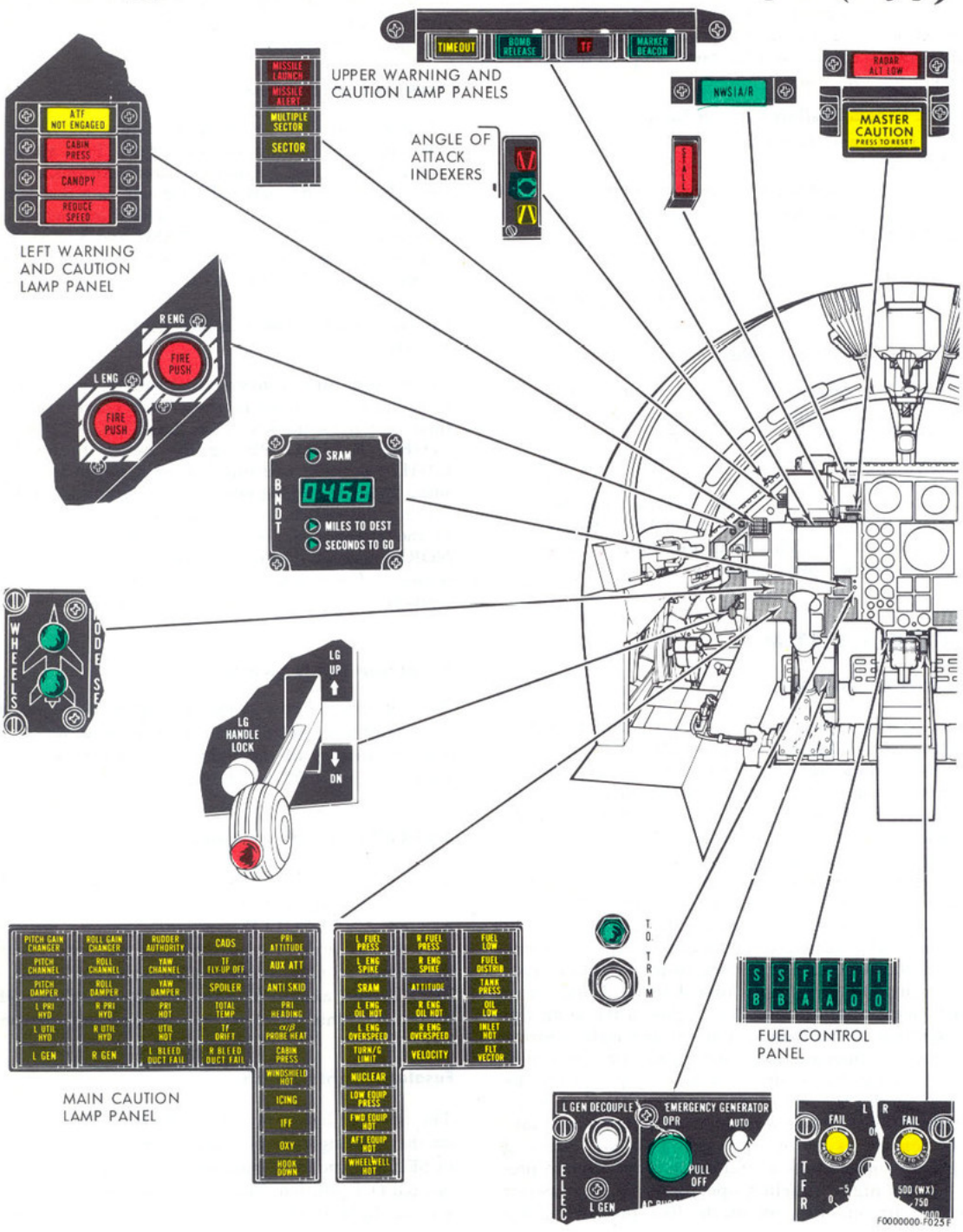


Figure 1-36. (Sheet 1)

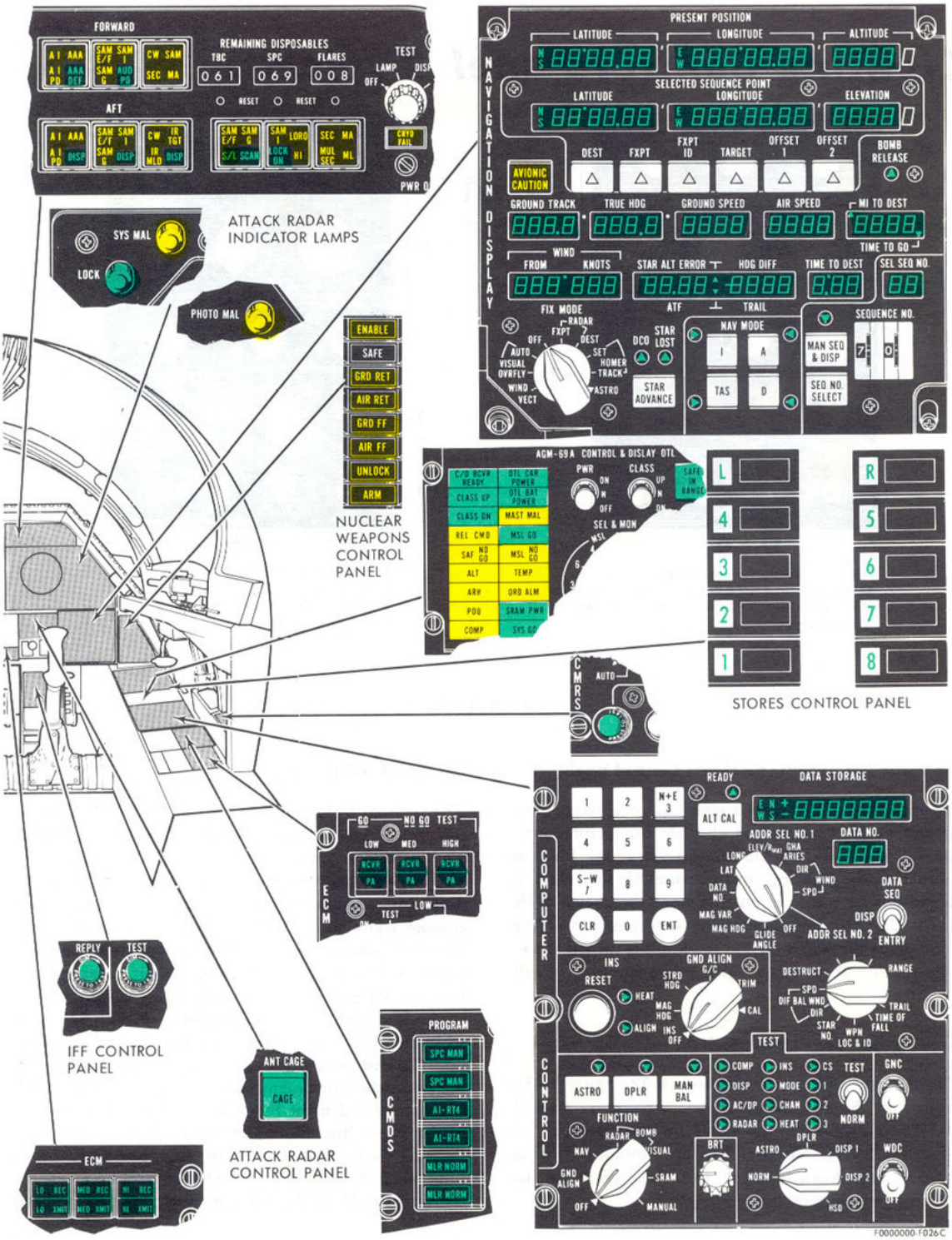
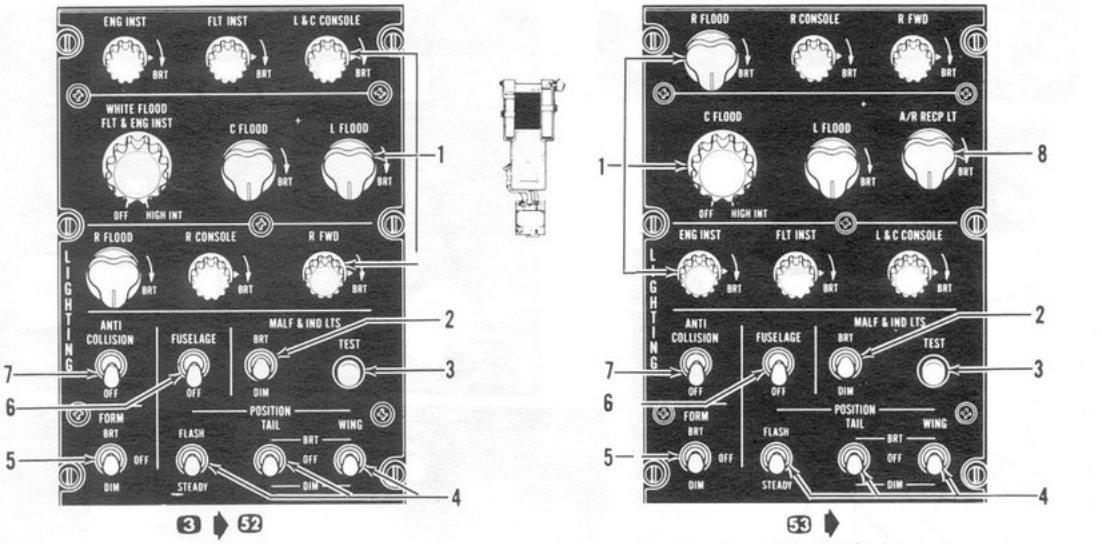


Figure 1-36. (Sheet 2)

Lighting Control Panel (Typical)



1. Internal Lighting Control Knobs.
2. Malfunction and Indicator Lamp Dimming Switch.
3. Malfunction and Indicator Lamp Test Button.
4. Position Lights Switches.
5. Formation Lights Switch.
6. Fuselage Lights Switch.
7. Anti-Collision Lights Switch.
8. Air Refueling Receptacle Lights Control Knob.

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Figure 1-37.

Air Refueling Receptacle Lights Control Knob.

On aircraft 3 ▶ 52, the air refueling receptacle lights control knob is located on the left main instrument panel (35, figure 1-6). On aircraft 53 ▶, the knob is located on the lighting control panel (3, figure 1-37). Regardless of the location, the knob is labeled A/R RECP LT. The full counterclockwise position of the knob turns the lights off. As the knob is turned clockwise the intensity of the lights varies from off to full brightness.

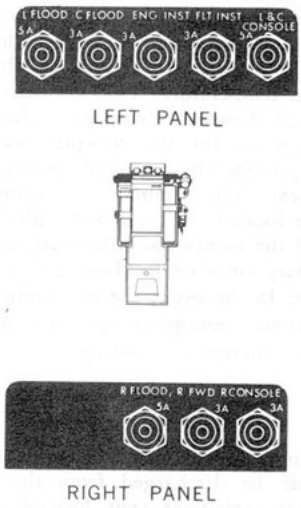
Landing and Taxi Lights Switch.

The landing and taxi lights switch (2, figure 1-73), located on the miscellaneous switch panel, is marked LANDING, OFF and TAXI. If the switch is left in either the LANDING or TAXI positions, a switch on the nose gear down lock will turn them off when the landing gear is retracted.

INTERNAL LIGHTING.

The internal lights include instrument panel and console lights, red and white flood lights and utility lights. The instrument panel and console lights consist of five circuits, each with an individual control knob, for the flight instruments, engine instruments, left and center console, right console and right main instrument panel. They are powered by 115 volt ac power from the right main ac bus through two fuse panels (figure 1-38) near the aft bulkhead. The flood lights consist of left, center and right red flood lights and high intensity white flood lights at various locations around the cockpit. The red flood lights provide cockpit lighting in the event the instrument panel and console lights fail. Each set of red flood lights has an individual control knob. The white flood lights provide high intensity lighting to prevent temporary blindness from lightning when flying in weather. One control knob adjusts the intensity of all the white flood lights. Both the red

Internal Lighting Fuse Panels (Typical)



F0000000-FI02

Figure 1-38.

and white flood lights receive 115 volt ac power from the ac essential bus. Two utility lights (50, figure 1-2 and 19, figure 1-30), one for each side of the cockpit, are provided for individual work lights. They are normally stowed on the left side of the aft console and on the right side of the right instrument panel but can be moved to various locations about the crew station. A slight rotation of the lamp will lock it in place on its mount. The front of each utility light can be rotated to change color from white to red and vice versa. A rheostat on the aft end of each light must be turned clockwise to turn the light on and set the desired intensity. The utility lights are powered by 28 volt dc from the engine start bus. Aircraft 53 ▸ are equipped with white instrument and flood lights only.

Internal Lighting Control Knobs.

The internal lighting control knobs (1, figure 1-37), located on the lighting control panel, control the various internal lighting circuits. The full counterclockwise position of each knob turns the lights off. As the knobs are turned clockwise, detent positions at spaced intervals vary the intensity of the lights from off to full brightness. Five of the knobs control the instrument panel and console lighting. Knobs are labeled and control the respective circuits as follows:

FLT INST—Left main instrument panel.

ENG INST—Engine instruments.

L&C CONSOLE—Left and center consoles.

R CONSOLE—Right console.

R FWD—Right main instrument panel.

The flood lights are controlled by individual knobs marked R FLOOD, C FLOOD and L FLOOD for the right, center and left flood lights respectively. On aircraft 3 ▸ 52, the knob marked WHITE FLOOD FLT & ENG INST controls all the white flood lights. On aircraft 53 ▸, the knob marked C FLOOD controls these white flood lights. In both cases this knob is marked OFF at the full counterclockwise position and HIGH INT (high intensity) near the full clockwise position. Turning the knob past HIGH INT turns all the white flood lights to maximum intensity. This will also turn on additional white flood lights. Once these lights are on, their intensity may be decreased by turning the knob counterclockwise. All of the white flood lights will be turned off when the knob is rotated to the OFF position.

CANOPY.

The canopy consists of left and right clam shell hatches hinged to a center beam assembly. The hatches open to a maximum of 65 degrees. Each hatch has an external and internal canopy latch handle for opening or closing. When the hatches are closed and latched, the internal handle locks in place to prevent inadvertent unlatching of the hatch inflight. Each hatch is manually raised or lowered with the aid of an air/oil counterpoise. The counterpoise will also hold the hatch in any position selected. A flush mounted plunger, located adjacent to the external handle, can be pushed in to unlock the internal handle from the outside.

INTERNAL CANOPY LATCH HANDLES.

Two canopy latch handles are located on the inside lower horizontal frame member of each canopy hatch (7, figure 1-2). An over-center spring-loaded canopy latch handle lock tab, in the face of each canopy latch handle, locks the handle in the latched position to prevent inadvertent opening inflight. When the lock tab is flush the canopy latch handle is locked. Pressing in on the forward part of the lock tab will cause the rear part of the tab to snap out, unlocking the canopy latch handle. The handle must then be pulled out and aft to a detent position to unlatch the hatch. Once the hatch is unlatched, pulling the handle further aft past the detent engages the counterpoise to aid in opening. When

the desired hatch position is attained, the handle must be returned to the detent position to lock the counterpoise and hold the hatch. Each handle is mechanically linked to a flush external canopy latch handle located outside of each hatch. Inflation of the canopy pressurization seal is automatically operated by closure of the canopy hatch. The actuator mounted on the hatch lower surface depresses a plunger in the canopy sill to inflate the seals and turn off the canopy unlock warning lamp.

CANOPY EXTERNAL LATCH HANDLES.

Two flush mounted canopy external latch handles are located on the lower horizontal frame member of each canopy hatch. Each handle is mechanically linked to its respective internal handle. Pressing in on the forward part of the handle will extend the rear portion of the handle so that it may be grasped to unlatch and raise the hatch. If the internal handle is locked in the closed position a flush mounted pushbutton plunger, located adjacent to the external canopy handle, is provided to unlock the internal canopy handle from the outside.

CANOPY UNLOCK WARNING LAMP.

A red canopy unlock warning lamp, located on the left warning and caution lamp panel (figure 1-36), will light when either hatch is not locked. When lighted, the word CANOPY is visible.

AIR CONDITIONING AND PRESSURIZATION SYSTEMS.

The air conditioning and pressurization systems (figure 1-39) combine to provide temperature-controlled, pressure-regulated air for heating, ventilating, pressurizing the cockpit and inflating the canopy seals. The system also provides air to the forward and aft electronic equipment bays, anti-icing and defog systems, windshield rain removal system, SRAM missile system and pneumatic pressure for throttle boost.

AIR CONDITIONING SYSTEM.

The air conditioning system provides temperature controlled air for the cockpit. The system also provides a temperature controlled flow of cooling air to the electronic equipment that requires a controlled environment for efficient operation. See figure 1-39. High pressure hot air is bled from the sixteenth stage compressor of each engine. This bleed air is directed through a tee fitting to a common duct and is routed through an air-to-air heat exchanger, where it is cooled by ram air that is circulated through the heat exchanger. The air is then routed through an air-to-water heat exchanger where it is further cooled and then enters the cooling turbine. The cooling turbine further cools the air to a temperature suitable for cooling the cockpit and electronic equipment bays. The cold air leaving the turbine passes through a water separator to remove most of the free moisture. A cabin temperature controller is fed signals from temperature sensors and from a pilot operated control panel. The temperature controller controls the setting of the cold air modulating valves. It also controls the setting of the cockpit hot air modulating and shutoff valve which allows hot air to mix with the refrigerated air stream, obtaining air at the selected temperature. This air then enters the cockpit through diffusers. Prior to engine start, the ground cooling air for the cockpit and electronic equipment is provided by a ground cooling cart or an auxiliary air cart. The receptacle for connecting the ground cart is located on the lower right side of the fuselage aft of the cockpit, and the receptacle for connecting auxiliary air is on the fuselage aft of the left main gear tire. In the event the air conditioning system malfunctions, emergency ram air operation is available for ventilation and cooling.

Note

- During operation at high power settings, steam may be discharged from the air-to-water heat exchanger vent located on the lower side of the fuselage between the weapons bay doors and the main landing gear door. This is a normal condition and should be no cause for concern.
- During operations under humid conditions, rapid increases in power may induce flash freezing in the water separator causing a loss of cooling air to the cabin and electronic equipment. Normally flash freezing is of short duration (2-3 minutes), and is self-correcting. This condition may be diagnosed by observing loss of cabin airflow followed by lighting of the forward equipment hot caution lamp from 30 to 120 seconds later. The duration of this condition may be reduced by selecting full warm on the cabin temperature selector. Refer to "Caution Lamp Analysis," Section III.
- During operation at low or idle power settings, the air conditioning system control valves will modulate toward an open position. When engine power is advanced, high airflow will occur until the cabin temperature control valves return pressure to normal. During high cabin airflow, cabin pressure may vary ±500 feet.

Cabin Air Distribution Control Lever.

A cabin air distribution control lever (49, figure 1-2), located on the aft bulkhead, controls distribution of airflow in the cockpit. The lever is labeled CABIN

AIR DISTR and has two positions marked FWD DEFOG and AFT. The normal position of the lever is the AFT position. In this position, airflow into the cockpit is separated between the rear bulkhead diffusers and the windshield defog system with approximately 85 percent directed to the diffusers. Moving the lever towards the FWD DEFOG position will decrease airflow through the air diffusers and increase airflow through the defog system. When the lever is in the full forward position all the airflow will be directed through the defog system. Although the AFT position is considered normal to obtain maximum airflow, desired crew comfort is accomplished by selecting any intermediate position between FWD DEFOG and AFT.

Note

During operation at high power settings, and when airflow to the cabin is high, moisture may be sprayed on the windshield and in the crew members' face. Moving the lever toward the aft position will decrease this possibility.

Air Source Selector Knob.

The air source selector knob (2, figure 1-40), located on the air conditioning control panel, has six positions marked OFF, L ENG, BOTH, R ENG, RAM, and EMER. The knob controls bleed air source or allows selection of emergency ram air operation when the normal system is not operating. The knob controls a series of valves which operate as follows in the different knob positions:

- In the OFF position, the left and right bleed air check and shutoff valves are closed. The cold air modulating valve is open (no flow). The cabin hot air modulating and shutoff valve is modulating (no flow). The pressure regulating and shutoff valve is closed.
- In the L ENG position, the left engine is the source of bleed air, and the right bleed air check and shutoff valve is closed. The cold air modulating valve and the cabin hot air modulating and shutoff valve will be modulating in response to the position of the temperature control knob. The pressure regulating shutoff valve will be regulating.
- In the BOTH position, the left and right bleed air check and shutoff valves are open. The cold air modulating valve and the cabin hot air modulating and shutoff valve will be modulating in response to the position of the temperature control knob. The pressure regulating and shutoff valve will be regulating.
- In the R ENG position, the right engine is the source of bleed air and the left bleed air check and shutoff valve is closed. The cold air modulating valve and the cabin hot air modulating and shutoff valve will be modulating in response to the position of the temperature control knob. The pressure regulating and shutoff valve will be regulating.
- In the RAM position, the engine bleed air check and shutoff valves are open. The cold air modulating valve is open (no flow). The cabin hot air modulating and shutoff valve is modulating in response to the position of the temperature control knob. The pressure regulating and shutoff valve is closed. The ram air door is open. The RAM position will dump cabin pressure and allow combined ram air flow and regulated engine bleed air to ventilate the cabin. Temperature control of this air is available by using the temperature control knob to control the amount of engine bleed air mixed with ram air. In the RAM position bleed air pressure is available to the wing seals, fuel tank pressurization system, electronic equipment, windshield wash and rain removal, throttle boost, and canopy seals.
- In the EMER position, the engine bleed air check and shutoff valves are closed and the ram air scoop is open. The EMER position dumps cabin pressure and allows ram air flow to ventilate the cabin. Temperature control is not maintained in this mode and heating is not available.

CAUTION

In OFF or EMER, air pressure is not available to the wing seals, fuel tank pressurization system, electronic equipment, windshield wash and rain removal, throttle boost, SRAM missile or canopy seals.

Air Conditioning System Mode Selector Switch.

The mode selector switch (6, figure 1-40), located on the air conditioning control panel, is a three position switch marked AUTO, OFF, and MAN. The switch is a lever-lock type switch which must be pulled out to change positions. In the AUTO position, the cockpit temperature is automatically controlled at the temperature selected by the temperature control knob. A signal goes to the controller which opens or closes the modulating valves to maintain the selected temperature. In the MAN position, the cockpit temperature controller is bypassed and control of the modulating temperature control valves is directly from the temperature control knob. In the OFF position, all power is removed from the system and the valves in the system, which control cabin temperature, will declutch and go to the full cool position.

Temperature Control Knob.

The temperature control knob (1, figure 1-40), located on the air conditioning control panel, is provided to select cockpit temperature. The knob can be rotated through a 300 degree arc and has mechanical stops at each end. The extreme counterclockwise end is marked COOL and the clockwise end is marked WARM. With

Air Conditioning and Pressurization System

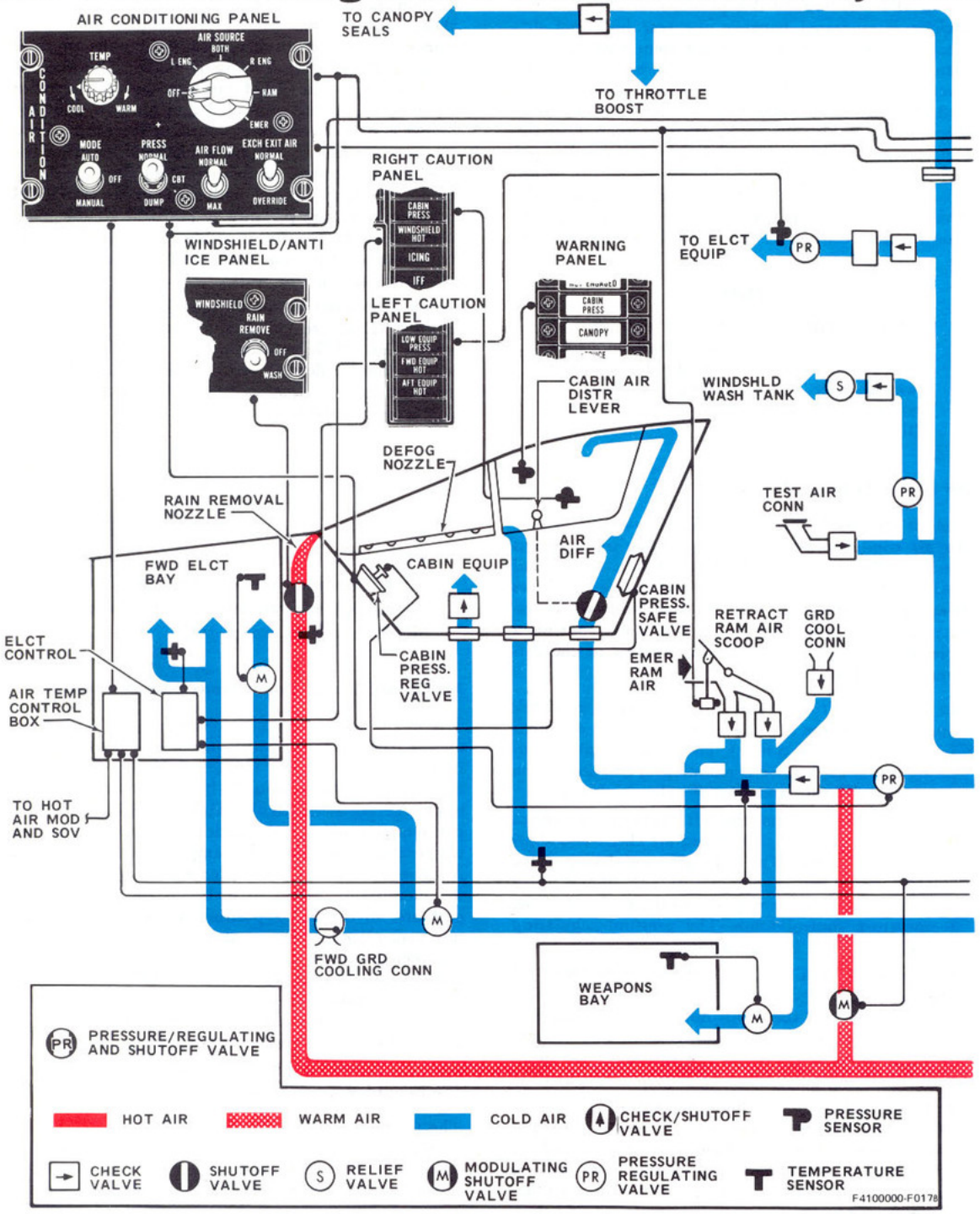


Figure 1-39. (Sheet 1)

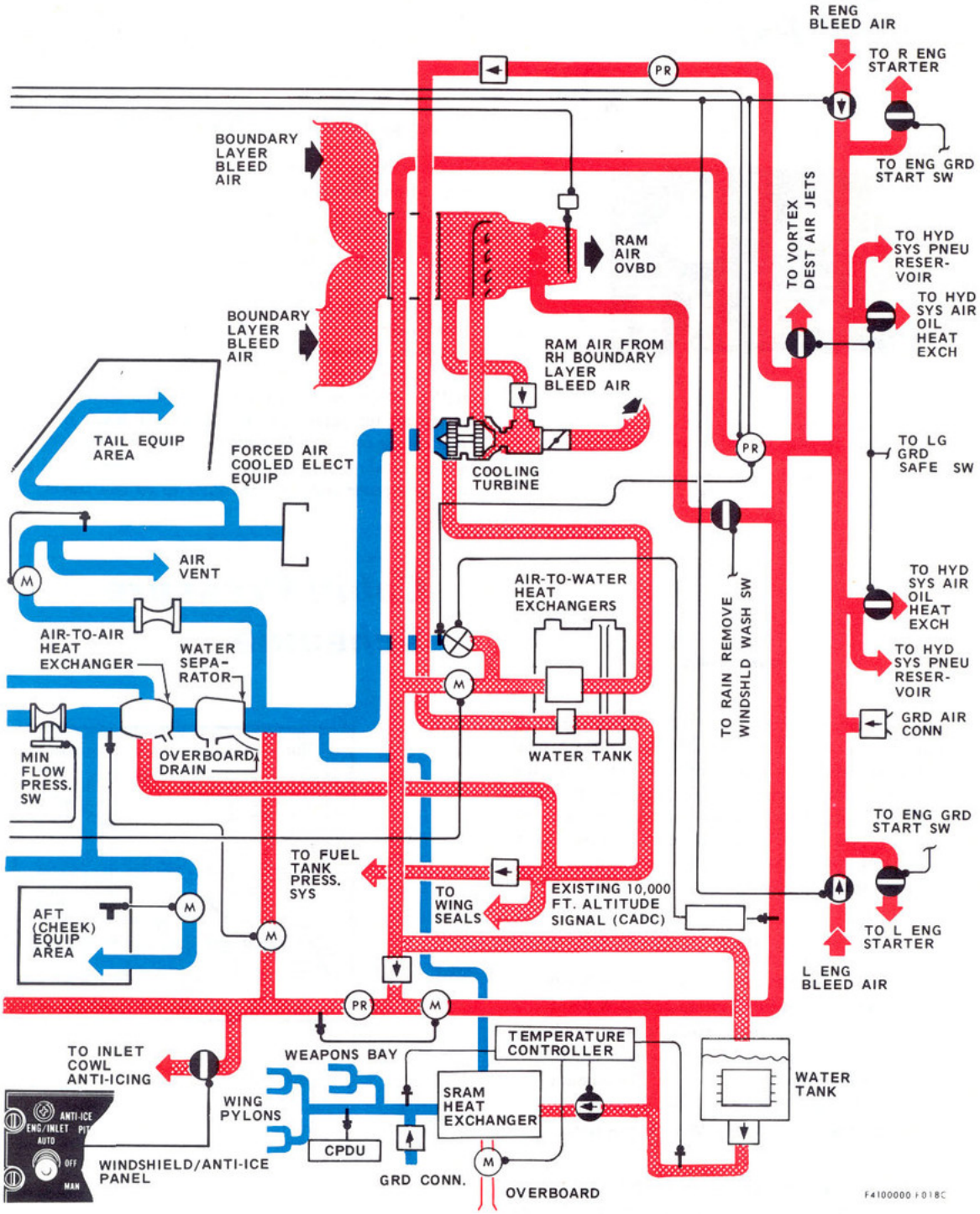
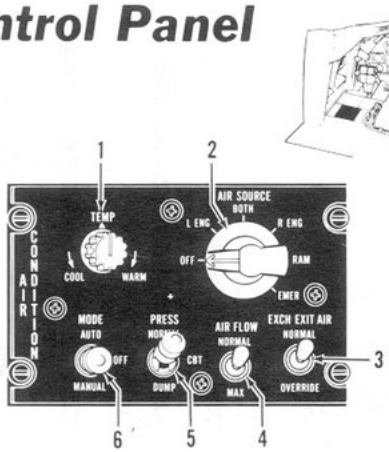


Figure 1-39. (Sheet 2)

Air Conditioning Control Panel



1. Temperature Control Knob.
2. Air Source Selector Knob.
3. Exchange Exit Air Control Knob.
4. Air Flow Selector Switch.
5. Pressurization Selector Switch.
6. Mode Selector Switch.

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Figure 1-40.

the mode selector switch in AUTO, rotating the knob in either direction sends a signal to the cockpit temperature controller which constantly positions the modulating temperature control valves to maintain the selected temperature. When the temperature control knob is positioned at the mid-point between COOL and WARM, the cockpit temperature is maintained at approximately 67 degrees F.

Note

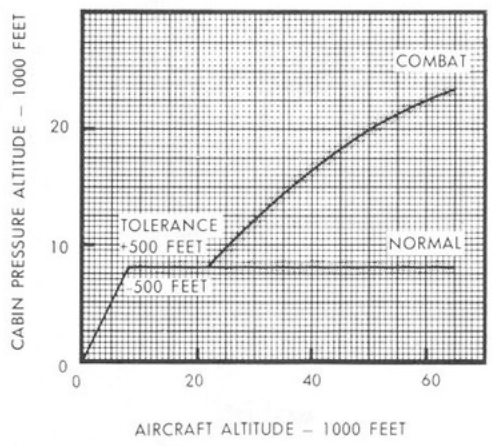
Operation with the temperature control knob at full COOL in warm weather or full WARM in cool weather with the mode selector knob in AUTO may result in an objectionable noise with the high flow in the cockpit. The amount of airflow can be reduced by backing the knob off the full COOL or WARM position.

With the mode selector switch in MANUAL, the signal goes directly to the modulating temperature control valves, opening or closing them as directed by the signal generated from the temperature control knob. During manual operation, the valves will respond only when the knob is held against a spring-loaded detent at either one of the extreme positions, COOL or WARM. Maximum valve travel time from maximum cold to maximum warm is approximately 45 seconds.

Exchange (Exch) Exit Air Control Switch.

The exchange exit air control switch (3, figure 1-40), located on the air conditioning control panel, is a two position switch marked NORMAL and OVERRIDE. The switch provides a means of controlling the amount of ram airflow through the air-to-air heat exchanger by opening or closing an exit door in the ram air discharge exit. In the NORMAL position, the central air data computer automatically controls the position of the door. When the outside air temperature is below 75 degrees F and airspeed is above 225 knots, the door will be closed to reduce drag. All other combinations of outside temperature and airspeed will result in automatic door opening. Placing the switch to OVERRIDE will override the automatic functions of the central air data computer and open the door to its full travel.

Cabin Pressure Schedule



F0000000-F028

Figure 1-41.

Note

- The air conditioning system water supply is required for supersonic flight conditions. When the water tank becomes empty at high speed flight, the system will cycle off. If this condition occurs, speed should be reduced to subsonic and the exchange exit air control switch placed to the OVERRIDE position to obtain maximum cooling. Normal system operation will be restored at subsonic flight conditions.
- When any external stores are installed, the exchange exit air control switch must be placed in the OVERRIDE position to prevent subsonic water boiling and possible shutdown of the air conditioning system.

Air Flow Selector Switch.

The air flow selector switch (4, figure 1-40) is a two position switch marked NORMAL and MAX. The switch provides a means of controlling the amount of airflow in the cabin. The NORMAL position provides cabin airflow for normal usage. The MAX position may be used for low level, high speed operation on a hot day. The MAX position may also be used during rain removal, windshield wash or for defogging.

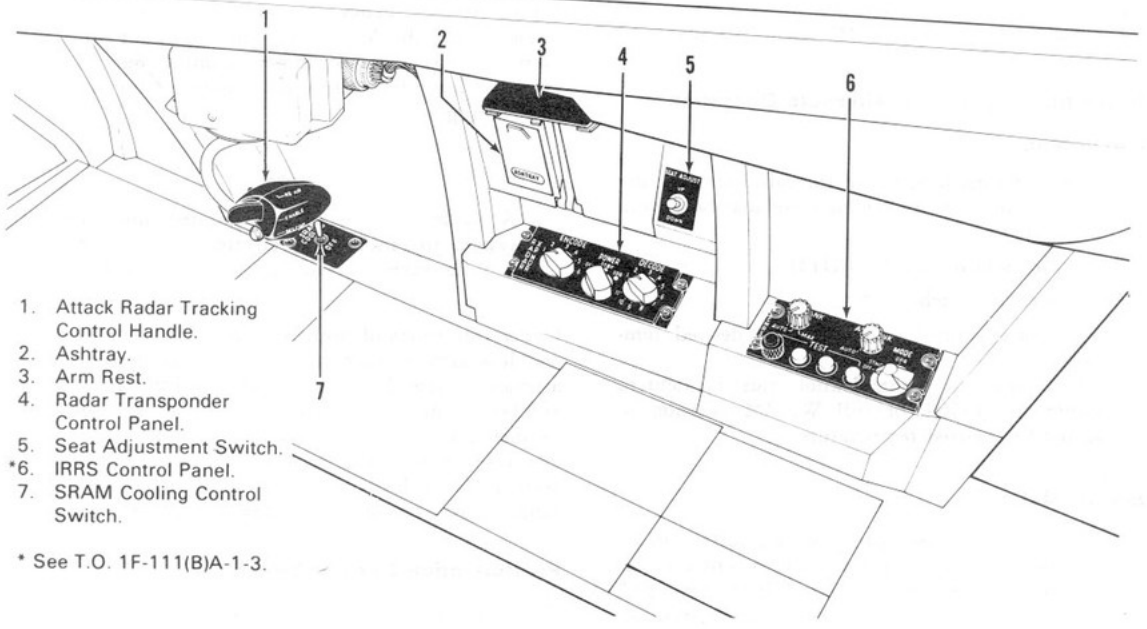
SRAM Cooling Control Switch.

The SRAM cooling control switch (7, figure 1-42), located on the right sidewall is a two position switch marked COOLING and OFF. The switch provides a means of controlling the cooling air and regulated pressure to the missile. The COOLING position provides automatically controlled cooling air and pressure regulated to 23 (±1) psi to the missile. The OFF position turns the SRAM cooling off.

CAUTION

To prevent unnecessary condensation within missile IMU or ingestion of sand and dust during low level operations, operate the SRAM cooling switch throughout the flight in accordance with the limitations established in T.O. 1F-111(B)A-30-1.

Right Sidewall (Typical)



1. Attack Radar Tracking Control Handle.
2. Ashtray.
3. Arm Rest.
4. Radar Transponder Control Panel.
5. Seat Adjustment Switch.
*6. IRRS Control Panel.
7. SRAM Cooling Control Switch.

* See T.O. 1F-111(B)A-1-3.

F0000000-F029D

Figure 1-42.

Bleed Air Duct Fail Caution Lamps. (After T.O. 1F-111-1151)

Two bleed air duct fail caution lamps (figure 1-36), located on the main caution lamp panel, are labeled L BLEED DUCT FAIL and R BLEED DUCT FAIL. If a bleed air duct failure (such as a rupture) occurs, the associated bleed air duct fail caution lamp will come on. Failure of a bleed air duct is critical because of the close proximity of a fuel cell and the high temperature of the exiting bleed air.

Forward Equipment Hot Caution Lamp.

The amber forward equipment caution lamp, marked FWD EQUIP HOT, is located on the main caution light panel (figure 1-36). The lamp will light if the cooling air flow is insufficient. The following equipment is listed in the order of heat generation. The list should be used as a guide for equipment shutdown, depending on flight requirements. Shutdown may be required to prevent degraded performance and/or equipment damage from overheating.

- ECM
- Attack Radar
- HF Radio (Transmit)
- Converter Set (in operation when either WDC or GNC in operation)
- IRRS
- RHAW
- UHF Radio (Transmit)
- INS
- Astro Compass
- TACAN
- One Computer (WDC or GNC)
- Doppler Radar
- Radar Altimeter
- HF Radio (Receive)
- IFF
- UHF Radio (Receive)

Air Conditioning System Alternate Operation.

Manual Mode.

In the event of a malfunction of the cabin temperature controller, cabin temperature may be manually controlled as follows:

1. Air source selector knob—BOTH.
2. Mode selector switch—MAN.
3. Temperature control knob—Set for desired temperature.
   The temperature control knob must be held to either full COOL or full WARM position to adjust for desired temperature.

Ram Air Mode.

In the event the air conditioning system fails, ram air mode can be used for cockpit and equipment cooling. Refer to "Ram or Emergency Mode Flight Envelope," Section V. During ram air mode operation, cockpit temperature can be controlled as follows:

1. Air source selector knob—RAM.
2. Mode selector switch—AUTO or MAN.
3. Temperature control knob—Set for desired temperature.
   If the mode selector switch is positioned to MAN, the temperature control knob must be held to either full COOL or WARM position to adjust for desired temperature.

Note

Cabin cooling, in the ram mode, is limited by the ambient ram air temperature; however, cabin heating in the ram mode can be maintained as the cabin hot air supply is still available to mix with the incoming ram airflow.

PRESSURIZATION SYSTEM.

Pressurization of the cockpit, canopy seals, throttle boost, attack radar, terrain following radar, track breaker and SRAM missile is provided by the pressurization system. Pressure in the cockpit is controlled by a pressure regulating valve located in the front of the cockpit. When the aircraft is below 8000 feet, the pressure regulating valve automatically maintains an unpressurized condition in the cockpit regardless of the schedule selected. When the aircraft is above 8000 feet, the pressure schedule is maintained by controlling the outflow of incoming ventilation air. A cabin pressure safety valve located at the rear of the cockpit will relieve pressure any time the cockpit pressure exceeds limits. An emergency ram air scoop, which can be opened into the airstream, will admit air into the crew and electronic equipment compartments in the event of loss of cooling and pressurization air from the cooling turbine.

Note

A priority valve is incorporated into the system to ensure that electronic equipment cooling takes precedence over the cockpit.

Under conditions of high cockpit pressure differential and low airflow, such as an idle power descent from altitude, a slow depletion of cabin altitude may occur and both temperature control and defog functions will be ineffective until engine power is increased. Failure of this valve may cause forward equipment hot caution lamp to light, loss of cabin cooling airflow, unstable cabin pressurization and temperature control.

Pressurization Selector Switch.

The pressurization selector switch (5, figure 1-40), located on the air conditioning control panel, is a three position lever-lock switch with positions NORMAL, CBT, and DUMP. In the NORMAL position, the cockpit pressure is selected to a schedule that will maintain

an 8000 foot cabin altitude from 8000 feet up to the operational ceiling of the aircraft. In the CBT (combat) position, the cockpit maintains an 8000 foot cabin altitude from 8000 feet up to 22,500 foot altitude and then maintains a constant 5 psi differential above ambient pressure. In the DUMP position, the cabin pressure regulator and the cabin pressure safety valve are open and the cockpit is not pressurized. See figure 1-41 for cockpit pressure schedule for normal and combat conditions.

Cabin Altitude Indicator.

A cabin altitude indicator (1, figure 1-15), located on the auxiliary gage panel, is provided to monitor cabin altitude.

Pressurization Caution Lamp.

An amber pressurization caution lamp, marked CABIN PRESS, is located on the main caution light panel (figure 1-36). The lamp will light when the cabin altitude is above 10,000 feet. When operating the cabin pressurization system in COMBAT, the cabin pressurization caution lamp will be lighted when aircraft altitude is above 26,000 feet.

Pressurization Warning Lamp.

A red pressurization warning lamp (8, figure 1-6), marked CABIN PRESS, is located on the left main instrument panel. The lamp will light when the cabin pressure is above 38,000 feet.

Equipment Low Pressure Caution Lamp.

An amber low equipment pressure caution lamp marked LOW EQUIP PRESS is located on the main caution light panel (figure 1-36). The lamp will light when the supply pressure to the pressurized electronic equipment drops below limits.

ANTI-ICING AND DEFOG SYSTEMS.

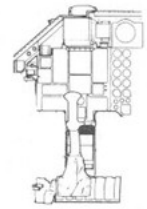
Probe Anti-Icing.

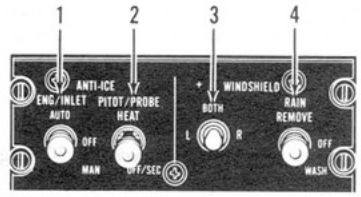
Heating elements powered by 115 volt ac are provided on the pitot-static, total temperature and angle-of-attack and sideslip (alpha/beta) probes for anti-icing. Power to the probe heater is controlled by a pitot/probe heater switch located on the windshield wash/anti-icing control panel. An alpha/beta probe caution lamp is provided to monitor the function of the heaters in the angle-of-attack and sideslip angle probes.

Pitot/Probe Heater Switch.

The pitot/probe heater switch (2, figure 1-43), located on the windshield wash/anti-icing control panel, has two positions marked HEAT and OFF/SEC (secondary). The switch performs the following functions:

Windshield Wash/ Anti-Icing Control Panel





1. Engine/Inlet Anti-Icing Switch.
2. Pitot/Probe Heater Switch.
3. Windshield Selector Switch.
4. Windshield Wash/Rain Removal Selector Switch.

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Figure 1-43.

On the ground.

- The OFF/SEC position turns off power to the probe heaters, and lights the alpha/beta probe heat caution lamp.
- The HEAT position furnishes power to primary alpha/beta heaters, and to the pitot-static probe.

Inflight.

- The OFF/SEC position provides power to the secondary alpha/beta heaters and the pitot-static heater. The total temperature heater is off.
- In the HEAT position, power is provided to the primary alpha/beta heaters, pitot-static heater and the total temperature heater.

On takeoff a safety switch on the landing gear actuates to arm the secondary heater circuits in the angle of attack and angle of sideslip probes. During flight if the primary heaters in either the angle of attack or angle of sideslip probes malfunctions, the alpha/beta probe heat caution lamp will light and the secondary heater in the failed probe will be automatically energized. Momentarily placing the switch to OFF/SEC should extinguish the caution lamp while the switch remains in that position, thereby verifying that the

secondary heaters are functioning properly. The total temperature probe heater does not receive power while the switch remains in the OFF/SEC position.

The HEAT position may be used to ground check the heater in the pitot-static probe and the primary heaters in the angle of attack and angle of sideslip probes. Proper operation of the primary heaters will be indicated by the alpha/beta probe heat caution lamp going out immediately after the switch is positioned to HEAT. The total temp probe heater may also be checked by placing the switch to HEAT and depressing the flight control master test switch and holding the CADC test switch to HIGH. Operation of the total temperature heater can be confirmed by observing an increasing temperature on the total temperature indicator.

Note

If the total temperature exceeds 50 degrees C during ground operation, the CADS caution lamp may light and remain lighted until total temperature drops below 50 degrees C.

The switch controls power to the heating elements in the pitot-static probe and total temperature probe, when the aircraft is airborne, and to the primary or secondary heating elements in the angle-of-attack (alpha) probe and side slip angle (beta) probes.

Alpha/Beta Probe Heat Caution Lamp.

The alpha/beta probe heat caution lamp, located on the main caution lamp panel (figure 1-36), provides indications that the angle-of-attack and/or sideslip angle probe heaters are not functioning properly as follows:

On the ground.

- Indicates the pitot/probe heater switch is in the OFF/SEC position.
- With the pitot/probe heater switch in the HEAT position, indicates the primary heater element(s) in either or both probes is malfunctioning or has overheated and has been deenergized by the thermostats.

Inflight.

- With the pitot/probe heater switch in the HEAT position, indicates the primary heater element(s) in either or both probes is not functioning.
- With the pitot/probe heater switch in the OFF/SEC position, indicates the secondary heater element in either or both probes is not functioning.

Note

The alpha/beta probe heat caution lamp is disabled at speeds above mach 1.10.

Engine Anti-Icing Systems.

The engine anti-icing system prevents formation of ice on the engine inlet guide vanes, and the engine nose cone. The engine anti-icing system uses regulated compressor bleed air. The engine inlet anti-icing system prevents formation of ice on the spike tip, the leading edge of the inlet cowl and inside the main engine air duct of the inlet cowl. The engine inlet anti-icing system uses air from the air conditioning system hot air manifold in the main landing gear wheel well.

Note

Idle rpm will not provide sufficient hot air for anti-icing.

The spike sensing probe anti-icing system prevents formation of ice on the spike local mach probe and spike lip shock probe. The probes are heated by 115 volt ac electrical heaters. Although the engine anti-icing, engine inlet anti-icing, and spike sensing probe anti-icing are three separate systems, they are controlled by a single, three position switch. Both automatic and manual modes of operation are provided. An electronic ice detector is located in the left engine air inlet. When icing conditions exist, a signal is transmitted to the icing caution lamp regardless of the position of the engine/inlet anti-icing switch.

Engine/Inlet Anti-Icing Switch. The engine/inlet anti-icing switch (1, figure 1-43), located on the windshield wash/anti-icing control panel, is a three position switch marked AUTO, MAN and OFF. The lever-lock type switch locks in all three positions. In the AUTO position, the anti-icing circuitry is armed, and when the electronic ice detector senses an icing condition, a signal is transmitted to the icing caution lamps. The signal also energizes a relay which turns on the elements in the spike sensing probe heaters and opens the engine anti-icing and engine inlet anti-icing control valves allowing the circulation of hot air through the anti-iced components. Approximately 60 seconds after the icing condition ceases, the hot air valves will close, the spike probe heating elements will be deenergized and the engine icing caution lamp will go out. When the switch is placed to MAN, the engine anti-icing and engine inlet anti-icing valves open and the spike probe heating elements are energized whether or not the ice detector senses an icing condition. Placing the switch to OFF shuts off air to the engine anti-icing and engine inlet anti-icing systems, and turns off the spike probe heating elements; however, the icing caution lamp will still be operational.

Engine Icing Caution Lamp. The engine icing caution lamp, located on the main caution lamp panel (figure 1-36), will light when the electronic ice detector senses an icing condition. While the icing condition exists, the caution lamp will remain lighted regardless of the

position of the engine/inlet anti-icing switch. The lamp will go out 60 seconds after the icing condition ceases. A malfunction within the electronic ice detector may cause the icing caution lamp to light, thus activating the engine/inlet anti-icing system if the Engine/Inlet Anti-icing switch is in AUTO. If a malfunction of the detector is known or suspected, operate the system manually by placing the engine/inlet anti-icing switch to OFF if engine/inlet anti-icing is not required, or to MAN when operation is required, under the following conditions.

Ground operation:

1. In visible moisture—Temperature between 35°F and −10°F.
2. In clean air—Relative humidity above 70 percent and dew point temperature 25°F to 35°F.

In-Flight Operation:

1. In visible moisture—Total temperature of +5°C or below.
2. In clear air—No limitations related to icing.

Inlet Hot Caution Lamp. The inlet hot caution lamp, located on the main caution lamp panel (figure 1-36), provides an indication that the temperature of anti-icing bleed air has exceeded limits. When the lamp lights, the words INLET HOT are visible and anti-icing air to the auxiliary cowls is automatically shut off, then the lamp will go out.

Windshield Defog System.

Air for windshield defogging and cabin air distribution share the same control lever. For description, refer to "Cabin Air Distribution Control Lever," this section.

WINDSHIELD WASH AND RAIN REMOVAL SYSTEM.

The windshield wash and rain removal system is provided to keep both the windshields clear of impinging rain and insects. Compressor bleed air is directed over the outside of the windshields by a fixed area nozzle. This hot air blast will evaporate impinging rain and prevent further accumulation of rain on the windshield. Windshield wash is accomplished by injecting a liquid wash solution into the rain removal nozzle. The windshield wash solution is contained in a one gallon tank located on the right side of the nose wheel well.

Windshield Wash/Rain Removal Selector Switch.

The windshield wash/rain removal selector switch (4, figure 1-43), located on the windshield wash/anti-icing control panel, has three positions marked RAIN REMOVE, WASH, and OFF. The switch is spring-loaded from the WASH to the OFF position and is locked out of the RAIN REMOVE position. The switch must be pulled out to move from OFF to RAIN REMOVE. Placing the switch to RAIN REMOVE will open the rain remove shutoff valves, allowing temperature and pressure regulated compressor bleed air to be directed to the windshield(s) selected. When the switch is placed to WASH, a time delay relay is energized to open the rain remove shutoff valve and the windshield wash shutoff valve selected by the windshield wash selector switch. While these valves are open, compressor bleed air and liquid windshield wash solution will be directed to the selected windshield(s). Allowing the switch to return from WASH to OFF will close the valves after a 5-second delay, shutting off the air and windshield wash solution.

Windshield Selector Switch.

The windshield selector switch (3, figure 1-43), located on the windshield wash/anti-icing control panel, has three positions marked L (left), R (right), and BOTH. Selection of any of the positions will determine the windshield(s) to be washed or receive rain removal air. For optimum performance of rain removal system, operate one side only. Selection of BOTH position will decrease airflow on each windshield.

Windshield Hot Caution Lamp.

The windshield hot caution lamp, located on the main caution lamp panel (figure 1-36), indicates windshield high temperature. An overheat switch, installed in the rain removal air supply duct upstream of the shutoff valve, will close when the air temperature exceeds limits. When the overheat switch closes, a circuit is completed to close the rain remove shutoff valves and light the windshield hot caution lamp. After the switch closes, the caution lamp will normally go out within 15 seconds.

WHEEL WELL OVERHEAT DETECTION SYSTEM.

A wheel well overheat detection system provides a visual indication of an overheat condition in the main wheel well area. The function of the system is similar to the engine fire detection systems. Sensing elements, located in the main wheel well and plumbing crossover areas aft of the main landing gear bulkhead, detect a rise in temperature and light the wheel well hot caution lamp when a predetermined temperature is reached. When an overheat condition is indicated, the air source selector knob, located on the air conditioning control panel, should be positioned to OFF or EMER. This will close the engine bleed air check and shut off valves to shut off the source of the hot bleed air and open the ram air door.

Note

- The engine bleed air check and shutoff valves are electrically controlled and pneumatically operated. Each valve is controlled by 28-volt dc power from the essential bus and is protected by a circuit breaker located in the right forward equipment bay. Loss of electrical power to a valve with 16th stage bleed air present will cause the valve to open.
- After T.O. 1F-111-946, sensing elements are also located in the weapons bay routing tunnel and near the ac power panel. These elements are incorporated into the wheel well overheat detection circuit.

WHEEL WELL HOT CAUTION LAMP.

A wheel well hot caution lamp (figure 1-36), located on the main caution lamp panel, provides an indication of an overheat condition in the main wheel wells and plumbing crossover areas aft of the landing gear bulkheads. The words WHEEL WELL HOT are visible when the lamp is lighted. After T.O. 1F-111-946, this lamp also provides indications of overheat in the weapons bay routing tunnel and ac power panel area.

AGENT DISCHARGE/FIRE DETECT TEST SWITCH.

The agent discharge/fire detect test switch (2, figure 1-6), located on the left main instrument panel, provides a means of checking the wheel well overheat detection system. The AGENT DISCH position serves no function with the wheel well detection system. Positioning the switch to FIRE DETECT TEST position will light the wheel well hot caution lamp, if the wheel well detection system is operational.

FUSELAGE OVERHEAT TEST SWITCH.

The fuselage overheat test switch (6, figure 1-27), located on the ground check panel, is marked LOOP 1, NORM, and LOOP 2, and is spring-loaded to NORM. This switch is used to individually check the two loops in the wheel well overheat detection system for shorts to ground. Positiining the switch to LOOP 1 puts an artificial signal (short to ground) on loop 2 and will cause the wheel well hot caution lamp to light if Loop 1 is shorted to ground. Loop 2 is checked in the same manner by positioning the switch to LOOP 2. Since the system must have a signal on both loops before it will cause the wheel well hot caution lamp to light, a short on a single loop will not prevent detection of an overheat condition.

OXYGEN SYSTEM.

The oxygen system consists of a normal (liquid) system located in the forward fuselage and cockpit and an emergency (gaseous) system located behind the cockpit aft bulkhead.

NORMAL OXYGEN SYSTEM.

The normal oxygen system consists of a 15 liter liquid oxygen converter, which converts the liquid oxygen to a gas; a heat exchanger, which heats the gas to a temperature suitable for breathing, an on-off control valve at each station and a diluter demand type regulator attached to each crew member's torso harness. A three position control knob on the regulator provides for selection of normal diluted oxygen, 100 percent oxygen and, in the event of an emergency, 100 percent oxygen under pressure.

CAUTION

Use care when connecting the oxygen regulator to the restraint harness and oxygen supply hose or when connecting the mask hose to the regulator. The valve port screens on the regulator may be easily damaged by careless or improper handling.

A quick disconnect is provided between the regulator and oxygen mask hose to expedite abandoning the aircraft on the ground. After each flight, the regulator is to be stowed on a wedge-shaped bracket (7, figure 1-46), located on the right side of each seat headrest. For the duration of the oxygen supply refer to figure 1-44. Refer to figure 1-83 for oxygen system servicing.

Oxygen Control Knob.

The oxygen control knob (1, figure 1-45), located on the oxygen regulator on the right side of the upper torso harness, has three positions marked NORM, 100 percent, and EMER (emergency). When the knob is positioned to NORM, oxygen proportionally diluted with air, is supplied to the mask at cabin altitudes from 0 to 17,500 feet. At cabin altitudes above 17,500 feet, 100 percent oxygen is supplied and pressure breathing will commence at approximately 35,000 feet. With the knob in the 100 percent position, 100 percent oxygen is supplied to the mask. Selecting the EMER position provides 100 percent oxygen under pressure to the mask.

Oxygen Control Levers.

Two oxygen control levers (1, figure 1-46) are provided to control flow of oxygen from the supply system to the oxygen regulator. When in the ON posi-

Oxygen Duration

With regulator at 100 percent or EMER position.

CABIN ALTITUDE	CONSUMPTION 2 MEN cu. ft./hr.	DURATION — HOURS*														
35,000	9.8	46.8	43.7	40.6	37.4	34.3	31.2	28.1	25.0	21.8	18.7	15.6	12.5	9.4	6.2	3.1
30,000	13.4	34.3	32.0	29.7	27.4	25.1	22.8	20.5	18.3	16.0	13.7	11.4	9.1	6.8	4.6	2.3
28,000	15.0	30.6	28.5	26.5	24.4	22.4	20.4	18.4	16.3	14.3	12.2	10.2	8.2	6.1	4.1	2.0
26,000	16.0	28.6	26.7	24.8	22.9	21.0	19.1	17.2	15.3	13.4	11.5	9.6	7.6	5.7	3.8	1.9
24,000	18.52	24.7	23.1	21.4	19.8	18.1	16.5	14.9	13.2	11.6	9.9	8.3	6.6	5.0	3.3	1.6
22,000	20.76	22.1	20.7	19.2	17.7	16.2	14.7	13.3	11.8	10.3	8.8	7.4	5.9	4.4	2.9	1.5
20,000	23.0	20.0	18.6	17.2	15.9	14.6	13.3	12.0	10.6	9.3	8.0	6.6	5.3	4.0	2.7	1.3
18,000	25.24	18.1	16.9	15.7	14.5	13.3	12.1	10.9	9.7	8.5	7.3	6.1	4.8	3.6	2.4	1.2
16,000	27.48	16.8	15.6	14.5	13.4	12.3	11.1	10.0	8.9	7.8	6.7	5.6	4.4	3.3	2.2	1.1
14,000	30.0	15.6	14.5	13.5	12.4	11.4	10.2	9.2	8.2	7.1	6.1	5.1	4.1	3.1	2.0	1.0
12,000	32.8	13.9	13.0	12.1	11.1	10.2	9.3	8.4	7.5	6.5	5.6	4.7	3.7	2.8	1.9	0.9
10,000	35.6	12.9	12.0	11.1	10.3	9.5	8.6	7.7	6.9	6.0	5.1	4.3	3.4	2.6	1.7	0.8
8,000	39.4	11.6	10.8	10.1	9.3	8.5	7.8	7.0	6.2	5.4	4.7	3.9	3.1	2.3	1.5	0.7
0	55.6	8.2	7.7	7.1	6.6	6.0	5.5	4.9	4.4	3.8	3.3	2.7	2.2	1.6	1.1	0.5
AVAILABLE OXYGEN	LITERS (LIQUID)	15	14	13	12	11	10	9	8	7	6	5	4	3	2	1
	Cu. Ft. GAS	459.0	428.4	397.8	367.2	336.6	306.0	275.4	244.8	214.2	183.6	153.0	122.4	91.8	61.2	30.6

With regulator in NORM position.

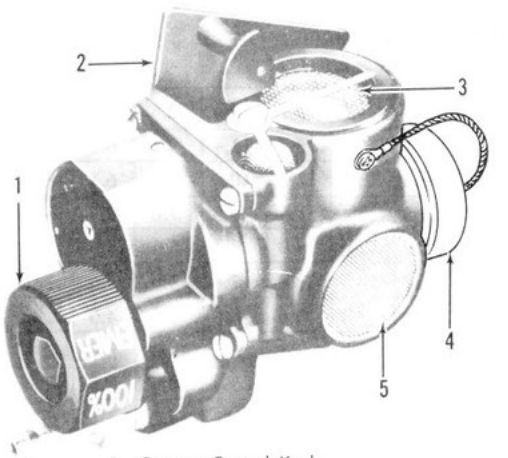
CABIN ALTITUDE	CONSUMPTION 2 MEN cu. ft./hr.	DURATION — HOURS*														
35,000	9.8	46.7	43.6	40.5	37.4	34.3	31.2	28.1	25.0	21.8	18.7	15.6	12.5	9.4	6.2	3.1
30,000	13.2	34.7	32.4	30.1	27.8	25.5	23.2	20.9	18.6	16.2	13.9	11.6	9.3	7.0	4.6	2.3
25,000	14.0	32.8	30.6	28.4	26.2	24.0	21.9	19.7	17.5	15.3	13.1	10.9	8.7	6.6	4.4	2.2
20,000	12.4	37.0	34.6	32.1	29.6	27.2	24.7	22.2	19.8	17.3	14.8	12.3	9.9	7.4	4.9	2.5
15,000	10.2	45.0	42.0	39.0	36.0	33.0	30.0	27.0	24.0	21.0	18.0	15.0	12.0	9.0	6.0	3.0
10,000	10.2	45.0	42.0	39.0	36.0	33.0	30.0	27.0	24.0	21.0	18.0	15.0	12.0	9.0	6.0	3.0
8,000	10.2	45.0	42.0	39.0	36.0	33.0	30.0	27.0	24.0	21.0	18.0	15.0	12.0	9.0	6.0	3.0
5,000	10.2	45.0	42.0	39.0	36.0	33.0	30.0	27.0	24.0	21.0	18.0	15.0	12.0	9.0	6.0	3.0
Sea Level	10.2	45.0	42.0	39.0	36.0	33.0	30.0	27.0	24.0	21.0	18.0	15.0	12.0	9.0	6.0	3.0
AVAILABLE OXYGEN	LITERS (LIQUID)	15	14	13	12	11	10	9	8	7	6	5	4	3	2	1
	Cu. Ft. GAS	459.0	428.4	397.8	367.2	336.6	306.0	275.4	244.8	214.2	183.6	153.0	122.4	91.8	61.2	30.6

*2 crew members (double duration for 1 crew member).

When available oxygen is less than 1 liter descend to below 10,000 feet MSL.

Figure 1-44.

Diluter Demand Oxygen Regulator



1. Oxygen Control Knob.
2. Wedge Mounting Plate.
3. Diluter Valve.
4. Oxygen Mask Outlet (Dust Cover Installed).
5. Anti Suffocation Valve.
6. Inlet Hose Fitting.

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Figure 1-45.

tion, oxygen is supplied from the converter to the regulator; when in OFF, oxygen flow is shut off at the control valve in the oxygen-suit control panel.

CAUTION

To prevent damage to the regulator do not actuate the lever ON or OFF with the dust covers installed.

Oxygen Quantity Indicator.

An oxygen quantity indicator is located on the right main instrument panel (16, figure 1-30). The indicator indicates the total quantity of liquid oxygen in the converter. The indicator dial is graduated from zero to 20 liters in increments of one liter. In the event of a power failure, the indicator pointer will freeze.

Oil/Oxygen Quantity Indicator Test Switch.

The oil/oxygen quantity indicator test switch (6, figure 1-15), located on the auxiliary gage panel, is provided to check oxygen quantity indicator. The switch is marked OXY, QTY, OIL QTY and is spring-loaded to the center unmarked OFF position. When the switch is held depressed to the OXY QTY position, on aircraft 7 ▸ 52 the indicator pointer will move to the zero-liter indication if the indicating system is operating properly. On aircraft 53 ▸ the indicator pointer will drive in a counterclockwise direction to the zero-liter indication and beyond, and will continue to drive ccw until the button is released. When the switch is released, the pointer will move to the original reading. The oxygen caution lamp will light during an indicator check when the pointer indicates a quantity of 2 liters or less. Refer to "Oil System", this section, for the oil quantity indicator test functions of the switch.

Oxygen Caution Lamp.

An amber caution lamp on the main caution lamp panel (figure 1-36) will light when oxygen quantity indicator indicates 2 liters or less or when oxygen system pressure is less than 42 (±2) psi. When the caution lamp lights, inspection of the oxygen quantity indicator will determine whether the lamp came on because of low quantity or low pressure. When the lamp is lighted, the letters OXY will be visible on the caution lamp panel, and the master caution lamp will light.

EMERGENCY OXYGEN SYSTEM.

The crew module is equipped with an emergency oxygen system consisting of two oxygen bottles, a presure reducer, a pressure gage, and a manual handle. The system is activated automatically during ejection or in event of failure of the automatic feature, it is manually activated by a handle. Also, during other phases of flight, this system provides an emergency oxygen supply in event of failure of the normal oxygen system. When activated either manually or automatically, high pressure gaseous oxygen flows to a pressure reducer where it is reduced to 50 to 90 psi. It is then routed into the normal oxygen system upstream of the oxygen control valves. Sufficient emergency oxygen is available for 10 minutes duration at 27,000 feet.

Note

EMER or 100% must be selected on the oxygen control knob (1, figure 1-45) to receive pure oxygen to the oxygen mask. Selecting NORM on the oxygen control knob supplies a proportionally diluted air/oxygen mixture.

Emergency Oxygen Handle.

The green emergency oxygen handle (6, figure 1-46) is located on the right crew member's oxygen panel. During ejection, this handle is used to manually

Oxygen Control Panel

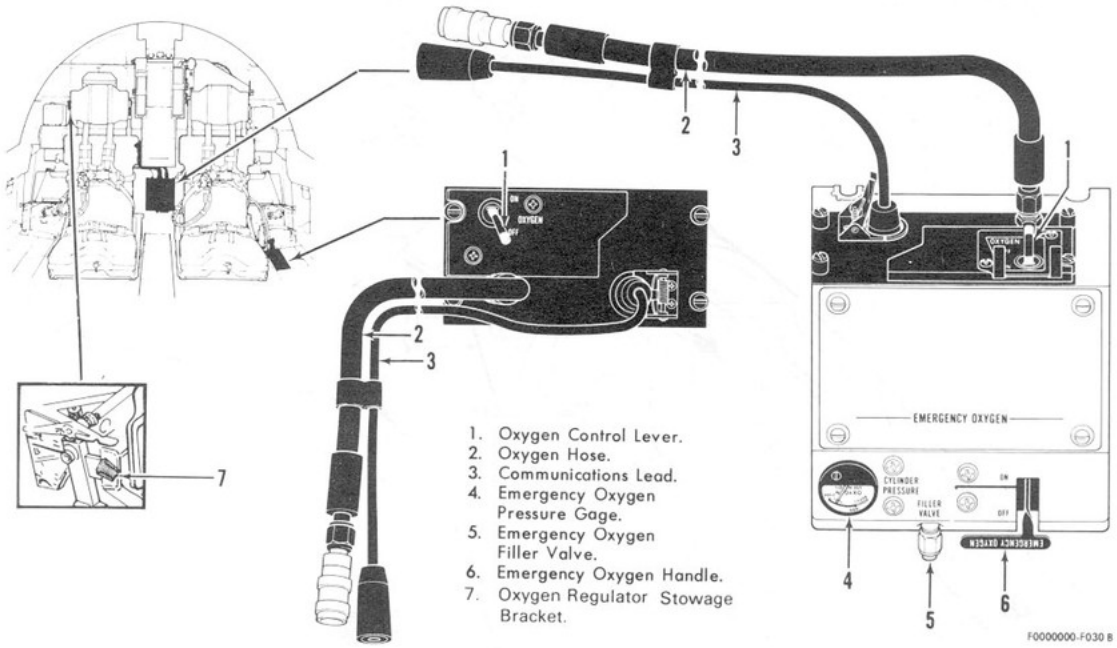


Figure 1-46.

activate the emergency oxygen system in the event automatic activation fails. Also, in event of failure of the normal oxygen system during other phases of flight, this handle is used to provide an emergency oxygen supply. Raising the handle will open the emergency oxygen pressure reducer allowing oxygen to flow to each oxygen control valve.

Emergency Oxygen Pressure Gage.

The emergency oxygen pressure gage (4, figure 1-46), located on the right crew member's oxygen control panel, indicates the pressure in the emergency oxygen bottles. The gage is marked REFILL in the red region and FULL in the black region with index marks at 1400 and 2500 psi.

OXYGEN SYSTEM ALTERNATE OPERATION.

If the normal oxygen system fails or is depleted, pull the emergency oxygen handle.

CREW MODULE ESCAPE SYSTEM.

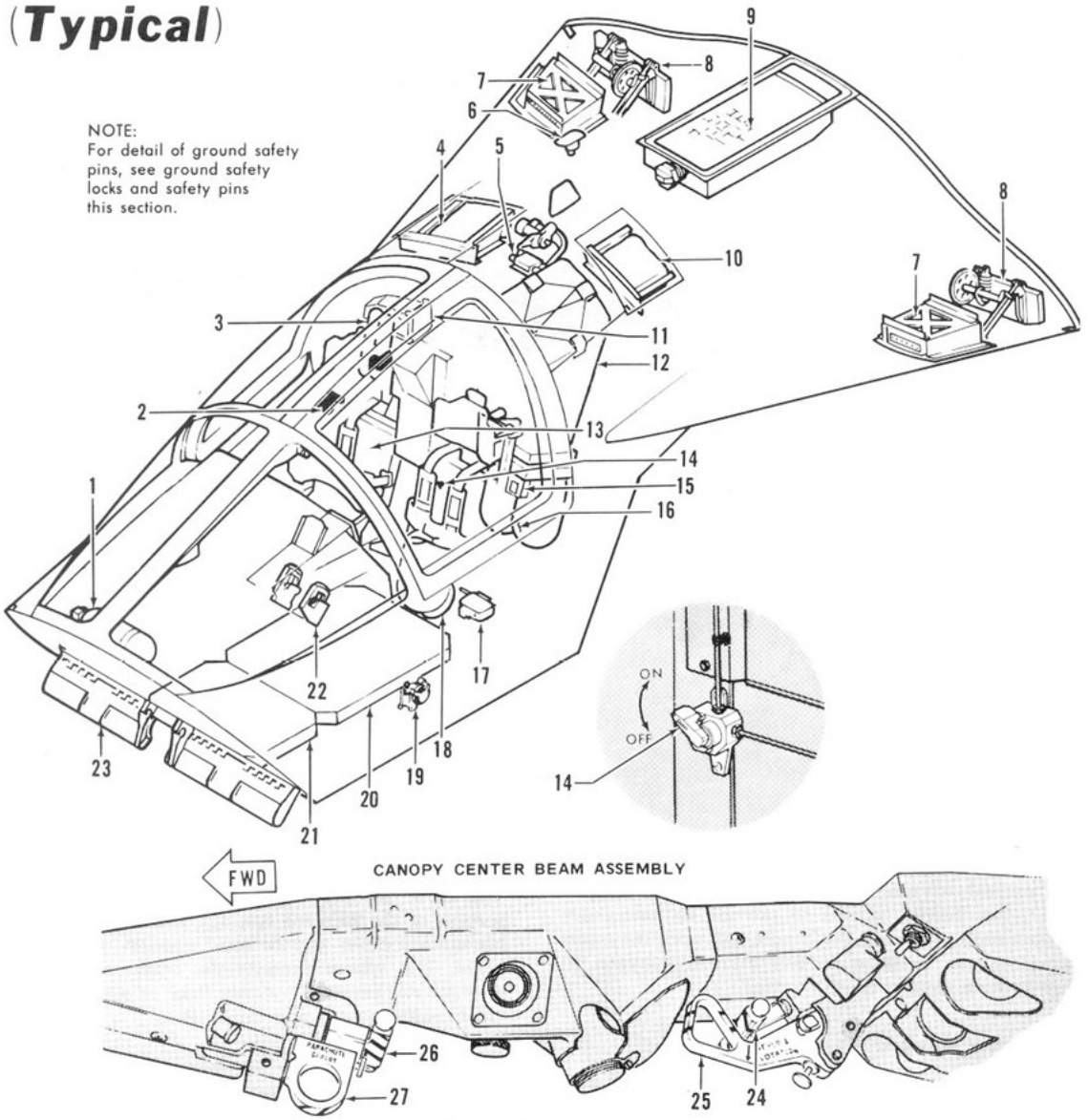
The crew module (figure 1-47) forms an integral portion of the forward fuselage and encompasses the pressurized cabin and forward portion of the wing glove. The system protects the occupants from environmental hazards on either land or water and provides underwater escape capabilities.

WARNING

The removal or addition of components in the crew module will change the center of gravity of the module and adversely affect its stability on ejection.

For additional information, refer to "Oxygen System", this section.

Crew Module General Arrangement (Typical)



1. Auxiliary Flotation Bag Pressure Bottle.
2. Canopy Center Beam Assembly.
3. Emergency Oxygen Bottles (2).
4. Right Self-Righting Bag.
5. Barostat Lock Initiator.
6. Emergency UHF Antenna.
7. Aft Flotation Bag.
8. Pitch Flap (2).
9. Stabilization-Brake Parachute.
10. Left Self-Righting Bag.
11. Quick Rescue Kit.
12. Recovery Parachute.
13. Survival Gear.
14. Chaff Dispenser Control Lever.
15. Radio Beacon Set.
16. Impact Attenuation Bag Pressure Bottles (2).
17. Chaff Dispenser.
18. Rocket Motor.
19. Bilge/Flotation Bag Inflation Pump.
20. Impact Attenuation Bag.
21. Auxiliary Flotation Bag.
22. Ejection Handles (2).
23. Chin Flaps (2).
24. Recovery Parachute Release Handle.
25. Severance and Flotation Handle.
26. Auxiliary Flotation Handle.
27. Recovery Parachute Deploy Handle.

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Figure 1-47.

CREW MODULE SEATS.

The crew module seats (figure 1-48) are electrically adjustable vertically and manually adjustable forward and aft. The seat headrest structure, which is attached to the aft bulkhead, and the seat pan are manually adjustable forward and aft. The forward adjustment of the headrest requires the inertia reel to be unlocked. Each seat is equipped with a restraint harness to protect the crew member during flight in turbulence and during ejection. The harness consists of shoulder straps and lap straps and a single point harness release attached to the front of the seat by an anchor strap. The shoulder straps are attached to the inertia reel and to the seat structure to prevent them from sliding off the crew member's shoulders. The lap straps are attached to each side of the aft seat pan. The ends of the shoulder and lap straps snap into the single point release. The single point release must be rotated 90 degrees, in either direction to release the straps. A detent at the 90 degree point will hold the release in that position. The release is spring-loaded to the center (locked) position when it is not in the detent. A wedge shaped plastic block is mounted on the right side of both types of harness to attach the oxygen regulator. Each seat is equipped with an inertia reel located behind the headrest. When unlocked, the inertia reel will allow the shoulder straps to extend or retract automatically to allow freedom of movement of the crew member. When an excessive g-force is encountered or when manually locked, the inertia reel will prevent further shoulder strap extension and will take up shoulder strap slack as the crew member returns to a normal position. The inertia reel is also equipped with an explosive cartridge in a power retraction device which, during ejection, will retract the shoulder straps and lock the reel. The right seat must be repositioned to gain access to the survival equipment compartment. Instructions for repositioning the seat are located on the back of the seat under the back cushion.

EJECTION EQUIPMENT.

The ejection equipment consists of the necessary initiators, severance components and the rocket motor. Actuation of either ejection handle initiator provides an explosive impulse sequenced to lock the shoulder harness inertia reels in the retract position, activate the emergency oxygen system, release the chaff dispenser, activate guillotine cutters, ignite the rocket motor, activate the severance components and to deploy the stabilization-brake and recovery parachutes and impact attenuation bag. The severance components consist of the flexible linear shaped charges (FLSC) and explosive guillotine cutters. The FLSC is located around the crew module so that detonation will cut the splice plate joining the crew module to the aircraft. FLSC is also used to remove the covers over the parachutes and flotation, self righting, and impact attenuation bags. The explosive guillotine cutters are provided to sever antenna leads, secondary control cables, and an oxygen line. Quick disconnects located in the crew module floor are used for separation of the normal air conditioning and pressurization system ducts, the flight controls, and the electrical wiring. The rocket motor, located between the crew members and behind the seat bulkhead, provides the thrust to propel the crew module up and away from the aircraft.

Rocket Motor.

The rocket motor has both a lower and an upper nozzle. The lower nozzle provides a normal thrust of 27,000 pounds below 300 (±30) knots. To avoid excessive g-forces to the crew members, the rocket motor is provided with two concentric upper nozzles. The small auxiliary nozzle in the center of the upper nozzle fires simultaneously with the lower nozzle to provide about 500 pounds of thrust to counteract slow speed crew module pitchup tendencies. Maximum upper nozzle thrust is achieved at aircraft speeds above 300 (±30) knots by severance of the rocket upper nozzle diaphragm. The increase in exhaust area results in reduced lower nozzle thrust (about 9000 pounds at the lower nozzle and 7000 pounds at the upper nozzle) and extended operating time. The rocket motor is located between the seat bulkhead and aft pressure bulkhead of the crew module. After ground cold soak at minus 50°F or below, it is necessary to direct warm air into the rocket motor compartment, between 80° to 120°F for 30 minutes, prior to takeoff.

Dual-Mode "Q"-Actuated Selector.

The dual-mode "q"-actuated selector continuously senses aircraft speed and selects the appropriate time delay. The symbol "q" denotes dynamic pressure. The "q"-actuated selector allows activation of a 1-second time delay initiator and blocks propagation to the rocket motor upper nozzle when aircraft speed is less than 300 (±30) knots. When aircraft speed is greater than 300 (±30) knots, the "q"-actuated selector blocks propagation to the 1-second time delay initiator and allows activation of SMDC to the rocket motor upper nozzle.

"G"-Sensor Initiator.

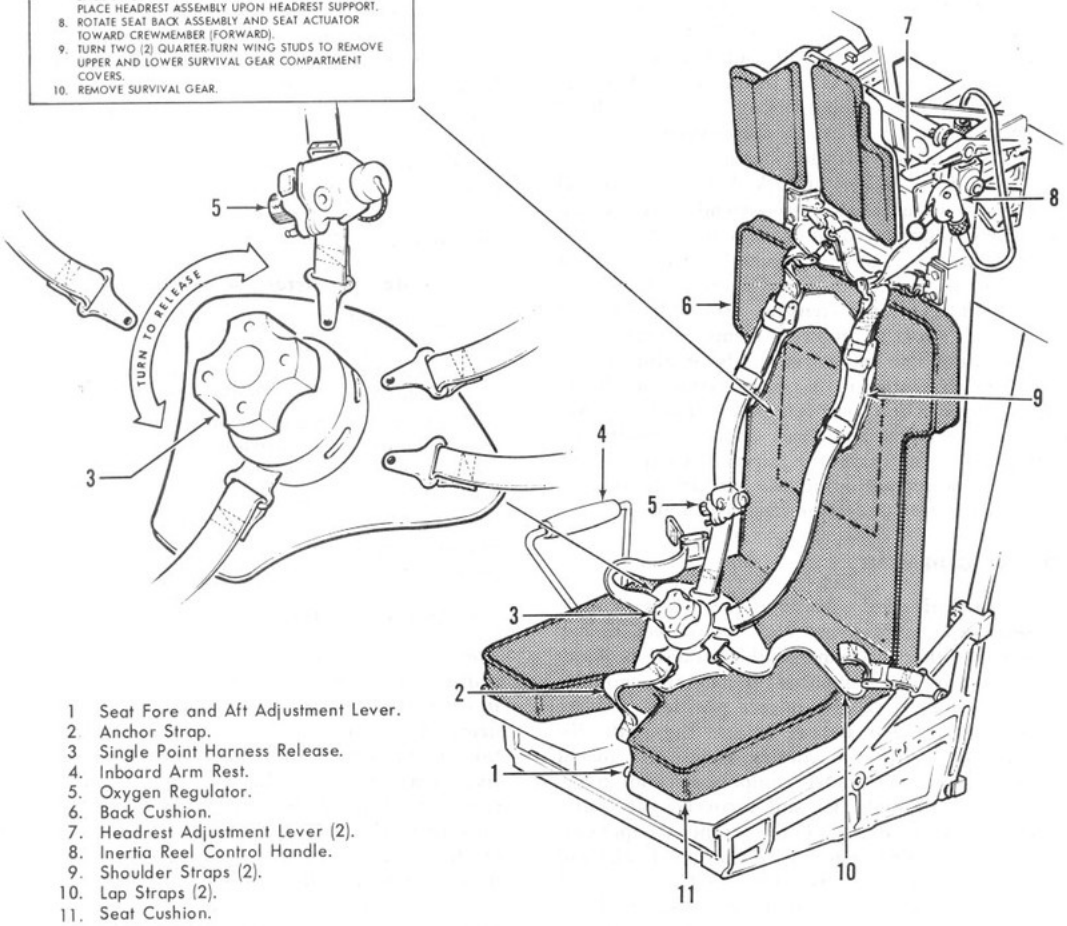
The "g"-sensor initiator consists of two operating trains. A rotating weighted arm, designated the rotor, in each explosive train is normally locked to prevent firing of the train. Firing of SMDC into the "g"-sensor inlet ports retracts the lock piston. Forward decelerative forces initially hold the rotors against stops. As the forces drop off, the spring-loaded rotors swing aft until they release dual firing pins. This initiates SMDC at the outlet ports and continues the detonation sequence to deploy the recovery parachute. The "g"-sensor initiator is located above the survival equipment compartment. A time delay fires the "g"-sensor initiator 1.6 seconds after rocket motor ignition. After the forward component of acceleration decreases to 2.2 "g's", the "g"-sensor initiator fires and activates the barostat lock initiator.

Crew Module Seats (Typical)

PEEL CLOTH FROM TAPE TO REMOVE INSTRUCTIONS

SURVIVAL GEAR ACCESS

1. DEPRESS EITHER HEADREST ADJUSTMENT LEVER FOR RH SEAT; SLIDE HEADREST FULLY FORWARD.
2. PULL UP ON RH SEAT PAN FORE AND AFT ADJUSTMENT HANDLE; SLIDE SEAT FULLY FORWARD.
3. LH CREWMEMBER REACHES BEHIND RH SEAT AND DISCONNECTS SEAT ADJUSTMENT ACTUATOR ELECTRICAL CONNECTOR.
4. LH CREWMEMBER REACHES BEHIND RH SEAT TO PULL BALL LOCK ATTACH PIN FROM SEAT ACTUATOR WHILE RH CREWMEMBER RELIEVES HIS WEIGHT ON SEAT TO AID PIN REMOVAL. PULL PIN OUT COMPLETELY. CAUTION: SEAT WILL SUDDENLY DROP DOWN WHEN PIN IS REMOVED. MAKE SURE THAT CREWMEMBER'S FEET ARE NOT UNDER SEAT PAN AND HANDS ARE ABOVE AND CLEAR OF SEAT SIDES.
5. WITH CREWMEMBER IN RH STATION FACING AFT, POSITION BOTH HEADREST STOPS LOCATED ON HEADREST TRACKS IN THE UP POSITION.
6. DEPRESS ONE OR BOTH HEADREST ADJUSTMENT LEVERS AND PULL HEADREST TOWARD CREWMEMBER (FORWARD) UNTIL HEADREST DISENGAGES FROM ITS SUPPORT TRACKS.
7. RAISE ENTIRE HEADREST ASSEMBLY UPWARD UNTIL SEAT BACK UPPER SUPPORTS CLEAR THE SEAT BACK TRACKS. PLACE HEADREST ASSEMBLY UPON HEADREST SUPPORT.
8. ROTATE SEAT BACK ASSEMBLY AND SEAT ACTUATOR TOWARD CREWMEMBER (FORWARD).
9. TURN TWO (2) QUARTER-TURN WING STUDS TO REMOVE UPPER AND LOWER SURVIVAL GEAR COMPARTMENT COVERS.
10. REMOVE SURVIVAL GEAR.



1. Seat Fore and Aft Adjustment Lever.
2. Anchor Strap.
3. Single Point Harness Release.
4. Inboard Arm Rest.
5. Oxygen Regulator.
6. Back Cushion.
7. Headrest Adjustment Lever (2).
8. Inertia Reel Control Handle.
9. Shoulder Straps (2).
10. Lap Straps (2).
11. Seat Cushion.

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Figure 1-48.

Barostat Lock Initiator.

The barostat lock initiator consists of two operating trains. An aneroid bellows of each explosive train in this initiator is normally locked to prevent firing of the train, constant cycling, and wearout. Firing of SMDC into the barostat inlet ports initiates an explosive charge that retracts the pins which normally lock the bellows. The aneroid bellows prevents the firing of the explosive train above 15,000 feet. Below this altitude, atmospheric pressure compresses the bellows sufficiently to release the firing pins that initiate booster caps and continue the detonation sequence to remove the recovery parachute and blade antenna severable cover and fire the recovery parachute catapult. The barostat lock initiator is located on the explosive component support bracket in the rocket motor compartment.

RECOVERY AND LANDING EQUIPMENT.

The recovery and landing equipment consists of stabilization components, the recovery parachute, landing and flotation components. The stabilization components consist of the stabilization glove, stabilization-brake parachute, pitch flaps and chin flaps. The stabilization glove which forms the forward portion of the wing glove is an integral part of the crew module. This glove section serves to stabilize the flight of the crew module until deployment of the recovery parachute. The pitch flaps, in the under surface of the glove section, and chin flaps under the forward section, assist in maintaining crew module horizontal stability. The stabilization-brake parachute, which is contained in a compartment in the center of the top aft section of the glove, is used to decelerate the crew module and assist in maintaining stable flight prior to recovery parachute deployment. The stabilization-brake parachute is a six foot flat diameter ribbon type parachute attached by two bridles to the outboard aft sections of the glove section. The recovery parachute has a ringsail canopy with a 70 foot flat diameter. The parachute is attached by two bridles to the crew module so that the module will maintain an upright and level attitude during descent. The parachute is housed in a container located between the seat bulkhead and the aft pressure bulkhead. This container rests on the parachute catapult pan. The catapult forcibly deploys the parachute at a velocity sufficient to ensure proper bag strip-off. A dynamic pressure "q" actuated selector, monitors aircraft speed to select one of three possible time delays. One time delay train includes an acceleration "g" sensor initiator that actuates when the crew module forward deceleration drops to 2 "g's". Continuation of the firing train next unlocks the barostat initiator. When below 15,000 feet, the barostat initiator, if unlocked will fire and in turn fire the catapult to deploy the recovery parachute. The parachute is initially deployed in a reefed configuration. The parachute is disreefed by three cutters which sever the reefing line shortly after line stretch is reached. The landing and flotation components consist of an inflatable landing impact attenuation bag, flotation bags and self-righting bags. The impact attenuation bag, located in the crew module floor, inflates automatically during descent and serves to cushion the landing impact. Regulated pneumatic pressure for inflation of the bag is contained in two storage bottles in the crew module. Pressure within the bag is maintained at 2 psi. Although the crew module is watertight and will float, additional buoyancy is provided by a flotation bag at each aft corner of the glove section and by an auxiliary flotation bag at the front of the crew module. Inflation of the aft flotation bags and auxiliary flotation bag is accomplished by manually pulling the initiator handles located on the canopy center beam. The pressure source for inflation of these bags is contained in storage bottles located in the crew module. The auxiliary flotation bag is provided for use only in event of cabin flooding and in that event to gain additional freeboard to open canopy hatches. It should be reserved and not used unnecessarily. Also, its deployment will cause the crew module to ride higher in the water, thereby becoming more affected by wave action to the detriment of crew comfort. In the event the aircraft is ditched, the crew module can be separated from the aircraft by pulling the severance and flotation initiator handle located on the canopy center beam. Pulling this handle will sever the crew module from the aircraft, inflate the aft flotation and self-righting bags and turn on the emergency oxygen supply.

WARNING

Do not apply pressure to the four canopy center beam handles until the release buttons have been fully depressed. To do so may bind the locking pins and prevent system activation.

SURVIVAL EQUIPMENT.

The survival equipment consists of locating aids, a combination bilge/flotation bag inflation pump, standard survival equipment, and a small quick rescue kit. The locating aids consist of a chaff dispenser, radio beacon set (AN/URT-27 or AN/URT-33), an AN/PRC-90 or ACR/RT-10A radio set, and a portable distress beacon light. The chaff dispenser, when armed, will activate to dispense chaff automatically during the ejection sequence. A control lever in the cockpit is provided to either arm or disarm the dispenser prior to ejection. This lever also arms the radio beacon set. The radio beacon set will emit an intermittent, modulated

tone to aid in rescue operations. The set is located behind the left seat. The set is connected to the crew module mounted emergency UHF antenna which erects upon ejection. The set may also be used as a portable utilizing its own retractable antenna. The AN/PRC-90 or ACR-RT-10A radio, located in the survival equipment stowage compartment, provides a means of two way voice communications. The portable distress beacon light, also located in the survival equipment stowage compartment, produces a powerful flashing light to aid in night rescue operations. The combination bilge/flotation bag inflation pump is operated by fore and aft motion of the control stick. This will cause simultaneous pumping of water overboard and inflation of the flotation bags. Over-inflation of the bags is prevented by relief valves. Standard survival equipment is provided for all climatic conditions. This equipment is stored in the survival equipment stowage compartment behind the right seat. Instructions on how to gain access to the survival equipment compartment are contained on a detachable instruction plate mounted on the back of the right seat behind the back seat cushion. The contents of the survival equipment stowage compartment will be determined by the applicable using command. A small quick rescue kit containing survival equipment useful for rescue aid in hostile territory when early rescue is expected is mounted on the rear bulkhead above the right headrest.

CREW MODULE EJECTION SEQUENCE.

WARNING

Under certain conditions of crew module weight and/or tail wind, zero altitude and zero airspeed, ejection capability may not be available. Because of the variables involved, ejection is not recommended at zero altitude with less than 50 KIAS.

The ejection sequence is initiated by squeezing and pulling either ejection handle located on the center console. All succeeding functions through landing are automatically actuated by dual explosive firing trains. Emergency oxygen system actuation is automatic, however, a manually actuated backup handle is available if required. After a delay of 0.35 second to allow for powered repositioning of crew members, the crew module is severed and the rocket motor is ignited. The noise of ejection will be loud but of short duration. Full thrust is sustained for approximately 1.0 second. Crew members may expect moderate ejection accelerations at low speeds becoming more severe at very high speeds. During the first six inches of crew module separation, pitch flaps and stabilization (chin) flaps rotate down into deployed position. Their function is to control crew module trim angle of attack and resulting spinal accelerations. The stabilization-brake parachute is deployed 0.15 second after crew module severance. This parachute provides necessary acceleration control and stabilization at speeds above 450 knots. Crew module pitch control at high speeds is provided by firing the rocket motor upper nozzle. The crew module usually rolls after ejection, with the roll rate dependent upon airspeed and the severity of roll-producing factors such as lateral c.g. eccentricity and ejection during sideslip. Rolls generally degenerate into an upright "Dutch roll" which is stopped by deployment of the recovery parachute, at which time a rapid change in crew module attitude occurs. Recovery parachute deployment is timed by a sequencing system which senses speed, acceleration, and altitude upon ejection. This system consists of three time delays (1.0, 1.6, and 4.4 seconds), a "q"-actuated selector, a "g"-sensor initiator, and a barostat lock initiator. The 4.4 second delay serves as a safety backup to the other sequencing components. At speeds below 300 (±30) knots and altitudes below 15,000 feet, the recovery parachute is deployed after a 1.0 second delay. At speeds above 300 (±30) knots and altitudes below 15,000 feet, recovery parachute deployment is controlled by a "g"-sensor initiator and is, thereby, delayed until crew module longitudinal (fore and aft) deceleration drops to a 2.2 "g's." This allows the crew module to decelerate to below the design limit airspeed of the recovery parachute. At altitudes above 15,000 feet, deployment is constrained by a barostat lock initiator. The barostat lock initiator is armed by one of three explosive trains, whichever fires first; these are from the "q"-actuated selector, the "g"-sensor initiator, or a 4.4 second time delay initiator. After the barostat lock initiator is armed and after the crew module falls through 15,000 feet, ambient pressure compresses the aneroid bellows causing the initiator to fire. The recovery parachute is then deployed upward at 45 feet per second. Manual deployment capability, which is operable at the crew member's discretion by means of the parachute deploy handle, is provided as a backup to the automatic barostat system.

WARNING

- The parachute deploy handle bypasses the barostat lock initiator. It should not be actuated above 15,000 feet; otherwise, failure of the recovery parachute may result.
- During ejection, the standby altimeter is rendered inoperative due to pyrotechnic gases entering and overpressurizing the pitot-static system. The instrument will then be unreliable, and should not be used for altitude indications after ejection.

The initial recovery parachute inflation and the associated opening shock loads are controlled by a reefing line that holds the parachute canopy opening to about 8 feet diameter. Parachute disreefing to full inflation occurs 2.5 seconds after suspension line stretch. Whereas free-fall from maximum altitude to 15,000 feet occurs in 85 seconds, the remaining descent time after recovery parachute deployment is about 7.5 minutes. Chaff deployment as an aid to radar tracking and automatic emergency UHF radio transmission occurs 3.0 seconds after ejection handles are actuated if the chaff control lever is in ON. The crew module repositions to its horizontal landing attitude and the emergency UHF antenna erects 7.0 seconds after recovery parachute deployment. A mild explosive report will be heard and a sudden dropping of the tail of the crew module will occur upon repositioning. Landing impacts are absorbed by the impact attenuation bag, which is fully inflated 7.25 seconds after recovery parachute deployment. Crew members may expect moderate or severe landing impact decelerations for the nominal weight crew module. In calm or no-wind conditions, it may not be necessary to release the recovery parachute.

DITCHING ESCAPE SEQUENCE.

If the aircraft is ditched, crew module severance and flotation bag deployment may be initiated manually by pulling the severance and flotation handle. When the handle is pulled the following sequence of events occurs: An initiator is fired to (1) fire the FLSC to separate the crew module from the aircraft, (2) remove the severable covers over the aft flotation bags and the self-righting bags, (3) fire the explosive valve in an air storage bottle to inflate the aft flotation bags and the left self-righting bag and (4) fire the explosive valve in an air storage bottle to inflate the right self-righting bag.

WARNING

Pulling the severance and flotation handle and the auxiliary flotation handle does not disable the rocket motor; it will still fire if either ejection handle is pulled. To preclude inadvertent firing of the rocket motor during the ditching sequence, both ejection handle safety pins must be installed.

Seat Fore and Aft Adjustment Lever.

The seat fore and aft adjustment lever (1, figure 1-48), located in front of the seat pan between the crewmember's legs, is provided to unlock the seat from the carriage to allow forward and aft adjustment. When the handle is pulled up, the seat will unlock to allow a maximum of 5 inches travel from full aft to full forward. Since this lever does not provide headrest adjustment, forward and aft adjustment of the seat will result in a tilting of the seat back.

Seat Adjustment Switches.

Vertical adjustment of each seat is provided by a switch located on each left and right sidewall (5, figure 1-25 and 5, figure 1-42) adjacent to the seat. Each switch has positions marked UP and DOWN and is spring-loaded to the center unmarked OFF position. Positioning a switch to either UP or DOWN energizes an electrical actuator to raise or lower the seat as selected. The seat has a maximum vertical travel of 5 inches.

CAUTION

Placement of objects under the right seat may cause damage to the fire warning system control boxes which are located on the floor, just forward of the right seat.

Headrest Adjustment Lever.

A headrest adjustment lever (7, figure 1-48), located on either side of each seat headrest is provided for fore and aft adjustment of the headrest. Depressing either lever will unlock the headrest allowing it to be moved either forward or aft. Releasing the lever will lock the headrest in place. Since the seat back is attached to the headrest, fore and aft movement of the headrest will cause the seat back to tilt.

CAUTION

Crews will not place objects behind the seat back as this may cause damage to the seat back and the adjustment mechanism.

Inertia Reel Control Handle.

The inertia reel control handle (8, figure 1-48), located on the left side of each seat headrest, is provided to lock or unlock the inertia reel. When the handle is in the up position, the inertia reel is locked. Pulling down on the handle unlocks the inertia reel, and the restraint harness can be pulled out.

Ejection Handles.

Two ejection handles (22, figure 1-47), one located on either side of the center console adjacent to the crewmember's seat, are provided to initiate the ejection cycle. When the lock release on the top of handle is depressed, the handle is released and may be pulled out. Pulling the handle out approximately ½ inch will fire the initiator to start the ejection sequence.

Recovery Parachute Deploy Handle.

The ring-shaped recovery parachute deploy handle (27, figure 1-47), located on canopy center beam assembly, is provided as an emergency means of deploying the recovery parachute should the normal method fail. Pressing a release button and pulling the handle will fire an initiator to deploy the parachute.

WARNING

Use caution to avoid inadvertent actuation of the recovery parachute deploy handle in flight prior to ejection. The parachute would deploy immediately with catastrophic results since there is no means of recovery parachute jettison with the module attached to the aircraft.

Recovery Parachute Release Handle.

The T-shaped recovery parachute release handle (24, figure 1-47), located on the canopy center beam assembly, is provided to release the recovery parachute from the crew module after landing. Pressing a release button on either side of the handle and pulling the handle fires the parachute release retractors at the bridle attaching points releasing the bridles from the crew module. The recovery parachute release handle cannot be pulled until the severance and flotation handle has been pulled.

WARNING

Do not pull the recovery parachute release handle prior to landing as the parachute will separate from the crew module allowing the crew module to free fall.

Auxiliary Flotation Handle.

The T-shaped auxiliary flotation handle (26, figure 1-47), located on the canopy center beam assembly, is provided to inflate the auxiliary flotation bag on the front of the crew module. Pressing a release button on either side of the handle and pulling the handle out fires an initiator which in turn removes the severable cover over the auxiliary flotation bag and fires an explosive valve in an air storage bottle to inflate the bag.

Severance and Flotation Handle.

The severance and flotation handle (25, figure 1-47), located on the canopy center beam assembly, is provided for escape in the event the aircraft is ditched. Pressing a release button on the lower side of the handle and pulling the handle down will fire the FLSC and guillotines, separating the crew module from the aircraft, and will inflate the aft flotation bags and the self-righting bags. Pulling the handle will also activate the emergency oxygen system. The rocket motor is not ignited in this sequence; however, it is not disabled and will still fire if either ejection handle is pulled.

WARNING

- Do not actuate the severance and flotation handle when personnel are within 50 feet of the crew module because of explosive severance of metal covers.
- Do not actuate the severance and flotation handle prior to impact. To do so may result in severe post-landing gyrations and rupture of the bags.

Bilge/Flotation Bag Inflation Pump.

The bilge-flotation bag inflation pump (figure 1-49) provides simultaneous inflation of the flotation bags and pumping of water overboard. The pump is operated by fore and aft motion of the control stick. After landing, the bilge pump lock pin is removed from the pin stowage hole and inserted in the operating hole. This connects the pump to the control stick. A plunger, adjacent to the pin stowage hole, must be pushed in to open the pump air and water outlet valves. Movement of the stick will then operate the pump.

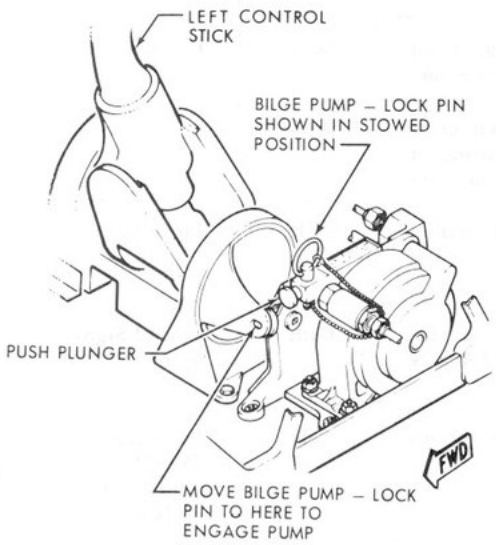
WARNING

Do not engage the bilge pump until after ejection. To do so will prevent the cabin pressure regulator from relieving cabin pressure and the canopy hatches may be blown off when they are opened during normal operation.

CAUTION

Failure to push in the plunger may result in damage to the pump and render it inoperative.

Bilge/Flotation Bag Inflation Pump



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Figure 1-49.

Chaff Dispenser Control Lever.

The chaff dispenser control lever (14, figure 1-47), located on the aft bulkhead, is used to arm or disarm the crew module chaff dispenser. The lever is labeled CHAFF and has two positions marked ON and OFF. Placing the lever to the ON (up) position opens a mechanical interrupt to allow explosive train propagation to the chaff dispenser release mechanism. When the crew module is ejected, the explosive train releases the chaff dispenser and the slip stream dispenses the chaff. Placing the lever to the OFF (down) position closes the mechanical interrupt, thereby disarming the dispenser. When the lever is in the ON position, the radio beacon set will be automatically activated as the crew module ejects.

COMMUNICATION EQUIPMENT.

For a listing and function of communications equipment see figure 1-50.

UHF COMMAND RADIO (AN/ARC-109).

The UHF command radios provide air-to-air and air-to-ground communications. Two complete UHF communication sets are installed in the aircraft and are designated No. 1 and No. 2. Each system consists of a receiver-transmitter (RT) unit, a control panel, an antenna selector, and blade type upper and lower antennas. See figure 1-69 for antenna locations. The two RT units and guard receiver are located in the right forward equipment bay. Both radio sets may be operated simultaneously except for transmission on the same channel, where interference between sets occurs. There are 3500 channels available in 50 kilohertz increments in the frequency range from 225.00 to 399.95 megahertz .The guard receiver monitors the guard frequency of 243.0 megahertz when guard function is selected. The control panel allows selection of 20 preset channels and manual selection of any frequency in the frequency range of the radio. The upper and lower antennas complement each other to provide omni-directional antenna coverage. An automatic feature allows the receiver to select the antenna which receives the first usable signal; however, either the upper or lower antenna may be manually selected.

Note

When operating the UHF in the automatic antenna selection mode, the antenna selector has a transmission memory circuit which automatically connects the transmitter to the antenna last used for transmission. If the channel or frequency is changed to another station or the aircraft position has changed relative to the station, this may be the wrong antenna for the next transmission and difficulty will be encountered in gaining contact. Should this occur, manually select the upper or lower antenna and repeat the transmission to gain contact.

The UHF-1 shares the same antennas with the TACAN and UHF-2 shares with the IFF. Number 1 UHF operates on 115 vac from the essential ac bus and 28 vdc from the essential dc bus. Number 2 UHF operates on 115 vac from the left main ac bus and 28 vdc from the main dc bus.

UHF Radio Function Selector Knob.

The UHF radio function selector knob (4, figure 1-51), located on the UHF radio control panel, has four positions marked OFF, MAIN, BOTH, and ADF. Rotat-

Communications and Avionics Equipment

Type	*Designation*	*Function*	*Operator*	*Range*
UHF RADIO	AN/ARC-109	Air-to-air and air-to-ground voice communication	Both	Line-of-Sight
HF RADIO	AN/ARC-123	Air-to-air and air-to-ground long range voice communications	Right	5000 miles
RADIO BEACON SET	AN/URT-27 or AN/URT-33	Provides a tone signal for rescue aircraft to home on	Both	Line-of-Sight
INTERPHONE	AN/AIC-25	Interphone between crew members and monitoring of all communications facilities	Both	
IDENTIFICATION RADAR (IFF-SIF)	AN/APX-64	Provides coded IFF replies to an interrogating ground radar station	Right	Line-of-Sight
TACAN	AN/ARN-84V	Provides bearing and distance informaton to TACAN stations	Both	Line-of-Sight up to 300 NM
ILS	AN/ARN-58A	Provides visual indications for ILS approaches	Left	Localizer 45 NM Glide Slope 25 NM
RADAR ALTIMETER	AN/APN-167	Provides precise altitude measurements from 0 to 5000 feet	Both	0-5000 feet
TERRAIN FOLLOWING RADAR	AN/APQ-134	Provides all weather, low altitude terrain following, obstacle avoidance and blind letdown capability	Left	Line-of-Sight up to 15 miles
BOMBING-NAVIGATION SYSTEM	MARK II B	Provides integrated bombing and navigation capabilities in conjunction with other systems in the aircraft	Right	
ATTACK RADAR	AN/APQ-114	All weather navigation, fix-taking, bombing, and tanker rendezvous	Right	Line-of-Sight up to approximately 200 miles
RADAR TRANSPONDER	AN/APX-78	Provides air-to-air and air-to-ground range and bearing determination	Right	Line-of-Sight up to approximately 200 miles
OPTICAL DISPLAY SIGHT	AN/ASG-25	Provides tanker rendezvous and air-to-ground attack capability and duplicate information as displayed on ADI for instrument flying	Left	Line-of-Sight

Figure 1-50.

ating the knob to the MAIN position activates the RT unit for normal transmission and reception on the channel selected; the guard receiver is inoperative. Rotating the knob to the BOTH position also activates the RT unit for normal use and in addition activates the guard receiver to allow monitoring guard frequency. The ADF position is inoperative and should not be selected.

UHF Radio Mode Selector Switch.

The three-position UHF radio mode selector switch (9, figure 1-51), located on the UHF radio control panel, is marked PRESET, MANUAL, and GUARD. The PRESET position is used when selecting one of the 20 preset frequencies. MANUAL is used for frequencies that are selected by the manual frequency selector knobs. GUARD tunes the main RT unit to the guard frequency of 243.0 megahertz.

UHF Radio Preset Channel Selector Knob.

The preset channel selector knob (8, figure 1-51), located on the UHF radio control panel, permits selection of one of twenty preset frequencies. With the mode selector switch at PRESET, movement of the preset channel selector knob changes the frequencies to that of the channel selected. There are 20 channels, numbered 1 through 20, that may be individually selected. The number of the channel selected is displayed in a window above the knob. Frequencies for each channel are written on a channel frequency log located under the pilot's glare shield. Frequencies of the preset channels can be changed during flight.

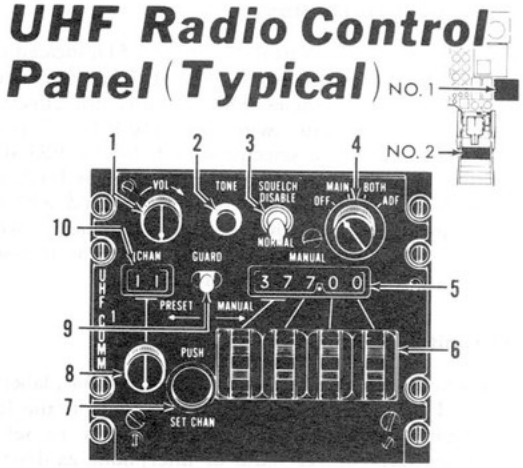
UHF Radio Manual Frequency Selector Knobs.

Four thumb actuated UHF radio manual frequency selector knobs (6, figure 1-51), located on the UHF radio control panel, are provided for manually selecting frequencies. Manual frequency selection can be made in steps of 50 kilohertz from 225.00 through 399.95 megahertz. The first selector knob will select the first two digits of the desired frequency. The second, third and fourth knobs select the third, fourth and fifth frequency digits respectively. The selected frequency is displayed in a window on the face of the UHF radio control panel.

UHF Radio Volume Control Knob.

The volume control knob (1, figure 1-51), located on the UHF radio control panel, increases the volume of the receiver when turned clockwise and decreases it when turned counterclockwise.

UHF Radio Control Panel (Typical)



1. Volume Control Knob.
2. Tone Button.
3. Squelch Switch.
4. Function Selector Knob.
5. Frequency Indicator Window.
6. Manual Frequency Selector Knobs (4).
7. Channel Set Pushbutton.
8. Preset Channel Selector Knob.
9. Mode Selector Switch.
10. Preset Channel Indicator Window.

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Figure 1-51.

Squelch Switch.

The squelch switch (3, figure 1-51), located on the UHF radio control panel, is a two-position switch marked DISABLE and NORMAL. The switch is provided so that the squelch can be selected for compatibility with the strength of the signal being received. Placing the switch to DISABLE turns off the squelch. Placing the switch to NORMAL turns the squelch on. If UHF reception is breaking up and intermittent, switching to DISABLE may restore reception.

Tone Button.

The tone button (2, figure 1-51), is located on the UHF radio control panel. With the UHF radio in operation, depressing the button will interrupt reception and transmit a continuous wave (CW) 1000-Hz tone signal on the selected frequency.

Channel Set Pushbutton.

The channel set pushbutton (7, figure 1-51), located on the UHF radio control panel, is used to set or change preset channel frequencies. The button is only effective when the mode selector switch is in the PRESET position. With the mode selector switch in the PRESET position and with the preset channel selector knob set to the desired channel, depressing the button will set the frequency selected in the manual frequency window into the desired channel. The button is recessed in a guard to prevent inadvertent actuation.

Transmitter Selector Knobs.

Two transmitter selector knobs (3, figure 1-54), labeled INT, UHF-1, UHF-2 and HF, are located on the left and right communications control panels to select either the UHF or HF radio or interphone as desired for transmission.

Microphone Switches.

A three-position spring-loaded, pivot type microphone switch marked TRANS and INPH with an unmarked off (center) position, is located on each right throttle (5, figure 1-5). It is spring-loaded to the center OFF position.

UHF Radio Antenna Selector Switches.

Two three-position UHF radio antenna selector switches (1 and 2, figure 1-59), located on the antenna select panel are labeled UHF-1 and UHF-2. The switches are marked UPPER, AUTO, and LOWER and control the selection of upper and lower antennas. Placing the switch to AUTO causes the antenna selector to control the antenna switching relay to select the correct antenna. Placing the switch in the LOWER or UPPER position controls the antenna relay directly to allow manual selection of either the upper or lower antenna.

UHF Radio Frequency Indicator Window.

The frequency indicator window (5, figure 1-51), located on the UHF radio control panel, indicates the frequency selected. The window has five digits, which are set by frequency selector knobs.

HF RADIO (AN/ARC-123).

The HF radio provides long range high frequency single side band communications. The radio operates in three modes: SSM, single side band; AME, amplitude modulation equivalent, and FSK, frequency shift keying. The FSK position is inoperative at this time. There are 280,000 channels available in 100 Hz increments in the frequency range of 2000 through 29,999.9 kilohertz. Components of the system include a receiver-transmitter (RT) unit, amplifier power supply, control panel, antenna, coupler, antenna coupler control, and remote capacitor. The RT unit, amplifier power supply and antenna coupler controls are located in the right forward electronic bay. The antenna coupler is located in the aft fuselage below the antenna which is a part of the vertical stabilizer and dorsal fin. The remote capacitor is located at the forward top of the dorsal fin. Refer to figure 1-69 for antenna location. The control panel is located on the center console. Once the system is placed in operation either crew member can use the equipment.

HF Radio Mode Selector Knob.

The HF radio mode selector knob (7, figure 1-52), located on the HF radio control panel has four positions marked OFF, SSB, AME and FSK. In the OFF position, the system is deenergized. Rotating the knob to SSB, provides single side band capability of operation. The AME position provides the capability of amplitude modulation equivalent capability of operation. The FSK position is inoperative.

HF Radio Frequency Selector Knobs.

Six frequency selector knobs (1, figure 1-52), located on the HF radio control panel, provide a means of setting desired frequencies. Each knob has an indicator line drawn to the window it controls on the frequency indicators. To prevent selection of frequencies below 2000 KHz, the 1000 KHz and 10,000 KHz knobs are interlocked. In order to select 0 with the 10,000 KHz knob, a 2 through 9 must be present in the 1000 KHz window. In order to select a 1 or 0 with the 1000 KHz knob, a 1 or 2 must be present in the 10,000 KHz window.

HF Radio Frequency Indicator Window.

The HF radio frequency indicator window (3, figure 1-52), located on the HF radio control panel, has six digits indicating the frequency selected. Each window has an indicator line drawn to its corresponding frequency selector knob.

RF Gain Control Knob.

The radio frequency gain control knob (6, figure 1-52), located on the HF radio control panel is labeled RF GAIN. The knob is used to adjust the radio frequency gain of the receiver as desired during normal operation, to provide better signal to noise ratio.

Note

The RF GAIN control knob should not be used for adjusting the audio volume.

Noise Blanking Control Knob.

The noise blanking control knob (5, figure 1-52), located on the HF radio control panel, is labeled NOISE BLANK. The knob is used to reduce the effects of impulse noise, either natural or man-made, including ECM.

Note

Adjustment of the noise blanking control knob is not critical. If high impulse noise is encountered, rotation of the knob will blank the noise without affecting voice signals.

Volume Control Knob.

The volume control knob (4, figure 1-52), located on the HF radio control panel, is labeled VOLUME. When the knob is rotated clockwise, the audio output is increased.

Note

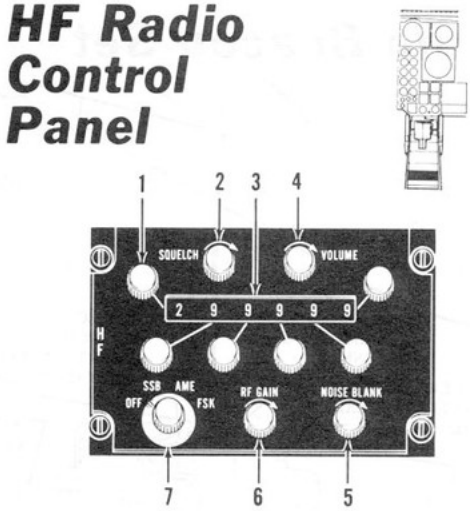
- For most applications, the volume control knob should be set to obtain approximately the same audio level as the UHF to balance the audio out of the interphone. After this balance has been obtained, any additional audio volume corrections should be made with the interphone volume controls.
- Care should be exercised in the use of this control as audio can be disabled if this control is set too far counterclockwise.

Squelch Control Knob.

The squelch control knob (2, figure 1-52), located on the HF radio control panel, is labeled SQUELCH and is used to adjust the threshold of squelch operation in the SSB or AME mode. Adjustment of the squelch must be done after the other controls have been set. First, place the squelch control fully clockwise. Then with no signal on the channel, turn the squelch control counterclockwise slowly until the noise is subdued. Rotate the control knob slowly clockwise and find the place where the set breaks in and out of squelch. This is the proper squelch adjustment for the prevailing noise conditions. If the noise conditions change, the squelch must be readjusted. If desired signals are found to break in and out of squelch, adjust to a higher clockwise position or operate without squelch.

Note

Care must be exercised in the use of this control, as the receiver audio can be disabled if the squelch is set too far counterclockwise.



1. Frequency Selector Knobs (6).
2. Squelch Control Knob.
3. Frequency Indicator Window.
4. Volume Control Knob.
5. Noise Blanking Control Knob.
6. RF Gain Control Knob.
7. Mode Selector Knob.

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Figure 1-52.

RADIO BEACON SET.

A radio beacon set (figure 1-53), located behind the left seat (15, figure 1-47), is provided for use as a survival radio to aid in crew rescue after ejection. The radio operates on a self-contained battery. The set is connected to a crew module mounted antenna which is automatically erected when the crew module ejects. An on-off switch on the face of the radio is provided to arm the set. A safety plug located adjacent to the on-off switch must be removed to place the set in operation. When the switch is positioned to ON and the safety plug removed, the radio will transmit an intermittent modulated tone signal for the rescue aircraft to home on. The radio can also be removed from the crew module and used as a portable rescue aid. A telescoping antenna, stowed in the radio, can be extended when the radio is used as a portable. When the on-off switch is placed to ON and the chaff dispenser control lever (14, figure 1-47) is ON, the radio will be automatically actuated whenever ejection occurs.

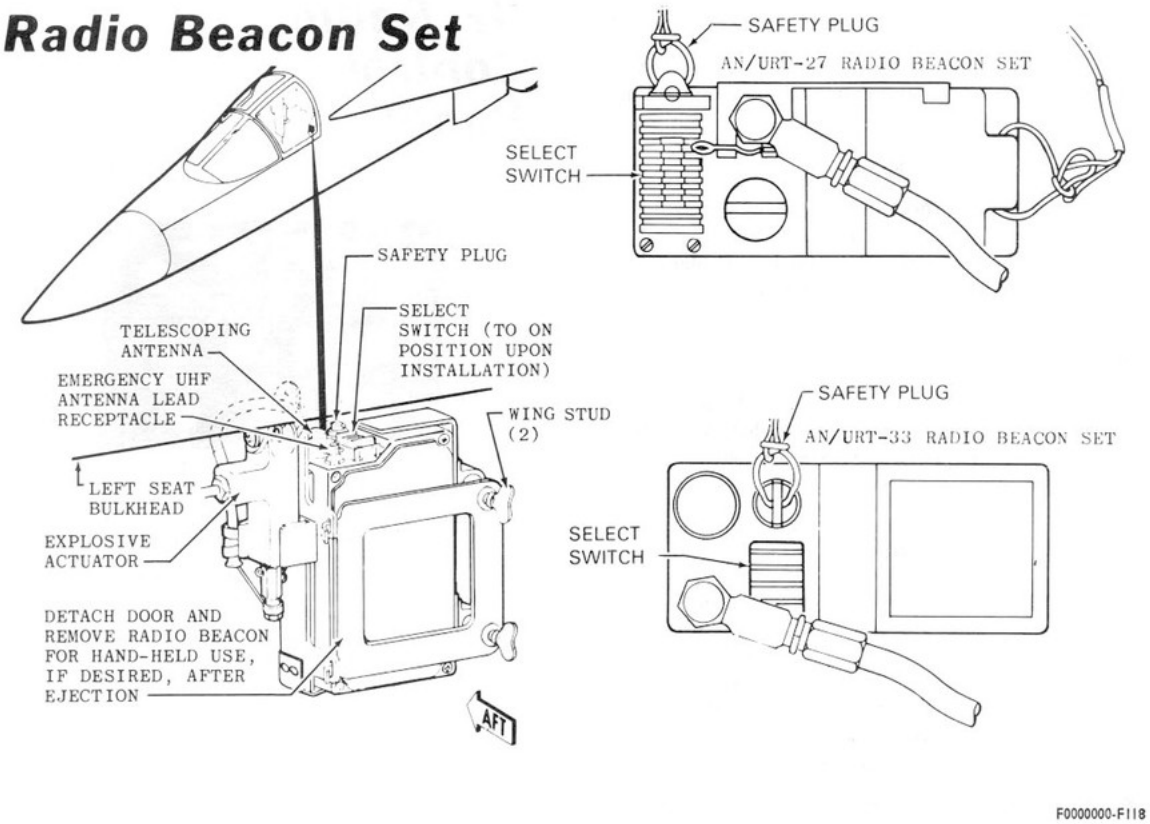


Figure 1-53.

INTERPHONE (AN/AIC-25).

The interphone provides the following functions: communications between crew members and between crew members and ground crew; monitoring and volume control UHF radio, HF radio, TACAN, ILS, RHAW and missile tone reception; and hot mic and call capability. Two identical communications panels (figure 1-54) are located on the left and right consoles. Interphone stations for ground crew operation are located in the nose wheel well, main landing gear well and in the ground power receptacle. Power is applied to the interphone whenever power is on the aircraft.

Communications Monitor Knobs.

Eight push-pull communication monitor knobs (1, figure 1-54), located on each communications panel, are marked and monitor the functions as follows:

INT	— Interphone
UHF 1	— UHF Command Radio 1
UHF 2	— UHF Command Radio 2 and IFF mode 4
HF	— HF Radio
ILS	— ILS and Localizer or TFR aural command
TACAN	— TACAN Identification
RHAWS	— Radar Homing and Warning System
LISTEN	— Hot Mic Reception

Other signals fed to the communications panel are the landing gear warning tone and the stall warning signal. The monitor knobs are pulled out to turn on and pushed in to turn off. Each knob (except HOT MIC TALK) may be rotated for volume control.

Master Volume Control Knob.

A master volume control knob (4, figure 1-54), located on each communications panel, controls the volume

of all inputs to the panel. If a change to an individual input volume is desired, it can be accomplished by rotating the appropriate monitor knob.

Hot Microphone Talk Buttons.

A push-pull (HOT MIC) hot microphone talk button (5, figure 1-54), located on each communications panel, provides a continually operating microphone when it is pulled. When this switch is pulled, the crew member can transmit without using the microphone switch, however, the crew member at each station must have his hot mic listen monitor buttons pulled to receive the transmission.

Note

The use of HOT MIC TALK when there is a high background noise level in the cockpit will interfere with UHF communications. Under these conditions, use of the INPH push-to-talk mode rather than HOT MIC TALK is recommended.

Interphone Call Button.

The interphone call button (2, figure 1-54), located on the communication panel, permits either crew member to call the other crew member or the ground crew. Depressing either call button boosts the interphone volume of the other stations and reduces the operator's side tone level, allowing the call signal to override the other station's reception. The call signal will override the reception at the other station regardless of the position of the communications monitor knob or transmitter selector knobs at either station.

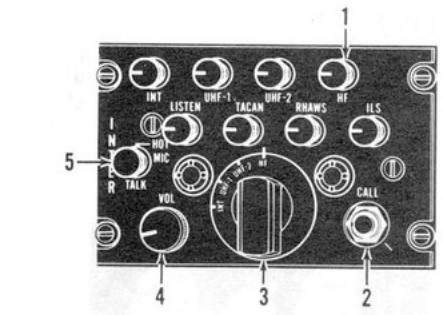
Transmitter Selector Knob.

Two seven-positon transmitter selector knobs (3, figure 1-54), located on each communications panel, are provided to select either UHF or HF radio. The knobs are marked INT, UHF 1, UHF 2, and HF. Three spare positions are unmarked. In either the HF or UHF positions only the radio transmitter selected will be keyed when the microphone switch is moved to the TRANS position. In addition, the UHF 1, UHF 2, or HF position will allow continuous monitoring of the respective receiver, regardless of the position of the communications monitor knob. Regardless of the position of the transmitter selector switch, the interphone may be used by moving the microphone switch on the throttle to the INPH position.

Microphone Switch.

A three-position pivot type microphone switch, marked TRANS and INPH with an unmarked OFF position, is located on each right throttle (5, figure 1-5).

Communications Panel





1. Monitor Knobs (8).
2. Interphone Call Button.
3. Transmitter Selector Knob.
4. Master Volume Control Knob.
5. Hot Microphone Switch.



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Figure 1-54.

The switch is spring-loaded to the center OFF position. When the transmitter selector switch is in the INT position, moving the switch to either position allows interphone use.

Transmit Foot Switch.

A foot switch, located by the right crew member's left foot, is marked TRANSMIT. The switch is spring-loaded to the off position and, when depressed, allows transmission.

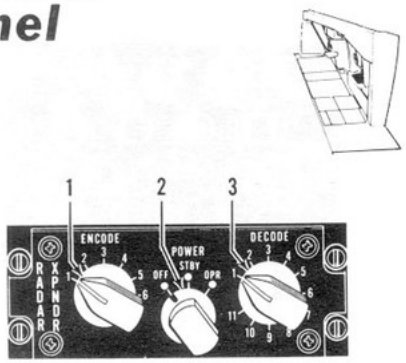
Interphone Foot Switch

A foot switch, located by the right crew member's right foot, is marked INTERPHONE. The switch is spring-loaded to the off position and, when depressed, allows interphone use.

Exterior Interphone Stations.

Exterior interphone stations in the nose wheel well, the main landing gear wheel well, and the ground power receptacle have a volume control knob, a call

Radar Transponder Control Panel



1. Encode Mode Selector Knob.
2. Power Selector Knob.
3. Decode Mode Selector Knob.

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Figure 1-55.

pushbutton, and a receptacle for ground cord plug in. The call pushbutton and volume control knob function the same as those controls on the interphone control panel.

I-BAND RADAR TRANSPONDER (AN/APX-78).

The I-band radar transponder enables ground and airborne I-band radar systems to identify and determine the range and bearing of the aircraft. The equipment consists of a radar transponder control panel, a radar transponder, a heat exchanger, and an I-band horizontally polarized antenna. The equipment does not perform interrogation but only transmits replies to selected interrogations. When interrogated by I-band pulses of the proper spacing, the radar transponder component transmits high level replies in the same frequency band. The I-band transponder is capable of replying to single or several spacings of double pulse interrogations, depending on the setting of the DECODE mode switch. The characteristics of the I-band transponder reply will be a single pulse or double pulse with several selections of spacing depending on the setting of the ENCODE mode switch.

ENCODE MODE SELECTOR KNOB.

The six-position encode mode selector knob (1, figure 1-55), located on the radar transponder control panel, is marked 1 through 6. In position 1, the transponder will reply with a single pulse. In position 2 through 6, the transponder will reply with a pulse pair of the selected spacing when properly interrogated. The pulse spacing of the pulse pairs is as follows:

Encode Switch Position	*Pulse Spacing*
2	4 NM
3	6 NM
4	8 NM
5	10 NM
6	12 NM

POWER SELECTOR KNOB.

The three-position power selector knob (2, figure 1-55), located on the radar transponder control panel, is marked OFF, STBY and OPR. When the knob is rotated to the OFF position, the system is deenergized. When the knob is rotated to the STBY position, power is supplied, and the receiver is operational, but replies are inhibited. When the knob is rotated to OPR, the system is operational and will respond to a proper interrogation. The I-band transponder will be operational within one minute after being changed from the OFF to OPR position and will meet all performance requirements within five minutes.

DECODE MODE SELECTOR KNOB.

The eleven-position decode mode selector knob (3, figure 1-55), located on the radar transponder control panel, is marked 1 through 11. When the knob is rotated to the number 1 position, the transponder will respond to pulse combinations single or multiple in the proper radio frequency band. In position 2 through 11, the transponder will respond to pulse pairs of the selected spacing.

TACTICAL AIR NAVIGATION SYSTEM (AN/ARN-84V).

The tactical air navigation system (TACAN) enables the aircraft to display continuous indications of its distance and bearing from any selected TACAN station located within a line-of-sight distance. There are 126 channels available for selection. The equipment consists of the TACAN receiver-transmitter and its control panel. Two antennas, one on top of the fuselage and the other beneath the fuselage (figure 1-69), function to keep the TACAN receiver locked on to the antenna receiving a usable signal. The TACAN equipment also has an air-to-air mode and can be used between two

aircraft having TACAN with air-to-air capability for range information only. The TACAN works in conjunction with the instrument system coupler, the bearing distance heading indicator, the optical display sight, the horizontal situation indicator, the attitude director indicator, and through the interphone control panel for audio output. The TACAN control panel is located on the center console.

TACAN FUNCTION SELECTOR KNOB.

The function selector knob (3, figure 1-56), located on the TACAN control panel, has four positions marked OFF, REC, T/R, and A/A. In the REC position, the set will receive bearing and audio identity signals only, but range information will not be displayed because the TACAN transmitter is not on. In the T/R position, both the receiver and the transmitter are operative, the system will receive and display both range and bearing of the station being interrogated, and audio identity signals are fed into the interphone system. In the A/A (air-to-air) position, the set will transmit and receive to and from another aircraft having air-to-air capability. To operate in this mode, the air-to-air mode in both aircraft must be selected, and the channels selected must be 63 channels apart. As an example, if the TACAN in one aircraft is on channel 10, the TACAN in the other aircraft must be selected to channel 73. In the A/A mode, the TACAN will provide information for up to 5 aircraft to home on a single aircraft.

TACAN CHANNEL SELECTOR KNOBS.

Two channel selector knobs (4, figure 1-56), located on the TACAN control panel, are used to select any of 126 TACAN channels. The left knob selects the tens and hundreds channel digits; the right knob selects the units channel digit. The selected channel is digitally displayed on the CHAN indicator above the TEST switch.

Note

Interference between the IFF and TACAN channels 1 thru 9, 63 thru 72, and 126 can prevent TACAN lock-on. Avoid using these channels in the air-to-air mode.

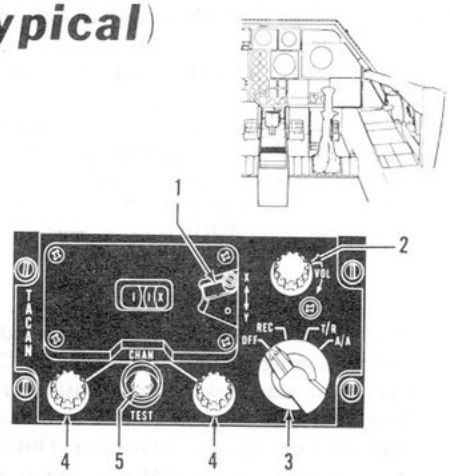
TACAN CHANNEL MODE SWITCH.

The channel mode switch (1, figure 1-56), located on the TACAN control panel, selects either the X or Y mode of pulse coding. The Y position is inoperative.

TACAN SELF TEST SWITCH.

When the self test switch (5, figure 1-56), located on the TACAN control panel, is depressed, test values of 180 (±3) degrees bearing and 0.00 (±0.5) miles will be displayed on the HSI and BDHI (if installed).

Tacan Control Panel (Typical)



1. Channel Mode Switch
2. Volume Control Knob
3. Function Selector Knob
4. Channel Selectors
5. Self Test Switch

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Figure 1-56.

TACAN VOLUME CONTROL KNOB.

A volume control knob (2, figure 1-56), located on the TACAN control panel, provides a means for controlling the volume of the audio identity code.

TACAN ANTENNA SELECTOR SWITCH.

The three-position TACAN antenna selector switch (3, figure 1-59) is located on the antenna select panel. The switch is marked UPPER, AUTO and LOWER. Placing the switch to AUTO causes the antenna selector to control the antenna switching relay to select the correct antenna. Placing the switch to UPPER or LOWER controls the antenna relay directly to allow manual selection of either the upper or lower antenna.

TACAN OPERATION.

1. Set function selector knob to desired mode.
2. Position antenna selector switch to AUTO.
3. Rotate channel selector knobs to desired channel.
4. Set channel mode switch to X.
5. Pull out TACAN communications monitor knob and adjust volume as desired.

6. Adjust volume control as desired.
7. Set ISC mode selector knob to TACAN.
8. Set the desired TACAN course in HSI course selector window.
9. Monitor ADI, ODS, HSI, and BDHI for proper indications.
10. To initiate a system self test, move the function selector knob out of the OFF position, momentarily depress the self test switch and observe 180° (±3°) bearing and 0.0 (±0.5) miles range on the HSI and BDHI (if installed).

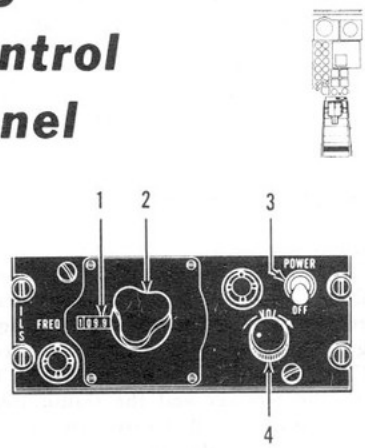
Note

It is possible that improperly adjusted or malfunctioning ground or airborne TACAN equipment may lock on to a false bearing. This error will probably be plus or minus 40 degrees or multiples of 40 degrees. This is an inherent error in the TACAN system, consequently, bearing information should be cross-checked against other navigation aids whenever possible. When false lock-on occurs, it is possible to correct the malfunction by switching to another channel and back to the desired channel or turning the set off and back on again. This deficiency does not affect the range display.

INSTRUMENT LANDING SYSTEM (ILS) (AN/ARN-58A).

The ILS provides the capability of making instrument approaches to runways equipped with localizer, glide slope and marker beacon equipment. The system consists of three receivers, one each for localizer, glide slope and marker beacon; four antennas, two for localizer and one each for glide slope and marker beacon, a control panel and a marker beacon light. The localizer and glide slope receivers operate on 20 fixed frequency channels which may be selected on the control panel. Localizer identification signals are supplied to the headset for station identification. Localizer and glide slope steering and deviation signals are provided to the instrument system coupler for display on the ADI, HSI, and ODS. Warning flags on the ADI become visible whenever the signal level on the selected frequency is too weak to be usable or is unreliable. Refer to "Instruments," this section, for the tie-in of the ILS and Integrated Flight Instruments. The marker beacon receiver operates on a fixed frequency of 75 megahertz and, when over a beacon facility, will provide a signal to the marker beacon lamp. Power is applied to the marker beacon receiver whenever power is on the aircraft. Refer to "Instrument Flight Procedures," Section VII, for instrument landing system operating procedures.

ILS Control Panel



1. Frequency Window.
2. Frequency Selector Knob.
3. Power Switch.
4. Volume Control Knob.

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Figure 1-57.

ILS FREQUENCY SELECTOR KNOB.

The frequency selector knob (2, figure 1-57), located on the ILS control panel, allows individual selection of 20 ILS channels ranging in localizer frequencies from 108.1 to 111.9 megahertz in 0.2 megahertz increments. There is a detent position of the knob for each channel. Each localizer frequency selected is paired with a glide slope frequency between 329.3 and 335.0 megahertz. The frequency of each channel selected is displayed in a digital window to the left of the knob.

ILS POWER SWITCH.

The power switch (3, figure 1-57), located on the ILS control panel, is a two-position switch marked POWER and OFF. When the switch is placed to POWER, power is applied to the localizer and glide slope receivers.

ILS VOLUME CONTROL KNOB.

The volume control knob (4, figure 1-57), located on the ILS control panel, adjusts the volume of the localizer station identification signal.

MARKER BEACON LAMP.

The marker beacon lamp (15, figure 1-6), located on the left main instrument panel, provides a visual coded station signal when the aircraft is over a marker beacon facility. When lighted, the words MARKER BEACON are displayed in green.

Note

The marker beacon lamp may blink during HF radio transmission on some frequencies due to electro magnetic interference. This is not a malfunction of either the marker beacon or HF radio and should be no cause for alarm.

IFF SYSTEM (AN/APX-64). (AIMS)

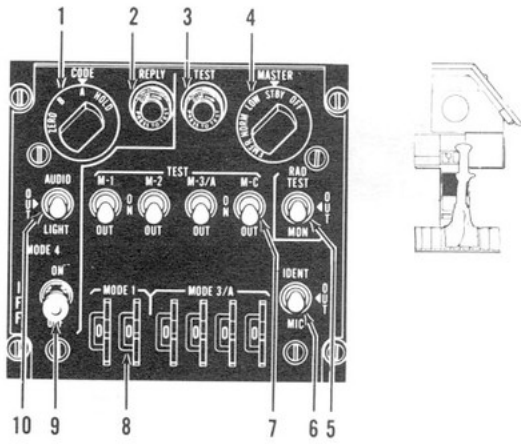
The air-to-ground IFF system provides for Mark X IFF with selective identification feature (SIF), automatic altitude reporting, and Mark XII (mode 4) encrypted IFF.

Operation is possible in any one of five modes, with the capabilities of I/P (identification of position) and emergency identification. The modes of operation have the following significance: Mode 1—Security Identity, Mode 2—Personal Identity, Mode 3/A—Traffic Identity, Mode 4—Encrypted Identity and Mode C—Altitude Interrogation. The equipment consists of an IFF control panel, a transmitter-receiver, a mode 4 computer, an antenna lobing switch, and two radiator-type antennas. The equipment does not perform interrogation but only transmits coded replies to correctly coded interrogations. Two blade type antennas, an upper and lower, are provided. See figure 1-69 for antenna locations. The lobing switch rapidly transfers contact of the transmitter-receiver from one antenna to the other. This constant alternation eliminates blank spots in the antenna pattern caused by aircraft structure. Mode 2 and 4 code settings are set into the receiver-transmitter on the ground and thus are fixed for any one flight. Mode 1 and 3/A codes are set up at the control panel. All modes can be turned on or off at the control panel. Replies to modes 1, 2, 3/A, 4 and C interrogations, as well as to I/P and emergency replies, are shown on the ground station equipment. The system will go to code 7700 in mode 3/A when operating in the emergency mode. The mode C provides altitude information from the CADC to the ground in 100 foot increments.

Note

Mode C should not be used with a known or suspected CADC malfunction, as this could result in the tarnsmission of erroneous altitude information.

IFF Control Panel



1. Mode 4 Code Control Knob.
2. Reply Lamp.
3. Test Lamp.
4. Master Control Knob
5. Rad Test/Monitor Switch.
6. Identification-of-Position Switch.
7. Mode Select/Test Switches (4).
8. Code Selector Wheels.
9. Mode 4 Control Switch.
10. Mode 4 Monitor Control Switch.

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Figure 1-58.

An optional low sensitivity setting provision restricts sensitivity so that replies are made only to local interrogations.

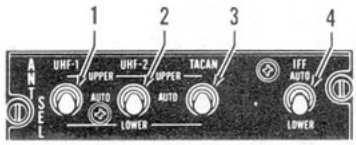
IFF MASTER CONTROL KNOB.

The five-position IFF master control knob (4, figure 1-58) is located on the IFF control panel. The knob positions are marked OFF, STBY, LOW, NORM and EMER. The OFF position removes power from the set and also zeroizes mode 4 code settings. In STBY, the equipment is turned on and warmed up but will not transmit. In LOW, only local (strong) interrogations are recognized and answered. In NORM, full range recognition and reply occurs. The knob must be pulled outward to position it to EMER. When the knob is positioned to EMER, an emergency-indicating pulse group is transmitted each time a mode 1, 2, or 3/A interrogation is recognized.

IDENTIFICATION-OF-POSITION SWITCH.

The identification-of-position (I/P) switch (6, figure 1-58), located on the IFF control panel, is used to control transmission of I/P pulse groups. The switch has

Antenna Select Panel





1. UHF-1 Radio Antenna Selector Switch.
2. UHF-2 Radio Antenna Selector Switch.
3. TACAN Antenna Selector Switch.
4. IFF Antenna Selector Switch.

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Figure 1-59.

three positions marked MIC, OUT, and IDENT. When the switch is momentarily held in the spring-loaded IDENT position, the I/P timer is energized for 15-30 seconds. If a mode 1, 2, or 3/A interrogation is recognized within this 15-30 second period, I/P replies will be made. When the switch is placed in the MIC position, the I/P pulse group will be transmitted in reply to a mode 1 or 3/A interrogation as long as a microphone switch is held to the TRANS position and for 15-30 seconds after the microphone switch is released. The transmitter selector knob, at the crew station being used, must be in the UHF position to allow transmission of I/P groups with the microphone switch. When the microphone switch is open, transmission of the I/P pulse groups will be withheld. Placing the switch to the OUT position prevents transmission of I/P groups.

IFF ANTENNA SELECTOR SWITCH.

The two-position antenna selector switch (4, figure 1-59), located on the antenna select panel, is marked AUTO and LOWER. When the switch is placed to AUTO, the antenna lobing switch rapidly cycles contact of the receiver-transmitter between the upper and lower antenna to provide thorough antenna pattern coverage. When the antenna selector switch is placed to LOWER, the lower antenna will be used to receive and reply to interrogation signals.

MODE SELECT/TEST SWITCHES.

Four mode select/test switches (7, figure 1-58), located on the IFF control panel, are marked TEST, ON and OUT. The switches are labeled M-1, M-2, M-3/A and M-C from left to right to correspond to mode 1, mode 2, mode 3/A and mode C. The OUT position for each switch disables the transmitter-receiver for the mode selected. The ON position for each switch enables the transmitter-receiver to reply to interrogations for the mode selected. If more than one switch is placed to ON, the transmitter-receiver will reply to interrogations for all modes selected. The switches are spring-loaded to the ON position from the TEST position. IFF modes 1, 2, 3/A and C can be tested by holding the associated mode select/test switch in TEST while the RAD test/monitor switch is in MON or OUT and the MASTER control knob is in LOW or NORM. If the test lamp lights while the mode select/test switch is held in TEST, the related mode is functioning properly.

Note

When any of the four mode select/test switches is placed to ON and the RAD test/monitor switch is in MON, the test lamp may light, either continuously or periodically, due to external interrogations. This also validates operation of the mode selected.

RAD TEST/MONITOR SWITCH.

The three-position rad test/monitor switch (5, figure 1-58), located on the IFF control panel, is used for control of the radiation—test and monitor provisions. The switch has three positions marked RAD TEST, OUT and MON. The switch is spring-loaded from the RAD TEST to the OUT position. With the switch in MON, the test lamp will light when external or self test interrogations are received and answered. The RAD TEST position is used by maintenance personnel for a more thorough self test of the system. When the switch is placed to OUT, the radiation test and monitor circuits are inoperative.

CODE SELECTOR WHEELS.

Two sets of thumb actuated code selector wheels (8, figure 1-58), located on the IFF control panel, are provided to set mode 1 and mode 3/A codes. The set of wheels labeled mode 1 consists of two wheels which

allow selection of 32 different codes. The set of wheels labeled mode 3/A consists of four wheels which provide the capability of setting 4096 codes. Code digits on each wheel are read in windows recessed in the face of the panel.

MODE 4 CONTROL SWITCH.

The mode 4 control switch (9, figure 1-58), located on the IFF control panel, has two positions marked ON and OFF. Mode 4 operation is enabled by placing the switch to ON. Placing the switch to OFF disables mode 4 operation. The switch toggle must be pulled out in order to move the switch between the ON and OFF positions.

MODE 4 CODE CONTROL KNOB.

The mode 4 code control knob (1, figure 1-58), located on the IFF control panel, has four positions marked ZERO, A, B, and HOLD. The knob must be pulled out before it can be moved to the ZERO position, and is spring-loaded from HOLD to the A position. Positions A and B select the preset code for the present and succeeding code periods, respectively. Placing the knob to ZERO will zeroize both code settings if the master control knob (4, figure 1-58) is in any position except OFF. Both codes will be automatically zeroized when the IFF is turned off after landing. However, both code settings can be retained by momentarily holding the knob in the spring-loaded HOLD position prior to turning the IFF off. The HOLD function is operative only when the landing gear handle is in the DN position, and requires that system power remain on for at least 15 seconds after HOLD is selected to allow mechanical latching of the code settings.

MODE 4 MONITOR CONTROL SWITCH.

The mode 4 monitor control switch (10, figure 1-58), located on the IFF control panel, has three positions marked AUDIO, OUT, and LIGHT. In the AUDIO position, monitoring of mode 4 interrogations and replies is provided by an audio tone on the interphone and by illumination of the reply lamp on the IFF control panel. The audio tone is controlled by the UHF-2 mixer switch on each communications panel. Placing the switch to LIGHT switches out the audio tone and provides monitoring only by the reply lamp. In the OUT position, both the audio tone and the reply lamp are inoperative.

REPLY LAMP.

The reply lamp (2, figure 1-58), located on the IFF control panel, lights to indicate mode 4 replies. This lamp is operative only when the mode 4 monitor switch is in AUDIO or LIGHT.

IFF CAUTION LAMP.

The IFF caution lamp is located on the main caution lamp panel (figure 1-36). The lamp will light whenever an inoperative mode 4 capability is detected, provided the mode 4 computer is installed in the aircraft and the master control knob is not in the OFF position. Specific discrepancies monitored by the IFF caution lamp are:

- Mode 4 codes zeroized
- Failure of the system to reply to proper interrogation
- Automatic self-test function of the mode 4 computer reveals a faulty computer

The letters IFF are visible on the lamp when lighted.

TEST LAMP.

When the mode select/test switch is held in TEST, the test lamp (3, figure 1-58), located on the IFF control panel, will light to indicate the related mode is operable.

BOMBING NAVIGATION SYSTEM.

The bombing navigation (bomb nav) system is an aided-inertial digital system consisting of the inertial navigation set (INS), the control and display set (CDS) and the digital computer complex (DCC). The system operates in conjunction with other weapon system avionics equipment to provide the following functions: Refer to figure 1-60.

- Platform Alignment—A capability is provided for alignment of the inertial platform either on the ground, or while in flight using reference data from other avionics equipment. In addition, a capability is provided for automatically calibrating the inertial components in the INS.
- Self-Contained Navigation — The capability is provided for autonomous inertial navigation using only the INS and the CDS or inertial navigation wherein the DCC utilizes inputs from the INS to provide navigation functions.
- Augmented Navigation—Utilizing reference inputs from the astrocompass, the doppler radar and/or the CADC, the system provides the capability for five different aided-interial navigation modes. In the event of failure of any one or more of the systems which provide inputs to the system, the system regresses to the next navigation mode automatically in accordance with a mode hierarchy selection. (See figure 1-61.)
- Auxiliary Navigation—In the event of failure of the INS, the system utilizes reference inputs from the doppler radar and the CADC provide dead-reckoning navigational functions.
- Supporting Navigation Functions—The system maintains a continuous knowledge of the following navi-

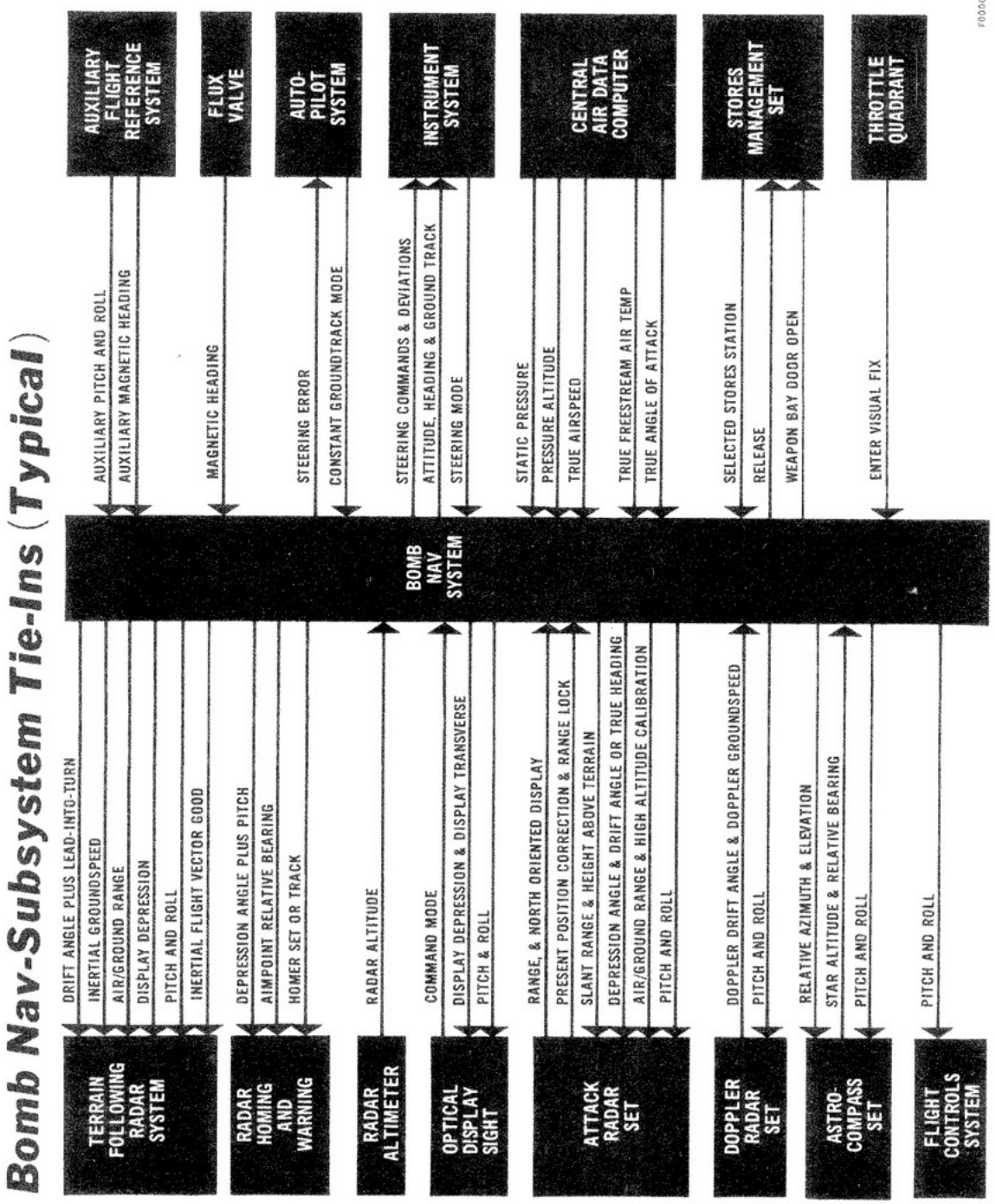
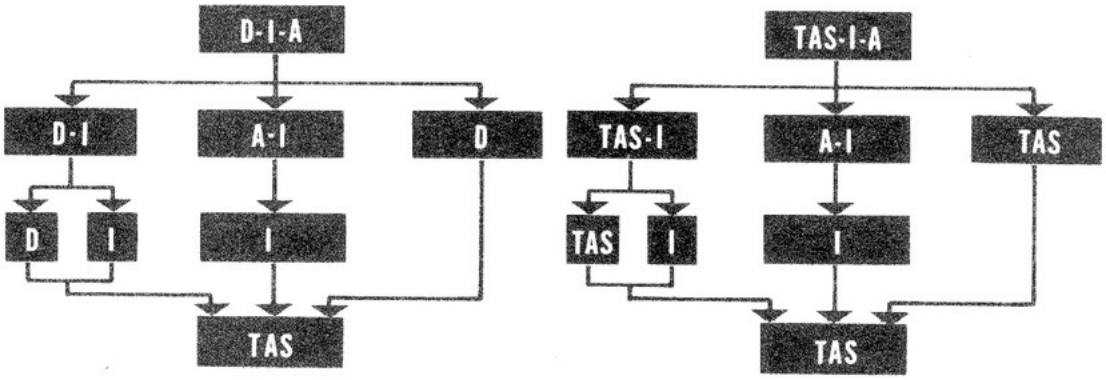

Bomb Nav-Subsystem Tie-Ins (Typical)
BOMB NAV SYSTEM
TERRAIN FOLLOWING RADAR SYSTEM
DRIFT ANGLE PLUS LEAD-INTO-TURN
INERTIAL GROUNDSPEED
AIR/GROUND RANGE
DISPLAY DEPRESSION
PITCH AND ROLL
INERTIAL FLIGHT VECTOR GOOD
RADAR HOMING AND WARNING
DEPRESSION ANGLE PLUS PITCH
AIMPOINT RELATIVE BEARING
HOMER SET OR TRACK
RADAR ALTIMETER
RADAR ALTITUDE
OPTICAL DISPLAY SIGHT
COMMAND MODE
DISPLAY DEPRESSION & DISPLAY TRANSVERSE
PITCH & ROLL
ATTACK RADAR SET
RANGE, & NORTH ORIENTED DISPLAY
PRESENT POSITION CORRECTION & RANGE LOCK
SLANT RANGE & HEIGHT ABOVE TERRAIN
DEPRESSION ANGLE & DRIFT ANGLE OR TRUE HEADING
AIR/GROUND RANGE & HIGH ALTITUDE CALIBRATION
PITCH AND ROLL
DOPPLER RADAR SET
DOPPLER DRIFT ANGLE & DOPPLER GROUNDSPEED
PITCH AND ROLL
ASTRO-COMPASS SET
RELATIVE AZIMUTH & ELEVATION
STAR ALTITUDE & RELATIVE BEARING
PITCH AND ROLL
FLIGHT CONTROLS SYSTEM
PITCH AND ROLL
AUXILIARY FLIGHT REFERENCE SYSTEM
AUXILIARY PITCH AND ROLL
AUXILIARY MAGNETIC HEADING
FLUX VALVE
MAGNETIC HEADING
AUTO-PILOT SYSTEM
STEERING ERROR
CONSTANT GROUNDTRACK MODE
INSTRUMENT SYSTEM
STEERING COMMANDS & DEVIATIONS
ATTITUDE, HEADING & GROUND TRACK
STEERING MODE
CENTRAL AIR DATA COMPUTER
STATIC PRESSURE
PRESSURE ALTITUDE
TRUE AIRSPEED
TRUE FREESTREAM AIR TEMP
TRUE ANGLE OF ATTACK
STORES MANAGEMENT SET
SELECTED STORES STATION
RELEASE
WEAPON BAY DOOR OPEN
THROTTLE QUADRANT
ENTER VISUAL FIX
F0500000-F020A

Figure 1-60.

Navigation Sensor Selection Hierarchy



1. The point at which the hierarchy is entered is determined by the state of the nav mode select pushbuttons on the NDU. For example D-I mode is selected if the D and I pushbuttons are depressed.
2. D-TAS or A modes are not exclusively selectable on the nav mode select pushbuttons.
3. If no pushbuttons are depressed, or the A only is depressed, TAS-I-A is selected.
4. If a unit fails, the computer selects the next lower mode as shown by the hierarchy chart. For example, if D-I is selected and D fails, the nav mode regresses to I.
5. During inflight alignments, if the selected reference velocity is lost, the system does not select a backup reference velocity.

F0000000-F024A

Figure 1-61.

gation-related parameters for display and/or transmission to the systems; present position latitude and longitude coordinates, roll and pitch attitude, groundspeed, ground track, true heading, magnetic heading, magnetic variation, drift angle, wind speed, wind direction, altitude above sea level, star altitude error and heading difference, and time and distance to destination.

- Mission Data Storage — A capability is provided for storage of the latitude, longitude, and elevation of 149 data points. These data points are assigned to two sequence tables; one table consists of 25 fixpoints, the other 75 destination/targets. Each destination or target may have two offset aimpoints associated with it. In addition, the capability is provided for the storage of the latitude and longitude of two fixpoints or targets of opportunity whose location is determined during a mission through the use of system sensors. A capability is also provided to store the ballistic constants of 15 different weapon types. Star data for 57 stars are stored for use in astro modes.
- Data Entry—A capability is provided for either automatic or manual entry of mission-related data into the computer complex; automatic data entry is accomplished through use of a flight-line tape reader.
- Mission Sequencing—Automatic sequence of destination/targets (including offset aimpoints) and fixpoints can be established in the order in which they will be encountered during a mission; these sequence points are then automatically selected at the appropriate time during the flight for steering, fixtaking, weapon delivery, etc. A further capability is provided to interrupt the automatic sequence at any time or to divorce the system entirely from the automatic sequence.
- Navigation Steering—The system provides steering error signals to the autopilot and HSI and bank steering commands to the ADI and ODS. In certain system operating modes, pitch steering commands are also provided to the ADI and ODS. Six different navigation steering modes are provided, including a self-contained airborne instrument low approach (AILA) mode for instrument landing on runways not equipped with ground-based letdown systems.

- Fixtaking—A capability is provided to effect present position corrections through the use of visual sighting techniques or with data provided by the attack radar or the astrocompass. Provisions are also made for determining and automatically storing the coordinates of fixpoints through the use of visual sighting techniques or the attack radar. In addition, the system operating in conjunction with the attack radar and the radar homing and warning system (RHAWS) provides a capability to determine and store the coordinates of ground-based radar emitters.
- Weapon Delivery—The system performs the computations and control functions required for air-to-ground weapon delivery against designated targets. The computations include automatic ballistics, steering error, time-to-go, weapon bay doors open signal, and weapon release point. Five modes are available: three synchronous and two visual. In addition, special provisions are included to supply navigation data to the SRAM system.
- System Self-Test/Built-in-Test—Comprehensive self-test and built-in-test features are provided by the system. The self-test (continuous, non-interfering test) and built-in test (manually, cockpit initiated tests) capabilities are provided to detect failures and identify the faulty subsystem and LRU to permit the operator to evaluate the effect of the failure on mission performance.

INERTIAL NAVIGATION SET.

The inertial navigation set (INS) is a fully automatic, self-contained inertial navigator which contains the following major components:

1. Navigation Computer Unit (NCU)—The NCU, located in the left-hand, forward equipment bay, is a self-contained digital navigation computer providing basic navigation computations when supplied inputs of velocity and heading. The NCU also provides the control and self-test functions for the inertial reference unit.
2. Inertial Reference Unit (IRU)—The IRU is located in the forward equipment bay. It contains a four gimbal, all attitude inertial platform and its associated electronics. The IRU provides basic outputs of velocity, heading, and aircraft attitude.
3. Battery Unit (BU)—The BU, located in the left-hand forward equipment bay, provides power for periods up to five seconds, to the NCU and IRU during aircraft electrical system transients. The BU also contains overtemperature sensing circuitry for the NCU, IRU, and itself.

CONTROL AND DISPLAY SET (CDS).

The CDS consists of two units, the navigation display unit (NDU) located on the right main instrument panel, and the computer control unit (CCU) located on the right console. The CDS provides the following functions:

- Control of primary electrical power to the avionics of the bomb nav system, doppler radar and astrocompass.
- Entering, changing, and recalling of digital computer data.
- Display of navigation, weapon delivery, and operational status data.
- Selection and control of functional and operational modes.
- Selection, display, and reset of built-in-test functions.
- Display of continuously monitored self-test results.

The CDS is capable of transmitting or receiving data from either general purpose digital computer of the DDC via the converter set (CS). The CDS also has the capability of transmitting or receiving data from the INS. The data channels to or from the INS are independent of the converter set (CS). The NDU is capable of receiving serial digital data over any one of three data channels at any one time. Two of the channels transmit data from the CS (one channel being redundant) and the third channel transmits data from the INS and is monitored by the NDU only during autonomous INS operation. The CCU transmits serial digital data over two data channels. One channel transmits data to the CS and the other channel transmits data to the INS for use during INS autonomous operation.

DIGITAL COMPUTER COMPLEX. (DCC)

The DCC is located in the forward electronics bay and is made up of the general navigation computer (GNC), the weapon delivery computer (WDC), and the converter set (CS). The GNC is programmed to handle all navigation functions, and has a weapon delivery backup capability. The WDC performs weapon delivery control, provides system self-test, and backup for navigation modes. The CS acts as a buffering logic and signal conversion device between the computers and other aircraft subsystems.

CONTROLS AND INDICATORS.

Function Select Knob.

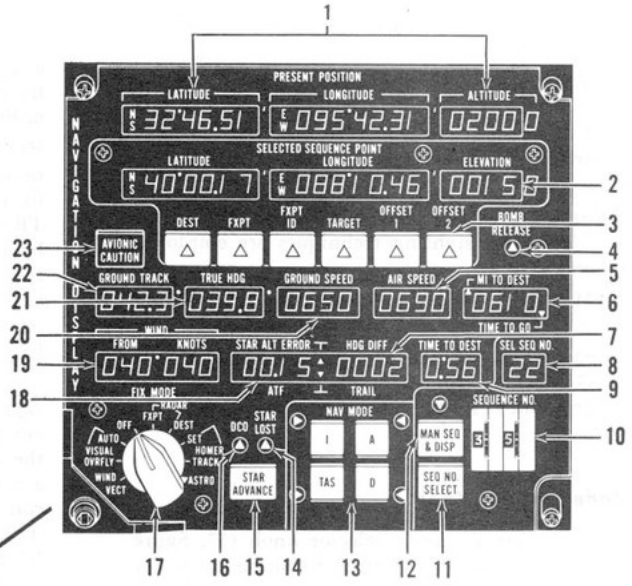
The seven-position function select knob (37, figure 1-62) controls power turn-on and system operating modes. The knob is pull-to-turn in the OFF and GND ALIGN positions. The knob markings and functions are as follows:

1. OFF—Inhibits power application to the astrocompass, doppler, and bomb nav system.
2. GND ALIGN—Enables any of the four ground alignment modes. Applies power to the bomb nav

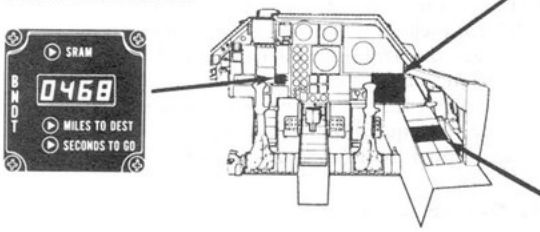
Bomb Nav Control Panels (Typical)

1. Present Position Displays.
2. Selected Sequence Point Displays.
3. Selected Sequence Point Pushbuttons (6).
4. Bomb Release Lamp.
5. Airspeed Display.
6. Miles To Destination/Time To Go Display.
7. Heading Difference/Trail Display.
8. Selected Sequence Number Display.
9. Time To Destination Display.
10. Sequence Number Set Wheels.
11. Sequence Number Select Pushbutton.
12. Manual Sequence and Display Pushbutton.
13. Nav Mode Select Pushbuttons.
14. Star Lost Lamp.
15. Star Advance Pushbutton.
16. Doppler Cutoff Lamp.
17. Fix Mode Selector Knob.
18. Star Altitude Error/Actual Time of Fall Display.
19. Wind Display.
20. Ground Speed Display.
21. True Heading Display.
22. Ground Track Display.
23. Avionic Caution Lamp.

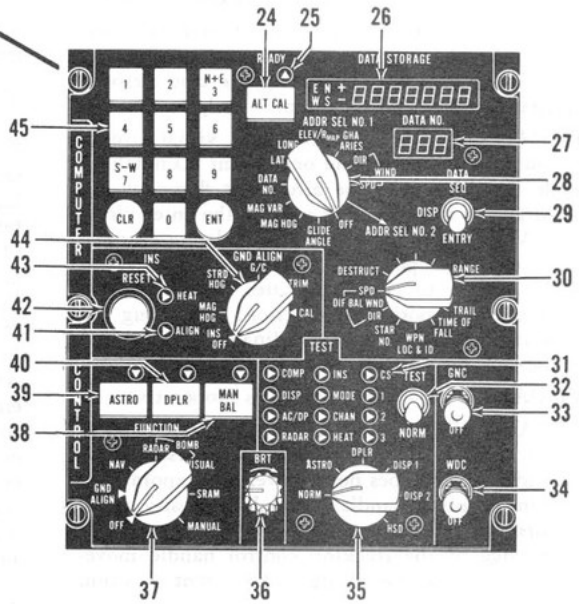
NAVIGATION DISPLAY UNIT PANEL



BOMB NAV DISTANCE TIME INDICATOR



COMPUTER CONTROL UNIT PANEL



24. Altitude Calibration Pushbutton.
25. Altitude Calibration Ready Lamp.
26. Data Storage Display.
27. Data Number Display.
28. Address Selector Knob (No. 1)
29. Data Switch.
30. Address Selector Knob (No. 2).
31. Test Indicator Lamps.
32. Test Switch.
33. General Navigation Computer Power Switch.
34. Weapons Delivery Computer Power Switch.
35. Test Selector Knob
36. Indicator Lighting Control Knob.
37. Function Select Knob
38. Manual Ballistics Pushbutton.
39. Astrocompass Pushbutton.
40. Doppler Radar Pushbutton.
41. INS Align Indicator Lamp.
42. INS Reset Button.
43. INS Heat Indicator Lamp.
44. INS Ground Align Knob.
45. Date Entry Pushbuttons.

F7310000-F013 8

Figure 1-62.

system, including the INS, provided the ground align knob is in any position other than INS OFF. Enables power to be applied to the astrocompass and the doppler radar provided the ASTRO and DOPP pushbuttons are depressed.

3. NAV—Enables all navigation modes and inflight alignment modes.
4. BOMB RADAR—Enables weapon delivery modes in which the attack radar is utilized to detect a target or its offset aimpoints. In this mode steering is provided to direct the aircraft to the weapon release point.
5. BOMB VISUAL—Enables weapon delivery modes in which visual sighting techniques are employed to detect a target.
6. SRAM—Same as NAV position except that all doppler outputs are isolated from the DCC if the INS is selected and operating.
7. MANUAL—Interrupts automatic sequencing and permits manual selection for navigation and bombing.

Fix Mode Selector Knob.

The nine-position fix mode selector knob (17, figure 1-62) is used to select the desired fixtaking mode for either present position correction or storage of the coordinates of fixpoints. The knob markings and functions are as follows:

1. OFF—Terminates and disables all fixtaking modes.
2. WIND VECT—Enables the wind vector fix mode. This mode is a back-up mode and is used to refine stored wind data for use during the true airspeed navigation mode and the doppler navigation mode.
3. VISUAL OVERFLY—Enables the visual overfly present position correction or fixpoint ID fixtaking mode. In this mode, the aircraft is flown over a point and at the instant of overfly, the enter visual fix (EVF) button on the throttle is depressed causing the present position counters to be up dated (100 percent) to the coordinates of the selected point (present position correction) or causing the present position counter coordinates to be stored in the fixpoint ID table.
4. VISUAL AUTO—This position is inoperative.
5. RADAR FXPT—In this mode, the attack radar is used to acquire the fixpoint. If the intersection of the radar cursors does not lie over the fixpoint, the tracking control handle is used to position the cursors on the point. The computer maintains knowledge of the tracking control handle movements to refine knowledge of present position coordinates.
6. RADAR DEST—This mode is identical to the radar fixpoint mode except that destination coordinates are applicable rather than fixpoint coordinates.
7. HOMER SET—The HOMER SET position on the fix mode selector knob causes a circular cursor to appear on the RHAW scope. The tracking control handle is used to position this cursor over the radar station return (the attack radar cursors are positioned over the same point). The radar station location is computed from its azimuth and elevation angles and height above terrain. Positioning the fix mode selector knob to OFF will store the coordinates of the cursor position.
8. HOMER TRACK—The cursors are initially positioned using the HOMER SET position. When the fix mode selector knob is rotated to the HOMER TRACK position, the circular cursor is replaced by a bearing line cursor on the RHAW scope. Further tracking control handle corrections along this bearing line will refine the locations of the station in range. Positioning the fix mode selector knob to OFF will store the coordinates of the cursor position.
9. ASTRO—Enables the astrocompass, two-star present position correction fixtaking mode. Assuming the astrocompass is already locked on and tracking a star, engagement of this mode causes the astrocompass to break lock and search for the next star within the field of view in the computer table. When track on the second star is achieved, the DCC utilizes the azimuth and elevation data from both stars to refine present position. After processing these data, this procedure is continuously repeated using subsequent stars. The knob must be pulled out to turn to or from the ASTRO position.

INS Ground Align Knob.

The ground align knob is used to select the desired INS alignment or calibration mode and to apply power to the INS. The knob is inactive unless the function select knob is in a position other than OFF. The knob must be pulled out to turn to or from the INS OFF, TRIM and CAL positions. The normal method of ground alignment is by gyrocompassing. This method requires adequate time allowances for gyro warm-up and azimuth alignment by sensing of earth rate. In this mode, the inertial platform initially is caged to zero roll and pitch for approximately 30 seconds. At the completion of the cage phase, the platform enters a partial alignment phase. The align lamp will come on steady 90 seconds (prior to T.O. 1F-111-1180) or 34 seconds (after T.O. 1F-111-1180) after power has been applied to the INS. When coarse level alignment is achieved, the I advisory lamp will come on and, after T.O. 1F-111-1180, will flash at a once per second rate, and the primary attitude caution lamp will go out. At this time the navigation function can be entered with degraded accuracy. When it is determined that the platform level is within prescribed limits, the partial align phase is completed and a primary attitude good signal is issued. The issuance of this signal will cause the primary attitude caution lamp to go out if

primary reference is selected, and will cause the inertial navigation (I) mode indicator lamp to come on. The heading caution lamp will go out provided the DCC heading good signal has been issued. This indicates to the operator that the INS is capable of functioning as an attitude reference and can be used to supply roll and pitch to the flight reference system at this time. The final alignment phase is signified by the flashing of the align lamp. Alternate methods of ground alignment of the inertial system are (1) alignment to stored heading, (2) alignment of stored magnetic variation, and (3) alignment by two-axis trim procedures. The alignment to stored heading is a rapid alignment of the bomb nav system utilizing information stored in the INS computer from previous operation. This mode is accurate only when a gyrocompass alignment and less than 30 minutes of navigation were accomplished prior to system turn on, and the aircraft has not been moved since previous system turn off. Alignment to stored magnetic variation is a rapid alignment of the bomb nav system utilizing information stored in the INS computer from previous operation. This mode assumes that the system was gyrocompass aligned, was in the navigation mode for less than 30 minutes, present position has not changed significantly (600 feet), that magnetic variation has remained essentially constant, and that aircraft heading has not changed by more than 2.5 degrees. These requirements allow the aircraft to be moved provided the aircraft is returned to approximately the same heading and magnetic variation has not been changed. Two axis trim provides the capability for trimming both the north and east uncompensated gyro drift rates during a ground alignment mode. This alignment consists of two gyrocompass alignments during which the platform azimuth is slewed 90 degrees under DCC control. An opportunity to enter the navigation mode with gyrocompass accuracy exists during the 20 seconds of flashing align lamp indication after completion of the first gyrocompass. If the navigation mode is not commanded during this 20-second interval, the DCC will cause the align lamp to cease flashing and will slew the stable element 90 degrees. Subsequent to this slewing, a second gyrocompass alignment is initiated. This alignment is complete when the align lamp comes on and starts flashing. In-flight alignment is accomplished by rotating the INS ground align knob from OFF to any position except TRIM or CAL when the function select knob is in any position except OFF or GND ALIGN.

Note

If the INS has been ground aligned and is operating in-flight and it is desired to initiate an in-flight alignment, the INS ground align knob must be positioned to OFF and back on as noted above.

When an in-flight alignment is initialized by the DCC, the align lamp will come on steady and the primary attitude and heading caution lamps will be on. When the platform has been leveled to within prescribed limits, the primary attitude and heading (provided the DCC heading good signal is present) caution lamps will go out and the inertial navigation mode lamp will light. Completion of in-flight alignment will be indicated when the align lamp goes out. In-flight alignment requires external position and/or velocity reference data. Consequently, the doppler and astro systems should be energized as desired, and the desired navigation mode pushbuttons must be selected. As soon as possible a position fix should be accomplished. If a navigation mode pushbutton is not selected a TAS-I in-flight alignment is commanded. If the inertial navigation mode pushbutton only is selected, numerous position fixes will be required to accomplish the alignment. The knob markings and functions are as follows:

1. INS OFF—Removes power from the INS.
2. MAG HDG—Initiates the stored magnetic variation rapid platform alignment mode.

Note

For an accurate stored magnetic variation alignment, the INS must have undergone an accurate gyrocompass alignment, the aircraft, if moved, must be returned to within 2.5 degrees of the original heading, and magnetic variation must have remained essentially unchanged.

3. STRD HDG—Initiates the stored heading, rapid platform alignment mode.

Note

For an accurate stored heading alignment the INS must have undergone an accurate gyrocompass alignment and the aircraft must not have been moved since the system was turned off.

4. G/C—Initiates the normal, ground gyrocompass alignment mode.
5. TRIM—Initiates the two-axis trim calibration mode. In this mode, the north and east uncompensated platform drift rates are automatically trimmed. The mode may be selected before, during or after any of the ground alignment modes. The first phase of this mode causes trimming of the north uncompensated drift rate; at the completion of this phase, the align lamp commences flashing. If at the end of 20 seconds a navigate mode is not

selected, the platform is automatically rotated 90 degrees and the east uncompensated drift rate is trimmed; the align lamp is on steady during this phase. Two-axis trim complete is signified by the flashing of the align lamp upon completion of the second phase.

Note

Erroneous primary heading information will result if the function select knob is placed in NAV while the platform is being slewed to a new heading.

6. CAL—This is a maintenance function only and requires that a special calibration program be loaded into the computer complex memory.

Address Selector Knobs.

The two twelve-position address selector knobs (28, and 30, figure 1-62) are labeled ADDR SEL NO. 1 and ADDR SEL NO. 2. The knobs are used to select automatic addressing for data words being entered into or recalled from the computers. Address selector knob No. 1 is marked: GLIDE ANGLE, MAG HDG, MAG VAR, DATA NO, LAT, LONG, ELEV (aircraft 63 ⏵ 52) or ELEV/R MAP (aircraft 53 ⏵) GHA ARIES, WIND DIR, WIND SPD, → (transfer), and OFF. The MAG HDG position is for display only. When this position is selected and the data switch is positioned to DISP, magnetic heading is displayed in the data storage display. When address selector knob No. 1 is in the → (transfer) position, the control of automatic addressing is transferred to address selector knob No. 2. When address selector knob No. 1 is in the OFF position, all address functions are disconnected. The OFF position is used with the INS reset button to clear the NCU error trap. Address selector knob No. 2 is labeled DESTRUCT, RANGE, TRAIL, TIME OF FALL, WPN LOC & ID, STAR NO., DIF BAL WIND DIR and DIF BAL WND SPD. Two additional positions on the address select knob No. 2 are operative as follows: the unmarked 11 o'clock position is used to notify and verify that the DG mode of AFRS operation is being used; the unmarked 12 o'clock position is used to zero doppler correction terms (boresight and scale factor) input to the DCC. The operator has the capability to manually enter all parameters except MAG HDG using the address selector knobs.

Test Selector Knob.

The five-position test selector knob (35, figure 1-62) is included in the test portion of the system. The knob is used to select builtin test f-unctions. The knob markings and functions are as follows:

1. NORM—Continuously monitored self-test modes.
2. ASTRO—Initiates the astrocompass built-in self-test mode.
3. DOPPLER—Selects the doppler radar built-in self-test mode.
4. DISP 1—Applies power to all alpha and numerical tests and display lamp segments to verify their operation.
5. DISP 2—Cycles all test and display lamps through all patterns to verify lamps and logic.

Note

The test switch must be in the TEST position to initiate ASTRO, DOPPLER, DISP 1 and DISP 2 tests.

Indicator Lighting Control Knob.

The indicator lighting control knob (36, figure 1-62) is labeled BRT. Turning the knob clockwise increases brightness on both NDU and CCU.

CAUTION

Use only the degree of brightness required to see the lights. Maximum brightness (full CW) greatly decreases the life of the elements and requires replacement of entire NDU and CCU panels.

Nav Mode Select Pushbuttons.

The four nav mode select pushbuttons (13, figure 1-62) labeled NAV MODE. An indicator lamp located by each button lights to provide an indication of the navigation mode currently in use. The buttons are marked and function as follows:

I — Selects the inertial mode of navigation and must be used for inflight alignment.

A — Selects the astrocompass mode of navigation.

When activated, the A indicator lamp will be lighted when the following conditions are met:

- The astrocompass system is locked onto a star.
- Astro and inertial data, if selected, are in reasonable agreement.

D — Selects the doppler radar mode of navigation.

When the pushbutton is activated, the doppler radar is placed in the operate (radiating) mode. The doppler radar pushbutton (40, figure 1-62) must be activated prior to selection of the "D" doppler mode.

Note

When the doppler operate lamp lights, the following conditions are indicated:

- D—doppler pushbutton has been selected, groundspeed is greater than 100 KIAS, drift is less than 25 degrees and the system is reliable.
- Doppler and inertial, if selected, are in reasonable agreements.

TAS—Selects the CADC augmented automatic navigation, or air data dead reckoning navigation modes. The indicator lamp will light if the DCC determines that the CADC is operating and is in use.

Data Entry Pushbuttons.

The pushbuttons (45, figure 1-62) are momentary-contact, push-on, release-off switches which are used to manually enter data into, or recall data from the DCC or the INS. Data entered by the keyboard is displayed on the data storage or data no. counters. Each push-button contains an integral lamp to light the switch lens engraving. Pushbuttons 0 thru 9 are for digit inputs. Pushbuttons 3 and 7 are also used to enter + or — signs or latitude, longitude coordinate symbols (N, E, W, S). The button marked CLR (clear) blanks the data storage counter. The ENT (enter) key causes information to be transferred from the data storage counter to the computers. The following data can be automatically entered by use of a flight line tape reader:

- Star data table
- Wind profile table
- Weapon location and ID table
- Blast radii table
- Fixpoint sequence table
- Destination/target sequence table
- Data points table
- Glide angle
- Weapon range, trail, and time of fall
- Differential ballistic wind
- Magnetic variation

Manual data entry of the following mission parameters can be accomplished when desired:

- Data points
- Greenwich hour angle
- Wind direction and speed
- Glide angle (for AILA)
- Magnetic variation
- Differential ballistic wind
- Weapon location and identification
- Weapon range
- Weapon trail
- Weapon time of fall
- Star number

The data entry pushbuttons may be used to notify the DCC that the DG mode of AFRS operation is in use, and to zero doppler correction terms (boresight and scale factor).

Altitude Calibration Pushbutton.

The altitude calibration pushbutton (24, figure 1-62), marked ALT CAL, is a push-on, push-off switch which initiates the pressure altitude calibration mode when pressed into an active state. The green lamp will light when the ALT CAL pushbutton is pressed into an active state, and will go out when the pushbutton is pressed into an inactive state.

Astro Compass Pushbutton.

The astro pushbutton (39, figure 1-62), marked ASTRO, is a push-on, push-off switch which is used to apply power to the astrocompass. An indicator lamp mounted adjacent to the pushbutton will light when the astro pushbutton is in an active mode. For a more complete description, refer to T.O. 1F-111(B)A-1-2.

Doppler Radar Pushbutton.

The doppler pushbutton (40, figure 1-62), marked DPLR, is a push-on, push-off switch which is used to put the doppler radar in standby. An indicator lamp will light when the doppler pushbutton is in an active mode. For a more complete description, refer to T.O. 1F-111(B)A-1-2.

Manual Ballistics Pushbutton.

The manual ballistics pushbutton (38, figure 1-62) is a push-on, push-off switch marked MAN BAL. When activated, the computer will use the hand entered values of trail/range and time-of-fall for weapons delivery. An indicator lamp will light when the MAN BAL button is in an active mode.

Star Advance Pushbutton.

The star advance pushbutton (15, figure 1-62), marked STAR ADVANCE, is a momentary-on switch. Depressing the pushbutton provides a signal to the DCC to designate the point selected by the radar cursors, when performing a wind vector or heading correction.

Note

The selection of the star to be tracked by the astrotracker is accomplished by manually entering the desired star number via the CCU.

A lamp labeled STAR LOST located by the pushbutton will light to provide an indication that the star selected for reference is not being tracked.

Sequence Number Select Pushbutton.

The sequence number select pushbutton (11, figure 1-62), marked SEQ NO. SELECT, enables the operator to make changes in the route point sequencing.

Manual Sequence and Display Pushbutton.

The manual sequence and display pushbutton (12, figure 1-62), marked MAN SEQ & DISP, is a push-on, push-off type. A lamp adjacent to the pushbutton will light when the button is depressed. Depressing the button enables the display of any set of route point coordinates, selected by the sequence number setwheels, and interrupts automatic sequencing.

Enter Visual Fix Pushbutton.

The enter visual fix pushbutton (1, figure 1-5), located on the left throttle of the left set, is marked EVF. The pushbutton, when depressed, is used for visual fix-taking and visual fixpoint ID.

Selected Sequence Point (SSP) Pushbuttons.

The six SSP pushbuttons (3, figure 1-62) are located on the NDU. The DEST, FXPT, TARGET, OFFSET 1 and OFFSET 2 pushbuttons define which type of sequence point coordinates are to be displayed, and also during manual data entry define for the DCC what type of sequence point coordinates are being entered. The FXPT ID pushbutton is used in conjunction with the fix mode selector knob to enable the fixpoint ID mode. In the fixpoint ID mode, the radar cursors can be driven to overlay the point of interest whose coordinates are unknown. The computer determines the coordinates of the unknown point from knowledge of the starting point of the cursors and the tracking handle corrections. The coordinates of the instantaneous position of the cursors are displayed on the SSP displays in this mode, and the selected sequence number display indicates the current fixpoint ID sequence number. When the operator terminates the mode, the coordinates of the cursors are stored in the fixpoint ID table in the computer. The coordinates of the fixpoint ID point can be subsequently recalled by depressing the fixpoint ID pushbutton, selecting the appropriate fixpoint ID sequence number, and depressing the manual sequence and display pushbutton. Only one button can be depressed at a time.

Sequence Number Set Wheels.

The two sequence number set wheels (10, figure 1-62), located on the NDU, are labeled SEQUENCE NO. The setwheels are marked 0 thru 9. Together they generate a four-bit code to establish the automatic route point sequence, select stored route point coordinates for display, and advance or retard the route point sequence.

INS Reset Button.

The INS reset button (42, figure 1-62), located on the CCU, is labeled RESET. When depressed, the button transmits a signal to the INS, commanding it to replace present position coordinates with those previously entered via the CCU panel keyboard pushbuttons. The reset button is operable only if the function select knob is in the GND ALIGN position, except during autonomous nav modes. The INS reset button is also used to clear the NCU error trap when the address selector knob No. 1 is in the OFF position.

Data Switch.

The three-position data switch (29, figure 1-62) is labeled DATA. The SEQ position is selected for entry of data pertaining to mission sequence. The DISP position is selected for recall and display of information for specific memory locations. The ENTRY position is used to enter data associated or not associated with a data number.

General Navigation Computer Power Switch.

This two-position switch (33, figure 1-62) is marked GNC and OFF. The switch is lever locked in the GNC position and supplies electrical power to the general navigation computer.

Weapon Delivery Computer Power Switch.

This two-position switch (34, figure 1-62) is marked WDC and OFF. It is lever locked in the WDC position and supplies electrical power to the weapon delivery computer.

Note

In the event of a failure of one computer and the operator desires to attempt to regain the two computer operation, it is necessary to manually switch both computers off and back to on. The computers must be turned on within 4 seconds of each other in this procedure. If only one computer and the INS are powered up simultaneously, the computer will halt.

Test Switch.

The two-position test switch (32, figure 1-62) is marked TEST and NORM. The TEST position enables a

particular built-in-test function to be performed. The NORM position enables continuous monitoring and reporting self-test routines.

Avionic Caution Lamp (ACL).

The ACL (23, figure 1-62), located on the NDU is labeled AVIONIC CAUTION. The ACL will light in conjunction with a fault legend display on the CCU. The ACL lamp and fault legend will remain lighted until the fault is acknowledged by the operator. Acknowledgement is accomplished by depressing the ACL once after which the ACL and fault legend will go out and remain out if no other fault exists. If another fault exists, the fault legend for that fault and the ACL will light until the new fault is acknowledged by the operator. Once acknowledged, the fault legend or ACL will not light again due to reoccurrence of the same fault. Reoccurrence of faults will be reported only after a recall has been performed. Recall is accomplished when the operator depresses the ACL when no fault legends nor the ACL is lighted. This action clears any record of previous faults which have been reported. If no faults are reported at this time, the ACL and fault legends remain out until a fault occurs. If a fault is reported at this time or if any fault occurs thereafter, the ACL and fault legend will light until the fault is acknowledged.

Test Indicator Lamps.

The 12 test indicator lamps (31, figure 1-62) are labeled TEST. The display is used to indicate the results of operator-initiated built-in tests, and of continuously monitored self-tests. As many as three lamps associated with the test message may light while a test is in progress, and will go out if the test is passed. If the test is failed, the ACL will light and the TEST display(s) will remain lighted on that test which failed. The test failure indications may be cleared by depressing the ACL. The TEST displays are marked as follows:

COMP—Computer complex.

INS—Inertial navigation set.

CS—Converter set.

AC/DP—Astrocompass/doppler radar.

1, 2, 3—Used to designate combinations of failures. For example, COMP 1 indicates a GNC failure, CS2 indicates that area 2 of the CS has failed, INS 3 indicates that the INS battery unit has failed, etc.

MODE—Incompatible mode; crew selection or failure of a subsystem indicates an invalid mode state.

CHAN—Serial digital communications channel failure.

HEAT—INS or DCC overtemperature.

DISP—Control and display set.

RADAR—Doppler or attack radar.

Note

Five different operations or sets of conditions which can cause an AC/DP CHAN test legend and ACL are:

1. The aircraft makes heading changes exceeding astrocompass maximum yaw rates while the astrocompass is in the search mode (i.e., turning during taxi maneuvers).
2. The aircraft makes attitude changes exceeding 24 degrees up, 16 degrees down in pitch or 20 degrees in roll while the astrocompass is in slew mode.
3. Star tracking is attempted with marginal sky conditions.
4. The attitude reference is switched from the IRU to the AFRS.
5. GHA of Aries is updated while the astrocompass is not tracking a star.

Except for item 4, the AC/DP CHAN test legend can be cleared by manually advancing the system to another star number, then depressing the ACL pushbutton. Item 4 requires cycling of astro power and then depressing the pushbutton. In each case, the star altitude error and heading difference displays will blank until the fault is cleared.

INS Align Indicator Lamp.

The INS align indicator lamp (41, figure 1-62) marked ALIGN will come on when a signal is received from the INS, indicating that the platform is aligning and will flash when the platform is fully aligned during ground alignment, or go out during inflight alignment.

INS Heat Indicator Lamp.

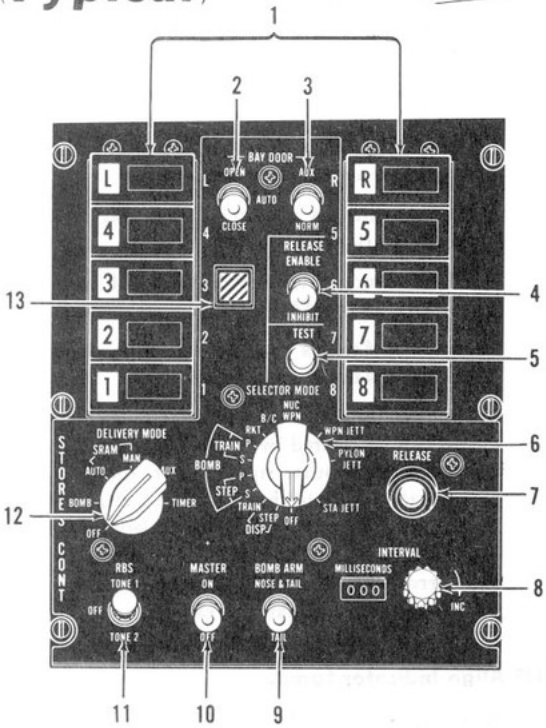
The INS heat indicator lamp (43, figure 1-62) marked HEAT will light when a signal is received from the INS, indicating that the IRU components have not reached operating temperature.

Bomb Release Lamps.

Two bomb release lamps, one on the upper warning and caution panel (figure 1-36) and the other on the NDU (4, figure 1-62), are labeled BOMB RELEASE. The bomb release lamp on the upper warning and caution panel will light when a bomb release signal is received from the SMS. The bomb release lamp on the NDU will light for 3.0 seconds when a bomb release signal is received from the CS.

Stores Control Panel (Typical)





1. Stores Station Selector Switches (10).
2. Weapons Bay Door Control Switch.
3. Weapons Bay Door Auxiliary Switch.
4. Release Enable Switch.
5. Stores Present Test Button.
6. Weapons Mode Selector Knob.
7. Weapon Release Button.
8. Intervalometer Set Knob.
9. Bomb Arming Selector Switch.
10. Master Power Switch.
11. RBS Tone Switch.
12. Delivery Mode Selector Knob.
13. Weapons Bay Door Position Indicator.

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Figure 1-63.

WARNING

To prevent inadvertent release with any abnormal station select lamp indication, or when a bomb release lamp remains lighted, care must be taken when positioning bombing system controls. Proceed as follows:

EWO — Position the release enable switch to INHIBIT and the bay door control switch to CLOSE, if applicable. Complete the necessary actions for subsequent weapon release and reposition the bay door control switch, if applicable, to OPEN and the release enable switch to ENABLE immediately prior to release.

TRAINING (with stores loaded) — Position the release enable switch to INHIBIT, perform the appropriate "Abort Procedures", Section IV, and do not subsequently reposition any of the controls or switches included in the "Abort Procedures" check list.

Primary Attitude Caution Lamp.

The primary attitude caution lamp (figure 1-36) will light when the attitude good signal from the INS is invalid or auxiliary reference is selected. When either of these situations occurs, the primary heading lamp will light and the source of attitude and heading signals will be switched to the AFRS. The letters PRI ATTITUDE are visible on the lamp when lighted.

Primary Heading Caution Lamp.

The primary attitude caution lamp (figure 1-36) will light when:

(1) the primary attitude signal from the INS is invalid or;
(2) the auxiliary reference is selected, or
(3) the primary heading is invalid.

When situations (1) and (2) occur, the source of the primary attitude and heading signals will be switched to the AFRS. In situation (3), primary attitude only is provided to the flight instruments. The letters PRI HDG will be visible on the lamp when lighted.

Present Position Displays.

The present position displays (1, figure 1-62) are labeled PRESENT POSITION. The displays are marked and provide the following indications.

1. LATITUDE—Display consists of one alpha numeric lamp followed by six numeric lamps. It displays latitude in degrees and minutes, to the nearest 0.01 of a minute.

2. LONGITUDE—Display consists of one alpha-numeric lamp followed by seven numeric lamps. It displays longitude in degrees and minutes, to the nearest 0.01 of a minute.

3. ALTITUDE—Display consists of four numeric lamps with a zero (0) engraved after the last lamp. It displays aircraft mean sea level pressure altitude in feet to the nearest 10 feet. For altitudes below sea level, the left counter displays a minus (—) sign.

Note

All latitude and longitude readouts can be in error by 0.01 of a minute of arc; for example N30 degrees 00.00 minutes may appear as N29 degrees 59.99 minutes.

Selected Sequence Point Displays.

The three selected sequence point (SSP) displays (2, figure 1-62) are labeled SELECTED SEQUENCE POINT. The displays are marked and provide the following indications:

1. LATITUDE—Display consists of one alpha-numeric lamp followed by six numeric lamps. It displays latitude of the sequence point in use as determined by the SSP pushbuttons and the automatic route point sequence. The display reads in degrees and minutes, to the nearest 0.01 of a minute.

2. LONGITUDE—Display consists of one alpha-numeric display lamp followed by seven numeric lamps. It displays longitude of the sequence point in use as determined by the SSP pushbuttons and the automatic route point sequence. The display reads in degrees and minutes, to the nearest 0.01 of a minute.

3. ELEVATION—Display consists of four numeric lamps, with a zero (0) engraved after the last lamp. It displays mean sea level elevation of the sequence point in use as selected by the SSP pushbuttons and the automatic route point sequence. It displays the elevation in feet to the nearest 10 feet. For altitudes below sea level, the left display displays a minus (—) sign.

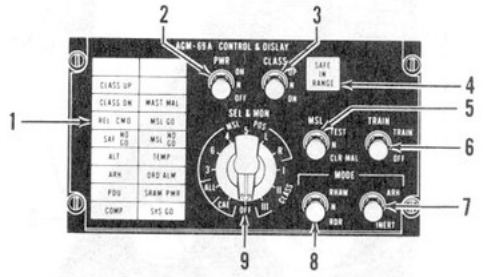
Ground Track Display.

The ground track display (22, figure 1-62) is marked GROUND TRACK and displays aircraft true ground track, in degrees to the nearest 0.1 degree.

True Heading Display.

The true heading display (21, figure 1-62) is marked TRUE HDG, and displays aircraft true heading, in degrees, to the nearest 0.1 degree.

SRAM Control Panel (Typical)



1. Status and Malfunction Indicator Lamps.
2. Power Switch.
3. Class Switch.
4. Safe-In-Range Indicator Lamp.
5. Missile Switch.
6. Train Switch.
7. Anti-Radiation Homer/Inertial Mode Switch.
8. RHAW/Radar Mode Switch.
9. Select and Monitor Control Knob.

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Figure 1-64.

Ground Speed Display.

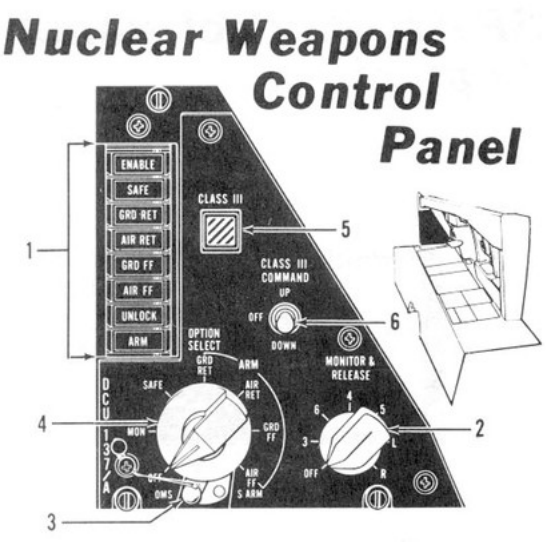
The groundspeed display (20, figure 1-62) is marked GROUND SPEED and displays aircraft groundspeed, in knots to the nearest knot.

Airspeed Display.

The airspeed display (5, figure 1-62) is marked AIR SPEED, and displays aircraft true airspeed in knots to the nearest knot.

Miles-to-Destination/Time-to-Go Display.

The miles-to-destination/time-to-go display (6, figure 62) is marked MI TO DEST/TIME TO GO. A lamp is mounted adjacent to each function, and when lighted, indicates that parameter is being displayed. In the NAV mode, the display displays distance to destination in nautical miles. In the RADAR BOMB modes, it displays time-to-go to weapon release in seconds. In VISUAL BOMB, the display will be blanked until either weapon release has been depressed, at which time it will display the time remaining until actual weapon release.



1. Nuclear Weapons Monitor Lamps.
2. Nuclear Weapons Monitor and Release Knob.
3. Nuclear Weapons Arm Knob Lock Lever.
4. Nuclear Weapons Arm Knob.
5. Class III Status Indicator.
6. Class III Command Override Switch.

F0000000-F098

Figure 1-65.

Wind Displays.

The two wind displays (19, figure 1-62) are labeled WIND FROM/KNOTS and continuously display the computed value of wind direction and magnitude. The display marked FROM displays the direction from which the wind is blowing, in degrees. The display marked KNOTS displays the wind speed in knots.

Star Altitude Error/Actual Time of Fall Display.

The star altitude error/actual time of fall display (18, figure 1-62) is marked STAR ALT ERROR and ATF. A lamp is mounted adjacent to each function and when lighted indicates that parameter is being displayed. In the NAV mode, the display displays star altitude error, in arc-minutes. The alpha-numeric lamp displays either an A (Away) or T (toward). In the RADAR or VISUAL BOMB modes, it displays the actual time of fall in seconds.

Heading Difference/Trail Display.

The heading difference/trail display (7, figure 1-62) is marked HDG DIFF and TRAIL. An indicator lamp is mounted adjacent to each function, and when lighted indicates that parameter is being displayed. In the NAV mode, it displays the heading difference, in arc-minutes. The first alpha-numeric lamp reads either a plus (+) or minus (—). In the RADAR or VISUAL BOMB mode, it displays trail to the nearest 10 feet.

Time-to-Destination Display.

The time-to-destination display (9, figure 1-62) is marked TIME TO DESTINATION. It consists of three numeric lamps and displays time to destination, in hours and minutes when in the navigation mode, and miles to target, to the nearest mile, when in bomb mode.

Selected Sequence Number Display.

The selected sequence number display (8, figure 1-62) is marked SEL SEQ NO. It consists of two numeric lamps and displays the sequence point in use as determined by the SSP pushbutton.

Data Storage Display.

The data storage display (26, figure 1-62) is marked DATA STORAGE. It consists of one alpha-numeric lamp and seven numeric lamps to display data being entered into, or recalled from, the DCC or the INS. Alpha characters (N, S, E, W, +, or —) are displayed on the left lamp and decimal characters 0 thru 9 appear on the other lamps from right to left, character-by-character.

Data Number Display.

The data number display (27, figure 1-62) is marked DATA NO. It consists of 3 numeric lamps to display the data number associated with the data being entered into or recalled from the DCC or the INS. A single data number for each set of coordinates (lat, long, elev) is entered from the keyboard switches prior to entering the data. The lamps are blanked when address selector No. 1 knob is rotated out of the LAT, LONG, or ELEV positions unless the data switch is in the DISP position.

Star Lost Lamp.

The star lost lamp (14, figure 1-62) is a green indicator lamp which, when lighted, indicates that the astrocompass is not tracking a star. In addition, the STAR LOST lamp will blink off and on to indicate that a heading fix is in progress.

Doppler Cutoff Lamp.

The doppler cutoff lamp (16, figure 1-62) is a green indicator lamp which, when lighted, indicates that the doppler is not tracking a good doppler return signal.

BOMB NAV SYSTEM OPERATION.

For bomb nav system operating procedures, refer to Section IV.

ARMAMENT SYSTEM.

The armament capability of the aircraft includes the delivery of conventional and nuclear weapons in various configurations. The stores are carried in the weapons bay and on eight wing pylons. Four of the wing pylons (3, 4, 5 and 6) pivot to remain streamlined with the longitudinal axis of the aircraft as the wing sweep is varied. The various equipment, controls and indications for the bombing or missile system, are covered in detail in the applicable delivery manual. T.O. 1F-111(B)A-25-1 for nuclear delivery, 1F-111(B)A-34-1-1 for conventional delivery and T.O. 1F-111(B)A-30-1 for missile delivery. Refer to Section V for a list of authorized stores.

EXTERNAL STORES GROUND JETTISON SWITCH.

The external stores ground jettison switch (8, figure 1-16) has two positions marked ARM and OFF and is guarded in the OFF position. In the ARM position, the main landing gear safety (squat) switches are bypassed to allow emergency jettison of external stores during ground operation. In the OFF position, external stores cannot be jettisoned while the weight of the aircraft is on the main landing gear.

BOMBING TIMER.

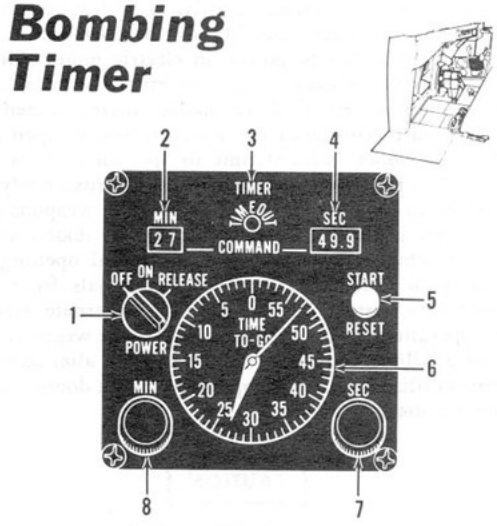
A bomb timer (figure 1-66) provides a backup for timed navigation. The weapon release function of the bombing timer has been deactivated. The timer has the capability of time-to-go for 30 minutes and 59.9 seconds in increments of 0.1 seconds.

Power Selector Knob. The power selector knob (1, figure 1-66), labeled POWER, has three positions marked OFF, ON and RELEASE. When rotated to the ON position, power is turned on to the timer circuit. When the time cycle is started and drives to zero, the time out lamp will come on. The RELEASE position is deactivated.

Minute Command Counter. The minute command counter (2, figure 1-66), labeled MIN, is driven by the minute command knob (8, figure 1-66), and has a range of 0 to 30 minutes in 1 minute increments. The counter is set to the value in minutes for desired countdown.

Seconds Command Counter. The seconds command counter (4, figure 1-66), labeled SEC, is driven by the seconds command knob (7, figure 1-66), and has a range of 0 to 59.9 seconds in 0.1 second increments. The counter is set to the value in seconds for desired countdown.

Start/Reset Pushbutton. When depressed, the start/reset pushbutton (5, figure 1-66), labeled START RESET, causes the time-to-go clock to drive to the



1. Power Selector Knob.
2. Minute Command Counter.
3. Time Out Lamp.
4. Seconds Command Counter.
5. Start/Reset Pushbutton.
6. Timer-To-Go Clock.
7. Seconds Command Knob.
8. Minutes Command Knob.

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Figure 1-66.

settings on the minute and seconds command counters. Depressing the pushbutton a second time starts the timing cycle.

Time Out Lamp. The time out lamp (3, figure 1-66), labeled TIME OUT, will come on in all cases when the time-to-go display clock has driven to zero.

Time Out Indicator Lamp. The time out indicator lamp (15, figure 1-6) will light when the time-to-go clock has driven to zero. When lighted, the letters TIME OUT are visible.

Time-To-Go Clock. The time-to-go clock (6, figure 1-66), labeled TIME-TO-GO, is a graduated scale with a range of 0 to 60 in 1 digit increments. The large pointer will display minutes and the small hand will display seconds. The clock is set and started by the start/reset pushbutton.

WEAPONS BAY DOORS.

The weapons bay doors enclose the weapons bay area located between the nose and main landing gear. The doors are constructed in left and right clam shell halves which fold outward as they are opened. Normal and alternate systems are provided to operate the doors.

The normal system utilizes hydraulic power from the utility hydraulic system to drive a hydraulic motor. The alternate system uses 115 volt ac power from the right main ac bus to power an electric motor. Either motor drives a gear reduction mechanism, which through a series of drive shafts interconnected to hinges on the inside of the weapons bay, to open and close the doors. Normal time to open or close is 2½ seconds. The alternate system takes approximately 30 seconds to open or close the doors. The weapons bay doors are controlled by the weapons bay door control switch which allows selection of manual opening or closing or automatic operation on signals from the bomb nav system. Selection of the alternate system for operating the doors is provided by the weapons bay door auxiliary control switch. When the alternate system is utilized, automatic operation of the doors should not be used.

CAUTION

To prevent damage to the door actuator motors during alternate operation there must be a 10-second interval between opening and closing the doors and no more than 3 complete door opening and closing cycles within a 15 minute period.

Weapons Bay Door Auxiliary Switch.

The weapons bay door auxiliary switch (3, figure 1-63), labeled BAY DOOR, has two positions marked AUX and NORM. With the switch in the NORM position, the weapons bay door operates on hydraulic pressure from the utility hydraulic system. Placing the switch to AUX provides electrical power to operate the weapons bay door.

Weapons Bay Door Control Switch.

The weapons bay door control switch (2, figure 1-63) has three positions marked OPEN, AUTO and CLOSE. The OPEN and CLOSE positions of the switch provide manual control of door operation by either normal hydraulic power or auxiliary electrical power depending on the position of the weapons bay door auxiliary control switch. The AUTO position functions in conjunction with the weapons bay door auxiliary switch. With the weapons bay door auxiliary switch in the NORM position, the doors will automatically open approximately 10 seconds from release or 10,000 feet actual range from target (whichever is sooner), remain open until approximately one second after weapon release and then close. The doors will automatically operate with the weapons bay door auxiliary switch in the AUX position, however since the doors take 30 seconds to operate in this configuration, an automatic weapon release cannot be obtained.

WARNING

If the position of the bay door control switch is not in agreement with the position of the weapons bay doors, the doors may actuate to the commanded position when hydraulic and/or electrical power is applied to the aircraft.

Weapons Bay Door Position Indicator.

The weapons bay door position indicator, located on the stores control panel, is a flip-flop type indicator, which displays a cross hatched indication for intermediate weapons bay door positions or electrical power interruptions, or OPEN or CLOSE for those positions.

WEAPONS BAY DOOR SAFETY PANEL.

A weapons bay door safety panel is located just aft of the left clam-shell half of the weapons bay doors. (See figure 1-17.) This panel houses the weapons bay door interlock switch, a safety lockpin, and the manual drive adapter. The interlock switch has two positions marked Normal and Safe. The normal position enables hydraulic and/or electrical (AUX) operations of the weapons bay doors. The ground safety lockpin is used as either a mechanical ground safety lock when inserted into the manual drive adapter, or can be used to open the weapons bay doors when hydraulic or electrical power are not available. Mechanical operation of the weapon bay doors via the ground safety lockpin is independent of the positioning of the door interlock switch.

CAUTION

Do not turn the manual drive adapter more than 271 turns from fully closed when manually opening the doors. Damage to the door drive mechanism may result.

After T.O. 1F-111-948, the ground safety access has been redesigned to prevent loss of the access door during flight. A slot in this access door permits installation and removal of the ground safety lockpin (figure 1-17) with the door closed. The modified door is held closed by screws. When not in use, the ground safety lockpin for the modified access door is stowed in the cockpit. The safety interlock switch must be positioned to NORMAL to permit either the modified or unmodified door to be closed.

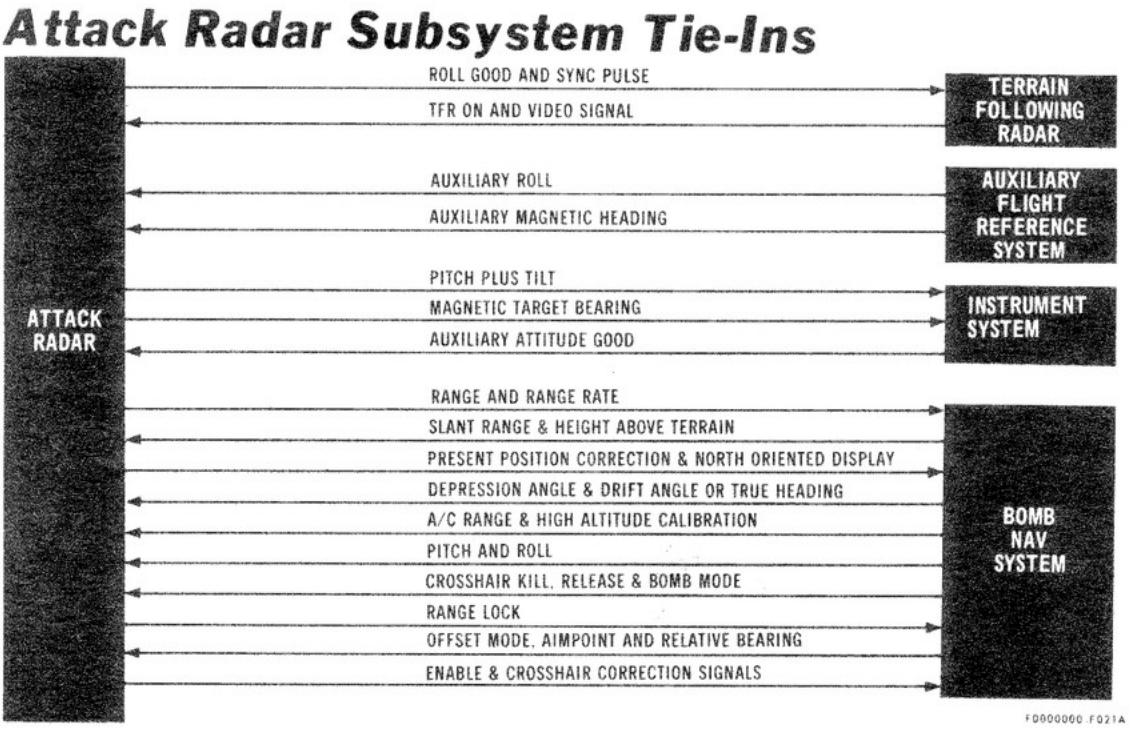


Figure 1-67.

CAUTION

After T.O. 1F-111-948, the ground safety lock-pin shall be removed prior to engine start to prevent ingestion during engine start.

ATTACK RADAR (AN/APQ-114).

The attack radar provides all weather operation for aid to navigation, bombing and air refueling rendezvous capability. See figure 1-67 for attack radar system tie-ins with other weapon system avionic equipment. The set performs the following functions:

- Ground mapping.
- Navigation fixtaking.
- Air-to-air search and tracking.
- Air-to-air rendezvous.

The system may be operated in three ground modes, an air mode and a beacon mode. The beacon mode can be selected in either the air or ground modes. Basic components of the system consist of an antenna, an antenna roll unit and antenna control, located in the radome, and a modulator-receiver-transmitter (MRT) and synchronizer, located in the forward electronic bay. The radar scope and the controls, including the tracking control handle, are located at the right crew station. A recording camera is provided to take radar scope photographs. The antenna, whose travel limits are ±45 degrees in roll and ±30 degrees in pitch, is automatically stabilized in pitch and roll by signals from the bomb nav system. In the event of a bomb nav system failure, the antenna is stabilized in roll only by signals from the auxiliary flight reference system (AFRS). The antenna can be caged in the event of pitch or roll stabilization failure to be in alignment with the aircraft longitudinal and lateral axis by means of the antenna cage pushbutton; however, the antenna will continue to sector in azimuth and tilt can be adjusted. For location of the antenna, see figure 1-69. The MRT operates in the J frequency band of 16 to 16.55 GHZ and has the capability of automatic frequency control (AFC-1 and AFC-2) along with manual frequency control (MFC). In the AFC-1 position, the receiver operates

Photo Data Recording

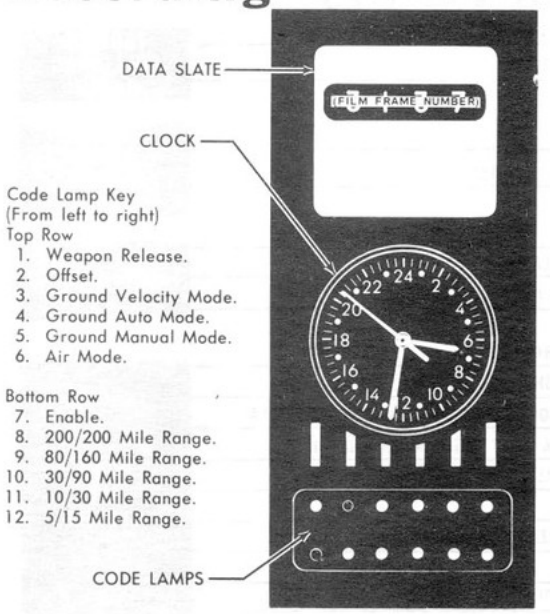


Figure 1-68.

in the automatic frequency control mode and the transmitter sweeps through the frequency band with random reversal. This provides a measure of immunity to many types of jamming and improves stability of the scope display. When operating in AFC-2 the receiver operates in the automatic frequency control mode and the transmitter is manually tuned. In the event of a malfunction of AFC-1 or AFC-2 the MFC position is used. In this position the transmitter operates on mid-band fixed frequency and the receiver is manually tunable by adjusting the knob over the MFC range tracking. The beacon mode of operation enables the attack radar to interrogate and receive replies from J-band transponders. This capability enables long range detection and identification of aircraft for rendezvous and in-flight refueling. The scope panel contains the radar scope, recording camera, and the necessary operating and tuning controls for the scope and camera. The recording camera is mounted behind the radar scope. A small window in the side of the cathode ray tube allows the camera to take exposures of the back of the radar scope. The image on the scope is reversed by optics so that the film exposure will represent the scope presentation as seen by the operator. A film exposure is taken automatically at weapon release on a signal from the bomb nav system or manually when desired. Automatic scope photography is provided to take scope pictures at rates of either 1 exposure each 40 scans to 1 exposure each 4 scans depending on the function select knob position. A lamp on the radar scope panel will blink each time an exposure is taken. A film magazine in the face of the radar scope panel provides a minimum of 1200 exposures of 35 millimeter film. A readout window on the magazine shows percentage of film remaining. The magazine is installed or removed by means of a handle recessed in the front of the magazine. Simultaneous film exposure of a clock, data slate and 12 code lamps is made with each scope exposure to identify each frame of the film as shown in figure 1-68. The clock and slate provide time of exposure and operator's name, date, mission, etc. The tracking control handle is used in conjunction with the bomb nav system to position the azimuth and range cursors for fixtaking, bombing, and antenna tilt control in the air mode. Self test features incorporated into the system are used for preflight, maintenance malfunction analysis, and troubleshooting. The system operates on 115 volt ac power from the left main ac bus and 28 volt dc power from the main dc bus.

ATTACK RADAR GROUND MODES.

Three ground modes of operation are provided for ground mapping, fixtaking, and weapon delivery. The ground modes are: ground manual (GND MAN), ground auto (GND AUTO) and ground velocity (GND VEL). In all ground modes, the antenna scans with a fan beam ±45 degrees centered about the longitudinal axis. In the ground auto mode, the display is a ground-track stabilized, or north oriented stabilized one-radius-offset PPI display. The bomb nav system supplies the radar with continuously computed signals which represent slant range and relative bearing of a point which may be a target, aiming point, or fixpoint, depending upon the mode of operation of the bomb nav system. These signals enable the radar to generate synchronous range and azimuth cursors (cross-hairs) which, for an errorless input, will appear exactly on the radar image of the desired point and will maintain track with the point as it moves relative to the aircraft. The azimuth cursor is positioned within ±45 degrees in azimuth by the relative bearing signals supplied from the bomb nav system. The range cursor is positioned from approximately zero nautical miles to 124 nautical miles in response to the computed slant-range-to-target signal supplied by the bomb nav system. Fore or aft movement of the tracking handle, when the tracking handle enable switch is also depressed, will cause correction signals to be sent to the bomb nav system to adjust the computed along-track target position; these signals will represent north-south corrections for the north-oriented display. Similarly, when the enable switch is depressed a left or right motion of the tracking handle will cause correction signals to be sent to the bomb nav system to correct the computed cross-track target position; these

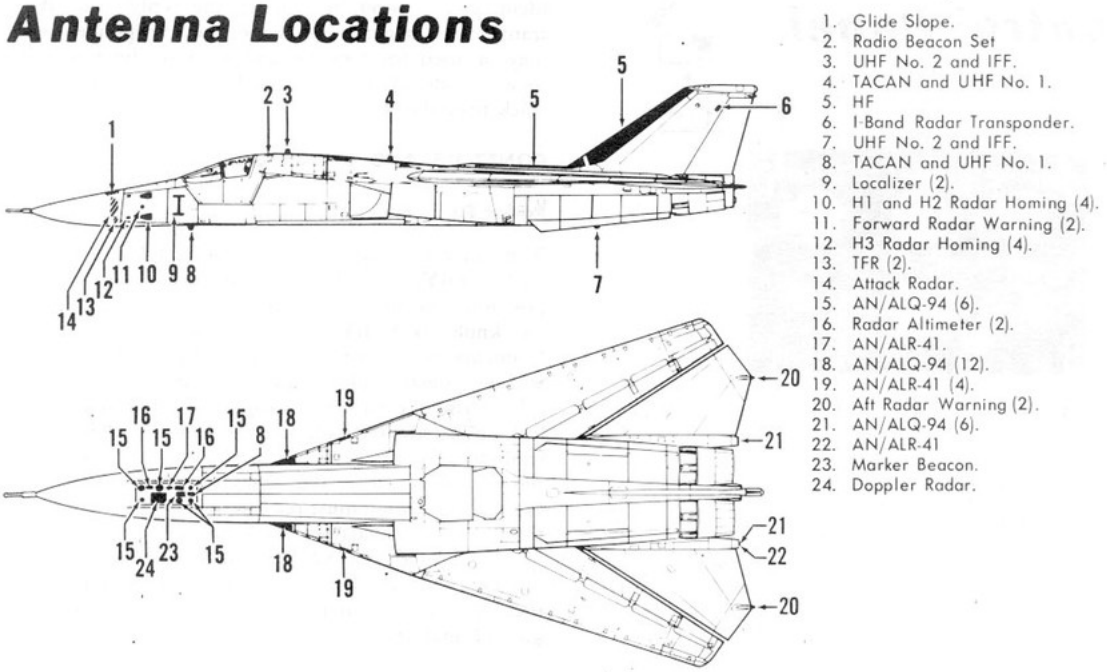


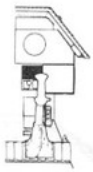
Figure 1-69.

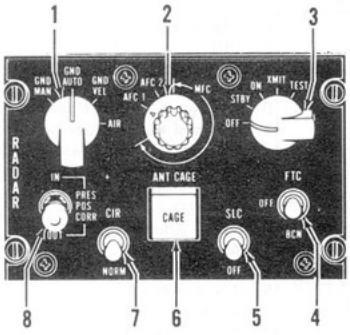
signals represent east-west corrections for north-oriented display. The present position correction switch is used in conjunction with the bomb nav system to permit present position correction. Present position is corrected when the bomb nav system is in the navigation mode by placing the switch to IN and moving the tracking handle so that the intersection of the radar cross-hairs coincides with the target. If the switch is placed to OUT, when operating in the navigation mode no movement of the radar cursors is possible by use of tracking handle movement signals. When the bomb nav system is in the bomb mode, the cursors are automatically placed on target. In this mode, if the switch is set to OUT, corrections are made by the tracking handle to the aircraft-target relationship and no correction to present position is made. If the switch is placed to IN while in bomb mode, movement of the tracking handle results in corrections to both the aircraft target relationship and the computed value of present position. In the ground velocity mode, the range and azimuth cursors will appear in the center of the display. The slant range and relative bearing to the target destination or offset are supplied by the bomb nav system and are used to drive the display so that the target or destination point will remain centered under the crosshairs in the center of the scope display. In the ground manual mode, the display is a ground-track stabilized, one radius-off-set PPI display and can operate independently of the bomb nav system. In the ground manual mode with the antenna caged, the scope display is not ground-track stabilized. In this situation, zero degrees on the radar scope represents the aircraft heading. The range and azimuth cursors are manually positioned by use of the tracking handle. The range of the cursor can be read on the radar scope panel range counter.

ATTACK RADAR AIR MODE.

The air mode is provided for long range contact, identification, and tracking of aircraft for rendezvous and air refueling. In the air mode, a pencil beam overlapping two-bar box scan is provided. In wide scan, the

Attack Radar Control Panel





1. Mode Selector Knob.
2. Frequency Control Knob.
3. Radar Function Knob.
4. Fast Time Constant/Beacon Switch.
5. Side Lobe Cancellation Switch.
6. Antenna Cage Pushbutton Indicator Lamp.
7. Antenna Polarization Switch.
8. Present Position Correction Switch.

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Figure 1-70.

antenna scans an area ±45 degrees about the longitudinal axis of the aircraft and can be positioned ±30 degrees in elevation. In narrow scan, the antenna scans an area ±10 degrees about the azimuth cursor. Wide scan is utilized for search and initial tanker position detection. Once the tanker is located, narrow scan can be selected to lock on and automatically track the tanker and provide the pilot with steering commands for the rendezvous. In order to track a target, the azimuth cursor must be positioned to within 10 degrees of the target. Actuation of the sector switch provides the narrow scan and activates the automatic range search and target acquisition circuitry. In the air mode, the IF gain is a fixed value unless the attack radar is in the beacon mode.

BEACON MODE.

The beacon mode of operation enables the attack radar to interrogate and receive replies from J-band transponders. The beacon mode of operation can be selected in both the air-to-ground and air-to-air modes. Once the beacon mode is selected, all normal targets and ground clutter will disappear from the radar scope, since the transmitter and receiver are tuned to different frequencies. Targets, if present, will be transponder replies to the radar pulse interrogation. Once a transponder reply is noted on the scope, the operator can identify the target by reading the reply code. As a transponder target closes range, the transponder reply may be used for lock on and tracking by switching to air mode of operation and employing the target track procedures.

CONTROLS AND INDICATORS.

Radar Function Knob.

The radar function knob (3, figure 1-70) is marked OFF, STBY, ON, XMIT, and TEST. In the OFF position, the entire system is de-energized. Placing the knob to STBY supplies power to all system filaments for warm-up and energizes a 40-second warmup delay and a 5-minute transmitter warmup delay. Also the antenna is caged in pitch and stowed full up in tilt and full left in azimuth. Placing the knob to ON energizes the entire system, except for the transmitter, after the 40-second warmup delay has expired. The knob must remain in the ON position for the 5-minute mandatory warmup period after which it may be placed to XMIT to place the entire system in operation. The TEST position allows self test of the system for malfunction troubleshooting and ground maintenance.

Note

After the radar function knob is turned from STBY to either ON, XMIT, or TEST, the mode selector knob should be placed briefly in the GND AUTO or GND VEL position. This action provides a master reset. Master reset is an internally generated signal that places the attack radar computer circuitry in a known state of valid operation. Necessity for reset may be indicated by an abnormal radar display such as distorted display, cursor hangup, or erratic range readout.

Attack Radar Mode Selector Knob.

The attack radar mode selector knob (1, figure 1-70) is marked GND MAN (ground manual), GND AUTO (ground auto), GND VEL (ground velocity) and AIR. In the GND MAN position, the range and azimuth cursors are positioned with the tracking control handle and antenna tilt is positioned with the antenna tilt control knob independently of the navigation computer. In the GND AUTO position, the cursors are automatically positioned and the antenna tilt normally is automatically positioned by signals from the bomb nav system. The tracking control handle is used to correct the bomb nav system present position, and the tilt control knob is used to refine the scope display. GND VEL is the same as GND AUTO, except the scope display is a ground velocity stabilized

magnified picture, and the intersection of the cursors remains in the center of the scope display. In the AIR position, the antenna is programmed for a box scan which can be raised or lowered in elevation with the tracking control handle during search, detection, and acquisition of an air target. Once a target has been acquired by the radar, tracking is automatic in range, azimuth, and elevation.

Note

When rotating the attack radar mode selector knob, allow one second for each mode. Rapid cycling of the mode selector knob through two or more modes may produce conditions requiring an additional master reset action.

Attack Radar Frequency Control Knob.

The frequency control knob (2, figure 1-70) is marked AFC 1 (automatic frequency control), AFC 2, and MFC (manual frequency control). In AFC 1, the receiver operates in the automatic frequency control mode, and the transmitter operates in a frequency agility mode in which the transmitter sweeps through the frequency band with random reversal. The changing frequency and rapid scanning rate in this position provides immunity to many types of jamming and improves stability of the PPI display. In the AFC 2, the receiver operates in the automatic frequency control mode, and the transmitter frequency is manually changed using the transmitter tuning control knob. The MFC position of the knob is variable over a range between the 12 and 8 o'clock position. In this position the transmitter operates in a mid-band fixed frequency, and the receiver is manually tunable by adjusting the knob over the MFC range.

Attack Radar Fast Time Constant/Beacon Switch.

The fast time constant beacon switch (4, figure 1-70) is marked FTC (fast time constant), OFF, and BCN (beacon). The FTC position provides leading edge discrimination of all returns. This discrimination highlights the leading edge of targets, blanks out the trailing edge, and provides a much clearer assessment of the relative position or pattern of the complex. The FTC position is used to minimize the effects of jamming in any mode of operation.

Note

FTC should be used in the air mode of operation when obvious jamming signals are present on the attack radar display or to reduce the effect of ground clutter during target detection.

The BCN position places the system in beacon mode of operation and can be selected in both air and ground modes. This mode enables the attack radar to interrogate and receive replies from J-band transponders. Once the beacon mode is selected, all targets except the transponder replies will disappear from the scope display.

Attack Radar Side Lobe Cancellation Switch.

The side lobe cancellation (SLC) switch (5, figure 1-70) is marked SLC and OFF. The SLC position cancels the energy received from the side lobes of the radar beam to reduce ground clutter. The SLC position may be selected in any mode of operation; however, it is most effective when operating in the air mode at low altitudes.

Antenna Polarization Switch.

The antenna polarization switch (7, figure 70) is marked CIR (circular) and NORM. In NORM, the antenna polarization is horizontal when operating in the ground mode and vertical when operating in the air mode. The CIR position changes antenna polarization to circular when operating in either ground or air modes. The CIR position may be used to reduce rain clutter interference on the scope. The polarization is always horizontal for beacon mode of operation.

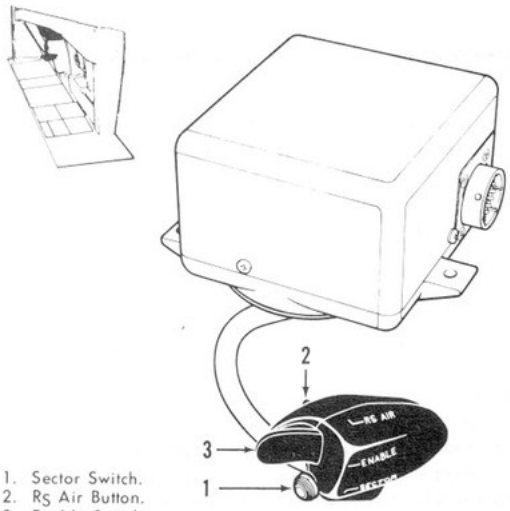
Present Position Correction Switch.

The present position correction switch (8, figure 1-70) is marked IN and OUT. The switch is used in the ground auto and ground velocity modes of operation when correcting the bomb nav system present position counters with the tracking handle.

Attack Radar Tracking Control Handle.

The tracking control handle (figure 1-71) is mounted on a pivot pedestal. The enable switch must be depressed and held to activate the handle. When operating in the air mode (with the range search button depressed) or in the ground manual mode, fore or aft displacement of the handle will slew the range cursor out or in respectively. When operating in the normal ground auto or ground velocity mode, fore or aft displacement will slew the cursor intersection in a direction parallel to aircraft ground track. For north-oriented display operation, this will move the cursor intersection point north or south. Moving the handle to a fore or aft position in the air mode without depressing the range search button will adjust antenna elevation down and up respectively. When operating in the air or ground manual mode, left or right displacement of the handle will slew the azimuth cursor left or right. When operating in the normal ground auto or ground velocity mode, left or right displacement will slew the cursor intersection in a direction perpendicular to aircraft ground track and will move

Attack Radar Tracking Control Handle



1. Sector Switch.
2. R_S Air Button.
3. Enable Switch.

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Figure 1-71.

the cursor intersection point west or east for north-oriented display operation. Slewing speed is proportional to the amount of handle deflection.

Attack Radar Range Search (R_S) Button.

The red range search button (2, figure 1-71) is labeled R_S AIR. The button is used in the air mode of operation and when depressed causes the radar set to break lock. With the sector switch (1, figure 1-71) in the aft position (wide antenna scan), depressing the R_s button permits the range cursor to be slewed rapidly to any desired position on the sweep (to a maximum range of 124 nautical miles) by moving the tracking handle fore or aft. When the button is released, the range cursor will remain stationary after slewing. With the sector switch in the forward position (narrow scan), depressing the R_s button overrides range lock (if established) and permits the range cursor to be slewed rapidly to any desired position on the sweep (to a maximum range of 124 nautical miles) by moving the tracking handle fore and aft. When the button is released, range searching will resume from the point at which the range cursor was positioned by slewing.

Attack Radar Sector Switch.

The sector switch (1, figure 1-71), labeled SECTOR, is a two-position, thumb actuated toggle switch. The switch is used in either the ground or air modes of operation to change the sector of antenna sweep. In the aft position (wide scan) antenna sweep is ± 45 degrees about the longitudinal axis of the aircraft. In the forward position (narrow scan) antenna sweep is ± 10 degrees about the azimuth cursor. In addition, the forward position causes the initiation of automatic acquisition and tracking in the air mode.

Photo Mode Selector Switch.

The photo mode selector switch (7, figure 1-72) is marked AUTO, MAN (manual) and OFF. The switch is spring-loaded from MAN to AUTO. A film exposure is taken when the switch is held to the MAN position. With the switch in AUTO, film exposures will be taken automatically at a rate of one photo per each 40 antenna scans or one photo per each four antenna scans when radar bomb mode is selected on the computer control unit. Regardless of switch position, a film exposure is taken automatically at weapon release on signal from the bomb nav system.

Note

Malfunctions of the attack radar system (ARS), which cause attack radar scope presentation abnormalities, should be recorded through use of the MAN position of the photo mode selector switch.

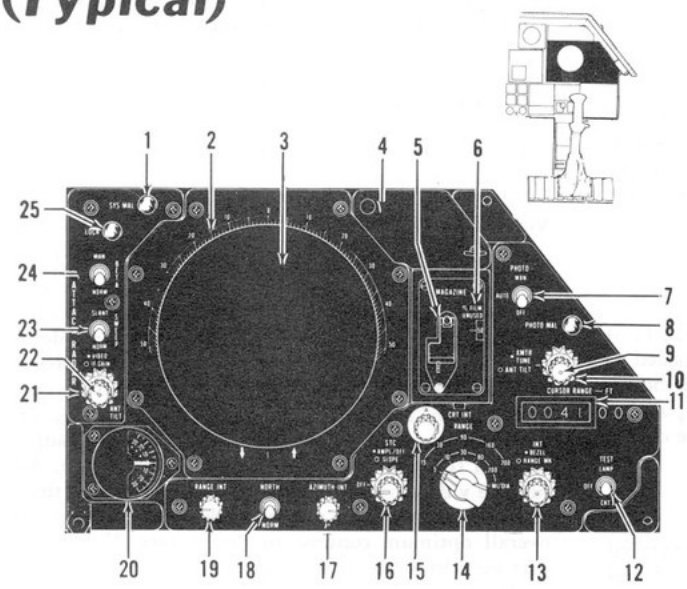
Intermediate Frequency Gain Knob.

The intermediate frequency gain knob (21, figure 1-72) is labeled IF GAIN and permits adjustment of receiver gain when operating in the ground modes. This control determines maximum usable sensitivity of the receiver circuits and functions primarily as a brightness control. To set IF gain, the video gain control must be advanced to a position to allow noise video to be painted on the scope even though the optimum setting of the video gain control must be made with the function selector knob in XMIT. This control is also operative in air mode if BCN is selected.

Antenna Tilt Control Knob.

The antenna tilt control knob (10, figure 1-72), labeled ANT TILT, provides a means of manually adjusting antenna tilt position when operating in the ground modes. In the ground mode, the knob is the only means of adjusting antenna tilt. In the ground auto and ground velocity modes, antenna tilt is automatically positioned by signals from the bomb nav system, and the knob is used to refine this position. The knob has a

Attack Radar Scope Panel (Typical)



1. System Malfunction Lamp.
2. Azimuth Bezel.
3. Radar Scope.
4. Ground Adjustment Access Door.
5. Film Magazine Removal Handle.
6. Unused Film Indicator.
7. Photo Mode Selector Switch.
8. Photo Malfunction Indicator Lamp.
9. Transmitter Tuning Control Knob.
10. Antenna Tilt Control Knob.
11. Cursor Range Counter.
12. Test Switch.
13. Bezel/Range Marks Intensity Control Knobs.
14. Range Selector Knob.
15. Scope Intensity Control Knob.
16. Sensitivity Time Control Knobs.
17. Azimuth Cursor Intensity Control Knob.
18. North Orientation Selector Switch.
19. Range Cursor Intensity Control Knob.
20. Antenna Tilt Indicator.
21. Intermediate Frequency Gain Control Knob.
22. Video Adjustment Knob.
23. Sweep Control Switch.
24. Beta Switch.
25. Range Lock Indicator Lamp.

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Figure 1-72.

detent corresponding to zero antenna tilt pisition for reference when the radar is operating in ground manual mode, or operating in ground auto or ground velocity mode with the beta switch in the MAN position. The detent indicates zero tilt correction when the radar is operating in ground auto or ground velocity mode with the beta switch in the NORM position. Rotating the knob fully counterclockwise tilts the antenna up to +30 degrees and rotating the knob fully clockwise tilts the antenna down to −30 degrees. However, with the terrain following radar (TFR) operating, the antenna can only be physically adjusted to −8 degrees (pitch plus tilt) to prevent interference with the TFR. Antenna position is indicated on the antenna tilt indicator (20, figure 1-72), located on the attack radar scope panel. The indicator is graduated in 5-degree increments from zero to ±30 degrees. The knob has no control over antenna tilt when operating in the air mode.

Attack Radar Scope Intensity Control Knob.

The attack radar scope intensity control knob (15, figure 1-72), labeled CRT INT, provides an adjustment of scope baseline intensity from zero to full brightness. To set the CRT intensity properly, operate the radar with the function selector in the ON position with the IF GAIN and VIDEO controls fully CCW. Advance the knob clockwise (increasing brightness) to a point where the sweep is visible.

Bezel/Range Marks Intensity Control Knobs.

Two coaxial knobs (13, figure 1-72) provide an adjustment of bezel and range marks intensity. The knobs are labeled INT. The outer knob is marked BEZEL, and the inner knob is marked RANGE MK. Turning either knob clockwise increases intensity from zero to full brightness. The range mark intensity should normally be adjusted at a slightly different level from the range cursor intensity to prevent confusion.

Range and Azimuth Cursor Intensity Control Knobs.

Two cursor intensity control knobs (17, 19, figure 1-72) provide an adjustment of range and azimuth cursor intensity. The knobs are labeled individually as RANGE INT and AZIMUTH INT. Turning either

knob clockwise increases intensity of the respective cursor from zero to full brightness. Adjustment of the range and azimuth cursor intensity to the lowest usable value will allow more precise placement of the cursors over the target.

North Orientation Selector Switch.

The north orientation selector switch (18, figure 1-72) is marked NORTH and NORM. In the NORTH position, the top of the radar scope is true north oriented. The north stabilized display is employed only in the ground auto and ground velocity modes of operation. The NORM position permits normal scope display presentation.

Attack Radar Test Switch.

The test switch (12, figure 1-70), labeled TEST, is marked LAMP, CKT and OFF and is spring-loaded to OFF. The LAMP position is used to check all the indicator lamps on the scope panel. The CKT position is used to make maintenance ground confidence checks to the system.

Attack Radar Sweep Control Switch.

The sweep control switch (23, figure 1-72), marked SLANT and NORM, is used in the ground modes of operation to provide a map-like presentation in the NORM position and a linear presentation in the SLANT position. In the AIR mode, the switch is inoperative and the sweep is automatically set to slant.

Attack Radar Beta Switch.

The beta switch (24, figure 1-72), marked MAN (manual) and NORM, functions in the ground auto and ground velocity modes to select automatic sighting angle in the NORM position and manual sighting angle in the MAN position. In the normal position, sighting angle is automatically positioned by signals from the bomb nav system, and the antenna tilt control knob can be used to refine this position. In the MAN position, sighting angle is adjusted with the antenna tilt knob. The switch is inoperative in the ground manual and air modes of operation.

Sensitivity Time Control Knobs.

Two coaxial rotary sensitivity time control (STC) knobs (16, figure 1-72) provide a means of equalizing radar intensity over the entire scope display when operating in the ground modes at low altitude. The outer knob, labeled AMPL/OFF, has an OFF position at nine o'clock and is used to obtain an initial adjustment of display intensity or to turn the STC function OFF in the event of a malfunction in the STC circuit. The STC function is fixed in air mode but must be turned on with the outer knob. The inner knob, labeled SLOPE, is used to balance the display intensity throughout the sweep. The STC slope function is inoperative in the air mode. The STC function is inhibited in beacon mode. With STC on, an altitude compensation signal is provided to maintain even ground paint as long as the toe of the antenna beam remains at 80 nautical miles. With the toe of the beam at shorter distances, the slope and amplitude controls must be adjusted for compensation. The amplitude control sets the receiver sensitivity level at zero range and the slope control adjusts the time required after zero range for the receiver to regain full sensitivity. This time would be less if the toe of the beam was on a point 30 miles ahead of the aircraft than for a condition with the toe 80 miles ahead.

Video Adjustment Knob.

The video adjustment control knob (22, figure 1-72), labeled VIDEO, is used to increase the amplitude of the video signal supplied to the attack radar scope when it is turned clockwise. The video control determines the brightness of target returns as opposed to the CRT intensity control setting the overall baseline brightness of the scope. To set the video control properly, advance the function knob to XMIT and adjust antenna tilt to see the most returns on the scope. Adjust the video control until the target returns are sharp and bright against the picture background and give an overall optimum contrast to the picture. Video gain may need to be decreased slightly to prevent blooming of the target on a bomb run as the range decreases and the return grows stronger.

Transmitter Tuning Control Knob.

The transmitter tuning control knob (9, figure 1-72) allows continuous tuning of the transmitter over its entire frequency range. The knob, labeled XMTR TUNE, may be used when the frequency control knob is in the AFC 2 position.

Attack Radar Range Selector Knob.

The range selector knob (14, figure 1-72) is marked RANGE with 15, 30, 90, 160, and 200 mile positions on an outer scale and miles/diameter with 5, 10, 30, 80, and 200 mile positions on an inner scale. The inner scale is used in the GND MAN, GND AUTO and AIR modes to select desired scope range. The inner and outer scales are used in conjunction with each other in the GND VEL mode. The outer scale then determines maximum display range and the inner scale determines the diameter of the scope range being displayed about the cursors.

Attack Radar Scope.

The radar scope (3, figure 1-72) provides a sector scan plan position indicator (PPI) display with a fixed one radius offset sweep in all modes of operation except in ground velocity mode. In ground velocity mode, the sweep is a variable offset with a maximum displacement of six radii. In normal stabilization, the aircraft position on the scope is at the bottom in vertical alignment with the center of the scope. In north orientated

stabilization, the top of the scope display will be toward true north. The scope is 6 inches in diameter. The sector displayed is a 90-degree area ahead of the aircraft when in wide scan and a 20-degree area centered on the azimuth cursor when in narrow scan. An azimuth bezel (2, figure 1-72), around the top of the scope is graduated in one-degree increments with each 10 degrees marked to show azimuth displacements up to 50 degrees either side of the aircraft heading or ground track. When operating in the air mode or when the antenna is caged in ground manual mode, zero degrees on the scale represents aircraft heading. In other ground modes, the scan is displaced in azimuth to compensate for drift, and zero degrees represents ground track. North is vertical on the scope when north-orientation is selected. In the air to air tracking mode of operation, two arrows in the bottom of the bezel indicate target vertical position relative to antenna scan. When both arrows are lighted, the target is in the center of the scan. Range and azimuth cursors are displayed on the scope for fixtaking and target tracking. The cursors are positioned with the tracking control handle. Fixed range markers are provided for various ranges of operations. For 5, 10, 30, 80, and 200 ranges each range mark represents 1, 2, 5, 20, and 40 mile range increments respectively, except there are no range marks displayed in ground auto or ground velocity modes when in 5 and 10 or 15/5 and 30/10 range scales, respectively. Scope brilliancy and intensity of the bezel, cursors and range marks are controlled by knobs on the scope panel.

Attack Radar Cursor Range Counter.

The cursor range counter (11, figure 1-72) automatically indicates slant range in all modes of operation. The counter has four digital readout windows capable of indicating distance up to 799,900 feet in increments of 100 feet.

Attack Radar System Malfunction Lamp.

An amber system malfunction lamp (1, figure 1-72), labeled SYS MAL, will not light when the function selector knob is in the STBY position. It will light indicating any of the following: (1) failure in the antenna system, or input to the antenna system; (2) failure in the transmitter system; (3) exceeding the roll limit of the antenna; (4) any time the combined antenna pitch plus tilt angle exceeds the vertical limits (vertical limits with TFR are —8 to +30 degrees); and (5) anytime the frequency control knob is moved to the MFC position, except during beacon mode operation, the lamp becomes a coarse tuning indicator.

Note

It will be normal for the attack radar system malfunction lamp to light intermittently during TFR operation.

If the system is not usable, the function selector knob should be placed to STBY to stow the antenna.

Attack Radar Range Lock Indicator Lamp.

A green range lock indicator lamp (25, figure 1-72), labeled LOCK, will light when a range lock is acquired on a target when operating in the air mode.

Photo Malfunction Indicator Lamp.

The amber photo malfunction indicator lamp (8, figure 1-72), labeled PHOTO MAL, provides an indication of camera operation or malfunctions. The lamp will blink each time a film exposure is made. The lack of a light indicates a camera shutter malfunction. A steady light indicates film breakage or failure of the film feed mechanism. If a photo malfunction occurs, the malfunction circuitry can be reset by cycling the radar to standby, and back to on or transmit.

Note

In case of film stoppage, the malfunction can usually be cleared by removing the film magazine and manually turning the drive shaft two or three turns.

Unused Film Indicator.

The unused film indicator (6, figure 1-72) is a digital readout indicator that displays the percent of film remaining in the magazine. When the indicator reads 100 percent a maximum of 1300 frames of film is present.

Antenna Cage Pushbutton Indicator Lamp.

The antenna cage pushbutton indicator lamp (6, figure 1-70), labeled ANT CAGE, provides a means of caging the antenna and the lamp provides an indication that the antenna is caged, either due to manually pushing the button or due to failure of the automatic pitch and roll stabilization circuitry. Depressing the button will cage the antenna pitch and roll axes and align the antenna with the longitudinal and lateral axes of the aircraft; however, the antenna will continue to sector in azimuth, and tilt can be adjusted. When operating in the ground manual mode with the antenna caged, the radar scope will display aircraft heading at zero degrees azimuth and range sweep will be slant range instead of ground range. When the antenna is caged, a lamp in the button will light displaying the word CAGE. Depressing the button again after the antenna has been caged will uncage the antenna and the lamp will go out. Should the bomb nav system stabilization platform fail, the lamp will light and remain on until the system is switched to AUX reference.

Note

Do not cage the attack radar antenna during TFR operation. To do so will cause a flyup.

Waveguide Low Pressure Pushbutton/Indicator Lamp.

The attack radar waveguide is pressurized to approximately 29 psi. This pressurization is to prevent arcing in the waveguide and subsequent damage to the magnetron. An amber pushbutton/indicator lamp, located under the attack radar ground adjustment access door, will light and the radar system will go into a standby condition in the event of a drop in waveguide pressure below approximately 23 psi. This circuit is operative only when the attack radar function knob is in the XMIT or TEST position. Opening the access door and depressing the lamp will override the standby condition and the radar system will return to a transmitting mode. The lamp will remain lighted until the waveguide pressure returns to normal. To decrease the effects of transmitting with reduced pressure, minimum transmit time techniques must be employed at altitudes as low as practicable. Radar range selection should be 30 NM or less if possible. If arcing indications are apparent, turn the radar function knob to ON and wait as long as possible between transmission cycles. Only under extreme conditions, such as avoiding thunderstorms or when present position is in doubt, should the attack radar be used above 8000 ft msl.

CAUTION

Transmitting with the attack radar with less than normal waveguide pressure and an arcing indication on the scope will reduce the life expectancy of the radar system.

RADAR ALTIMETER SYSTEM (AN/APN-167).

The radar altimeter system is a dual channel low altitude radar system which provides precise absolute altitude, rate of altitude change, and preselected minimum altitude warnings. Rate of altitude change from 0 to 500 feet per second is furnished to the terrain following radar. Minimum altitude flyup signals are provided to the integrated flight instruments in the ILS and AILA mode of instrument system coupler operation. The system is composed of two receiver-transmitter (RT) units; two antennas, one for transmitting and one for receiving; a distribution box; a radar altimeter indicator and the necessary controls. The RT units are located in the forward electronic equipment bay. When the system is placed in operation, one RT unit is activated and the other is in standby for use in the event the operating unit malfunctions. In the event of a malfunction, the standby RT unit must be manually selected. The RT unit in operation is connected to the antennas and its outputs are distributed to other aircraft systems by circuits in the distribution box. The radar altimeter will break lock if the bank angle exceeds 45 degrees or if pitch angle exceeds ±20 degrees. The system incorporates a self-test feature for checking reliability. Refer to figure 1-69 for antenna location.

WARNING

High frequency radar waves can penetrate snow, sand, dry lake beds, etc. When operating in these areas, the radar altimeter may indicate a greater terrain clearance than actually exists.

RADAR ALTIMETER CHANNEL SELECTOR SWITCH.

The radar altimeter channel selector switch (5, figure 1-73), located on the miscellaneous switch panel, is labeled RADAR ALTM and has two positions marked CHAN 1 and CHAN 2. Placing the switch in either position will allow the RT unit in the respective channel to transmit and receive.

RADAR ALTIMETER BYPASS SWITCH.

The radar altimeter bypass switch (4, figure 1-73), located on the miscellaneous switch panel, is a two position switch marked NORMAL and BYPASS. Placing the switch to BYPASS when above 5000 feet over the terrain provides a signal to the TFR to permit blind letdowns. The switch will go to NORMAL as 5000 feet is passed during descent. When the switch is in the NORMAL position, blind letdowns from below 5000 feet above terrain may be accomplished. Positioning the switch to BYPASS, when performing TF operations below 5000 feet AGL, will cause a TF fail and a flyup maneuver. This will also cause the radar altitude low warning lamp to light unless actual AGL is within ±10 percent of the 1875 foot pseudo radar altitude from the bypass switch.

RADAR ALTIMETER.

The radar altimeter (20, figure 1-6), located on the left main instrument panel, provides absolute altitude indications from 0 to 5000 feet. Indications are provided by a pointer on a dial graduated in increments of 10 feet from 0 to 500, 50 feet from 500 to 1000, and 500 feet from 1000 to 5000. An OFF warning flag in a window on the right side of the dial will appear when power is removed from the system, when the system malfunctions, or when roll or pitch limits are exceeded.

WARNING

If power is lost on the system, the OFF warning flag will appear on the dial and the pointer will remain at the last powered indication.

The radar altimeter control knob on the lower right of the altimeter serves three functions; as an on-off control, to set a minimum altitude index pointer on the dial, and as a test button to check the system. Initially turning the knob clockwise applies power to the system; further rotation of the knob rotates the index pointer from zero to any desired minimum altitude setting. Depressing and holding the knob activates the self-test feature of the system and, if the RT unit is operating properly, provides an indication of 300 (±15) feet.

CAUTION

The radar altimeter must be turned off after each flight to prevent damage to the receiver-transmitter unit, should power be applied without cooling air.

The self-test feature may be used at any time and at any altitude below approximately 38,000 (±5000) feet.

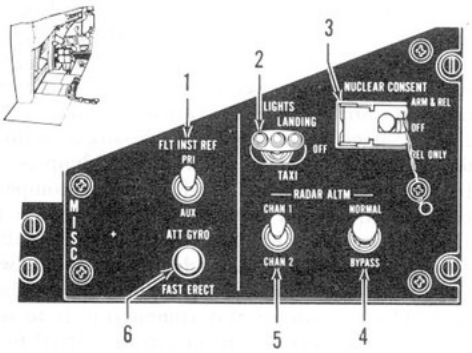
WARNING

Initiation of the self-test during TF operations at the 200-and 300-foot set clearances, while the radar altimeter signal to the TFR is controlling the aircraft, may result in a dive command from the TFR. In this case, the 83 percent flyup will not be operable.

Note

Performing a self-test of the radar altimeter during terrain following at set clearances of 400 feet or higher will cause a flyup, since the false altitude (300 ±15 feet) that the radar altimeter locks on during self-test is less than 83 percent of these clearances. If the radar altimeter signals are controlling the aircraft and self-test is performed at the 200 and 300 feet set clearances, a dive may result. The 83 percent flyup will be inhibited during radar altimeter self-test at the 200 and 300 feet set clearances.

Miscellaneous Switch Panel



1. Flight Instrument Reference Select Switch.
2. Landing and Taxi Lights Switch.
3. Nuclear Consent Switch.
4. Radar Altimeter Bypass Switch.
5. Radar Altimeter Channel Selector Switch.
6. AFRS Gyro Fast Erect Button.

F0000000-F022

Figure 1-73.

RADAR ALTITUDE LOW WARNING LAMP.

The radar altitude low warning lamp (21, figure 1-6), located on the left main instrument panel, will light when the absolute altitude of the aircraft is at or below the minimum altitude set into the radar altimeter. When lighted, the letters RADAR ALT LOW are displayed on the face of the lamp.

This lamp is also used in conjunction with the low altitude monitor. The monitor will cause this lamp to light for any of the following conditions.

1. The radar altitude data to the radar altimeter differs from that being supplied to the TFR by 10 percent or more.
2. The aircraft has penetrated below 83 percent of selected set clearance.
3. The selected set clearance plane on the TFR control panel and the set clearance plane received by the TFR computer differ by a predetermined value.
4. The low altitude monitor has failed. When the warning lamp lights due to any of these conditions, a fail flyup maneuver will occur. A fail flyup will not occur if the lamp lights due to indicated radar altitude being less than that set by the index pointer.

LOW ALTITUDE MONITOR.

The low altitude monitor provides a redundant 83 percent flyup function to back up the 83 percent flyup circuitry in the TFR. If, during terrain following operations, the aircraft descends below 83 percent of the selected clearance for any reason, the low altitude monitor will (1) cause the radar altitude low warning lamp to light (if it is not already lighted as a result of the aircraft being below the minimum altitude set into the radar altimeter), and (2) interupt the TF data good signal to the flight control system, which will cause initiation of a flyup maneuver. In addition, the low altitude monitor compares the two altitude signal outputs from the selected radar altimeter channel. If these signals differ by 10 percent or more, the radar altitude low warning lamp will light and a flyup maneuver will be initiated. The purpose of this comparator is to detect a failure in the radar altimeter altitude signal to a high limit or a failure of the bypass switch to properly switch from BYPASS to NORMAL. The low altitude monitor also contains an internal monitor to detect a malfunction in the selected clearance circuitry. If a malfunction is detected, this monitor will cause the radar altitude low warning lamp to light and a flyup maneuver to be initiated. The monitoring functions of the low altitude monitor are enabled only when one or both TFR channels are in a TF mode and the radar altimeter is locked-on.

WARNING

It is possible, during any phase of operation, for a low altitude monitor malfunction to cause a flyup maneuver when below the minimum altitude set on the radar altimeter. This can occur regardless of the position of the TFR channel mode selector switches. If this malfunction occurs, the radar altimeter should be turned off for the remainder of the flight.

Note

If the radar altitude low warning lamp lights because of a failure sensed by the low altitude monitor, the lamp will not go out when the autopilot release lever is depressed.

TERRAIN FOLLOWING RADAR (AN/APQ-134).

The terrain following radar (TFR) provides low altitude terrain following, obstacle avoidance, and blind letdown capability. The TFR consists of left and right antenna receivers, synchronizer transmitters, power supplies and computers in a dual channel configuration; a radar scope panel and a control panel. Each channel may be operated independently of the other in any one of three modes; terrain following (TF), situation display (SIT), or ground mapping (GM). During auto or manual TF turning flight maneuvers, the TFR antenna is positioned to look along the anticipated flight path, and the climb/dive commands are compensated for the aircraft being in a roll attitude. The TFR receives inputs from the radar altimeter, attack radar, converter set, flight controls system, doppler radar, central air data computer. Refer to figure 1-75 for TFR system tie-ins with other weapon system avionic equipment.

TERRAIN FOLLOWING (TF) MODE.

The TF mode allows the aircraft to be flown manually or automatically at a preselected terrain clearance. Climb and dive signals generated in this mode can be furnished to the ADI, ODS and to the flight control system. The set terrain clearance can be manually maintained by flying pitch steering commands on the ADI and ODS. In auto TF operation, the set terrain clearance will be automatically held. Should the aircraft descend to below 83 percent of the selected terrain clearance altitude setting, or if the aircraft is banked more than 45 degrees, the ADI and ODS will indicate a flyup command and a flyup maneuver will be initiated, if the aircraft is being flown with the auto TF switch in either the AUTO TF or OFF positions. A failure detected by the low altitude monitor will light the radar altitude low warning lamp and cause the flight control system to initiate a flyup maneuver. However, the TF failure warning lamp and pitch steering bars may not indicate a fail condition.

Note

The initial aircraft response to the flyup command may be as much as +2.8 absolute "g's."

The TF mode can also be used to make blind letdowns to a preselected terrain clearance. When using this capability, descent can be made manually using the pitch steering commands on the ADI and ODS, or automatically by placing the auto TF switch to the AUTO TF position. The descent is limited to a 12 degree dive. Only one channel at a time can be operated in TF mode. If both channels are placed to TF the second channel placed to TF will go to a standby condition as a backup and will automatically take over should the operating channel fail. A failure in the operating TFR channel will cause a flyup command to be sent to the pitch damper with the auto TF switch in either AUTO TF or OFF and a flyup steering command will be displayed on the ADI and ODS pitch steering bars.

Note

The flyup command can be interrupted by holding the autopilot release lever depressed (autopilot release/pitch control stick steering lever to the first detent after T.O. 1F-111 (B)A-593). To get rid of the flyup signal, the TFR channel controlling the aircraft must be turned out of the TF mode. If the auto TF switch is in the AUTO TF position, it should be positioned to OFF. The flyup signal is locked out to prevent automatic flyup when the flight control system is in the take-off and land configuration.

Refer to figure 1-74 for information pertaining to tie-in of the TFR with the flight control system. In the TF mode, antenna scan is vertical and the scope display is in the form of a non-linear E type presentation. A zero command line, displayed on the scope, provides a terrain clearance reference. The slope of the zero command line will vary with the speed of the aircraft, terrain clearance setting and the type of ride selected. Range displays on the scope are from left to right on a nonlinear scale so that ranges up to two miles are displayed over three-fourths of the scope and the remaining one-fourth of the scope displays returns up to ten miles. Elevation of returns along the ground track are displayed vertically on the scope.

Note

Electromagnetic interference from HF radio transmission on some frequencies may cause a flyup maneuver when operating the TFR in the TF mode. This interference may also cause degradation of the TFR scope displays. If HF radio use is essential and interference is noted when operating in the TF mode, the terrain should be cleared visually or, if this is not possible, the aircraft climbed to the minimum enroute altitude.

SITUATION (SIT) MODE.

This mode of operation is used in conjunction with TF mode for obstacle avoidance. Antenna scan is in azimuth, 30 degrees either side of ground track. Antenna tilt cannot be adjusted. Returns of the terrain that are at or above the aircraft altitude are displayed on the radar scope in a one-radius offset PPI presentation. Ground track is stabilized vertically along the center of the scope. Range graduations in this mode are linear.

GROUND MAPPING (GM) MODE.

The GM mode provides a scope presentation of the terrain that is ahead of the aircraft. Antenna tilt can be adjusted for best picture. This mode is used primarily for navigation. The antenna scan and the type of scope display are the same as when operating in SIT mode.

TFR CONTROL AND INDICATORS.

TFR Channel Mode Selector Knobs.

Two five-position rotary channel mode selector knobs (2, figure 1-76), located on the TFR control panel, permit selection of the desired operating mode in each of the two channels. The knobs are labeled L and R for the respective channel and are individually marked OFF, STBY, TF, SIT and GM. In the STBY position, power is applied to the channel for warmup. The TF, SIT and GM positions provide terrain following, situation display or ground mapping modes of operation respectively. Normally, each channel is operated in a different mode; however, unless one channel is in TF mode, the reference not engaged lamp (ATF not engaged caution lamp after T.O. 1F-111(B)A-593) will light when AUTO TF is selected. If both knobs are positioned to TF, the second channel will automatically go to a standby condition; then, should the operating channel fail, the one in standby will automatically take over.

Note

For normal operations, one channel should be in TF and the other in SIT or GM. Crews will not rely on automatic take over.

The channel that failed can be checked by rotating the mode selector knob from TF to STBY and back to TF again. The channel fail lamp will go out, but is not an indication that the malfunction has cleared until the operating channel selector knob is rotated out of TF causing the first channel to take over in the TF mode. If the channel fail caution lamp does not light, the malfunction has cleared and the channel is operating properly. When channel switching occurs over water, it is normal for the second channel to initially fly the aircraft above the set clearance with a gradual decrease in the command until the set clearance is re-established. This condition is caused by the standby channel having a full climb present from the altimeter override command which receives zero altitude while in standby.

CAUTION

If both channels are not placed in STBY or above when operating the TFR or attack radar, the crystal diodes in the receiver of the channel in OFF may be ruined, thus rendering the receiver inoperative.

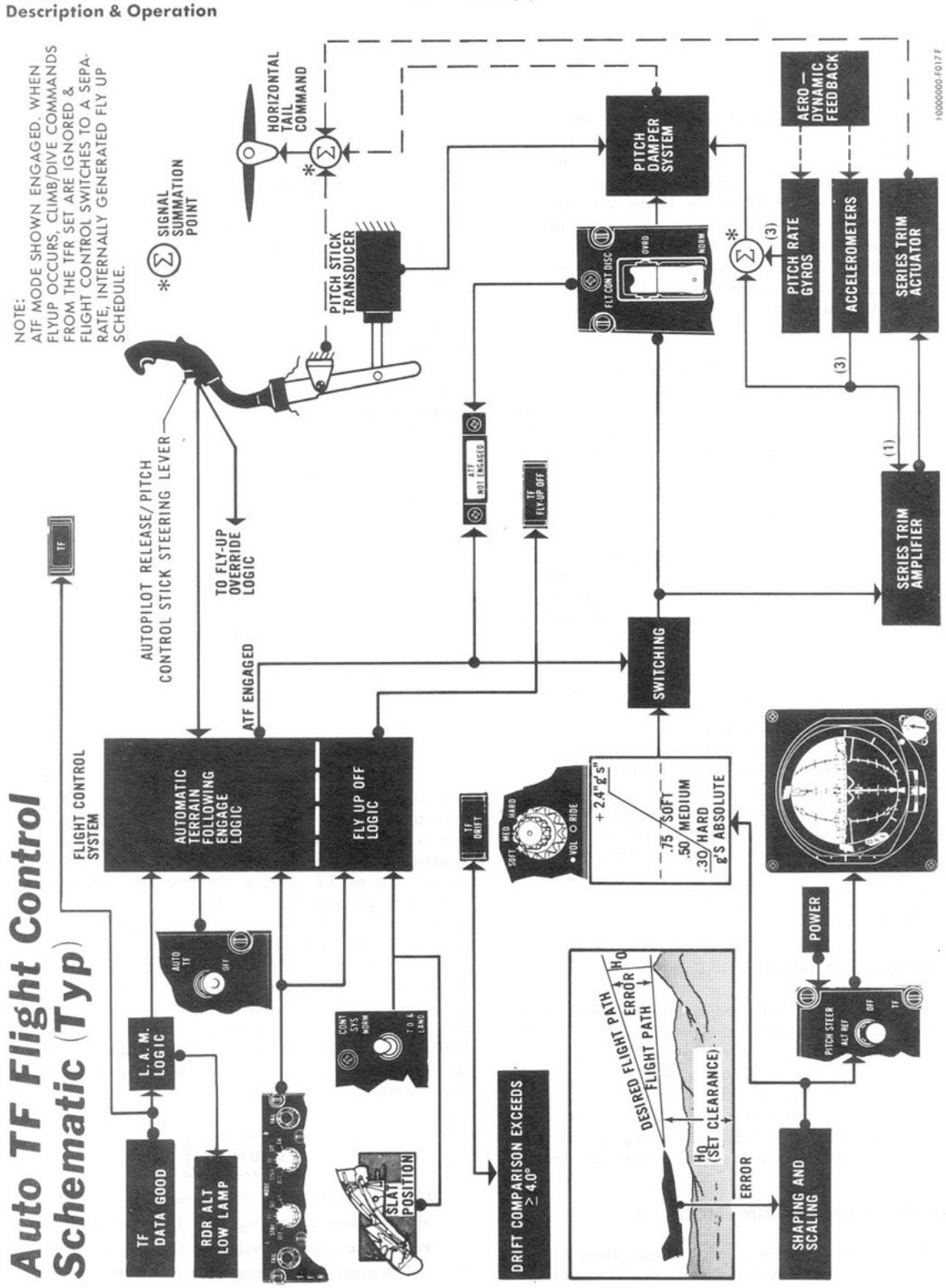

Auto TF Flight Control Schematic (Typ)
NOTE:
ATF MODE SHOWN ENGAGED. WHEN FLYUP OCCURS, CLIMB/DIVE COMMANDS FROM THE TFR SET ARE IGNORED & FLIGHT CONTROL SWITCHES TO A SEPARATE, INTERNALLY GENERATED FLY UP SCHEDULE.
HORIZONTAL TAIL COMMAND
SIGNAL SUMMATION POINT
PITCH STICK TRANSDUCER
PITCH DAMPER SYSTEM
AERO—DYNAMIC FEEDBACK
PITCH RATE GYROS
ACCELEROMETERS
SERIES TRIM ACTUATOR
SERIES TRIM AMPLIFIER
AUTOPILOT RELEASE/PITCH CONTROL STICK STEERING LEVER
TO FLY-UP OVERRIDE LOGIC
ATF NOT ENGAGED
TF FLY UP OFF
ATF ENGAGED
SWITCHING
FLIGHT CONTROL SYSTEM
AUTOMATIC TERRAIN FOLLOWING ENGAGE LOGIC
FLY UP OFF LOGIC
TF
L.A.M. LOGIC
TF DATA GOOD
RDR ALT LOW LAMP
AUTO TF
CONT SYS NORM
T.O. & LAND
SLAT POSITION
TF DRIFT
DRIFT COMPARISON EXCEEDS ≥ 4.0°
SOFT MED HARD
VOL RIDE
+ 2.4 g's
.75 SOFT
.50 MEDIUM
.30 HARD
g'S ABSOLUTE
DESIRED FLIGHT PATH
ERROR
FLIGHT PATH
HO
HO (SET CLEARANCE)
ERROR
SHAPING AND SCALING
POWER
PITCH STEER ALT REF
OFF
TF
FLT CONT DISC
GYRO
NORM
(3)
(1)
0000000-F017 F

Figure 1-74.

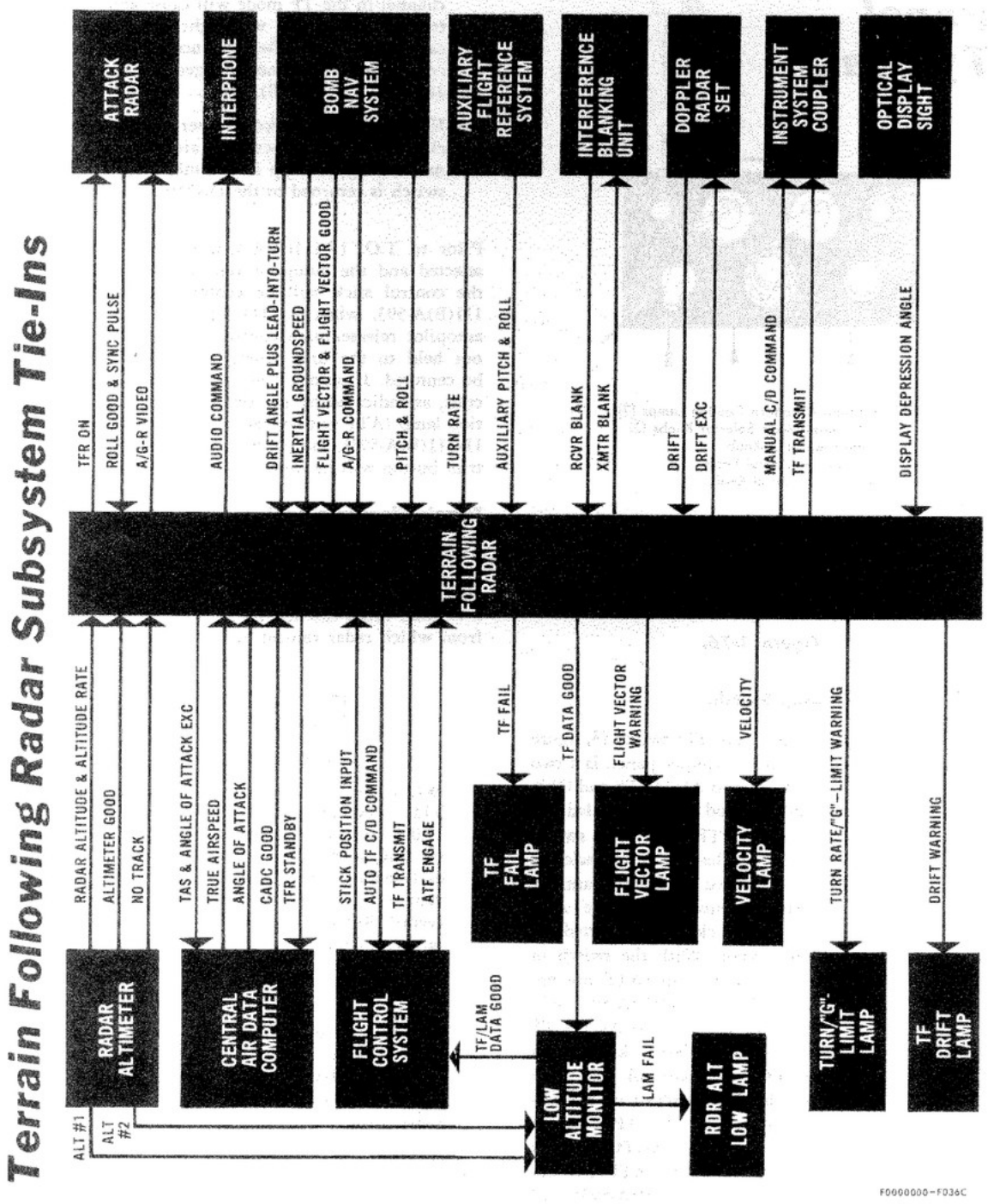
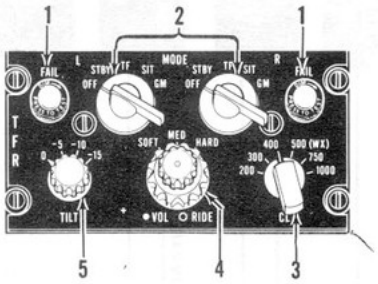

Terrain Following Radar Subsystem Tie-Ins
ATTACK RADAR
INTERPHONE
BOMB NAV SYSTEM
AUXILIARY FLIGHT REFERENCE SYSTEM
INTERFERENCE BLANKING UNIT
DOPPLER RADAR SET
INSTRUMENT SYSTEM COUPLER
OPTICAL DISPLAY SIGHT
TERRAIN FOLLOWING RADAR
TFR ON
ROLL GOOD & SYNC PULSE
A/G-R VIDEO
AUDIO COMMAND
DRIFT ANGLE PLUS LEAD-INTO-TURN
INERTIAL GROUNDSPEED
FLIGHT VECTOR & FLIGHT VECTOR GOOD
A/G-R COMMAND
PITCH & ROLL
TURN RATE
AUXILIARY PITCH & ROLL
RCVR BLANK
XMTR BLANK
DRIFT
DRIFT EXC
MANUAL C/D COMMAND
TF TRANSMIT
DISPLAY DEPRESSION ANGLE
RADAR ALTIMETER
RADAR ALTITUDE & ALTITUDE RATE
ALTIMETER GOOD
NO TRACK
ALT #1
ALT #2
CENTRAL AIR DATA COMPUTER
TAS & ANGLE OF ATTACK EXC
TRUE AIRSPEED
ANGLE OF ATTACK
CADC GOOD
TFR STANDBY
FLIGHT CONTROL SYSTEM
STICK POSITION INPUT
AUTO TF C/D COMMAND
TF TRANSMIT
ATF ENGAGE
TF/LAM DATA GOOD
LOW ALTITUDE MONITOR
LAM FAIL
RDR ALT LOW LAMP
TF FAIL LAMP
TF FAIL
TF DATA GOOD
FLIGHT VECTOR LAMP
FLIGHT VECTOR WARNING
VELOCITY LAMP
VELOCITY
TURN/"G"-LIMIT LAMP
TURN RATE/"G"—LIMIT WARNING
TF DRIFT LAMP
DRIFT WARNING
F0000000-F036C

Figure 1-75.

TFR Control Panel (Typical)



1. TFR Channel Failure Caution Lamps (2).
2. TFR Channel Mode Selector Knobs (2).
3. Terrain Clearance Knob.
4. Ride/Volume Control Knob.
5. Antenna Tilt Control Knob.

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Figure 1-76.

Auto Terrain Following Switch.

The auto terrain following (auto TF) switch (3, figure 1-26), located on the autopilot/damper panel, is a two position lever lock switch marked AUTO TF and OFF. The switch is locked in OFF and must be pulled out to move from OFF to AUTO TF. When the switch is in OFF and either TFR channel mode selector knob is in TF, the aircraft must be flown manually using the pitch steering commands on the ADI and ODS to hold the terrain clearance selected on the TFR terrain clearance knob. With the switch in OFF, the reference not engaged lamp (ATF not engaged caution lamp after T.O. 1F-111(B)A-593) will remain on. When the switch is placed to AUTO TF and either TFR channel mode selector knob is in TF, signals from the TFR will control the pitch damper to automatically hold the aircraft on the terrain clearance setting selected by the TFR terrain clearance knob. With the switch in AUTO TF, the reference not engaged caution lamp (ATF not engaged caution lamp after T.O. 1F-111(B)A-593) will go out.

Note

- Selecting AUTO TF without at least one TFR channel in the TF mode will cause abnormal trim operation and will light the flyup off caution lamp and the reference not engaged caution lamp (ATF not engaged caution lamp after T.O. 1F-111(B)A-593).
- If the pitch autopilot/damper switch is cycled during TFR operation, aircraft response may be degraded up to one minute after the switch is returned to the DAMPER position.

Prior to T.O. 1F-111(B)A-593, when AUTO TF is selected and the autopilot release lever is not held, the control stick will be centered. After T.O. 1F-111(B)A-593, when AUTO TF is selected and the autopilot release/pitch control stick steering lever is not held to the first detent, the control stick will be centered. If auto TF mode is controlling the aircraft, as indicated by the reference not engaged caution lamp (ATF not engaged caution lamp after T.O. 1F-111(B)A-593), the pitch trim function of the stick trim button will be inoperative.

Terrain Clearance Knob.

The terrain clearance knob (3, figure 1-76), located on the TFR control panel, has six positions marked 200, 300, 400, 500, 750 and 1000. The 500 position provides a weather mode marked (WX) which limits the area from which radar returns are processed.

WARNING

When the 500-foot position is selected, the TFR does not command on targets beyond 15,000 feet or above the aircraft fuselage center line, and will not anticipate the terrain early enough to provide a command to prevent terrain impact if airspeed limits are exceeded. Refer to Section V for airspeed limitations for this clearance setting.

Note

When flying at one clearance setting and the knob is positioned to a higher setting, a TF failure and flyup may be generated until the aircraft is maneuvered outside the 83 percent radar altimeter flyup range.

The TFR should maintain flight within the following tolerances. If these tolerances are not maintained, the equipment should be written up after the flight. If

an under-shoot occurs to the extent that an 83 percent flyup occurs, change to the other channel. If the flyup again occurs select the next higher setting. Do not operate at the lower setting where the 83 percent flyup occurred.

Selected Clearance	Terrain Clearance	
	Min.	Max.
200	170	300
300	260	425
400	350	550
500	440	650
750	675	950
1000	900	1200

During automatic terrain following flight over level to rolling terrain or over water with HARD ride selected, the radar altimeter should indicate the altitude above the terrain corresponding to each clearance knob setting within these tolerances. Terrain clearances when cresting peaks will usually be slightly less than the stabilized, level terrain clearances. The above tolerances are not directly applicable to terrain following flight over rugged terrain; however, terrain clearances when cresting peaks should not consistently be below these tolerances.

Ride/Volume Control Knob.

The ride/volume control knob (4, figure 1-76) is located on the TFR control panel. The inner knob labeled RIDE is a three position rotary knob marked SOFT, MED, and HARD. This knob controls the magnitude of the negative "g" forces imposed on the aircraft by the flight control system as it maintains a set altitude clearance above the terrain. The commanded dive "g" forces of 0.3, 0.5 and 0.75 will be experienced in the HARD, MED, and SOFT positions respectively. Progression of the ride control from HARD to SOFT will compute an earlier (softer) pullup command upon approach to an obstacle. The system will automatically provide a pullup command if necessary to avoid an obstacle regardless of the ride selected. The outer knob labeled VOL adjusts the volume of aural commands heard through the interphone. The aural commands consist of 2500 Hz frequency tone for climb and 500 Hz frequency tone for dive at 20 pulses per second per "g." Any time the pitch steering bars on the ADI and ODS are centered there will be no tone.

Note

- When not using TFR, the knob should be turned as required to prevent interference on interphone.
- During TFR operation, secondary volume control of the TFR aural is provided by the ILS monitor knob on each communications panel.
- The aural climb/dive command will correspond to the ADI/ODS pitch commands during TF operation.

Antenna Tilt Control Knob.

The antenna tilt control knob (5, figure 1-76), located on the TFR control panel is used to position antenna tilt between zero and —15 degrees for the best ground return when operating in the GM mode. The knob has antenna tilt angles of 0, —5, —10 and —15 marked for reference only.

Range Selector Knob.

The range selector knob (5, figure 1-77), located on the TFR scope panel, has four positions marked 5, 10, 15 and E. The first three positions change range of the scope presentation when using SIT or GM modes. The E position is used with the TF mode only.

Radar Scope Tuning Control Knobs.

Four radar scope tuning control knobs (4, figure 1-77) are located on the TFR scope panel. The knobs are labeled CURSOR, MEMORY, CONTRAST and VIDEO from top to bottom. The cursor knob adjusts the brilliance of the range cursors. The memory knob increases or decreases scope storage retention time. The contrast knob adjusts scope contrast for optimum viewing. The video control adjusts the video return brightness to desired level.

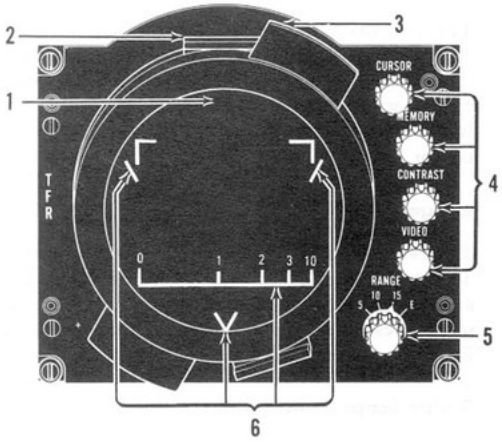
Radar Scope.

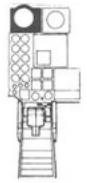
The radar scope (1, figure 1-77), located on the TFR scope panel, provides a direct viewing presentation of either an E (vertical scan) display when in TF mode or a sector PPI (azimuth scan) display when operating in SIT or GM modes. The scope overlays provide a rectangular grid with a 0 to 10 nautical mile scale at the bottom of the scope for TF mode and a "V" shaped grid for sector PPI presentations in SIT or GM modes. A self-test pulse (10, figure 1-77) is located at approximately 1.5 miles range at the top of the E scope presentation when the system is in TF mode of operation. Absence of this pulse, in TF mode, indicates improper system operation. The polaroid filter controls around the face of the scope can be rotated to adjust polarization of light for the best display under various degrees of light. A red scope presentation for night vision adaptation can be obtained with the filter controls. The ear-type handles on each side of the scope are provided to facilitate removal or installation of the unit.

TFR Channel Fail Caution Lamps.

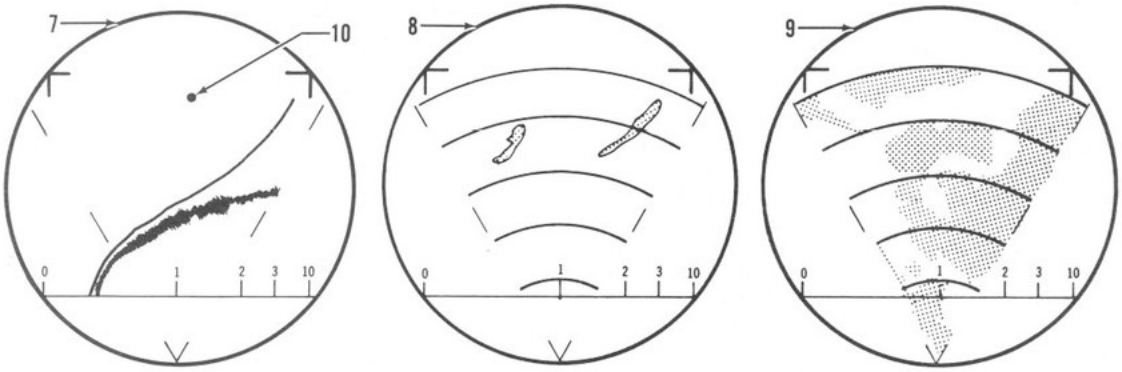
Two amber channel failure caution lamps (1, figure 1-76), located on the TFR control panel, are individually marked FAIL and are labeled L and R for the respective left and right channels. When the channel mode

TFR Scope Panel and Presentations

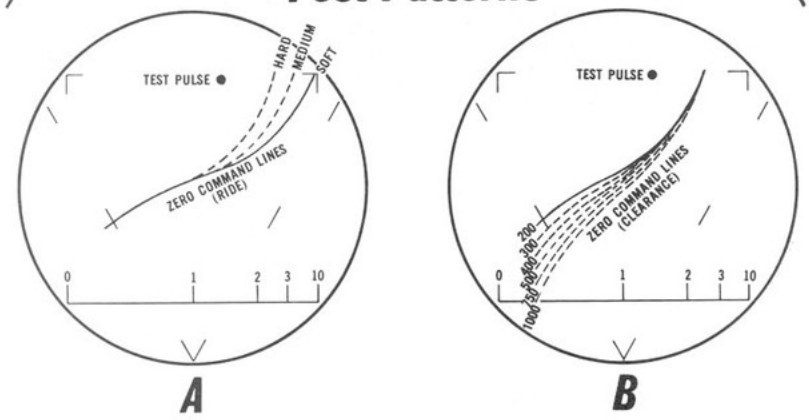




1. Radar Scope.
2. Polaroid Filter Control (2).
3. Scope Removal Handle (2).
4. Radar Scope Tuning Control Knobs (4).
5. Range Selector Knob.
6. Scope Overlay.
7. TF Mode Terrain Following E Display.
8. Sit Mode Lateral Terrain Search PPI Display.
9. GM Mode Ground Mapping PPI Display.
10. Test Pulse.



Test Patterns



F7341300-F002A

Figure 1-77.

selector knob is placed from OFF to STBY, the fail lamp will light to indicate that channel is not yet ready to operate. The lamp will go out after approximately 3 minutes indicating the channel is ready. After the channel is ready, a fail light with the mode selector switch in TF, SIT or GM position indicates a malfunction in that channel. The lamps will also light when the aircraft is above 5000 feet absolute altitude, and the radar altimeter bypass switch is in the NORMAL position. A press-to-test feature allows each lamp to be checked.

Note

- The lamps will momentarily blink when changing the TFR terrain clearance knob from low to higher clearance settings or switching into or out of the 500(WX) setting. This is a normal indication.
- The lamps will also blink rapidly when more than 0.5 "g"s are commanded if the instrument system coupler pitch steering mode switch is not in the TF position when operating in the manual TF mode.

TF Drift Caution Lamp.

The TF drift caution lamp located on the main caution lamp panel (figure 1-36) provides an indication that the primary and secondary drift plus lead into turn signals do not compare within 4 degrees. When this occurs, the TFR antennas are locked at zero degrees azimuth and therefore can no longer lead the aircraft into turns. When the lamp is lighted, the letters TF DRIFT are visible. Under IFR conditions, TF flight should not be performed unless drift and turn rate are zero when the lamp is lighted. Under VFR conditions, fly TF in slow turns or with small drift angles only if the terrain clearance can be verified.

WARNING

When the drift caution lamp is lighted, TF flight should not be performed with drift angles in excess of 3 degrees, and bank angles should be limited to 10 degrees unless terrain clearance can be assured under VFR conditions.

Reference Not Engaged Caution Lamp.

The reference not engaged caution lamp is located on the left main instrument panel (figure 1-36). The following conditions must be satisfied to put the reference not engaged caution lamp out during TFR operation:

1. Either TFR channel selector knob in the TF position,
2. The auto TF switch in the auto TF position, and
3. "G" command controlling the aircraft with the TFR checking safe.

While operating in auto TF, the lamp can come on due to conditions not associated with TFR operations. For example, if operating in auto TF with either constant track or heading navigate autopilot, lateral control stick steering will disengage the autopilot heading sub-mode and cause the lamp to light. With the lamp lighted under this situation, a subsequent failure in the "g" command logic will go undetected and could result in the aircraft flying into the ground. For detailed description of the lamp and its function with other systems, refer to Autopilot System, this section.

WARNING

When the reference not engaged (ATF not engaged after T.O. 1F-111(B)A-593) caution lamp lights while in auto TF, immediate action must be taken to put the lamp out or terminate auto TF flight to prevent possible terrain impact.

ATF Not Engaged Caution Lamp.

After T.O. 1F-111(B)A-593, the ATF not engaged caution lamp (8, figure 1-6), located on the left main instrument panel, will light under the following conditions:

1. Either TFR channel mode selector knob is in the TF position and the auto TF switch is OFF.
2. The auto TF switch is in AUTO TF and neither TFR channel mode selector knob is in the TF position.

WARNING

If the autopilot release lever is depressed to the first detent during manual or auto TF operation, the TF flyup off caution lamp will not light although 83 percent flyup protection is not available.

3. The flight control disconnect switch is placed to the OVRD (override) position.

The letters ATF NOT ENGAGED are visible in the face of the lamp when lighted.

Note

The ATF not engaged lamp is not connected in any way to the autopilot system.

Flight Vector Caution Lamp.

The flight vector caution lamp, located on the main caution lamp panel (figure 1-36), provides an indication that the TFR is no longer making a comparison check of the aircraft flight vector and that the TFR may be computing using the backup flight vector. The lamp will light when either the inertial flight vector data good signal or the CADC data good signal is lost to the TFR. If the CADC data good signal is not received at the TFR, the lamp will light and the primary flight vector will be used in the TFR computation. If the inertial flight vector data good signal is not received at the TFR, the lamp will light and backup flight vector (aircraft pitch angle minus aircraft angle-of-attack) will be used in the TFR computation. The letters FLT VECTOR are visible on the lamp when it is lighted.

Turn Rate/G-Limit Caution Lamp.

The turn rate "g"-limit caution lamp located on the main caution lamp panel (figure 1-36) provides an indication when lighted that any or all of the following conditions exist:

1. The aircraft is executing a turn in which the heading rate exceeds the maximum safe value (2.0 degrees per second) for which the TFR antenna lead-into-turn stabilization is reliable.

Note

If excessive roll rates or rapid reversal of bank are accomplished, the turn rate/g limit caution lamp may not light immediately due to the lag in the turn rate caution circuitry.

2. The aircraft is in a roll attitude and the computed roll compensated climb command exceeds 2.47 "g"s.
3. The caution lamp will light whenever the aircraft is at a zero roll and close enough to an obstacle that the TFR is generating its maximum 2.47 "g" command.

Note

The TFR is limited to commanding a maximum of 2.47 "g" maneuvers.

The lamp should be monitored in turns. When the lamp lights, the bank angle should be reduced to put the lamp out. When lighted, the letters TURN/G-LIMIT are visible on the lamp.

Velocity Caution Lamp.

The velocity caution lamp located on the main caution lamp panel (figure 1-36) indicates the TFR is computing with a mach 0.8 airspeed. The lamp will light when the true airspeed and DCC groundspeed differ by 130 knots. The letters VELOCITY are visible on the lamp when it is lighted. Terrain following flight shall be restricted to a velocity range of 0.7 mach to 0.9 mach when the lamp is lighted. Should the comparison return to its normal level, the lamp will go out and operation will be normal.

WARNING

Airspeed should be held between mach 0.7 and 0.9 when the velocity caution lamp is on. If these speeds are exceeded the TFR will not properly anticipate terrain clearance commands and the aircraft may fly into the ground.

Note

The velocity caution lamp will light during TFR ground checks.

TF Fly-up Off Caution Lamp.

The TF fly-up off caution lamp, located on the main caution lamp panel (figure 1-36) provides an indication that fly-up protection is not available. The letters TF FLY-UP OFF are visible on the face of the lamp when it is lighted. The fly-up maneuver is inhibited until the fly-up circuit is armed. The fly-up off lamp will be lighted during TF mode selection until the fly-up circuit is armed. Arming of the fly-up circuit will result in the TF fly-up off lamp going out. The fly-up circuit can be armed by one of two ways (a) The TF set must initially check safe (TF fail warning lamp goes out) or (b) Prior to T.O. 1F-111(B)A-593, the autopilot release lever momentarily depressed; after T.O. 1F-111(B)A-593, the autopilot release/pitch control stick steering lever momentarily depressed to the first detent. After the fly-up circuit is armed, a subsequent TF fail signal (TF fail warning lamp comes on) will result in a fly-up maneuver. The fly-up circuitry is interlocked with the flight control system switch and slat extension mechanism to prevent inadvertent fly-ups during take-off and landing operations. When airborne and with the L or R TFR channel mode selector knob in the TF

position, the lamp will light when the flight control system switch is: (1) in the T.O. & LAND position, or (2) in the NORM position and the slats are extended. Certain power failures within the TFR system will also cause the lamp to light. The lamp will also light if the auto TF switch is placed to the AUTO TF position and neither TFR channel is in a TF mode, however this is an abnormal switching configuration and should be avoided. Terrain following flight should not be attempted when the lamp is lighted.

WARNING

If the autopilot release lever is depressed to the first detent during manual or auto TF operation, the TF flyup off caution lamp will not light and flyup protection is not available.

TF Failure Warning Lamp.

A TF failure warning lamp (15, figure 1-6), located on the left main instrument panel, provides a more apparent indication of TFR channel malfunctions. If each channel is being operated in a different mode the lamp will light when the channel in TF mode malfunctions. If both channels are in TF mode, the lamp will momentarily light when the channel in operation fails and the backup channel takes over. Should the backup channel in turn fail, the lamp will light and remain on.

TF FLYUP CAUSES.

The TF flyup is caused by one of several conditions. These conditions are: (1) the TFR system detects an internal malfunction, (2) the TFR system detects an excessive error in comparison of certain inputs to the TFR, (3) the TFR system detects a loss of one or more input data good signals from other systems, and (4) altitude (AGL) is less than 83% of selected set clearance, and (5) the low altitude monitor detects an error in the radar altimeter outputs or the redundant 83% flyup circuits or the set clearance fails to a low value. Refer to figure 1-78.

OPERATIONAL CONSIDERATIONS.

Operating Configurations.

The recommended operating configuration for the TFR is to use one channel in the TF mode and the other channel in the SIT mode for maximum utilization of the terrain following and obstacle avoidance capabilities of the system. It is specifically recommended that the automatic channel switchover feature (both channel mode selector knobs in TF) not be used at clearances below 500 feet. Should a failure occur at the lower clearances in the channel operating in the TF mode, a clearance of 1000 feet should be flown after manually switching to TF in the opposite channel until proper performance is confirmed. It is also recommended that the index pointer in the radar altimeter be set at the minimum terrain clearance tolerance for the selected clearance to be flown.

TFR Warning & Caution Lamp Indications

DATA SIGNALS				LAMPS LIGHT
ROLL	LARA	CADC	INERTIAL	
FAIL	GOOD	GOOD	GOOD	L & R FAIL Caution Lamps & TF Warning Lamp
GOOD	FAIL	GOOD	GOOD	L & R FAIL Caution Lamps & TF Warning Lamp
GOOD	GOOD	FAIL	GOOD	FLT VECTOR Caution Lamp
GOOD	GOOD	GOOD	FAIL	FLT VECTOR Caution Lamp
GOOD	GOOD	FAIL	FAIL	L & R FAIL Caution Lamps & TF Warning Lamp

Figure 1-78.

If the low altitude warning lamp lights, the crew should monitor aircraft altitude and attitude, and anticipate an 83 percent flyup signal. If the aircraft descends through the 83 percent tolerance and the 83 percent flyup does not occur, immediately climb to a safe altitude and recheck the flyup circuitry of both TFR and radar altimeter channels prior to resuming TF flight.

Note

The minimum terrain clearances depicted in this section are based on hard ride, and flight over level to rolling terrain or over water.

When flying auto or manual TF, the radar altimeter indicator must be monitored for an altitude error, especially while flying over terrain having low radar return. The altitude information from the radar altimeter is used to generate the TF climb/dive command when forward video is lost. If there is an error on the radar altimeter indicator, e.g., indicating approximately 3000 feet while flying level at less than 1000 feet, and forward video is lost, the climb/dive command will be a dive and the 83 percent flyup will not occur because the altitude being in error indicates a condition of being above selected set clearance.

WARNING

- Terrain following flight should not be attempted when there is an obvious altitude error indicated on the radar altimeter indicator.
- Both crew members must monitor aircraft terrain clearance during TF flight. During periods when forward video is lost (due to water or low surface reflectivity), aircraft altitude will be monitored by using outside visual reference (if possible), AVVI interpretation, and close observation of TF system operation. The crew should be especially watchful for unexplained descents while over water or level terrain.

Changing Terrain Clearance Settings.

When changing selected clearances on the TFR to a higher setting, the terrain clearance control knob should be progressively rotated through each intermediate setting and the aircraft allowed to stabilize at the intermediate clearances until the desired clearance is reached. Instantaneous switching beyond more than one incremental clearance may cause the aircraft to assume a climb angle greater than 20 degrees, thus possibly causing the radar altimeter to lose range track. In this event, the aircraft will not level out at the desired clearance setting and the flyup must be overridden by manual control.

Blind Let-Down.

When initiating a blind let-down in the TF mode from above 5000 feet absolute, the radar altimeter will not be locked and will drop out its data good signal to the TFR. To prevent this signal from generating a flyup, the altimeter by-pass switch must be positioned to BY-PASS to give the TFR a pseudo altitude and data good signal. Upon passing through 5000 feet, the altimeter should lock, the by-pass switch will drop out, and normal altitude and data good signals will be supplied to the TFR. If the terrain is mountainous, the clearance may suddenly increase beyond 5000 feet and a fail signal will result. Should this occur, immediately reposition the by-pass switch to BY-PASS and resume the let-down. If the let-down is being made to 1000 feet set clearance, the dive angle will be limited to 10 degrees until the radar altimeter locks. At this time the dive angle may increase to 12 degrees. At other set clearances, the dive limits are 12 degrees above and below 5000 feet. It is recommended that an initial set clearance of 1000 feet be used for all blind letdowns. At this clearance setting, a climb command should be indicated on the ADI/ODS command bars at approximately 2000 feet AGL, and aircraft rotation toward level flight should begin no later than 1600 feet AGL. When a let-down is made from high altitude in the vicinity of mountainous or rugged terrain, the following is recommended to minimize the annoying radar altimeter break-lock when passing through the lock-on altitude and also to reduce the rate of descent at the set clearance level-off. Manual descent to MEA is recommended with the autopilot release lever depressed, while the aircrew monitors the pitch steering bar and E scope display. After the aircraft is established at MEA and the attack radar and E scope reveal no abrupt ridges along the aircraft flight path, an auto TF descent should then be initiated. Aircrews should be aware that while flying at MEA, the 5000 foot limit of the radar altimeter may be exceeded resulting in a TF fail-safe flyup.

WARNING

When a blind let-down is made, the radar altimeter must be monitored to assure that it locks on after passing through 5000 ft absolute. If the let-down is made over water or areas of low radar energy return, the radar altimeter is the only source of a signal to compute a level-off at the set clearance.

Note

Blind letdowns, started at an altitude that does not allow dive angle to reach 10 (±1) degrees prior to LARA ring-in (5000 feet AGL), do not verify proper radar altimeter operation.

Low Altitude Flyup.

Should the aircraft descend below 83 percent of the set clearance, a TF fail occur or a low altitude monitor fail occur, a fail-safe flyup maneuver will be commanded. The flyup command generated by descending below 83 percent of the set clearance will not be terminated until the aircraft climbs through the 83 percent point. The pitch trim function of the stick trim button is disabled and the control stick is centered during TFR flyup maneuvers.

WARNING

If the condition that has caused the aircraft to initially descend below 83 percent is still present and the TFR is allowed to control the aircraft, a pushover can result which will dive the aircraft back through the 83 percent point at an angle from which a flyup command cannot recover.

Should a TF fail or a low altitude monitor fail occur and a fail-safe flyup maneuver be initiated, the maneuver should be overridden by depressing the autopilot release lever (autopilot release/pitch control stick steering lever to the first detent after T.O. 1F-111(B)A-593) on either control stick as the aircraft attains a 20-degree pitch angle or limit angle-of-attack, whichever is less. Maintain this angle-of-attack or attitude until sufficient altitude has been gained to clear all known obstacles in the immediate area. Before releasing the lever, the pilot should check that the channel failure caution lamp on the TFR control panel for the channel in the TF mode is not lighted and that the radar altitude low warning lamp is not lighted. If either lamp is still on, releasing the autopilot release lever will allow the flyup maneuver to be resumed. Should the flyup maneuver not be terminated by depressing the autopilot release lever (autopilot release/pitch control stick steering lever to the first detent after T.O. 1F-111(B)A-593), the TFR should immediately be switched out of the TF mode, the auto TF switch turned to OFF, and the flyup overridden by stick force if necessary. If the flyup continues, turn the radar altimeter off.

Aircraft Pitch Attitude.

If the pitch attitude of the aircraft exceeds 20 degrees of pitch during a flyup command, the radar altimeter may break track. If this occurs, the flyup must be manually overridden until the radar altimeter can regain track and allow the TFR to check safe.

Turning During TF Flight.

TF flight can be performed up to 45 degrees of roll angle or 2 degrees per second of turn rate. If the roll limit is exceeded, the TFR will declare a fail and flyup command. If the turn rate limit is exceeded, the turn/"g"-limit caution lamp will come on. The TFR antennas lead the aircraft in a turn but cannot scan terrain which might be in the aircraft path in turns greater than 2 degrees per second. In a turn, the turn/"g"-limit caution lamp should be closely monitored. If this caution lamp comes on, reduce the turn rate or roll angle until it goes out. In the event of a TF fail in a roll, go to zero roll when terrain permits.

WARNING

- If the turn/"g"-limit caution lamp comes on during a turn, reduce the turn rate or roll angle until it goes out.
- If a TF fail occurs in a roll, go to zero when terrain permits.

Stick Pitch Inputs During Auto TF.

If in the process of making small heading corrections during auto TF, some stick force is applied in pitch, the aircraft response to auto TF commands may be degraded up to 5 seconds after the stick control forces are released. During TF fail safe flyups, control of the aircraft should be achieved by depressing the autopilot release lever (autopilot release/pitch control stick steering lever to the first detent after T.O. 1F-111(B)A-593) and using stick force until a decision is reached regarding continued TF operation. Angle-of-attack should not be allowed to exceed operating limits included in Section V. Maintain safe terrain clearance during this period.

Aircraft Trim.

When the auto TF switch is placed to the AUTO TF position, and the autopilot release lever is not held, the control stick will be centered. When the auto TF mode is controlling the aircraft, the pitch trim function of the stick trim button is inoperative.

WARNING

Improper operation will result if the auxiliary pitch trim switch is operated during auto TF or pitch autopilot modes.

Auxiliary pitch trim should not be used during auto TF operation. If nose up trim is used, the series trim actuator will drive nose down until the trim effect is neutralized. This reduces the amount of remaining down elevator available through the series trim actuator. When a down command is subsequently received from the TFR, there will be insufficient elevator remaining to keep the aircraft on the set clearance and ballooning on the back side of hills will occur.

Tower Detection Capabilities.

Towers and power lines may not reflect sufficient returns to cause the TFR to fly the aircraft over the obstacles. Flights should be planned to avoid these areas.

Effect of Precipitation.

Preflight planning must include consideration of enroute weather. If heavy rain or thunderstorm activity is forecast, TFR operation may not be possible. Use of attack radar to vector around weather cells is recommended and the 500(WX) set clearance will provide more capability in dense clouds or rainfall. The E scope should be monitored for video returns from weather during TFR flight. The back scatter from moderate to heavy precipitation will often be visible on the E scope. If the operator cannot determine where the terrain ends and the precipitation begins on the E scope, the automatic signal detection circuitry will also be incapable of discrimination and a climb command will result. As video returns from rain approach the zero command line from the right side of the E scope, a climb command can be expected. When a climb command occurs, the pilot should not allow the auto TF to exceed the angle-of-attack limits or exceed a 20-degree climb attitude and should add power to maintain airspeed. If either of these angle limits are encountered, depress the autopilot release lever (autopilot release/pitch control stick steering, after T.O. 1F-111(B)A-593), maintain the climb, and level off at a safe altitude (not less than MEA). When the video from weather disappears from the E scope and normal ground return is present, TFR operation can be resumed.

WARNING

- Certain weather conditions can cause blanking (no video presentation) of the E scope and/or attack radar with no associated flyup. This blanking eliminates radar returns from terrain and precipitation. The TFR system interprets the blanking as the ground return from a flat, low radar energy return surface such as a body of water and reverts to radar altimeter override mode. In this mode, the TFR will not provide safe flight over other than known level terrain.
- If E scope blanking due to weather conditions is observed or suspected, an immediate climb must be initiated to MEA.

The improved weather penetration capabilities of the planar array antennas were achieved as a trade-off against degraded detection capabilities of smooth, regular objects (i.e., towers, water tanks, etc.). Flights should be planned to exclude these targets from the flight path.

Flight Over Wooded Terrain.

During flight over sparsely wooded terrain, dead or defoliated trees, the TFR may not command sufficiently on the back scatter from the trees to maintain the selected clearance over the tops of the trees thus the tops of isolated trees may project above the 83 percent flyup threshold at the lowest clearance. To avoid unsafe flight conditions a minimum clearance of 200 feet greater than the tops of the trees should be used during TF flight over areas of dead trees or extremely tall trees. Over densely wooded areas, the radar altimeter will tend to measure aircraft clearance alternately between the ground and the tops of the trees, thus the indicated radar altitude may become very erratic and may generate a number of short duration flyups.

Flight Over Sloping Terrain.

When flying over smooth sloping areas of low reflectivity such that radar altimeter commands are controlling the aircraft, the aircraft will fly at an offset from the selected clearance proportional to the magnitude of the slope. If the flight path is up an extended slope, the aircraft will fly below set clearance by 87 feet/degree of slope and the 83 percent flyup threshold may be reached. When flying down an extended slope, the aircraft will fly at an offset above the set clearance.

Flight Over Mountainous Terrain.

When flying auto TF over steep mountainous terrain, a flyup command without a failure indication may occur shortly after cresting a peak which has a large extended slope with a small hill on the back side. This flyup is an inherent characteristic of the terrain following radar system and the frequency and severity is a function of the terrain, set clearance, airspeed, ride setting and aircraft gross weight. Flyups for a duration of up to 4 seconds may be experienced with angle-of-attack approaching 15 degrees with buffet onset. Flyups of this nature can be minimized, if desired, by performing flight over rough mountainous terrain at a higher set clearance and/or selection of medium or soft ride.

Use of E-Scope.

The E-scope presentation is intended for an advisory display and is not a primary command display. The pilot can fly manual TF by keeping the video below the command line but due to the lack of proper feedback to the display it should not be used as the primary manual display command below 500 feet set clearance; however, the E-scope display should be monitored at all clearances to determine if forward video returns are being received. When the video on the TFR scope becomes weak or barely visible, the commands will be generated from the radar altimeter. Under these conditions the pilot must assure that the surface is water or other low reflectivity smooth surface by visual means or by comparison with the other TFR channel.

Note

At clearance settings of 750 feet, the start of the TFR zero command line (ZCL) may have a short vertical line extending downward. Terrain video may penetrate through the vertical portion when flying close to the selected clearance. Also, below 750 feet clearance, there may be a short gap in the ZCL near the center of the presentation that can change with ride control, selected clearance or airspeed. These are normal conditions on the E-Scope presentation and will not affect system performance.

SIT Mode Display.

The SIT mode of the TFR provides a scope display of targets that are at or above the aircraft altitude. Since this mode does not provide a margin of vertical clearance, the scope display should only be used as a reference for avoiding obstacles and not for overflying obstacles.

Primary Attitude Reference.

The TFR is dependent upon the bomb nav system for roll, pitch, flight vector, drift and turn signals. The AFRS provides backup pitch and roll signals. Since all sources of possible error are not continuously tested, the flight crew should monitor all available cockpit indications (such as cross-checking primary vs. auxiliary attitude indicators, large differences between heading and ground track, unrealistic winds, radar scope stabilization etc.) to detect any malfunctions that may not be detected by the bomb nav system. If an abnormal condition is observed, TF flight should be terminated.

Auxiliary Attitude Reference.

The auxiliary flight reference system (AFRS) provides pitch and roll signals to the TFR as the primary source when the inertial navigation system is inoperative or not selected. During terrain following flight on the AFRS, primary attitude reference will not be available for a comparative check to detect errors or failures on critical inputs to the TFR system. When using the auxiliary reference, the primary attitude, heading, and flight vector caution lamps will be lighted indicating that the primary flight vector is not being used.

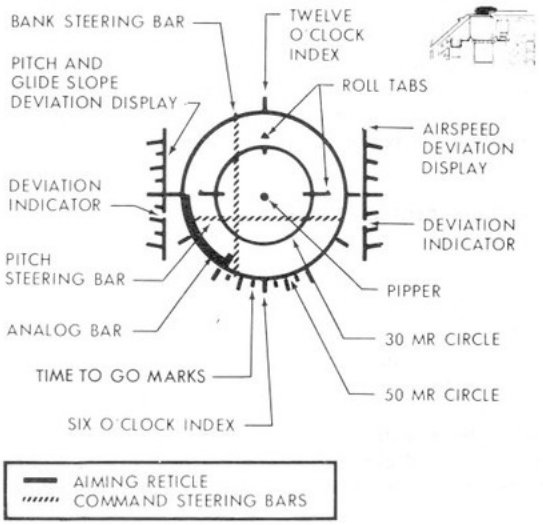
WARNING

Terrain following flights should be limited to day VFR conditions when the AFRS is furnishing attitude signals to the aircraft subsystems.

OPTICAL DISPLAY SIGHT (AN/ASG-25) (ODS).

The optical display sight (ODS) is integrated with other systems in the aircraft to deliver bombs and missiles, and provide homing, navigation and landing information. The system consists of the optical display sight and control panel, located at the left crew station, and an amplifier located in the forward electronics bay. The optical display sight provides indications in the form of two presentations: an aiming reticle, lighted in red, and a set of command steering bars, lighted in green. Refer to figure 1-79 for aiming reticle and command steering bar presentations. The aiming reticle consists of a 2-milliradian center pipper, a 30-milliradian circle, roll reference tabs, a 50-milliradian circle, analog bar reference tabs which serve as a time-to-go scale, an analog bar presentation, and two deviation indicators. All elements of the aiming reticle are

Aiming Reticle & Steering Bar Presentations (Typ)



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Figure 1-79.

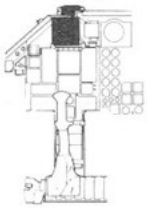
fixed with respect to one another as the aiming reticle display moves about on the combining glass. The analog bar represents time-to-go prior to weapon release and appears as a bar of light on the lower half of the 50-milliradian circle. The 50-milliradian circle has fixed indices located at the 3, 4, 5, 6, 7, 8 and 9 o'clock positions, each index denoting 100 seconds when in the 300-second scale, or 10 seconds when in the 30-second scale. The three movable indices of the reticle display are the roll tabs. The roll tabs provide an indication of the roll attitude of the aircraft. Roll tab reference indices are located at 9, 12, and 3 o'clock positions on the 30 milliradian circle. Deviation indicators to the left and right of the reticle rings provide indications of aircraft attitude and speed deviations from preset conditions. The left deviation indicator provides "fly-to" pitch or glide slope deviation indications, depending on the mode selected with the instrument system coupler mode selector knob. When operating in the ILS or AILA mode, the left deviation indicator presents deviation from the glide slope. Full scale represents a glide slope deviation of 0.7 degrees. When operating in any mode other than ILS or AILA, the left deviation indicator presents the pitch deviation from the pitch preset value set into the ODS. Full scale represents a pitch deviation of 10 degrees. The right deviation indicator provides airspeed deviation indications from a preset indicated airspeed value. Each graduation mark represents 10 knots, and a full scale deviation is 30 knots. Low airspeed deviation indications show airspeed below hand-set values, and vice versa. When the bomb nav system is operating in the autonomous mode, the airspeed deviation indications are not reliable. The command steering bars consist of a pitch steering bar and a bank steering bar superimposed over the aiming reticle. The center pipper of the aiming reticle is the zero reference for the steering bars. When the bars are centered, part of the bars are blanked out to provide a window around the pipper. The steering bars function independently of the ODS mode select knob, and provide duplicate indications of the ADI steering bars. Large right azimuth deviations without a corresponding right roll maneuver may cause the bank command bar to drive out of view to the right. Large down pitch steering deviations without a down pitch rate may cause the pitch command bar to drive out of view at the bottom of the sight reticle. Mechanical stops prevent the bars from driving out of view to the left or to the top of the sight reticle. For detailed explanation of the ODS command steering bar indications, refer to "Instruments" this section. The ODS functions in six primary modes of operation.

ODS MODE SELECT KNOB.

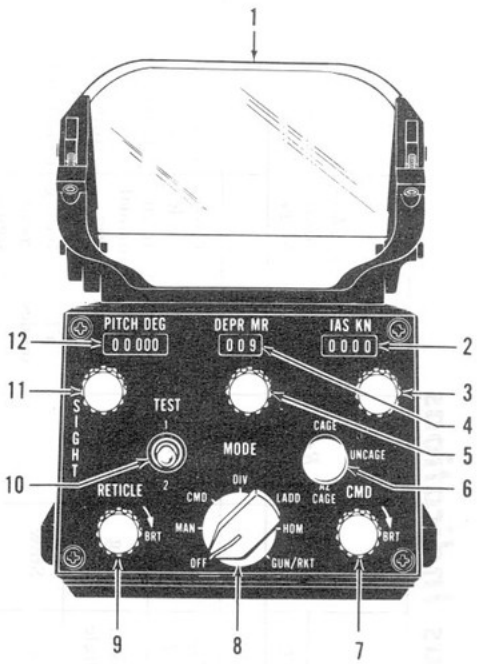
The ODS mode select knob (8, figure 1-80) has 7 positions marked OFF, MAN (manual), CMD (command), DIV (dive), LADD (low angle drogue delivery), HOM (homing), and GUN/RKT (gun/rockets). In the OFF position, all functions of the ODS are inoperative. Refer to figure 1-81 for ODS indications in the various knob positions. Each knob position provides the following indications and functions:

- The MAN (manual) position is used when the pilot desires manual control of the reticle.
- The CMD (command) position is utilized in the ILS or AILA modes and with other systems in four sub modes to provide indications used for radar or visual bombing or combinations of both, updating the bomb-nav system digital computer complex (DCC), and air refueling tanker rendezvous. For bombing, the ODS is utilized with the attack radar in one of the ground modes, the bomb nav system in either radar bomb or visual bomb mode and with the instrument system coupler in bomb nav mode. Indicated airspeed and pitch angle are preselected and set on the ODS control panel. The attack radar can be utilized to locate the target and provide steering bar indications on the ODS to steer the aircraft toward the target. If visual conditions exist, the ODS pipper can also be used to line up on the target. If a radar run is made the ODS analog bar will provide time-to-go indications to bombs away and the weapon will be released automatically on a release signal generated by the bomb nav system on inputs from the attack radar. If

Optical Display Sight Control Panel



1. Optical Sight.
2. Preset Indicated Airspeed Indicator.
3. Indicated Airspeed Set Knob.
4. Reticle Depression Indicator.
5. Reticle Depression Set Knob.
6. Aiming Reticle Cage Lever.
7. Command Bar Brightness Knob.
8. Mode Select Knob.
9. Aiming Reticle Brightness Knob.
10. Test Switch.
11. Pitch Degree Set Knob.
12. Pitch Degree Indicator.



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Figure 1-80.

a visual run is made, the pilot will fly the aircraft so that the target will drive down the reticle toward the pipper. When the target is superimposed under the pipper, a manual release can be made utilizing either weapon release button. When used to update the bomb nav system, the ODS is utilized to make a visual run on a fixpoint similar to making a bomb run except, when the fixpoint is under the pipper, the enter visual fix (EVF) button is depressed to enter the updated information in the bomb nav system. For air refueling rendezvous, the ODS is used in conjunction with the attack radar in the air mode and the instrument system coupler in the tanker rendezvous (TKR RV) mode to provide steering bar indications on the ODS to steer the aircraft to the tanker.

Note

In this configuration, the bank steering bars on the ODS and ADI will oscillate after the attack radar locks-on the target. This is normal and is due to the automatic angle tracking of the radar. The amplitude of the oscillation will increase as the range is decreased; however, the frequency of the oscillation will remain constant.

- The dive mode is inoperative.
- The LADD mode is inoperative.
- The HOM (homing) position is used to aid in visual detection of ground radar targets. The radar homing and warning system (RHAW) will generate signals representative of the angular position of the ground radar. The signals are used by the ODS to position the reticle over the ground radar within the field of view of the ODS sight and the pilot can visually search for the ground radar in the vicinity of the reticle pipper.
- The GUN/RKT (gun/rocket) mode is inoperative.

PITCH DEGREE (PITCH DEG) SET KNOB.

The pitch degree set knob (11, figure 1-80), located on the optical display sight control panel, is used to set in the desired pitch angle. A pitch degree indicator (12, figure 1-80), located directly above the knob, will display the pitch in degrees set in by the knob.

RETICLE DEPRESSION (DEPR MR) SET KNOB.

The reticle depression set knob (5, figure 1-80), located on the optical display sight control panel, is used to set in desired depression angles of the aiming reticle.

ODS Mode Select Knob Positions Versus Indications

Mode Select Knob Positions		Bomb Nav Function Select Knob	Fix Mode Selector Knob	Instrument System Coupler Mode Sel. Knob	Aiming Reticle		Analog Bar	Roll Tabs	Airspeed	Pitch or Glide Slope	Command Bars
					AZ.	EL.					
OFF											
MAN. (Manual Mode)		Any	Any	Any	Cage	Man. Dep.	Stow	Available in all modes	Indicated airspeed deviation from manual set in all modes	Pitch angle from manual set, unless landing mode signal is present then glide slope deviation	Available in all modes to display steering commands
CMD (Command Mode)	Radar Bombing (Sub-mode)	Bomb Radar	Optional	Bomb Nav	CCIP	Fixed Angle	Tg Rel				
	Visual Bombing (Sub-mode)	Visual Bomb	Optional	Optional	Positioned to CCIP solution		Stow				
	Air Refueling (Sub-mode)	Not Bomb	Not Visual Auto	Tanker Rendezvous	Cage	Man. Dep.	Stow				
HOM (Homing Mode)		Any	Any	Any	RHAW	RHAW	Stow				

Figure 1-81.

A reticle depression indicator (4, figure 1-80), located directly above the set knob, indicates in milliradians the reticle depression set by the depression set knob. The maximum depression setting is approximately 200 MR below the fuselage reference line.

INDICATED AIRSPEED (IAS KN) SET KNOB.

An indicated airspeed set knob (3, figure 1-80), located on the optical display sight control panel, is used to set in a desired indicated airspeed. A preset indicated airspeed indicator (2, figure 1-80), located directly above the set knob, indicates the indicated airspeed in knots set in by the indicated airspeed set knob.

TEST SWITCH.

The test switch (10, figure 1-80), located on the optical display sight control panel, is provided to allow an operational check and a fault isolation check to be performed on the ODS while installed in the aircraft without the aid of test equipment. The switch has positions 1 and 2 and is spring-loaded to the center OFF position. Position 1 is used for performing in-flight and ground self tests. Position 2 is used for performing ground fault isolation tests only.

AIMING RETICLE CAGE LEVER.

The aiming reticle cage lever (6, figure 1-80), located on the optical display sight control panel, has three positions labeled CAGE, UNCAGE, and AZ CAGE. In the CAGE position the aiming reticle is mechanically caged to 3 degrees below the fuselage reference line and 0 degrees in azimuth. In the UNCAGE position, the aiming reticle is free to move in azimuth and in elevation. In the AZ CAGE position, the aiming reticle is mechanically caged to 0 degrees in azimuth.

AIMING RETICLE BRIGHTNESS KNOB.

The aiming reticle brightness knob (9, figure 1-80), located on the optical display sight control panel, is provided to adjust the brilliance of the aiming reticle.

COMMAND (CMD) BAR BRIGHTNESS KNOB.

The command bar brightness knob (7, figure 1-80), located on the optical display sight control panel, is provided to adjust brilliance of the command steering bars. Rotating the knob full clockwise will provide full brilliancy. Rotating the knob full counterclockwise will turn off the command steering bars.

PENETRATION AIDS.

Penetration aids control panels are shown in figure 1-82. For further information pertaining to penetration aids, refer to classified supplement T.O. 1F-111(B)A-1-3. Refer to Section II for turn-on procedures and to Section IV for self test and pre-set procedures.

MISCELLANEOUS EQUIPMENT.

THERMAL RADIATION PROTECTION.

Thermal radiation protection for the crew is provided by side curtains on the canopy hatches and hinged forward panels located between the glare shield and windshield.

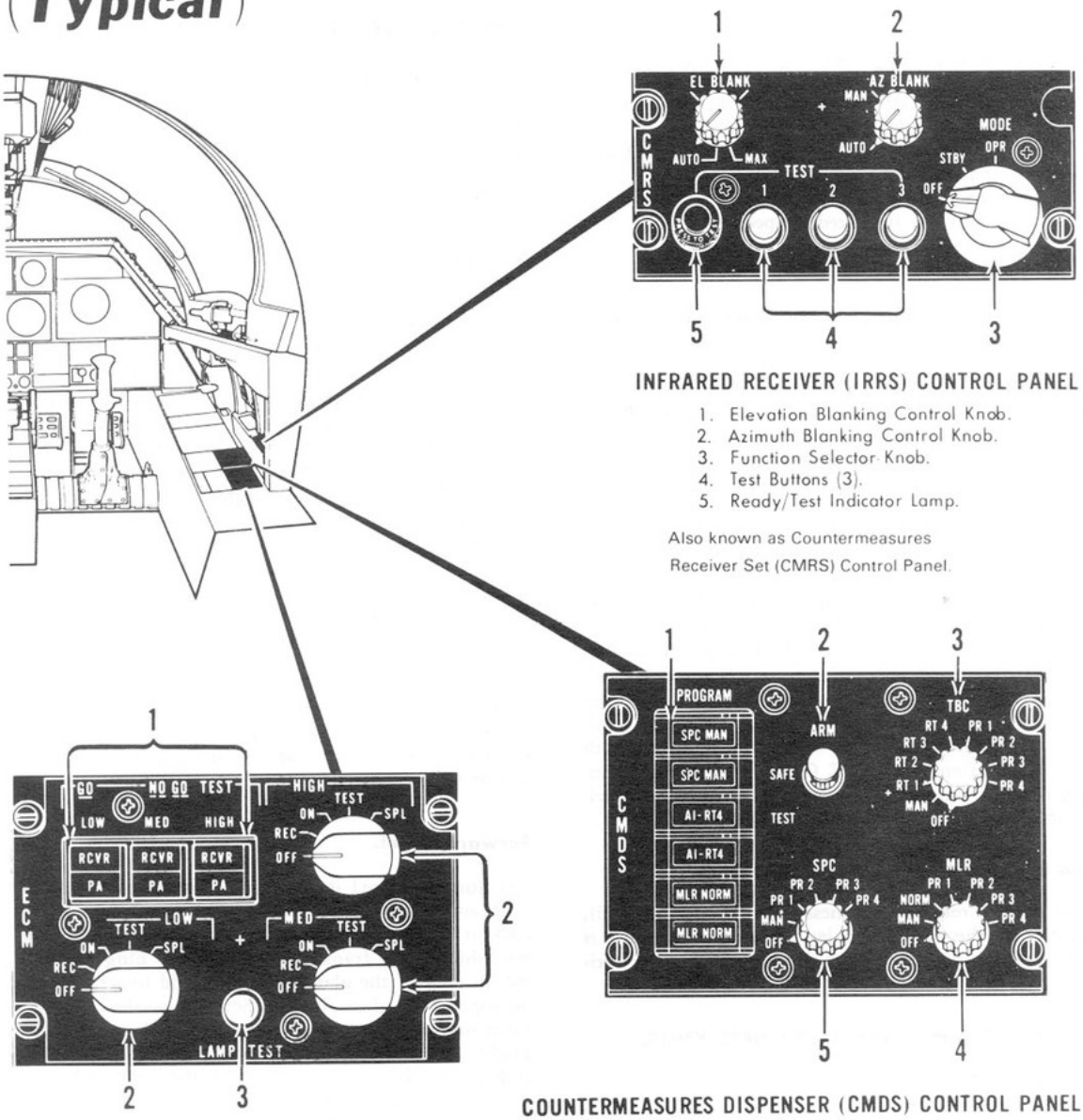
Side Curtains.

The side curtains (11, figure 1-2) are mounted along the upper edge of each canopy hatch on either side of the center canopy beam. When stowed, the curtains are folded as an accordion in the shape of a fan with the hinge forward. As each curtain is extended it unfolds to form an arc from the top rear to the bottom forward edge of the hatch. The rim of the arc rides in a track to form a light seal. When fully extended, the forward edge of the curtain forms a light seal against the forward hatch structure, thus completely covering the canopy hatch glass. The curtain is retained in the stowed position by a two-position magnetic latch to assist in proper operation of the curtain. A handle, labeled RADIATION CURTAIN, is provided on the forward edge of the curtain to extend or retract the curtain. A position latch on the forward seal locks the curtain in the extended position. A pushbutton, labeled CURTAIN RELEASE, must be depressed to release the curtain for retraction. A decal, located adjacent to the curtain release button, contains instructions for extending or stowing the curtain.

Forward Panel.

The forward panel is constructed in sections to form a thermal radiation shield across the front of the cockpit between the top of the glare shield and the windshield arch frame. The panels are hinged along the aft edge of the glare shield and fold forward to lie on top of the glare shield when not needed. A slide catch on each side secures the panels against the glare shield. A cable lanyard attached to each of the two slide catches is provided to unlatch the catch and erect the shields. The right section must be raised first. When erected a friction catch retains the upper edge of each section against the windshield arch to provide a light seal. To stow the panel, each crew member disengages the friction catch by pushing forward on his section adjacent to the catch. When disengaged, the panels will fall forward on the glare shield. The slide catches on each side should be engaged to retain the panel in the stowed position. A decal, located on the forward canopy hatch structure, contains instructions for erecting and stowing the panels.

Penetration Aids Control Panels (Typical)



INFRARED RECEIVER (IRRS) CONTROL PANEL

1. Elevation Blanking Control Knob.
2. Azimuth Blanking Control Knob.
3. Function Selector Knob.
4. Test Buttons (3).
5. Ready/Test Indicator Lamp.

Also known as Countermeasures Receiver Set (CMRS) Control Panel.

ECM CONTROL PANEL

1. Pushbutton Indicator Lamps (3).
2. Control Knobs (3).
3. Lamp Test Button.

COUNTERMEASURES DISPENSER (CMDS) CONTROL PANEL

1. Dispenser Program Module Indicator Lamps (6).
2. CMDS Arming Switch.
3. Track Break Chaff Mode Selector Knob.
4. Missile Launch Response Selector Knob.
5. Special Purpose Chaff Mode Selector Knob.

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Figure 1-82.

CREW ENTRANCE LADDERS AND STEPS.

Crew entrance ladders and steps, located on each side of the fuselage, provide crew access to the cockpit without the aid of ground support equipment. When not in use, both sets of ladders and steps are retracted into the sides of the fuselage. Each left or right ladder and step can be electrically extended from inside the cockpit. Slotted head releases are provided on the outside of the fuselage to manually extend the ladders and steps from the ground. The ladders and steps must be manually stowed from the ground.

Entrance Ladder Switch.

The entrance ladder switch (4, figure 1-27), located on the ground check panel, is labeled LADDER and has three positions marked L (left), R (right) and OFF. Placing the switch to L or R will provide 28 volt dc power to solenoids in the respective ladder and step to release and extend the ladder and step. The switch is spring-loaded to the center OFF position.

MIRRORS.

Four rear view mirrors, two on each side of the cockpit canopy frame (9, figure 1-2) are installed to permit the crew rearward vision without moving from their normal sitting position. The mirrors are adjustable in tilt only.

MISSION DATA BOX STOWAGE.

A mission data box (38, figure 1-2) is located at the aft end of each left and right console at the bottom of the aft bulkhead. The boxes are provided to stow classified mission data in the aircraft for each crew member. Each box has provisions for a combination lock to secure the lid shut and a lanyard from the box to the aircraft. The boxes are secured in their mounting brackets by sliding type fasteners. The lid on each box is spring-loaded to the open position. Provisions are available on the box for a security seal to be installed when required.

CHART STOWAGE.

Two chart stowage compartments are located on each side of the lighting control panel (46 and 52, figure 1-2). The left compartment is labeled LETDOWN CHART HOLDER; the right compartment is labeled LETDOWN CHARTS. Each compartment is provided with a strap and fastener to secure the charts and holder.

SAFETY PIN STOWAGE.

A stowage compartment (53, figure 1-2), located on the aft bulkhead outboard of the left seat, is provided to stow the safety pins.

SPARE LAMP AND FUSE HOLDER STOWAGE.

A stowage compartment (44, figure 1-2), located on the aft bulkhead outboard of the right seat, is provided for stowing spare light bulbs and fuses.

FOOD AND LIQUID STOWAGE.

A food and liquid stowage compartment (42, figure 1-2) for each crew member is located on the aft bulkhead outboard of each crew member's seat. The stowage is in two sections, one for food, the other for liquid. The door on the food section is spring-loaded to the open position and is closed with a lever type fastener. The liquid bottle is secured below the food compartment with sliding fasteners. A valve located on the bottom of the bottle dispenses a given amount of liquid when the cup is properly engaged and depressed. The stowage compartment may be rotated out of the way when not in use by activating the unlock lever at the front and swinging the whole unit aft and up about a pivot point until it latches in the up position. When use of the food and liquid is desired, grasp the latch and pull down. This will unlatch the stowage compartments, and further pulling on the latch handle will rotate the compartment to a using position.

DRINKING CUPS.

Drinking cups (40, figure 1-2), located on the left and right sidewall are provided for each crew member. The cups are stowed in a spring clip when not in use. The cups are equipped with an anti-splash baffle to reduce accidental spilling.

ASH TRAYS.

An ash tray (7, figure 1-25 and 2, figure 1-42), located on the left and right sidewall, is provided for each crew member. Each ash tray may be opened by pulling a handle at the top and closed by pushing the handle outboard.

RELIEF CONTAINER STOWAGE.

A relief container (39, figure 1-2), located just aft of the left and right consoles, is provided for each crew member. Each container lid is held shut with a lever type latch. A bracket inside each lid holds four relief bags.

MANUALS STOWAGE.

Manuals stowage (41 and 56, figure 1-2) is provided on the aft bulkhead outboard of the left and right seats. The manuals are stowed in a bracket and held in place by lids which are spring-loaded to the open position. To close the lids, push aft until they come in contact with a lever latch. Lifting up on the latch releases the lids which open automatically.

IFR CHART STOWAGE.

The IFR chart stowage (57, figure 1-2) is located at the aft end of the left console, outboard of left seat. The charts are stowed and removed by pushing a spring-loaded keeper to one side, which springs back in position when released. This stowage box is mounted in the aircraft with slide fasteners and must be removed to gain access for removal of the mission data box.

HOOD STOWAGE COMPARTMENT.

A hood stowage compartment (51, figure 1-2) is located on the left seat above the headrest and provided to store the attack radar scope hood.

CORRECTION CARD HOLDERS.

Card holders are provided under the glare shield for EPR setting, compass correction and UHF frequency channels. Each card holder is attached by spring tensioned hinges riveted to the glare shield. The card holders are pulled out into position for reading purposes and spring back against the lower side of glare shield when released.

CHART HOLDER.

A lighted chart holder is provided to clearly display approach charts where they can be easily followed during instrument letdowns. The holder is a rectangular transparent pane the size of an approach chart and is attached in a swivel socket on the canopy center beam. It can be swivelled to the left or right and latched in place for use by either crew member. The holder has both red and white lighting which can be mixed as desired by control knobs located on the top of the holder. The holder is stowed in a receptacle (52, figure 1-2), located in the aft console, when not in use.

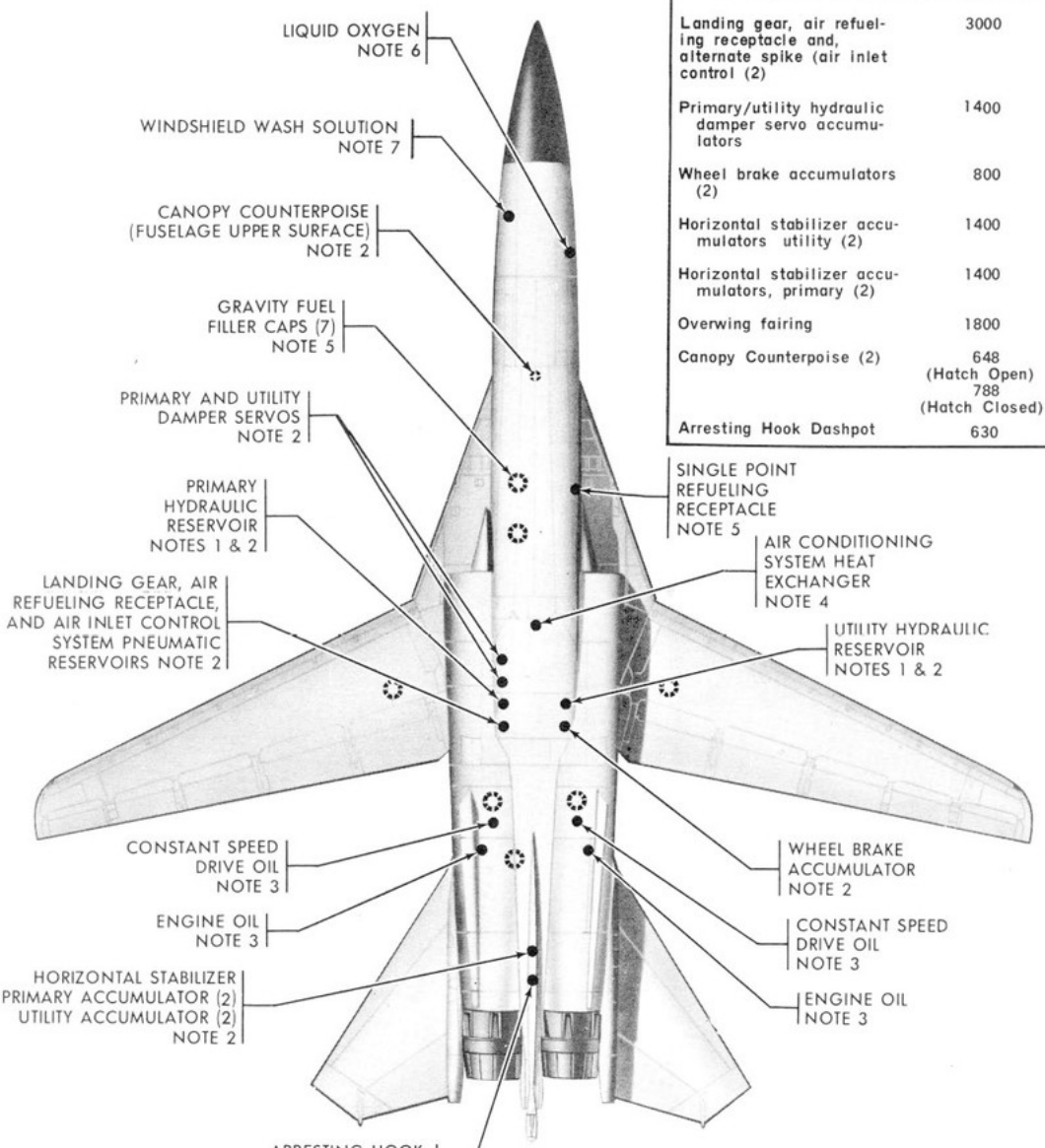
STARTER CARTRIDGE STOWAGE CONTAINER.

A starter cartridge stowage container, located on the left forward side of the main landing gear wheel well, is provided to carry two spare starter cartridges. The container is made of plastic and has a detachable cover to allow servicing or access to the spare cartridges when needed.

Servicing Diagram

PNEUMATIC SERVICING REQUIREMENTS TABLE

SYSTEM or COMPONENTS	PRESSURE (PSIG) (AT 70°F, 21°C)
Landing gear, air refueling receptacle and, alternate spike (air inlet control (2)	3000
Primary/utility hydraulic damper servo accumulators	1400
Wheel brake accumulators (2)	800
Horizontal stabilizer accumulators utility (2)	1400
Horizontal stabilizer accumulators, primary (2)	1400
Overwing fairing	1800
Canopy Counterpoise (2)	648 (Hatch Open) 788 (Hatch Closed)
Arresting Hook Dashpot	630



NOTES:

1. Hydraulic Oil – MIL-H-5606
2. Nitrogen-FS BB-N-411, Type 1, Class 1, Grade B or Air – MIL-P-5518
3. Oil – MIL-L-7808
4. Demineralized Water – MIL-D-83132 (Tap Water when MIL-D-83132 is not available. See T.O. 1F-111(B)A-2-15-1 for restrictions.)
5. Fuel – MIL-T-5624 (JP-4)
6. Liquid Oxygen – MIL-O-27210, Grade B, Type II
7. Wash Solution – Refer to T.O. 1F-111(B)A-2-1.

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Figure 1-83.