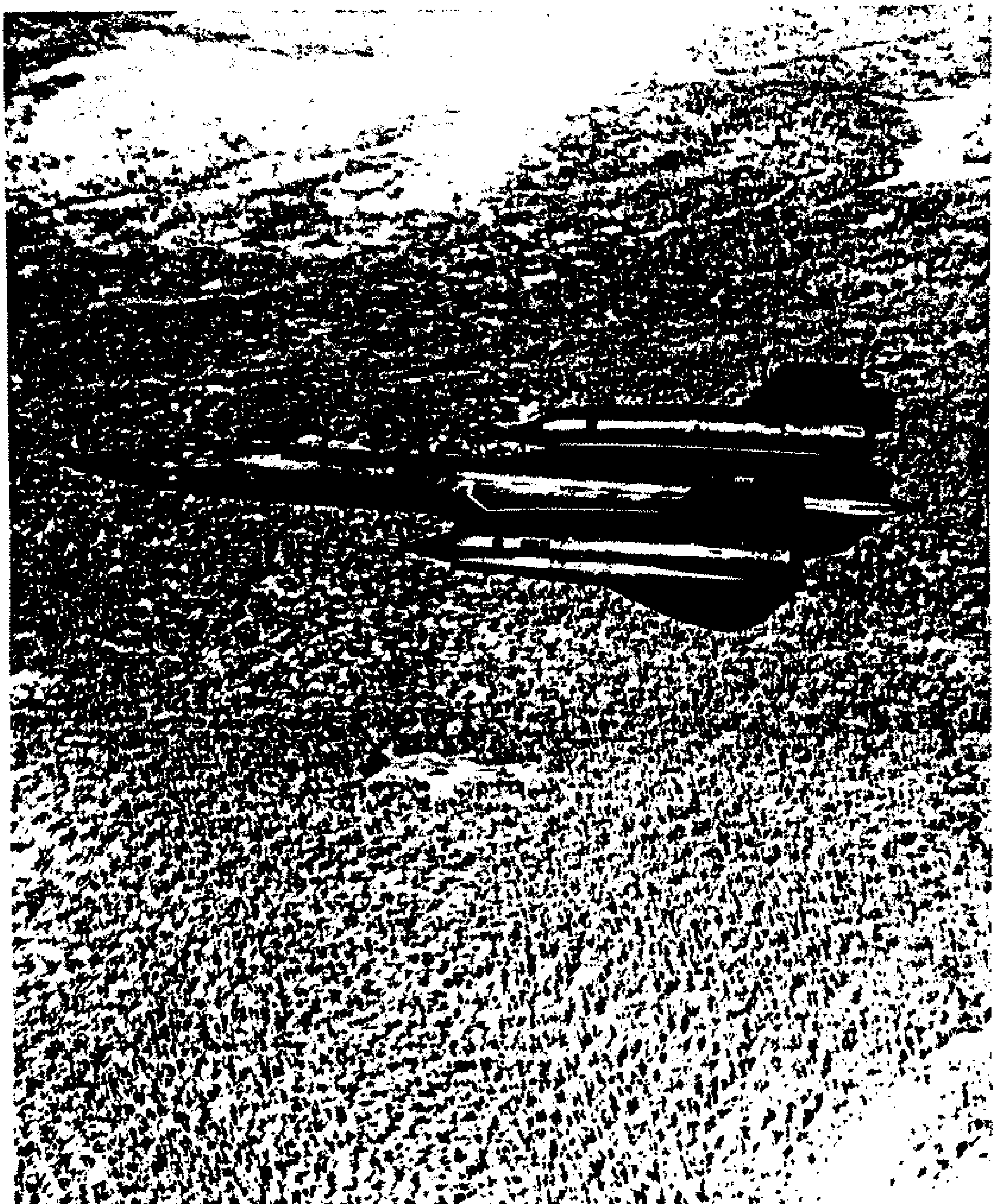


SR-71A

INTRODUCTION

This section describes SR-71A reconnaissance aircraft. Subsection 1A describes the SR-71B.

THE AIRCRAFT

The SR-71 is a delta-wing, two-place aircraft powered by two axial-flow turbojet engines. The aircraft, built by the Lockheed California Company, features titanium construction and is designed to operate at high altitudes and high supersonic speeds. The aircraft has very thin wings, twin canted rudders mounted on top of the engine nacelles, and a pronounced fuselage "chine" extending from the nose to the leading edge of the wing. The propulsion system uses movable spikes to vary air inlet geometry. Surface controls are elevons and rudders, operated by irreversible hydraulic actuators with artificial pilot control feel. The aircraft can be refueled either in-flight or on the ground through separate receptacles that feed into a common refueling line. A drag chute is provided to augment the six-mainwheel brakes. The aircraft is painted black to reduce internal temperatures when at high speed.

Dimensions

Length (overall)	107.4	ft.
Height (to top of rudders)	18.5	ft.
Wing span	55.6	ft.
Wing area (reference)	1605 sq.	ft.
Tread (MLG middle wheel centerlines)	16.67	ft.

Gross Weight

The loaded gross weight of the aircraft varies from approximately 135,000 to over 140,000 pounds. Zero fuel weight varies from 56,500 to more than 60,000 pounds.

NOTE

Use SR-71 manual of weight and balance data applicable to specific aircraft to compute aircraft performance.

FUEL

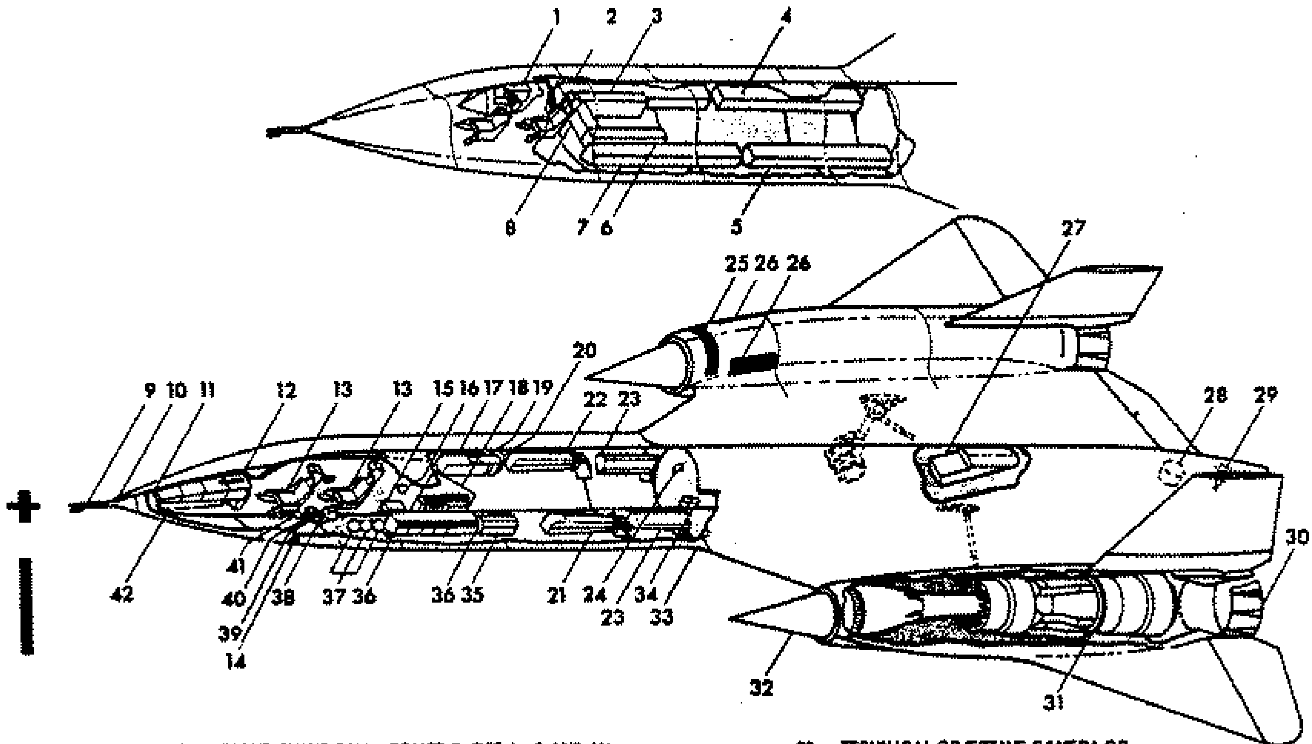
The operating envelope of the JT11D-20 engine requires special fuel. The fuel is not only the source of energy but is also used in the engine hydraulic system. During high Mach flight, the fuel is also a heat sink for the various aircraft and engine accessories which would otherwise overheat at the high temperatures encountered. This requires a fuel having high thermal stability so that it will not break down and deposit coke and varnishes in the fuel system passages. A high luminometer number (brightness of flame index) is required to minimize transfer of heat to the burner parts. Other items are also significant, such as the amount of sulfur impurities tolerated. Advanced fuels, JP-7 (PWA 535) and PWA 523E, were developed to meet the above requirements.

JP-7 and PWA 523E contain one gallon of PWA 536 lubricity additive per 5200 gallons of fuel to insure adequate lubrication of fuel hydraulic pumps.

ENGINE AND AFTERBURNER

Thrust is supplied by two Pratt & Whitney JT11D-20 bleed bypass turbojet engines with afterburners. (See Figure 1-2.) The engines are designed for continuous operation at compressor inlet temperatures above 400°C, which are associated with high Mach flight. The engine has a single-rotor, nine-stage, 8.8:1 pressure ratio compressor utilizing a compressor bleed bypass at high Mach. When opened, bypass valves bleed air from the fourth stage of the compressor, and six ducts route it around the rear stages of

GENERAL ARRANGEMENT AND BAY LOCATOR DIAGRAM



1 RIGHT CHINE BAY - COMPT D (DEF A, C AND M)
2 RIGHT FORWARD MISSION BAY - COMPT L AND N
3 RADIO EQUIPMENT BAY - COMPT R
4 RIGHT AFT MISSION BAY - COMPT Q AND T
5 LEFT AFT MISSION BAY - COMPT P AND S
6 ELECTRONICS BAY - COMPT E
7 LEFT FORWARD MISSION BAY - COMPT K AND M
8 CAMERA BAY - COMPT C
9 PITOT MAST
10 HF ANTENNA
11 LOCALIZER ANTENNA
12 RADAR OR OBC EQUIPMENT - COMPT A
13 EJECTION SEAT
14 FORWARD UHF ANTENNA (LEFT SIDE)
15 ANS PLATFORM AND COMPUTER
16 IFF ANTENNA
17 RADAR RECORDER
18 ELECTRICAL LOAD CENTER
19 AIR REFUELING RECEPTACLE
20 MISSION RECORDERS
21 TECHNICAL OBJECTIVE CAMERA
22 TECHNICAL OBJECTIVE CAMERA OR RADAR RECORDER
23 EIP
24 AFT UHF ANTENNA (RIGHT SIDE)
25 FORWARD BYPASS DOORS
26 POROUS BLEED AIR OUTLETS
27 DRAG CHUTE RECEPTACLE
28 ROLL AND PITCH MIXER
29 CW RECEIVE ANTENNA (DEF H)
30 EJECTOR FLAPS
31 J-58 ENGINE
32 MOVABLE SPIKE
33 VHF ANTENNA (LEFT SIDE)
34 SAS GYROS
35 DIGITAL AND AR1700 RECORDERS (EIP)
36 DEF H
37 LIQUID OXYGEN CONTAINERS
38 TACAN ANTENNA
39 DEF H CENTERLINE RECEIVE ANTENNA
40 UHF-ADF ANTENNA
41 GLIDE SLOPE ANTENNA
42 SLR ANTENNA

SEE SECTION IV FOR COMPLETE LIST OF BAY DESIGNATIONS AND EQUIPMENT LOCATIONS

F203-4(m)

Figure 1-1

JT11D -20 ENGINE

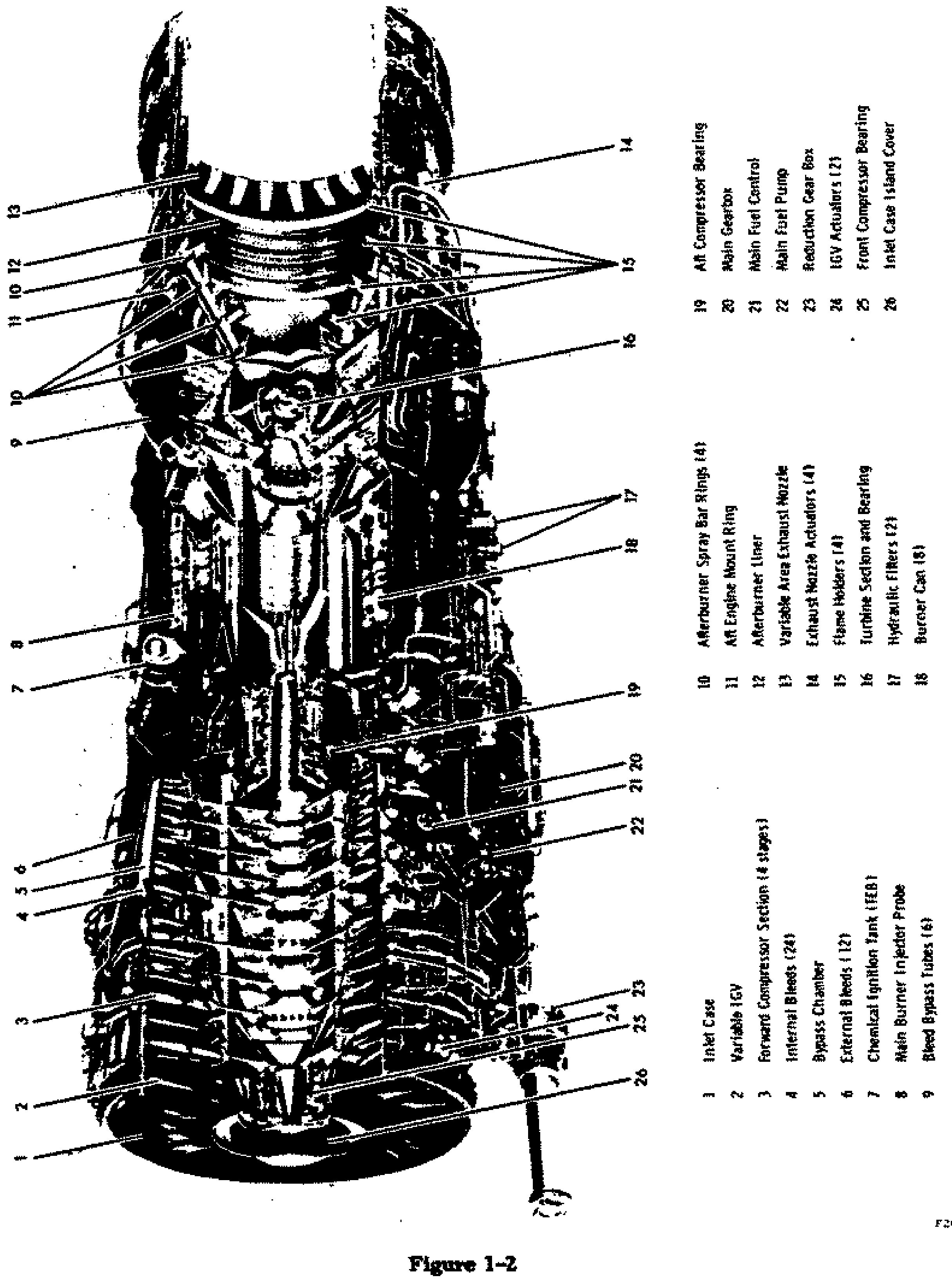


Figure 1-2

the compressor, the combustion section, and the turbine. The bleed air re-enters the turbine exhaust around the front of the afterburner where it is used for increased thrust and cooling. The transition to bypass operation is scheduled by the main fuel control as a function of compressor inlet temperature and engine speed. The transition normally occurs in a CIT range of 85° to 115°C, corresponding to a Mach range of 1.8 to 2.0.

When on the ground, or at low Mach numbers, engine speed varies with throttle movement when the throttle is between IDLE and slightly below the Military stop. At higher settings, up to maximum afterburner, the main fuel control schedules engine speed as a function of CIT and modulates the variable area exhaust nozzle to maintain approximately constant rpm. Throttle movement in the afterburning range only changes the afterburner fuel flow, nozzle position, and thrust. At high Mach number and constant inlet conditions, engine speed is essentially constant for all throttle positions down to and including IDLE. At a fixed throttle position, engine speed will vary according to main fuel control schedule when CIT (Mach) changes.

The engine contains a two-stage turbine. Turbine discharge temperatures are monitored by exhaust gas temperature indication. A chemical ignition system is used to ignite the low vapor pressure fuel. A separate engine-driven hydraulic system, using fuel as hydraulic fluid, operates the exhaust nozzle, chemical ignition system dump, compressor bypass, starting bleed systems, and Inlet Guide Vanes (IGV). The main fuel pump, engine hydraulic pump and tachometer are driven by the main engine gearbox. The afterburner fuel pump is powered by an air turbine, driven by compressor discharge air.

<u>Maximum Rated Thrust</u>

Maximum rated thrust is obtained in afterburning by placing the throttle against the quadrant forward stop. The maximum afterburning <u>uninstalled</u> thrust of each engine at sea level, static condition, and standard day is 34,000 pounds. Takeoff thrust in maximum afterburner is illustrated in Figure 1-3 at sea level pressure altitude. It shows the variation in thrust with ambient temperature and the effect of airspeed during the takeoff acceleration.

<u>Partial Afterburning Thrust</u>

Afterburning fuel flow and thrust are modulated by moving the throttle between the Military detent and the quadrant forward stop. Minimum afterburning thrust is obtained with the throttle just forward of Military and is approximately 85% of maximum afterburning thrust at sea level and

TAKEOFF THRUST

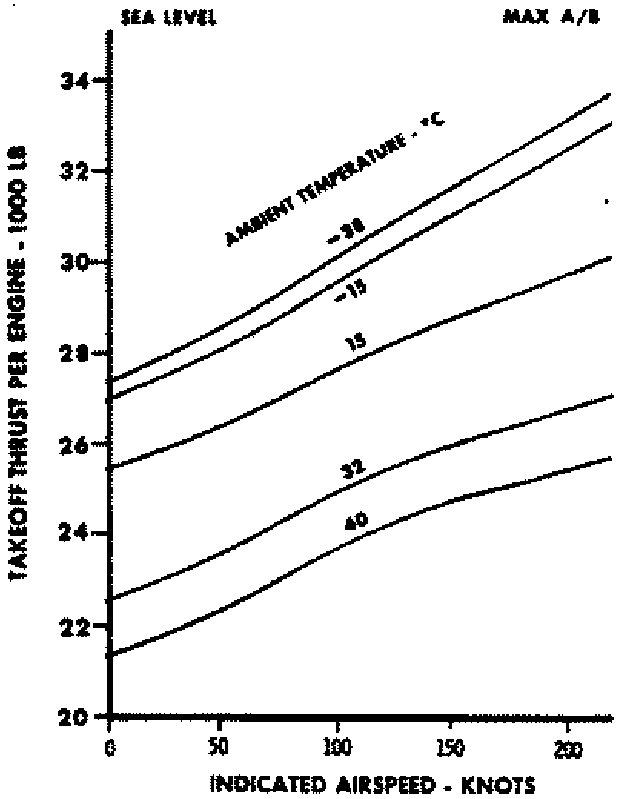


Figure 1-3

approximately 55% at high altitude. Afterburner ignition is automatically actuated when the throttle is advanced past the detent. The time required to obtain afterburner ignition after moving the throttle past the detent is a function of afterburner fuel manifold fill time. The fill time can be up to three seconds at sea level and up to seven seconds at altitude. Afterburner fuel flow is terminated when the throttle is retarded below the detent. The basic engine operates at Military rated thrust during all afterburning operation.

MILITARY THRUST

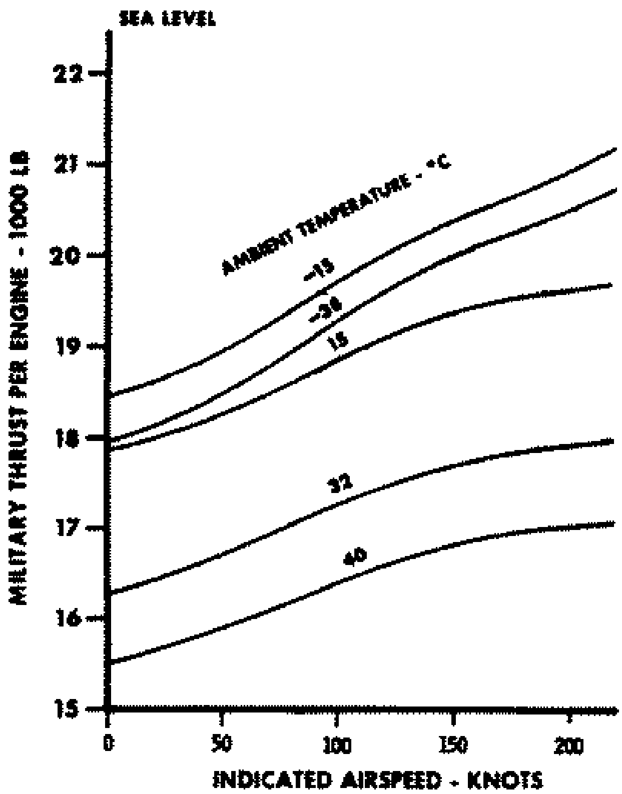


Figure 1-4

<u>Military Thrust</u>

Military thrust is the maximum non-afterburning thrust and is obtained by placing the throttle against the aft side of the Military detent. At sea level static conditions, military thrust is approximately 70% of maximum thrust. At high altitude, military thrust is approximately 28% of the maximum thrust available. Figure 1-4 illustrates the variation in military thrust with ambient temperature and airspeed at sea level pressure altitude.

<u>Idle Thrust</u>

Idle thrust is the minimum non-afterburning thrust level. With the throttle in IDLE, the engine operates at approximately 3975 rpm up to 60°C (140°F). At higher ambient temperatures, rpm increases approximately 50 rpm per 1°C. Idle thrust is illustrated in Figure 1-5, at sea level pressure altitude, for airspeeds typical of a landing and deceleration.

IDLE THRUST

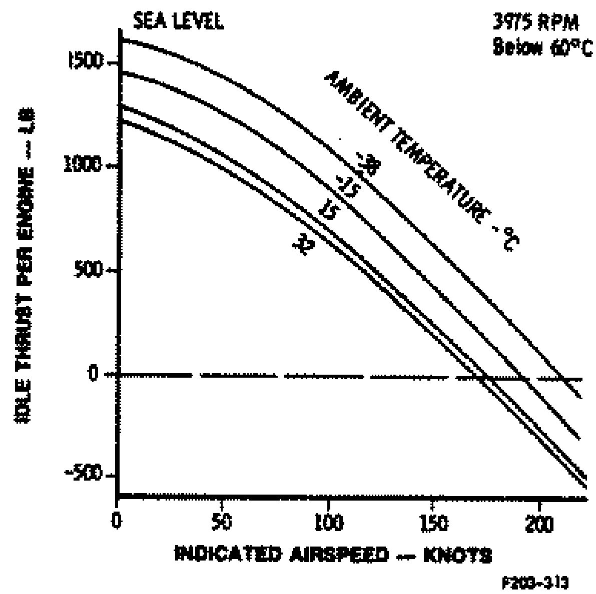


Figure 1-5

THROTTLES AND THROTTLE SETTINGS

Two throttle levers are located in a quadrant on the pilot's left forward console. The right throttle is mechanically linked to the right engine main fuel control. The left throttle is linked to the left engine afterburner fuel control. The afterburner and the main fuel controls are interconnected by a closed loop cable. The throttle quadrant has three labeled positions, OFF, IDLE, and AFTERBURNER, and an unlabeled Military power stop. See Figure 1-6. The non-afterburning operating range of the engine is between IDLE and Military.

A spring-loaded pushbutton switch is located on the right throttle knob. When depressed, the pilot's microphone is connected to the selected radio transmitter.

A throttle air inlet restart switch is located on the inboard side of the right throttle. The switch is used for restarting both air inlets simultaneously.

Off

In the OFF position, the windmill bypass valve cuts off fuel from the burner cans and routes it back to the aircraft system. This provides cooling for engine oil, fuel pump, and fuel hydraulic pump when an engine is windmilling.

Idle

When a throttle is moved forward from OFF to IDLE, a roller drops over a hidden ledge at the IDLE position. This ledge prevents the engine from being inadvertently cut off when the throttles are retarded to IDLE. The throttles must be lifted to be moved from IDLE to OFF.

Afterburner

The throttle must be slightly raised and pushed forward to clear the Military stop before additional forward movement of the throttle can initiate afterburner ignition. The AFTERBURNER range extends from the Military stop to the quadrant forward stop.

Start

There is no distinct throttle position for starting. Starting is accomplished by moving the throttle from OFF to IDLE as the engine is accelerated by the starter. As the proper engine speed is reached, fuel is directed to the engine burners by actuation of the windmill bypass valve and the chemical ignition system is actuated by fuel pressure.

Throttle Friction Lever

Throttle friction is controlled by a lever located on the inboard side of the throttle quadrant. Moving the lever forward, as the INCREASE FRICTION label indicates, progressively increases friction.

TEB Remaining Counters

Mechanical digital counters, aft of each throttle,indicate the number of TEB shots remaining for each engine. The counters are spring wound and set to 16 prior to engine start. Each time a throttle is moved forward from OFF to IDLE, or from Military to AFTERBURNER, the corresponding counter indication decreases by 1.

Tachometers

Tachometers for each engine are mounted on the right side of the pilot's instrument panel. They indicate engine speed in revolutions per minute by means of a main pointer and dial calibrated to 10,000 rpm, and a smaller dial and subpointer which make one complete revolution for each 1000 rpm. The tachometers are self-energized and operate independently of the aircraft electrical system.

ENGINE FUEL SYSTEM

Engine fuel system components include the engine-driven fuel pump, main fuel control,

THROTTLE QUADRANT

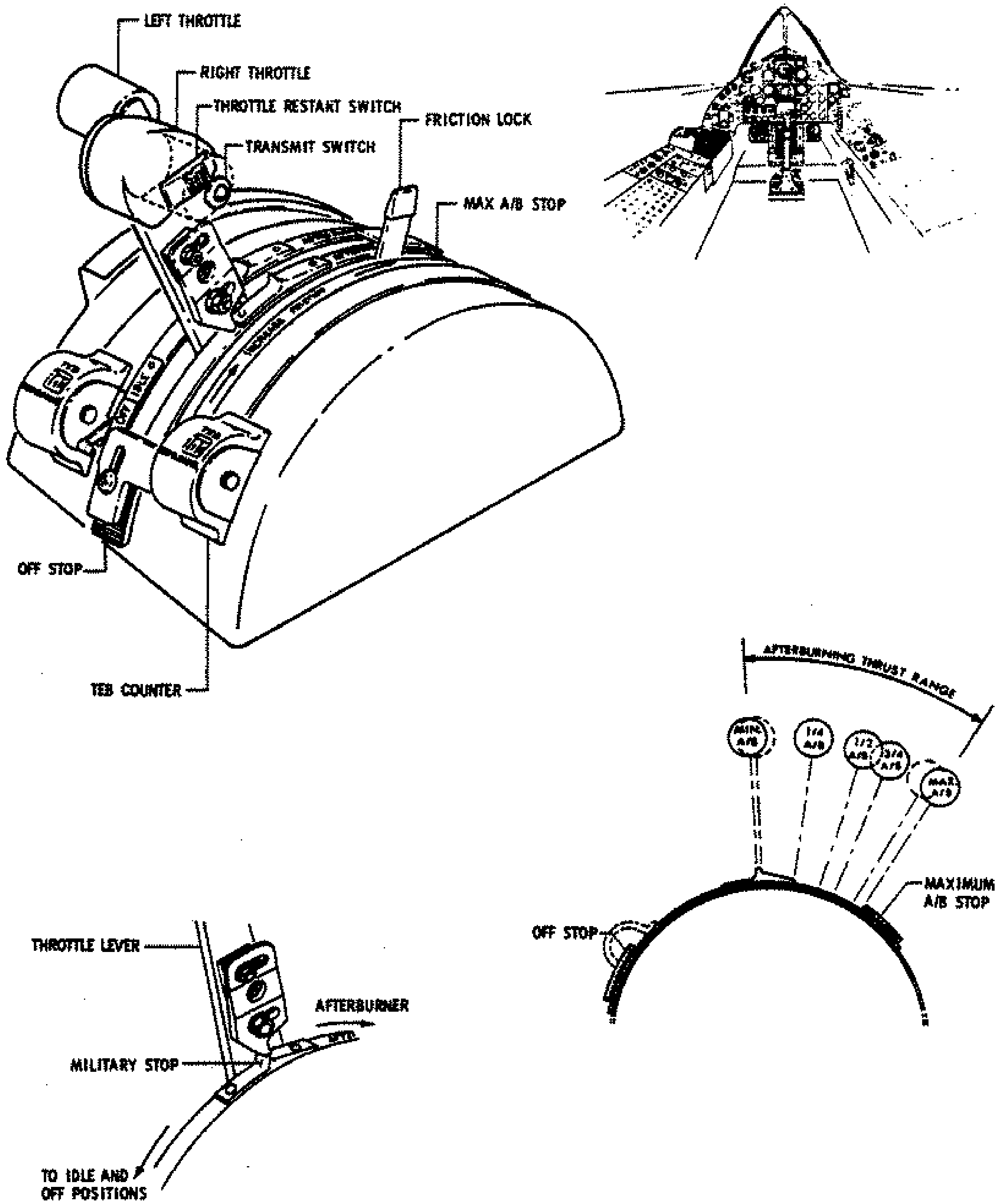


Figure 1-6

windmill bypass valve and variable area fuel nozzles in the main burner section. (See Figure 1-7.)

Main Fuel Pump

The engine-driven main fuel pump is a two-stage unit. The first stage, a single centrifugal pump, acts as a boost stage. The second stage consists of two parallel gear-type pumps with discharge check valves. The parallel pump and check valve arrangement permits continued operation if either pump fails.

Main Fuel Control

The main fuel control meters main burner fuel flow, controls the bleed bypass, start bleed valves, IGV, and exhaust nozzle modulation. It regulates main engine thrust as a function of throttle position, compressor inlet air temperature, main burner pressure, and engine speed. The bypass, start-bleed valve positions and IGV are controlled as a function of engine speed, biased by CIT. Afterburner operation is always at Military-rated engine speed and EGT. The control has a remote trimmer for EGT regulation. There is no emergency fuel control system.

Windmill Bypass and Dump Valve

The windmill bypass and dump valve directs fuel to the engine burners for normal operation or bypasses fuel to the recirculation system for accessory, engine component, and engine oil cooling during windmilling operation. The valve responds to signals from the main fuel control. The valve opens to drain the engine fuel manifold when the engine is shut down.

Fuel Nozzles

The engine has an eight-unit can-annular combustion section with 48 variable-area, dual-orifice fuel nozzles. The nozzles are arranged in clusters of six nozzles per burner. Each nozzle has a fixed area primary metering orifice and a variable area secondary metering orifice, discharging through a common opening. The secondary orifice opens as a function of primary pressure drop.

Combustion Chamber Drain Valve

The main engine ignition system plumbing is equipped with a fuel purge or "Dribble Tee". This allows fuel from the main fuel pump interstage to flush residual ignition fluid (TEB) from the ignition probe. It prevents "coking" from occurring which would restrict the ignition probe and prevent engine ignition. Hence, fuel in small quantity should drain from the main burner case overboard drain fitting anytime there is fuel pressure to the engine pump inlet, due to fuel boost pump operation or tank pressure developed by the LN_2 fuel tank pressurization system. If fuel does not drain normally, either the chemical ignition system probe is plugged or the burner drain has malfunctioned. The normal leakage from the main burner case overboard drain should be confirmed before start. If the overboard drain is restricted, it will increase the "wetted" fuel area in the burner and could result in severe torching during engine start.

Fuel Flow Indicators

Fuel flow indicators for each engine, mounted on the right side of the pilot's instrument panel, display total fuel flow (engine and afterburner) plus tank return flow, if any. Dial calibrations are provided in 5000 pound per hour increments to 95,000 pph. Five center digit windows show fuel flow to the nearest 100 pph. Power for the indicators is supplied from the essential ac bus through the L and R FLOW circuit breakers on the pilot's right console.

ENGINE AND AFTERBURNER FUEL SYSTEM

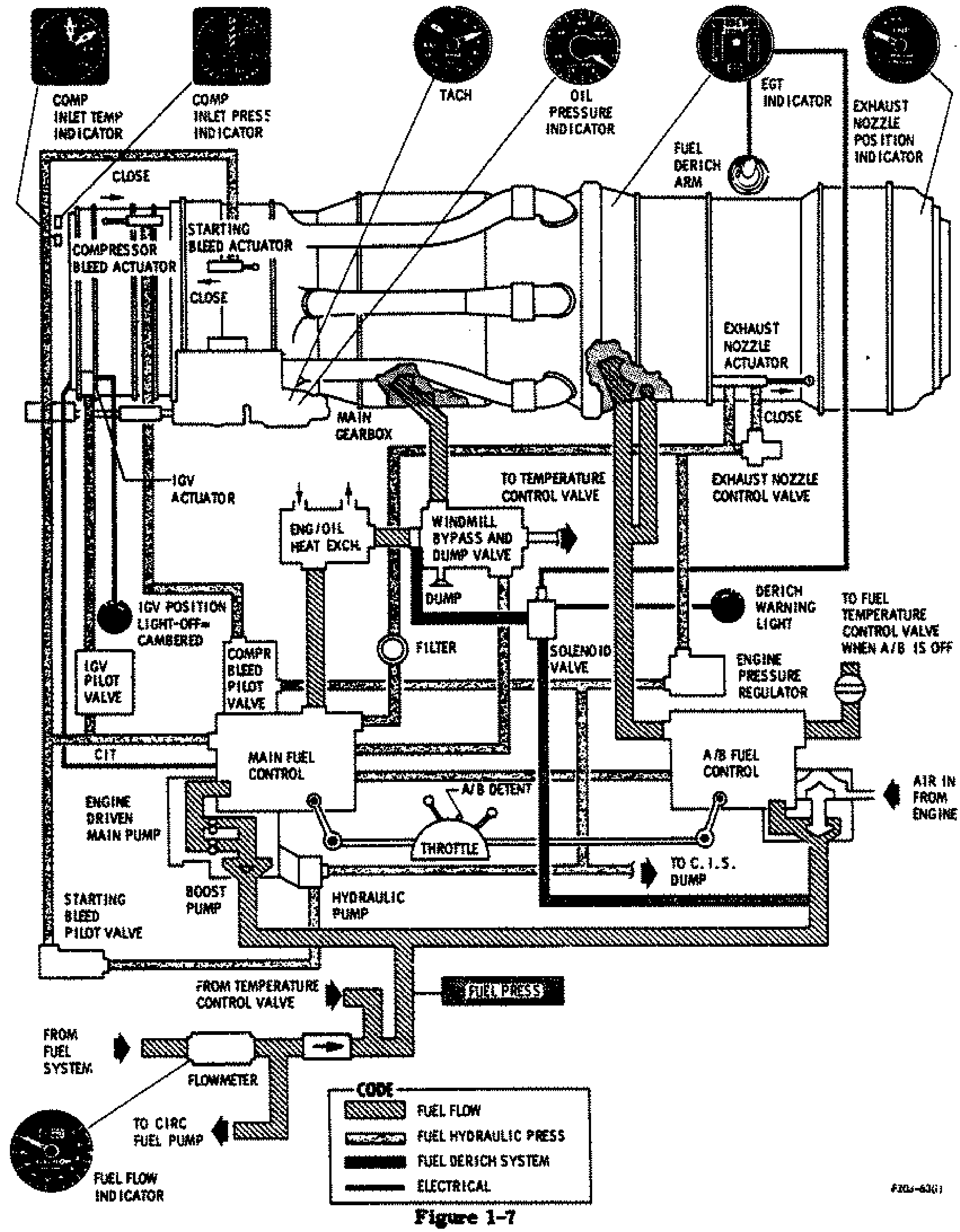


Figure 1-7

NOTE

Tank return flow forms an appreciable portion of the indication when at or below Military.

AFTERBURNER FUEL SYSTEM

Afterburner Fuel Pump

The afterburner fuel pump is a high speed, single stage centrifugal pump. The pump is driven by an air turbine, operated by engine compressor discharge air. The compressor discharge air supply is regulated by a butterfly valve in response to the demand of the afterburner fuel control.

Afterburner Fuel Control

The hydromechanical afterburner fuel control schedules fuel flow as a function of throttle position, main burner pressure, and compressor inlet temperature. Fuel flow is metered to discharge fuel from the four concentric afterburner spraybar rings.

FUEL DERICH SYSTEM

A derichment system on each engine protects against severe turbine overtemperature. If the respective EGT indicator reaches 860°C while the system is armed, the fuel/air ratio in the engine burner cans is automatically reduced (deriched). A signal from the EGT gage actuates a solenoid-operated valve which bypasses metered engine fuel from the fuel/oil cooler to the afterburner fuel pump inlet. See Figures 1-7 and 1-8. Once actuated, the solenoid valve is held open until the system is turned off or rearmed. A red, flashing FUEL DERICH warning light illuminates when the valve for the respective engine is open.

Derichment at sea level decreases thrust approximately 5% in maximum afterburner; 7% at Military. Derichment during supersonic cruise decreases engine thrust approximately 45% in maximum afterburner (overall engine/inlet thrust loss is about 10% for the deriched engine since derichment has no effect on CIP) and may cause the afterburner to blow out. Continuous operation with the engine deriched does not harm the engine (provided derichment can reduce EGT to normal limits).

After an unstart, do not move the fuel derich switch from ARM unless inlet roughness has cleared and the inlet has restarted (CIP recovered); otherwise, severe overtemperature can result. Do not attempt to relight the afterburner while deriched. Lighting the afterburner while deriched can result in engine speed suppression of up to 750 rpm.

Fuel Derich Arming Switch

A three-position FUEL DERICH arming toggle switch is located on the pilot's left instrument side panel. In the ARM (center) position, the derich circuit is armed and the derich solenoid valve will open and remain open if the respective EGT indication reaches 860°C. In the OFF (down) position, the derich solenoid valve is closed and the system cannot provide derichment. The REARM (up) position is spring-loaded and allows the pilot to rearm the fuel derich system without moving the switch to OFF. Power for the switch is furnished by the essential dc bus through the L and R FUEL DERICH circuit breakers on the pilot's left console.

Fuel Derich System Test Switch

A three-position toggle switch, labeled FUEL DERICH SYSTEM, is located on the pilot's left console. The switch is labeled L (left), R (right) and is spring-loaded to the center (off) position. When the switch is moved to the L or R position, the digital indication on the respective EGT gage slews toward 1198°C. When the EGT gage indication exceeds 860°C, the red jewel light in the gage illuminates; and, if the FUEL DERICH switch is in ARM, the respective fuel derich warning light flashes and the derich solenoid valve opens.

Fuel Derich Warning Light

Two red, flashing fuel derich warning lights are located on the right side of the pilot's instrument panel. A light flashes when the fuel derich system for the respective engine is activated (derich solenoid valve open) and continues flashing until the fuel derich system is rearmed or off.

EXHAUST GAS TEMPERATURE (EGT) TRIM SYSTEM

EGT Gages

Two EGT gages, one for each engine, are located on the right side of the pilot's instrument panel. Each gage has a digital indicator that shows turbine discharge temperature from 0° to 1198°C, a HOT (red) and COLD (yellow) condition flag, a red "jewel" overtemperature warning light that illuminates when the EGT digital indication reaches 860°C, and a power OFF warning flag. Each indicator receives power from the essential ac bus through its respective L or R EGT IND circuit breaker on the pilot's right console.

EGT Trim Switches

Two four-position EGT trim switches, one for each engine, are located on the pilot's left console. The positions are labeled AUTO (left), INCR (up), DECR (down) and HOLD (center). When a switch is held in INCR, a small electric motor on the engine fuel control increases the ratio of main burner fuel flow to main burner combustion pressure and thus increases the turbine discharge temperature (EGT). Holding a switch in DECR runs the motor in the opposite direction and decreases EGT. The switches have no effect on rpm as long as the nozzles are modulating to provide the scheduled engine speed. However, engine speed will increase (or decrease) with increasing (or decreasing) EGT when nozzle position is limited full closed (or open).

An EGT "permission" circuit prevents automatic trimming in either direction and manual uptrim when the respective throttle is positioned below Military or the engine is deriched. Manual EGT downtrim remains available even when the permission circuit is on (throttle below Military or engine deriched). Power for the trim motors is furnished by the essential ac bus through the L and R EGT TRIM circuit breakers on the pilot's right console. Power for the permission circuit is furnished by the essential dc bus through the L and R EGT circuit breakers on the pilot's left console.

Automatic EGT Trim

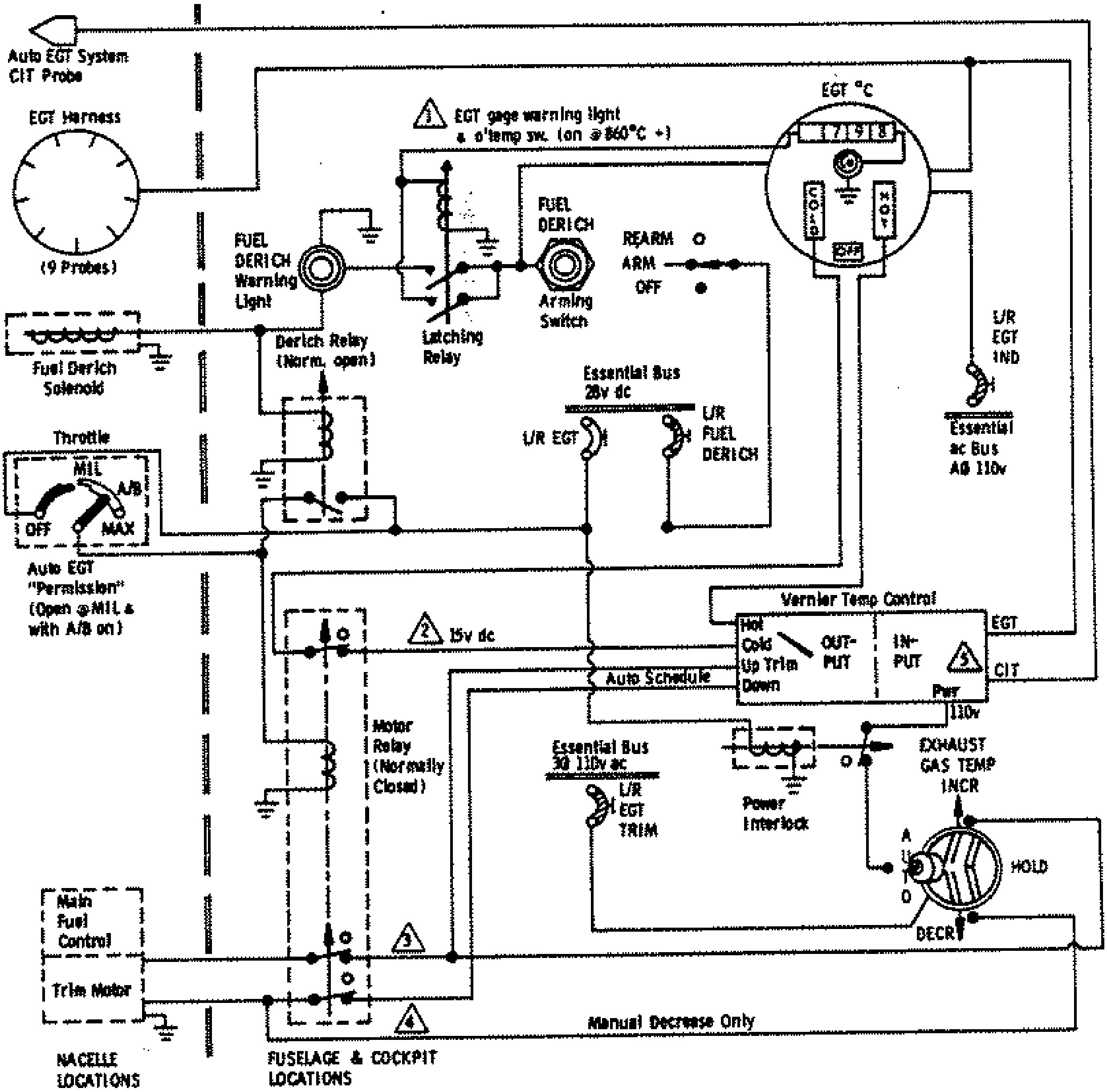
When an EGT trim switch is in AUTO, the respective throttle is positioned at or above Military, and that engine is not deriched, the permission circuit is opened (off) to allow automatic EGT trim. The electric trim motor for the respective engine regulates its main fuel control and automatically provides EGT within a 10°C nominal deadband. See Figure 1-9.

If the EGT for an engine is either above or below the deadband, the system trims EGT toward its deadband. The rate of trimming depends on temperature deviation from the deadband. If EGT is more than 10°C above the deadband, the system downtrims at its maximum rate (approximately 8°C per second), and the HOT flag is displayed on the EGT gage. When the EGT is more than 10°C below the deadband, the system uptrims at 1°C per second and the COLD flag is displayed. When EGT is 10°C or less from its deadband, the system trims at only $1/3^\circ$C per second and the condition flags retract.

CAUTION

If EGT tends to hunt or has an abnormal tendency to uptrim or downtrim while AUTO is selected, the corresponding engine should be operated in manual trim and the condition reported following flight. An EGT overtemperature may occur if continued operation in AUTO is attempted.

EGT INDICATION AND CONTROL SYSTEM



NOTE

1. Switch closes and light illuminates as 860°C reached. Switch stays closed and light remains on above 860°C. When EGT then decreases below 860°C, jewel light is extinguished but "latching" relay maintains power to derich solenoid until derich arming switch is cycled.

2. COLD flag does not operate when below Military power or when deriched. HOT and COLD flag operating power (15v dc) is produced within the vernier temperature control.

3. Direction of trim is controlled by ac power phase-sequencing.

4. Auto trim and manual uptrim are inoperative when below Military power or when deriched. Manual downtrim is always available if three-phase power is supplied.

5. Auto EGT vernier temperature control is powered only when AUTO is selected.

F203-159(c)

Figure 1-8

NOMINAL EGT SCHEDULE

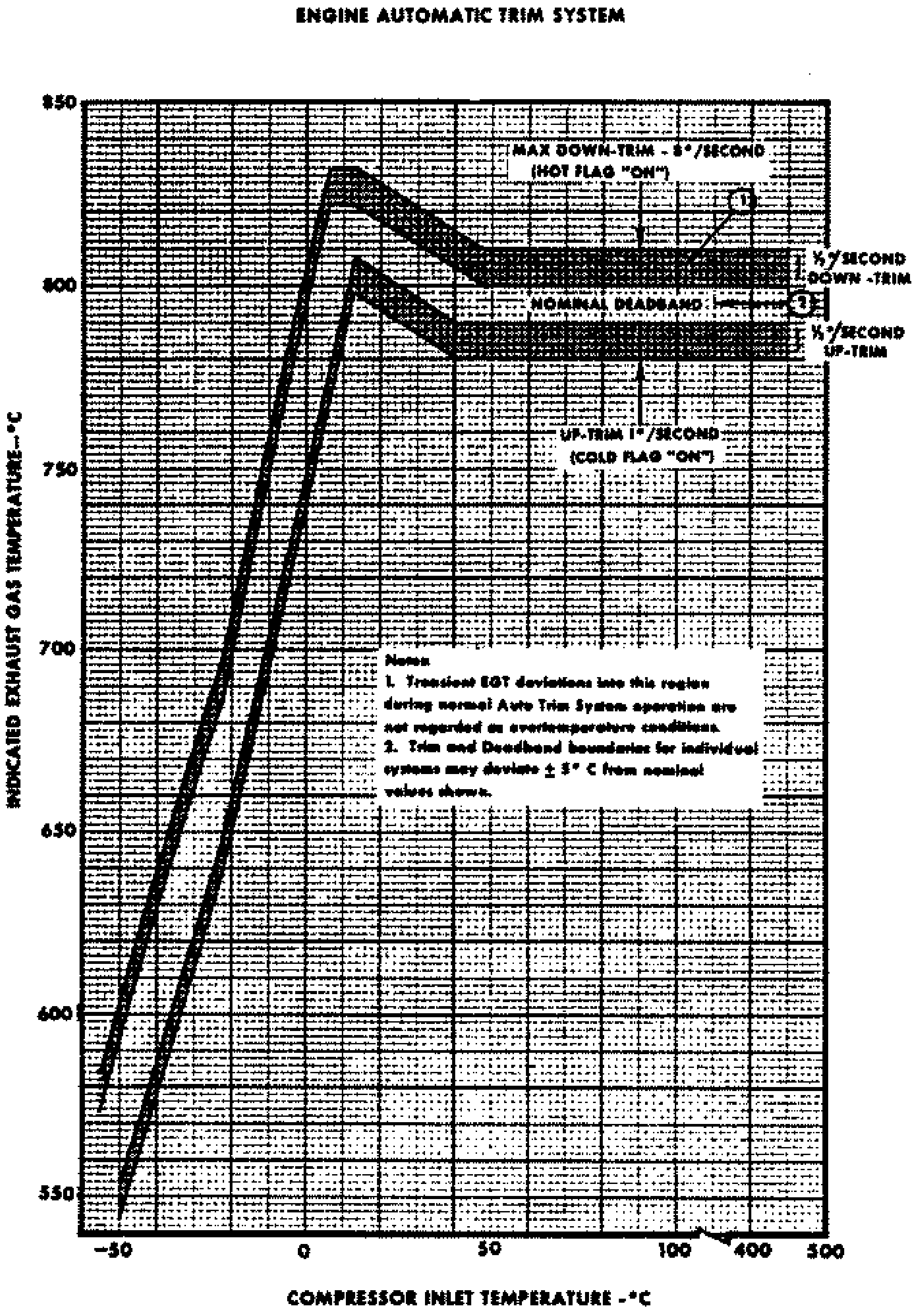


Figure 1-9

NOTE

- o The EGT condition flags do not operate when manual EGT trim is used.
- o With AUTO EGT selected, the COLD condition flag will not operate when the permission circuit is on (throttle below Military or deriched).
- o With AUTO EGT selected, the HOT condition flag will operate regardless of the condition of the permission circuit.

Auto EGT is normally used during the engine trim check; however, if the system is obviously uptrimmed above normal limits, reduce power and manually downtrim prior to resetting military power.

The hot flag overtemperature warning circuit is continually energized when in AUTO EGT. A hot flag may appear at throttle positions below the permission switch and should be corrected by manually downtrimming or retarding the throttle.

Repeated rapid throttle movements forward and back through the permission switch operating range can cause military EGT to increase sufficiently to produce derichment. This EGT ratcheting is caused by a lag in thermocouple response and can be avoided by selecting manual trim when repeated rapid throttle movements to and below Military are expected.

Selecting the spring-loaded HOLD (center) position deactivates automatic trimming and maintains the existing EGT trimmer setting.

NOTE

When AUTO EGT is selected, a brief spurious flag indication may occur in response to the EGT and CIT conditions that existed when AUTO EGT trim was last used. Valid flag indications and normal AUTO EGT trim operation should occur within five seconds.

INLET PARAMETER INDICATIONS

Compressor Inlet Pressure (CIP) Gage

The CIP gage, on the left of the pilot's instrument panel, has an L and R needle which indicate inlet static pressure at the face of each compressor, and a striped third needle that indicates expected normal CIP. The position of the striped pointer is governed by DAFICS using pressures sensed by the pitot-static system. The indication varies with Mach and KEAS so that the striped needle shows "normal" CIP for the flight condition. A substantial difference between the striped needle and actual CIP indicates improper inlet operation. Higher actual pressure at normal speeds and altitudes may produce unstarts. Lower pressure indicates poor pressure recovery due to improper spike and/or bypass door settings except at abnormal angles of attack or yaw conditions, where inlet operation is automatically biased to less than normal recovery. The spread between inlet CIP indications (L and R needles) should not exceed 1 psi. The needle can be used as a guide for bypass door settings during manual operation of one or both inlets; however, it is preferable to keep the L and R needles slightly below the "normal" indication to maintain a margin below unstart pressures. Automatic or manual inlet operation at pressures below the "normal" indication reduces aircraft range.

NOMINAL CIP RANGE

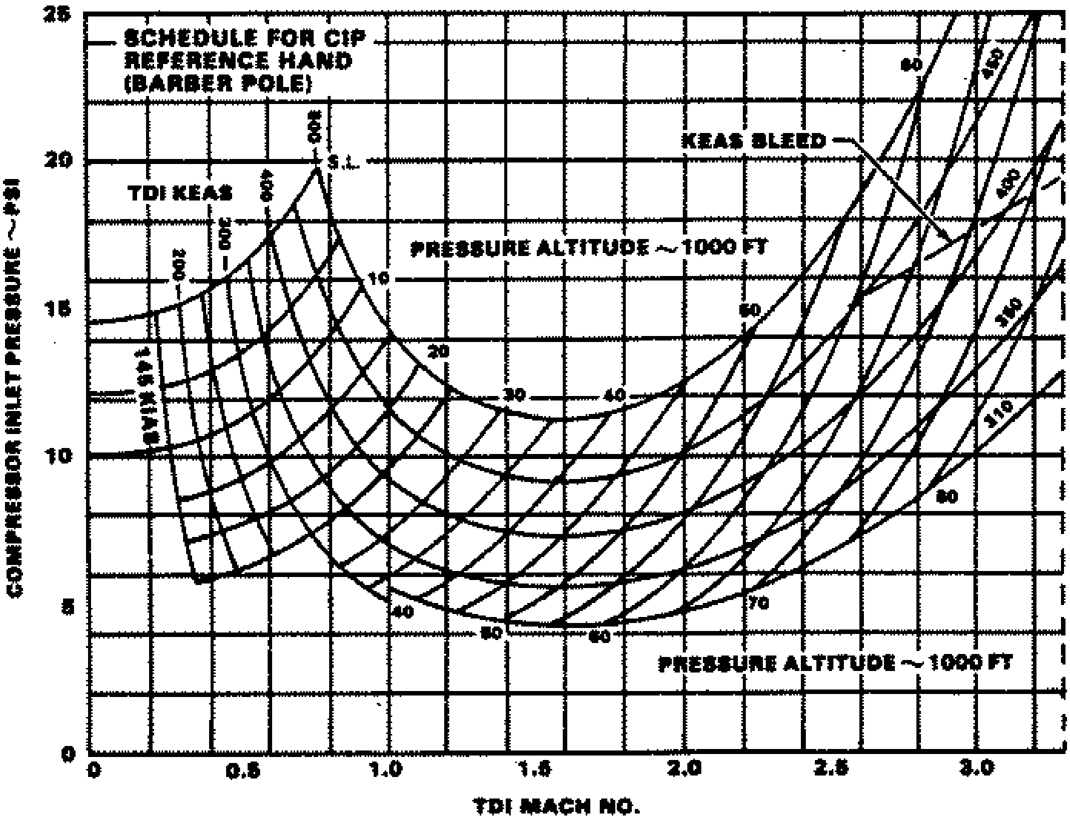


Figure 1-10

Power is supplied by the essential ac bus through the CIP circuit breaker on the pilot's right console.

Compressor Inlet Temperature (CIT) Gage

A dual indicating CIT gage is mounted on the left side of the pilot's instrument panel. L (left) and R (right) needles indicate the total (ram) air temperature forward of the first compressor stage in the corresponding engine inlet. Major calibrations are marked from 0°C to 500°C. The dial has 50° incremental markings below 300°C. Above 300°C, the gage has increased sensitivity and incremental marks are supplied for each 10°C. Slight differences between left and right CIT indications can be expected; however, differences of more than 15°C while at supersonic cruise speeds should be reported as a discrepancy. Power is furnished by the essential ac bus through the L and R CIT circuit breakers on the pilot's right console.

EXHAUST NOZZLE AND EJECTOR SYSTEM

The variable-area, iris-type, engine afterburner nozzle is comprised of segments operated by a cam and roller mechanism and four hydraulic actuators. The actuators are operated by fuel hydraulic system pressure. The engine afterburner nozzle is enclosed by a fixed-contour, convergent-divergent ejector nozzle to which free floating trailing edge flaps are attached. In flight, the inlet shock trap bleed and aft bypass doors (when open) supply secondary airflow between the engine and nacelle for cooling. During ground operation, suck-in doors in the aft nacelle area provide cooling air. Intake doors around the nacelle, just forward of the ejector, normally supply tertiary air to the ejector nozzle when subsonic. The tertiary doors and trailing edge flaps are free to open and close with varying internal nozzle pressure (a function of Mach and engine thrust).

Nozzle Actuation

The exhaust nozzle control and actuation system is composed of four actuators to position the exhaust nozzle, and an exhaust nozzle control which modulates pressure at the actuators in response to engine speed signals from the main fuel control. The exhaust nozzle control is mounted on the aft portion of the engine.

Exhaust Nozzle Position (ENP) Indicators

Two ENP indicators, one for each engine, are located on the right side of the pilot's instrument panel. They are marked from 0 to 10 as an index of nozzle position from closed to open. Each indicator responds to an electrical transducer located near its exhaust nozzle. The transducer is cooled by fuel and is operated by the afterburner nozzle feedback link. Power for the indicators is supplied from the essential ac bus through the L and R ENP circuit breakers on the pilot's right console.

ENGINE EXTERNAL AND INTERNAL BLEEDS

The internal bypass bleed control and actuation system consists of four two-position actuators to move the bleed valves and a pilot valve, within the main fuel control, to establish the pressure to the actuators. The pilot valve controls the bleed valve position in response to a mechanical signal from the main fuel control. Bleed valve position is scheduled within the main fuel control as a function of engine speed and CIT. The external bleed control and actuation system is similar to the internal bleed system, except that three actuators are used.

ENGINE INLET GUIDE VANES (IGV)

The engine compressor inlet case houses a two-position inlet guide vane (IGV) system. The guide vanes can be either in the cambered position, which is normal for cruise, or in the axial position which is normal for takeoff and acceleration to intermediate supersonic speed. The IGV axial position (parallel to the airflow) results in more thrust. Actuation to the cambered position occurs at a CIT of 85° to 115°C (about Mach 1.9) during acceleration. The cambered position is mandatory when operating continuously at CIT above 125°C (approximately Mach 2.0). Shifting is normally controlled by the main engine fuel control; however, the shift to the axial position from cambered is prevented if the IGV Lockout Switch is positioned to LOCKOUT. Refer to Figure 1-11 for IGV shift scheduling information.

IGV Lockout Switches

The shift schedules for the internal bypass bleeds and the inlet guide vanes are identical. There is a positive locking feature which prevents unscheduled IGV shift to axial after the cambered position has been reached. In addition, two-position IGV lockout switches (one for each engine) on the pilot's right console, can lock out IGV shift from cambered to axial. With a lift-loc switch in LOCKOUT, the respective IGV is maintained in the cambered position regardless of internal bleed position. The LOCKOUT position is ineffective until the guide vanes are in the cambered position. The switches cannot cause or prevent IGV shift from axial to cambered. With IGV NORM selected, IGV shift occurs with internal bleed shift. Power for the IGV lockout solenoid circuits is supplied from the essential dc bus through the L and R IGV circuit breakers on the pilot's left console.

Inlet Guide Vane Position Lights

Two rotate-to-dim, amber inlet guide vane (IGV) position lights are installed on the right side of the pilot's instrument panel. An indicator is provided for each engine, identified by L or R adjacent to the appropriate light. An IGV light illuminates when the inlet guide vanes of the respective engine shift to the axial position as scheduled by the main fuel control. The light extinguishes when the IGV reaches the cambered position. Inlet guide vane position is sensed by a switch on the engine compressor case which operates when the

COMPRESSOR BLEED AND IGV SHIFT SCHEDULE
IGV/Bypass Bleed Functions

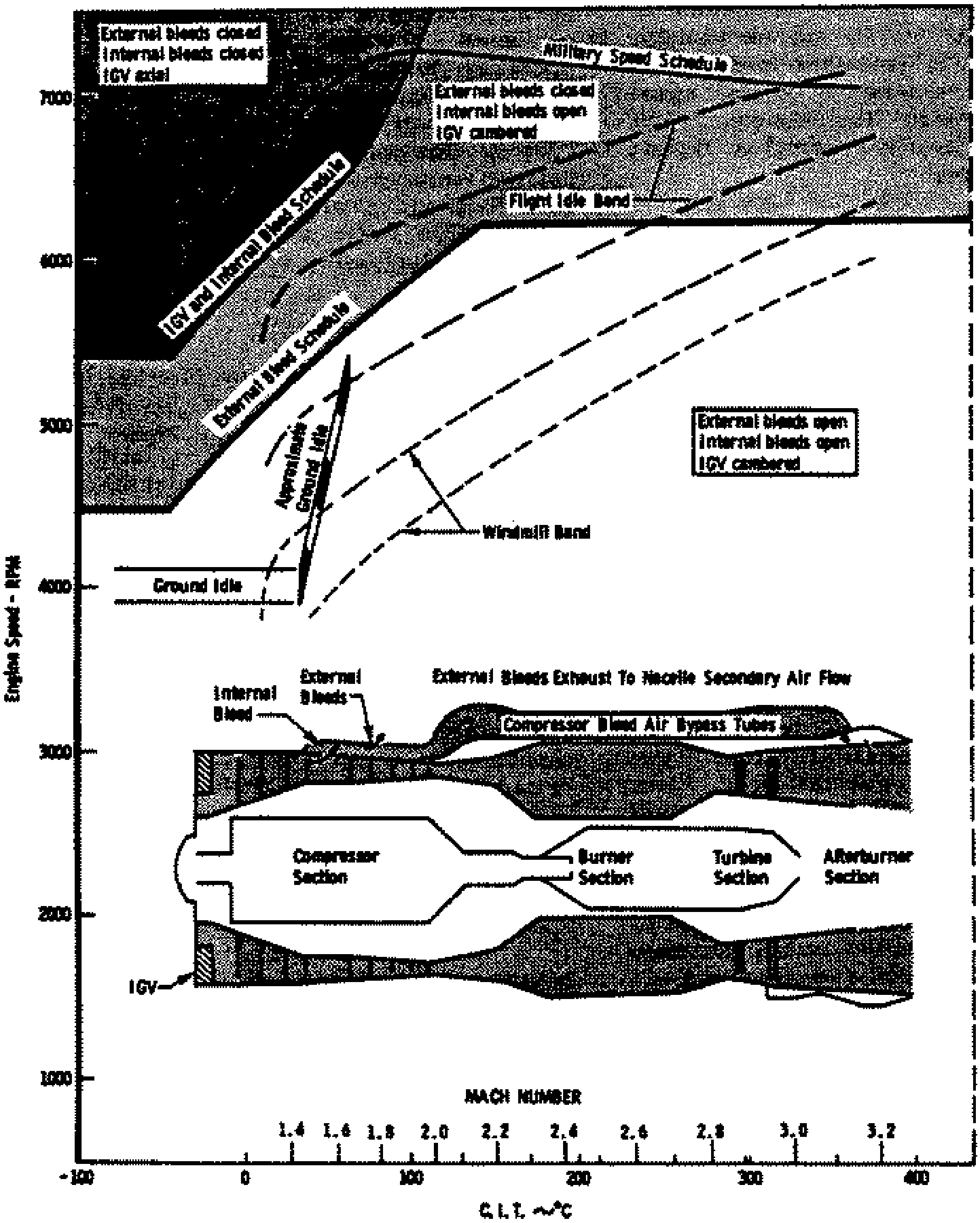


Figure 1-11

guide vanes reach or leave the cambered position. Power for the lights is furnished by the essential dc bus through the WARN 2 light circuit breaker on the pilot's left console.

a. The IGV lights must be off (IGV cambered) during start and at idle.

b. The IGV lights must be on (IGV axial) for takeoff.

c. The IGV lights must be off (IGV cambered) above 150°C (approximately Mach 2.2).

OIL SUPPLY SYSTEM

The engine and speed-reduction gearbox are lubricated by an engine-contained, "hot tank", closed system. The oil is cooled by circulation through an engine fuel-oil cooler. The oil tank is mounted on the lower right side of the engine compressor case. Tank volume is 6.7 US gallons. The oil tank is serviced to 5.15 US gallons. The oil is gravity-fed to the main oil pump which forces the oil through a filter and the fuel-oil cooler. The filter is equipped with a bypass in case of clogging. The oil is distributed to the engine bearings and gears from the fuel-oil cooler. Oil screens are installed at the lubricating jets for additional protection. Scavenge pumps return the oil to the tank where it is de-aerated. A pressure regulating valve keeps flow and pressure relatively constant during all flight conditions. Oil quantity warning lights are provided.

The approved oil is MIL-L-87100 (PWA 524). At low ambient temperatures, oil may be diluted with trichlorethylene, Federal Specification O-T-634, Type 1.

Main Fuel-Oil Cooler

Engine oil temperature is controlled by engine fuel which passes through the main fuel-oil heat exchanger. A bypass valve in the cooler passes additional fuel around the cooler when engine requirements are greater than the flow capacity of the cooler (approximately 12,000 pounds per hour).

Engine Oil Pressure Gages

An oil pressure gage for each engine is provided on the right side of the pilot's instrument panel. Each gage indicates output pressure of the respective engine oil pump in pounds per square inch, using electrical signals from a fuel-cooled transmitter. The dials are calibrated from 0 to 100 psi in 5-psi increments. Power for the gages is furnished by the essential ac bus 26 volt instrument transformer through the L and R OIL PRESS circuit breakers on the pilot's annunciator panel.

Engine Oil Temperature Lights

The L and R OIL TEMP annunciator lights are not functional. The OIL TEMP lights only illuminate when the IND & LT TEST push-button switch is depressed.

Low Oil Quantity Lights

L and R OIL QTY warning lights, on the pilot's annunciator panel, illuminate when oil quantity in the respective engine oil tank is less than 2-1/4 gallons.

ENGINE FUEL HYDRAULIC SYSTEM

Each engine is provided with a fuel hydraulic system for actuation of the afterburner exhaust nozzle, IGVs, and the start and bypass bleed valves. Fuel hydraulic pressure is also required to dump the chemical ignition system. An engine-driven pump maintains system pressures up to 1800 psi with a maximum flow of 50 gpm (approximately 19,700 pph) for transient requirements. Engine fuel is supplied to the pump from the main fuel pump boost stage. Some high-pressure fuel is diverted from the hydraulic system to cool the nonafterburning recirculation line and the windmill bypass valve discharge line. This fuel is returned to the aircraft fuel system. Low-pressure fuel from the hydraulic pump case is returned to the main fuel pump boost stage. Hydraulic system loop cooling is provided by the compensating fuel supplied by the main fuel pump.

ACCESSORY DRIVE SYSTEM (ADS)

An ADS is mounted forward of the engine in each nacelle. Its three major components include a constant speed drive, an accessory gearbox, and an all-attitude oil reservoir. Input power from the engine is transmitted to the ADS through a reduction gearbox on the engine and a flexible drive shaft. At the ADS, a constant speed drive unit converts the variable shaft speed to a constant rotational speed to power the ac generator. Two hydraulic pumps and a fuel circulating pump are also mounted on the ADS gearbox. The two hydraulic pumps supply power for the (A and L) or (B and R) hydraulic systems. The fuel circulating pump supplies fuel to the aircraft heat sink system. The speeds of these pumps vary directly with engine rpm.

The ADS is lubricated by an independent dry sump system with its own pump, using oil from an all-attitude reservoir. The reservoir is pressurized with nitrogen gas from the aircraft LN_2 system and supplies oil to the accessory gearbox, the constant speed drive, and the ac generator regardless of flight attitude. (Loss of the LN_2 supply to the ADS does not affect ADS operation.) The oil is cooled by circulation through a fuel-oil heat exchanger which is part of the heat sink system. Reservoir capacity is approximately 8 quarts.

EXTERNAL STARTER SYSTEM

An external starting unit is required for ground starts. This may be a compressed air supply, a self-contained gas engine cart, or a multiple air-turbine cart. The output drive gear of either cart connects to a starter gear on the main gearbox at the bottom of the engine. There are no aircraft controls for this system. It is turned on and off by the ground crew in response to instructions from the pilot.

CHEMICAL IGNITION (TEB) SYSTEM

Triethylborane (TEB) is used for starting ignition of main burner and afterburner fuel. Catalytic igniters attached to the afterburner flame holders tend to maintain afterburner operation after initial ignition.

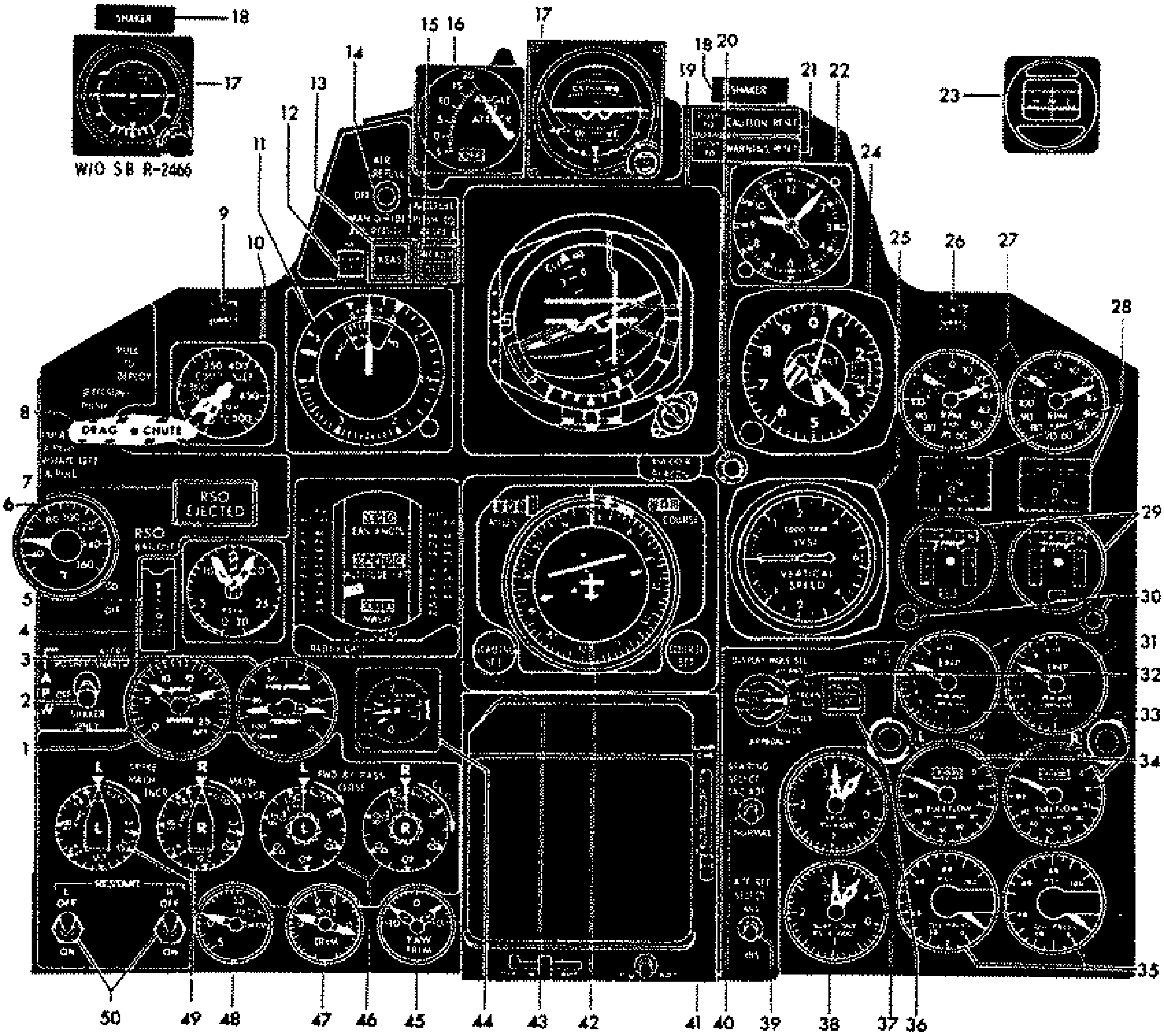
A 600 cc (1-1/4 pint) TEB storage tank is installed on each engine. The tanks are pressurized with nitrogen gas prior to flight to provide inerting and operating pressure. Special handling procedures are required for TEB as it will burn spontaneously with exposure to air above -5°C. The TEB tank is cooled by main burner fuel flow. A rupture disk is provided for each tank which will allow vaporized TEB and nitrogen gas to be discharged through the afterburner section if tank pressure exceeds a safe level. No indication of TEB tank discharge is provided to the flight crew.

At least 16 metered TEB injections can be made with one full tank of TEB. The system is controlled by engine fuel pressure signals while the engine is rotating. Throttle advancement from OFF to IDLE provides main burner ignition, and from Military into the AFTERBURNER range provides afterburner ignition. Main burner ignition occurs almost immediately during an airstart with the engine windmilling. The time required to obtain afterburner ignition is a function of the afterburner fuel manifold fill time, (up to three seconds at sea level and seven seconds at altitude).

Igniter Purge Switch

An IGNITER PURGE toggle switch is located on the pilot's right instrument side panel. When the switch is held in the up position, a solenoid-operated valve supplies fuel-hydraulic system pressure to the chemical ignition system dump valve if the engine is rotating. This allows the TEB to be dumped into the afterburner section. While dumping, engine speed should be above 5000 rpm to avoid afterburner liner damage. The purge switch must be actuated for at least 40 seconds to dump a full load of TEB. At the end of the dump period, the switch should be released and re-cycled to clean out the lines. Electrical power for purging is furnished from the essential dc bus through the IGN PURGE circuit breaker on the pilot's left console.

CENTER INSTRUMENT PANEL - Forward Cockpit

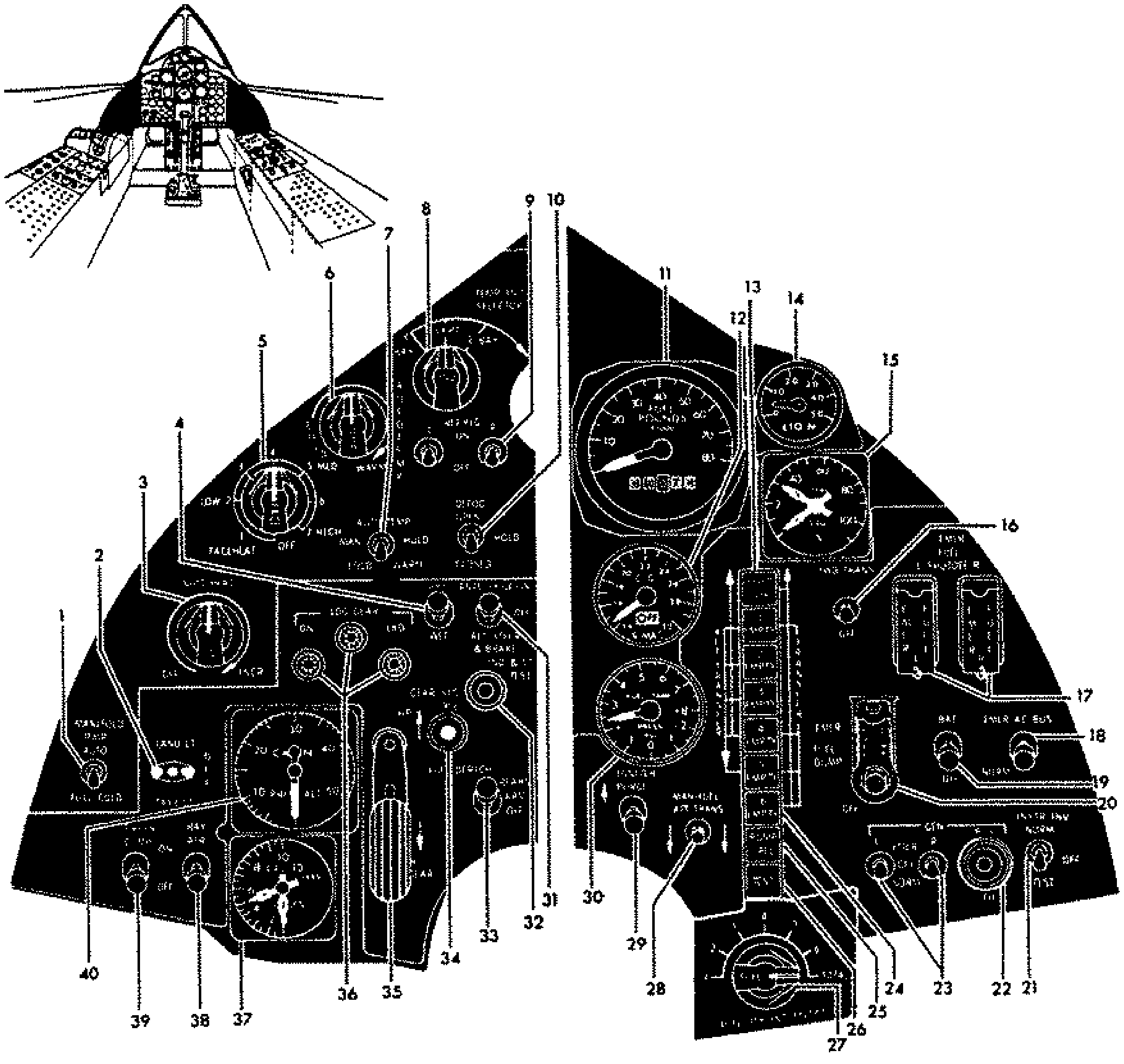


1. Spike Position Indicator
2. Pusher/Shaker Switch
3. Forward Bypass Position Indicator
4. Compressor Inlet Pressure Gage
5. RSO Bailout Switch
6. Temperature Indicator
7. RSO Ejected Indicator Light
8. Drag Chute Handle
9. Left Inlet Unstart Light
10. Compressor Inlet Temperature Gage
11. Airspeed - Mach Meter
12. Nosewheel Steering Engaged Light
13. KEAS Warning Light
14. Air Refuel Switch
15. Air Refuel Ready - Disc Pushbutton and Light
16. Angle of Attack Indicator
17. Standby Attitude Indicator
18. Shaker Indicator Light
19. Attitude Director Indicator
20. Marker Beacon Light
21. Master Caution and Warning Lights
22. Elapsed Time Clock
23. Standby Compass (In Canopy)
24. Altimeter
25. Inertial - lead Vertical Speed Ind.
26. Right Inlet Unstart Light
27. Tachometers
28. Fire Warning Lights
29. Exhaust Gas Temperature Inds.
30. Fuel Derich Lights
31. Exhaust Nozzle Position Indicators
32. Display Mode Select Switch
33. IGV Lights
34. Fuel Flow Indicators
35. Oil Pressure Indicators
36. Tacan Control Transfer Switch
37. L and R Hydraulic Systems Pressure Gage
38. A and B Hydraulic Systems Pressure Gage
39. Attitude Reference Select Switch
40. Bearing Select Switch
41. Nav Map Display
42. Horizontal Situation Indicator
43. Triple Display Indicator
44. Accelerometer
45. Yaw Trim Indicator
46. Forward Bypass Switches
47. Roll Trim Indicator
48. Pitch Trim Indicator
49. Spike Switches
50. Inlet Restart Switches

F200-77(2)(p)

Figure 1-12

INSTRUMENT SIDE PANELS - Forward Cockpit



1 Manifold Temperature Switch
2 Landing and Taxi Light Switch
3 Suit Heat Rheostat
4 Wet-Dry Switch
5 Face Heat Rheostat
6 Cockpit Temperature Control
7 Cockpit Temperature Control and O-Ride
8 Temperature Indicator Selector Switch
9 L and R Refrigeration Switches
10 Defog Switch
11 Fuel Quantity Indicator
12 Center Of Gravity Indicator
13 Fuel Crossfeed Switch
14 System 3 Nitrogen Quantity Indicator
15 Liquid Nitrogen Quantity Indicator
16 Fuel Forward Transfer Switch
17 Emergency Fuel Shutoff Switches
18 Emergency AC Bus Switch
19 Battery Switch
20 Fuel Dump Switch
21 Instrument Inverter Switch
22 Generator Bus Tie Switch
23 L and R Generator Switches
24 Fuel Boost Pump Switches
25 Pump Release Switch
26 Fuel Boost Pump Light Test Switch
27 Fuel Quantity Indicator Selector Switch
28 Manual Fuel Aft Transfer Switch
29 Igniter Purge Switch
30 Fuel Tank Pressure Indicator
31 Brake Switch
32 Indicators and Light Test Switch
33 Fuel Derichment Switch
34 Gear Signal Release Switch
35 Landing Gear Lever
36 Landing Gear Indicator Lights
37 Liquid Oxygen Quantity Indicator
38 Bay Air Switch
39 Cockpit Pressure Dump Switch
40 Cabin Altitude Indicator

F203-3144(a)

Figure 1-13

[redacted]

LEFT CONSOLE - Aft Cockpit

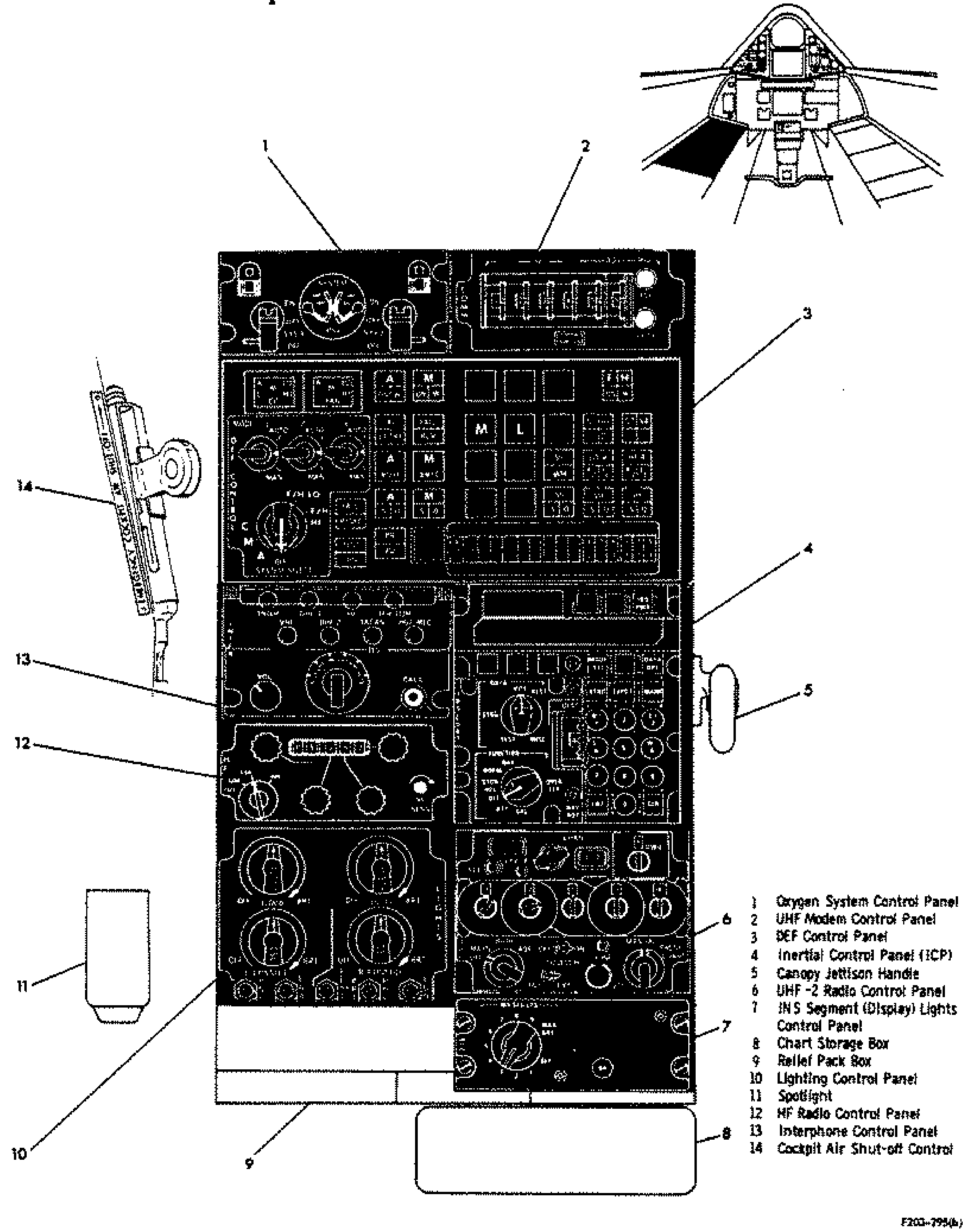
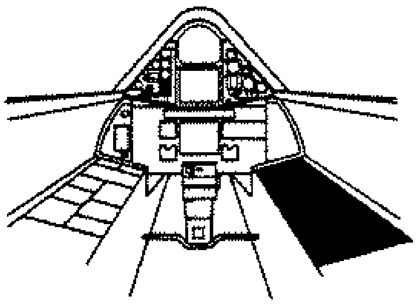
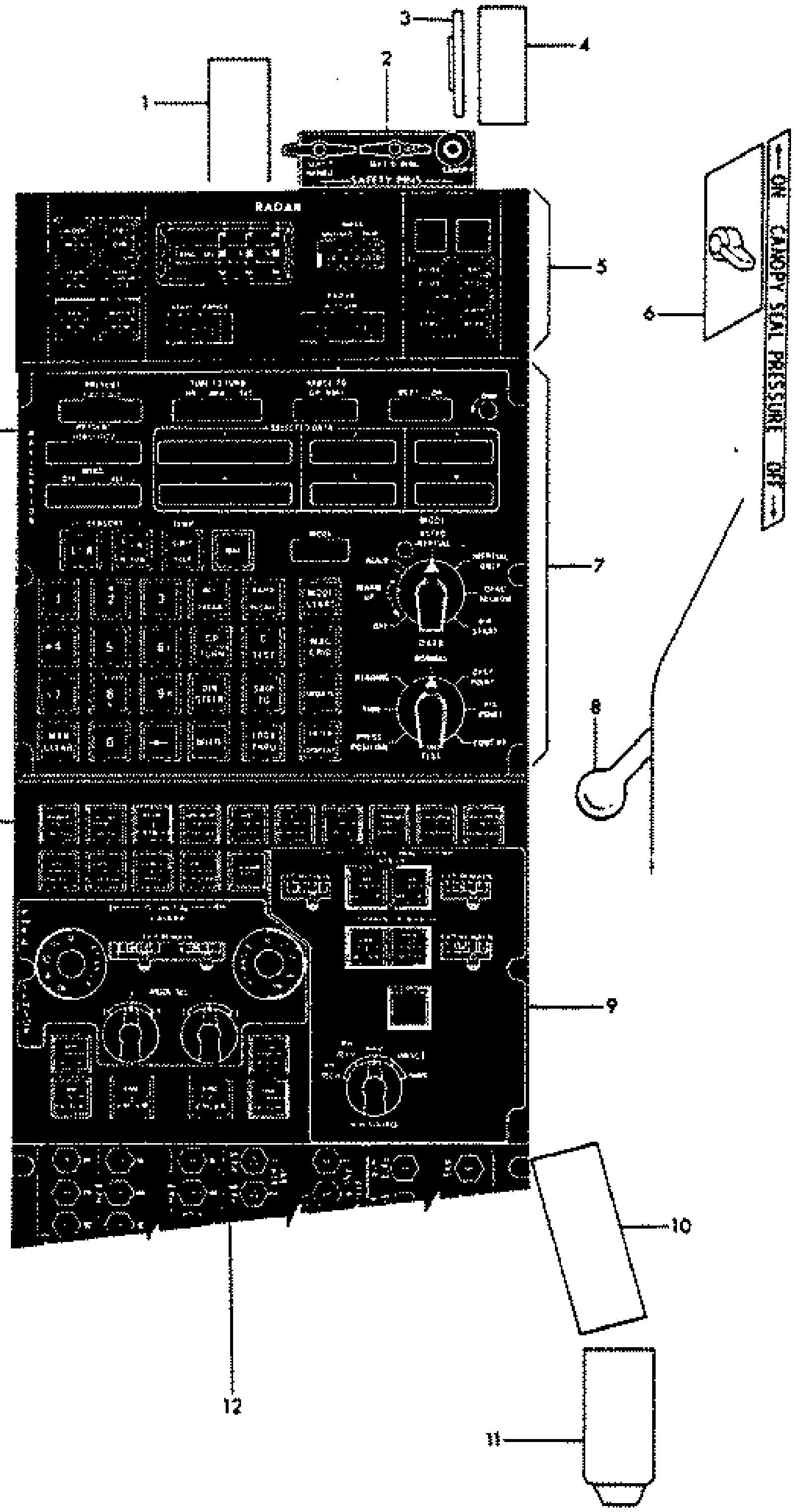


Figure 1-18

[redacted]

RIGHT CONSOLE - Aft Cockpit





1. Water bottle box
2. Safety pins
3. Dinghy stabber
4. Chart storage box
5. Radar control panel
6. Canopy seal pressure valve (sill)
7. Astroinertial navigation control panel
8. Canopy latch handle (sill)
9. Mission equipment power and sensor control panel
10. Storage box
11. Spotlight
12. Circuit breaker panel
13. Chart storage box

F203-265(c)

Figure 1-19

AIR INLET SYSTEM

The air inlet system in each nacelle includes the cowl structure, a moving spike to provide optimum internal airflow characteristics, variable forward and aft bypass openings, a spike porous centerbody bleed, and an internal shock trap bleed for internal shock wave positioning and boundary layer flow control. Each inlet is canted inboard and downward to align with the local airflow pattern. (See Figures 1-20 and 1-21.)

The forward and aft bypass openings control airflow characteristics within the inlet and mass flow to the engine. Normally, the spike and forward bypass are operated automatically by DAFICS and the aft bypass is scheduled manually. Overriding manual controls are provided for the spike and forward bypass. The forward bypass can be operated manually when the spike is in automatic operation; however, when the spike is controlled manually, the forward bypass is also in manual control. Manual operation of the spike alone while the forward bypass control is in the AUTO position will cause the forward bypass to open 100%.

INLET SPIKES

The spikes are automatically locked in the forward postion for ground operation and for flight below 30,000 feet. They are unlocked above this altitude, but remain in their forward positions until Mach 1.6. During automatic operation above Mach 1.6, the spikes retract approximately 1-5/8 inch per 0.1 Mach number. Total spike motion is approximately 26 inches. This increases the captured stream tube area 112%, from 8.7 square feet to 18.5 square feet. The throat closes down to 4.16 square feet, 54% of the area at Mach 1.6.

During automatic operation, DAFICS schedules spike position as a function of Mach number with biasing for vertical acceleration, angle of attack, and angle of sideslip. See Figures 1-23, 1-25 and 1-27. DAFICS air data (sensed at the nose-mounted pitot mast and measured by the PTAs) are used to compute Mach, angle of attack, and angle of sideslip for automatic spike control.

Spike position can also be set manually by cockpit controls.

The spike centerbody is equipped with small slots which remove spike boundary layer air and prevent flow separation. The air is ducted overboard through the nacelle louvers after passing through the spike and its supporting struts.

INLET FORWARD BYPASS

The forward bypass openings in each inlet provide overboard exhausts for inlet air which is not required by the engine. The openings are a rotating band of ports located a short distance aft of the inlet throat. The air exits overboard through louvers located forward of the louvers for the spike centerbody bleed. Bypass position is automatically modulated by DAFICS to control inlet pressure aft of the normal shock (major internal shock wave) to position this shock properly near the throat.

In automatic operation, the forward bypass remains closed until above Mach 1.4; then it is released to modulate in accordance with DAFICS schedules. In automatic operation, the forward bypass returns to closed when below Mach 1.3. The inlet usually "starts" between Mach 1.6 and Mach 1.8; that is, the normal shock moves from in front of the inlet to a position near the shock trap bleed in the

AIR INLET SYSTEM - Spike Forward

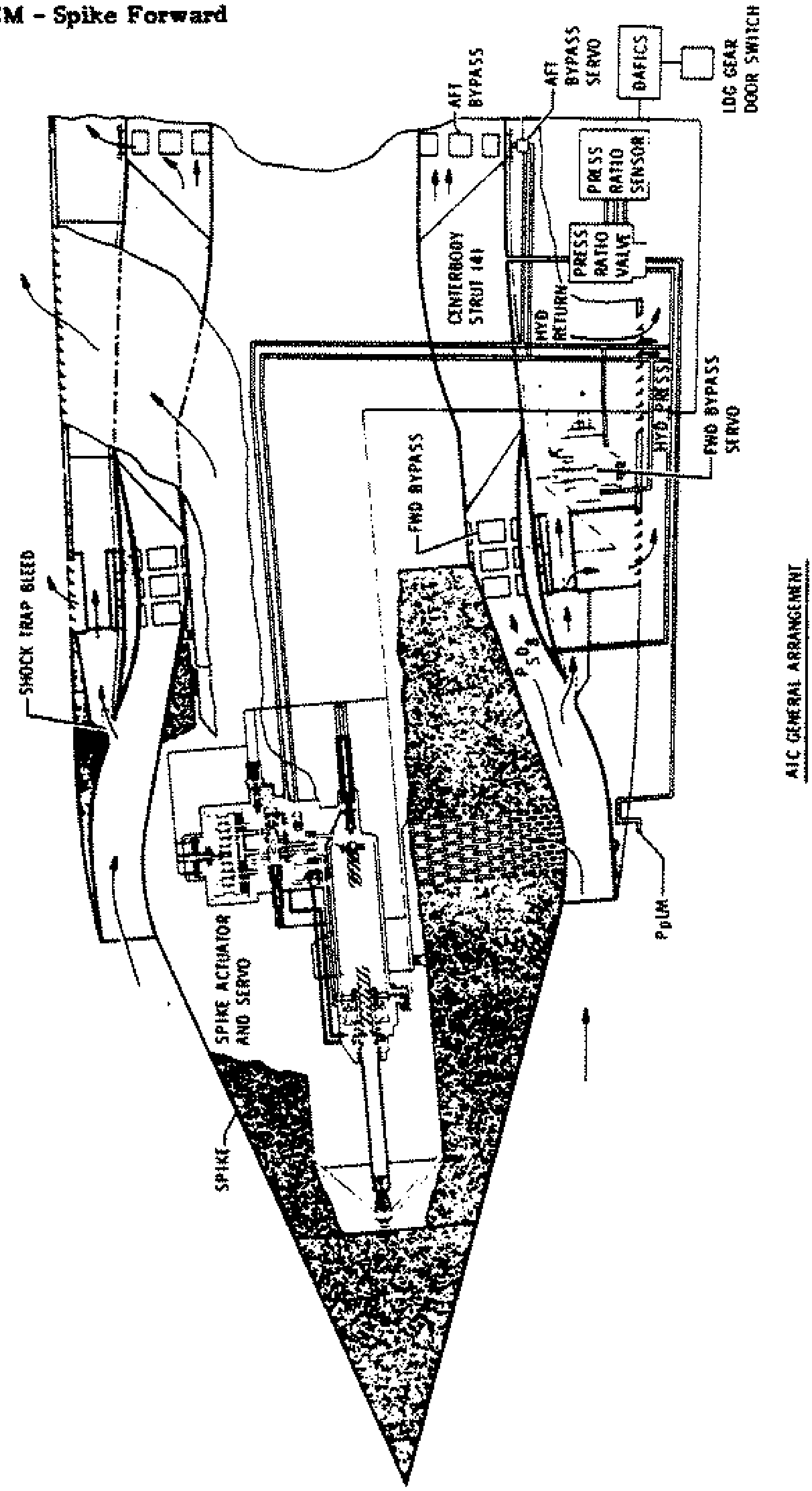


Figure 1-20

AIRFLOW PATTERNS

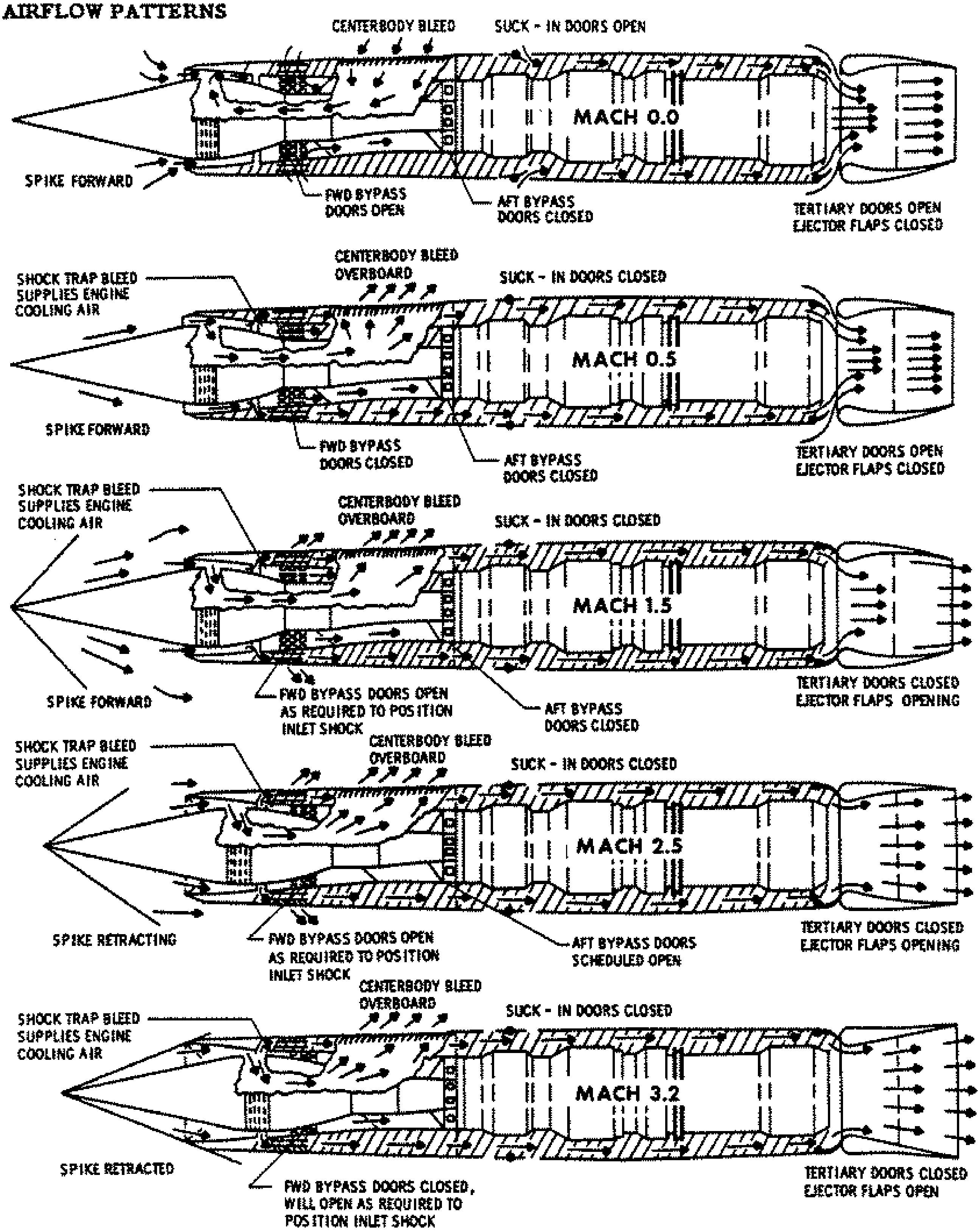


Figure 1-21

BARBER POLE C.I.P.

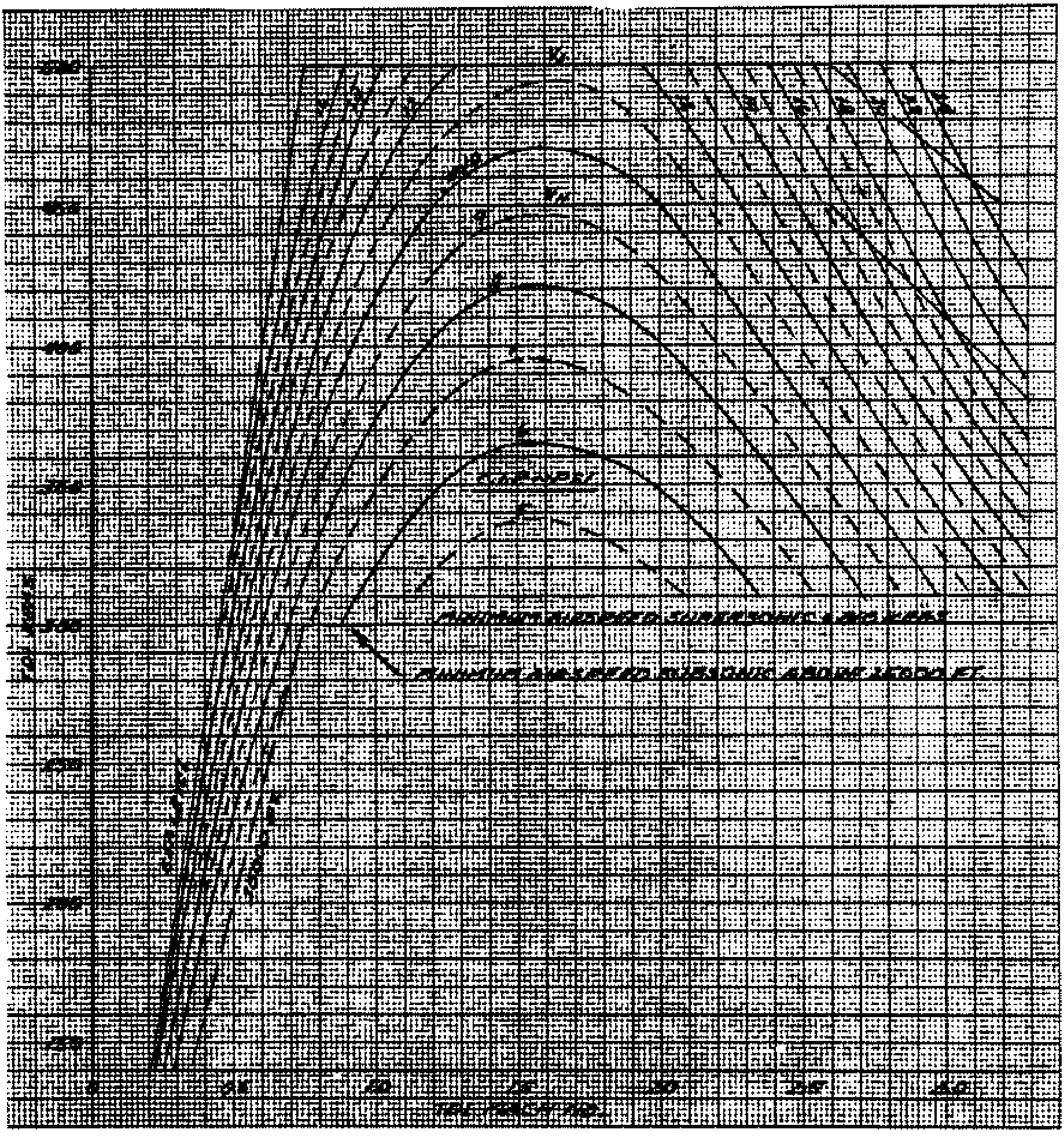


Figure 1-22 (Sheet 1 of 5)

AIR INLET SPIKE SCHEDULE

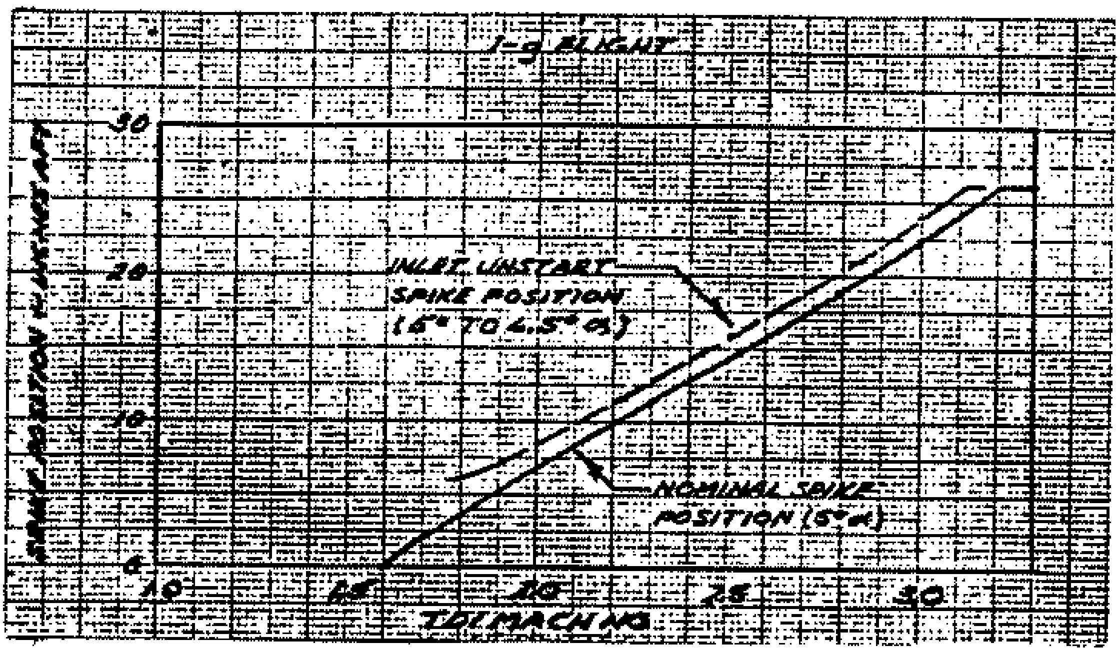


Figure 1-22 (Sheet 2 of 5)

throat. As Mach increases, the forward bypass position must modulate automatically, or by manual control, to keep the normal shock near the throat. When in manual control, the opening can be set in accordance with a Mach schedule or by referring to the position of the opposite inlet (if it is operating automatically). See Figure 3-5.

In automatic control, DAFICS positions the forward bypass to maintain a duct pressure ratio (DPR) schedule as a function of Mach number, with biasing for vertical acceleration, angle of attack, and angle of sideslip. (DPR = P_{sD8}/P_{pLM}, inlet duct static pressure immediately aft of the primary shock wave divided by total free stream pressure or pitot pressure outside the cowl.) (See Figure 1-20.) Four inlet duct static pressure taps measure P_{sD8}. These are located circumferentially on the inlet duct wall, aft of the shock trap bleed. The total free stream pressure, P_{pLM}, is sensed by two pitot probes located on the outside of each nacelle.

When the landing gear is down, the forward bypass is held 100% open by an override signal from a main gear door switch. Control of the forward bypass reverts to the mode selected by the pilot when the gear has retracted.

INLET AFT BYPASS

The inlet aft bypass consists of a rotating band of ports located just forward of the engine face. When rotated to an open position, inlet air is allowed to bypass the engine and join the flow from the shock trap bleed. The combined flow passes through the space between the engine case and the nacelle structure, cools these spaces, then augments the exhaust gas flow in the ejector area. See Figure 1-21.

FORWARD BYPASS POSITION - Mach 2.20 to 2.80

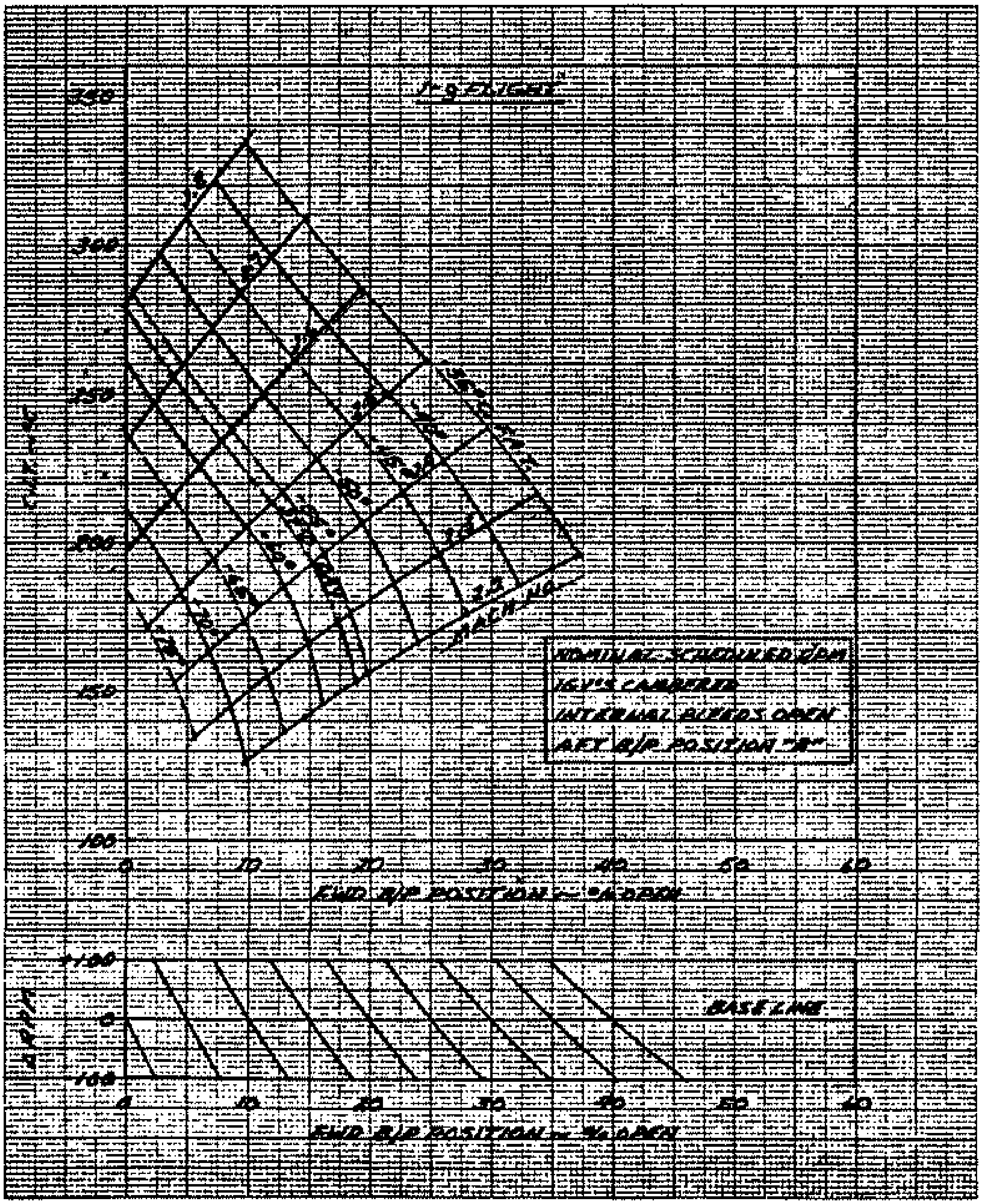


Figure 1-22 (Sheet 3 of 5)

FORWARD BYPASS POSITION - Mach 2.20 to 3.20

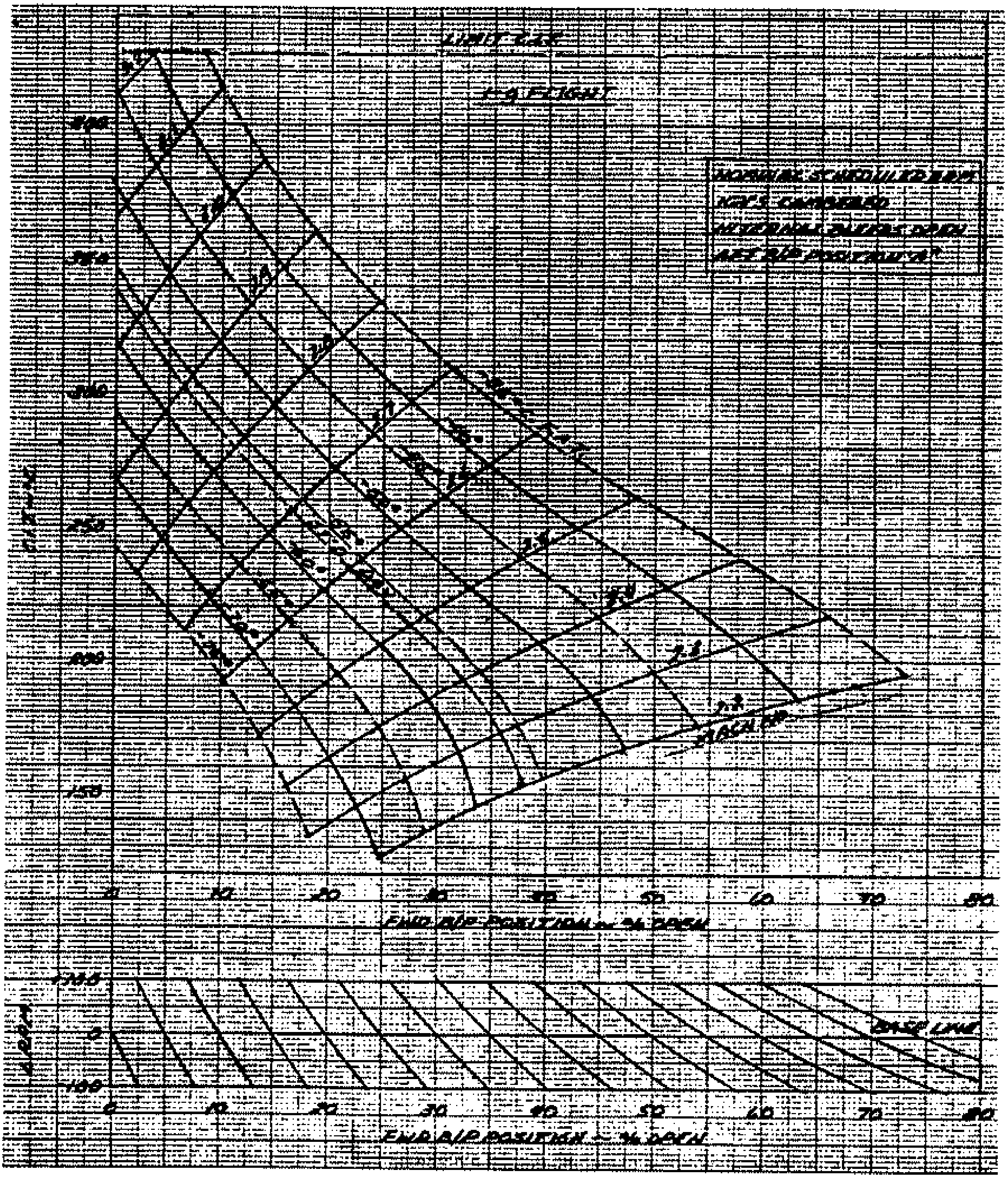


Figure 1-22 (Sheet 4 of 5)

FORWARD BYPASS POSITION - Mach 2.90 to 3.20

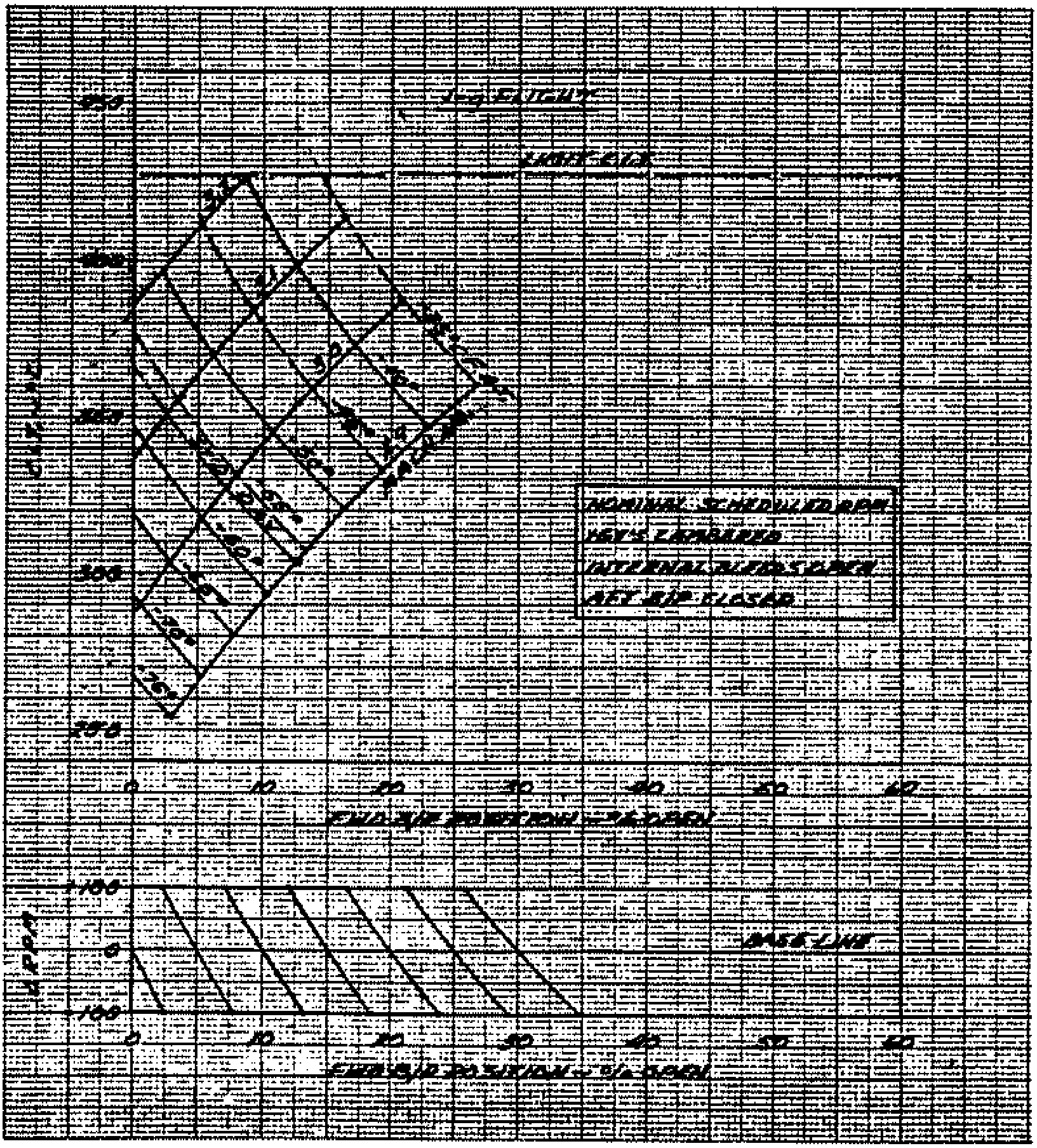


Figure 1-22 (Sheet 5 of 5)

The position of the aft bypass in each inlet is controlled by rotary selector switches in the pilot's cockpit. When the gear is extended, the aft bypass ports are held closed by an override signal from the nose gear downlock switch.

INLET CONTROL PARAMETERS

When the aircraft is supersonic, inlet airflow is controlled by positioning the spike and forward bypass so that the locations of shock waves ahead of the inlet and at the inlet throat produce maximum practical pressure recovery at the engine face, and supply the proper amount of air to the engine. Refer to spike position vs Mach number and forward bypass position vs Mach number and CIT, Figure 1-22. Manual operation of the aft bypass provides for those conditions where additional bypass area is required or when a reduction in forward bypass flow is desired. The forward and aft bypass openings and the spikes for the left and right inlets are operated by the L and R hydraulic systems, respectively. The inlet control also includes a shock expulsion sensor and restart system.

In automatic control, DAFICS schedules the spike and forward bypass positions as a function of Mach, with biasing for load factor, angle of attack, and angle of sideslip. Bias does not override a manually operated spike or forward bypass.

Load Factor Bias

The g-bias function in DAFICS schedules the spike forward and the forward bypass more open when load factor is greater than 1.5 g or less than 0.7 g during automatic inlet operation. The biasing noticeably decreases CIP and reduces the possiblity of unstarts. Figures 1-23 and 1-24 illustrate spike and door biasing, respectively.

SPIKE POSITION BIAS DUE TO LOAD FACTOR

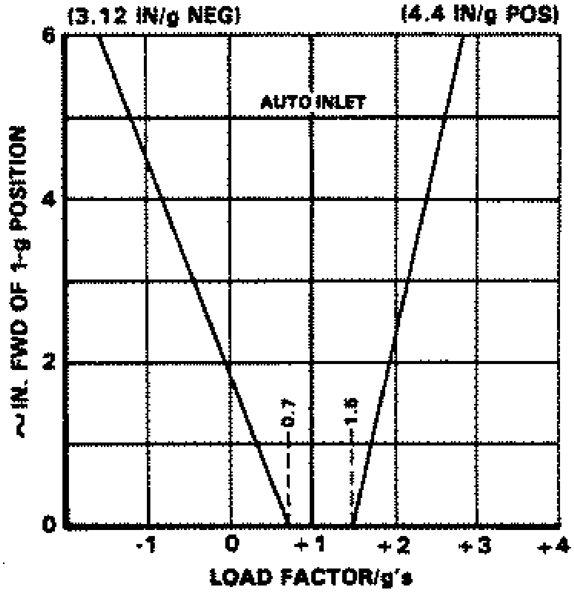


Figure 1-23

FWD BYPASS POSITION BIAS DUE TO LOAD FACTOR AND DPR

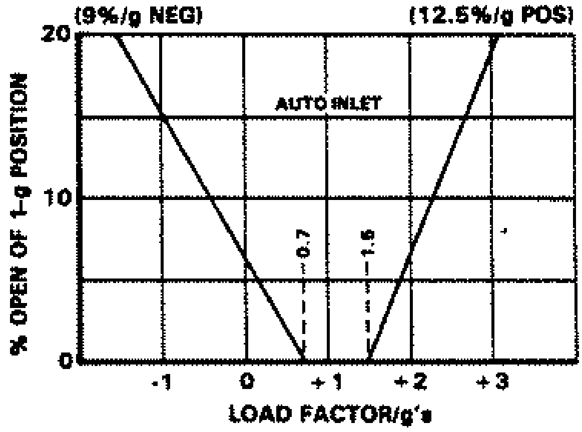


Figure 1-24

Angle of Attack Bias

During automatic inlet operation above Mach 1.6, spikes are biased forward when angle of attack deviates from five degrees (Figure 1-25). Add the angle of attack bias and load factor bias to obtain total bias in a turn or pull-up.

Figure 1-26 illustrates the effect of angle of attack bias on forward bypass position.

SPIKE POSITION BIAS DUE TO ANGLE OF ATTACK

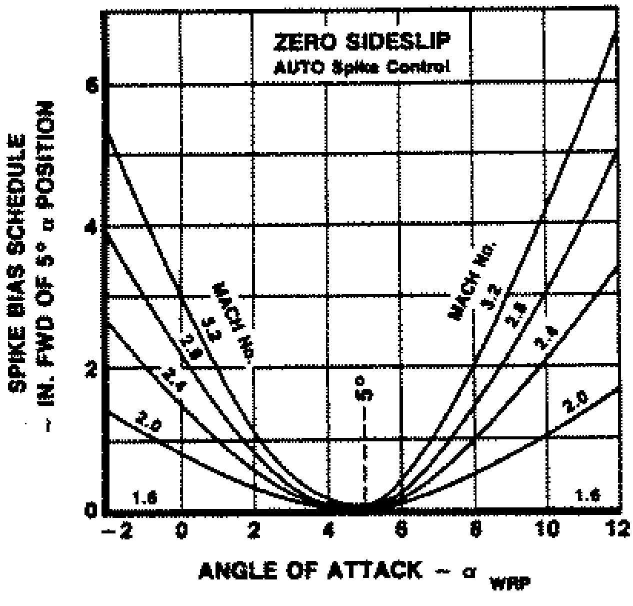


Figure 1-25

Angle of Sideslip Bias

During automatic inlet operation, spike position is also biased forward due to sideslip (yaw) angle. Figure 1-27 shows the bias for the left inlet. Because of the nacelle inlet cant, the chart is not applicable to the right inlet unless read with a reverse sign for sideslip angle, i.e., enter the chart with $-\beta$ for a nose-left condition.

EFFECT OF ANGLE OF ATTACK ON FORWARD BYPASS POSITION

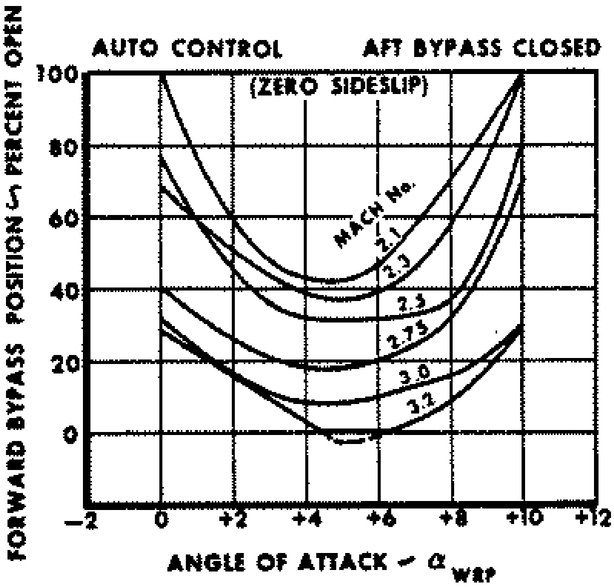


Figure 1-26

Sideslip angles of up to one degree are common. The result can be a "split" spike position indication. Because the spike control positioning tolerance is ± 0.2 inches, this could result in a split indication of up to 0.9 inches with a sideslip angle of 1° (with no indicator instrument error).

In automatic operation, the duct pressure ratio (DPR) is also biased lower due to sideslip angle. Figure 1-28 shows forward bypass position changes that result from spike and DPR biasing when flying with a sideslip at typical cruise speeds. Note that a 1° sideslip can result in a 3% difference between the two inlets. An additional 3% difference is possible due to the tolerance in DPR scheduling.

SPIKE POSITION BIAS DUE TO SIDESLIP (YAW)

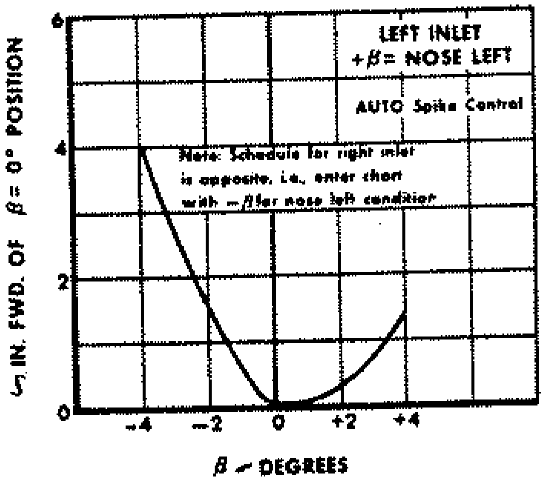


Figure 1-27

EFFECT OF SIDESLIP (YAW) ON FORWARD BYPASS POSITION

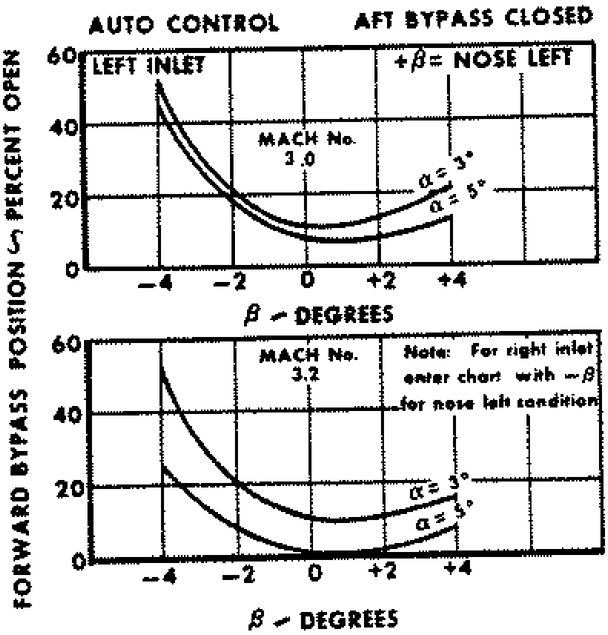


Figure 1-28

Effect of RPM on Forward Bypass Position

Engine speed also affects bypass position, as rpm directly affects engine airflow requirements. This amounts to approximately 4% increase in bypass opening per 100 rpm decrease in engine speed when at typical cruise conditions and with the bypass nearly closed. The influence of rpm change increases significantly as the forward bypass condition approaches 50% open, where a 10% difference in forward bypass indication per 100 rpm change is not unusual.

Malfunction Biases

During automatic inlet operation, the forward bypass door duct pressure ratio (DPR) schedule is biased slightly lower for: loss of DAFICS computer(s), loss of reliable M PTA air data for sideslip computations, or unstart(s).

The DPR schedules of both inlets are biased 40 mpr (milli-pressure ratios) lower if a DAFICS A, B, or M computer fails (appropriate A, B, or M CMPTR OUT caution light on the pilot's annunciator panel illuminated).

If DAFICS is not receiving sideslip angle data from the M PTA, or the sideslip angle data from the M PTA is unreasonably high (as determined by comparison with the yaw SAS lateral accelerometers), the angle of sideslip used for automatic inlet operation computations is zero and the DPR schedules for both inlets are biased 20 mpr lower.

Each time the shock expulsion sensor (SES) detects an unstart (as indicated by illumination of the L or R UNST light), the DPR schedule for the inlet that unstarted is reduced 10 mpr. The DPR schedule for the other inlet is not changed.

The total reduction in DPR schedule on each inlet will not exceed 40 mpr regardless of how many malfunctions occur. For comparison, a 40 mpr differential is the range of adjustment provided to maintenance for scheduling inlet DPR.

AIR INLET SYSTEM CONTROLS

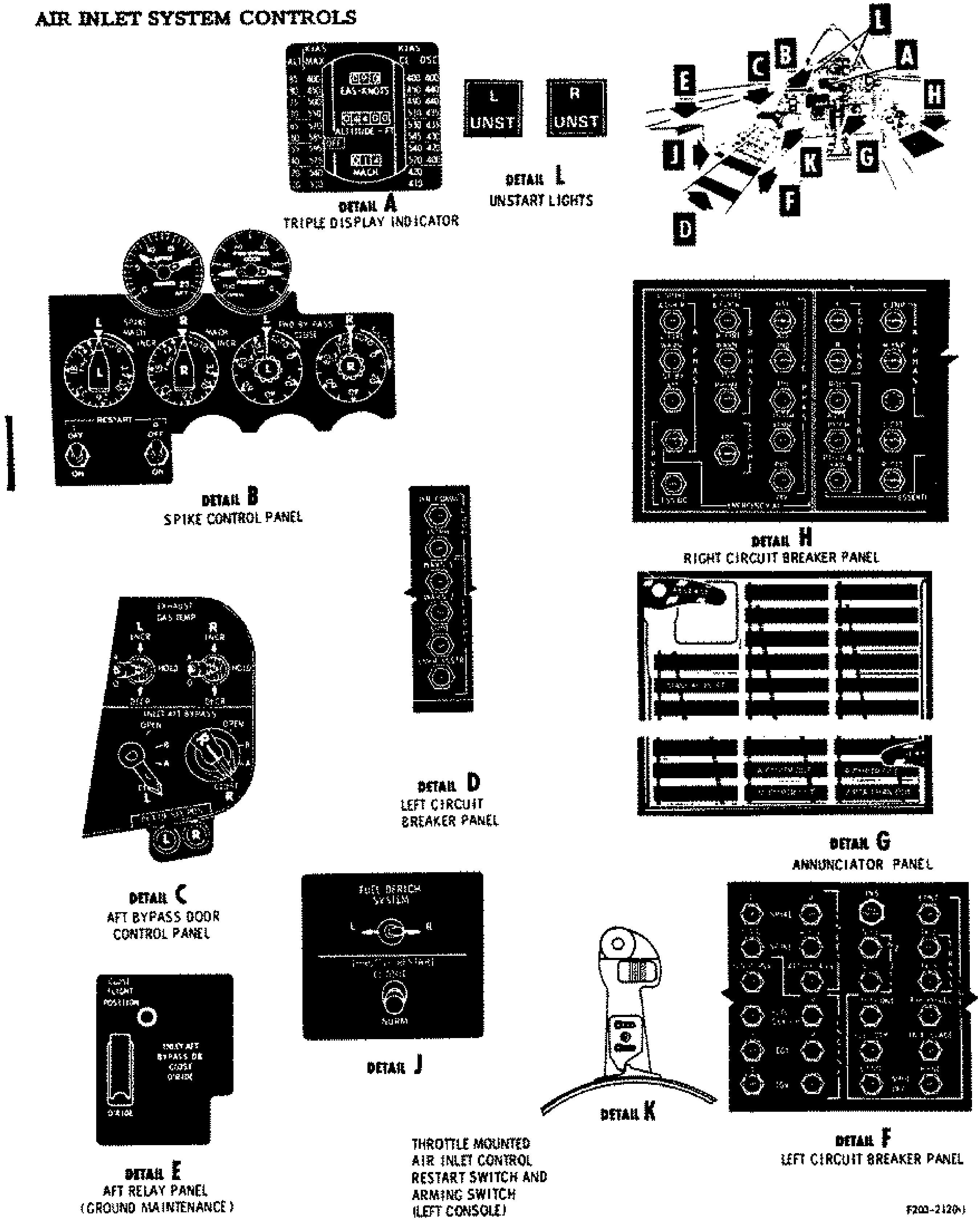


Figure 1-29

AUTOMATIC RESTART

The inlet control includes a shock expulsion sensor (SES) and an auto restart feature which operate automatically above Mach 1.6 (normally, SES is effective above Mach 2.0). If the inlet normal shock wave is expelled, the SES overrides the auto spike and forward bypass schedule (for <u>both</u> inlets above Mach 2.3). This "cross tie" feature keeps the other inlet from unstarting, reduces asymmetric thrust, and minimizes undesirable sideslip angles. During the automatic restart cycle, the forward bypasses open fully and the spikes move forward as much as 15 inches. Spike retraction starts 3.75 seconds after the expulsion is sensed. After the spikes return to their scheduled position, the forward bypasses return to automatic operation. In automatic operation, the forward bypass door duct pressure ratio schedule of the inlet that unstarted is biased slightly lower after each unstart, for up to 4 unstarts. The "cross tie" function is locked out below Mach 2.3 so that only the inlet which has unstarted will perform the automatic restart cycle.

Compressor inlet pressure (CIP) is the SES reference. The SES system actuates when a momentary CIP decrease of more than 23% occurs. Rapid CIP decrease is characteristic of an unstart; however, the SES can also be actuated by compressor stalls if CIP decreases rapidly more than 23%. Successive unstarts or compressor stalls may cause the SES reference pressure (CIP) to decay. The SES cannot operate if the momentary pressure drop is less than 23% of the existing reference pressure. In this event, manual restart will be necessary.

If an inlet remains unstarted (i.e, the shock is not recaptured) after automatic restart, the SES will not actuate since CIP will be low and will not change. In this event, manual restart will be necessary.

The automatic restart and cross tie features do not override any manually positioned inlet control. If an unstart occurs on the inlet in manual spike operation, neither that inlet nor the opposite inlet will respond. With only one forward bypass door in manual control, an unstart on either inlet will result in automatic actuation of both spikes and the door in automatic control. If the manually controlled door is not opened manually, an unstart and possible engine stall can be expected on that inlet, even if the opposite inlet unstarts. Manual operation of the air inlet restart switch overrides the SES automatic restart cycle and manual position settings of that inlet.

INLET CONTROLS, INSTRUMENTS AND INDICATOR LIGHTS

Spike Switches

Individual rotary control knobs for each spike are on the left side of the pilot's instrument panel. Each switch has two detent positions, AUTO and FWD, indexed to the instrument panel lubber line engraved at the 12 o'clock position. The knobs are graduated counterclockwise in Mach from 1.4 through 3.2 to represent the manual override range of control. With the knob in the AUTO detent, the corresponding spike is positioned automatically. The FWD setting moves the spike to its forward stop, bypassing the automatic circuitry. The manual override settings schedule the spike to any desired position from full forward at the Mach 1.4 - 1.6 setting to full aft (26-inch position), at the Mach 3.2 setting. There is no automatic bias for load factor or angles of attack and sideslip during manual control.

NOTE

> Inlet restart switches override all other inlet controls (the forward bypass door opens and the spike is driven forward).

The emergency ac bus provides power for manual and automatic control of the spikes and forward bypass doors through the L and R SPIKE AND DOOR circuit breakers on the pilot's right console. Power for the automatic control circuit relays is furnished from the essential dc bus through the L and R SPIKE circuit breakers on the pilot's left console. Power for spike unlock above 30,000 feet and Mach 1.4 is furnished from the essential dc bus through the L and R SPIKE SOL circuit breakers on the pilot's left console.

Spike Position Indicator

A dual position indicator for the left and right spikes is located on the left side of the pilot's instrument panel. This instrument is marked in one inch increments from 0 (forward) through 26 (full aft). Instrument power is provided from the essential dc bus through the R SPIKE circuit breaker on the pilot's left console.

Inlet Forward Bypass Switches

Two rotary control switches, one for each forward bypass system, are provided on the pilot's instrument panel. Each switch has two detent positions, AUTO and OPEN, indexed to the instrument panel lubber line engraved at the 12 o'clock position. The knobs may be rotated clockwise from the OPEN position to schedule the bypass between full open and full closed. The 100% OPEN detent positions the bypass full open. The manual settings allow the pilot to schedule forward bypass position, overriding the automatic system. Power for the circuit is furnished by the essential dc bus through the L and R SPIKE circuit breakers on the pilot's left console. Power for inlet forward bypass unlock above Mach 1.4 is furnished from the essential dc bus through the L and R SPIKE SOL circuit breakers on the pilot's left console.

MANUAL INLET FORWARD BYPASS BIAS VS. SPIKE KNOB MACH NUMBER

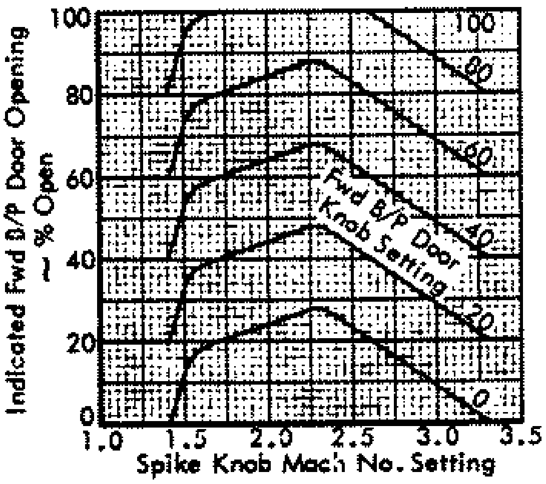


Figure 1-30

Forward Bypass Position Indicator

A dual position indicator for the left and right forward bypass is located on the left side of the pilot's instrument panel. The indicator shows the positions of the forward bypass from 0% (fully closed) to 100% (full open). The dial is marked in 20% increments, with additional marks at each 10% position. Instrument power is provided from the essential dc bus through the L SPIKE circuit breaker on the pilot's left console.

Forward Bypass Bias for Manual Spike Operation

A manual forward bypass bias schedule is applied to the forward bypass whenever the spike and the forward bypass are operated manually. This schedule prevents inadvertent closure of the forward bypass into a range where unstarts are likely. The indicator will show the forward bypass as being more open than the setting selected by the forward bypass switch. With the spike in manual:

a. With the forward bypass switch in AUTO or 100% position - The forward bypass will open fully.

b. With the forward bypass switch in any manual position, other than 100%, the forward bypass will close as far as the bias schedule will permit.

Inlet Aft Bypass Switches and Indicators

Two rotary INLET AFT BYPASS switches, one for each aft bypass system, are located left of the throttle quadrant. The switch positions are: CLOSE, A (15% open), B (50% open), and OPEN (100%). Two rotate-to-dim,amber lights, labeled L and R, are located below the switches. The lights illuminate when an aft bypass position and switch setting do not correspond and extinguish when the aft bypass reaches the setting selected by the aft bypass switches. Approximately 5 seconds is required for the aft bypass ring to move from full closed to full open. The aft bypass actuator control circuits and indicator lights are powered by the essential dc bus through the AFT BYPASS circuit breaker on the pilot's left console.

NOTE

> Occasional momentary illumination of either or both aft bypass out-of-position lights may occur with any switch position selected. This is caused by door actuator drift away from the commanded position limit switch. The door is then commanded back to the selected position and the door out-of-position light extinguishes. Such uncommanded movement of the aft bypass doors are generally too small to be detected by changes in auto inlet position indications, and will not affect aircraft performance.

A ground maintenance switch, labeled INLET AFT BYPASS DR CLOSE O'RIDE, is located on the left console relay panel in the forward cockpit. It overrides the aft bypass close signal when the landing gear is extended.

Inlet Restart Switches

Two inlet restart switches are installed on the lower left edge of the pilot's instrument panel. The two positions are off (center), and RESTART ON (down). When the switch is moved to RESTART ON, the respective inlet spike is moved forward and the forward bypass is positioned full open. Selection of RESTART ON overrides all other spike and forward bypass control settings for that inlet. Power for the respective inlet restart switches is furnished by the essential dc bus through the L and R SPIKE SOL circuit breakers on the pilot's left console.

NOTE

> The forward bypass door linear voltage differential transformer (LVDT) is not temperature compensated. With RESTART ON, the forward bypass is 100% open but the gauge may indicate as low as 80% open.

Throttle Restart Switch

A throttle restart switch, located on the inboard side of the right throttle, restarts both inlets simultaneously. The three-position slide switch can be moved fore and aft by the pilot's left thumb. In the forward (off) position, the inlet spikes and forward bypass are controlled by settings of the inlet spike and forward bypass control switches, or by the individual inlet restart switches. The center position opens the forward bypass of both inlets but does not affect the position of the spikes. The aft position causes the forward bypass to open and the spikes to move full forward in both inlets. This switch is operative only when the throttle restart cutout switch is in NORM. Power for throttle restart switch control of the respective spike and forward bypass positions is from the essential dc bus through the L and R SPIKE SOL circuit breakers on the pilot's left console.

NOTE

> The forward bypass door linear voltage differential transformer (LVDT) is not temperature compensated. With the throttle restart switch in the center or aft position, the forward bypass is 100% open but the gauge may indicate as low as 80% open.

Throttle Restart Cutout Switch

The THROTTLE RESTART CUTOUT switch is on the pilot's left console. It arms the circuit to the throttle restart switch. In the NORM position, the throttle restart switch is operable. If the throttle restart switch malfunctions, the throttle restart cutout switch should be placed in CUTOUT.

Inlet Unstart Lights

The inlet unstart lights, labeled L UNST and R UNST, are located on the pilot's instrument panel. The L UNST or R UNST light illuminates when the shock expulsion sensor (SES) detects an unstart in the corresponding inlet and extinguishes after 3.75 seconds when the SES resets. Power for the lights is furnished by the essential dc bus through the L and R SPIKE SOL circuit breakers on the pilot's left console.

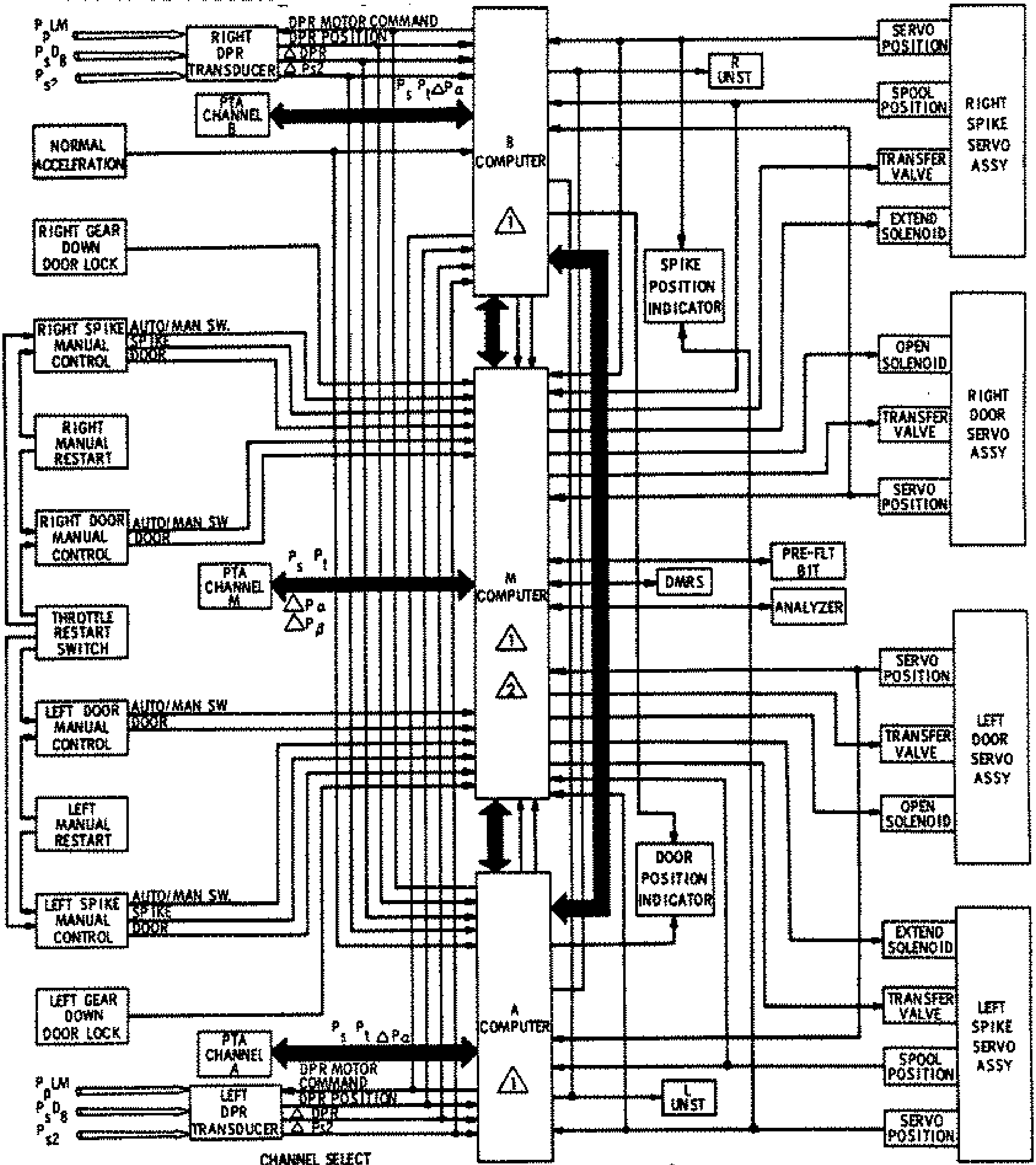
WARNING

> Near design Mach, the inlet may remain unstarted after the L or R UNST light extinguishes.

NOTE

> For multiple unstarts or an inlet which has not restarted, the unstart light(s) will not illuminate unless the subsequent unstart(s) is sensed by the SES.

INLET CONTROL SYSTEM



CHANNEL SELECT

COMPUTER CONDITION	CHANNEL DRIVING SERVOS ⚠1				COMMENTS
	LEFT SPIKE	RIGHT SPIKE	LEFT DOOR	RIGHT DOOR	
NORMAL	A	B	A	B	⚠2
FAIL A	B	B	B	B	⚠2
FAIL B	A	A	A	A	⚠2
FAIL A, B	M	M	M	M	MANUAL CONTROL ONLY

⚠1 Automatic spike and door control commands originate in "A" and "B" channels, but are output by the "M" channel computer.

⚠2 Manual control of individual servos through the "M" channel may be selected.

◄━━► Digital Interface

━━━▷ Pneumatic Pressure

F200-374(a)

Figure 1-31

Manual Inlet Indicator Light

The MANUAL INLET caution light on the pilot's annunciator panel illuminates when one or more of the four rotary spike and/or forward bypass control knobs is not in AUTO, or an inlet restart switch is not in off.

INLET CONTROL SYSTEM

When in AUTO, the left and right spikes and forward bypass doors are controlled by the A and B computers, respectively, output through the M computer to the appropriate servo assemblies. Each computer (A and B) receives nonredundant inputs from its own duct pressure ratio (DPR) transducer assembly, but they share a common normal acceleration signal. Left and right DPR signals are crossfed, so that either A or B computer can control both inlets if the other computer fails. Left inlet auto control requires at least one phase of emergency ac power to the A computer through the COMPUTER circuit breakers in the aft cockpit. Right inlet auto control requires at least one phase of emergency ac power to the B computer through the COMPUTER circuit breakers in the aft cockpit.

Control of inlet unstart lights is direct from each computer (A and B), with cross control if either computer fails.

Manual spike and door control (both inlets) requires M computer A and C phase or B and C phase emergency ac power through the COMPUTER circuit breakers in the aft cockpit.

Left and right spike and door position signals are supplied to forward cockpit indicators. DMRS, PREFLIGHT BIT, and DAFICS Analyzer interfaces are via the M computer. Refer to Figure 1-31 for channel select table.

FUEL SYSTEM

There are five individual fuselage tanks, tanks 1A, 1, 2, 4, and 5, and two wing-fuselage tank groups, tanks 3 and 6. (See Figure 1-32.) Tank 6 is further divided into 6A and 6B. Interconnecting plumbing and electrically driven boost pumps are utilized for fuel feed, transfer, and dumping. Other components of the system include pump controls, nitrogen inerting, scavenging, pressurization and venting, single-point refueling receptacles, and a fuel quantity indicating system. The fuel heat sink system cools cockpit air, engine oil, accessory drive system oil, and hydraulic fluid.

The fuel system moves the aircraft center of gravity by:

a) automatic tank sequencing to the engine feed manifolds to control the fuel level in the various tanks;

b) early depletion of tank 1 fuel to a pre-selected shut-off level, which is dependent upon the zero fuel weight and C.G. position, and whether the flight is subsonic or supersonic;

c) automatic aft fuel transfer into tank 5 when tank 5 pumps are energized and tank 2 contains fuel. In addition, the manual aft transfer system is automatically energized if tank 1 has less than 1500 pounds of fuel, the tank 2 boost pumps are off, and the boost pumps in tanks 3 and 5 are on;

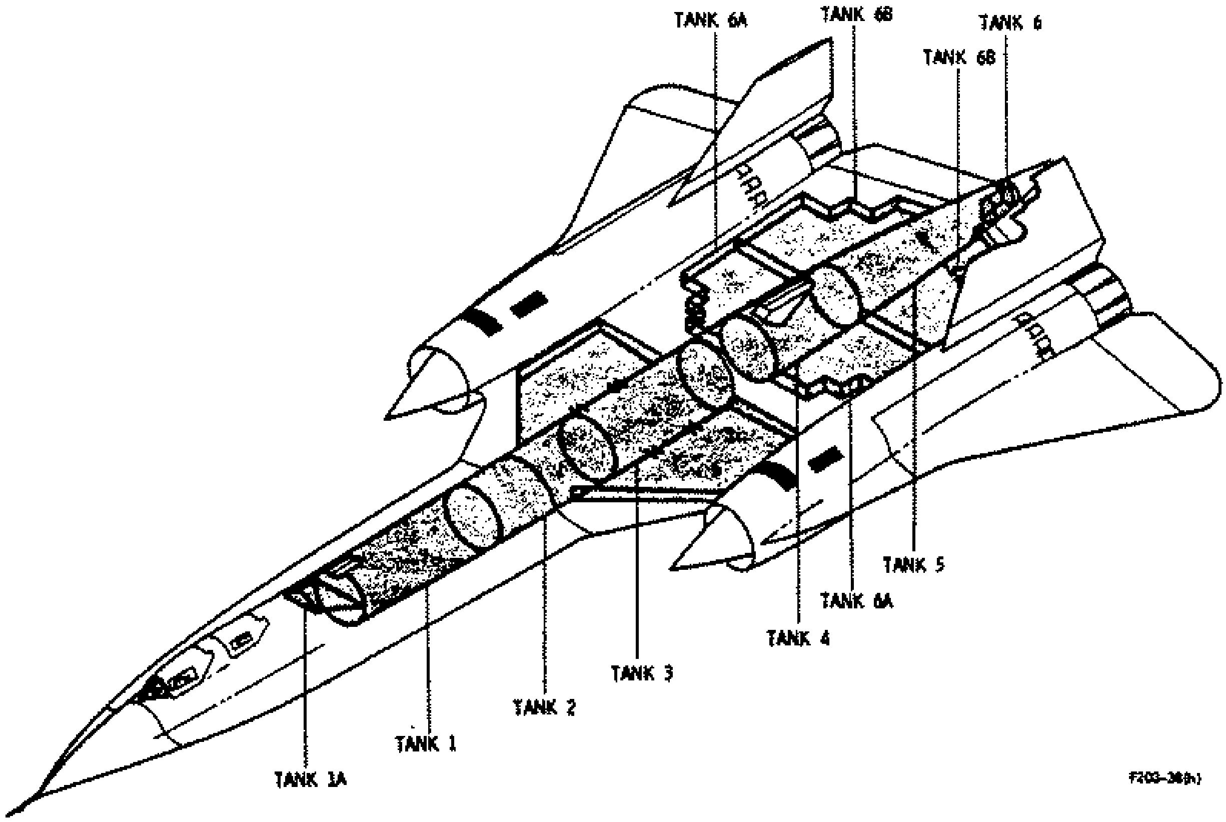
d) manual aft transfer into tank 5 when it is less than full;

e) manual forward transfer to tank 1. See Section 5, Use of Forward Transfer.

For automatic center of gravity scheduling to be effective, the fuel loading distribution given in T.M. SR-71-5 Weight and Balance Manual must be followed.

The automatic fuel feed system is designed to maintain the c.g. within the band depicted in Figure 1-33.

FUEL QUANTITY



FUEL TANK CAPACITIES
Normal Flight Attitude

Tank	Fuel/Gal	Fuel (JP-7)
1A	251.1	1650 lb.
1	2095.9	13770 lb.
2	1974.1	12970 lb.
3	2459.7	16160 lb.
4	1453.6	9550 lb.
5	1758.0	11550 lb.
6A (forward)	1158.3	7610 lb.
6B (Aft)	1068.5	7020 lb.
Total	12219.2	80280 lb. *

* At average fuel density of 6.57 lb./gal.
(46.2° API, Fuel temperature = 78°F)

Figure 1-32

C.G. VS. GROSS WEIGHT
OFF TANKER WITH FULL FUEL LOAD - SUPERSONIC CRUISE - AUTOMATIC FUEL SEQUENCING

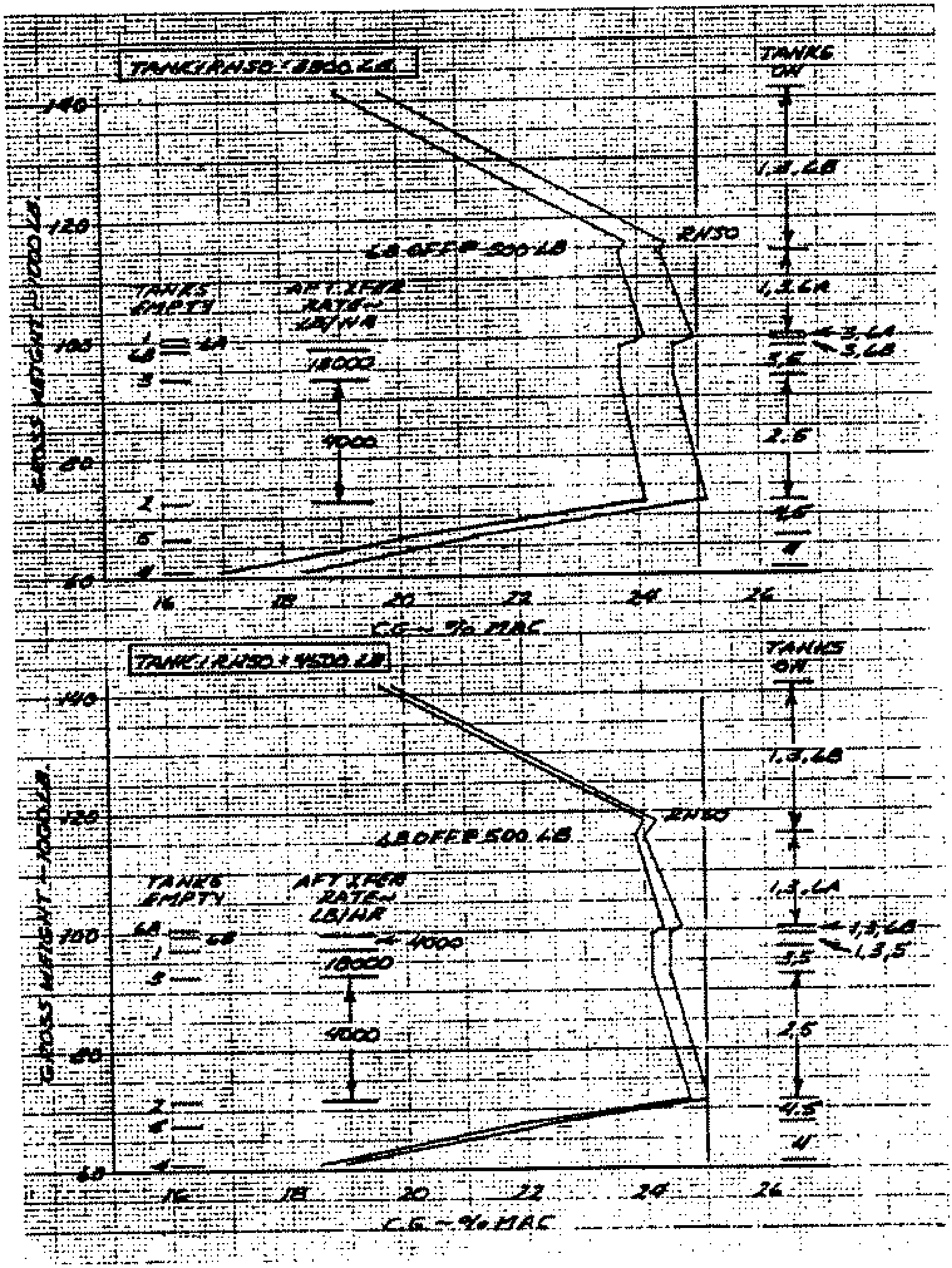


Figure 1-33

ZERO FUEL WEIGHT C.G. BANDWIDTH SUPERSONIC FLIGHT	
RHSO SETTING TANK 1 -LB	ZERO FUEL WEIGHT C. G. RANGE -% MAC
3300	17.0 to 18.4
4500	18.5 to 18.9
5000	19.0 to 19.5
5500	19.6 to 20.1
6000	20.2 to 20.8
6800	20.9 to 21.6
7500	21.7 to 22.3

Figure 1-34

FUEL TANKS

Tanks 1, 1A, 2, 4, and 5 are entirely contained in the fuselage. Tank 1A is a small tank located immediately forward of and feeding into tank 1. Tanks 3 and 6 consist of three and five tank groups, respectively. The No. 3 tank group is comprised of the forward section of each wing and a fuselage tank. The No. 6 tank group is located in the wings on either side of tanks 4 and 5 and includes a small sump tank (approximately 12 gallons) at the extreme aft end of the fuselage which contains the boost pumps for the group. All fuselage tanks are interconnected by a single vent line. Each wing tank is vented to its nearest fuselage tank. Submerged boost pumps are contained in the fuselage tanks. The pumps supply fuel through the left and right manifolds and transfer fuel for center of gravity control. The manifolds can be connected by opening a crossfeed valve. A fuel dump valve is installed in each manifold.

FUEL FEEDING AND SEQUENCING

The left engine is supplied from the left fuel manifold which is normally fed from tanks 1, 2, 3, and 4. The right engine is supplied from the right fuel manifold which is normally fed from tanks 1, 4, 5, and 6. Although crossfeed can be used to feed either engine from any tank, the normal automatic fuel sequencing schedule is:

Left engine supply tanks	Right engine supply tanks	Until
1, 3, 4*	1**, 6, 4*	6 empty
1, 3	1**, 5	1 empty
3	5	3 empty
2	5	2 or 5
2 or 4***	5 or 4***	empty
4	4	See note.

* If tank 4 is full initially, pumps 4-1 and 4-2 operate briefly to provide approximately 850 pounds ullage space. Tank 4 indicator light will not illuminate.

** Termination of tank 1 supply to the right engine depends on the right-hand shutoff float switch selection.

*** Depending on the right-hand shutoff float switch selection for tank 1, tank 2 may or may not shut off before tank 5 is depleted.

NOTE

Forward transfer is necessary during subsonic operation to maintain c.g. ahead of the aft limit.

Tank sequence is controlled automatically by float switches, see Figure 1-35. These switches optimize the center of gravity for supersonic cruise.

FUEL BOOST PUMP AND FLOAT ARRANGEMENT

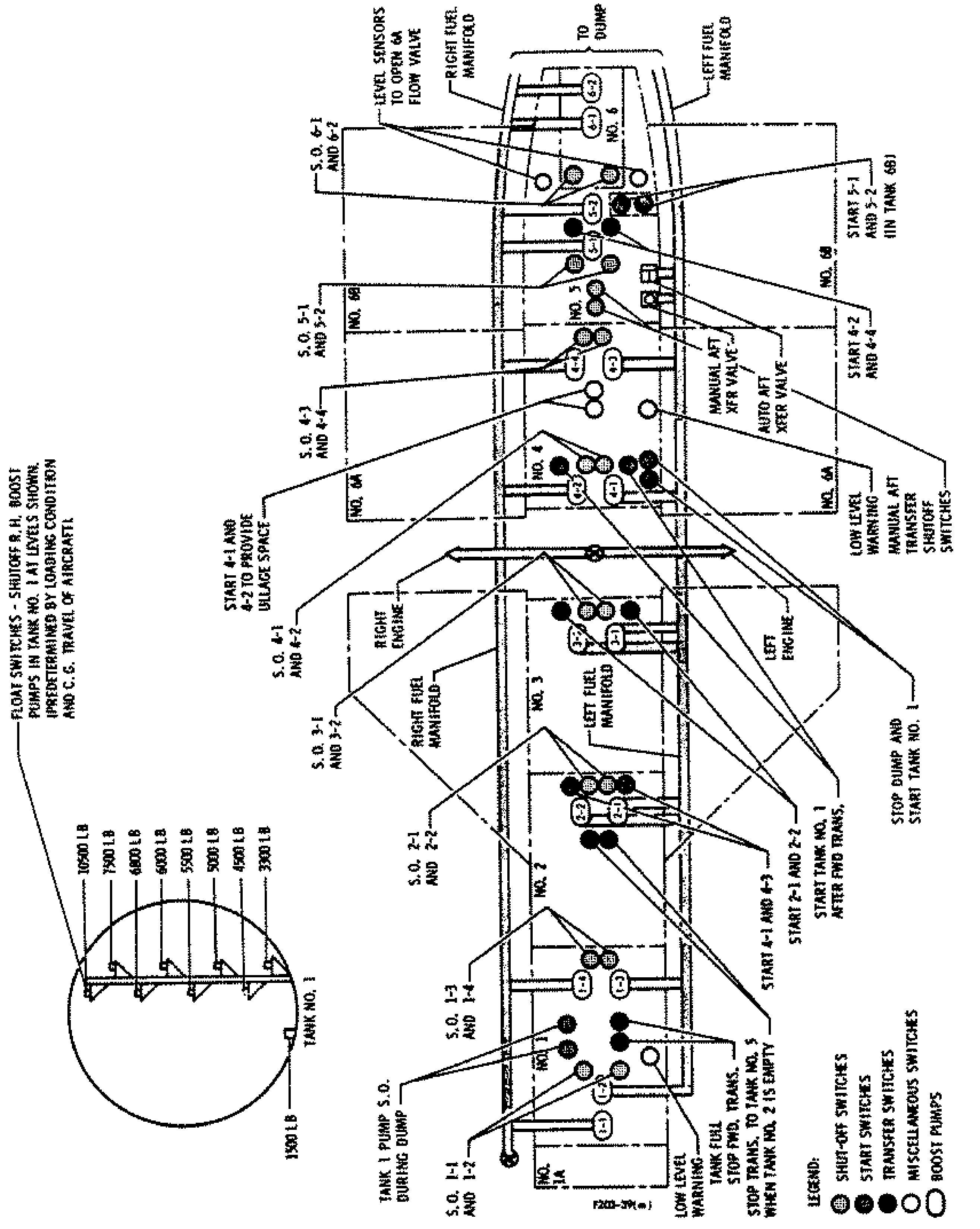


Figure 1-35

ATTITUDE EFFECTS ON FUEL TANK FLOAT SWITCH ACTUATION

TANK NO.	SWITCH	ANGLE OF ATTACK DEG	
		0	+6.2
		Tank Fuel Remaining - Lb (6.7 lb/gal.)	
1	STOP FORWARD TRANSFER	13300	13700
	TANK 1 PUMP SHUTOFF DURING DUMP	4550	4850
	LOW LEVEL WARNING	4200	5400
	SHUTOFF 1-1 AND 1-2	1650	2850
	SHUTOFF 1-3 AND 1-4	400	65
2	STOP TRANSFER TO TANK 5 WHEN TANK 2 IS LOW	600	400
	START 4-1 AND 4-3	500	100
	SHUTOFF 2-1 AND 2-2	300	65
3	START 2-1 AND 2-2	500	175
	SHUTOFF 3-1 and 3-2	250	75
4	START 4-1 AND 4-2 TO PROVIDE ULLAGE SPACE	9550	9400
	LOW LEVEL WARNING	3750	4050
	STOP NORMAL DUMP AND START TANK NO. 1	3500	3950
	START TANK NO. 1 WITH FORWARD TRANSFER ON	400	1250
	SHUTOFF 4-3 AND 4-4	300	75
	SHUTOFF 4-1 AND 4-2	250	1100
5	MANUAL AFT TRANSFER SHUTOFF	10950	10950
	START 4-4	1700	350
	START 4-2	1600	300
	SHUTOFF 5-1 AND 5-2	1400	200
6	6A TO 6B FLOW CONTROL VALVES BEGIN SCHEDULING	10900	8100
	START 5-1 (SWITCH IN TANK 6B)	6475	300
	START 5-2 (SWITCH IN TANK 6B)	5200	150
	SHUTOFF 6-1 AND 6-2 (SWITCH IN TANK 6 SUMP TANK)	10	20

Figure 1-36

FUEL FEED SYSTEM

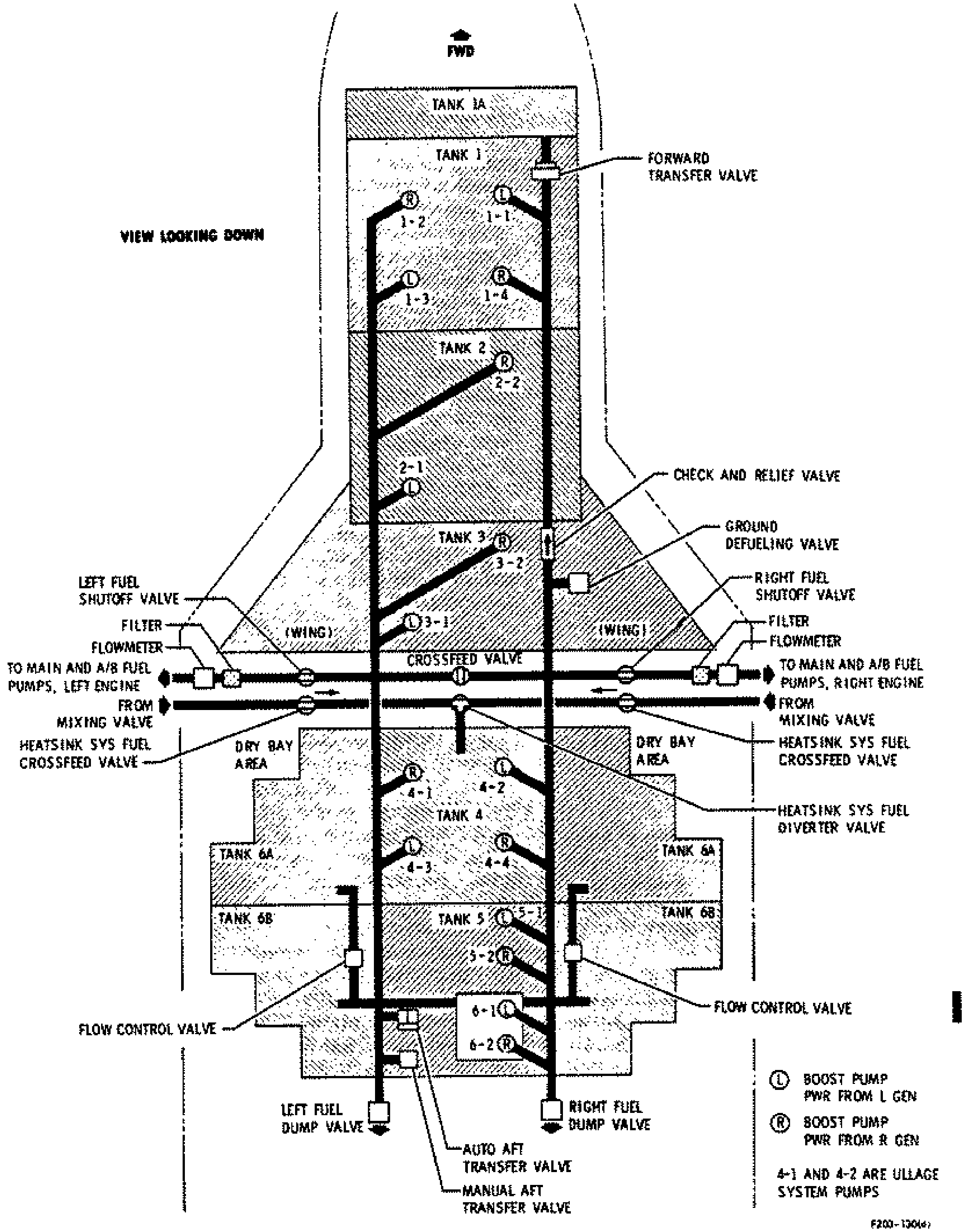


Figure 1-37

A stack of float switches, called right-hand shutoff switches, is installed in tank 1. These switches allow the automatic c.g. control system to compensate for variations in aircraft weight and c.g. due to sensor loading. The selection of a float switch determines the level of fuel remaining in Tank 1 at which boost pumps 1-1 and 1-4 (which supply the right fuel manifold) are shut off. Since boost pumps 1-2 and 1-3 continue to supply the left fuel manifold, the rate of fuel flow from tank 1 is reduced approximately 50% when shutoff occurs. This stops the rapid aft shift of c.g. experienced when tank 1 is supplying both fuel manifolds. Setting of the right-hand shutoff switches is accomplished by maintenance personnel prior to flight. Float switch selection is based on aircraft zero fuel weight and c.g. position as scheduled in the SR-71-5 Handbook of Weight and Balance Data.

NOTE

> Recommended weight and/or c.g. limits can be exceeded by seemingly normal loading arrangements. Check loading documents carefully.

Eight settings are provided between 3300 pounds and 10,500 pounds of fuel remaining in tank 1. Proper selection results in a schedule for c.g. in a band that closely approaches the aft limit during cruise.

The preselected float switch setting is overridden if tank 4 contains less than 3600 ±1200 lb of fuel. In this case, tank 1 supplies fuel to both manifolds until the tank 1 fuel low level switches shut off the boost pumps.

The tank 6 boost pump sump tank is fed by tank 6B. Two flow control valves are located between tanks 6A and 6B. When the valves are open, fuel in tank 6A can flow into tank 6B. The tank 6 flow control valves are held open by fuel pressure provided by the tank 6 boost pumps or by pump 5-1. Fuel pressure from the tank 6 boost pumps is provided to the valves only when the quantity of fuel in tank 6B is below approximately 500 pounds at cruise attitude. If pump 5-1 is not on, the valves function to maintain approximately 500 pounds of fuel in tank 6B until tank 6A is empty. If pump 5-1 is on, the valves are open regardless of the fuel quantity in tank 6B.

Because of the aft position of the tank 6 outlets, tank 6 may stop feeding if a nose down flight attitude is established while fuel remains in tank 6 wing tanks. A nose-down attitude and/or a deceleration shifts fuel forward in tanks 6A and/or 6B and uncovers the tank outlets. (Normally, the tank 6 group empties during climb and level cruise). With the wing tank outlets uncovered, fuel does not flow to the sump tank which contains the tank 6 boost pumps. As a result, tank 5 is sequenced on by the low level float switch in the aft portion of tank 6B. The tank 6 pumps are shut off when the sump tank empties, even though a substantial amount of fuel may remain in the wing tanks. An unusual c.g. condition can result if the incorrect sequencing persists. The c.g. may also move out of the desired range if a substantial mismatch of fuel flow to the engines exists (see Figure 2-8). Refer to Fuel Management in Section II.

FUEL BOOST PUMPS

Sixteen single-stage, centrifugal, ac-powered boost pumps supply the fuel manifolds. (See Figures 1-35 and 1-37). Tanks 1 and 4, which normally feed both engines, are equipped with four pumps each and tanks 2, 3, 5, and 6 have two pumps each. Either pump of a pair is capable of supplying sufficient pressure to permit engine operation at reduced afterburning thrust if the other pump fails. Two pumps are required for maximum afterburning fuel flow at lower altitudes. The pumps may be manually operated by use of individual tank boost pump control switches, located on the pilot's right instrument side panel. Manual control of the tank pumps supplements but does not terminate automatic tank sequencing. Manual selection of any tank pumps will change the programmed c.g. schedule and may cause a serious c.g. condition to develop. The boost pumps are cooled and lubricated by the fuel; therefore, manual activation of pumps should be terminated (by pressing the pump release switch) when the tank is empty.

[redacted]

In automatic operation, each pump is protected by a float switch that deactivates the pump when the tank is empty. Individual circuit breakers for each pump are loacted in the E-bay and are not accessible in flight. Three-phase ac power for the pumps is furnished by the generator buses. Odd-numbered pumps (except 4-1) are powered by the left generator ac bus and even numbered pumps (except 4-2) are powered by the right generator ac bus.

NOTE

- In tank 4, pump 4-2 and 4-3 are powered by the left generator bus and pump 4-1 and 4-4 are powered by the right generator bus. This precludes flameout due to fuel starvation if a single generator fails while the bus tie is split.
- If the left engine generator fails and the bus tie is split, fuel in tank 6A may become trapped due to insufficient tank 6 pump pressure and the unavailability of pump 5-1. Pump 5-1 pressure is ported directly to the level control and flow valves controlling tank 6A fuel feed.

FUEL TRANSFER SYSTEM

Fuel transfer systems control aircraft c.g. A manual forward transfer system and an automatic and a manual aft transfer system are provided.

Forward Transfer

Fuel may be transferred forward through the right fuel feed manifold from tanks 4, 5, and 6 into tank 1 at a maximum rate of approximately 950 lb/min. Fuel may be transferred from the left fuel feed manifold (tanks 2 and 3) by opening the crossfeed valve. Fuel transfer is initiated by the forward transfer valve in the forward end of the right fuel feed manifold opening into tank 1. This valve is controlled by the FWD TRANS fuel transfer switch. With RHSO settings of 6000 lbs or higher, forward transfer may be required more often.

Automatic Aft Transfer

Fuel can be transferred from the left fuel feed manifold into tank 5 either by automatic or manual operation. The automatic aft transfer system is started when the tank 5 boost pumps are energized if tank 2 contains fuel above the level of its low fuel quantity float switch. The automatic aft transfer stops when the tank 5 boost pumps are de-energized or when tank 2 fuel level reaches the low fuel quantity float switch. The rate of fuel transfer is determined by throttle position. The fuel automatic aft transfer rates are: 65 lb/min (4000 lb/hr) when both throttles are in afterburner, 23 lb/min (1400 lb/hr) otherwise. In addition, the 233 lb/min (14,000 lb/hr) manual aft transfer rate is automatically added when:

1. Tank 1 contains less than 1500 lbs, and
2. Tank 2 contains fuel, but it's boost pumps are not on, and
3. Tank 3 and 5 boost pumps are on.

Manual Aft Transfer

The manual aft transfer system permits the pilot to transfer fuel into tank 5 from any tank(s) whose pumps are supplying the left fuel manifold.

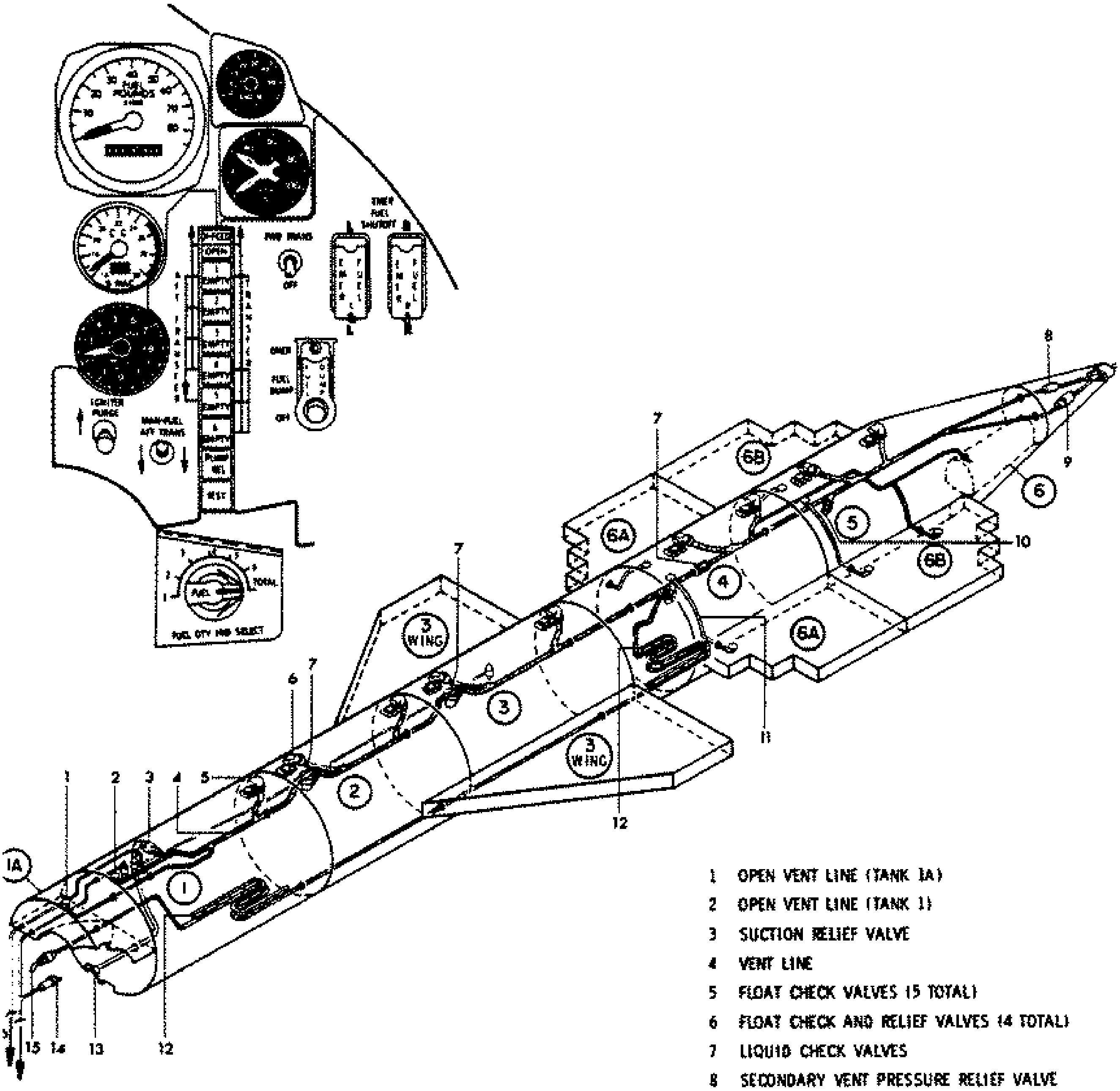
The manual aft transfer rate is approximately 233 lb/min (14,000 lb/hr).

NOTE

Forward transfer operates at a rate of approximately 950 lb/min, a rate more than sufficient to overcome the effects of automatic and manual aft transfer.

[redacted]

FUEL SYSTEM - Pressurization



1 OPEN VENT LINE (TANK 1A)
2 OPEN VENT LINE (TANK 1)
3 SUCTION RELIEF VALVE
4 VENT LINE
5 FLOAT CHECK VALVES (5 TOTAL)
6 FLOAT CHECK AND RELIEF VALVES (4 TOTAL)
7 LIQUID CHECK VALVES
8 SECONDARY VENT PRESSURE RELIEF VALVE
9 PRIMARY VENT PRESSURE RELIEF VALVE
10 VENT DRAIN VALVE
11 VENT LINE, WING TANK TO TOP OF FUSELAGE TANK (4 TOTAL)
12 FUEL LINE FROM FUEL PUMP OUTLETS
13 SUCTION RELIEF ORIFICE (NOSE WHEEL WELL)
14 TANK PRESSURE TRANSMITTER
15 LN_2 FLOW FROM DEWAR TANKS
16 TO NITROGEN TANK PRESSURE SENSORS

Figure 1-38

FUEL HEAT SINK SYSTEM

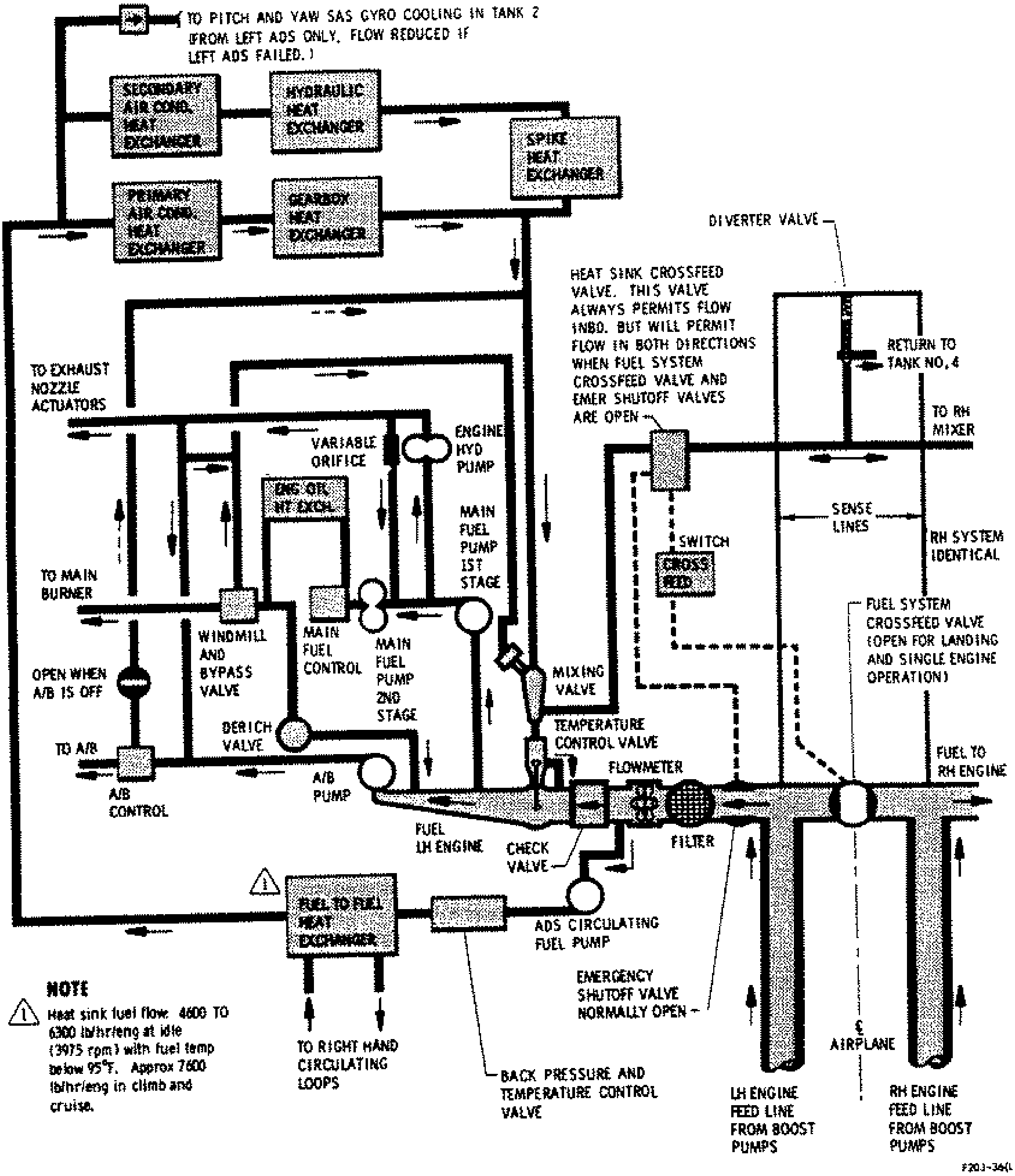


Figure 1-39

Manual aft transfer is controlled by the MAN-FUEL AFT TRANS switch on the pilot's right instrument side panel. Two ullage float switches automatically terminate manual aft transfer if the switch is held on while tank 5 is full; however, automatic aft transfer is not stopped.

FUEL TANK PRESSURIZATION SYSTEM

The fuel tank pressurization system (Figure 1-38) consists of indicators, three Dewar flasks and associated valves and plumbing to the fuel tanks. Two Dewar flasks, each containing 106 liters of liquid nitrogen, are located in the nosewheel well. The third Dewar flask, containing 50 liters of liquid nitrogen, is installed in the left forward chine (B bay). The nitrogen flasks are equipped with automatic ac-powered heaters to change the liquid nitrogen to gas. The nitrogen from the flasks is routed through heat exchangers (in tanks 1 and 4) to ensure that the nitrogen has become gaseous. The nitrogen gas is then ported to a common vent line and to the top of all tanks. The nitrogen gas pressurizes each fuel tank to 1.5 (±0.25) psi above ambient pressure and inerts the ullage space above the heated fuel to prevent autogenous ignition.

The fuel tank pressurization system provides 1.5 (±0.25) to 3.25 (±0.25) psi differential pressure to fuel tanks for a more positive fuel supply to the engines. Fuel tank positive pressure is prevented from exceeding 4.15 psi by a secondary pressure relief valve in the tail cone vent line.

Positive tank pressure is maintained in three ways:

1. The inerting system allows nitrogen flow into the tanks whenever fuel tank pressure differential drops below 1.5 (±0.25) psi.
2. Fuel vapor pressure tends to maintain 1.5 (±0.25) to 3.25 (±0.25) differential during supersonic cruise when fuel warms up.
3. Ascent into less dense atmosphere causes the pressure differential in the tanks to increase to the setting of the pressure relief valve of 3.25 (±0.25) psi at which time the tank will continue venting until level-off.

A small quantity of nitrogen is required for taxi, runup, and takeoff. After takeoff, little or no nitrogen is required until descent for refueling or landing. As the aircraft descends, atmospheric pressure increases causing a demand on the nitrogen pressurization system to keep the internal pressure of the tank higher than the increasing external pressure. For reliability, the nitrogen system is separated into two independent systems, each of which is capable of supplying the required flow for a normal descent. Two liquid nitrogen quantity indicators on the pilot's right instrument side panel indicate the quantity remaining in each flask. The system 3 liquid nitrogen quantity indicator, marked LIQ N, is above the quantity indicator for systems 1 and 2. Fluctuations of the system 3 gage are normal.

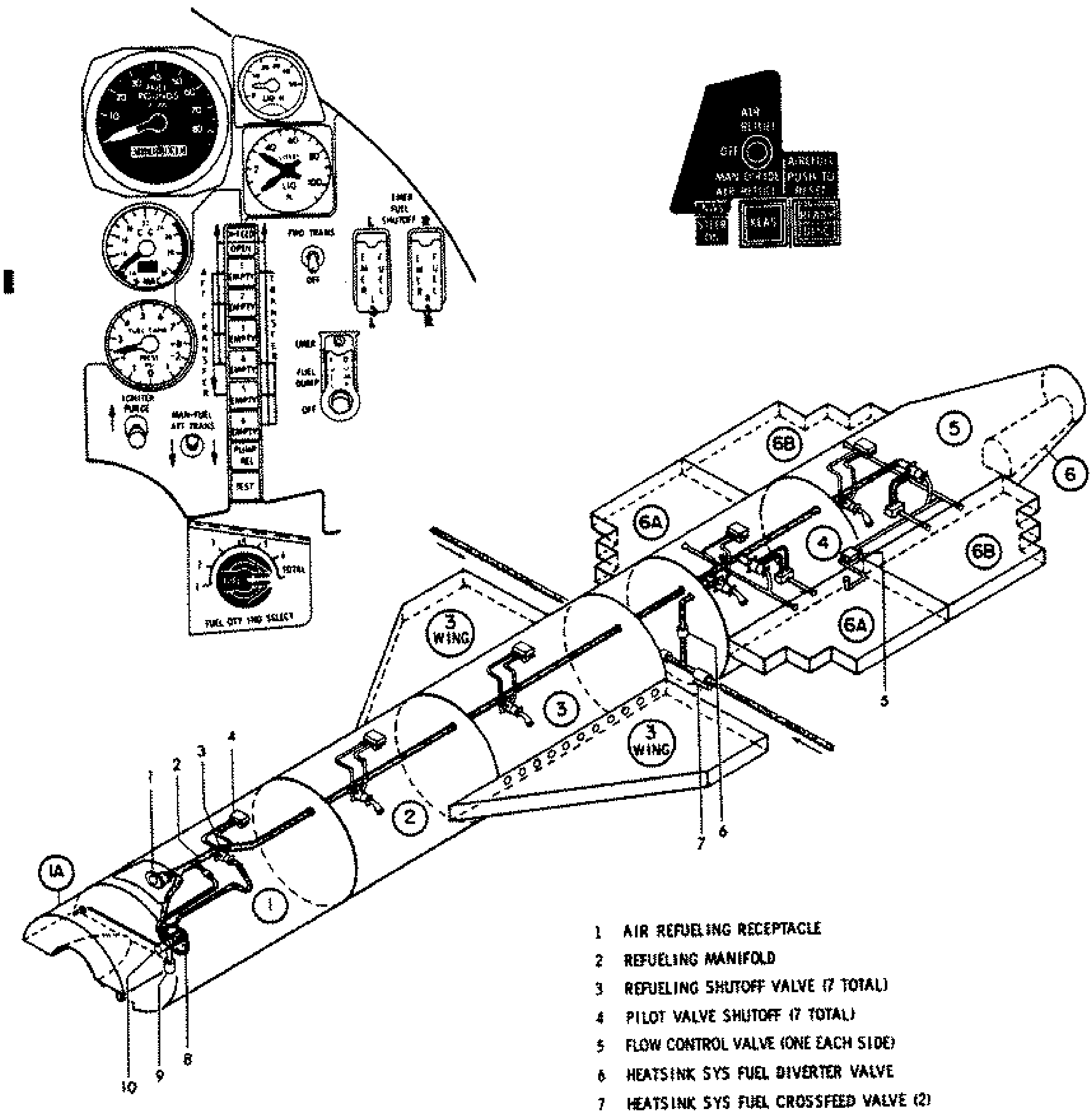
NOTE

> The nitrogen systems may deplete at an uneven rate; consequently, the quantity gages may show different amounts remaining.

FUEL HEAT SINK SYSTEM

Fuel is used to cool the air-conditioning systems, the aircraft hydraulic fluid, and engine and accessory drive system oil. (See Figure 1-39.) Circulated fuel also cools the TEB tank and the control lines which actuate the afterburner nozzle. Engine oil is cooled by main engine fuel flow through an oil cooler, located between the main fuel control and the windmill bypass valve. This fuel is then directed to the main burner section. The other cooling is accomplished by fuel circulation through several cooling loops. If within engine consumption requirements, the

FUEL SYSTEM - Refueling



1 AIR REFUELING RECEPTACLE
2 REFUELING MANIFOLD
3 REFUELING SHUTOFF VALVE (7 TOTAL)
4 PILOT VALVE SHUTOFF (7 TOTAL)
5 FLOW CONTROL VALVE (ONE EACH SIDE)
6 HEATSINK SYS FUEL DIVERTER VALVE
7 HEATSINK SYS FUEL CROSSFEED VALVE (2)
8 TORUS FUEL SPRAY RING
9 GROUND REFUELING RECEPTACLE
10 TANK 1A REFUEL AND FLOW VALVE

F203-13(5)(F)

Figure 1-40

hot fuel returning from the accessory drive system heat exchanger, the primary and secondary air-conditioning heat exchangers, the hydraulic fluid heat exchanger, the spike heat exchanger, and the exhaust nozzle actuators is circulated through a mixing valve and temperature limiting valve and returned to the main engine and afterburner fuel manifold. The quantity in excess of engine requirements is diverted to tank 4.

NOTE

> When cooling loop fuel temperature is less than 96°F, flow through the loop should be between 4600 and 6300 pph at idle rpm. Loop flow increases to approximately 7600 pph at military rpm. The cooling loop flow automatically increases approximately 3600 pph as a result of temperature control valve operation when loop fuel temperature is above 96°F. Loop flow and cockpit fuel flow indication are equal until engine consumption becomes greater than flow through the loop. Excess flow is returned to the fuel tanks when engine consumption is less than cooling loop flow.

If the temperature of the mixed cooling loop and incoming engine fuel exceeds 290°F, the temperature control valve starts to close and some of the cooling loop fuel is prevented from mixing with the incoming engine fuel. A pressure-operated valve routes the hot fuel to tank 4. The temperature control valve is completely closed at 300°F and all cooling loop fuel is returned to tank 4. If tank 4 is full, the return fuel will be diverted to the next tank that has space for it. During single engine operation with the inoperative engine throttle in OFF, actuation of the fuel crossfeed valve allows the hot recirculated fuel from the windmilling engine to cross over and mix with the cooling loop and incoming fuel for the operating engine. If within engine consumption requirements and if the mixed fuel temperature is below 290°F, all of the hot fuel will be burned by the operating engine and afterburner. If the mixed fuel temperature is above 300°F, all hot fuel from both engines is returned to tank 4 or to the next tank with space available. Placing the emergency fuel shutoff switch to the shutoff (up) position terminates heat sink fuel to the windmilling engine. When either of the fuel shutoff valves is closed, the corresponding heat sink crossfeed valve is deenergized to prevent fuel circulation through the inoperative engine.

AIR REFUELING SYSTEM

The air-refueling system can receive fuel at approximately 6000 pounds per minute from KC-10 or KC-135 boom-equipped tanker aircraft. Calibrated orifices for each tank allows all tanks to be filled simultaneously in 12 to 15 minutes with a refueling pressure of 65-70 psi. A shutoff pilot valve in each tank terminates refueling flow when the tank or tank group is full. The air refueling system consists of a boom receptacle, receptacle doors, hydraulic valves, hydraulic actuators, a signal amplifier, control switches, and panel indicator lights. The refueling doors are held closed by hydraulic pressure; if hydraulic pressure is lost, as when the engines are shut down on the ground, the doors open by spring action. This enables air refueling if both the L and R hydraulic systems fail. The system normally requires hydraulic actuating power from the L hydraulic system to operate the doors and boom receptacle. If L hydraulic pressure is below 2200 psi, the refueling doors and receptacle can be operated by R system pressure if the brake switch is in ALT STEER & BRAKE. Electrical power is required from the essential dc bus for operation of the controls and indicators.

Unless the SR-71 receiver is using the manual boom latching procedure for refueling (air refueling switch in MAN O'RIDE) the boom will automatically disconnect if fuel pressure exceeds 70 psi. Pressure disconnect is normal when tanks reach full if refueling with a KC-135 tanker. Because the KC-10 refueling system automatically reduces flow to maintain normal refueling pressure, pressure disconnect does not normally occur if refueling with a KC-10 tanker.

FUEL SYSTEM CONTROLS AND INDICATORS

Crossfeed Switch

The push-button crossfeed switch is mounted above the column of fuel boost pump switches. When depressed, a motor operated valve between the left and right fuel manifolds opens to join the two fuel manifolds, so either or both manifolds can feed either or both engines. The fuel heat sink systems are also interconnected. The crossfeed switch must be depressed a second time to terminate crossfeed operation. The legend XFEED illuminates as the valve starts to open and an OPEN legend illuminates when the valve is fully opened. The OPEN legend extinguishes when the valve starts to close, but the XFEED legend remains on until the valve is fully closed. Circuit control power is furnished by the essential dc bus through the fuel CONT circuit breaker on the pilot's left console. Three phase power for the crossfeed valve is furnished by the essential ac bus through circuit breakers in the E bay.

Fuel Boost Pump Switches and Indicator Lights

Six self-illuminated, square plastic fuel boost pump pushbutton switches are installed in a vertical line on the pilot's right instrument side panel. The switches control manual operation of the fuel boost pumps. The switches have an electrical hold and bail arrangement that allows manual selection of only one tank of tank group 1, 2, 3 and one tank of tank group 4, 5, 6 at the same time.

NOTE

> Manual operation supplements, but does not terminate, automatic fuel tank sequencing.

When a set of boost pump relays are actuated, either automatically or manually, a clear numeral on an illuminated green background in the upper half of the pushbutton illuminates. When a tank is empty, an amber EMPTY light in the lower half of the push button illuminates and that pump group stops, unless manually selected.

NOTE

> If all the fuel is used from a manually selected tank, its EMPTY light will not illuminate until the tank which normally turns on that tank is also empty.

Automatic operation of the ullage system pumps 4-1 and 4-2 does not affect the Tank 4 indicator light. The Tank 4 indicator light only illuminates if pump 4-3 or 4-4 (or both) is on. In automatic operation, pumps 4-3 and 4-1 are turned on by individual float switches in Tank 2; and pumps 4-4 and 4-2 are turned on by individual float switches in Tank 5.

When manually depressed, the boost pump switch will hold down electrically and the pumps continue to operate until the pump release switch is pressed. Power for the boost pump switch circuits is furnished by the essential dc bus through the fuel CONT circuit breaker on the pilot's left console. Power for the indicator lights is furnished by the ac hot bus through the INSTR light circuit breaker on the pilot's right console and the FUEL CONT circuit breaker on the light control panel on the pilot's left console.

NOTE

> Pulling the TK 5 TRANS circuit breaker will disable tank 2, 3, and 5 fuel boost pump indicator lights. The fuel boost pumps are not disabled.

Pump Release Switch

A push-button PUMP REL switch is located below the fuel boost pump switches. When the PUMP REL switch is depressed, any boost pump switch that has been manually actuated is released. Power for the circuit is furnished by the essential dc bus through the fuel CONT circuit breaker on the pilot's left console.

CAUTION

A manually selected boost pump should be released when the tank indicates EMPTY.

Tank Lights Test Switch

A push-button tank lights TEST switch, is below the pump release switch. When the switch is depressed, the fuel boost pump lights, crossfeed, pump release, and test lights illuminate. Power is furnished by the ac hot bus through the INSTR light circuit breaker on the pilot's right console and the FUEL CONT TEST circuit breaker on the pilot's light control panel.

Fuel Forward Transfer Switch

The two-position forward transfer switch, labeled FWD TRANS (up) and OFF (down), is located on the pilot's right instrument side panel. If tank 1 is not full, moving the switch to FWD TRANS shuts off all tank 1 boost pumps and opens the forward transfer valve, allowing fuel to transfer into tank 1. Forward transfer is stopped automatically, when tank 1 is full, by the tank 1 "full" float switches; however, tank 1 boost pumps will not resume operation until the forward transfer switch is OFF or tank 4 is almost empty. Power for the circuit is furnished by the essential dc bus through the TK 1 TRANS circuit breaker on the pilot's left console.

Fuel Forward Transfer Light

With S/B R-2691, a FWD TRANSFER light on the RSO's instrument panel illuminates when the fuel forward transfer switch in the forward cockpit is in FWD TRANS. Power for the light is from the TRANS TK1 circuit breaker on the pilot's left circuit breaker panel.

Fuel Aft Transfer Switch

The fuel aft transfer switch, located on the pilot's right instrument side panel, is spring-loaded to the off (up) position. Holding the switch in the down position operates a solenoid controlled valve in the left fuel manifold, if tank 5 is not full, and allows fuel to enter tank 5. If tank 5 is full, the two float switches prevent the solenoid valve from operating. Control power for the manual aft transfer valve solenoid is from the essential dc bus through the TK 5 TRANS circuit breaker on the pilot's left console.

NOTE

Pulling the TK 5 TRANS circuit breaker will disable tank 2, 3, and 5 fuel boost pump indicator lights. The fuel boost pumps are not disabled.

Fuel Dump Switch

A guarded, three-position, lift-lock FUEL DUMP switch is installed on the pilot's right instrument side panel. The switch positions are: EMER (up), FUEL DUMP (center), and a guarded OFF (down) position. The switch must be pulled out and up to move to the EMER position. In the FUEL DUMP position, dual solenoid dump valves in each fuel manifold open to commence dumping. All fuel tanks continue to feed in automatic sequence until tank 1 reaches approximately 4700 pounds, depending on aircraft attitude (see Figure 1-36), then tank 1 boost pumps stop. Fuel pumps in all other tanks continue to operate in automatic sequence until tank 4 reaches approximately 3700 pounds (again, a function of aircraft attitude). At 3700 pounds, fuel dump ceases and, if there is any fuel in tank 1, tank 1 boost pumps will start. With the switch in the EMER position, dumping is identical <u>except</u> that at the 3700 pound level in tank 4, fuel dump does not cease and dumping will continue to empty tanks.

WARNING

Emergency fuel dumping must be terminated by positioning the dump switch to OFF (or FUEL DUMP) or all tanks will empty.

The nominal dump rate is 2500 pounds per minute for both FUEL DUMP and EMER switch positions, but the rate varies with the amount of fuel remaining and the number of boost pumps operating. (Refer to Section II, Fuel Dumping). Power for the circuit is furnished by the essential dc bus through the fuel DUMP CONT circuit breaker on the pilot's left console.

A switch in the forward cockpit indicator causes the CG annunciator warning light in both cockpits to illuminate when cg reaches 25.3% to 25.6% aft cg or 16.4% to 17.0% forward cg. Power is furnished to the indicators by the ac hot bus through a FUEL QTY circuit breaker in each cockpit. An OFF warning flag is displayed only on the affected indicator if power is interrupted. If power to the forward cockpit indicator is lost, cg indication on both indicators will remain at the cg shown at the time of power interruption, but the aft cockpit indicator will not display an OFF warning flag.

NOTE

- An erroneous cg indication can be expected during steady sideslip, as during single-engine operation, when fuel remains in tank 6.
- Fluctuations of the cg indications of about 5 percent can be expected with keying or modulation of the HF transmitter.

CG Indicator Mode Selector

A mode selector for the cg indicator is located on a bulkhead aft and left of the seat in the forward cockpit. Values representing reference fuel density, aircraft weight without fuel, and the corresponding moment value must be set with the mode selector knobs. The mode selector control settings can be viewed by the pilot when entering the aircaft, but re not accessible in flight.

NOTE

The correct cg indicator mode selector settings must be set on the ground to obtain a proper cg indication in-flight. If the correct settings are not made, the cg indicator will appear to operate properly, but will be erroneous, and manual cg computations will be necessary.

Fuel Low Pressure Lights

Warning lights for each engine, labled L and R FUEL PRESS, are located on the pilot's annunciator panel. The light illuminates when fuel pressure in the respective main fuel manifold decreases to less than 7 (± 1/2) psi. The light extinguishes when fuel pressure rises above 10 psi.

NOTE

The L and/or R FUEL PRESS warning light(s) may illuminate sporadically during fuel dumping. The light for the left manifold may illuminate during manual aft transfer when tank sequencing occurs. The light for the right side may illuminate during forward transfer.

Fuel Quantity Low Light

A FUEL QTY LOW caution light on the pilot's annunciator panel is illuminated by the closing of low-level float switches in tanks 1 and 4. The switches are connected in series and both must close to illuminate the caution light. The switch in tank 1 closes when the fuel level has dropped below 4200 pounds at 0 degrees pitch angle and below 5400 pounds at +6.2 degrees pitch angle. The tank 4 switch closes when the fuel level has dropped below 3750 pounds at 0 degrees pitch angle and below 4050 pounds at +6.2 degrees pitch angle. Therefore, the light may illuminate at any condition of tanks 1 and 4 between 3750 pounds (0-degree pitch angle, fuel in tank 4 only) and 9450 pounds (+6.2 degree pitch angle and fuel in both tanks). This light will stay on after initial sensing and may be reset by moving the refuel switch to AIR REFUEL, then to OFF.

Fuel Tank Pressure Indicator

A fuel tank pressure indicator is installed on the pilot's right instrument side panel. The gage senses tank 3 pressure, which is common to all tanks, and is marked from -2 to +8 in increments of 1 psi. Power is supplied by the essential ac bus 26 volt instrument transformer through the FUEL TK PRESS circuit breaker on the pilot's annunciator panel.

Fuel Tank Low Pressure Light

A TANK PRESS warning light is located on the pilot's annunciator panel. The light illuminates when tank pressure is less than +0.25 psi.

Liquid Nitrogen Quantity Indicators

Two liquid nitrogen quantity indicators are installed on the pilot's right instrument side panel.

The dual needle lower indicator displays the quantity of liquid nitrogen remaining in the system 1 and system 2 Dewar flasks. The indicator is marked in 5-liter increments from 0 to 110 liters. When the indicators and warning lights test button (IND & LT TEST) is depressed, the quantity indicator needles move toward zero. Power for the indicator is furnished by the essential dc and ac busses through four circuit breakers. Two, labeled N2 QUAN NO 1 and NO 2, are located on the pilot's left console, and two labeled N QTY NO 1 and NO 2 are located on the pilot's right console.

NOTE

The Dewar flasks may deplete at an unequal rate.

A second liquid nitrogen quantity gage is installed above the dual needle indicator. The gage is marked LIQ N and displays the quantity of liquid nitrogen remaining in the system 3 Dewar flask. The dial is marked in 10 liter increments from 0 to 50 liters. Fluctuations of the gage are normal. The instrument indication is not affected by operation of the indicators and warning lights test button. Power is furnished from the essential dc bus through a circuit breaker in the C-Bay.

Nitrogen Quantity Low Indicator Lights

Two nitrogen quantity low caution lights, one for each system, are located on the pilot's annunciator panel. The lights are labeled SYS 1 N QTY LOW and SYS 2 N QTY LOW. When illuminated, the respective nitrogen quantity is less than 3 liters. Operation of the lights can be checked by depressing the indicators and warning lights test button until the quantity gauge indicates below 3 liters. When the test button is released, the nitrogen quantity low lights will remain illuminated momentarily.

NOTE

- If necessary, loiter in accordance with emergency procedures to cool the fuel tanks if LN_2 has been depleted. Cooling is not required if speed has not exceeded Mach 2.6.
- There is no caution light to indicate depletion of the system 3 liquid nitrogen supply.

Air Refuel Switch

An air refuel switch, located at the top of the pilot's instrument panel, has three positions: AIR REFUEL (up), OFF (center), and MAN O'RIDE (down). When the switch is placed in the AIR REFUEL or MAN O'RIDE position, the refueling door opens, the receptacle lights illuminate, and the READY light in the air refuel reset switch illuminates. In AIR REFUEL, the boom latches are automatically armed. In MAN O'RIDE, opening and closing of the boom latches must be controlled manually by the air refuel disconnect trigger switch.

CAUTION

Before opening or closing the refueling door, ensure that the probe is clear.

Air Refuel Reset Switch and Indicator Light

A dual-indicating, self-illuminating push-button switch, labeled PUSH TO RESET, is located on the top of the pilot's instrument panel. The upper half of the push-button switch illuminates green and displays the word READY when the air refuel ready switch is in either AIR REFUEL or MAN O'RIDE and the refueling system signal amplifier is on. The READY light extinguishes when the boom is seated and latched. If the boom disconnects from the refueling receptacle, the lower half of the push-button switch illuminates amber and displays DISC. The push-button switch must then be depressed (or the air refuel switch recycled) to illuminate READY before the boom can be reengaged. The READY light illuminates and the DISC light does not illuminate when a disconnect occurs while refueling in MAN O'RIDE.

Air Refuel Disconnect Trigger Switch

The trigger switch on the forward side of the control stick grip can be used to disconnect the refueling boom.

When refueling with the air refuel switch in MAN O'RIDE, the trigger switch is used to open and close the boom latches. Depressing the trigger switch opens the boom latches and holding it depressed keeps the boom latches open. When the boom is seated, the READY light extinguishes and the switch can be released to close the boom latches, locking the boom in the receptacle.

ELECTRICAL SYSTEM

Electrical power is normally supplied by two ac generators, rated at 60 KVA each. The generators are mechanically driven by their respective engines through constant speed drive (CSD) units and operate in parallel. They provide 115/200 volt, 400-cycle, three-phase power to five ac buses and to two, two-hundred ampere transformer-rectifiers (T-Rs). Three 28-volt dc buses are normally energized by these T-Rs. Either generator is capable of supplying the normal ac and dc power requirements of the aircraft. Figure 1-44 shows the ac and dc power supplies, control circuits, and power distribution system.

The electrical system operates automatically and the protective features and the emergency dc system are automatically available after the generators have been set and the batteries switched on. An emergency ac generating system is provided which may be available after some types of electrical system failures. It must be selected manually, using the generator control switches.

Electrical system back-up controls provided in the forward cockpit for emergency conditions include the generator bus tie, instrument inverter, and emergency ac bus switches. Seven caution lights indicate: generator(s) out, generator bus tie open, transformer-rectifier(s) out, instrument inverter on, and emergency battery on.

BATTERIES

Two 25 ampere-hour batteries are provided for emergency service. If both generators are off or inoperative, or both transformer-rectifiers fail, each battery individually supplies one of the two essential dc buses. The battery relays will not engage unless sufficient charge remains in the No. 1 battery.

The No. 1 battery energizes the No. 1 essential dc bus, which supplies the SAS pilot valves. It also energizes the emergency ac bus through an instrument inverter, rated at 1 KVA. See Emergency DC Power Supply, Figure 1-46.

The No. 2 battery energizes the No.2 essential dc bus which supplies SAS control power, DAFICS computers, and all other essential dc system loads. The No. 2 battery is always connected to the BAILOUT and PILOT EJECTED warning lights, regardless of the battery switch position.

The maximum duration of the dual-battery power system is approximately 40 minutes if unnecessary equipment is turned off. Figure 1-43 lists power requirements of equipment energized from the essential dc buses.

EMERGENCY AC POWER SYSTEM

Each main generator has an emergency operating mode which may be available if neither generator will function normally. To be usable, the generator(s) selected must still be rotating with intact windings. Then, setting the system to the emergency mode may generate usable, but unregulated, ac power. There is no control of voltage or frequency in the emergency mode since generator speed is not governed and the No. 2 battery provides direct excitation of the generator field. If either generator is operating in EMER, the bus tie will open. Emergency ac power is applied directly to the ac hot bus and to the generator bus associated with that engine. See Figure 1-45.

CAUTION

In-flight, do not operate either generator in EMER unless both generators have failed.

Primarily, the emergency ac system will supply power for the fuel boost pumps and will power the ac hot bus to provide fuel transfer, cross-feed, and pitch axis trim capability. It also supplies power to the corresponding T-R unit and, if sufficient voltage is generated, the dc buses will be powered by the T-R. With neither generator operating in NORM, the emergency ac bus is energized by the No. 1 essential dc bus through the instrument inverter and the essential ac bus is dead; the monitored dc bus is also dead.

If sufficient ac generator voltage is not available to the T-Rs, the essential dc buses and the instrument inverter will be powered by the batteries.

WARNING

During emergency ac operation, the normal automatic fuel sequencing system is disabled and the pilot must manually select tanks. The automatic aft transfer and ullage systems are inoperative. Normal generator fault protection is not provided.

EXTERNAL POWER

An external power receptacle in the nosewheel well accepts a six-wire type cable connection from an MD-3 or MD-4 (or equivalent) ground support unit. The supply must provide 115-120/200-208 volt, 400-cycle, three-phase ac power with A-B-C phase rotation, and 28 volt dc power. (The 28 volt dc supply only energizes the aircraft external power relay. Aircraft dc power is obtained from the external ac supply through the ship's T-R units.)

Transfer to Internal Power

The aircraft generator switches are ineffective when the engines are stopped. Normally, the generator switches are set to NORM after the engines start. The generators remain disengaged until the right engine CSD reaches a speed which allows its generator to synchronize with the frequency and phasing of the external supply. However, if the generator switches are set to NORM before the engines are started, the resulting paralleling transient may cause the ANS to trip off. The external power contactor is automatically opened when the right engine reaches a parallel condition with the external supply and the generator line contactor closes. The external power connector can be removed after this occurs, but its removal is normally delayed until both generators are on-line and the system operating normally - a condition indicated by all seven electrical system caution lights being off.

CIRCUIT BREAKER PANELS - Fwd Cockpit

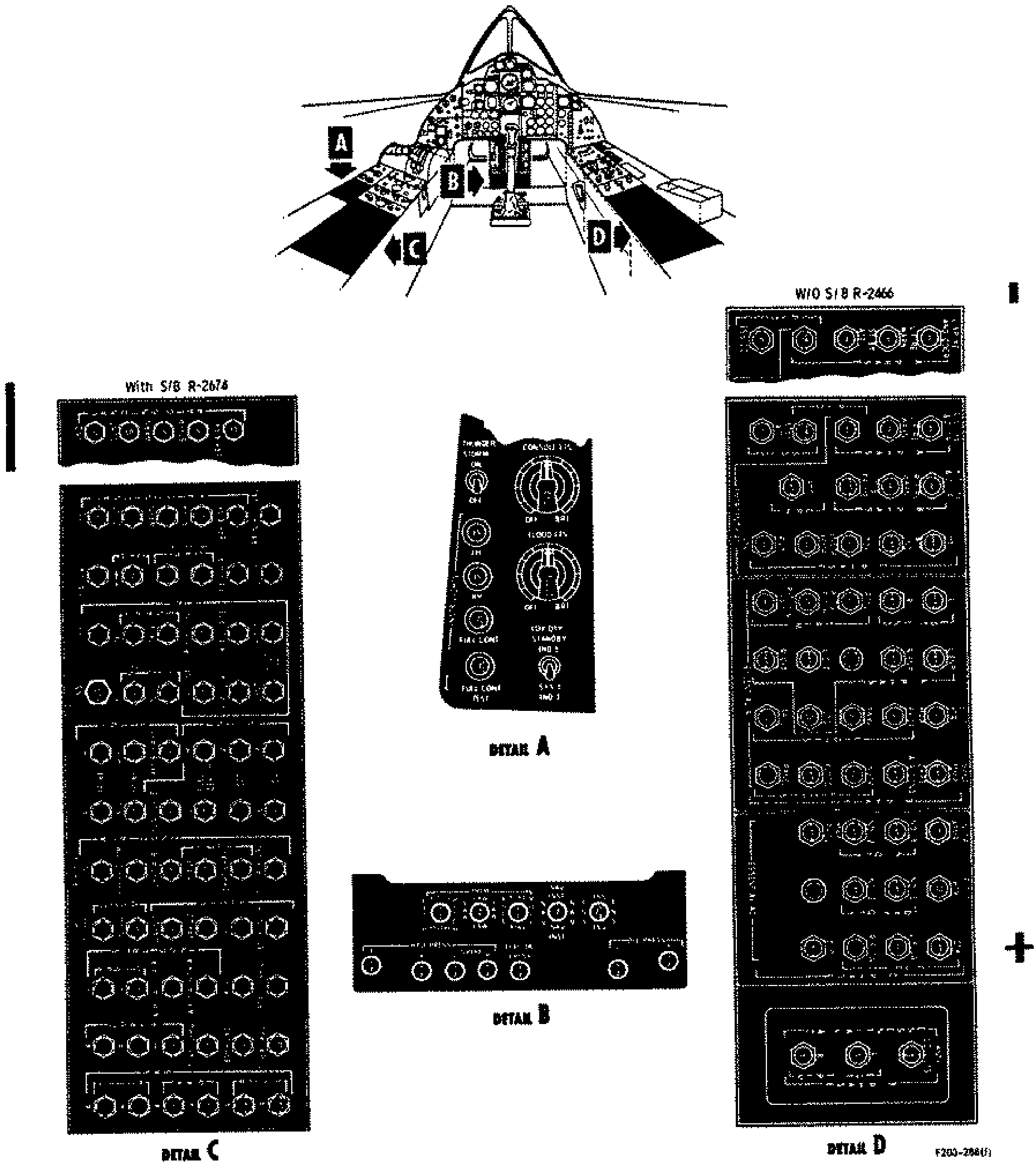


Figure 1-41

CIRCUIT BREAKER PANEL
Aft Cockpit

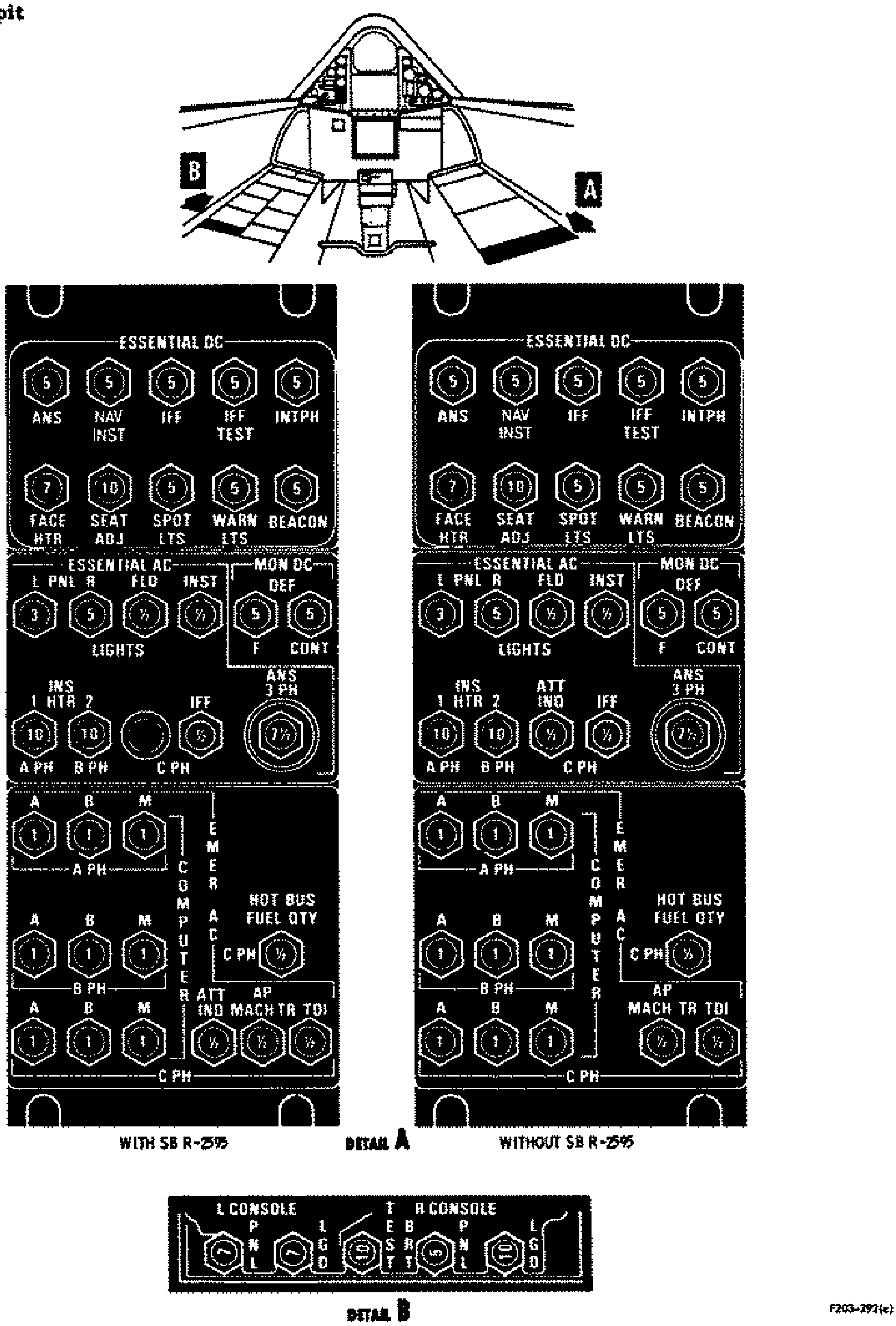


Figure 1-42

The left engine generator automatically synchronizes its output with the right generator, then its contactor closes and the generators operate in parallel. If the right generator is not on line, the left generator can be set in NORM, but it may not parallel the external power supply and it will not come on line until external power is disconnected or shut off. A momentary power surge can result if the phasing is not synchoronized. The L GEN OUT caution light will remain on until power is transferred.

Engine starting without external electrical power is possible, but not recommended. Instrument indications, including EGT, require essential ac bus power and would not be available until the generators are turned on.

CIRCUIT BREAKERS

Circuit breaker panels, located in the forward and aft cockpits, contain pullout/push-to-reset breakers for certain ac and dc circuits. See Figures 1-41 and 1-42. Services interrupted by opening these circuit breakers are listed by Figure 1-47. Other circuit breaker panels which are not accessible during flight are located in the C and E bays.

Differential Protection Relays

A differential protection relay (DPR) is a part of each generator system. The DPRs provide automatic protection by disconnecting the associated generator for a significant fault within the generator, in the generator feeder lines (to the buses), and/or in the generator line contactors.

ELECTRICAL SYSTEM CONTROLS AND INDICATOR LIGHTS

Generator Switches

A 3-position lift-loc control switch for each generator is located on the pilot's right instrument side panel. The switch positions are NORM (down), OFF (center), and EMER (up). Placing either switch in NORM will return the respective generator to normal operation if it has been removed from the bus for any reason other than generator or system failure. The NORM position is locked to prevent accidental actuation to OFF or EMER. In OFF, the corresponding generator is removed from service.

NOTE

> If its protective circuits trip a generator and it does not reset automatically, the generator switch must be moved to OFF and then back to NORM to attempt a manual reset.

In EMER (used when both generators have failed), a 28 volt dc excitation current from the essential dc bus is applied to the respective generator exciter fields through a single 5 amp EMER EXC circuit breaker. If the generator is rotating and the windings are operative, an unregulated voltage and frequency ac current is developed which will power the hot bus and corresponding generator bus. The essential ac bus will be dead. Extinction of the corresponding L and/or R GEN OUT light(s), after selecting EMER, indicates successful emergency system operation. The corresponding L and/or R XFMR RECT OUT light may extinguish; if this occurs, the EMER BAT ON light will extinguish if either the battery voltage is exceeded by the transformer-rectifier output, or the battery output is less than 10 amperes.

Bus Tie Switch

A push-button bus tie switch is located on the pilot's right instrument side panel. If the GEN BUS TIE OPEN light illuminates simultaneously with indication of generator failure, depressing the switch should retie the L and R generator buses and extinguish the light, if the failure was in the generator or its control system.

ELECTRICAL LOAD ANALYSIS WHEN NO AC GENERATOR AVAILABLE

DC ESSENTIAL BUS ITEM	POSSIBLE LOAD IN AMPERES	CAN BE USED OR TURNED OFF BY SWITCH	CAN BE USED OR TURNED OFF BY CB	CANNOT BE USED-TURNED OFF BY SWITCH	CANNOT BE USED-TURNED OFF BY CB
AIR REFUEL	1.20	X			
AIR SHUTOFF CONTROL	2.80	X			
APW	2.50	X			
AUTOPILOT/MACH TRIM A	0.30		X		
AUTOPILOT/MACH TRIM B	0.30		X		
BRAKE AND SKID	1.60	X			
COCKPIT AIR	0.10		X		
COCKPIT AND BAY TEMP	0.30	X			
COMNAV-50	2.75			X	
COMPUTER A	0.20		X		
COMPUTER B	0.20		X		
COMPUTER M	0.20		X		
DRAG CHUTE	3.00	X			
EGT TRIM, L AND R	1.00		X		
EMER FUEL S/O, L AND R	0.40			X	
EMER GEN EXCITER	1.00	X			
FACE HEAT	0.88	X			
FUEL CONTROL	0.40				X
FUEL DERICH, L AND R	2.40		X		
FUEL DUMP	2.30				
FUEL FWD TRANSFER	1.50	X	X		
FUEL TANK 5 TRANSFER	2.40				X
GROUND AIR VALVES	0.70				X
IFF	0.18			X	
IFR INTERCOMM	0.07		X		
IGNITER PURGE, L AND R	2.00	X			
ILS	1.83	X			
INLET AFT BYPASS, L AND R	4.00		X		
INLET GUIDE VANES	0.08				
INS	2.00	X	X		
INSTRUMENT INVERTER	28.50	X			
INTERPHONE	0.32		X		
LANDING GEAR CONTROL			X		
LANDING GEAR INDICATORS			X		
LANDING GEAR WARNING			X		
LN_2 QUANTITY, NO. 1 AND NO. 2	2.00				X
MANIFOLD TEMP	1.30		X		
NAVIGATION INSTRUMENTS	0.86		X		
NAVIGATION SYSTEM					X
NOSE STEERING	2.15				X
PILOT VALVES	0.00				

Figure 1-43 (Sheet 1 of 2)

ELECTRICAL LOAD ANALYSIS WHEN NO AC GENERATOR AVAILABLE (CONT.)

DC ESSENTIAL BUS ITEM	POSSIBLE LOAD IN AMPERES	CAN BE USED OR TURNED OFF BY SWITCH	CAN BE USED OR TURNED OFF BY CB	CANNOT BE USED-TURNED OFF BY SWITCH	CANNOT BE USED-TURNED OFF BY CB
PITOT HEAT	0.00			X	
PRESSURE DUMP	0.40	X			
RAIN REMOVAL (W/O S/B 2674)	1.50	X			
RUDDER LIMITER, L AND R	3.30	HANDLE			
SAS A	8.80	X	X		
SAS B	8.80	X	X		
SEAT ADJUST	5.00	X			
SPIKE AND DOOR POS. IND.	0.80		X		
SPIKE CONTROL L	1.00		X		
SPIKE CONTROL R	1.00		X		
SPIKE SOL L	3.50		X		
SPIKE SOL R	3.50		X		
SPOT LIGHTS	0.36	X			
STALL WARNING	0.15		X		
STANDBY ATT. IND., 3-INCH	2.00		X		
TACAN (ARN-118)	2.00			X	
TANK 4 ULLAGE	0.15				X
T-STORM LIGHTS	1.43	X			
TURN AND SLIP INDICATOR	0.25		X		
UHF ANTENNA ACTUATOR	3.50	X			
VHF (ARC-186)	2.96	X			
WARNING LIGHTS	1.41		X		
WINDSHIELD DEICE	0.70	X			

EMERGENCY AC BUS ITEM	POSSIBLE LOAD IN VA	CAN BE USED OR TURNED OFF BY SWITCH	CAN BE USED OR TURNED OFF BY CB	CANNOT BE USED-TURNED OFF BY SWITCH	CANNOT BE USED-TURNED OFF BY CB
ANGLE OF ATTACK IND.	5.00				
ATTITUDE DIRECTOR IND.	20.00		X		
ATTITUDE IND-RSO (S/B 2595)	20.00		X		
AUTOPILOT/MACH TRIM	30.00		X		
COMPUTER A	124.00		X		
COMPUTER B	124.00		X		
COMPUTER M	124.00		X		
EMER INSTR TRANSFORMER	18.20		X		
FIRE WARNING, L AND R	2.20		X		
HSI	50.00		X		
INLET, L	20.00		X		
INLET, R	20.00		X		
INS	220.00	X			
PITCH INDICATOR	0.21		X		
STANDBY ATT. IND., 2-INCH	13.20		X		
TDI	39.00		X		

Figure 1-43 (Sheet 2 of 2)

CIRCUIT BREAKER FUNCTION TABLE

CIRCUIT BREAKER	EFFECT OF POWER INTERRUPTION
	ESSENTIAL AC BUS (Forward Cockpit - Cont.)
ANTI COLL LTS (3Ø)	Disabled: Anti-collision/fuselage lights.
EGT TRIM (3Ø)	
L	Disabled: Left engine EGT trim motor.
R	Disabled: Right engine EGT trim motor.
	26 VOLT AC ESSENTIAL BUS (Forward Cockpit/Center Pedestal)*
HYD PRESS	
A	Disabled: A-System hydraulic pressure indication.
B	Disabled: B-System hydraulic pressure indication.
SPIKE L	Disabled: L-System hydraulic pressure indication.
SPIKE R	Disabled: R-System hydraulic pressure indication.
FUEL TK PRESS	Disabled: Fuel system tank pressure indication.
OIL PRESS	
L	Disabled: Left engine oil pressure indication.
R	Disabled: Right engine oil pressure indication.
TRIM	
PITCH	See 26 V Emergency AC BUS
YAW	Disabled: Yaw trim condition indication.
ROLL	Disabled: Roll trim condition indication.
NAV INST	Disabled: ANS attitude signal to ADI and AI. ANS heading signal to HSI and BDHI.
INS	See 26 V Emergency AC BUS
	*NOTE: Essential AC Bus INSTR XFMR circuit breaker must be in for power to 26 V Essential AC BUS.
	ESSENTIAL DC BUS (Aft Cockpit)
ANS	Disabled: ANS
BEACON	Disabled: G-band beacon when S/B 1763K installed:
FACE HTR	Disabled: Helmet face heat, aft cockpit.
IFF	Disabled: All IFF modes.
IFF TEST	Disabled: IFF transponder self test capability, Modes 1, 2, 3A, and C.
INTPH	Disabled: All audio to aft cockpit headset (except trainer EMERG ICS).
NAV INST	Disabled: Navigation display relays. Attitude reference reverts to: Pilot-INS RSO-ANS. Autopilot will not engage with Pilot's ATT REF switch in ANS, HSI displays mag hdg and TACAN DME. BDHI displays true hdg and TACAN DME.
SEAT ADJ	Disabled: Aft cockpit seat adjustment. Seat remains as set.
SPOT LTS	Disabled: Aft cockpit spot lights and flex point lights.
WARN LTS	Disabled: All aft cockpit caution lights.
	MONITORED DC BUS (Aft Cockpit)
DEF F	Disabled: DEF H
DEF CONT	Disabled: ALL DEF control systems.

Figure 1-47 (Sheet 6 of 7)

CIRCUIT BREAKER FUNCTION TABLE

CIRCUIT BREAKER	EFFECT OF POWER INTERRUPTION
	EMERGENCY AC BUS (Aft Cockpit)
COMPUTER — A, B, M (AØ); A, B, M (BØ); A, B, M (CØ)	NOTE: • With one circuit breaker opened to each computer, no capability is lost. • With two circuit breakers opened to a computer, that computer is likely to shut down. The corresponding pitch, yaw and/or roll sensors and servos are disabled if the following pairs of circuit breakers are opened: A Computer: A & B or A & C phase; B Computer: A & B or B & C phase; M Computer: A & C or B & C phase. • With all three A computer circuit breakers open, left auto inlet is disabled (inlet goes to restart). Manual control of left spike and door required. • With all three B computer circuit breakers open, right auto inlet is disabled (inlet goes to restart). Manual control of right spike and door required. • Manual inlet control is disabled if the M computer A and C phase or B and C phase circuit breakers are opened.
ATT IND (W/SB R-2595) (CØ)	Disabled: Aft cockpit attitude indicator.
AP MACH TR (CØ)	Disabled: Autopilot and Mach Trim systems, Analytical Redundancy, AOA, TDI OFF flags in view (indications remain valid), Bank Command steering bar remains centered.
TDI (CØ)	Disabled: Aft cockpit TDI. TAS to Pilot's & RSO's Map Projector - Automatic Map Rate.
	AC HOT BUS (Aft Cockpit)
FUEL QTY	Disabled: Fuel quantity and c.g. indications, aft cockpit.
	ESSENTIAL AC BUS (Aft Cockpit)
INS HTR 1 (AØ)	Disabled: No. 1 INS heater. INS performance will degrade.
LIGHTS (AØ)	
PNL L	Disabled: All functions of L CONSOLE PNL and L CONSOLE LGD circuit breakers on left console c/b panel.
PNL R	Disabled: All functions of R CONSOLE PNL, R CONSOLE LGD AND TEST & BRT circuit breakers on left console c/b panel.
FLD	Disabled: Aft cockpit flood lights.
INST	Disabled: Aft cockpit instrument panel lights.
INST LIGHTS (AØ)	Left Console, aft cockpit
L CONSOLE	
PNL	Disabled: Left console panel lights.
LGD	Disabled: Left console legend lights.
TEST & BRT	Disabled: Legend test function and, if right console rheostat switch is off, legend and panel lights associated with the right console rheostat.
R CONSOLE	
PNL	Disabled: Right console panel lights.
LGD	Disabled: Right console legend lights.
INS HTR 2 (BØ)	Disabled: No. 2 INS heater. INS performance will degrade.
ATT IND (W/O SB R-2595) (CØ)	Disabled: Aft cockpit attitude indicator.
IFF (CØ)	Disabled: IFF Mode 4 capability.
ANS 3 PH (3Ø)	Disabled: ANS

Figure 1-47 (Sheet 7 of 7)

batteries are furnishing at least 10 amperes to the essential dc buses. The services in Figure 1-43 can be powered by the batteries and instrument inverter.

If dc power from the T-Rs is interrupted, a time delay of ten seconds occurs before the EMER BAT ON light illuminates. The EMER BAT ON light will not illuminate if T-R dc power is restored within ten seconds.

NOTE

> Occasional flickering of the EMER BAT ON caution light can be disregarded if not accompanied by other indications of abnormal electrical system operation.

Generator Out Caution Lights

The L or R GEN OUT caution light on the pilot's annunciator panel illuminates when the corresponding generator is disconnected automatically or the respective generator control switch is OFF.

HYDRAULIC SYSTEMS

Four separate hydraulic systems are provided, each with its own pressurized reservoir and engine-driven pump. (See Figure 1-48). Normally, all the systems are independent. The pumps for the A and L system are driven from the left engine accessory drive system (ADS) and the B and R pumps from the right engine ADS. The A, B, and L system reservoirs are serviced to 2.8 gallons of hydraulic fluid. The R system reservoir is serviced to 4.5 gallons of hydraulic fluid. Hydraulic fluid is cooled by fuel-oil heat exchangers, using the aircraft fuel supply for cooling.

The A and B hydraulic systems power the flight controls. An accumulator is provided in each of these two systems. The A hydraulic system also powers the APW system stick pusher.

The L hydraulic system powers the left engine air inlet system and normally powers the landing gear (including uplocks and door cylinders), brakes, air refueling door and receptacle, and nosewheel steering. The R hydraulic system powers the right engine air inlet system. In addition, if L system pressure is less than 2200 psi, the R system will automatically power the landing gear retraction cycle, and the pilot may select the R system to power nosewheel steering and the air refueling system (by setting the brake switch to ALT STEER & BRAKE). Regardless of L system pressure, the pilot may always select the R system to power the brakes. Antiskid braking is available with either L or R hydraulic system.

Hydraulic System Pressure Gages

Two dual-indicating hydraulic gages are installed on the pilot's instrument panel. The bottom (SURF CONT) gage indicates A and B system hydraulic pressures, and the top (SPIKE) gage indicates L and R system hydraulic pressures. The gages are calibrated in 100-psi increments from 0 to 4000 psi. Pressure indication is sent from remote transmitters in the individual systems.

Power for the gages is furnished by the essential ac bus 26 volt instrument transformer through the A, B, L SPIKE and R SPIKE circuit breakers on the pilot's annunciator panel.

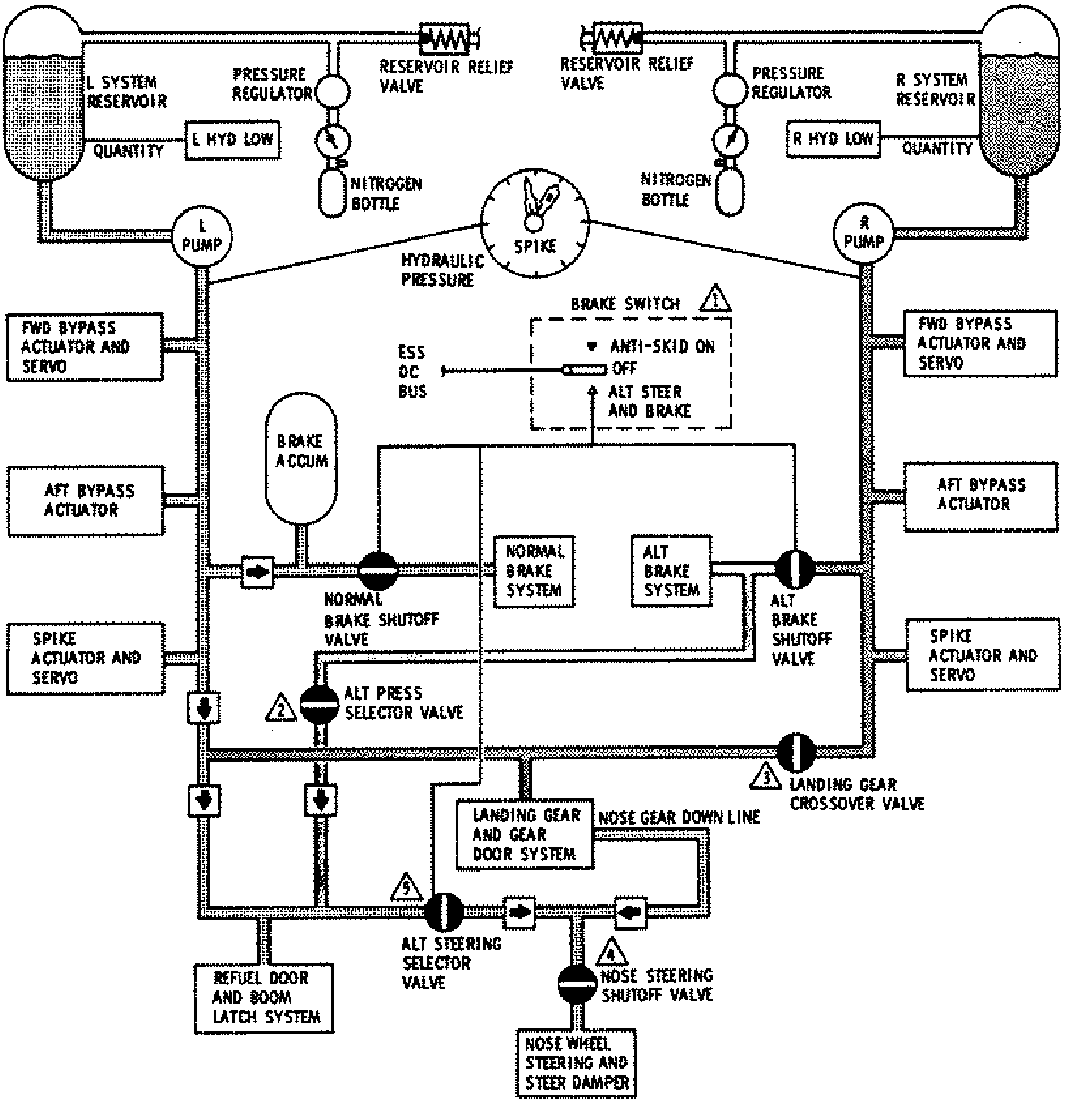
L and R Hydraulic Quantity Low Warning Lights

The L or R HYD warning light on the pilot's annunciator panel illuminates when hydraulic quantity in the respective reservoir decreases below 1.2 gallons.

A and B Hydraulic Quantity/Pressure Low Warning Lights

The A or B HYD warning light on the pilot's annunciator panel illuminates when hydraulic quantity in the respective reservoir decreases below approximately 1.2 gallons and/or the respective hydraulic pressure decreases below 2200 (±150) psi.

L AND R HYDRAULIC SYSTEMS



NOTE

1. OFF: "L" system powers brakes, no anti-skid protection.

   ANTI-SKID ON: Provides anti-skid protection on "Normal" brake system.

   ALT STEER AND BRAKE: Closes "Normal" and opens "Alt" brake shutoff valves, arms alternate system selector valves, and energizes "Alt. Anti-skid" system

2. With brake switch in "Alt Steer and Brake" position, valve is opened if "L" system pressure decreases below 2200 psi.

3. Crossover valve opens automatically if "L" system pressure decreases below 2200 psi, but only for gear retraction.

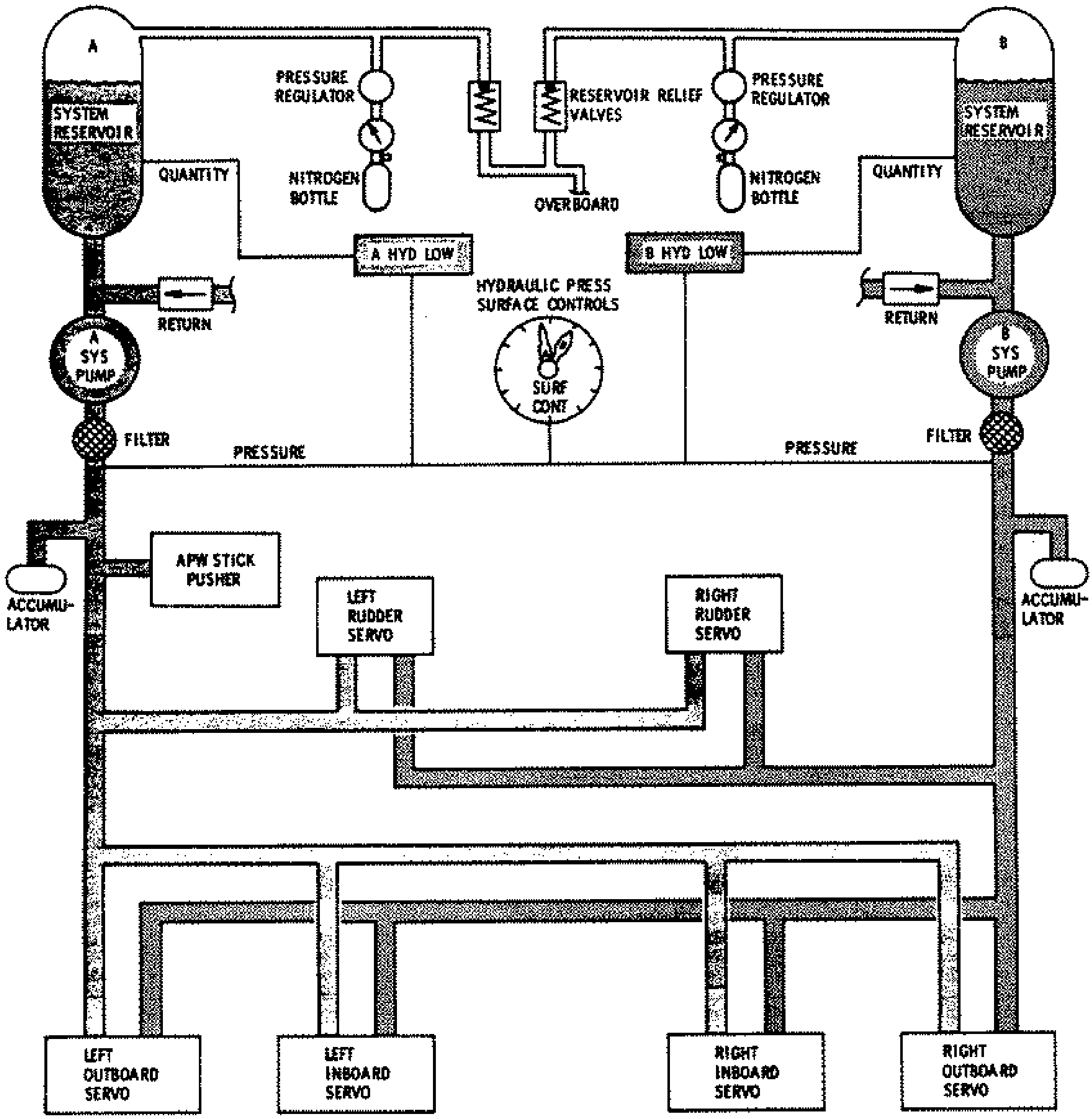
4. Steering controlled by CSC/NWS switch on control stick to provide nose steering on normal or alternate system pressure.

5. Valve opens when alternate steering and braking selected regardless of pressure in the L system

F203-16(1)

Figure 1-48 (Sheet 1 of 2)

A AND B HYDRAULIC SYSTEMS



NOTE

Each hydraulic system provides actuation power to half the actuating cylinders at each servo assembly.

HYD LOW lights are illuminated by decreasing quantity with 1.2 gallons remaining in the respective reservoir, and/or by decreasing pressure at approximately 2200 psi.

F203-15(1)

Figure 1-48 (Sheet 2 of 2)

NOTE

Rapid control surface deflection while near idle rpm may result in temporary illumination of the A and/or B HYD warning light(s). The light(s) should extinguish when flow demands diminish and normal pressure is restored.

LANDING GEAR SYSTEM

The tricycle landing gear and the main wheel well inboard doors are electrically controlled and hydraulically actuated. The main gear outboard doors and the nose gear doors are linked directly to the respective gear struts. Each three-wheeled main gear retracts inboard into the fuselage and the dual-wheel nose gear retracts forward into the fuselage. The main gear is locked up by the inboard doors and the nose gear by an uplock which engages the strut. There is no hydraulic pressure on the gear when it is up and locked. Downlocks inside the actuating cylinders hold the gear in place in the extended position. Normal gear operation is by L hydraulic pressure. L system hydraulic pressure is also on the gear when in the extended position. Normal gear retraction and extension time is 12 to 16 seconds. Should L hydraulic pressure drop to 2200 psi during retraction, the power source automatically changes to the R hydraulic system. If the L system fails, R system pressure cannot be used to extend the gear and the manual gear release must be used.

A landing gear strut damper system controls gear "walking," a fore-and-aft oscillation of the main landing gear strut associated with brake application. The system is sensitive to less than 1-g change in fore-and-aft acceleration. The damping is controlled through a g-monitoring valve which automatically increases or decreases the brake pressure as required. Hydraulic pressure for the damper system is provided by the L system. The damper does not function with the brake switch in ALT STEER & BRAKE. A strut damper shutoff valve removes L system pressure from the damper valve when the landing gear is retracted.

Landing and taxi lights are on the nose gear strut. Refer to Lighting Equipment, this section.

Landing Gear Handle

A wheel-shaped landing gear handle, on the pilot's left instrument side panel, has two positions: UP and DOWN. An up-lock latch prevents the gear handle from being inadvertently placed in DOWN. An up-lock release lever which extends from the top of the gear handle, must be pushed forward to release the up-lock latch. A safety-lock solenoid prevents the gear handle from being inadvertently placed UP while the aircraft is on the ground. A manual solenoid release button is located just above the gear handle. Depressing the release button overrides the safety-lock soleoid and allows the gear handle to be moved to UP. In UP, the gear will retract if hydraulic pressure is available except that the landing gear control circuit is interlocked with the gear scissor switches to prevent retraction of the gear on the ground.

A red warning light is located in the transparent gear handle. Power for the circuit is furnished by the essential dc bus through the landing gear CONT circuit breaker on the pilot's left console.

Manual Landing Gear Release Handle

A manual GEAR RELEASE T-handle is located on the pilot's annunciator panel. If the L hydraulic system fails, the landing gear handle should be placed DOWN and the CONT circuit breaker should be pulled before pulling the GEAR RELEASE handle. If the landing gear handle cannot be placed DOWN and the landing gear CONT circuit breaker is not pulled, the landing gear will retract if there is pressure in the R hydraulic system. When the GEAR RELEASE handle is pulled, the gear extends by gravity within 90 seconds. Up to 65 pounds of force and approximately 9-1/3 inches extension of the handle is required to release the gear. The uplocks are released in the following sequence as the cable extends: nosewheel, right main gear door aft latch, right main door forward latch, left main gear door aft latch, and left main door forward latch. Gear retraction can be accomplished after emergency extension if L or R hydraulic system pressure is available.

CAUTION

The landing gear must not be retracted while the manual release handle is pulled, as damage to the system can result. Stow the handle before retracting the gear.

Landing Gear Position Lights

Three green lights on the pilot's left instrument side panel illuminate when each respective landing gear is down-and-locked. The location of each light corresponds to the gear it monitors. The lights also illuminate when the IND & LT TEST button is depressed. Power is furnished by the essential dc bus through the landing gear IND circuit breaker on the pilot's left console.

Landing Gear Warning Light and Audible Warning

The red warning light in the landing gear lever handle illuminates when:

1. Gear is cycling.
2. Gear system is not locked in the position (UP or DOWN), programmed by the landing gear handle.
3. Gear is UP and throttles are within approximately 1 inch of the IDLE stop, while below 10,000 ± 500 feet.

A pulsed-tone warning signal is produced in the pilot's and RSO's earphones when the throttles are retarded below minimum subsonic cruise setting, the landing gear is not in the down and locked position and aircraft altitude is below 10,000 (± 500) feet. The pulsed tone circuit is isolated from the gear handle light circuit so that if an emergency gear extension is necessary with the gear handle up, the tone will not occur if the gear is locked down and the throttles are retarded. The tone sounds if the IND & LT TEST button is depressed while below 10,000 ±500 feet. Power for the light and audible warning is furnished by the essential dc bus through the landing gear WARN circuit breaker on the pilot's left console.

Landing Gear Warning Cutout Button

The aural gear warning may be silenced by depressing the GEAR SIG REL button on the pilot's left instrument side panel. The circuit is reactivated when the throttles are advanced above the minimum cruise setting. Power is furnished by the essential dc bus through the landing gear WARN circuit breaker on the pilot's left console.

Landing Gear Ground Safety Pins

Removable ground safety pins are installed in the landing gear assemblies to prevent inadvertent gear retraction. Warning streamers direct attention to their removal before flight. Extra pins are provided in a container on the pilot's aft bulkhead left of the ejection seat.

NOSEWHEEL STEERING SYSTEM

The nosewheel steering system provides power steering while on the ground. It can be engaged when aircraft weight is on any gear by aligning rudder pedal position with nosewheel angle and depressing the CSC/NWS button on the control stick. A holding relay circuit keeps steering engaged when the button is released. The button must be depressed and released again to disengage steering.

A nosewheel steering engaged light is provided on the top left of the pilot's instrument panel. Illumination of the green STEER ON legend indicates nosewheel steering engagement. The light extinguishes if steering is disengaged. Steering disengages automatically when weight is not on any gear. With weight on a gear, steering disengages with loss of hydraulic pressure or when manually disengaged by the pilot.

The steering angle obtained is directly proportional to rudder pedal deflection. The nosewheel is steerable 45° either side of center. Minimum steering radius is approximately fifty-five feet. See Figure 2-3 for clearance requirements while turning.

WARNING

The landing gear side load strength is critical. Side loads during takeoff, landing, and ground operation must be kept to a minimum.

CAUTION

Do not engage nosewheel steering before nosewheel touchdown; otherwise, excessive strut and fuselage forebody loads could result from steering angles developed before nosewheel contact.

A hydraulically actuated clutch is located within the steering damper unit. The clutch engages and disengages nosewheel steering when the CSC-NWS switch is actuated.

Rudder control in-flight with the gear down is severely restricted if the clutch jams and nosewheel steering does not disengage.

WARNING

Retract the landing gear immediately to relieve restriction of rudder movement if jamming of the nosewheel steering clutch is suspected while in-flight.

NOTE

Approximately 6° of rudder would be available, through cable stretch, by applying 180 pounds of force at the rudder pedals. Rudder restriction would not be noted with the gear up.

A mechanically operated centering cam automatically centers the nosewheel when the gear retracts.

Rudder pedal movement controls a hydraulically operated nosewheel steering and shimmy damper unit by means of a cable system when steering is engaged. While on the ground with the brake switch in the ANTI-SKID ON or OFF position, hydraulic power for steering is obtained from the L system through the nose landing gear down line and the nose steering shutoff valve. If L system pressure decreases below 2200 psi, selection of the brake switch ALT STEER & BRAKE position makes R system hydraulic power available for steering. Then hydraulic power is supplied to the nose steering shutoff valve through the alternate brake shutoff valve, alternate pressure selector valve, and the alternate steering selector valve. See sheet 1 of Figure 1-48.

NOTE

With ALT STEER & BRAKE selected, the R hydraulic system cannot supply hydraulic pressure for nosewheel steering until L system pressure decreases below 2200 psi. The L system continues to supply hydraulic power while above 2200 psi.

NOTE

After the landing gear CONT circuit breaker has been opened, ALT STEER & BRAKE must be selected to open the alternate steering selector valve and obtain hydraulic power for nosewheel steering.

Electrical power for control of the nosewheel steering system is obtained from the essential dc bus through the STEER, BRK & SKID, and CONT circuit breakers. See sheet 1 of Figure 1-47 for functions lost when any of these circuit breakers is open.

WHEEL BRAKE SYSTEM

The aircraft is equipped with hydraulic operated power brakes, controlled through toe-action of the rudder pedals, and provided with artificial feel. Two interrelated brake systems are provided: a normal system using

L system hydraulic power, and an alternate system using R system hydraulic power. Selection of normal or alternate brake system is controlled by the brake switch on the pilot's left instrument side panel. Both systems use a hydraulically operated relay system to control metering of hydraulic pressure to the multiple-disc brake assemblies on each main gear wheel. Braking follows toe pressure command within one-half second.

The normal brake system has a small accumulator and a strut damper system. The accumulator is charged by L system pressure and may provide up to three brake applications if L system fails. The brake accumulator is not required to hold a charge. The probability that the accumulator will provide braking decreases as time from the loss of L hydraulic system pressure increases. Accumulator braking is not available with the brake switch in ALT STEER & BRAKE. The strut damper system dampens fore and aft oscillations of the main gear struts (associated with brake applications). Strut oscillation ("strut walk") occurs at approximately 10 cps. Strut damping is operational only on the normal brake system.

Antiskid protection is available to both the normal and alternate brake systems. The antiskid system senses wheel skid as a function of wheel rpm. Wheel rpm decreasing too rapidly causes the antiskid system to relieve brake pressure to the affected main gear. Pressure is relieved until rpm increases sufficiently to permit further brake application without skidding. If wheel rpm does not increase within 2.7 seconds after antiskid relieves brake pressure: the antiskid fail-safe circuit deactivates antiskid and illuminates the ANTI-SKID OUT annunciator caution light, and braking without antiskid protection becomes available.

The antiskid system is operational above 12 miles per hour. The brake system permits near full system pressure at the brake assemblies under extreme braking. With very heavy braking, locked wheels may occur at speeds of less than 25 knots. Momentary lockup and unlocking does not affect overall braking performance. A touchdown safety feature in the antiskid system prevents landing with the brakes applied when the brake switch is in ANTI SKID ON or ALT STEER & BRAKE.

A DRY-WET switch (on left instrument side panel) permits selection of antiskid sensitivity. The WET position increases the sensitivity of the antiskid system to wheel spindown and improves antiskid operation on wet or icy runways. The sensitivity of the antiskid system to wheel spindown in the WET mode is such that the deceleration from normal drag chute action above 90 knots will relieve brake pressure to both main gears.

BRAKE CONTROL SWITCHES AND INDICATORS

Brake Switch

The three-position brake switch is on the pilot's left instrument side panel. In OFF (center), L hydraulic pressure is available for braking without antiskid protection. In ANTI SKID ON (up), L hydraulic pressure is available for braking with antiskid protection unless (after S/B R-2695) the trigger switch is depressed. In ALT STEER & BRAKE (down), R-hydraulic pressure is available for braking immediately with antiskid unless (after S/B R-2595) the trigger switch is depressed, and nosewheel steering from R system is available when L system pressure drops below 2200 psi.

CAUTION

If L hydraulic pressure is not available, R hydraulic system will not be available for braking or steering unless the brake switch is in ALT STEER & BRAKE.

Power is supplied to the brake switch from the essential dc bus through the BRK & SKID circuit breaker on the pilot's left console.

Antiskid Disconnect Trigger Switch

With S/B R-2695, antiskid system operation is interrupted while the trigger switch is held depressed. The hydraulic power source for brakes remains as selected by the brake switch.

WET-DRY Switch

The two-position wet-dry switch is on the pilot's left instrument side panel. In DRY (up), antiskid braking is compatible with wheel spindown characterisitics on a dry runway. The WET (down) position increases brake antiskid sensitivity to optimize braking on a wet or icy runway and reduces the probability of a blown tire and subsequent loss of antiskid braking. Electrical power is from the essential dc bus through the BRK & SKID circuit breaker.

ANTI-SKID OUT Light

The ANTI-SKID OUT caution light on the pilot's annunciator panel illuminates when the brake switch is OFF if the landing gear is down and there is weight on a gear, or if the antiskid system fail-safe circuit detects a fault.

After S/B R-2695, the ANTI-SKID OUT light also illuminates while the trigger switch is depressed if the landing gear is down and there is weight on a gear.

DRAG CHUTE SYSTEM

The drag chute system reduces landing roll and aborted takeoff rollout distance. The drag chute is stowed in an aft fuselage compartment above fuel tank 4. The drag chute attachment rides free in the compartment until locked to the aircraft during the initial stage of deployment. The drag chute and extraction system are packed in a deployment bag which contains a 42-inch vane-type pilot chute, a 10-foot extraction chute which produces aerodynamic lift, and a 40-foot ribbon-type drag chute. Normal chute deployment and jettisoning is accomplished electrically. Emergency deployment is accomplished mechanically. The drag chute handle operates both modes.

Drag Chute Handle

The T-shaped DRAG CHUTE handle is located on the upper left of the instrument panel. In the stowed (or jettison) position, the handle is horizontal, fully forward, and a red band on the shaft is not visible. Normal deployment is accomplished by pulling straight aft on the handle to the limit of its travel (approximately 1 inch). After chute deployment, jettisoning is accomplished by pushing the handle full forward. These handle positions operate switches which control the chute deploy and jettison actuator motors. Power for normal drag chute operation is provided by the essential dc bus through two DRAG CHUTE circuit breakers on the pilot's left console.

If normal deployment fails, emergency deployment is possible by rotating the handle 90 degrees counterclockwise from the normal deploy position and pulling to its aft travel limit (approximately 8 inches). The maximum pull force required is 60 pounds. When the handle is released, it is returned to the receptacle by the tension of a slack takeup spring. The drag chute cannot be jettisoned after emergency deployment because the actuator motor switches are disconnected during the manual deploy sequence.

CAUTION

If the DRAG CHUTE handle is pulled to the emergency deploy position and released immediately, damage could result to cockpit items as the handle snaps back to the receptacle.

NOTE

To avoid inadvertent emergency deployment, do not rotate the DRAG CHUTE handle during normal deployment.

Drag Chute Unsafe Light

The DRAG CHUTE UNSAFE caution light on the pilot's annunciator panel illuminates when: (1) the drag chute mechanism has been actuated to some degree either mechanically or electrically and is in an unsafe condition, or (2) power to both linear actuator dc motors is lost.

PRIMARY FLIGHT CONTROLS

The full-power irreversible flight control system consists of cockpit controls (stick and rudder pedals), four elevons, and two full-moving rudders. Two elevons are hinge-mounted to the upper trailing edge of each wing, one inboard and one outboard of the respective engine nacelle. A tetrahedral-shaped rudder is mounted to a fixed stub fin on the upper aft portion of each engine nacelle. Each rudder assembly is canted inward 15 degrees.

A servo assembly at each control surface meters dual system (A and B) hydraulic power for positioning the control surfaces. The control stick and rudder pedals are connected to the servos by cable and mechanical input systems. Feel springs in each axis provide the pilot with control feel proportional to the degree of control deflection. Artificial feel is provided since air loads are not felt by the pilot. Pilot inputs are limited to the force necessary to move the metering valve in the servos.

Control Stick

A conventional control stick operates the elevons. See Figure 1-49. Movement of the stick is approximately 9° forward and 16° aft of its neutral position and (with the control surface deflection limiters engaged) approximately 5 1/2° laterally. With the limiters disengaged, the stick can be positioned laterally approximately 8° from center at any point, and approximately 9° when not near its extreme forward or aft position. Similarly, full forward and aft stick positioning capability is reduced somewhat at the extreme "corners" of the deflection "box" when the limiters are disengaged. (This results because the elevon deflection angles are additive for combined pitch and roll commands. If surface deflection limits are reached, maximum stick pitch and roll command angles are also reached.) Full lateral movement requires approximately 10 pounds force. Approximately 25 pounds push force and 45 pounds pull force are required to reach the full forward and aft stick positions, respectively.

Three switches are located on the top and one switch on the right side of the grip. On the top left, a three-position communications switch labeled TRANS (up) and INPH (down) is springloaded to an unmarked off (center) position. The pilot's microphone is connected to the interphone system or to a selected radio transmitter when INPH or TRANS is selected, respectively. The microphone is disconnected when the center position is used unless the throttle-mounted microphone switch is pressed for radio communication or the interphone HOT MIC knob is pulled out. See Communications and Avionic Equipment, this section. A four-way (center-off) pitch and yaw axis TRIM switch operates as described under Manual Trim System, this section. A dual purpose CSC/NWS push-button switch, located to the right of the trim switch, either activates the autopilot Control Stick Command feature (if the autopilot is on) or engages/releases nosewheel steering while on the ground. Before S/B R-2674, pressing a rain removal system switch on the right side of the grip causes a quantity of rain removal fluid to be applied to the forward windshield panels if the windshield deicing switch on the left side of the pilot's annunciator panel has been set to RAIN REMOVAL ARM ON (up). After S/B R-2674, rain removal is deactivated.

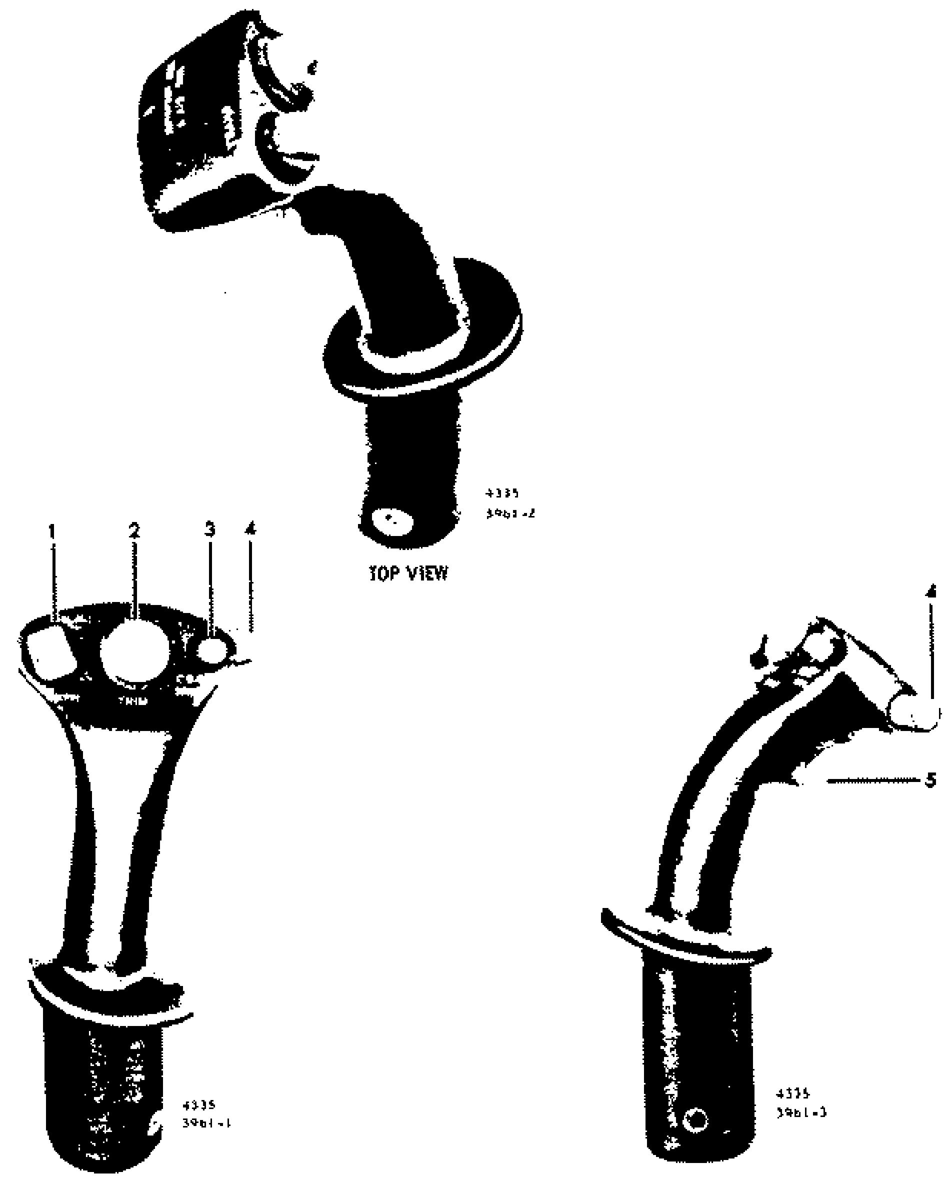
NOTE

> Do not operate the rain removal system unless the windshield is wet. The white fluid will stick to the glass and permanently obscure visibility if applied while the windshield is dry.

A multipurpose trigger switch is on the forward side of the grip. Operation of this switch disconnects air refueling, disengages autopilot, interrupts ac power to the pitch and yaw manual trim actuator motors (disabling the control stick trim switch and the RH RUDDER SYNCHRONIZER switch), interrupts the APW system stick pusher, and with S/B R-2695 interrupts antiskid system operation.

A stick-shaker motor is installed below the grip. It warns of a potentially high angle of

CONTROL STICK GRIP



NOTE

1. Transmitter-interphone microphone selector switch
2. Pitch and yaw trim switch
3. Control stick command-nosewheel steering button
4. Rain removal switch (Deactivated with S/B R-2674)
5. Trigger switch - disconnects air refueling and autopilot; interrupts pitch and yaw trim, APW stick pusher, and antiskid (with S/B R-2695).

Figure 1-49

attack and/or pitch rate. Operation of the shaker is controlled by the APW and High Alpha Warning systems. A red SHAKER warning light, located near the apex of the pilot's glareshield, illuminates when the shaker is on. A stick-pusher mechanism is also part of the APW system. The pusher displaces the stick forward to initiate corrective action in the pitch axis. (Refer to APW and High Alpha Warning Systems, this section.) Hydraulic pressure from the A system is required for pusher operation.

ELEVON CONTROL SYSTEM

The delta wing configuration uses elevons for pitch and roll control. The elevons respond to control inputs from the control stick, pitch and roll trim systems, SAS, and the autopilot. All control inputs are applied to the inboard elevon servo assemblies, which control actuation power for elevon positioning. The outboard elevons are mechanically slaved to their respective inboard elevon.

Elevon Control Cables

The control stick is connected by mechanical linkage to pitch and roll tension regulators in the cockpit. Dual control cables run from each tension regulator to respective cable quadrants above the mixer assembly in the tail cone.

Elevon Mixer Assembly

The mixer assembly provides the mechanical geometry necessary to sum the pitch and roll inputs from their respective control cable systems. The combined inputs are then applied as a single control input to the inboard elevon servo assemblies. The pitch and roll trim actuators are an integral part of the mixer assembly and are summed along with control stick inputs. All pitch and roll trim (Mach trim and auto trim included) is applied through the mixer assembly. The pitch and roll feel springs are in the mixer. Since the trim actuators are mounted downstream of the feel springs, the control stick is not displaced by elevon trim (pitch and roll) actuations.

Inboard Elevon Servos

All pitch and roll control inputs (stick, trim, SAS, and autopilot) are applied to the inboard elevon servos. In response to these inputs, the servos meter A and B hydraulic pressures to the actuating cylinders, which position the inboard elevon surface. Each hydraulic system provides power to three actuating cylinders at each inboard servo assembly.

Input Mechanism and Spring Cartridge

The outboard elevons are slaved to their respective inboard elevon by a mechanical input which connects the inboard surface to the outboard elevon servo. Thus, any movement of the inboard surface moves the outboard servo, which positions the outboard elevon.

A spring-loaded cartridge pushrod "shotgun" is installed in the inner wing portion of the input mechanism. Cartridge spring loads maintain four rollers in a detent during normal surface control operation, permitting the cartridge to act as a solid pushrod. If excessive backloads (approximately 900 pounds) are imposed on input mechanism movement by binding or seizure outboard of the cartridge, the rollers "jump their detent" to permit operation of the inboard elevon independent of the affected outboard elevon. Without this feature, failure at the outboard servo assembly or within the input mechanism could result in structural damage and/or loss of control in pitch and roll.

When operating the controls on only one hydraulic system, half of the actuating cylinders are inoperative and sufficient back pressure may be imposed on the input mechanism to cause the spring cartridge rollers to jump the detent at lower system pressures (approximately 1500 psi).

CAUTION

When starting engines, do not move the control stick until at least 1500 psi can be maintained on the A or B hydraulic system.

SURFACE CONTROL DEFLECTION LIMITS AND RATES

MODE OF OPERATION	SURFACE AUTHORITY LIMITS ⚠1			SURFACE MAXIMUM RATES		
	PITCH	ROLL ⚠2	YAW	PITCH	ROLL ⚠2	YAW
MANUAL	10° DOWN ⚠3 TO 24° UP ⚠4	24°	20° LEFT TO 20° RIGHT	32.5°/SEC	65°/SEC	37°/SEC
LIMITED MANUAL		14°	10° LEFT TO 10° RIGHT			
APW PUSHER EXTENDED ⚠6	10° DOWN ⚠3 TO 10° UP ⚠5					
SAS	6.5° UP TO 2.5° DOWN	4°	8° LEFT TO 8° RIGHT	15°/SEC	30°/SEC	28°/SEC
AUTOPILOT	2.3° DOWN TO 2.3° UP ⚠7					
AUTO-TRIM MACH TRIM	5.0° DOWN TO 8.5° UP			NOMINAL 0.113° SEC		
MANUAL TRIM		9°	10° LEFT TO 10° RIGHT	NOMINAL 1.13°/SEC	NOMINAL .96°/SEC	NOMINAL .90°/SEC

NOTE

⚠1 Combined pitch and roll application is limited by actuating cylinder stroke extremes at 20° down to 35° up.

⚠2 Roll figures reflect differential roll applied.

⚠3 10° down if pitch trim indicator is at or below zero. Trim above zero indication decreases down elevon authority.

⚠4 24° up if pitch trim indicator is at or above zero. Trim below zero indication decreases up elevon authority.

⚠5 Assumes pitch trim indicator at zero. Trim above zero indication increases up elevon authority. Trim below zero indication decreases up elevon authority.

⚠6 Pusher deflects elevons 1.7° down from trimmed position. Up elevon movement above trimmed position requires additional force to overcome pusher.

⚠7 Autopilot authority limits are 2.3° above FL500, and 1.6° below FL500.

F203-128(a)

Figure 1-50

SECTION I

FLIGHT CONTROL SYSTEMS

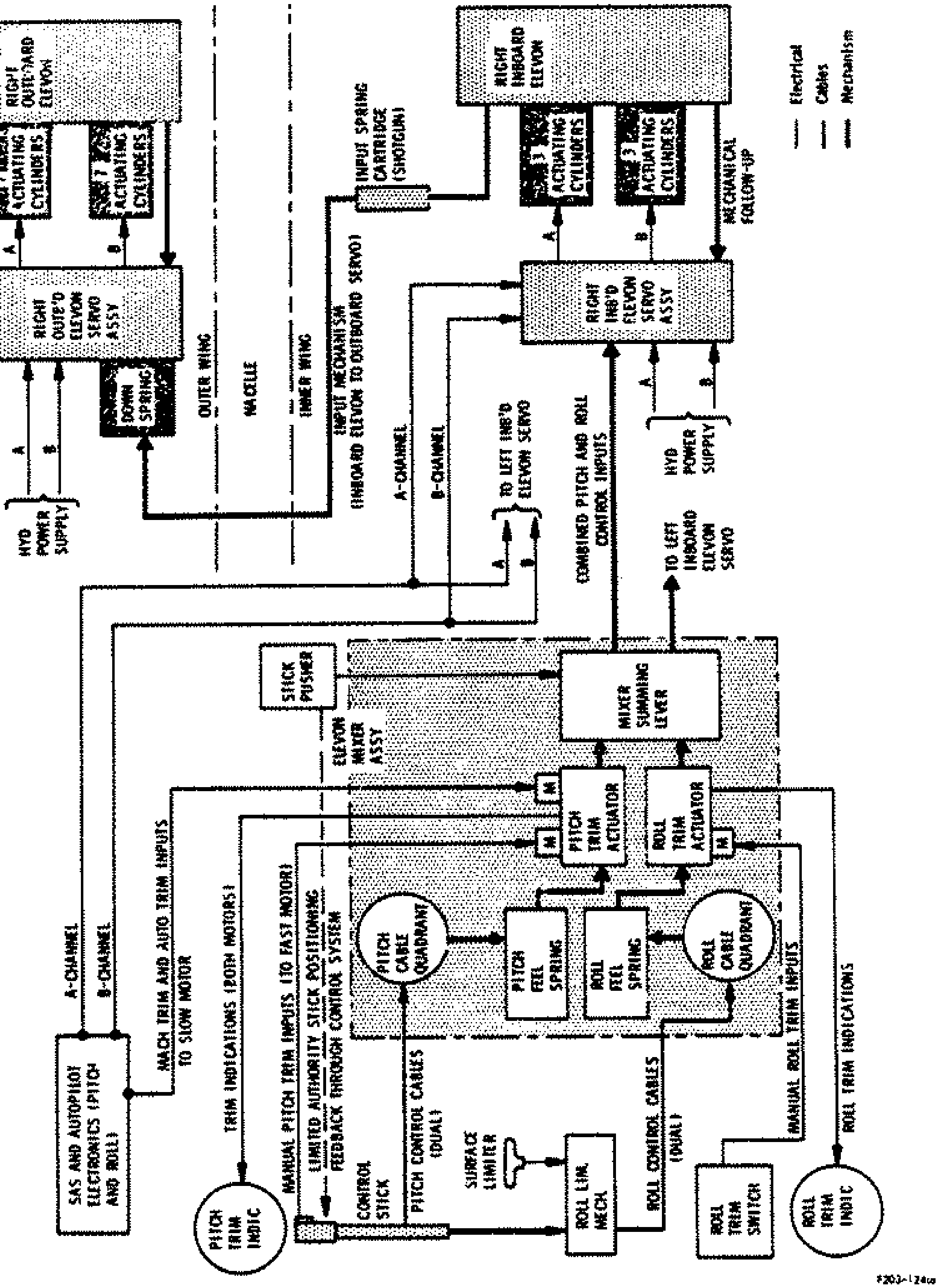


Figure 1-51

RUDDER CONTROL SYSTEM

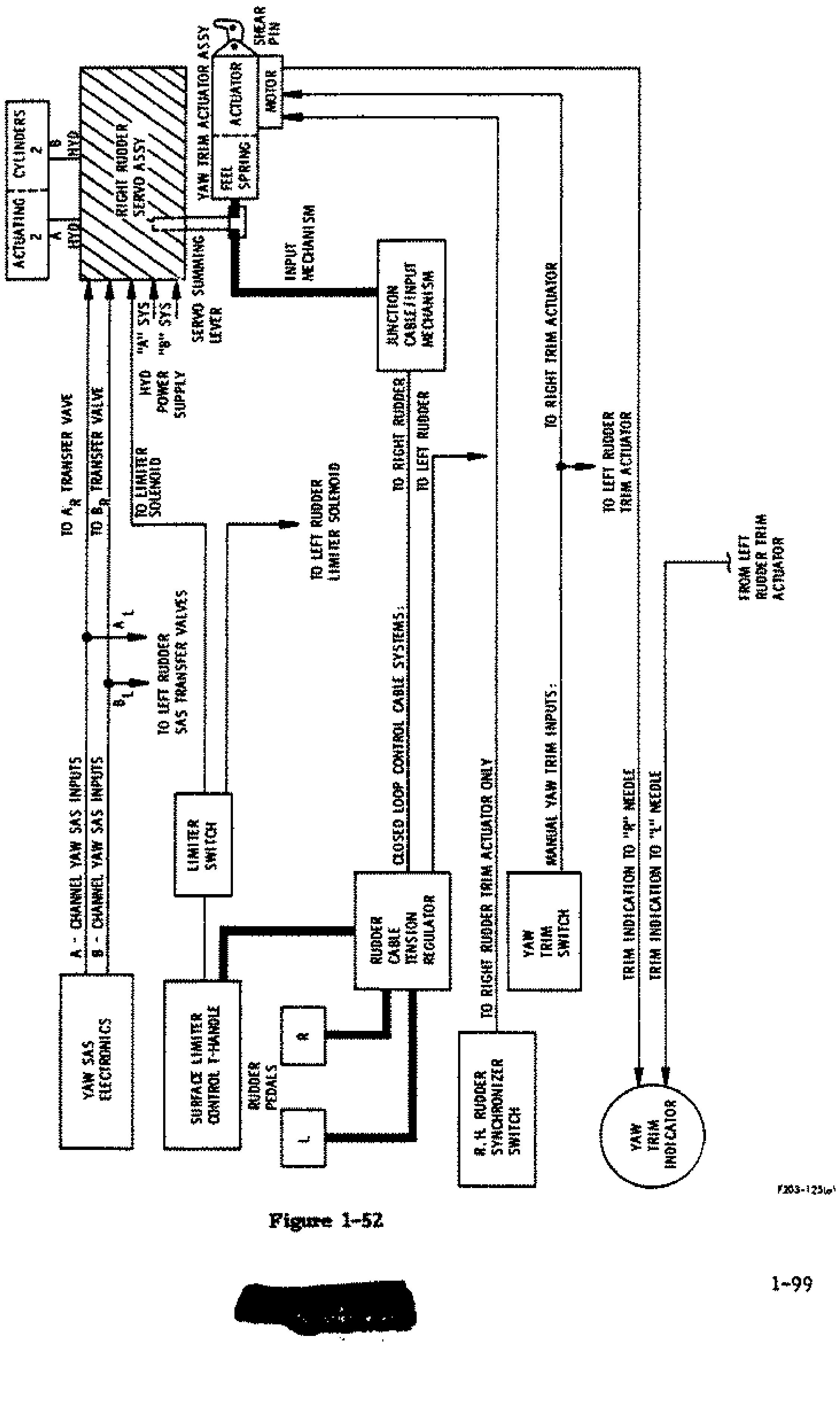


Figure 1-52

Outboard Servo and Limiter Spring

Inboard elevon movement is transmitted through the input mechanism to the outboard servo, which meters A and B hydraulic pressures to 14 actuating cylinders at each outboard elevon. Half of the cylinders are powered by each hydraulic system.

A limiter spring at the outboard servo input lever ensures that the outboard elevon does not travel full down if the input mechanism fails. Limiter spring force, in conjunction with servo bias spring loads, will position and maintain the outboard elevon at a three-degree-down position if disconnected. (Normal function of the servo bias spring is to apply a down-elevon load on the input mechanism to eliminate hysteresis.)

RUDDER CONTROL SYSTEM

The two full-moving rudders, which provide yaw (directional) control and stability, are positioned by control inputs from the rudder pedals, manual yaw trim system, and yaw SAS. Individual servo assemblies, which include a trim actuator and yaw feel spring, are installed in the fixed stub fin section of each rudder assembly.

Rudder Pedals and Input Mechanism

The rudder pedals connect to the yaw tension regulator by push-pull rods and bellcranks. A single closed-loop cable system to each rudder originates at the tension regulator and terminates at an input mechanism at the inboard side of each engine nacelle. The input mechanism transmits cable movement to the input lever at each rudder servo.

The rudder pedals are also used for main wheel braking (toe action) and nosewheel steering.

Pedal position is adjusted by the PEDAL ADJ T-handle, located at the bottom of the annunciator panel. To adjust rudder pedals, hold pedals and pull the PEDAL ADJ T-handle; the pedals are free to move fore-and-aft. Push or release the pedals to the desired position and release the T-handle to lock the pedals in place.

Rudder Servo Assembly

All yaw control inputs (pedals, manual trim, and SAS) are applied through the rudder servo which meters A and B hydraulic pressures to four actuating cylinders for positioning the rudder. (Two actuating cylinders at each servo powered by each hydraulic system.) A yaw trim actuator, incorporating a yaw feel spring, is installed at each servo. Yaw (rudder) trim is reflected in proportional rudder pedal movement.

MANUAL TRIM SYSTEM

All aircraft trim is achieved by positioning the main control surfaces. The pitch and roll trim actuators are an integral part of the elevon mixer assembly and a yaw trim actuator is installed at each rudder servo assembly.

Power for the manual pitch and yaw actuator motors is furnished by the ac hot bus through the PITCH & YAW circuit breaker on the pilot's right console. Power for the roll actuator motor is furnished by the essential ac bus through the ROLL circuit breaker on the pilot's right console.

Trim Power Switch

A TRIM POWER switch is located on the left side of the pilot's annunciator panel. In OFF, all trim (manual and auto) is inoperative. In ON, the trim system is operable.

Pitch, Roll, and Yaw Trim Indicators

Separate pitch, roll, and yaw trim indicators are on the pilot's instrument panel. The pitch trim indicator displays the sum of manual and Auto/Mach trim. SAS inputs are not shown. The roll trim indicator displays differential roll trim from 0 to 9 degrees. The yaw trim indicator displays the position of the left and right actuators individually on L and R needles. These needles are aligned (superimposed) when equal trim is applied at both rudders.

Power for the roll and yaw trim indicators is furnished by the essential ac bus 26 volt instrument transformer through the YAW and ROLL circuit breakers on the pilot's annunciator panel. Power for the pitch trim indicator is furnished by the emergency ac bus 26 volt instrument transformer through the PITCH circuit breaker on the pilot's annunciator panel.

Pitch Trim System

The pitch trim actuator may be operated by manual control, using a fast motor, or automatic control (including Mach Trim), using a slow motor. The manual (fast) trim motor trims ten times faster than the Auto/Mach (slow) trim motor. Pitch trim indication signals are provided by two position transmitters on the actuator. The slow motor transmitter also provides feedback signals to the autotrim system.

Pitch and Yaw Trim Switch

The manual pitch trim control switch is combined with the yaw trim switch on the control stick grip. Moving the switch up and down (from the spring-loaded center off position) applies nose-down and nose-up trim, respectively. Pitch trim application is limited to 5° ($-1/2^{\circ}$ $+1^{\circ}$) down and 8.5° (-1°, $+1/2^{\circ}$) up by actuator stroke limitations.

Moving the switch left and right applies left yaw and right yaw trim, respectively, to both rudders.

Manual pitch and yaw trim can be disconnected by pressing the trigger switch on the control stick grip. Mach trim is not affected.

CAUTION

To avoid runaway trim due to a sticking trim switch, assure positive switch movement to neutral after each actuation.

Roll Trim Switch

A three-position roll trim switch, located forward of the throttle quadrant, provides manual control of the roll trim actuator in the elevon mixer assembly. The self-centering switch may be toggled left and right to apply left roll and right roll trim, respectively. Roll trim application is limited (by actuator stroke length) to $\pm$ 4.5 degrees, for a maximum of 9 degrees differential roll trim.

Yaw Trim System

Yaw trim is applied by two trim actuators, one at each rudder servo assembly. The yaw function of the pitch and yaw trim switch on the control stick grip energizes both rudder trim actuators simultaneously. A right hand rudder synchronizer switch, located forward of the throttle quadrant, energizes the right rudder trim actuator only. Yaw trim is limited to approximately 10 degrees left and right by actuator stroke limits.

A shear pin is in the attach fittings of each actuator so rudder pedal inputs cannot be blocked by failure of a yaw feel spring in the trim actuator. If a seizure does occur, pedal inputs can break the shear pin and free the actuator attach point. When an actuator shear pin is broken, some yaw feel is lost and little or no yaw trim is available at the affected rudder.

NOTE

The trim actuator remains operative and the trim indicator needles will indicate normal trim operation, even though trim is applied at only one rudder. Rudder centering will be poor.

Right Hand Rudder Synchronizer Switch

Due to variations in rudder actuator motor speeds, yaw trim may not be applied equally to both rudders. A split in rudder position and yaw trim needle indications may result. A three-position, self-centering, RH RUDDER SYNCHRONIZER switch (located left of the roll trim switch) should be used to equalize left and right rudder trim positions. The switch may be toggled left and right to move the trailing edge of the right rudder left and right, respectively.

SURFACE LIMITER SYSTEM

Lateral (roll) control stick travel and rudder displacement are restricted by the surface limiter system. The system should be released (full travel mode) at speeds below Mach 0.5, and engaged (limited mode) at higher speeds. The surface limiter system is controlled by the SURF LIMIT RELEASE T-handle on the left side of the pilot's annunciator panel.

The T-handle is spring-loaded in the full forward (limited) position, and must be pulled aft to disengage the limiters and rotated 90 degrees clockwise to lock the handle aft. The T-handle is released (to engage the limiters) by rotating the handle 90 degrees counterclockwise.

A SURFACE LIMITER caution light on the pilot's annunciator panel illuminates if the limiters are in the wrong mode for the existing aircraft speed. The light is controlled by T-handle switch contacts and Mach inputs from DAFICS. With the limiters released, the light illuminates above Mach 0.5, and goes out when limiting is engaged. With limiters engaged, the light illuminates below Mach 0.5, and goes out when the full travel mode is selected.

Roll Travel Limiter System

With limiters engaged, a pin is inserted into a cam at the base of the control stick, physically limiting elevon travel to 14 degrees roll differential. Unlimited, the maximum roll differential is 24 degrees.

Rudder Travel Limiter System

Rudder travel is reduced from 20 degrees to 10 degrees left and right when the limiters are engaged. Rudder travel is limited by two different methods to ensure that no combination of rudder pedal, yaw trim, or yaw SAS can position the rudders beyond the limits. One method (mechanical), limits rudder pedal inputs by insertion of a pin between two stops on the rudder tension regulator in the forward cockpit. The second method (electro-hydraulic) limits travel of the rudder servo input lever at each servo by hydraulically extending solenoid limiter stops to restrict input lever movement.

The limiter control solenoids, located at each rudder servo, control hydraulic pressure for extension or retraction of the solenoid limiter stops. The control solenoids must be energized to permit hydraulic retraction of the stops when the limters are released. Power for solenoid control is furnished by the essential dc bus through individual RUD LIM circuit breakers on the pilot's left console. If limiter control power is lost, the solenoid stops extend to the limited position.

If power to one of the solenoid limiters is lost (loss of dc control power, relay failure, control circuit breaker trip, etc), rudder travel on the affected rudder will be limited to 10 degrees and the SURFACE LIMITER light will remain illuminated when full travel is selected. With one rudder limited and full travel selected, the non-limited rudder can be positioned well beyond the 10 degree limit by applying abnormal pressure on the rudder pedals.

DIGITAL AUTOMATIC FLIGHT AND INLET CONTROL SYSTEM (DAFICS)

The Digital Automatic Flight and Inlet Control System (DAFICS) comprises five major subsystems: stability augmentation system, autopilot/Mach trim system, automatic pitch warning and high angle of attack system, automatic/manual inlet control system, and air data system. The autopilot utilizes inputs from the ANS or the INS.

Air pressures measured independently by the A, B, and M channels of the pressure transducer assembly (PTA) are transmitted to the respective (A, B, and M) DAFICS computers, then shared between computers to determine best available measurements. DAFICS computes air data for schedules in DAFICS subsystems and cockpit displays (including the TDI).

The stability augmentation system provides automatic stability augmentation in the pitch, roll and yaw axis. The autopilot provides automatic flight control in the pitch and roll axes, and the Mach trim system provides speed stability augmentation in the pitch axis. The automatic pitch warning system provides a control stick shaker and stick pusher when approaching flight limits in the pitch axis. The inlet control system provides automatic and manual control of inlet air flow. (Refer to Figures 1-53 and 1-54.)

<u>Computer Reset Switches</u>

Individual A, B, and M CMPTR RESET switches on the pilot's annunciator panel allow manual restart of a computer. Holding a reset switch in the up position stops the respective DAFICS computer. Releasing the reset switch to the spring-loaded down position initiates restart. Deliberate computer(s) shutdown inflight is not authorized. The switches are powered by their respective A, B, and M CMPTR dc circuit breakers on the pilot's left console.

COMPUTER BUILT IN TEST

Computer inflight BIT performs a series of tests throughout flight to insure A, B, and M computer health and if any test fails, the computer will automatically shut itself down and turn on the appropriate CMPTR OUT light on the pilot's annunciator panel.

DAFICS PREFLIGHT BUILT IN TEST

The DAFICS preflight built in test (BIT) is normally accomplished before and after flight. The DAFICS PREFLIGHT BIT switch is located on the pilot's right console.

If the following requirements are not satisfied, the DAFICS PREFLIGHT BIT switch will not engage.

1. Hydraulic pressure - A system.
2. CMPTR OUT lights (3) not illuminated.
3. CSC/NWS switch - Released.
4. APW switch - PUSHER/SHAKER.
5. SPIKES & FWD BYPASS doors -AUTO.
6. RESTART switches - Off.
7. Throttle Restart switch - Off.
8. SAS channel engage switches - ON.
9. AUTOPILOT PITCH & ROLL switches - ON.
10. KEAS HOLD switch - ON.
11. HEADING HOLD switch - ON.

If B, L, or R hydraulic pressure is not available, the DAFICS PREFLIGHT BIT will fail.

During the DAFICS PREFLIGHT BIT, the APW pusher operates momentarily, the SHAKER warning light flashes momentarily, the A, B and M CMPTR OUT annunciator lights flash several times, the SAS SENSOR and SERVO lights flash momentarily, control surfaces move slightly, and the spikes move. The test cycle can be terminated manually by stopping any DAFICS computer. The BIT TEST light illuminates steady green while the test is running.

The test takes about one minute to complete. If the test is successful, the BIT TEST light flashes green at completion of the test. If any malfunctions are detected, the test continues, but at completion of the test the BIT TEST light extinguishes and the BIT FAIL light illuminates steady red.

When the DAFICS preflight BIT check is finished, indications are:

1. BIT FAIL light - Extinguished.
2. BIT TEST light - Flashing green.

or

1. BIT FAIL light - Steady red.
2. BIT TEST LIGHT - Extinguished.

3. MASTER CAUTION light - On.
4. SAS OUT annunciator panel light - Flashing.
5. Autopilot pitch and roll switches - Off.
6. HEADING HOLD switch - Off.
7. KEAS HOLD switch - Off.
8. AUTOPILOT OFF light - On.
9. OFF flags on both TDI's.
10. CIP barber pole at zero.
11. Spikes full forward.
12. DAFICS PRELIGHT BIT switch - OFF, automatically.

When the BIT terminates, if a steady red BIT TEST light, any SENSOR light, any SERVO light, or any CMPTR OUT light illuminates, notify maintenance.

The DAFICS remains in the test mode. Pressing one of the six SENSOR/SERVO recycle switches resets DAFICS to the flight mode. When DAFICS is reset to the flight mode, indications are:

1. A, B, and M CMPTR OUT lights - Flash, momentarily.

2. BIT TEST light - Off.

3. CIP Barber pole - Normal position.

4. Both TDI's will initiate resynchronization and run up to 55,000 ft, Mach 2.0, and 300 KEAS. AOA will indicate 10°. AOA will return to 0° in approximately 1 minute and 15 seconds and TDI indications will return to normal in approximately 2 minutes and 15 seconds after resetting DAFICS to the flight mode.

WARNING

- Failure to recycle a SENSOR/SERVO recycle switch will cause the DAFICS system to remain in the ground test mode. The SAS is non-functional while in the ground test mode. DAFICS will not operate normally until the system is reset.

- Do not attempt to activate the DAFICS PREFLIGHT BIT switch during flight. The DAFICS PREFLIGHT BIT check is inhibited unless there is weight on wheels, Mach number is less than 0.09, or KEAS is less than 101.

NOTE

Once DAFICS is reset to the flight mode, a flashing red BIT FAIL light indicates loss of SAS analytical redundancy.

STABILITY AUGMENTATION SYSTEM (SAS)

The stability augmentation system (SAS), a combination of electronic and hydraulic equipment, is an integral part of the basic aircraft control system. The system is normally engaged in all flight conditions, although it can be disengaged manually. Each axis of SAS (pitch, roll and yaw) is provided with two SAS channels. The SAS detects aircraft attitude changes and initiates control surface deflections to counteract the changes. Normally, the DAFICS A computer runs the A channel in pitch and yaw and the DAFICS B computer runs the B channel in pitch and yaw. The DAFICS M computer can take over for A computer or B computer or both in the pitch and yaw axis should A and/or B computer fail. The M computer drives through servo amplifiers in the A and B computer to provide surface control. The roll SAS is configured so that either A or B computer is capable of driving both roll servo channels. Sensor and servo monitors provide detection and automatic disengaging capability for faults.

During normal flight conditions, the aircraft experiences many small changes in attitude due to air loads or control inputs. These attitude changes are sensed by pitch, yaw, and roll sensors in each axis (three rate gyros in the pitch axis, three rate gyros plus three lateral accelerometers in the yaw axis, and two rate gyros in the roll axis). Analytical redundancy derived by the DAFICS computers from attitude displacements provides added redundancy for pitch, yaw and roll rate gyros, but not for the lateral accelerometers. The attitude changes detected by the sensors are sent to the DAFICS computers which electrically command the transfer valve positions of the SAS servos. The transfer valve converts the electrical signal into a proportional hydraulic flow into the SAS servo actuators. The SAS

COMPUTER INPUTS

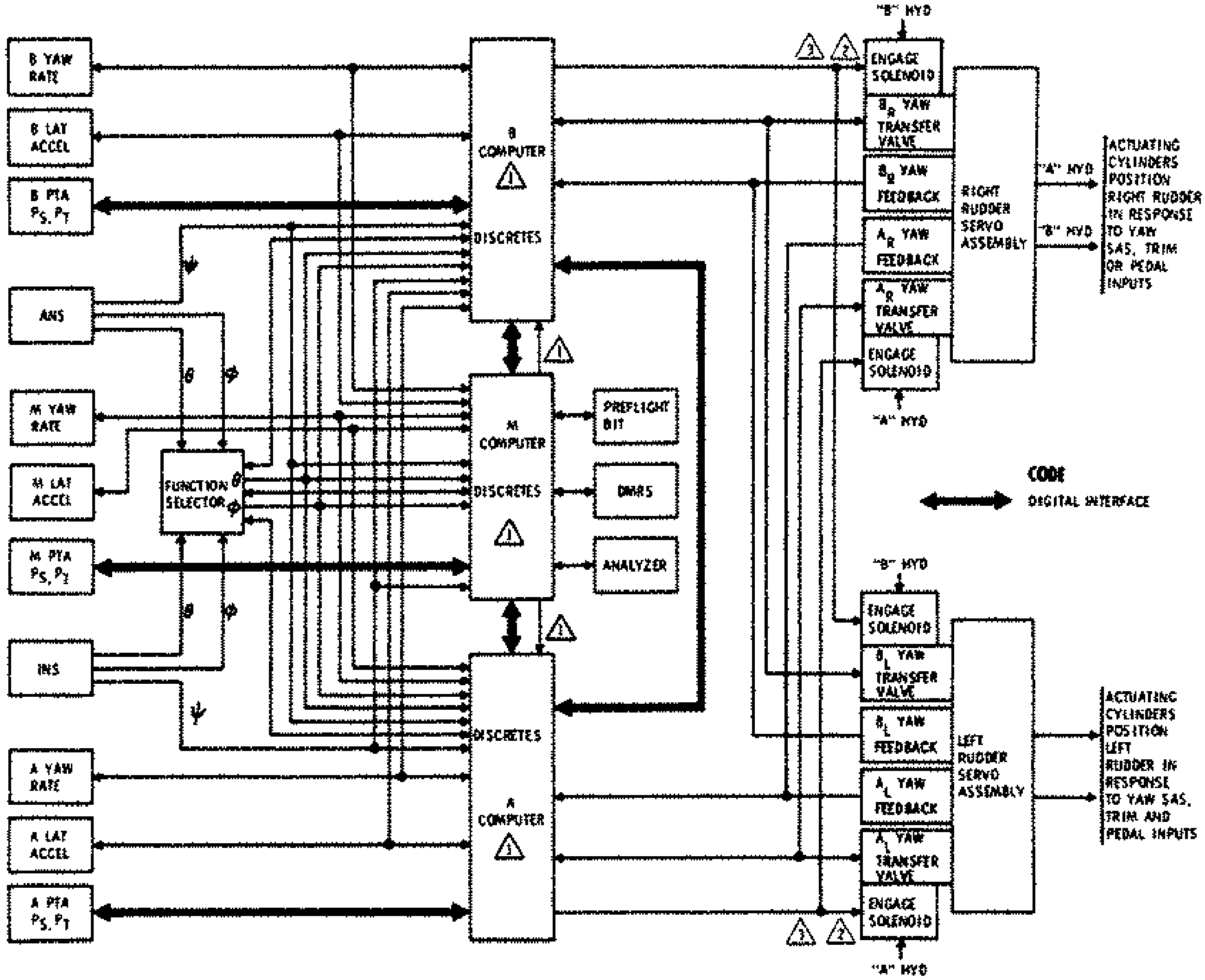
COMPUTER INPUTS	DAFICS: SAS PITCH	SAS YAW	SAS ROLL	AUTOPILOT & MACH TRIM	AIR DATA	APW & HIGH q WARNING	AIR INLET SYSTEM	COMPUTERS A	COMPUTERS B	COMPUTERS M
A CMPTR RESET Switch								X		
B CMPTR RESET Switch									X	
M CMPTR RESET Switch										X
P_s	X	X	X	X	X	X	X	X	X	X
P_T	X	X	X	X	X	X	X	X	X	X
ΔP_α						X	X	X	X	X
ΔP_β							X			X
Duct Pressure Ratio Transducers (Left and Right)							X	X	X	
INLETS: AUTO SPIKE Switch Positions							X	X	X	X
AUTO FWD BYPASS Switch Positions							X	X	X	X
Manual Spikes							X			X
Manual Forward Bypass Doors							X			X
Manual BIAS Schedule Controls							X			X
D-BIAS (Δ_T)							X	X	X	-
Total Temperature (From Left Inlet) (1)								X	X	X
ELEVON SERVOS (2 Left and 2 Right)	X		X					X	X	X
RUDDER SERVOS (2 Left and 2 Right)		X						X	X	X
SURFACE RELEASE T-Handle Position					X			X	X	
SAS: A Pitch Rate Sensor	X			X		X		X	X	
B Pitch Rate Sensor	X			X		X			X	X
M Pitch Rate Sensor	X			X		X		X		X
A Yaw Rate Sensor		X						X	X	
B Yaw Rate Sensor		X							X	X
M Yaw Rate Sensor		X						X		X
A Roll Rate Sensor			X	X				X	X	
B Roll Rate Sensor			X	X				X	X	
A Lateral Accelerometer Sensor		X						X	X	
B Lateral Accelerometer Sensor		X							X	X
M Lateral Accelerometer Sensor		X						X		X
A & B Pitch Channel Switches	X							X	X	X
A & B Yaw Channel Switches		X						X	X	X
A & B Roll Channel Switches			X					X	X	
A PITCH SENSOR Switch/Light	X							X	X	
B PITCH SENSOR Switch/Light	X								X	X
M PITCH SENSOR Switch/Light	X							X		X
A YAW SENSOR Switch/Light		X						X	X	
B YAW SENSOR Switch/Light		X							X	X
M YAW SENSOR Switch/Light		X						X		X
A PITCH SERVO Switch/Light								X		X
B PITCH SERVO Switch/Light									X	X
A YAW SERVO Switch/Light								X		X
B YAW SERVO Switch/Light									X	X
A ROLL SERVO Switch/Light								X	X	
B ROLL SERVO Switch/Light								X	X	
DAFICS PREFLIGHT BIT Switch	X	X	X	X	X	X	X	X	X	X
AUTOPILOT: All AUTOPILOT Panel Switches & Trim Wheels				X				X	X	
CSC Switch				X				X	X	
Trigger Switch (A/P Pedal Disengage Switch)				X		X		X	X	
ATT REF SELECT Switch	X		X	X		X		X	X	X
INS: Pitch	X	X	X	X				X	X	X
Roll	X	X	X	X				X	X	X
Heading	X	X	X	X				X	X	X
Attitude Ready				X				X	X	
Heading Valid				X				X	X	
ANS: Pitch	X		X	X		X		X	X	X
Roll	X		X	X		X		X	X	X
Heading	X		X	X		X		X	X	X
Steering Commands				X				X	X	
NAV READY				X				X	X	
Hydro Pressure	X	X	X	X		X	X	X	X	X
Weight-On-Wheels (WOW)				X		X		X	X	X
Gear Up Locks						X		X	X	X
Gear Down Locks							X			X

(1) Used in ANS and Map Projector

F308-27(c)

Figure 1-53

YAW SAS BLOCK DIAGRAM



SENSOR (RATE) SELECT

SENSOR CONDITION	SENSOR USED A	SENSOR USED B	SENSOR USED M	SENSOR OUTPUT SELECTED
Normal	A	B	M	Median of A, B, M
Fail A	M	B	M	Average of B, M
Fail B	A	A	M	Average of A, M
Fail M	A	B	B	Average of A, B
RATE GYRO WITH ANALYTICAL REDUNDANCY				
Fail A, B	M	-	M	M
Fail B, M	A	A	-	A
Fail A, M	-	B	B	B
RATE GYROS WITHOUT ANALYTICAL REDUNDANCY OR LATERAL ACCELEROMETER				
Fail A, B	-	-	-	None
Fail B, M	-	-	-	None
Fail A, M	-	-	-	None

COMPUTER SELECT

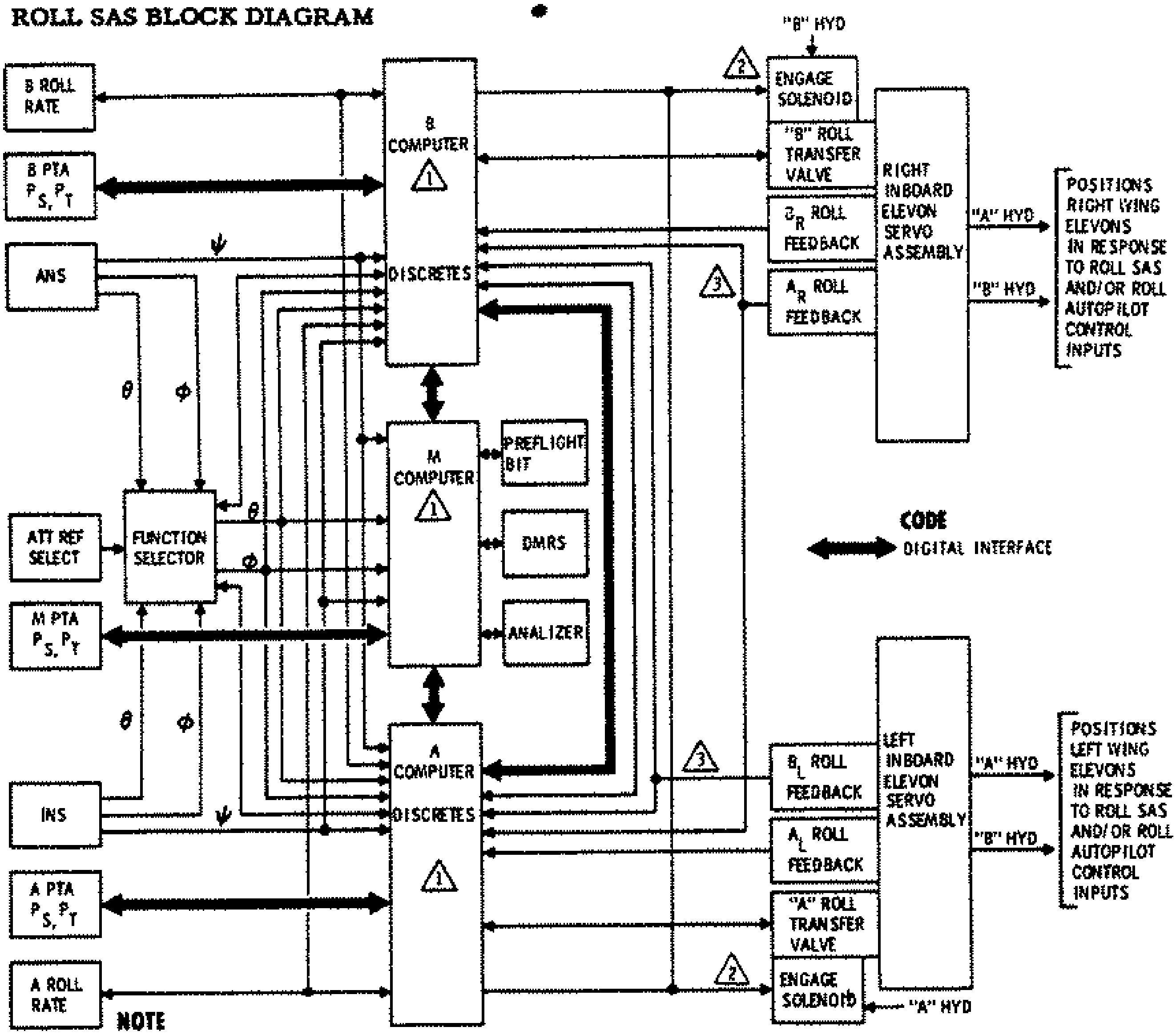
COMPUTER CONDITION	COMPUTER DRIVING SERVOS A_L	A_R	B_L	B_R	BACKUP
NORMAL	A	A	B	B	M for A, B
FAIL A	M	M	B	B	M for B
FAIL B	A	A	M	M	M for A
FAIL M	A	A	B	B	None
FAIL A, B	M	M	M	M	None
FAIL A, M	-	-	B	B	None
FAIL B, M	A	A	-	-	None

NOTE

1. YAW SAS Control
2. Logic outputs are generated by a "difference" between compared inputs, which disengage the faulty channel and energize SAS lamps to indicate whether fault was in SAS sensors or SAS servo. Gain is doubled in the remaining active channel to maintain effective Yaw SAS.
3. Controls engagement of SAS channel by controlling hydraulic flow through the transfer valve. With switch OFF or logic tripped, hydraulic flow is blocked and SAS signals cannot position rudders. Switch A controls A_L and A_R solenoids. Switch B controls B_L and B_R solenoids.

Fi[redacted]

ROLL SAS BLOCK DIAGRAM



NOTE

Roll SAS Control

a. Both switches OFF to ON engages both SAS channels and "ARMS" roll logic.

b. Sensor Failures
   - First sensor failure - Normal roll SAS capability.
   - Second sensor failure - No roll SAS capability.

⚠1 SENSOR (RATE) SELECT

SENSOR CONDITION	SENSOR USED COMPUTER A	SENSOR USED COMPUTER B	SENSOR OUTPUT SELECTED
NORMAL	A	B	AVERAGE OF A, B
WITH ANALYTICAL REDUNDANCY			
Fail A	B	B	B
Fail B	A	A	A
Fail A, B	-	-	None
WITHOUT ANALYTICAL REDUNDANCY			
Fail A	-	-	None
Fail B	-	-	None

⚠2 Controls engagement of SAS channel by controlling hydraulic flow through transfer valve. With switch OFF or logic "TRIPPED", hydraulic flow is blocked and SAS/Autopilot control signals cannot position elevons.

c. Servo Failures
   - Servo logic "trip" disengages both channels.
   - For a hard failure, single channel operation can be selected by disengaging both A and B engage switches and re-engaging channel opposite from failed servo light.
   - Single channel operation overrides all servo logic protection.
   - For momentary failure, recycle either engage switch to resume normal operation.

COMPUTER SELECT

COMPUTER CONDITION	COMPUTER DRIVING SERVOS LEFT	COMPUTER DRIVING SERVOS RIGHT	BACKUP
Normal	A	B	Other Computer
Fail A	B	B	None
Fail B	A	A	None

⚠3 Inputs indicate direction and degree of elevon deflection applied by SAS servos. An imbalance equal to 0.6 degrees elevon position will generate a logic "TRIP" signal.

F203-27(14)

Figure 1-57

SAS, differential elevon available is reduced from 4 to 2 degrees with some coupling into the pitch and yaw axes.

Analytical redundancy in the DAFICS computers derives a third roll rate to use for voting after a single rate gyro has failed. This capability provides dual sensor redundancy. (See Figure 1-57 for Roll SAS block diagram.)

SAS Controls and Indicators

Manual engage, disengage, and recycle control of SAS is provided by controls on the function selector panel on the pilot's right console. Indicator lights (recycle light/-switches) on the function selector panel display status of the system sensors and servos. When not in the ground test mode, the red BIT FAIL light flashes if analytical redundancy is not available. The SAS OUT light on the annunciator panel and the master CAUTION light illuminate for SAS failures but not for analytical redundancy failure. Refer to Figure 1-58, SAS Controls and Indications Table, for details of switch functions.

SAS LOGIC

DAFICS logic monitors voltages within all three axes of SAS so that malfunctions within the SAS can be detected at an early stage. The system is extremely sensitive and voltage tolerances are small, so even minor errors are detected to provide maximum protection. The logic system will disengage the affected servo when a difference in voltage is detected which is equivalent to approximately 2 degrees surface deflection error in pitch and yaw, and 0.6 degree in roll.

Pitch SAS

Sensors

Computer sensor select capability selects median between A, B and M gyro outputs and uses that signal for A, B or M computer SAS control. If tracking requirements are exceeded (first failure) the gyro that is out of tolerance is voted out and the appropriate PITCH SENSOR light on the function selector and the SAS OUT light on the annunciator panel are turned ON. The remaining two gyro outputs are now averaged and that signal is used for A, B or M computer SAS control. If the remaining two gyros exceed the tracking requirement (second failure) both gyros are compared with analytical redundancy derived pitch rate to pick out the failed gyro (appropriate PITCH SENSOR light on function selector and SAS OUT light on annunciator panel are turned ON). The remaining good gyro is now used for A, B or M computer SAS control. If the remaining gyro now exceeds tracking limit (third failure) all pitch SAS channels are disengaged (A, B and M PITCH SENSOR lights on function selector and SAS OUT light on annunciator panel are turned on).

If analytical redundancy is not available (flashing BIT FAIL light), when second gyro failure occurs, both pitch SAS channels are disengaged (A, B, and M PITCH SENSOR lights on function selector and SAS OUT lights on annunciator panel are turned on).

Servos

Computer servo monitoring compares left servo LVDT's with right servo LVDT's. If tracking requirement is exceeded, both LVDT's are compared with a servo model to isolate failure (appropriate PITCH SERVO light on function selector and SAS OUT light on annunciator panel are turned on) and that servo channel is disengaged.

Pitch servo LVDT primary and secondary monitors determine if impedence is within acceptable limits. If limits are exceeded, appropriate PITCH SERVO light on function selector and SAS OUT light on annunciator panel are turned on and the failed servo channel is disengaged.

Yaw SAS Logic

Sensors

Yaw sensor failure can be either a rate gyro or lateral accelerometer. Yaw rate gyro failures are indicated by a steady A, B, or M YAW SENSOR light. Lateral accelerometer failures are indicated by a flashing A, B, or M YAW SENSOR light. If the same yaw rate gyro and lateral accelerometer fail (A, B or M sensor), that YAW SENSOR light will be on steady. Computer sensor select capability for yaw rate gyros is the same as for pitch rate gyros, therefore, yaw SAS is available for single or dual gyro failures (if analytical redundancy is operative). Analytical redundancy is not provided for lateral accelerometers. Computer sensor select capability for lateral accelerometers is the same as for pitch rate gyros when analytical redundancy is not available, therefore, yaw SAS is not operative with dual lateral accelerometer failures.

Servos

Computer servo monitoring is the same as pitch SAS.

Yaw servo LVDT primary and secondary monitors are the same as for pitch SAS.

Roll SAS Logic

Sensors

Computer sensor select capability averages the A and B gyro outputs and uses that signal for A and B computer SAS control. If A and B gyro outputs exceed tracking requirement (first failure) both gyros are compared with the analytical redundancy derived roll rate signal. The remaining gyro that is within tolerance is selected and used for A and B computer SAS control (the ROLL SENSOR light on function selector and SAS OUT light on annunciator panel are turned on).

If analytical redundancy is not available (flashing BIT FAIL light), when first gyro failure occurs both roll SAS channels are disengaged (roll sensor light on function selector and SAS OUT light on annunciator panel are turned on). With analytical redundancy a second failure disengages roll SAS (ROLL SENSOR light on function selector and SAS OUT light on annunciator panel remain ON).

Servos

Computer servo monitoring compares left servo LVDT's with right servo LVDT's. If tracking requirement is exceeded both LVDT's are compared with a servo model to isolate failure and appropriate ROLL SERVO light on the function selector and SAS OUT light on annunciator panel are turned on and Roll SAS disengages. To select single channel Roll SAS engage the channel opposite from failed ROLL SERVO light on function selector.

Roll servo LVDT primary and secondary monitors determine if resistance is within acceptable limits. If limits are exceeded appropriate ROLL SERVO light on function selector and SAS OUT light on annunciator panel are turned on and Roll SAS disengages.

A subsequent servo failure in the reengaged good channel will neither disengage the servo nor illuminate the SERVO light in that channel.

AUTOPILOT

DAFICS A and B computers control the pitch and roll autopilot. Both the pitch and roll channels can be engaged, or either one may be operated independently. The autopilot pitch channel may be operated in the basic attitude hold mode, or with one of two other special features (Mach hold or KEAS hold). An automatic pitch trim system is energized when the pitch channel of the autopilot is engaged. A Mach trim system, not part of the autopilot, is enabled when the pitch channel of the autopilot is off. The roll channel may be operated in the basic attitude hold mode, or with one of two other features (heading hold or automatic navigation).

The autopilot pitch channel cannot be engaged unless at least one pitch SAS channel is engaged, and the autopilot roll channel cannot be engaged unless at least one roll SAS channel and one yaw SAS channel is engaged. Elevon control responses due to SAS and/or autopilot inputs are not reflected in control stick movement as they are applied

SAS CONTROLS AND INDICATIONS

CONTROL OR INDICATOR	POSITION OR INDICATION	FUNCTION
CHANNEL ENGAGE SWITCHES. One ON-OFF toggle switch for channel (A & B) of each axis. (On function selector.)	ON (A or B)	Energizes A or B channel of respective axis by energizing the engage solenoids of associated transfer valves.
	Off (A or B)	Disengages respective channel by de-energizing associated transfer valve engage solenoid. SERVO legend(s) corresponding to disengaged channel(s) illuminate.
SENSOR/SERVO Light-Switches, SENSOR half. One SENSOR light for each sensor (A, B & M) of Pitch SAS and Yaw SAS. (On function selector.)	All SENSOR lights Off.	Indicates normal pitch and yaw SAS conditions with all sensor inputs tracking within required limits.
	A, B or M SENSOR light On steady.	In pitch SAS, indicates failure and exclusion of sensor input. SAS remains engaged and operative. In yaw SAS, indicates exclusion of sensor input due to rate gyro failure. SAS remains engaged and operative.
	Push-button Recycling	Pushing any of the three push-buttons for that SAS axis recycles SAS logic to re-input the excluded sensor. SEE NOTE.
	A, B or M YAW SENSOR light Flashing. Δ1	Indicates exclusion of yaw sensor input due to lateral accelerometer failure. SAS remains engaged and operative.
	Push-button Recycling	Pushing any of the three push-buttons for the yaw axis recycles SAS logic to re-input the excluded sensor. SEE NOTE.
	A & M or B & M or A & B SENSOR lights On steady. Δ2	In pitch SAS, indicates failure and exclusion of two sensor inputs (analytical redundancy operating). SAS remains engaged and operative. In yaw SAS, indicates exclusion of two sensor inputs due to rate gyro failures (analytical redundancy operating). SAS remains engaged and operative.
	Push-button Recycling	Pushing any of the three push-buttons for that SAS axis recycles SAS logic to re-input one sensor. Pulse aircraft both directions in that axis, then press any of the three push-buttons again to re-input the other sensor. SEE NOTE.

Δ1 If both the yaw rate gyro and the lateral accelerometer fail in the same yaw sensor, that YAW SENSOR light will be On steady.

Δ2 Assumes at least two DAFICS computers operating.

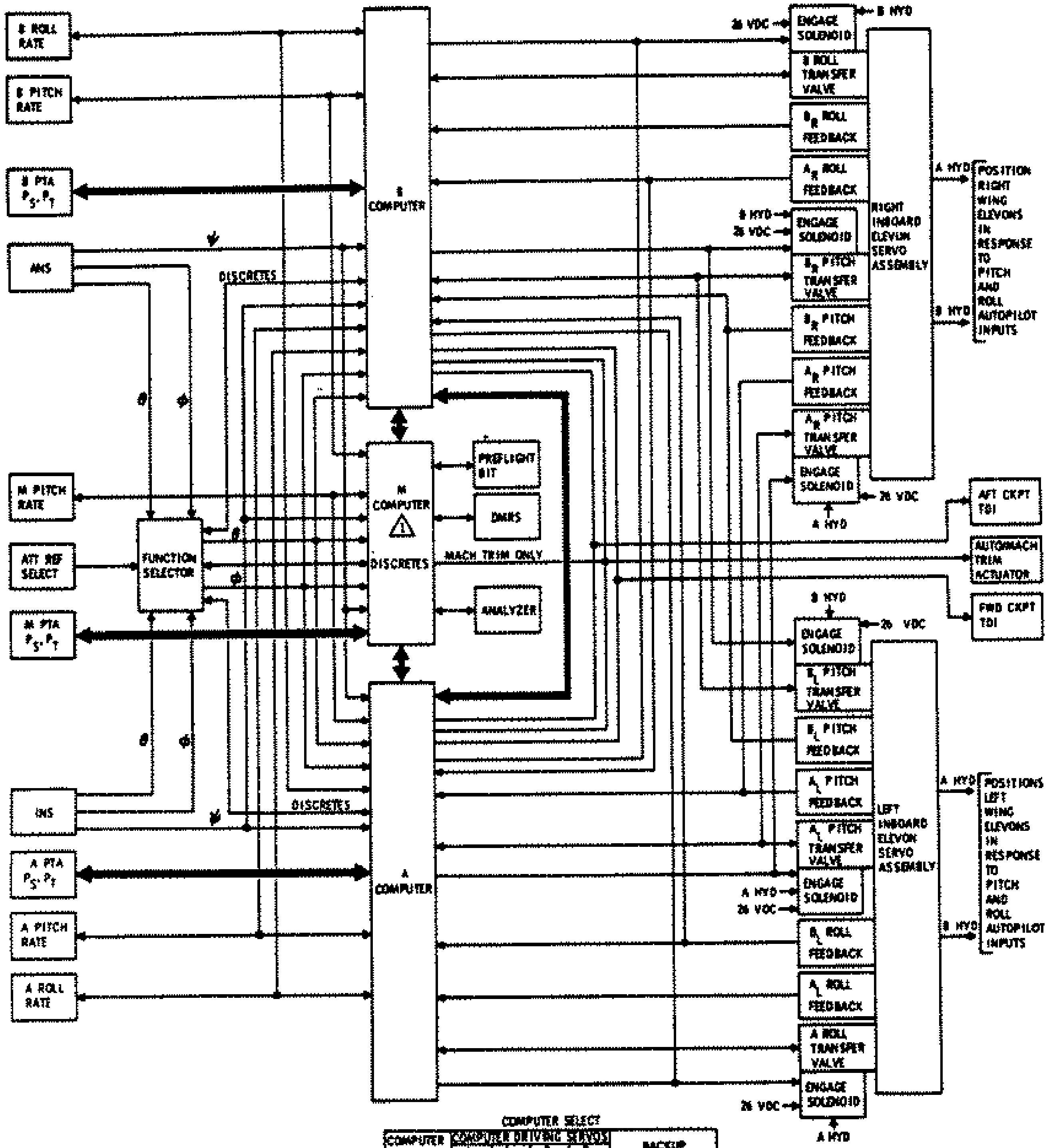
Figure 1-58 (Sheet 1 of 4)

SAS CONTROLS AND INDICATIONS (Cont.)

CONTROL OR INDICATOR	POSITION OR INDICATION	FUNCTION
SENSOR/SERVO (Cont.)	A, B & M SENSOR lights On steady.	In Pitch SAS, indicates failure of all three sensors, or of two sensors and analytical redundancy. Both pitch servos disengage. In yaw SAS, indicates failure of all three sensors due to rate gyro malfunctions or of two rate gyros and analytical redundancy. Both yaw servos disengage.
	Push-button Recycling	Pushing any of the three push-buttons for that SAS axis recycles SAS logic to re-input one sensor. Pulse aircraft both directions in that axis, then press any of the three push-buttons again to re-input the next sensor. Repeat for last sensor. SEE NOTE.
	A, B & M YAW SENSOR lights Flashing	Indicates failure of at least two yaw sensors due to lateral accelerometer malfunctions. Both servos disengage.
	Push-button Recycling	Recycling has no effect.
Roll SENSOR Light. (On function selector between ROLL engage switches.)	Light Off	Indicates normal roll SAS condition with both sensor inputs tracking within required limits.
	Light On	Indicates failure and exclusion of one sensor, both sensors, or one sensor and analytical redundancy. Two failures cause both roll servos to disengage. With failure and exclusion of one sensor, roll SAS remains engaged and operative.
	Engage Switch Recycling	Recycling either SAS ROLL engage switch recycles SAS logic to re-input one sensor. If light remains on, roll aircraft both left and right, then recycle either engage switch to re-input the other sensor. SEE NOTE.

Figure 1-58 (Sheet 2 of 4)

AUTOPILOT AND MACH TRIM BLOCK DIAGRAM



COMPUTER SELECT

COMPUTER CONDITION	COMPUTER DRIVING SERVOS A_L	A_R	B_L	B_R	BACKUP
Normal	A	A	B	B	A for B, B for A
FAIL A	B	B	B	B	NONE
FAIL B	A	A	A	A	NONE

NOTE

⚠ Mach Trim is triple redundant and can also be commanded from the M computer.

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⟷ DIGITAL INTERFACE

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Figure 1-61

Pitch autopilot rate capability may be exceeded when elevon deflection requirements exceed autopilot authority plus autotrim rates. For example, this condition could appear as a loss of pitch trim wheel effectiveness during high pitch rate transonic acceleration maneuvers and supersonic turning decelerations.

Roll Channel

AUTO NAV commands a maximum bank angle of 45°.

The auto-nav mode can be disengaged by using either the AUTO NAV switch, the HEADING HOLD switch, or the CSC button. There should not be any transient roll attitude changes unless the alignment indices show an out-of-synchronization condition.

AUTO NAV will not engage unless the ATT REF SELECT switch is in ANS.

HEADING HOLD cannot be engaged if INS is the selected attitude reference and the INS is operating in the ATTITUDE mode.

Autopilot Operation

To engage autopilot in attitude hold:

1. Engage SAS
2. Manually trim the aircraft.
3. Pitch and/or roll engage switches - ON.

For minor changes to attitude, use pitch or roll trim wheels.

For major changes to engaged attitude:

4. CSC button - Hold depressed until aircraft is stabilized in new attitude.
5. CSC button - Release.

   Autopilot will reengage in existing pitch and roll attitude.

To engage Mach hold or KEAS hold:

6. With the pitch autopilot engaged, stabilize the aircraft for a few seconds at the desired Mach or KEAS before engaging either mode.
7. Mach hold/KEAS hold switch - As desired.

To engage Heading hold:

8. Heading hold switch - ON.

To engage Auto-Nav:

9. Attitude reference select switch - ANS.

   If the switch is in INS, the autopilot will disengage when the switch is moved to ANS.

10. Auto-nav switch - ON.

    Turns should be anticipated while in AUTO NAV and manual control stick inputs applied, if necessary, to avoid excessive roll or bank angles.

To disengage autopilot:

11. Alignment indicators - Monitor.

    Continuous deflections of indicator needle in same direction indicates that a transient will probably occur on disengagement.

To disengage all but attitude-hold modes:

12. Depress CSC button.

To disengage entire autopilot:

13. Depress control stick trigger switch.

To disengage respective channels:

14. Move corresponding pitch and/or roll axis engage switches to OFF.

To extinguish the annunciator panel AUTO PILOT OFF light:

15. Depress A/P OFF light.

MACH TRIM SYSTEM

Speed stability augmentation is provided by the Mach trim system while accelerating or decelerating in the 0.2 to 1.5 Mach range. Mach trim is scheduled by the DAFICS computers to restore conventional stick forces and trim requirements by applications of pitch trim through the slow-speed motor of the pitch trim actuator. No control switch is provided, although Mach trim is enabled with pitch autopilot OFF, and is disabled with pitch autopilot ON. All manual and auto trim (including Mach trim) can be disabled by moving the TRIM POWER switch on the annunciator panel to OFF.

Power is supplied to the Mach trim system by the essential dc bus through the A/P MACH TR A, A/P MACH TR B, and CMPTR M circuit breakers on the pilot's left console and the emergency ac bus through the A/P MACH TR circuit breaker in the aft cockpit.

NOTE

> Mach trim redundancy is reduced when one of the two A/P MACH TR dc circuit breakers is opened and redundancy is lost when both A/P MACH TR circuit breakers are opened. To disable the Mach trim system, both A and B channel circuit breakers and CMPTR M circuit breaker must be opened <u>or</u> the AP MACH TR ac circuit breaker in the aft cockpit must be opened.

AUTOMATIC PITCH WARNING AND HIGH ANGLE OF ATTACK WARNING SYSTEMS

An Automatic Pitch Warning (APW) system provides control stick pusher and stick shaker features. APW stick shaker operation warns that a potentially unsafe condition of angle of attack plus pitch rate has been reached. For those situations where both warnings are provided, the boundary for stick pusher operation is slightly more extreme than the boundary for the stick shaker. In addition to the APW System, a separate High Angle of Attack (High Alpha) warning system can also operate the stick shaker, but not the pusher.

The stick shaker can operate throughout the flight regime without restriction due to Mach number or stick or gear position. APW system inputs to the stick shaker can be disabled by setting the APW switch to OFF. The High Alpha shaker warning system starts the shaker anytime its Mach number vs angle of attack schedule is exceeded regardless of the APW switch position. Proximity to the APW stick shaker boundary is displayed by the glide slope indicator needle on the Attitude Director Indicator when the DISPLAY MODE SEL switch is not in ILS or ILS APPROACH. This display is independent of APW switch position. The SHAKER warning light near the pilot's glareshield illuminates when the stick shaker operates. An APW caution light on the pilot's annunciator panel illuminates if the APW switch is turned off, if the APW fails self check, or if the 2 PTA CHAN OUT light illuminates (the 2 PTA CHAN OUT light remains on until reset by maintenance, but the APW light may subsequently extinguish).

The stick pusher can only operate while subsonic with two of the three landing gear doors locked in the up position. Two of the three DAFICS computers must be operating. The pusher warning can be disabled manually by the APW switch, by depressing the control stick trigger switch, or by positioning the control stick more than 2.5 degrees (1.5 inches) forward of neutral.

The APW and High Alpha Warning systems are required because there is no buffet or other natural warning of approach to unstable angle of attack conditions. The pitch boundary indication on the ADI can provide the pilot with a display of angle of attack plus pitch rate relative to the APW stick shaker boundary. If the boundary is broached, activation of the stick shaker warns the pilot that he has reached a potentially dangerous angle of attack or that pitch rate is such that a potentially dangerous condition will be reached unless immediate action is taken. Refer to Figure 1-63 for the shaker boundary schedules. The shaker warning is a 30 cps vibration of the control stick.

The stick pusher starts control stick correction in pitch. An abrupt push force of approximately 30 pounds is applied when the pusher boundary has been reached until the stick is 2.5° (1.5") forward of neutral. The push force can be overcome by the pilot, if desired. The pusher function can be disabled by the control stick trigger switch or by moving the APW switch to OFF or SHAKER ONLY.

The boundaries for operation of the APW stick shaker and pusher warnings are functions of angle of attack and pitch rate summation vs Mach number. The warning schedule reflects the changing susceptibility to pitch-up at subsonic and supersonic speeds. The boundary for operation of the High Alpha system shaker warning is fixed at angles of attack of 14° for low speed and 8° for high speed operation, with the switchover at Mach 1.4 or 1.55 (decelerating or accelerating, respectively). Observance of the APW and High Alpha system warning boundaries provides adequate margin to avoid pitch-up at all speeds.

WARNING

> Avoidance of the stick shaker or pusher/shaker warning boundaries does not, by itself, assure that load factor or angle of attack limits will be observed.

Reduce angle of attack and adjust attitude nose-down if a high angle of attack warning occurs or if an alpha limit is approached.

WARNING

> When subsonic, if an APW system or high angle of attack warning occurs, or if angle of attack and airspeed are not within limits, make angle of attack and speed corrections before adjusting the throttles. These actions alone may clear engine stall conditions, and are mandatory to avoid pitch-up, if at high angle of attack and/or low airspeed.

Essential dc power for the stick shaker is obtained either through the APW or STALL WARN circuit breaker on the pilot's left console. The power routing depends on which system activates the shaker. This APW circuit breaker also supplies power for the stick pusher solenoid which controls operating power from the "A" hydraulic system. The A and/or B computer must be operative for either the APW or High Alpha Warning system to operate. The High Alpha Warning system also requires essential dc power from the BRK & SKID circuit breaker on the pilot's left console. This power causes the main gear scissors switches and relays to close when aircraft weight is removed from the main gear. With the right main gear relay open, no alpha signals are received by the High Alpha Warning system.

APW AND HIGH ALPHA WARNING SYSTEM CONTROLS AND INDICATORS

There is no single control of all APW and High Alpha Warning system functions. The ADI pitch boundary indication is controlled by the Display Mode Select switch. APW control stick warning is controlled by the APW PUSHER/SHAKER switch. The APW stick shaker boundary changes and the stick pusher is cut out when two of the three landing gear doors are not locked in the up position. Each gear-up signal is routed to one DAFICS computer so if two DAFICS computers are inoperative the APW reverts to gear-not-up logic. The angle of attack input to the APW is inhibited with weight on the left main gear. The High Alpha Warning System shaker is inhibited with weight on the right main gear. The stick pusher can be cut out manually by the APW switch, the control stick trigger switch and by stick positioning. APW system status is indicated by the ADI glide slope needle, SHAKER warning light, stick shaker and pusher, APW caution light, and DAFICS A and B CMPTR OUT caution lights. No controls are provided for the High Alpha Warning system. The High Alpha Warning system control stick shaker function is always operative while in flight.

APW Pusher/Shaker Switch

A three-position APW pusher/shaker switch is provided on the left side of the pilot's instrument panel. The APW system stick shaker and stick pusher are inoperative when OFF (center) is selected; however, the High Alpha Warning system remains operative and the pitch boundary indication to the ADI continues to operate. See Figure 1-66.

The APW stick shaker is operative when the SHAKER ONLY (down) position is selected. The APW stick shaker and stick pusher are operative when the PUSHER/SHAKER (up) position is set. The APW caution light illuminates when the APW switch is OFF. APW angle of attack and pitch rate schedules are shown by Figure 1-63, Sheet 1 and 2.

With a normally operating APW system, do not position the APW pusher/shaker switch to OFF.

Shaker Warning Light

A red SHAKER warning light is located near the apex of the glareshield in the pilot's cockpit. It illuminates whenever dc power is supplied to operate the stick shaker motor by either the High Alpha Warning or APW system. The light can be tested by pressing the pilot's IND & LT TEST push-button switch. Essential dc power for the light is obtained through the APW or STALL WARN circuit breaker, whichever system is controlling the stick shaker motor.

APW Caution Light

An amber APW caution light on the pilot's annunciator panel illuminates when the APW system stick warning switch is OFF. With PUSHER/SHAKER or SHAKER ONLY selected, the light illuminates when both A and B computers are operative and the APW outputs from the A computer do not match the B computer. The APW caution light also illuminates (although it may subsequently extinguish) when the 2 PTA CHAN OUT annunciator light illuminates. If both A and B CMPTR OUT annunciator lights illuminate, both the APW system and the High Alpha Warning System are inoperative, but the APW caution light will not illuminate.

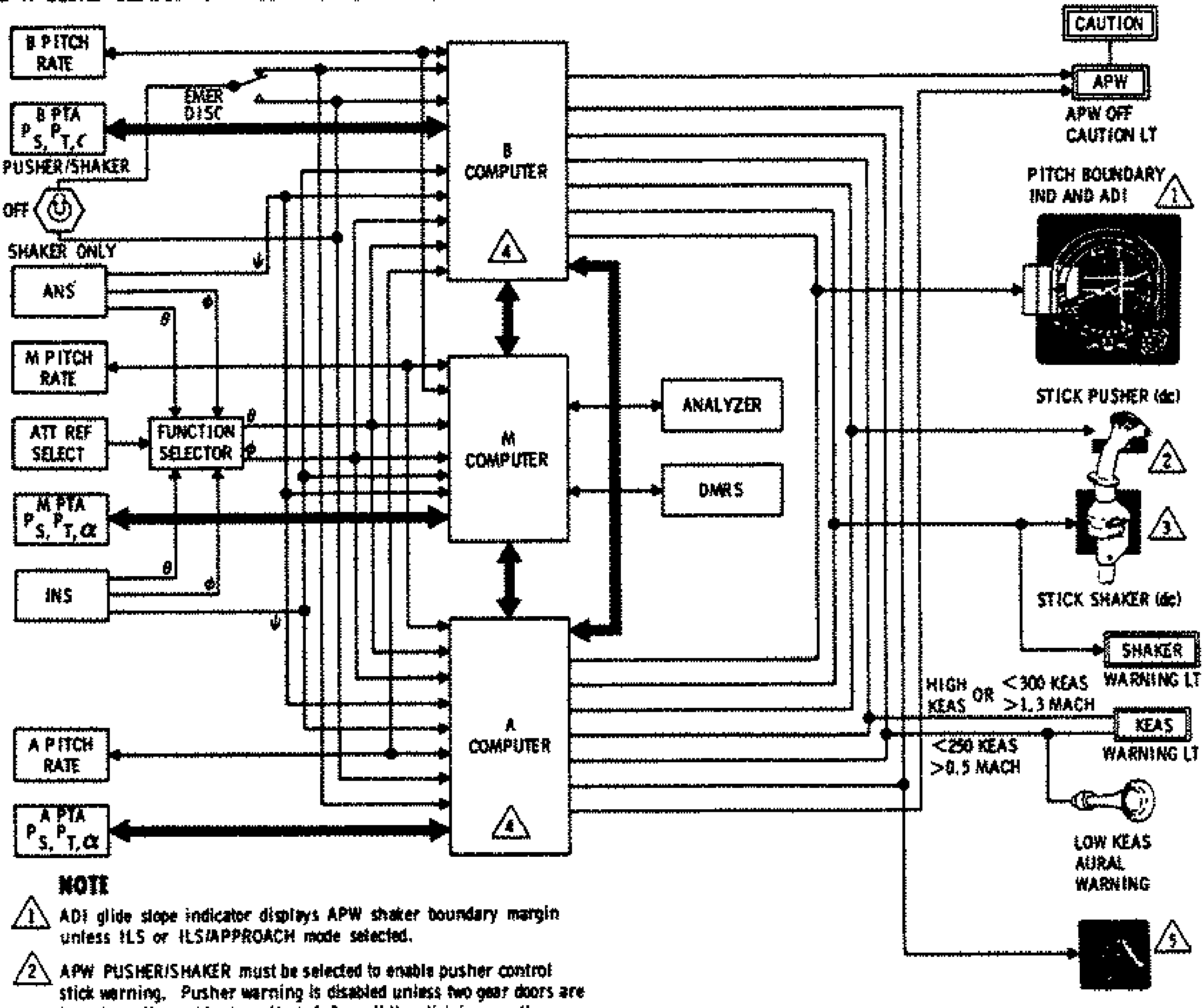
NOTE

The APW stick shaker and pusher are disabled when the APW annunciator light is on.

APW AND HIGH ALPHA WARNING SYSTEM OPERATION

APW computations are performed by the A and B DAFICS computers. Both computers use the same triple redundant digital input values for angle of attack and pitch rate as shown by Figures 1-62, and 1-64. If both the A and B computers are operating, both computers activate the shaker and pusher when the appropriate boundaries for the existing Mach number are exceeded. Pusher warning can not occur unless the APW shaker warning is also on. If the digital APW outputs from the A computer do not exactly match the digital APW outputs from the B computer, the APW annunciator light illuminates and both the pusher and shaker are deactivated. If either the A or B CMPTR OUT light illuminates, the remaining computer continues all APW functions including pusher, shaker, and ADI pitch boundary indication, but redundancy is lost.

APW AND HIGH α WARNING SYSTEM



NOTE

1. ADI glide slope indicator displays APW shaker boundary margin unless ILS or ILS/APPROACH mode selected.
2. APW PUSHER/SHAKER must be selected to enable pusher control stick warning. Pusher warning is disabled unless two gear doors are locked up, if speed is above Mach 1.0, or if the stick is more than 2.5° (1.5") forward of neutral. Also, stick pusher is inoperative if APW light is illuminated or any two computers have failed. Stick trigger disables pusher. Operation of pusher or stick trigger disengages autopilot.
3. Stick shaker operates if α exceeds either 8° or 14° (depending on status of Mach switch). Unless APW OFF selected, shaker operates if (α + q) APW shaker boundary is exceeded. With PUSHER/SHAKER selected, pusher operates when (α + q) APW pusher boundary is exceeded (unless pusher is disabled or its operation inhibited).
4. Mach switch is closed above Mach 1.55 with speed increasing and is open below Mach 1.4 with speed decreasing.
5. With DAFICS power on, alpha indicator reads 0 degrees below 101 KEAS and actual $\Delta P \alpha$ above 101 KEAS
6. Automatic pitch warning and stall warning is available if A or B or both computers are up and ready. M computer provides redundancy for air data and sensor select.
7. APW PUSHER/SHAKER operation is inhibited with weight on the left main gear. High α SHAKER operation is inhibited with weight on the right main gear.

SENSOR (RATE) SELECT

SENSOR CONDITION	SENSOR USED COMPUTER A	COMPUTER B	COMPUTER M	SENSOR OUTPUT SELECTED
Normal	A	B	M	Median of A, B, M
Fail A	M	B	M	Average of B, M"
Fail B	A	A	M	Average of A, M
Fail M	A	B	B	Average of A, B
WITH ANALYTICAL REDUNDANCY				
Fail A, B	M	-	M	M
Fail B, M	A	A	-	A
Fail A, M	-	B	B	B
WITHOUT ANALYTICAL REDUNDANCY				
Fail A, B	-	-	-	None
Fail B, M	-	-	-	None
Fail A, M	-	-	-	None

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⟷ DIGITAL INTERFACE

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Figure 1-62

If both (A and M) CMPTR OUT or (B and M) CMPTR OUT annunciator caution lights illuminate, the APW pusher function is deactivated and the shaker boundary changes to gear down values.

If the A and B CMPTR OUT annunciator caution lights illuminate, both the APW system and High Alpha Warning system are inoperative, but the APW annunciator caution light will not illuminate.

The DAFICS A and B computers operate the high angle of attack warning system stick shaker function independently from the APW System. When activated by the right main gear scissors switch and relay, DAFICS angle of attack and Mach number inputs to the A and B computers are used to determine when the High Alpha system stick shaker is reached. The High Alpha Warning system shaker is inoperative with weight on the right main gear.

APW Pitch Boundary Indication (PBI)

While in flight, the pitch boundary indication (PBI) on the Attitude Director Indicator shows proximity to the APW stick shaker boundary (unless ILS or ILS APPROACH Display Mode is selected). A three-quarters scale deflection of the glide slope displacement indicator (second dot from the top of the scale) indicates that the APW shaker boundary has been reached. Refer to Figures 1-62, 1-64, and 1-66. The PBI pitch rate signal is obtained as shown on the Sensor (rate) Select chart of Figure 1-63 whether the pitch SAS is engaged or not. The PBI displays the angle of attack plus pitch rate continuously, regardless of APW switch positioning.

A slightly-below-center indication corresponds to optimum supersonic cruise (5° to 6° angle of attack with zero pitch rate). Three-quarters scale deflection will be approached at supersonic maximum cruise altitude and during intermediate altitude turns.

At cruise speeds near Mach 0.9 three-quarters scale deflection represents 11° to 12° angle of attack (with zero pitch rate). At the angle of attack for optimum cruise the PBI is slightly below center scale.

NOTE

> Since alpha + pitch rate is the function displayed, the PBI does not reflect angle of attack unless pitch rate is zero.

It is possible to cause the APW stick shaker to operate momentarily while taxiing. When on the ground, the PBI is normally deflected to the position of the lowest dot marking if the ANS, INS, or TAC/ADF display mode is selected. No angle of attack signal is received by the APW system while the left main gear scissors switch is open, and there would be no pitch rate while in a static condition. However, needle fluctuations will occur if pitch rate gyros sense fuselage motion in the pitch axis. It is possible, but unlikely, that while taxiing, pitch rates exceeding 14.6 degrees per second could be experienced which would cause the APW stick shaker to operate momentarily. The high angle of attack shaker warning can not occur, as the right main gear scissors switch relay opens this warning circuit while on the ground.

NOTE

> Positioning of the PBI at the bottom index while in one "g" flight could indicate no angle of attack input to the APW system. This could result from failure of the left main gear scissors switch.

High Alpha Stick Shaker Boundaries

In flight, the High Alpha Warning system powers the stick shaker when its angle of attack boundary is reached —<u>regardless of pitch rate and the pitch boundary indication.</u> The angle of attack settings for this system are 14° when below Mach 1.4 (or when below Mach 1.55 if accelerating) and 8° when above

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STICK WARNING SCHEDULES

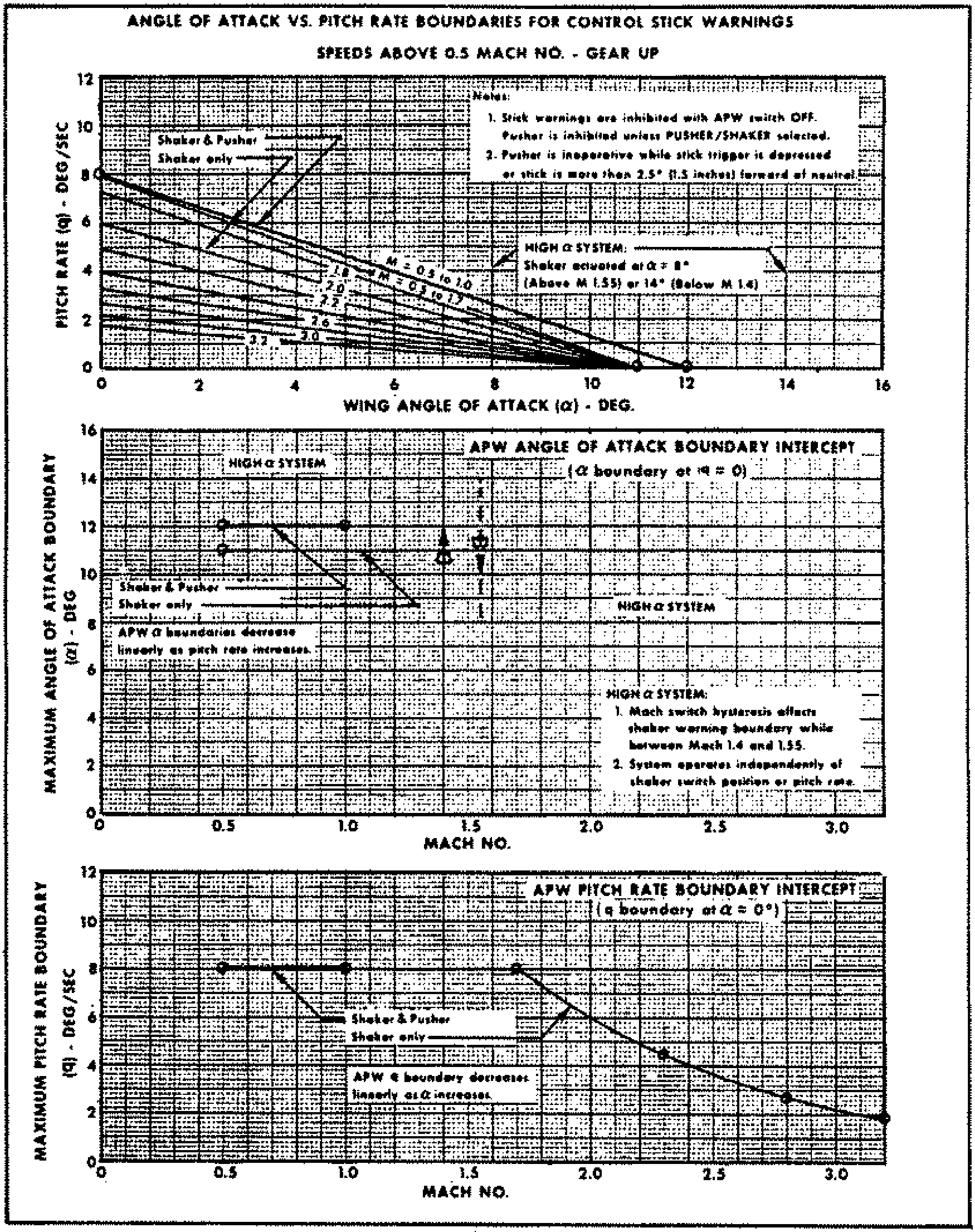


Figure 1-63 (Sheet 1 of 2)

STICK WARNING SCHEDULES

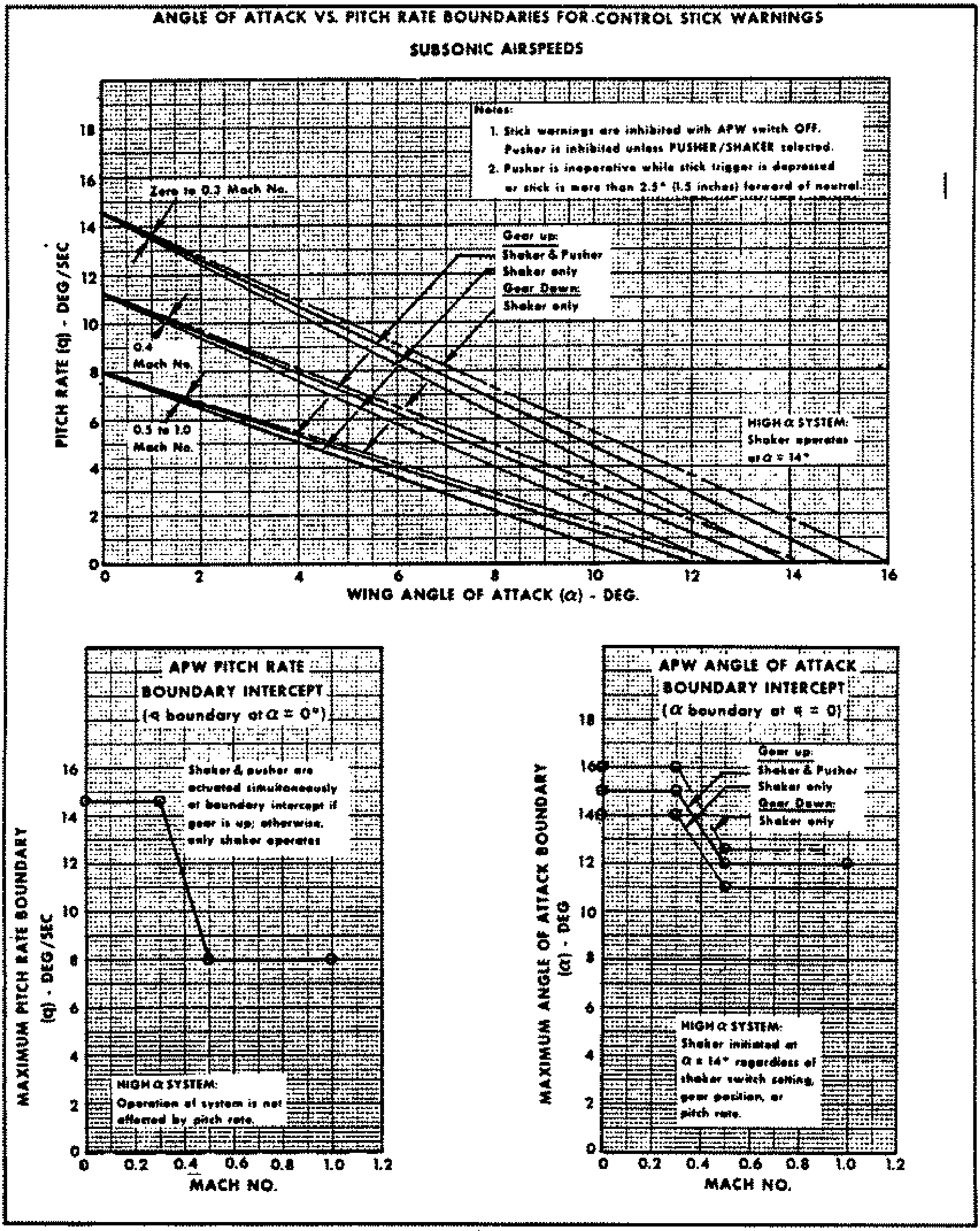


Figure 1-63 (Sheet 2 of 2)

Mach 1.55 (or when above Mach 1.4 when decelerating). See Figure 1-63.

APW Control Stick Warning Schedules

Refer to Figure 1-63, Sheet 1 and 2. Note that Sheet 1 only shows control stick warning schedules for gear up operation at speeds above Mach 0.5. These schedules apply to the major portion of flight. Schedules are included for lower speeds, typical for conditions before landing or after takeoff.

Notice that the APW alpha + pitch rate (q) boundary schedules, shown for the stick shaker and stick pusher, represent 100% of the APW boundary condition when the opposite factor (pitch rate or angle of attack) equals zero. For example: at Mach 3.2, an APW system angle of attack signal of 11° or a pitch rate signal of 1.8 degrees per second will cause the stick shaker to operate when the opposite pitch rate or alpha factor is zero. (See Note). Similarly, a 5.5° angle of attack obtained simultaneously with a 0.8 degree per second pitch rate will cause the shaker to operate. This is illustrated by the Mach 3.2 line on the angle of attack vs pitch rate boundary chart. Each of the 5.5° and 0.8°/sec angle of attack and pitch rate factors is 50% of its boundary intercept for Mach 3.2 speed. Any similar combination for the same speed would start the stick shaker.

NOTE

> Since the high angle of attack warning system setting is 8° in the high speed region, any angle of attack of 8° or more would start the stick shaker.

The stick pusher boundary intercept for gear up operation is 1° above the shaker boundary at zero pitch rate, as shown by Figure 1-63, Sheet 2 of 2. The difference decreases to zero at the pitch rate boundary intercept (alpha = 0°). The pusher is inhibited unless two gear are up, to eliminate pusher activation close to the ground immediately after takeoff or before landing. The shaker warning boundary is automatically shifted to a higher alpha schedule when two gear are not up, to minimize nuisance operation.

PITOT-STATIC SYSTEMS

A dual pitot-static and alpha/beta system supplies the total and static pressure necessary to operate the basic flight instruments and DAFICS. The pitot-static and alpha/beta probe is mounted on the nose of the aircraft. Pressure entering the tip of the probe is divided internally to produce two separate total pressure sources. Eight static ports on the probe furnish two separate static sources. One pitot and one static source are referred to as system 1 and the other two sources are referred to as system 2. System 1 supplies pressure to the pilot's altimeter, vertical speed indicator, and Mach-airspeed indicator. System 2 supplies pressure to the 3 channels of the pressure transducer assembly (PTA) which supplies digitized pressure information to the DAFICS computers. An alpha/beta probe provides angle of attack and sideslip pressure signals to the PTA. (See Figure 1-64.)

Heating elements for the probes are controlled by the pitot heat switch. Although there is no time limit for operation of the heater system while on the ground, the life span of the pitot heater element is lowered if pitot heat is left on unnecessarily.

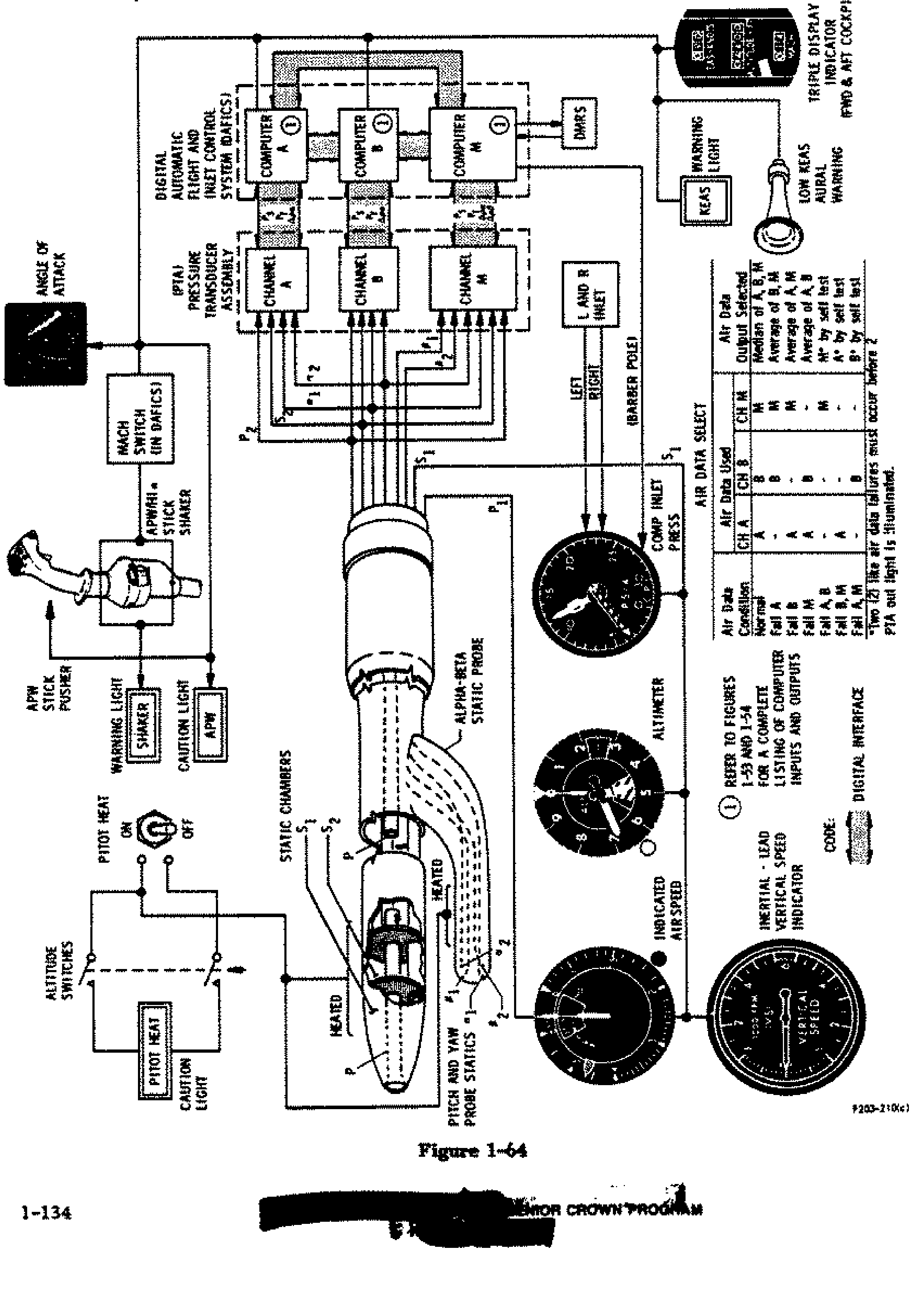
Pitot Heat Switch and Indicator Light

A two-position PITOT HEAT switch is located on the pilot's annunciator panel. In ON, the essential ac bus powers heating elements of the pitot-static and alpha-beta probes through the PITOT HTR circuit breaker on the pilot's right console.

The circuit also incorporates a PITOT HEAT annunciator caution light, and two altitude pressure switches. The light illuminates when the pitot heat switch is not in the correct position for the aircraft altitude. With the pitot heat switch ON, the light is on when above 63,000 ± 4000 feet. With the pitot heat switch OFF, the light illuminates below 40,000 ± 2000 feet, and during climb extinguishes between 40,000 and 50,000 feet.

SECTION I

PITOT-STATIC, PTA, INLET, SAS, AUTOPILOT, & APW INTERFACE



AIR DATA SELECT

Air Data Condition	Air Data Used CH A	CH B	CH M	Air Data Output Selected
Normal	A	B	M	Median of A, B, M
Fail A	-	B	M	Average of B, M
Fail B	A	-	M	Average of A, M
Fail M	A	B	-	Average of A, B
Fail A, B	-	-	M	M* by self test
Fail B, M	A	-	-	A* by self test
Fail A, M	-	B	-	B* by self test

*Two (2) like air data failures must occur before 2 PTA out light is illuminated.

Figure 1-64

PRESSURE TRANSDUCER ASSEMBLY (PTA)

The PTA changes pressures (P_s, P_T, $\Delta P\alpha$, $\Delta P\beta$) to electrical signals for DAFICS. The PTA has three electrically independent channels (A, B, M) which produce digitized P_s, P_T, and $\Delta P\alpha$ data. In addition the M channel transduces $\Delta P\beta$ data. The PTA is located in the left forward chine bay, aft of the nose bulkhead and is cooled by cockpit exit air.

CAUTION

> Loss of cockpit pressurization or cockpit air off at supersonic cruise will cause overheat and subsequent failure of the PTAs if a descent is not begun within 15 minutes.

Each PTA Channel (A, B, and M) only sends data to its corresponding (A, B, and M) DAFICS computer. The DAFICS computers share PTA data and select the best air data for DAFICS computations. (See Air Data Select chart, Figure 1-64) All operating DAFICS computers use the same selected PTA data. If a DAFICS computer is inoperative, the corresponding PTA channel is also inoperative.

The A, B, and M channels of the PTA are powered by the emergency ac bus through the three A, B, and M COMPUTER circuit breakers for the respective computer located in the aft cockpit. With two circuit breakers opened to a computer, the corresponding PTA channel is disabled.

FLIGHT AND NAVIGATION INSTRUMENTS

TRIPLE DISPLAY INDICATOR (TDI)

A TDI on the instrument panel in each cockpit provides digital displays of airspeed (KEAS), pressure altitude, and Mach number as computed by DAFICS.

The altitude indication range of each TDI is 0 to 99,950 feet. At 100,000 feet, the first digit is dropped, indicating 09,950 feet at 109,950 feet (the maximum limit of the DAFICS signal to the instruments).

The TDI Mach number indication range is 0 to 3.99. The minimum indication at static conditions normally varies from 0.11 to 0.2 Mach number and the maximum limit of the DAFICS signal to the instruments is Mach 3.5.

The TDI knots equivalent airspeed (KEAS) indication range is 0 to 599 KEAS. The minimum indication normally varies from 25 to 110 KEAS. The maximum limit of the DAFICS signal to the instruments is 560 KEAS.

OFF flags appear on the faces of the TDI instruments and TDI values are unusable if the DAFICS A and B computers fail. OFF flags appear on both TDI instruments if the emergency ac bus AP MACH TR circuit breaker in the aft cockpit is open, but indications remain valid. The OFF flags will not necessarily appear in conjunction with a 2 PTA CHAN OUT annunciator caution light.

If the OFF flags should appear with the 2 PTA CHAN OUT caution, it indicates that none of the available PTA channels is passing self check. Refer to 2 PTA Channels Out Procedures, Section III.

Power for both instruments is from the emergency ac bus through the TDI circuit breakers. The forward cockpit TDI circuit breaker provides power to the forward cockpit TDI; the aft cockpit TDI circuit breaker provides power to the aft cockpit TDI, which in turn provides TAS signals to the Pilot's & RSO's Map Projector - Automatic Map Rate.

AIRSPEED - MACH METER

A combination airspeed and Mach meter, operating directly from pitot-static pressure, is installed on the pilot's instrument panel. Airspeed and Mach values shown are indicated values as opposed to equivalent airspeed and true Mach displayed on the TDI. A limit airspeed needle varies with altitude to show the KIAS limit corresponding to a preset KEAS vs altitude schedule. See Airspeed - Mach Meter, Section V.

NOTE

At low altitudes and airspeeds the high end of the Mach scale will show in the window.

ALTIMETER

A pressure altimeter is installed on the pilot's instrument panel. The instrument has 1000-foot and 100-foot pointers, and a 10,000 foot pointer (with a triangular marker at its extremity) which extends to the edge of the dial. The center of the instrument has a cutout through which yellow and black warning stripes appear when aircraft altitude is less than 16,000 feet. The barometric pressure scale appears in a cutout at the right side and barometric pressure is set by a knob located at the lower left side of the instrument.

NOTE

The static altimeter and the IVSI exhibit appreciably more lag while supersonic than subsonic.

Altimeter Correction Card

An altimeter altitude correction card, located on the pilot's right sunshade, shows the indicated corresponding pressure altitudes to fly for a corresponding true pressure altitude. The difference between the columns is altimeter instrument error only.

INERTIAL-LEAD VERTICAL SPEED INDICATOR (IVSI)

An IVSI, installed on the pilot's instrument panel, shows the rate of change of altitude in feet per minute up to 6000 feet per minute. See Figure 1-64. It is operated by static pressure changes sensed by the system 1 static ports. It operates like conventional vertical velocity instruments except that an internal mechanism, sensitive to load factors between 0.7 and 1.4 g's, momentarily alters static pressure within the IVSI under changing load factors. The pressure change causes a corresponding change in rate-of-climb or rate-of-descent indication when the load factor is increasing or decreasing, respectively. The indication change will usually precede aircraft altitude changes. Changing static pressure conditions, such as during climb or descent, cause the vertical speed indications to continue registering in the appropriate direction as the effect of the load factor pulse diminishes. The instrument appears to be "lag free" when subsonic. At high altitudes, reduced air density decreases instrument response and accuracy. Indications may be less than 1/3 of actual conditions. Response to acceleration is retained, but to a lesser degree.

Below approximately 40,000 feet, IVSI indications respond quickly to pitch control. In a level turn the IVSI will show a small rate of climb during roll-in and rate of descent during roll-out due to load factor changes. Expect a change of approximately 50 fpm at cruise.

ANGLE OF ATTACK (AOA) INDICATOR

An angle of attack indicator is mounted on the pilot's instrument panel. The dial displays positive and negative angles of attack in one-degree increments through a range from -5° to $+22^{\circ}$.

The instrument responds to synchro signals from DAFICS which uses differential pressures sensed by the alpha ports of the alpha/beta probe ($\Delta P\alpha$) related to total pressure (P_T). See Figure 1-64. A and/or B computer must be operating to provide angle of attack.

An OFF flag is displayed on the face of the indicator if power to the instrument is interrupted.

The AOA indicator reads 0° on the ground and reads aircraft actual AOA at 100 KEAS or higher. In level flight, pitch angle is 1.2° greater than the wing angle of attack because of the negative angle of incidence of the wing. See Angle of Attack, Section VI.

ATTITUDE-DIRECTOR INDICATOR (ADI)

An ADI, located on the pilot's instrument panel, combines an attitude indicator, turn-and-slip indicator, and a flight director with ILS glide slope presentation. Pitch and roll signals from the ANS or INS systems are routed to the instrument through the ATT REF SELECT switch. The attitude sphere allows presentation of pitch and roll through 360 degrees. The sphere moves behind a small aircraft symbol fixed at the center of the instrument face. A trim knob allows manual positioning of the sphere in pitch.

The sphere is marked with a horizon line, small dots for 5-degree increments, short lines for 10-degree increments, and numeral markers for each 30-degree increment; large dots indicate the poles. Indices are provided for bank angles of 10, 20, 30, 45, 60 and 90 degrees.

The turn-and-slip indicator is mounted at the bottom of the ADI. A deflection of one-needle-width indicates a 4-minute, 360-degree, standard rate turn. The rate-of-turn transmitter receives power from the essential dc bus through the TURN GYRO circuit breaker on the pilot's left console.

Bank and pitch steering command bars are superimposed on the ADI. The bank steering bar shows bank required to position the aircraft on a desired heading or course and will center when (1) on course with wings level, (2) flight attitude is correct for return to course, or (3) bank is correct for rollout on-course. When in AUTO NAV, the maximum bank angle command is 45°. The maximum bank command is 35° in INS, TACAN ADF and ILS modes. The maximum bank and pitch commands are 15° and 10°, respectively, in ILS APPROACH mode.

The pitch steering bar indicates pitch attitude change required to intercept the ILS glide slope. The bar centers when (1) on glide slope with proper pitch angle, (2) pitch angle is correct for return to glide slope, or (3) pitch angle is correct for leveling-out on glide slope.

The steering bars indicate corrective action required and not direction or displacement from a desired course or glide slope. The vertical (bank) and horizontal (pitch) steering bars will be centered when power is off. The pitch steering bar is stowed at the bottom of the instrument when the DISPLAY MODE SEL switch is in INS, or TACAN ADF. The bank steering bar receives command signals in all display modes.

The pitch steering bar indicates rate of climb or descent when the DISPLAY MODE SEL switch is in ANS. See Figure 1-66. The bar indicates zero vertical velocity when it is aligned with the small airplane symbol. Full scale deflection of the bar to the top or bottom dot on the glide slope displacement scale represents a vertical velocity of 3484 fpm or more. This provides a scale sensitivity of approximately 1000 fpm for each quarter-inch displacement. The minimum vertical speed which can be sensed by the ANS and displayed is approximately 55 fpm. The displacement of the steering bar is opposite in direction to the vertical velocity indicator, to indicate the direction of pitch attitude change required to offset the climb or descent. The pitch steering bar vertical velocity indication can be used to maintain precise control of altitude at high Mach.

A warning flag comes into view at the top of the ADI if the instrument receives unreliable steering information. If INS attitude reference is selected and the INS is in the attitude mode, the flag indicates that heading and INS steering information are not reliable. If ANS attitude reference is selected and the DISPLAY MODE SEL switch is in ANS, the flag indicates the ANS is not ready and attitude, true heading, and steering information are not reliable. The glide slope pointer at the left center of the instrument shows aircraft position above or below the ILS glide slope when ILS or ILS APPROACH display mode is selected and valid ILS signals are received. The glide slope warning flag appears if there is insufficient ILS signal strength. The glide slope pointer also is the pitch boundary indicator when ILS or ILS APPROACH display mode is not selected. Refer to Figure 1-66 and to the APW and High Angle of Attack

Warning Systems description, this section. A power failure warning flag appears in the lower left of the instrument when ac power is interrupted, ATT REF SELECT switch is in INS and the INS ready signal is not present, or the ATT REF SELECT switch is in ANS and the nav-ready signal is not present.

Display Mode Select Switch

The DISPLAY MODE SEL switch on the pilot's instrument panel has five positions: ANS, INS, TACAN ADF, ILS, and ILS APPROACH. The switch controls display inputs to components of the HSI and ADI. See Figure 1-66.

Attitude Reference Select Switch

The ATT REF SELECT switch on the pilot's instrument panel has two positions: ANS (up) and INS (down). It selects which system supplies pitch and roll signals to the attitude director indicator, and pitch, roll, and heading signals to the autopilot. Autopilot AUTO NAV is only operative when the ATT REF SELECT switch is in ANS. Changing the ATT REF SELECT switch disengages the autopilot. Autopilot HEADING HOLD will not engage if INS attitude reference is selected and the INS is in attitude mode.

WARNING

INS reference should be selected in both cockpits if ANS nav-not-ready warnings (pilot's ANS REF and RSO's ANS FAIL caution lights) appear. Otherwise, unreliable attitude information may be displayed.

PERIPHERAL VISION DISPLAY (PVD)

The PVD is a twilight and night attitude orientation device that projects on the pilot's instrument panel, a laser-generated, thinly focused red line parallel to the horizon. A mask in the PVD projector prevents display of the line on the ADI. Like an attitude indicator, the PVD line remains fixed in inertial space. The PVD is not intended to be consciously included in the pilot's instrument crosscheck. Instead, the laser line is perceived indirectly through peripheral vision and subconsiously supports spatial orientation just as a visible outside horizon supports orientation.

The PVD derives pitch and roll attitude from the attitude source (ANS or INS) selected by the pilot's ATT REF SELECT switch. Roll movement of the line corresponds with changes in roll attitude and has no limit. Pitch movement of the line corresponds with changes in pitch attitude. Pitch movement of the line stops when a display boundary at the edge of the instrument panel is reached. The pitch angle at which the line will reach a boundary depends on the position of the pitch adjust switch and the pitch scale switch. The line flashes at 4 Hz to warn of a potentially hazardous flight condition if pitch angle exceeds 35 degrees nose up or 15 degrees nose down.

Twelve regularly spaced variations in line intensity create subtle segments in the line that remain fixed in heading relative to inertial space. Movement of the segments across the instrument panel allows a sense of heading change during turns. Because rate of turn decreases at high speed, the effect is most noticable at low speed. DAFICS M computer provides rate of turn inputs to the PVD processor.

The PVD processor, located in the R bay, continually performs built-in-test. If an internal processing fault is detected, the laser line will not be generated. If DAFICS rate of turn input is lost or a rate of turn fault is detected, the line segments will not move (zero turn rate), but pitch and roll will still be displayed. To inhibit the PVD when inputs from the selected attitude reference are unreliable, the laser line is not generated if a DAFICS analytical redundancy (ANR) failure occurs (flashing DAFICS Preflight BIT FAIL light).

Power for the PVD is provided through three PVD circuit breakers on the pilot's right console: 115 volt emergency ac, 26 volt emergency ac and 28 volt essential dc. The laser line will not be displayed if any circuit breaker is open.

PVD Projector

The PVD projector is located on the lower right side of the forward canopy.

Laser Safety

The location of the PVD projector makes it difficult for a pilot in a pressure suit to inadvertently expose an eye to direct laser energy. Laser beam reflections are not hazardous. The PVD should be OFF when the canopy is open to prevent ground support personnel from inadvertently looking directly into the laser beam.

WARNING

Do not look directly into the laser beam.

PVD Control Panel

The PVD control panel is located on the pilot's right console. See Figure 1-65.

Roll Adjust Switch

The ROLL adjust switch allows ±5 degrees of display adjustment in roll. Normally the switch should be aligned with the horizontal index.

Pitch Adjust Switch

The PITCH adjust switch allows vertical adjustment of the reference point on the instrument panel for level flight. When the switch is aligned with the index mark, the laser line will be between the ADI and the HSI when the aircraft is in the nominal attitude for supersonic cruise (6 degrees pitch angle).

Intensity Switch

The OFF detent, at the counterclockwise position of the intensity switch, deenergizes the PVD. Intensity of the laser line increases with clockwise rotation of the switch. The full clockwise position is labeled BRT. The PVD may be manually reset by rotating the switch to OFF and back on or by depressing the white pushbutton projecting above the center of the switch.

Pitch Scale Switch

The pitch SCALE switch allows the pilot to select the ratio of laser line movement in pitch compared to movement of the real horizon. In the normal (NORM) position, the angular movement of the laser line, when observed by the pilot, is the same as the angular movement of the outside horizon. In the 2, 1/2, 1/4 and 1/8 positions, the line moves 2 times, 1/2, 1/4 and 1/8 as much as the angular movement of the outside horizon, respectively.

The position of the laser line changes when the SCALE switch is moved, unless the aircraft is in the nominal attitude for supersonic cruise.

PVD CONTROL PANEL

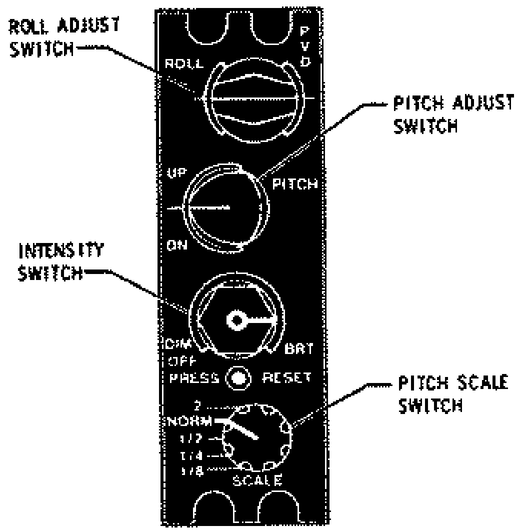


Figure 1-65

2-INCH STANDBY ATTITUDE INDICATOR (Without S/B R-2466)

A self-contained standby attitude indicator is located at the top of the pilot's instrument panel. A cylinder rotates to indicate angles through 80° climb, 90° dive, and 360° roll. The cylinder is inscribed with a horizon and 5° graduations for pitch angle. Bank angle is marked on the instrument case in 10° increments up to 30° and then 30° increments up to 90°. See Figure 1-12.

A cutout disables the self-erection feature when pitch or roll angle exceeds $12 \pm 1/2^\circ$ to prevent the indicator from erecting to a false vertical. This puts the indicator into a free drift mode when above this angle.

The indicator has a built-in 7° pitch bias. With this bias, the gyro horizon will center relative to the case index marks when the pitch attitude is 7° nose-up (5.8° angle of attack). When the aircraft is at a 0° pitch attitude on the ground, the gyro will erect to a 7° nose-down indication.

A combination pitch trim and caging knob is provided on the lower right corner of the indicator. The knob can be turned to position the miniature airplane from 5° up to 5° down. The knob is also used to cage the gyro by pulling the knob out to its fully extended position. The OFF flag appears when the caging knob is pulled. Releasing the knob allows the gyro to erect to 7° nose down and 0° roll within five minutes if the aircraft is level. Pulling the knob out and turning it clockwise locks the cylinder in the caged position. The cylinder should be unlocked during normal use. The nominal erection rate is 2.5° per minute to the apparent vertical.

The indicator receives emergency ac power through the STBY ATT circuit breaker on the pilot's right console. An OFF flag appears if power is interrupted. The indicator displays useful pitch and roll information for at least nine minutes after power loss.

Standby Attitude Indicator Characteristics

The standby attitude indicator has a maximum free drift rate of 0.9 degrees per minute. In level flight, any tendency to drift is continuously corrected to the apparent vertical at a rate of two to three degrees per minute. An acceleration sensor disables the erection system at bank angles greater than twelve degrees to prevent bank errors during turns. In this free gyro mode, a random combination of errors due to gyro drift, earth's rate and earth's profile can accumulate at a maximum of 1.6 degrees per minute.

In wings-level flight, the normal aircraft acceleration and deceleration does not disable the erection system and the indicator will erect to the apparent vertical. During climbout and descent, pitch errors up to approximately four and eight degrees, respectively, can be expected. If a turn is initiated immediately after climbout or descent, the free gyro errors may add to the acceleration induced error. For example, the indicator error could reach four degrees in roll after a normal supersonic descent and a 90° subsonic turn (with bank angle greater than 12°).

If the bank or pitch error does not exceed approximately twelve degrees, the indicator will automatically correct the above errors at the normal erection rate when the aircraft is in level flight and at constant velocity. However, the indicator can be aligned more rapidly by pulling the caging knob. This control should be limited to level flight conditions, as the gyro is erected to the case reference.

NOTE

> If a bank or pitch error of more than twelve degrees is accumulated, the cut-out feature prevents automatic correction when level. Manual caging will be necessary.

3-INCH STANDBY ATTITUDE INDICATOR (With S/B R-2466)

A self-contained standby attitude indicator is located at the top of the pilot's instrument panel. A cylinder rotates to indicate angles of 85° climb and dive, and 360° roll. The cylinder is inscribed with an artificial horizon and 5° graduations for pitch angle. Bank angle is marked on the instrument case in 10° increments up to 30° and then 30° increments up to 90°. See Figure 1-12.

The indicator has a built-in 7° pitch bias. With this bias, the gyro horizon will center relative to the case index marks when the pitch attitude is 7° nose-up (5.8° angle of attack). When the aircraft is at a 0° pitch attitude on the ground, the gyro will erect to a 7° nose-down indication.

A combination pitch trim and caging knob is provided on the lower right corner of the indicator. The knob can be turned to position the miniature airplane from 5° up to 10° down. When the caging knob is pulled to the fully extended position, the OFF flag appears and the gyro horizon is caged to the case level-flight index. Pulling the knob out and turning it clockwise locks the cylinder in the caged position. The cylinder should be unlocked during normal use. The nominal erection rate is 2.5° per minute to the local vertical.

The indicator receives essential dc power through the STBY ATT circuit breaker on the pilot's right console. An OFF flag appears if power is interrupted. The indicator displays pitch and roll information accurate to within $\pm 5^\circ$ for at least nine minutes after power loss.

Attitude Indicator Operating Characteristics

The attitude indicator is maintained erect to the apparent vertical at a rate of two to three degrees per minute when the gyro spin axis displacement is less than 7° in pitch and roll (level flight). The gyro erects to gravity at a reduced rate of 0.75°/minute during turns and fore and aft velocity changes that exceed the 7° erection criteria.

The indicator will erect to the apparent vertical. During climbout and descent, pitch errors of approximately four and eight degrees, respectively, can be expected.

HORIZONTAL SITUATION INDICATOR (HSI)

The HSI, located on the pilot's instrument panel, integrates information from the TACAN, ANS, INS, and the ILS receiver. See Figure 1-66. Power for the HSI is furnished by the essential ac bus.

Range Indicator

The range indicator on the upper left side of the HSI displays distance in nautical miles. A shutter covers the numerals when a distance signal is not present. The maximum range readout is 1999 nautical miles. A "K" shutter covers the first digit of the range readout window when range is less than 1000 miles. When the shutter opens (over 999 NM range) a fixed numeral 1 is displayed. The shutter remains closed unless the DISPLAY MODE SEL switch is in ANS and the BEARING SELECT switch is in NORMAL.

With the DISPLAY MODE SEL switch in ANS and the BEARING SELECT switch in NORMAL, the range readout is to the ANS destination point (DP) or to the ANS-computed turn point, depending on the mode (DP or TURN, respectively) selected on the ANS control panel. Refer to Navigation Control and Display Panel DP/TURN switch, Section IV. With the DISPLAY MODE SEL switch in INS and the BEARING SELECT switch in NORMAL, the range readout is to the INS DP. With the BEARING SELECT switch in TAC/ADF, the range readout is to the selected TACAN station. With the DISPLAY MODE SEL switch not in ANS or INS, range readout is to the selected TACAN station regardless of BEARING SELECT switch position.

Course Selector Window

The COURSE window displays ANS command course when the DISPLAY MODE SEL switch is in ANS. In all other modes, it displays the course that is manually selected with the COURSE SET knob.

Bearing Select Switch

A BEARING SELECT switch, located below the DISPLAY MODE SEL switch on the pilot's instrument panel, has two positions, NORMAL (down) and TAC/ADF (up). This switch is operated with the DISPLAY MODE SEL switch to select inputs to the HSI bearing pointer and HSI range indicator. See Figure 1-66.

Rotary Compass Card

The rotating azimuth ring is read at a stationary lubber line at the 12 o'clock position.

The card displays ANS true heading with the DISPLAY MODE SEL switch in ANS; in any other position, the display is INS computed magnetic heading. The RSO can provide true heading by manually selecting "0" mag var on the INS control panel. When the INS is operating in ATT mode (either by RSO selection or INS computer failure), the heading is set by the RSO using the heading slew knob.

Heading Marker

The heading marker is a rectangular marker located just outside of the rotating compass card. The marker is manually set by the HEADING SET knob except when the DISPLAY MODE SEL switch is in ANS, then, the heading marker displays INS heading.

Bearing Pointer

The bearing pointer is a heavy arrowhead located outside of the rotating compass card. Operation of the bearing pointer is affected by three switches: DISPLAY MODE SEL, BEARING SELECT and the UHF radio function selector.

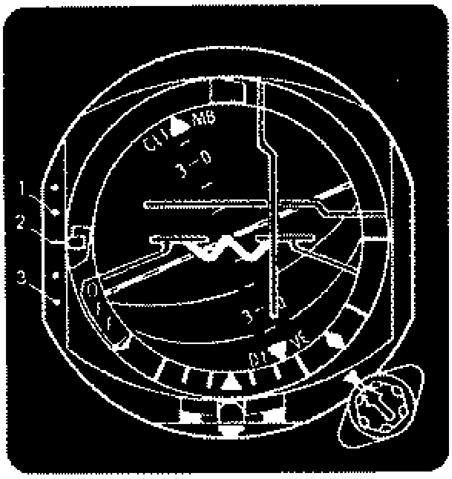
With the DISPLAY MODE SEL switch in ANS, the bearing pointer provides true bearing to a selected TACAN station or UHF transmitter regardless of the BEARING SELECT switch position. With the DISPLAY MODE SEL switch in INS, the bearing pointer is referenced to INS heading and with the BEARING SELECT switch in NORMAL, the bearing pointer displays bearing to the INS DP; with the BEARING SELECT switch in TAC/ADF, it displays bearing to a selected TACAN station or UHF transmitter. With the DISPLAY MODE SEL switch in TACAN ADF, the bearing is INS magnetic to a selected TACAN station or UHF transmitter, regardless of the BEARING SELECT switch position. With the DISPLAY MODE SEL switch in ILS or ILS APPROACH, the bearing pointer is servoed to the lubber line if the BEARING SELECT switch is in NORMAL; if in TAC/ADF, it will display INS magnetic bearing to a selected TACAN station or UHF transmitter.

Selecting the ADF position of the function selector switch on the UHF radio overrides TACAN information and provides UHF/ADF bearing.

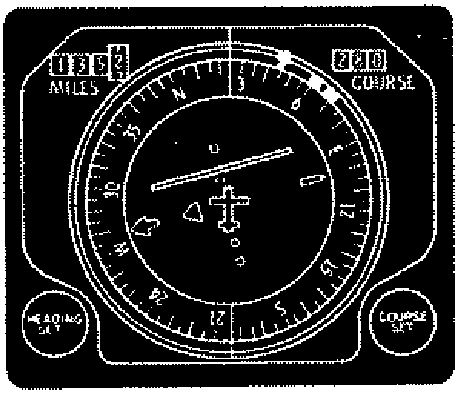
Course Arrow and Course Deviation Bar

The course arrow and course deviation bar are on the face of the rotating compass card. The course arrow displays a commanded or selected course and the course deviation bar displays displacement from the commanded or selected course. When the DISPLAY MODE SEL switch is in ANS, the course arrow and course window display desired ground track as computed by the ANS. The course deviation bar indicates left or right location of the desired course. Full deflection of the course deviation bar is proportional to 1 nm off-course. With the DISPLAY MODE SEL switch in INS, the course arrow is manually set to the desired INS course and the course deviation bar indicates deviation from the INS course. With the DISPLAY MODE SEL switch in TACAN ADF, the course may be selected manually with the COURSE SET knob and course deviation bar signals are received from the TACAN. With the DISPLAY MODE

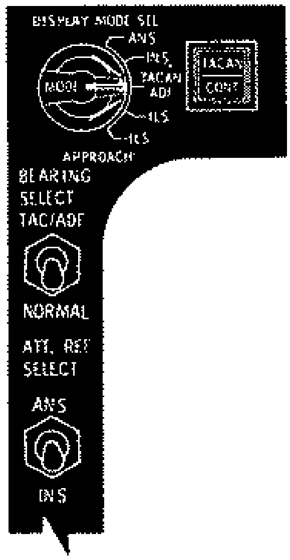
NAVIGATION INSTRUMENTS - Forward Cockpit



ATTITUDE DIRECTOR INDICATOR



HORIZONTAL SITUATION INDICATOR



* With ANS, INS, or TACAN/ADF display mode selected.

1. Indicator position if APW shaker boundary reached.
2. Pitch boundary or glide slope status indicator.
3. Indicator position for Angle of Attack and pitch rate = 0.

◆ 45° Index marks provided on ADI.

NOTE

▲ When UHF/ADF is operating it has priority on bearing pointer

The attitude reference select switch determines which system, ANS or INS, supplies pitch and roll to the attitude director indicator, pitch, roll, and heading to the autopilot. (The autopilot receives steering signals only in the ANS position).

⚠1 If the INS is operating in the attitude mode and the DISPLAY MODE SEL switch is in any position other than ANS, the course warning flag comes into view.

		DISPLAY MODE SELECTOR SWITCH									
		ANS		INS		TACAN/ADF		ILS		ILS/APCH	
		BEARING SEL. SW.		BEARING SEL. SW.		BEARING SEL. SW.		BEARING SEL. SW		BEARING SEL. SW.	
INDICATOR	INDICATOR FUNCTION	NORMAL	TAC/ADF	NORMAL	TAC/ADF	NORMAL	TAC/ADF	NORMAL	TAC/ADF	NORMAL	TAC/ADF
HORIZONTAL SITUATION INDICATOR	HEADING MARKER	INS Heading		Manually Set							
	BEARING POINTER	To Selected TACAN or ADF ▲		INS DP Bearing	TO SELECT TACAN OR ADF ▲	To Selected TACAN or ADF ▲		SERVOED TO LUBBER LINE	TO SELECT TACAN OR ADF ▲	SERVOED TO LUBBER LINE	TO SELECT TACAN OR ADF ▲
	COURSE ARROW	Command Course		Manually Set INS Course		Manually Set TACAN Course		Manually Set Inbound Localizer Course			
	COURSE DEVIATION	Left or Right of Command Course		Deviation from INS Course		Deviation From TACAN Course		Deviation From Localizer Course			
	TO-FROM INDICATOR ▲	Out of View		To Selected INS DP		To Selected Tacan Station		Out of View			
	RANGE INDICATOR	Dist. to DP/TURN	Dist to TACAN Station	Distance to INS DP	Distance to TACAN Station						
	K SHUTTER	Dist. to DP/TURN	Masked								
	AIRCRAFT HEADING	True		INS Heading (Magnetic or Alternate Mode Selected Heading)							
ATTITUDE DIRECTOR INDICATOR	BANK STEERING BAR	Command Course		INS Selected course		TACAN Course		Localizer Course			
	PITCH STEERING BAR	Vertical Velocity		Out of View						Pitch Direction for Glide Slope Beam	
	PITCH BOUNDARY OR G/S INDICATOR	Out of View / Angle of Attack and Pitch Rate Status Relative to APW Boundary						Position Relative to Glide Slope Beam			
⚠1	COURSE WARNING FLAG	← Out of View → ANS Valid		INS Heading Valid		TACAN Valid		Localizer Valid			
	GLIDE SLOPE WARNING FLAG	← Out of View →						Glide Slope Valid			
	POWER WARNING FLAG	Out of View - Attitude Reference Valid									

F203-277(b)

Figure 1-66

SEL switch in ILS or ILS APPROACH, the course arrow should be manually set to the localizer course with the COURSE SET knob; course deviation bar signals are from the ILS.

<u>To-From Indicator</u>

The to-from arrow is located radially between one end of the course arrow and the miniature aircraft at the center of the instrument.

With the DISPLAY MODE SEL switch in INS, to-from information is relative to an INS DP. With the DISPLAY MODE SEL switch in TACAN ADF, to-from information is relative to the selected TACAN station. The to-from arrow is hidden from view in all other display modes.

BEARING, DISTANCE, HEADING INDICATOR

The bearing, distance, heading indicator (BDHI) on the RSO's instrument panel contains a rotating compass card, range window, and two pointers.

The compass card displays ANS or INS heading, depending on the position of the BDHI SEL-HEADING switch. The No. 1 pointer displays ADF, INS DP, or TACAN bearing, depending on the position of the BDHI SEL-NO. 1 NDL switch. The No. 2 pointer always displays ANS command course.

Range Readout Window

The range readout window behind the shutter is a three-digit counter which displays slant range to a TACAN station in nautical miles. The maximum range readout is 999 nautical miles. When TACAN information is unreliable, a shutter covers the range readout.

<u>BDHI Heading Select Switch</u>

The two-position BDHI heading select switch on the RSO's instrument panel is labeled HEADING. With the switch in INS (up), the compass card displays INS heading. With the switch in ANS (down), the compass card displays ANS heading.

<u>Heading Slew Control</u>

The HEADING SLEW control, on the RSO's instrument panel, is used to set the INS heading when the INS is in the ATT (attitude) mode.

<u>BDHI SEL NO. 1 Pointer Select Switch</u>

The BDHI SEL NO. 1 pointer select switch on the RSO's instrument panel has three positions: ADF (up), INS (center) and TACAN (down). The No. 1 pointer displays an ADF, INS DP, or TACAN bearing, as selected.

ATTITUDE INDICATOR-RSO

An attitude indicator on the RSO's instrument panel receives pitch and roll signals from either the ANS or INS. Power for the indicator is furnished by the emergency ac bus through the ATT IND circuit breaker on the RSO's right console.

<u>Attitude Indicator Selector Switch</u>

The ATT IND switch on the RSO's instrument panel has two positions: INS (up) and ANS (down). The switch selects the attitude reference source for the attitude indicator and for ready signals to the power-off flag.

ACCELEROMETER

A mechanical accelerometer on the pilot's instrument panel has one indicating pointer and two recording pointers (one each for loads greater or less than 1-G). The recording pointers remain at the maximum travel positions reached by the indicating pointer, thus providing a record of maximum positive and negative G-loads encountered. To return the recording pointers to the 1-G position, press the button on the lower left corner of the instrument.

MAGNETIC COMPASS

A magnetic compass is located on the forward part of the pilot's canopy. An on-off toggle light switch is located below and to the left of the compass.

COMMUNICATIONS AND AVIONIC EQUIPMENT

The communications and avionic equipment includes:

AN/AIC-18	Interphone system.
COMNAV-50	UHF communication and navigation system (UHF-1 and UHF-2).
AN/ARA-48	Automatic direction finder equipment, used with COMNAV-50.
AN/ARC-186(V)	VHF communication system.
618-T	HF radio.
ARC-190	HF radio.
51RV-1	Instrument landing system.
Wilcox 914X-2	IFF transponder.
G-Band Beacon	For radar tracking during special tests.
I-Band Beacon	For radar tracking during special tests.
AN/ARN-118(V)	TACAN system.

Refer to Section IV for descriptions of the Inertial and Astro-inertial Navigation systems, the sensor equipment, and the defensive systems. Some instruments used to display avionic systems navigation information are described under Flight and Navigation Instruments, this section.

Pilot's Microphone Switches

The pilot's microphone is connected to the radio transmitter selected on the interphone control panel when the pilot either depresses the microphone switch on the right throttle knob or selects the TRANS position of the microphone switch on the control stick grip. A transmission side tone is heard while the transmitter is keyed. The button and switch are spring loaded to the off position. When not transmitting, the pilot can receive communication signals selected on the interphone control panel. When the switch on the control stick is moved to INPH, the pilot's microphone is only connected to the interphone circuit. The Mission Recorder System records voice signals when either TRANS or INPH is selected or the interphone panel HOT MIC switch is on.

RSO's Microphone Switches

The RSO's radio transmit switch is located in the left footrest and the interphone switch is located in the right footrest. A quiet/listen (muting) switch is provided on the right side of the cockpit floor to inhibit all audio input to the RSO's headset except pilot's CALL signals. The quiet/listen switch operates only when the RSO's interphone selector knob is in the MUTE or INTER position. The Mission Recorder System records the RSO's voice signals whenever the aft cockpit radio transmit or interphone switch is depressed or the interphone panel HOT MIC switch is on.

INTERPHONE SYSTEM, AN/AIC-18

The interphone system provides crew intercommunication plus crewmember-to-ground-crew interphone and crewmember radio communication. Each crewmember's microphone and headset are connected to the communication equipment selected by the selector knob on individual interphone control panels. The system provides:

1. Either press-to-talk or HOT MIC interphone communication between cockpits.
2. HF, UHF-1, UHF-2, and VHF radio transmission and reception for the pilot and RSO.
3. A CALL button in each cockpit for emergency communication between cockpits.
4. Interphone communication with the ground crew by means of an external receptacle in the nosewheel well.

5. Landing gear system unsafe audio warning (pulsed tone) and low KEAS aural warning (steady tone) to pilot and RSO.

6. Monitoring of navigation radio audio signals.

7. Interphone communication with tanker during aerial refueling.

8. Recording of interphone system signals on the Mission Recorder System (MRS) whenever the pilot's or RSO's transmit or interphone switches are keyed, or when HOT MIC is selected.

Interphone Control Panel

An interphone system control panel is located on the pilot's right console and on the RSO's left console. Power is supplied by the essential dc bus through individual INTPH circuit breakers in each cockpit. Each panel contains a volume control, seven monitoring knobs, a HOT MIC knob, a six-position rotary selector, and a CALL button. See Figure 1-67.

Monitoring Switches

The seven combination volume-control and push-pull switches at the top of the interphone panel are labeled: INTER, UHF-1, HF, IFR COM, VHF, UHF-2, and TACAN/ILS. Pulling out a switch connects the output of the labeled equipment to the headset of the crewmember. Rotating a monitor switch knob varies the volume of the signal. The VOL (volume control) knob establishes the maximum strength of all seven monitor switches. The in-flight refueling communications (IFR COMM) monitor switch is used to monitor the tanker interphone system after refueling contact.

The TACAN and ILS systems share a single monitor button. TACAN signals are available except when the DISPLAY MODE SEL switch is in ILS or ILS APPROACH (when ILS signals become available).

Selector Knob

The rotary selector knob, labeled INTER, UHF-1, HF, UHF-2, VHF, and MUTE, connects the microphone and headset of the crewmember in that cockpit to the communication systems as follows:

INTER - Provides normal voice communication with the opposite cockpit and with the ground crew (if connected).

UHF-1 - For transmission and reception using the UHF radio in the forward cockpit.

HF - For transmission and reception using the HF radio.

UHF-2 - For transmission and reception using the UHF radio in the aft cockpit.

VHF - For transmission and reception using the VHF radio.

MUTE - Provides the RSO with a quiet/listen (muting) capability. When this position is selected and the quiet/listen switch on the aft cockpit floor is depressed, all interphone signals are muted except those from the pilot's CALL. This position is not normally used unless isolation from communication system distractions is required.

Hot Microphone Switch

The push-pull HOT MIC switch permits continuous interphone conversation between crewmembers without depressing a microphone button. Two-way hot microphone operation results if each crewmember pulls out the HOT MIC knob and sets the rotary selector switch to any position except INTER. (HOT MIC transmissions are not possible when the rotary selector in that cockpit is set to INTER.) In one-way operation, the HOT MIC signals are not received unless the other crewmember has the interphone monitor switch pulled up and/or the rotary selector switch set to INTER.

INTERPHONE CONTROL PANEL

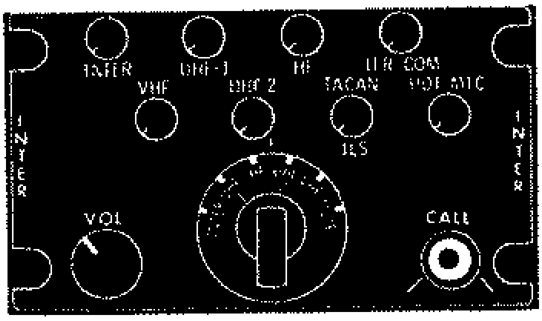


Figure 1-67

Volume Control

The volume (VOL) control knob adjusts the maximum audio level for all monitored signals.

Call Button

Depressing the spring-loaded CALL button permits interphone conversation to override any signals being received by the other crewmember. The calling volume is preset at a higher level and cannot be adjusted. A microphone switch does not have to be depressed to use CALL.

INTERPHONE SYSTEM NORMAL OPERATION

Normal communication between cockpits is accomplished by depressing the INPH switch (Pilot's Control Stick Grip and RSO's right side communication button on aft cockpit footrest). The crewmember in the opposite cockpit can hear if the interphone (INTER) monitor switch is pulled out and set to normal volume.

COMNAV-50 UHF RADIO

The COMNAV-50 UHF radio provides voice transmission and reception on any of 7000 channels in the P-Band frequency range. A direction-finding capability is also supplied. The radio is conventional except for a capability to operate in either an "Internal" mode (compatible with conventional UHF radio equipment), and an "External" mode (not compatible with other types of UHF radios). In the external mode, coded communication is only possible with other COMNAV-50 (or equivalent) radios. This mode has high resistance to jamming, allows message privacy, and has a range measuring capability.

Two independent UHF radios are provided, designated UHF-1 (front cockpit) and UHF-2 (aft cockpit). They can be used independently, within limits, for internal mode communication. A modulator/demodulator (Modem) control (COMM) panel in the aft cockpit controls coding of external mode signals and discrete selection of the ranging partner. The Modem controls can be switched by the RSO to become a part of either UHF system and give either UHF-1 or UHF-2 an external mode operating capability. Internal mode voice communication capability (without ADF function) is maintained by the opposite system; however, external mode transmissions can interfere with reception by the system operating in the internal mode if proper frequency separation is not maintained. (See Figure 1-68).

Transmitter Power Output

Transmitter power output in the internal mode is adjustable in five steps from 8.0 microwatts to 30 watts in the frequency range from 225 to 399.975 MHz. Power positions 5 and 6 have the same transmitter power output in the internal mode.

WARNING

ILS reception can be affected by UHF transmission at high power settings.

NOTE

When making an ILS approach, set power 4 or lower.

The transmitter power output in the external mode is adjustable in six steps from 8.0 microwatts to 100 watts in the frequency range from 230 to 394.975 MHz. Transmitter power should be kept as low as practical to reduce the possibility of detection. The following tables list recommended transmitter power levels for air-to-air communication in the external mode.

For Voice Communication and Ranging

Power Level	Estimated Distance Capability
6	300 plus nm
5	100 to 300 nm
4	10 to 100 nm
3	1 to 10 nm
2	less than 1 nm
1	less than 0.1 nm

For Direction Finding

Power Level Of Other Transmitter	Distance To Other Transmitter
6	100 to 200 nm
5	30 to 100 nm
4	3 to 30 nm
3	0.3 to 3 nm

External Mode Signal Characteristics

In the external mode, the transmitted signal is encoded to appear as noise to all receivers except those equipped with a compatible decoding device. The "mission code" feature prevents intelligible reception by stations which possess the necessary equipment but do not have the code. Ranging can occur only between two stations with the address code. Automatic direction finding (ADF) can also be accomplished with an addressed station in the external mode concurrently with ranging. Voice/ranging communication and ADF operation have distinct range differences for the same power level. Range measurements can be accomplished only in the external mode.

UHF Radio Equipment Location and Power Supplies

The UHF radio units, modulator/demodulator (Modem coding unit), and the ARA-48 automatic direction finding (ADF) equipment are located in the radio bay and cooled by the environmental control system. Power is furnished by the essential ac bus and the monitored dc bus.

CAUTION

If either canopy is open, the aft canopy latch handle must be in the aft position or the cockpit air handle must be in the forward (off) position for adequate equipment cooling. Otherwise, most of the cooling air would exit through the cockpit openings instead of the bays.

UHF Control Transfer Switch

The push-on/push-off control transfer switch, labeled UHF TRANS, is located on the left side of the RSO's instrument panel. The switch determines which UHF radio has ADF and EXT mode capabilities. When the UHF TRANS switch is on (illuminated), UHF-2 is connected to the ADF antenna, the UHF Modem, and the forward UHF blade antenna; and UHF-1 is connected to the aft UHF blade antenna (INT mode voice communication capability only). Depressing the UHF TRANS switch when it is illuminated extinguishes the light and reverses UHF-1/UHF-2 capabilities. ADF and EXT mode communication/ranging can only be accomplished by the UHF radio connected to the ADF antenna, UHF Modem, and the forward antenna by the UHF TRANS switch. INT mode voice communication is always possible on either UHF radio. Refer to Figure 1-69.

NOTE

- Either crewmember can monitor or transmit on either the UHF-1 or UHF-2 system.
- The RSO always controls the Modem panel.

COMNAV-50 UHF COMMAND RADIO CONTROL PANELS AND INDICATORS

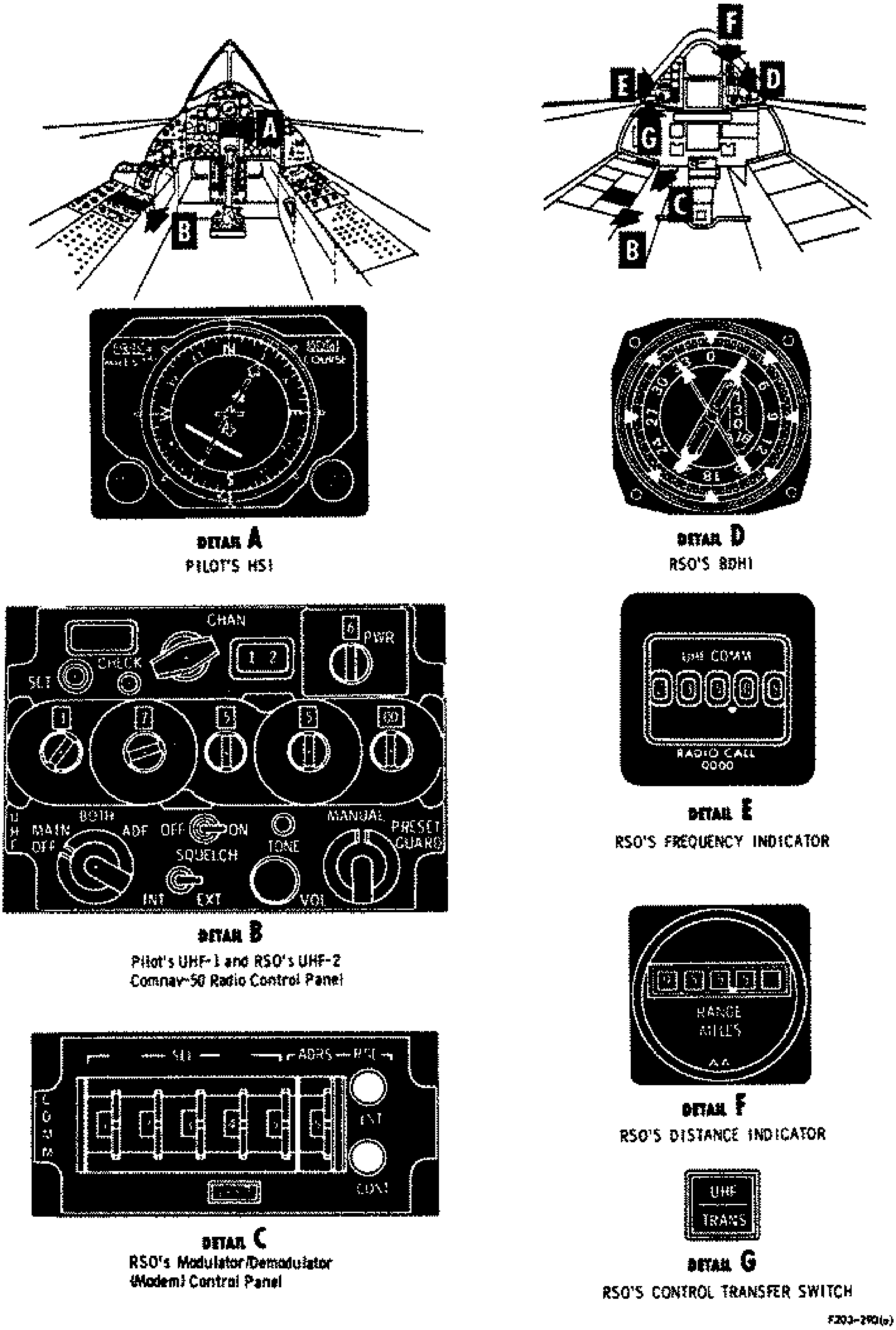


Figure 1-68

UHF RADIO CONTROL PANELS

A UHF radio control panel, labeled UHF, is located in each cockpit. Each crewmember can independently control operation of a UHF transmitter and its guard channel receiver. The ADF function on a radio is only operative when the RSO has selected control of the ARA-48 direction finding equipment (and the forward UHF antenna) for that radio, using the UHF TRANS switch.

Function Selector Switch

A four-position rotary function selector switch turns the radio on and OFF and selects MAIN, BOTH or ADF. ADF is operable with UHF-1 if the UHF TRANS switch light is off, and with UHF-2 if the UHF TRANS switch light is on.

The UHF radio is not energized when the function selector switch is OFF. In MAIN, only the transmitter and main receiver operate. In BOTH, the transmitter, and main and guard receivers operate. The Modem unit (used for external operation) is in standby when the selector is not OFF. In ADF, the ARA-48 is energized, and the main receiver and transmitter operate. Directional signals from the ARA-48 can be displayed on the forward cockpit HSI bearing pointer and the aft cockpit BDHI No. 1 needle.

NOTE

The guard receiver is inoperative in the ADF position.

Manual-Preset-Guard Selector

The MANUAL-PRESET-GUARD switch controls frequency selection. In MANUAL, the manual frequency selector switches are functional. In PRESET, the preset channel selector switch is functional. In GUARD, guard channel frequency (243.0 MHz) is set on the main receiver and transmitter.

Preset Channel Selector Switch

The preset channel selector switch, labeled CHAN, selects one of twenty preset frequencies when the MANUAL-PRESET-GUARD selector is in PRESET. The channel number appears to the right of the selector.

Manual Frequency Selector Switches

Five rotary switches located across the middle of the UHF radio panel selects any one of 7000 frequencies when the MANUAL-PRESET-GUARD selector knob is in MANUAL. The frequency is displayed above the switches.

INT-EXT Mode Switch

The two-position transmitting mode selector switch is labeled INT-EXT. In INT, the UHF radio transmits and receives narrow-band AM signals. This position is used for conventional UHF transmitting and receiving. In EXT, the radio and modulator/demodulator (Modem) are used together to receive and transmit the wide-band pseudonoise encoded signal. Range information in EXT is displayed on the distance indicator in the aft cockpit. Direction-finding using the ARA-48 ADF can be done with the switch in INT or EXT. The UHF radio which is not controlling external mode operation has only internal mode communication capability and no ADF function.

Power Selector Switch and Indicator

A rotary PWR switch controls transmitter output power in the internal and external modes. A digit above the knob indicates the relative power output, from "1" through "6". Power output can be set from a maximum of 30 watts (position 6 and 5) to a low of 8.0 microwatts (position 1) in the INT, or narrow band mode. In the EXT, or wide band mode, power is set from a maximum of 100 watts (position 6) to approximately 8.0 microwatts (position 1).

Volume Control

Clockwise rotation of the VOL knob increases receiver volume. The normal setting is nearly full clockwise so that the interphone controls can be set to make UHF volume compatible with other signals.

Tone Button

The TONE push-button generates a 1020-cycle tone for audio checking or homing. The tone can be generated in either the internal or external mode; however, the tone is inoperative in the external mode when the CONT switch light is illuminated.

Squelch Switch

The two-position SQUELCH switch provides ON and OFF positions for the control of receiver background noise. OFF position allows reception of weak signals and noise.

Frequency Set Switch

Pressing the SET button changes the frequency of that preset channel to the frequency set by the manual frequency selector switches.

Check Switch

Pressing the CHECK button displays the manual frequency corresponding to the selected preset channel in the display window above the switch.

REMOTE FREQUENCY INDICATOR

A UHF remote frequency indicator on the RSO's instrument panel shows the setting selected in the forward cockpit UHF. When the MANUAL-PRESET-GUARD switch on the UHF-1 radio control is in MANUAL, the selected frequency is displayed numerically on the indicator. The selected preset channel number is displayed when in PRESET. The indicator displays "G" when in GUARD.

MODULATOR/DEMODULATOR (MODEM) CONTROL PANEL

The modulator/demodulator (Modem) coding equipment provides the UHF radio with the following capabilities:

1. Communications with discreetly selected partners.
2. Message privacy.
3. Ranging (semi-automatic or automatic).
4. Combined direction finding and ranging.

The Modem control (COMM) panel is located on the forward portion of the RSO's left console. The Modem controls only function in the EXT mode of the UHF radio (UHF-1 or UHF-2) that is using the forward UHF antenna (as selected by the RSO's UHF TRANS switch).

Code Selector Switches

Five rotary selector switches, labeled SEL, have positions from 0 through 7. These switches establish the signal code in the external mode. A code of all zeros cannot be used. Stations must have identical code settings to communicate in the external mode.

Range Address Switch

The right rotary selector switch, labeled ADRS, provides selective ranging. It has eight settings: 0 through 5 plus "A" and "T". The "0" position is an off position which prevents another terminal from ranging, although discrete voice communication capability is retained. Positions 1 through 5 provide range addresses. The "A" position allows a range measurement on any terminal which can respond (having the same synchronization code setting), regardless of its range address code. This is considered an emergency code. "T" is a test position for checking indicator lights on the Modem control panel.

Interrogate Switch and Indicator Light

Depressing the interrogate (INT) push-button switch initiates range and bearing interrogations in the external mode. The INT switch illuminates while the range (or direction and range) is obtained. The switch light extinguishes after three seconds. If the UHF radio function switch is in MAIN, BOTH, or ADF the one-time range measurement in nautical miles and tenths of miles is displayed on the RSO's Distance Indicator. If the UHF radio function switch is in ADF, a one-time bearing is provided to the pilot's HSI and RSO's BDHI. The INT switch is also used with the continuous ranging (CONT) switch to establish automatic ranging and direction finding.

Continuous Ranging Switch

A continuous ranging (CONT) switch and indicator light at the right of the Modem panel initiates continuous ranging. Illumination of the CONT switch light shows that the Modem is in continuous ranging or in continuous ranging and direction finding. The mode depends on setting the UHF radio function selector switch to MAIN, BOTH, or ADF, as discussed above. The CONT switch illuminates at both stations while in continuous ranging or continuous ranging and direction finding; however, the station addressed must have activated its continuous ranging switch to maintain the sequence. The RSO's Distance Indicator is updated each five seconds in the continuous mode. The automatic cycle terminates if the CONT switch is depressed, if a microphone switch at either terminal is depressed for at least three seconds after tone terminates, or the INT/EXT mode switch is placed to INT. The distance indicator at the transmitting station will then reset to zero and the indicator at the receiving station will retain the last distance value. After the cycle is broken by depressing a microphone switch, the operator of either station can reestablish the mode by depressing the INT pushbutton.

Response Light

A respond (RESP) indicator light at the bottom of the Modem control panel illuminates while the other UHF system operating in the external mode is performing a range measurement.

DISTANCE INDICATOR

A distance indicator, on the RSO's instrument panel, displays the distance in nautical miles and tenths of miles between two COMNAV-50 (or equivalent) radio systems ranging in the external mode. The maximum indication is 999.9 miles. A negative contact indicates 000.0.

UHF ANTENNAS

Two fixed UHF blade antennas and a flush-mounted direction finding (ADF) antenna are provided. The forward (No. 1) blade antenna is under the left chine, abeam the cockpit. The aft (No. 2) antenna is under the right chine by the wing leading edge. The ADF antenna is on the centerline of the fuselage, below the aft cockpit.

The RSO's UHF TRANS switch controls access to the forward UHF blade and ADF antennas. See UHF Control Transfer Switch under COMNAV-50 UHF Radio, this section.

UHF RADIO OPERATION

The pilot controls UHF-1. The RSO controls UHF-2, the UHF TRANS switch and the Modem (COMM) panel. When the UHF TRANS switch is on (illuminated), only UHF-2 can use the direction finding and/or external communication modes. To use ADF and/or EXT mode functions on UHF-1, the RSO must select UHF TRANS off (not illuminated). The RSO controls the Modem for EXT operation with either radio. See Figure 1-69.

Operations in Internal Mode

Normal Operation:

UHF radio panel:

1. Mode switch - INT.
2. Volume control - Nearly full clockwise.

   Use the interphone panel controls for volume adjustments. If necessary, decrease the UHF volume control level to maintain compatibility with the range of adjustments available in the interphone panel.

COMNAV-50 UHF AND ARC-186(V) VHF
RADIO CONTROL AND SIGNAL CHANNELS
UHF TRANS on, RSO on UHF-2, Pilot on UHF-1, VHF off.

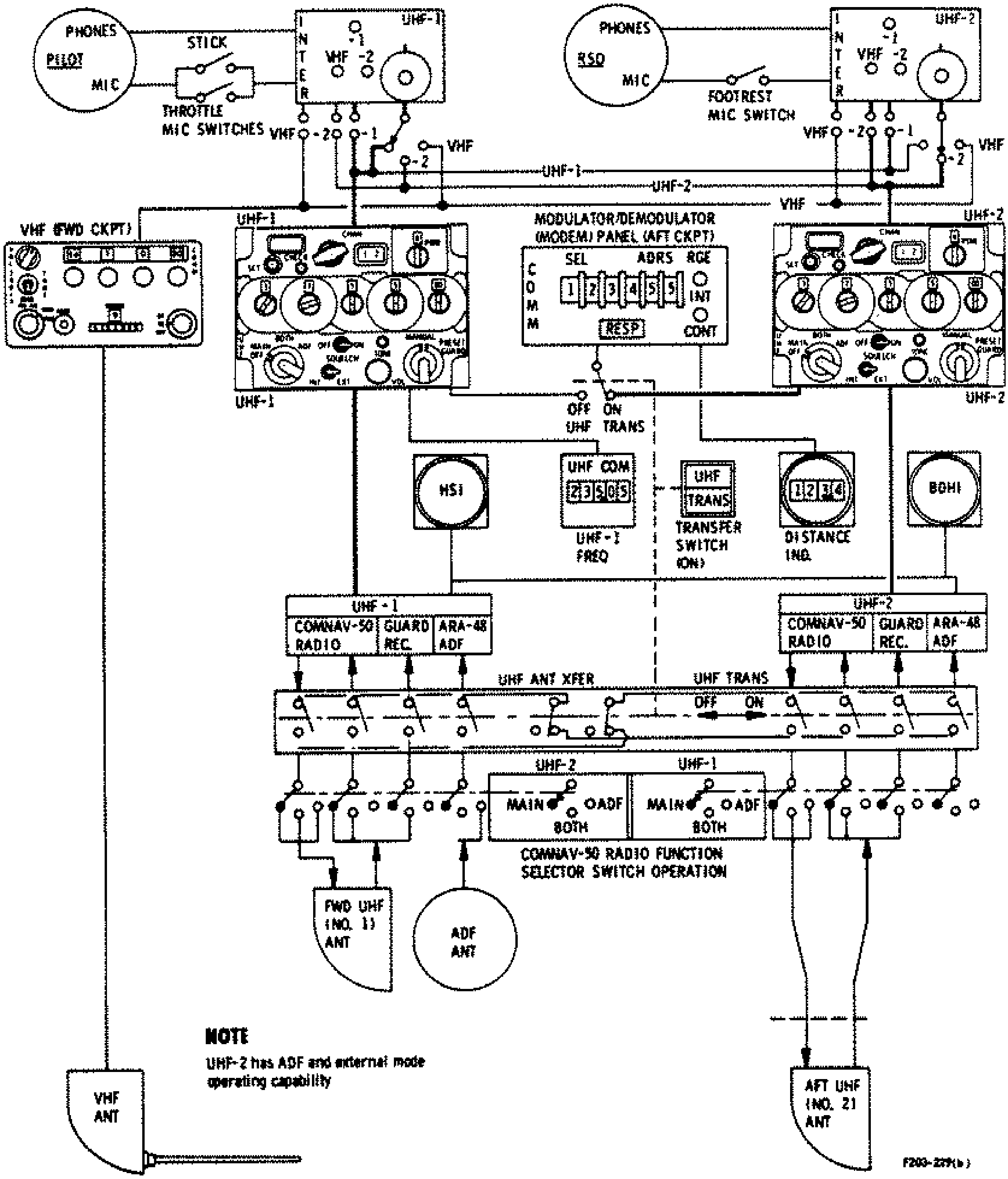


Figure 1-69 (Sheet 1 of 3)

COMNAV-50 UHF AND ARC-186(V) VHF RADIO CONTROL AND SIGNAL CHANNELS

UHF TRANS off, RSO on UHF-2, Pilot on UHF-1, VHF off.

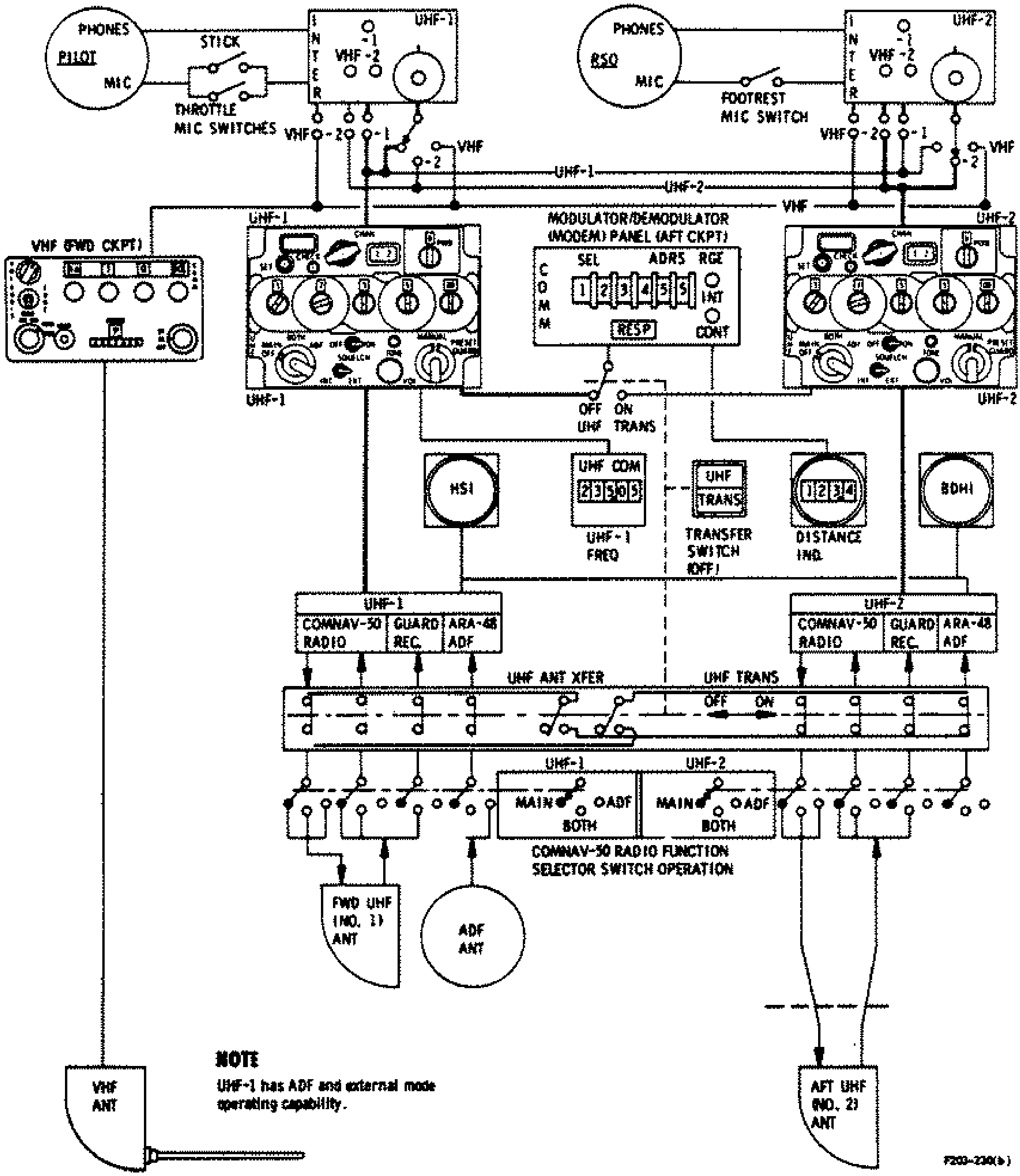


Figure 1-69 (Sheet 2 of 3)

The RSO can operate the UHF-2 system independently when the UHF TRANS switch is on (illuminated). If UHF-1 is to be operated in the external mode, the UHF TRANS switch must be off and the RSO must operate the Modem panel. When operating one UHF in the external mode, the other UHF can only be operated in the internal mode. External mode transmissions by one radio can interfere with reception by the other radio.

UHF radio panel:

2. Mode switch - EXT.

NOTE

Communication in the external mode is possible only with another station having the same capability.

3. Volume control - Nearly full clockwise.

Use the interphone panel controls for volume adjustments. If necessary, decrease the UHF volume control level to maintain compatability with the range of adjustments available in the interphone panel.

4. Power switch - Set.

Position 6 is normal; however, position 5 is normally the maximum within the U.S.

5. Frequency - Set PRESET or MANUAL.

Set the channel number with the CHAN knob if in PRESET.

Use the manual frequency selector switches if in MANUAL.

With GUARD selected the main receiver and transmitter are switched to the internal mode. GUARD has priority over the external mode. ADF is functional.

6. Function selector switch - MAIN or BOTH.

Set BOTH if guard channel monitoring is desired. Guard channel monitoring is not functional in ADF mode.

Approximately ten minutes is required for warm-up of the external mode equipment.

Modem (COMM) panel:

(T7) Code selector switches - Set as briefed.

Set 0 to 7 on each of the five code selector switches.

(T8) Range address switch - Set 0, or as briefed.

The zero setting prevents ranging (unless the other station's ADRS switch is in A) but does not restrict voice communication capability.

For voice communication:

9. Interphone controls - Set.

Select UHF-1 or UHF-2 with the interphone panel rotary selector knob and pull appropriate monitor button. Adjust volume by interphone volume control and the monitor button.

To transmit:

10. Radio transmit switch - Hold depressed.

A one-second tone will be heard. Begin transmission after tone.

For semiautomatic (one-time) ranging or range and bearing:

Accomplish steps (1) through (8) above, then:

For bearing display:

1. Bearing select switch - TAC/ADF

Required only if the DISPLAY MODE SEL switch is in INS, ILS or ILS APPROACH.

(2.) No. 1 pointer select switch - ADF

3. Function selector switch - ADF

Guard channel monitoring is not available in ADF.

To initiate ranging:

(T4.) Interrogate (INT) push-button - Depress momentarily.

The light in the INT switch illuminates for approximately three seconds. A range indication appears on the RSO's distance indicator when the light goes out and bearing will be indicated on the HSI and BDHI if in ADF mode.

To update range or range-and-bearing indications:

(T5.) INT push-button - Depress momentarily.

To communicate with range partner:

6. Microphone switch - Press.

Wait for tone to mute.

For automatic (continuous) ranging or range and bearing:

Accomplish steps (1) through (8) as for Operations in the External Mode. Then:

For bearing display:

1. Bearing select switch - TAC/ADF.

Required only if the DISPLAY MODE SEL switch is in INS, ILS, or ILS APPROACH.

(2.) No. 1 pointer select switch - ADF.

3. Function selector switch - ADF.

Guard channel monitoring is not available in ADF.

To initiate automatic ranging cycle:

(T4.) CONT push-button - Depress.

(T5.) INT push-button - Depress.

NOTE

If ranging stops or can not be initiated, but voice communication is satisfactory, it may be due to marginal signals or to a temporary condition. Attempt to resume ranging by pressing the INT switch. If ranging does not resume, increase power at one or both stations.

(T6.) CONT light - Check illuminated.

(T7.) INT and RESP lights - Check alternating illumination.

Both stations will update readings: each 5 seconds if ranging only; each 8 seconds if ranging with one way ADF; or each 12 seconds if ranging with two way ADF.

NOTE

- The equipment will automatically reinterrogate once if a ranging interrogation cycle is not completed after continuous ranging has been established. The digital range indication will be held for approximately 10 seconds and then reset to zero, if ranging is not reestablished.
- An erroneous range may occasionally appear, but the proper value will be updated during the next interrogate cycle.

To terminate cycling:

8a. Microphone switch - Press.

Wait for tone to mute. A tone will be heard: from 0 to 5 seconds (ranging only), 0 to 8 seconds (one way ADF), or 0 to 12 seconds (two way ADF), depending on progress of the cycle.

NOTE

If ranging only, and a transmission is begun within 1.5 seconds after muting, that transmission will be to the ranging partner only. Subsequent transmissions will be heard by all stations having identical code selections.

OR

b. INT/EXT mode switch - INT.

OR

(T) c. CONT push-button - Depress

Check that CONT light goes out.

As an alternate method of receiving ADF bearing during air refueling rendezvous, the UHF radio may be used to provide instantaneous direction finding (without ranging) in the external mode. This procedure is advantageous at close range.

The RSO should request the tanker to discontinue continuous ranging and transmit a tone in the external mode for five seconds every 15 seconds. The SR-71 UHF radio should be in the external mode with ADF selected. When the tone is transmitted, the ADF bearing will lock on steady and not tend to oscillate. Ranging with the tanker can be reestablished intermittently by requesting the tanker to reengage continuous ranging at one minute intervals.

AN/ARA-48 AUTOMATIC DIRECTION FINDER (ADF)

The AN/ARA-48 ADF antenna is used with the UHF system. When the UHF internal (INT) mode is used, the direction finder will point to emissions from any standard UHF radio transmitting on the same frequency. When the UHF external (EXT) mode is used, the ADF function is only compatible with other COMNAV-50 (or equivalent) equipment.

The ADF antenna is located under the RSO on the aircraft centerline. ADF is selected by the UHF radio function selector switch. Directional information can be displayed by the bearing pointer of the pilot's HSI and by the No. 1 pointer of the RSO's BDHI. Power is obtained from the essential ac and monitored dc buses. (See UHF Radio System Operation, this section).

AN/ARC-186(V) VHF RADIO

The ARC-186(V) VHF radio provides AM transmission and reception from 108.000 to 151.975 MHz. Frequency spacing is 25 KHz and 20 channels can be preset in addition to the preset guard frequency (121.5 MHz). Either narrowband or wideband operation is available, but must be preset by maintenance personnel. The FM capability of this radio is not operative.

The receiver/transmitter is located in the right forward chine bay. The antenna is located on the lower left fuselage, opposite the UHF-2 antenna. The radio control panel (Figure 1-70) is on the pilot's right console. Electrical power is provided by essential dc bus number 2.

Mode Select Switch

The rotary mode select switch has three positions:

OFF - Removes power.
TR - Applies power.
DF - Inoperative.

Frequency-Control/Emergency-Select Switch

The rotary frequency-control/emergency-select switch has four positions:

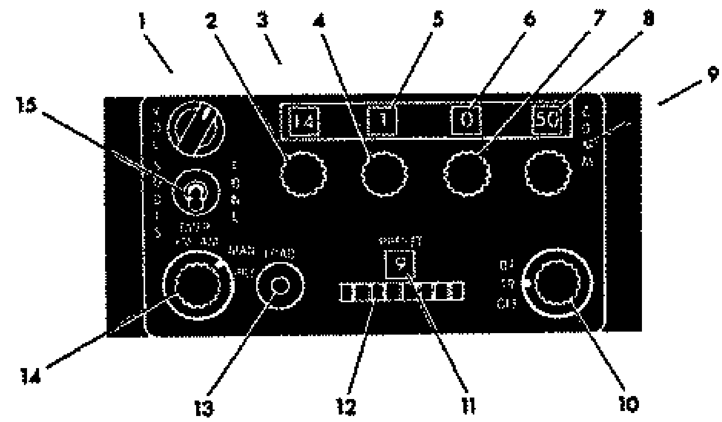
EMER FM - Inoperative.

EMER AM - Selects guard frequency (121.5 MHz).

MAN - Enables manual frequency selection.

PRE - Enables preset channel selection.

VHF CONTROL PANEL



1	VOLUME CONTROL	9	0.025 MHz SELECTOR
2	10.0 MHz SELECTOR	10	MODE SELECT SWITCH
3	10.0 MHz INDICATOR	11	PRESET CHANNEL INDICATOR
4	1.0 MHz SELECTOR	12	PRESET CHANNEL SELECTOR
5	1.0 MHz INDICATOR	13	LOAD SWITCH
6	0.1 MHz INDICATOR	14	FREQUENCY CONTROL/EMERGENCY SELECT SWITCH
7	0.1 MHz SELECTOR	15	SQUELCH DISABLE TONE SELECT SWITCH
8	0.025 MHz INDICATOR		

F203-225

Figure 1-70

Squelch-Disable/Tone-Select Switch

The squelch-disable and tone-select toggle switch has three positions:

Center Position	-	Enables squelch.
SQ DIS	-	Disables squelch.
TONE	-	A spring-loaded-to-center position that transmits a tone at 1000 Hz for audio checking or homing.

Volume Control

Clockwise rotation of the volume control knob, labeled VOL, increases volume. The normal setting is nearly full clockwise so that the interphone controls can be set to make VHF volume compatible with other signals.

Manual Frequency Selector Switches

Four rotary switches are used to manually select a frequency when the frequency-control/emergency-select switch is in MAN. The windows above each switch display the selected frequency.

Preset Channel Selector

A rotary preset channel selector switch is used to select one of 20 preset frequencies when the frequency-control/emergency-select switch is in PRE. The selected channel number is displayed above the switch.

Load Switch

Pressing the recessed LOAD push-button switch changes the preset channel to the frequency set by the manual frequency selector switches.

VHF OPERATION

The VHF radio control panel is in the forward cockpit.

1. Mode select switch - TR.

2. Frequency-control/emergency-select switch - PRE or MAN.

   Set the channel number with the preset channel selector if in PRE.

   Use the manual frequency selector switches if in MAN.

3. Volume control - Nearly full clockwise.

4. Interphone controls - Set.

   Select VHF with the interphone panel rotary selector switch and pull VHF monitor button. Adjust volume by interphone volume control and the monitor button.

5. Squelch-disable/Tone-select switch - Center position.

To transmit:

6. Transmit switch - Hold depressed.

HF RADIO SYSTEM, 618-T

The 618-T HF radio is a long-range voice communications transceiver. The modes of transmission are single sideband (SSB) and amplitude modulation (AM). The frequency range is 2 to 30 MHz, tunable in 1-KHz steps. Because of the nature of the antenna (comprising the pitot boom and the insulated forward portion of the aircraft nose) it is not advisable to transmit at frequencies below 4 MHz. The equipment includes a transceiver and semiautomatic antenna coupler/coupler control, mounted in the radio bay in the right chine. The HF radio control panel is on the RSO's left console (Figure 1-71). Electrical power is supplied by the essential ac and essential dc buses.

Long-range HF communication is highly sensitive to hourly and seasonal variations in propagation conditions. For best results, frequency assignment planning should be based on HF propagation predictions.

Function Switch

The four-position function selector switch, labeled OFF, USB, LSB, and AM, energizes the equipment and selects the desired operating mode. In USB (upper sideband), only the upper sideband signal is transmitted or received. This is the sum of the voice signal and the radio frequency (rf) signal. In the LSB (lower sideband) position, only the lower sideband signal is transmitted or received. This signal is the difference of the voice signal and the rf signal. In the AM position, the signal is amplitude modulated and both sidebands and the original rf signal are transmitted and received.

Frequency Selector Knobs

Four rotary switches manually select a frequency. The windows in the middle of the panel display the frequency selected.

RF Sensitivity Knob

The RF sensitivity knob, labeled RF SENS, adjusts the receiver sensitivity level to control the signal-to-noise ratio.

Normal Operation:

1. Interphone controls - Set.

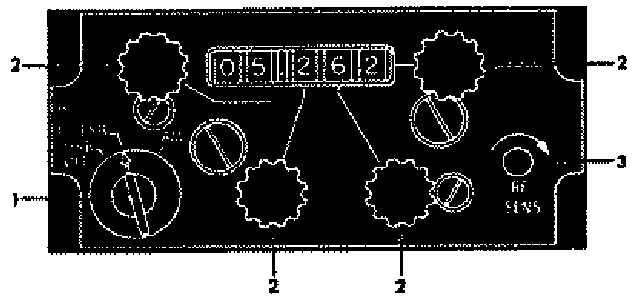
   Select HF with the interphone panel rotary selector switch and pull HF monitor button. Adjust volume by interphone volume control and the monitor button.

2. Function switch - USB, LSB, or AM.

3. Frequency selector knobs - Set.

   The muting of sound in the headset indicates the receiver is tuning to the new frequency.

HF CONTROL PANEL - 618T



1 FUNCTION SWITCH
2 FREQUENCY SELECTOR SWITCHES
3 RF SENSITIVITY SWITCH

Figure 1-71

When background sound is again heard:

4. RF sensitivity knob - Adjust.

   Turn the RF SENS knob clockwise until a distinct crackling background noise is heard. At this setting, the receiver is at maximum sensitivity and further rotation of the knob will not improve signal reception. Adjusting the RF SENS knob until there is no distinct background noise lowers receiver sensitivity and incoming signals may not be received. The background noise level varies with locality and propagation conditions, and small adjustments may be necessary to maintain the optimum sensitivity setting.

5. Transmit switch - Depress. Wait for the equipment to tune.

   A 1000 Hz tone will be heard until tuning is complete. Tuning may require 38 seconds.

   When the tone ceases, the transmitter is tuned. Adjust the interphone HF monitor switch volume.

   NOTE

   The HF radio should be retuned after takeoff to match the antenna inflight impedance condition.

   WARNING

   Rf energy from the HF radio during tuning or transmission has caused erroneous light and instrument indications.

__Emergency Operation__

If an overload exists in the power supply output, a protective circuit turns off the HF equipment. Attempt to restore normal operation as follows:

1. Function switch - OFF, then back to desired operating mode.

If the antenna coupler makes several consecutive tuning cycles, a thermal relay de-energizes the equipment. To restore operation:

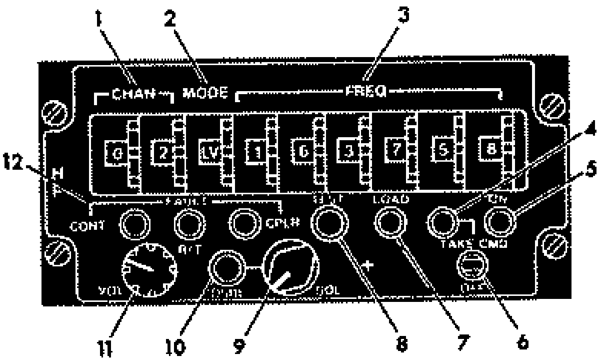
1. Function switch - OFF. After 2 minutes the thermal relay will cool.

2. Function switch - Set to desired operating mode.

__HF RADIO SYSTEM, AN/ARC-190 (V)__

The ARC-190 receives and transmits on 280,000 frequencies in a band from 2 to 29.9999 MHz spaced at 100 Hz. Frequencies below 4 MHz should not be used due to the nature of the antenna (comprising the pitot boom and the insulated forward portion of the aircraft nose). Modes of operation are upper and lower sideband, amplitude modulation and continuous wave. System components are: a receiver-transmitter (R/T) and antenna coupler located in the R bay;

HF CONTROL PANEL - ARC 190



1 CHANNEL SELECT THUMBWHEELS
2 MODE SELECT THUMBWHEEL
3 FREQUENCY SELECT THUMBWHEELS
4 TAKE COMMAND LIGHT
5 ON LIGHT
6 TAKE COMMAND/OFF SWITCH
7 LOAD PUSHBUTTON
8 TEST PUSHBUTTON
9 SQUELCH SWITCH
10 SQUELCH ENABLE/DISENABLE PUSHBUTTON
11 VOLUME SWITCH
12 FAULT INDICATOR LIGHTS

F20-312

Figure 1-72

antenna and antenna loading coil in the nose; and the HF control panel on the RSO's left console (Figure 1-72). Essential ac power is supplied through a circuit breaker in the C bay.

Automatic tuning occurs in both receive and transmit. Receive tuning requires less than 10 milliseconds. Transmit tuning requires 35 milliseconds on any of the 30 preset channels and 1 second on a manually selected frequency. The first transmission on a new frequency, even a momentary transmission, initiates the transmit tune cycle. A 1000-Hz audio tone is heard until tuning is complete.

Long-range voice HF communication is highly sensitive to hourly and seasonal variations in propagation conditions. For best results, frequency assignment planning should be based on HF propagation predictions.

Channel (CHAN) Switches

Two thumbwheel switches used to select 30 preset channels 00 through 29.

Mode Switch

An 8-position thumbwheel switch used to select modes of operation: LV, lower sideband voice; UV, upper sideband voice (must be set when selecting manual frequencies); LD, lower sideband data (not used); UD, upper sideband data (not used); CW, continuous wave; AM, amplitude modulation; P, preset (must be set when using the 30 preset channels); A, undefined (CONT FAULT will illuminate with A selected).

Frequency (FREQ) Switches

Six thumbwheel switches used to manually select frequencies.

ON Light

Illuminates when radio is on.

TAKE CMD/OFF Switch

Three position switch, spring-loaded to center position used to turn the radio on and off. Momentary TAKE CMD (forward) position turns the radio on. Momentary OFF (aft) position turns the radio off.

LOAD Switch

Pushbutton used to load preset channels in memory.

TEST Switch

Pushbutton used to initiate self-test. When pressed, a receive self-test cycle is initiated and all FAULT lights (3) illuminate. When released, all FAULT lights extinguish unless a component fault is present where indicated. A transmit self-test is initiated by depressing the transmit microphone switch (pilot or RSO with HF selected on interphone panel). A fault is indicated by illumination of the fault light(s).

NOTE

The transmit self-test can only be initiated after completion of a receive self-test.

FAULT Lights

CPLR - Indicates a coupler malfunction.

R/T - Indicates a receiver-transmitter malfunction or illuminates when an unloaded preset channel is selected with P selected in the MODE switch.

CONT - Indicates a control panel malfunction, FREQ switches set below 02.0000 MHz, or the MODE switch is set to A.

Squelch (SQL) Switch

Rotary switch that provides squelch threshold in 3 preset levels. Squelch is disabled in the fully counterclockwise position.

Disable (DSBL) Switch

Pushbutton that alternately enables and disenables the squelch (SQL) switch.

Volume (VOL) Switch

8-position rotary switch sets transmit-receive audio and transmit audio sidetone at 7 preset levels.

Normal Operation

1. Interphone rotary selector knob - HF.
2. Interphone HF monitor switch - Pull and rotate to approximately 12 o'clock position.
3. TAKE CMD/OFF switch -TAKE CMD.

   Momentarily position the switch to TAKE CMD to turn the radio on. The ON and TAKE CMD lights will illuminate.

4. MODE switch - Set.

   Set P for preset channel.

   Set UV for manually selected frequency.

5. CHAN or FREQ - Set.

   To use a preset channel, select the desired channel on the CHAN thumbwheels. If an unloaded preset channel is selected, the R/T FAULT light illuminates.

   To manually select a frequency, set the desired frequency on the FREQ thumbwheels. Do not use frequencies below 04.0000. If a frequency below 02.0000 is set, the CONT fault light illuminates.

6. VOL switch - Nearly full clockwise.

   Use the interphone panel controls for volume adjustments. If necessary, decrease the HF volume to maintain compatibility with the range of adjustments available in the interphone panel. The VOL switch also controls the transmit audio sidetone.

7. SQL and DSBL switches - Set.

   Rotate the squelch switch fully clockwise. If background noise is audible in the full clockwise position, depress the DSBL pushbutton to enable squelch. Rotate the SQL counterclockwise until background noise is audible. This position allows reception of any audible signal but with some continuous background noise. For normal operation rotate the switch one position more clockwise to set the threshold of reception just above the background noise.

8. TEST switch - Press.

   Check all three fault lights illuminate. When the switch is released, check no fault lights remain illuminated.

To transmit:

9. Microphone switch - Depress.

A 1000-Hz tone is heard during transmit tuning on the first transmission on a new frequency. Tuning is complete within one second.

WARNING

Rf energy from the HF radio during tuning or transmission has caused erroneous light and instrument indications.

To program a preset channel:

1. Mode switch - UV.
2. FREQ. - Set.
3. CHAN - Set.
4. LOAD pushbutton - Depress.

   The select channel will be programmed to the selected mode and frequency.

   Repeat as required for other channels.

To turn system off:

1. TAKE CMD/OFF switch - OFF.

   Momentarily position the switch to OFF. The ON and TAKE CMD lights will extinguish.

Emergency Operation

If the R/T FAULT light illuminates while receiving or transmitting:

1. Cycle a FREQ thumbwheel (if in UV mode) or a CHAN thumbwheel (if in P mode) at least three times.

Illumination of the R/T FAULT light when an unloaded preset channel is selected is normal.

If the R/T FAULT light remains on:

2. Attempt a short transmission.

CAUTION

If the ARC-190 HF transmits normally with the R/T FAULT light illuminated, use short transmissions sparingly to prevent overheat.

If transmission and/or reception is inoperative:

3. TAKE CMD/OFF switch - OFF.

   Attempt to restore normal operation by cooling the system at least two minutes. Reattempt normal operation as desired.

INSTRUMENT LANDING SYSTEM (ILS), 51RV-1

An ILS receiver supplies signals to the bank and pitch steering bars and glide slope indicator on the ADI, and to the course deviation indicator (CDI) on the HSI. Refer to Attitude-Director Indicator, Horizontal Situation Indicator, this section, and to Figure 1-66.

An invalid localizer or glide slope signal is indicated by a red warning flag appearing behind the bank steering bar or glide slope indicator, respectively.

WARNING

ILS reception can be affected by UHF transmission at high power settings.

The receiver operates on 20 frequencies. Localizer frequencies range from 108.1 to 111.9 MHz, and glide slope frequencies range from 329.3 to 335.0 MHz. The proper glide slope frequency is automatically tuned when the localizer frequency is selected.

ILS CONTROL PANEL

An ILS control panel (Figure 1-73), labeled NAV, is located on the pilot's right console. The panel contains concentric power (outer ring) and frequency (inner knob) switches. The power switch has two positions: OFF and PWR. The panel also has concentric volume control (outer ring) and frequency (inner knob) switches. Turning the VOL (outer ring) clockwise increases localizer identification volume. The TACAN/ILS interphone switch must be pulled and the DISPLAY MODE SEL switch must be in ILS or ILS APPROACH to hear the ILS identifier and/or marker beacon. A window, labeled MC, displays the localizer frequency (selected by rotating the inner knobs of the concentric switches). The ILS TEST buttons, labeled UP/L and DN/R, test the ILS system (excluding the antenna). The VOR button is not operable.

MARKER BEACON

A conventional 75 MHz ILS marker beacon receiver illuminates the amber MARKER BEACON light on the pilot's instrument panel and generates an audio signal when the aircraft is over an ILS marker. Power for the receiver is furnished by the essential dc bus.

NOTE

> The marker beacon antenna is located inside the right nosewheel door and is operational only when the nosewheel is down.

ILS Operation

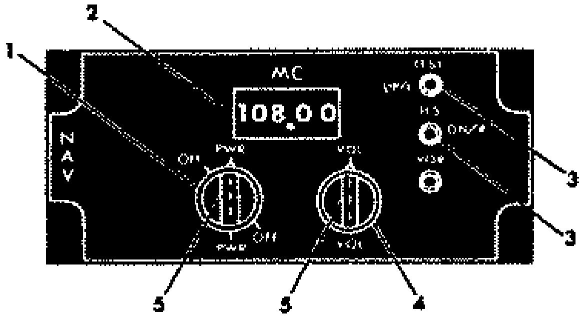
1. Interphone ILS monitor switch - Pull and set volume.
2. DISPLAY MODE SEL switch - ILS or ILS APPROACH.
3. Power switch - ON.

   Allow 90 seconds for warmup.
4. Volume - Nearly full clockwise.

To self-test the ILS:

1. Select any localizer frequency.
2. Align the HSI course arrow with the lubberline.

ILS CONTROL PANEL



1. POWER SWITCH
2. FREQUENCY WINDOW
3. TEST BUTTON
4. VOLUME CONTROL
5. FREQUENCY SELECTOR

Figure 1-73

3. DISPLAY MODE SEL switch - ILS APPROACH.
4. Press UP/L.

   The localizer and glide slope warning flags disappear, (the INS must be in NAV for the localizer warning flag to disappear), the glide slope indicator moves 1-dot up, the localizer moves 1-dot left, and the pitch and bank steering bars move half-scale up and left, respectively.
5. Press DN/R.

   The same actions occur as in step 4, but the directions are down and right.

AN/APX-108(V) IFF (W/SB R-2668) WILCOX 914X-2 IFF (W/O SB R-2668)

The IFF transponder responds to radar interrogation. The system includes altitude reporting, selective identification, and emergency reporting features. The Mode 4 function provides an encrypted IFF capability.

IFF CONTROL PANEL

The IFF control panel is located on the RSO's instrument panel. The controls for Mode 4 are on the left side of the IFF control panel, and the controls for Modes 1, 2, 3A and C are on the right side. See Figure 1-74.

Master Selector Switch

The rotary master selector switch has five positions:

OFF - Removes power. The switch must be pulled out before it can be rotated to OFF.

STBY - Standby. Applies power, but transmission is inhibited.

LOW - Only responds to strong (local) interrogations. Used at the request of a controller.

NORM - Normal operation. All modes selected have full sensitivity.

EMER - Emergency. Responds to interrogations in Modes 1, 2 and 3A. Mode 3A responds with code 7700. Mode C and 4 operate normally if selected. The switch must be pulled out before it can be rotated to EMER.

Mode 1, 2, 3A and C Controls

Mode 1, 2, 3A and C Control Switches

Four three-position toggle switches select Modes 1, 2, 3A and C. The ON (center) position places the corresponding mode in operation and the OUT (down) position disables the corresponding mode. The momentary TEST (up) position is used to test each mode. Continuous illumination of the TEST indicator light, while a mode control switch is held in TEST and the master selector switch is in NORM, indicates successful self-test of that mode.

Mode C responds with altitude information pulses in addition to the framing pulses (for tracking) sent in Mode 3A.

RAD-OUT-MON Switch

The three-position RAD-OUT-MON switch is provided for testing and monitoring the IFF system. The spring-loaded momentary RAD position is used with ground equipment for maintenance test. In the MON (monitoring) position, the TEST light will illuminate intermittently when responding to radar interrogations in the mode(s) selected. In the OUT position, the TEST light will only respond to self-test inputs using the mode control switches and will not indicate response to radar interrogation.

Mode 1 and 3A Code Selector Switches

Six thumb-wheel switches select the codes for Mode 1 and Mode 3A. The two switches on the left select Mode 1 codes and the other four switches select Mode 3A codes. The left Mode 1 switch is numbered 0 through 7, the other is numbered 0 through 3 on each half of the drum. The Mode 3A switches are numbered 0 through 7.

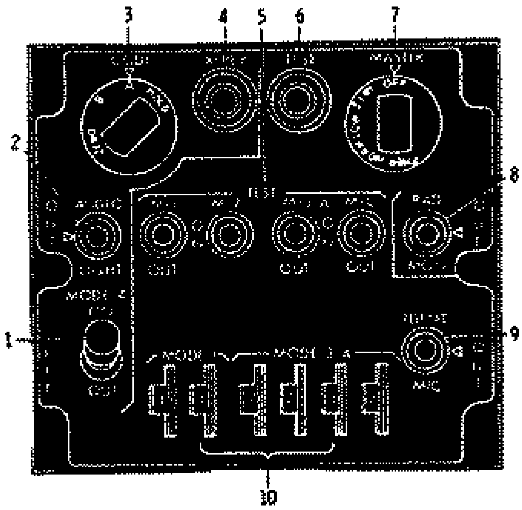
Identification-of-Position Switch

The three position identification-of-position (I/P) switch controls transmission of I/P pulse groups. The I/P timer is energized for thirty seconds when the switch is momentarily held in spring-loaded IDENT,and I/P replies will be made if a Mode 1, 2, or 3A interrogation is recognized within thirty seconds. The I/P pulse group is not transmitted when the I/P switch is in OUT. The MIC position is inactive.

Test Indicator Light

A rotate-to-dim, press-to-test, green TEST light indicates satisfactory operation of the transponder. With the master selector switch

IFF CONTROL PANEL



1. Mode 4 On-Out Switch
2. Mode 4 Indication Switch
3. Mode 4 Code Select Switch
4. Mode 4 Reply Light
5. Mode Control Switches
6. Test Indicator Light
7. Master Selector Switch
8. Rad-Out-Mon Switch
9. Identification-of-Position Switch
10. Code Selector Switches

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Figure 1-74

Mode C Altitude Reporting and Identification Capability Schedule

Mode Selector Switch Settings		Response	
Mode 3A	Mode C	Mode 3A	Mode C
ON	ON	Normal	Normal (Altitude Code)
ON	OUT	Normal	*Framing Pulses Only
OUT	ON	None	Normal (Altitude Code)
OUT	OUT	None	None

*Response indicates aircraft with Mode C altitude reporting capability to interrogating station.

Notes:

Mode 3A selector switch enables Mode 3A and Mode C decoder.

Mode C selector switch enables Mode C decoder and information pulses.

Mode C altitude reporting information is enabled by the DAFICS M computer. If the M computer fails, Mode C will continue to report the altitude at the time the computer failure occurred.

Figure 1-75

in NORM, the light illuminates when a mode switch (1, 2, 3A, or C) is placed in TEST if the self-test is satisfactory. When the system is operating (master selector switch in LOW or NORM), and the RAD-OUT-MON switch is in MON, the light blinks when the IFF responds to interrogation in the mode(s) selected.

Mode 4 Controls

Mode 4 is also controlled by the master selector switch.

Mode 4 Code Select Switch

The Mode 4 code select switch is labeled ZERO, B, A, and HOLD. Placing the switch in ZERO cancels (zeroizes) both the code settings. The switch must be pulled out before it can be rotated to ZERO. When the switch is in A or B, the transponder will respond to Mode 4 interrogation sources using the same code. The spring-loaded HOLD position is used (before power is removed from the transponder) to retain codes when the aircraft is on the ground.

NOTE

Weight must be on the nosegear before the HOLD position is functional.

To retain code settings, the switch must be held in HOLD for at least 5 seconds and transponder power must be left on for an additional 15 seconds. Otherwise, the code settings may not be mechanically latched and will zeroize when the master selector switch is turned off or power is disconnected.

Mode 4 Reply Light

A rotate-to-dim, press-to-test, green REPLY light illuminates when the Mode 4 indication switch is in either AUDIO or LIGHT, if transponder Mode 4 replies are satisfactory. Press-to-test is only operative in the AUDIO or LIGHT position.

Mode 4 Indication Switch

A three-position AUDIO-OUT-LIGHT toggle switch controls Mode 4 indications.

W/O SB R-2668, with the indication switch in AUDIO (up), an audio tone is heard in the aft cockpit only if a proper Mode 4 interrogation is received, and the REPLY light illuminates when Mode 4 generates a reply.

With SB R-2668, with the indication switch in AUDIO (up) an audio tone is heard in the aft cockpit if Mode 4 does not respond to a proper interrogation and the REPLY light will not illuminate.

NOTE

The RSO's IFR COM knob controls the volume of the Mode 4 audio.

In LIGHT (down), the REPLY light illuminates without audio when Mode 4 replies are transmitted. In OUT (center), both light and audio indications are inoperative.

Mode 4 On-Out Switch

A two-position toggle switch controls Mode 4 operation. A Mode 4 response cannot be transmitted unless: ON (up) is selected; the Master Selector Switch is in LOW, NORM, or EMER; and the Mode 4 code has not been zeroized. The switch must be pulled out to be moved to OUT (down).

Mode 4 IFF Caution Light

A rotate-to-dim, amber IFF CAUTION light, is located on the RSO instrument panel. The light illuminates each time the transponder fails to reply to a proper interrogation. If the Mode 4 codes are zeroized, or if self-test detects a system fault, the light will be on steady. Pressing the LAMP TEST switch checks the light .

IFF NORMAL OPERATION

Modes 1, 2, 3A and C

1. Master Selector Switch - STBY.

   Three minutes are required for warm-up.

2. Master Selector Switch - NORM.

   LOW should only be used at the request of a controller.

3. Mode 1 and 3A Code Selector Switches - Set.

NOTE

The Mode 2 Code is preset on the ground.

4. Mode Switch(es) - As required.

To test operation of individual Modes:

1. RAD-OUT-MON switch -OUT.

   If the switch is in MON, the TEST light will illuminate for both self-test and monitor functions.

2. Individual mode switch - TEST.

   Illumination of the TEST light indicates the corresponding Mode is operational. Repeat for each mode.

NOTE

Test modes individually.

3. Mode switches - As required.

To monitor operation of individual Modes:

1. RAD-OUT-MON switch - MON.
2. Individual Mode switch - ON.

   The TEST light will blink to indicate responses to interrogation. Repeat for other modes.

To transmit identification of position:

1. Identification switch - IDENT.

__Mode 4__

1. Mode 4 ON-OUT switch - ON.
2. Code select switch - A or B, as required.

   Code A is the code for a prescribed 24-hour period; Code B is for the next 24-hour period.

3. Audio/Light indicator switch - As desired.

To zeroize the codes:

1. Mode 4 code select switch - ZERO.

   The codes can also be zeroized by turning the master selector switch OFF, if the HOLD function has not been used.

NOTE

Once zeroized, Mode 4 is inoperative until the codes are reinserted on the ground.

To retain the codes after landing:

1. Mode 4 code select switch - HOLD.

   Place the switch in the HOLD position for 5 seconds, then wait another 15 seconds before turning equipment OFF.

To turn IFF off:

1. Master selector switch - OFF.

IFF EMERGENCY OPERATION

__Modes 1, 2, and 3A__

1. Master selector switch - EMER.

NOTE

In EMER, Mode 4 and Mode C replies are normal.

__Mode 4__

Illumination of the IFF CAUTION light indicates the transponder will not respond to Mode 4 interrogations.

With IFF CAUTION light illuminated:

1. Master selector switch - Check NORM.

2. Mode 4 ON-OUT switch - Check ON.

3. Mode 4 code select switch - Check.

   Check A or B.

G-BAND BEACON

The G-band (formerly C-band) beacon is a radio frequency transponder used to aid radar tracking of aircraft during special tests. The beacon responds to interrogation in the 5400 to 5900 MHz range. The transmitter radiates with at least 400 watts peak power at 500 pps. Receive and transmit frequencies are displaced at least 50 MHz for protection of the beacon receiver. Transmitter operation can not be adjusted while in-flight.

The beacon is controlled by two two-condition push-button switches located left of the viewsight controls on the RSO's instrument panel. Power is controlled by the BEACON switch. Actuating the switch illuminates an ON legend in the lower half of the switch. A warm-up period of at least three minutes is desired. A 15-minute warm-up period results in maximum signal stability. Another switch actuation turns the beacon off and extinguishes the ON legend. Two antennas are provided. Actuating the ANT (antenna) switch illuminates either a B (bottom antenna) or T (top antenna) legend in the lower quarters of the switch when the beacon is on.

Power for the beacon is from the Essential DC Bus through the BEACON circuit breaker on the RSO's right console.

I-BAND BEACON

The I-band beacon is provided when Mission Kit 4AT1030 is installed. The antenna can be installed in the left EIP door, right EIP door, left TECH door, or right TECH door.

The I-band beacon is a radio frequency transponder used to aid radar tracking of aircraft during special tests. The beacon responds to interrogation in the 8500 to 9600 MHz range. Receive and transmit frequencies are displaced at least 50 MHz for protection of the beacon receiver. Transmitter operation can not be adjusted while in flight.

The beacon is controlled by the OOC LH OPER power switch if the antenna is installed on the left EIP door or the left TECH door or by the OOC RH OPER power switch if the antenna is installed on the right EIP door or the right TECH door. Actuating the switch illuminates the ON legend in the lower half of the switch. System warm-up is from 20 seconds to two minutes. Another switch actuation turns the beacon off and extinguishes the ON legend. (The FAIL light is not functional and will not illuminate.)

Power for operation of the beacon is provided from the Monitored DC Bus through the left or right OOC CONT AND PWR circuit breaker in the C-bay.

TACAN SYSTEM AN/ARN 118(V)

The Tactical Air Navigation (TACAN) system operates with ground stations and cooperating aircraft. Continuous slant range and bearing information is obtained from ground stations. Range and bearing are also obtained through mutual transponding with cooperating aircraft; however, bearing information can not be transmitted by the SR-71 to the other aircraft. Operational range can be over 300 nautical miles at high altitudes.

126 X-mode and 126 Y-mode channels are available.

Transmitter interrogation frequencies range from 1025 to 1150 MHz with 1-MHz separation. Receiver frequencies range from 962 to 1024 MHz and 1151 to 1213 MHz when operating in the air-to-ground (T/R) mode.

NOTE

> TACAN reception can be affected by UHF transmissions at high power settings in external mode.

In the air-to-air mode, receiver frequencies used are the same as for transmitter interrogation; however, a pair of channels with 63 MHz separation is required. Since the air-to-air modes can operate at or near the IFF system frequencies (1090 and 1030 MHz for transmit and receive), IFF interference can cause unreliable TACAN operation on channels 1-11, 58-74, and 121-126.

The IFF receiver is suppressed during TACAN transmission and the TACAN receiver is suppressed during IFF transmissions, to protect the receivers. If no signal is received when a strong TACAN or IFF signal is expected, a malfunction of the suppression circuits can be identified by momentarily turning off one of the systems.

When TACAN channels are changed, acquisition time for the new channel is less than one second. No more than three seconds are required for bearing lock-on.

If the received TACAN distance signal is invalid, the distance displays on the HSI and BDHI are covered. If the DISPLAY MODE SEL switch is in TACAN, the steering bar warning flag appears if TACAN bearing is not valid. The identifier of the interfering station is purposely garbled if co-channel interference occurs in T/R. If signal loss occurs, velocity memories keep the bearing and range indications tracking for up to 3 and 15 seconds, respectively, or until the signal is reacquired. An automatic self-test is performed after each signal loss and the TEST light illuminates if the test fails.

Power for TACAN operation is provided by the essential ac bus and by the monitored dc bus through circuit breakers in the C-bay.

The TACAN antenna is a high temperature annular slot type (identical to the IFF antenna) located on the bottom centerline of the fuselage forward of the nosewheel well.

TACAN CONTROL PANEL

A TACAN control panel and a TACAN control transfer switch are provided in each cockpit (Figure 1-76).

Channel Selectors

Two channel selector knobs and an X-Y mode selector, control TACAN frequency. The channel number (01 through 126), selected by rotating the tens and units channel selector knobs, is displayed on the panel. Rotating the ring which surrounds the base of the units channel selector knob selects X or Y mode.

Mode Selector Switch

The rotary mode selector switch has five positions:

OFF - Removes power.

REC - Bearing to selected ground station. Range is not available.

NOTE

- After tuning a new station, the bearing pointer may slew to a bearing 90° greater than the actual bearing and remain there for about two seconds. This is normal.
- After signal loss and re-acquisition, the bearing pointer will slew to 270° and remain there for about seven seconds during the automatic self-test. After self-test, the bearing pointer may slew to a bearing 90° greater than the actual bearing and remain there for about two seconds.

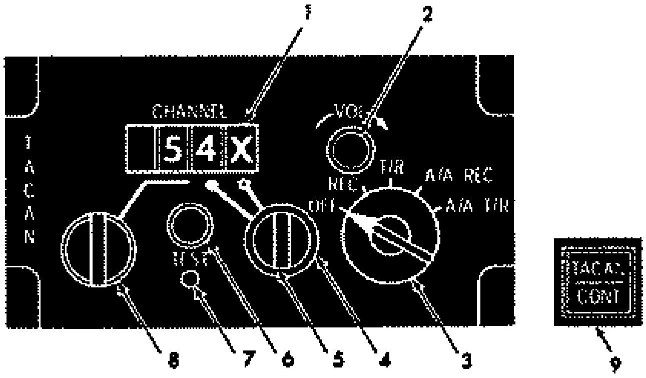
T/R - Bearing and slant range to a selected ground station.

A/A REC - Bearing to a suitably equipped cooperating aircraft (also in A/A with 63 channel separation). Range is not available.

A/A T/R - Bearing and slant range to a cooperating aircraft.

AN/ARN-118(V) TACAN CONTROLS

TACAN CONTROL PANEL
PILOT RIGHT CONSOLE
RSO LEFT INSTRUMENT PANEL



1 CHANNEL DIGITAL DISPLAY
2 STATION IDENTIFICATION VOLUME CONTROL
3 MODE SELECTOR CONTROL
4 X-Y CHANNEL SELECTOR CONTROL
5 UNITS CHANNEL SELECTOR CONTROL
6 R/T STATUS INDICATOR LIGHT
7 TEST SWITCH
8 TENS CHANNEL SELECTOR CONTROL
9 TACAN CONTROL TRANSFER SWITCH - PILOT'S AND RSO'S INSTR PANELS

F200-201(b)

Figure 1-76

NOTE

Some aircraft, including the SR-71, cannot transmit bearing. Range is available to both aircraft.

Volume Control Knob

Rotating the volume (VOL) control clockwise increases the TACAN identification signal audio level. The TACAN/ILS interphone switch must be pulled and the DISPLAY MODE SEL switch must be in ANS, INS, or TACAN ADF to hear the TACAN identifier.

Self-Test Push-button

A self-test of the TACAN equipment and its interface with the HSI and BDHI is initiated by actuating the TEST push-button. The adjacent indicator light flashes momentarily when the TEST switch is pressed.

NOTE

If the TEST light illuminates other than at the start of self-test, a TACAN fault exists.

A self-test can be terminated immediately by rotating a channel selector or the mode selector switch to another position.

NOTE

- Bearing and/or distance indications may still be present when the TEST light is on. Such indications may be accurate, but are unreliable.
- Be prepared for failure of TACAN equipment if the TEST light illuminates.

Small recesses on the outside of each side of the canopies provide lifting points so the canopies can be opened from outside. A latch on top of the nitrogen counterbalance cylinder engages when the canopy is fully open and holds the canopy in that position until the latch is released by pressing the latch release lever. The two canopies are independent in operation, except for the external jettison feature.

CAUTION

- Canopies shall be opened or closed only when the aircraft is stopped. To prevent wind forces from shearing the canopy hinge pins, hold canopy securely when opening.
- Maximum taxi speed with the canopy open is 40 knots. Gusts should be included as part of the 40 knot limit.

Canopy Latching Mechanism

Each canopy is latched closed by four hooks (two in each canopy sill). The canopy is latched and unlatched by a handle at the forward right side of each cockpit. Moving the handle rotates a transverse torque tube behind the seat to simultaneously position all four hooks. With the canopy closed, forward movement of the handle latches and locks the canopy, while aft movement releases the hooks and unlocks the canopy. The ejection or canopy jettisoning sequence unlatches the canopy by gas pressure from the canopy unlatch thruster behind the seat.

The aft canopy latching torque tube is mechanically connected to a cockpit air shutoff valve in the air-conditioning system. To conserve cooling air for electronic equipment, this valve shuts off air flow to both cockpits when the canopy latch handle is in the aft position. This valve is also operated by the RSO's Cockpit Air handle. The forward (off) position of the RSO's Cockpit Air handle will shut off air to both cockpits even if the aft canopy latch handle is forward. Refer to Environmental Control System Controls, Cockpit Air Valve, this section.

CAUTION

If either canopy is open, the aft canopy latch handle must be in the aft position or the cockpit air handle must be in the forward (off) position for adequate equipment cooling. Otherwise, most of the cooling air would exit through the cockpit openings instead of the bays.

Canopy Counterbalance System

Normal opening and closing of the canopy is assisted by gaseous nitrogen pressure from the canopy counterbalance cylinder at the left rear of the seat. Without counterbalance assistance, the canopy is difficult to raise (requiring approximately 112 pounds of force), and can drop with sufficient force to injure personnel. A counterbalance system pressure gage is located above the nitrogen cylinder and the pointer should indicate in the green if system pressure is normal.

WARNING

To avoid injury, verify that counterbalance pressure indication is normal before pushing the canopy latch release.

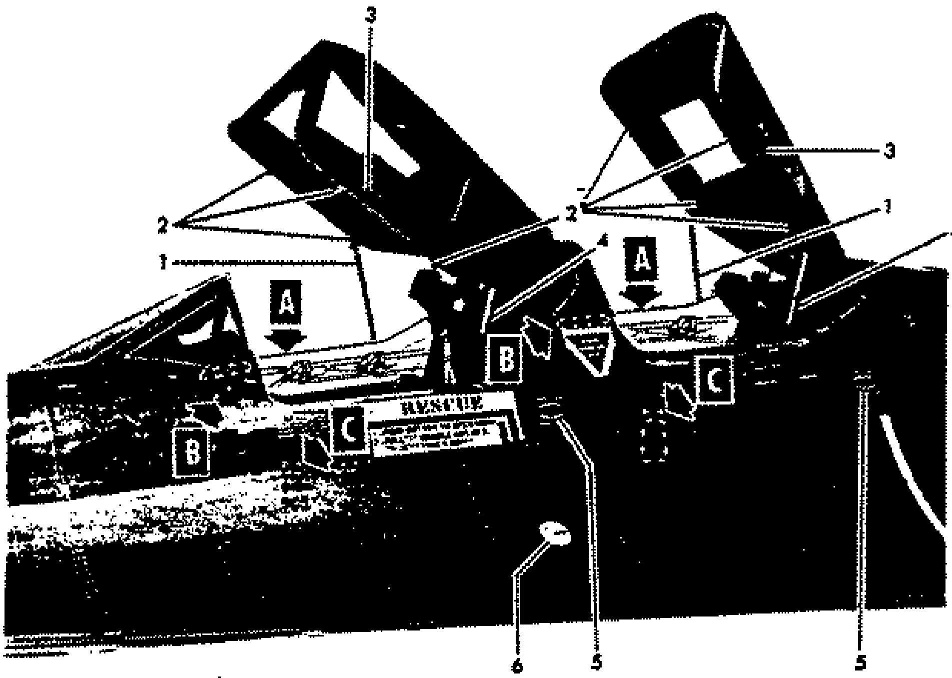
CAUTION

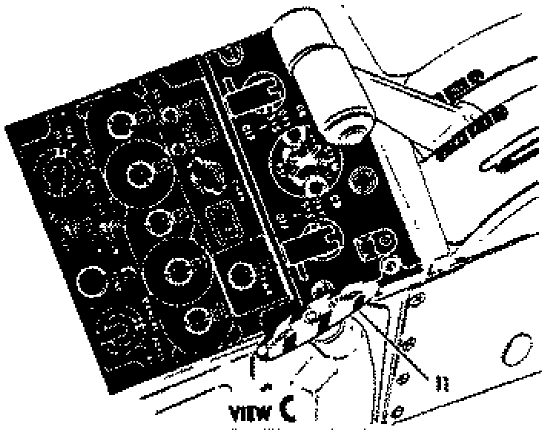
Do not rotate the canopy upward beyond its normal full-open position or canopy shear pins may be severed.

Canopy Seal System

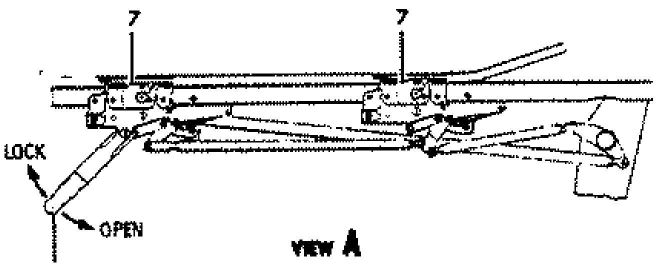
An inflatable seal is installed along the edge of each canopy frame. The seal is inflated by engine bleed pressure to provide an airtight seal between the canopy and the canopy sills and windshield. A pressure regulator and control valve is provided in each cockpit. The seal selector lever is located in the right forward corner of each cockpit and is used to inflate or deflate the seals. Seal pressure is regulated to no more than 23 psi.

CANOPIES, CANOPY CONTROLS and INDICATOR

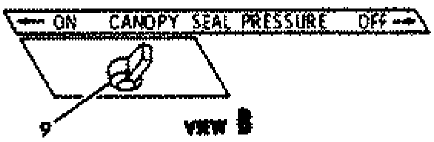




VIEW C
PILOT'S LEFT CONSOLE
(SEE FIGURE 1-18 FOR RSO LEFT CONSOLE)

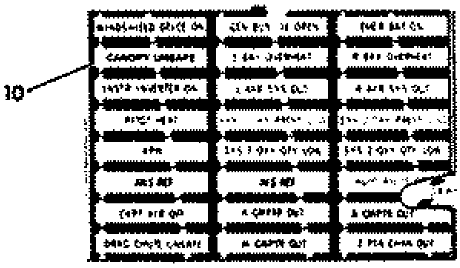


VIEW A
RIGHT SILLS



VIEW B
RIGHT SIDE PANELS

1 CANOPY PROP (GROUND HANDLING)
2 CANOPY LATCH ROLLER BRACKETS
3 CANOPY LIFTING HOLES
4 CANOPY COUNTERBALANCE AND UPLOCK
5 CANOPY EXTERNAL LATCH CONTROLS
6 CANOPIES EXTERNAL JETTISON HANDLE (HIDDEN)
7 CANOPY LATCH HOOKS (BOTH COCKPITS)
8 CANOPY LATCH HANDLES (BOTH COCKPITS)
9 CANOPY SEAL CONTROLS (BOTH COCKPITS)
10 CANOPY UNSAFE CAUTION LIGHT (PILOT'S INSTRUMENT PANEL)
11 PILOT'S CANOPY INTERNAL JETTISON HANDLE
(SEE FIGURE 1-18 FOR RSO CANOPY INTERNAL JETTISON HANDLE)



ANNUNCIATOR PANEL

F203-59(a)

Figure 1-77

REAR VIEW PERISCOPE

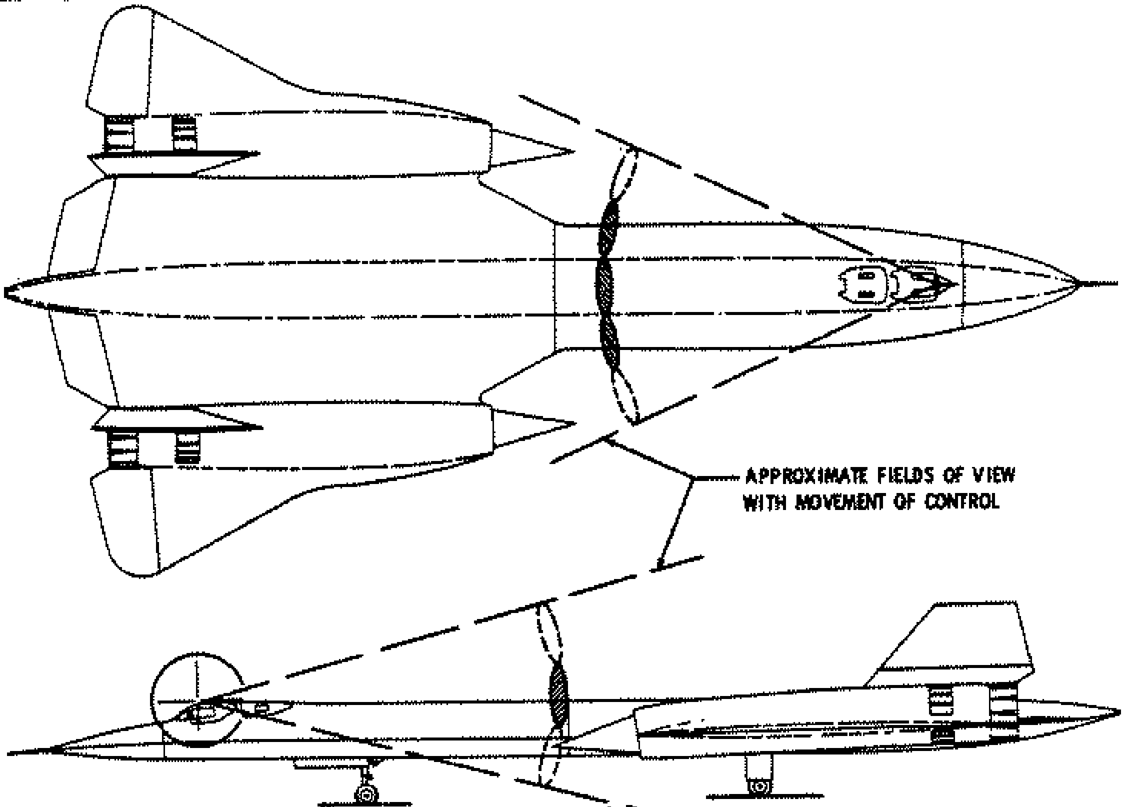


Figure 1-78

F205-112(b)

> **CAUTION**
>
> - Do not inflate seals unless the canopy is latched closed, or damage to seals may occur.
> - The OBC must be off unless both canopies are closed and the canopy seals are on.

Canopy Unsafe Warning Light

Illumination of the CANOPY UNSAFE annunciator caution light indicates that one or both canopies is not latched down and/or properly sealed.

Canopy Jettsion System

The canopy is jettisoned by gas pressure from a canopy unlatch thruster on the aft bulkhead of the respective cockpit. The unlatch thruster may be fired by pulling the ejection seat D-ring, or the CANOPY JETTISON T-handle located outboard of the left forward corner of each ejection seat.

When the unlatch thruster fires, it rotates the canopy hooks to the unlatch position and releases the canopy. The thruster charge is then ported to the canopy-removal thruster and seal-hose cutter. The seal-hose cutter severs the canopy seal hose, and the canopy-removal thruster forces the canopy up and aft, shearing the hinge attach points and thrusting the canopy clear of the aircraft.

The canopy can also be removed in-flight by manually unlatching it and pushing it up into the airstream. If the cockpit is pressurized when the canopy latches are released, the canopy will be blown upward into the airstream by cockpit pressure.

> **WARNING**
>
> Do not enter or leave the cockpit unless ground safety pins are installed in the seat ejection D-ring, the seat ejection T-handle, and the canopy jèttison T-handle.

Canopy External Latch Controls

Individual external latch release fittings are flush-mounted on left side of the fuselage, just below the canopy hinge points. Fittings accept a 1/2-inch-square bar extension to open the canopies from outside the aircraft. The canopy must be raised manually after releasing the latch hooks externally.

Canopy External Jettison System

Both canopies may be jettisoned on the ground by the external jettison system. An external jettison handle is located under an access panel on top of the left chine, at the aft end of the forward cockpit. When the external jettison handle is pulled, both canopies jettison in sequence, the forward canopy first, the aft canopy 1 second later. A long jettison cable allows the person pulling the handle to be well clear of the fuselage during jettison.

Cockpit Sunshades (Bat Wings)

Two 12.5" x 8" sunshades (bat wings) in the front cockpit are used to block the intense sunlight prevalent at high altitude, and to reduce cockpit light reflections at night. A sunshade is located on each canopy side-frame, below the side windows, and is attached to a rod mounted 5" aft of the forward end of the canopy. Each sunshade is individually adjustable: the sunshade can be extended along the rod, the rod elevated, and the sunshade rotated. They may be joined together to form an extension of the glare shield (refer to Night Flying, Interior Light Reflections, Section VII). A lever to adjust rotational friction is located on the top, inboard portion of the left bat wing; similar levers at both rod attachment points adjust elevation friction and allows the bat wings to be joined together. The right bat wing may be folded lengthwise and stowed above the PVD projector. Both bat wings have extension panels, located on the lower side, which increase the area by 50% when pulled out. The altimeter correction card is attached to the outboard side of the right bat wing.

REAR-VIEW PERISCOPE

A manually extended rear-view periscope is mounted in the top of the pilot's canopy. It is moved by using a white nylon handle mounted on the aft side of the viewing tube. Pushing the handle left unlocks the tube, allowing the periscope to be extended. Pushing the tube upward to a spring-detented position makes the rear view available. Cockpit pressure assists extension and resists retraction. The cone of view is approximately 10 degrees across; however, head movement extends the viewing cone to approximately 30 degrees total angle. (See Figure 1-78.) When extended, the periscope can be rotated horizontally to move the center of the viewing arc up to 10 degrees from the aft centerline. The lens provides a 2 to 1 reduction ratio.

MAP PROJECTORS

A strip-film projector is installed in each cockpit to provide a strip map of the route to be flown with mission data and emergency information. (See Figure 1-79.) Film strips are provided for each mission. The pilot and RSO projectors are of different design and are independently controlled.

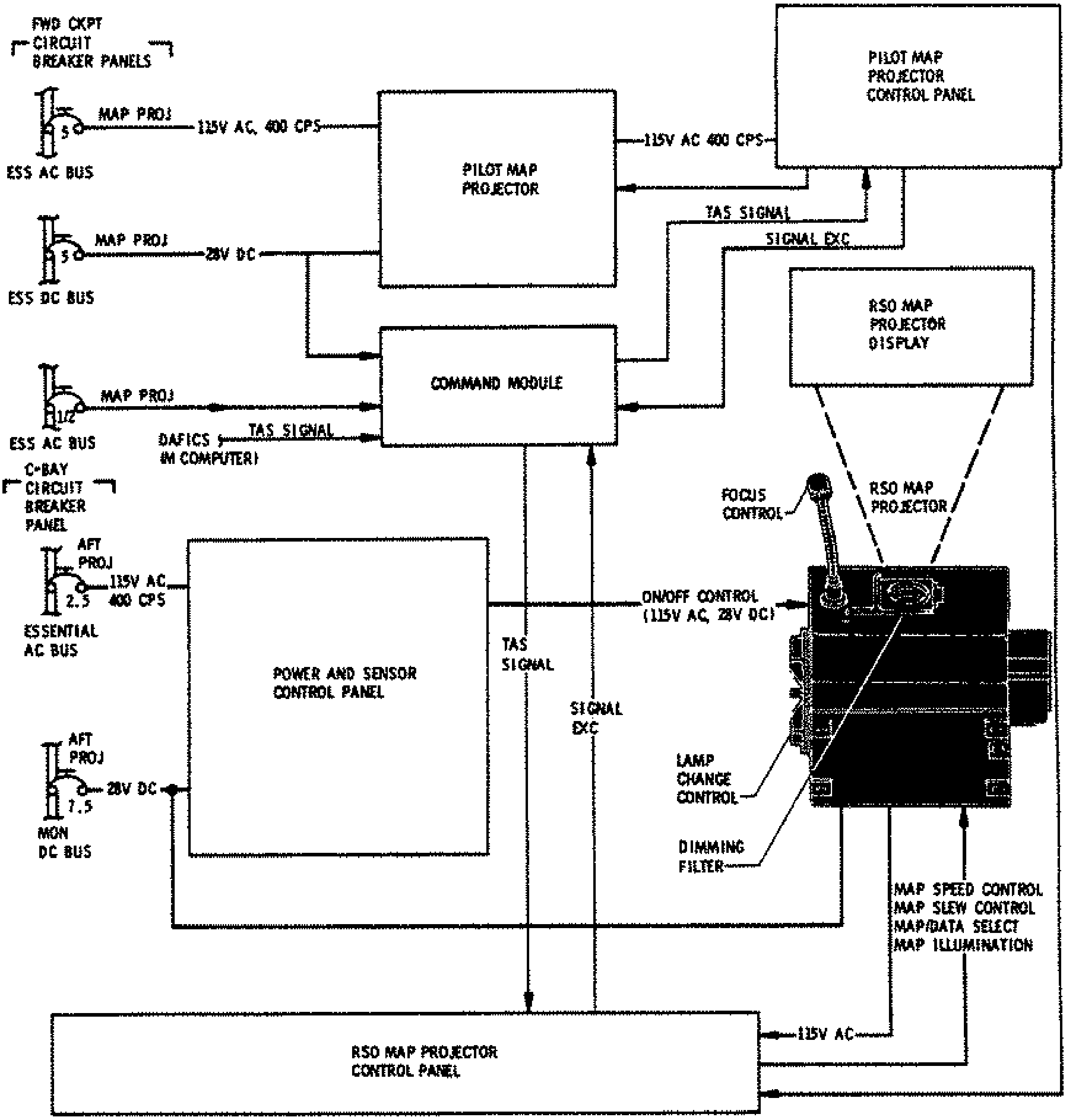
PILOT'S MAP PROJECTOR

The pilot's map projector is located in the bottom center of the instrument panel. Projector controls are located on the panel borders of the 4-1/4 by 4-1/4 inch display and on the map projector control panel on the left console, forward of the throttle quadrant. The projector can display up to 25 feet of 35 mm film.

Illumination Control

The illumination control slide switch at the bottom edge of the navigation map display is a combination power on-off switch and illumination rheostat. When the control is in the left-hand detent, the projector is deenergized; moving the control down and sliding it to the right energizes the projector lamp, blower, and film-drive motors. Moving the

MAP PROJECTORS



F203-24(2)(a)

Figure 1-79

control from DIM towards BRT increases the display brightness.

Projector Lamp DIM-BRT Switch

A three-position projector lamp dimming toggle switch is provided on the bottom edge of the navigation map display, to the right of the illumination control. In DIM, low voltage is applied to the navigation map projector lamp and the range of image brightness is reduced. In BRT the maximum range of image brightness is available. The unmarked middle position provides medium brightness.

Lamp Change Control

The lamp change control on the right border of the nav map display is labeled LAMP CHG. The lever switch has detent positions (labeled 1, 2, and 3) to provide a choice of three bulbs. If a bulb burns out, another may be selected by moving the lever to the right, out of detent, and repositioning it to one of the other two marked positions.

Slew Control Switch

The rotary slew control switch on the map projector control panel, is spring-loaded to the center (off) position. Counterclockwise rotation of the switch causes reverse slew and clockwise rotation causes forward slew of the film strip. The rate of film movement is variable in both directions. Increased rotation of the switch increases the slew speed.

NOTE

> To return to a specific position, reverse slew, then slew forward and stop on the point of interest. This eliminates any delay in automatic film drive advance.

Drive Switch

The drive switch on the map projector control panel, selects either manual rate control (MANUAL MACH), automatic map rate synchronized to true airspeed (AUTO), or OFF (film stop).

Map Rate Control and Indicator

The map rate control on the map projector control panel, is a single-turn potentiometer calibrated in Mach. This control permits manual synchronization between map film speed and aircraft groundspeed when the drive switch is in the MANUAL MACH position. The control has a dial face with seven major outer divisions, representing Mach, and a single pointer. The inner O marking with + and - arrows provides a means of biasing the map rate when the drive switch is in AUTO. For normal operation the pointer should be in the O position to provide zero bias. Counterclockwise rotation from the O position decreases map film speed; clockwise rotation increases map film speed. The control has no effect when the drive switch is in OFF. The projector is intended for use with map films scaled at 365 nautical miles per inch.

CAUTION

> Attempting to turn the control clockwise past 35 or counterclockwise past zero will damage internal stops and cause a loss in dial reference.

PILOT'S MAP PROJECTOR OPERATION

1. Illumination control - Slide right to turn on and obtain desired brilliance.
2. Map DRIVE switch - AUTO or MANUAL MACH, as desired.
3. RATE control - O position for AUTO or as required for MANUAL MACH.

   Adjust + or -, as necessary.
4. SLEW control - REV or FWD, as required.

RSO's MAP PROJECTOR

The RSO's map projector system includes the projector, projector controls, and a 9 by 9-inch display screen. Some map projector controls are located on the viewsight control panel (Figure 4-33 without S/B R-2538, Figure 4-33A with S/B R-2538). When the RCD is installed, the map projector is mounted on the floor and projects onto a hinged viewing screen below the RCD. When the RCD is not installed, the projector is mounted horizonally at a higher position on the bottom of the instrument panel, and projection is then onto the display screen mounted in place of the RCD.

NOTE

> The map film continues to move when data film is selected for viewing, even though the film is not projected. The rate of movement always corresponds to 250 nautical miles per inch.

Projector Power Switch

The projector power switch is located on the PWR & SENSOR control panel. A white illuminated legend PROJ appears in the top half of the switch. A green ON legend, that illuminates alternately when the pushbutton is depressed, and a red non-functional FAIL legend occupy the lower quarters of the switch.

Lamp Change Control

A rotary lamp change control is located on the side of the projector case. The control has three detented positions for different bulb selections. When the RCD is installed, the control is located on the right side of the projector case and may be operated by the RSO's foot when a bulb burns out. When the RCD is not installed, the control is located on the left side of the projector case and can be reached by hand.

Projector Focus Control

Map projector focus is adjustable by turning the focus control knob flexible stem on top of the map projector assembly.

Map Drive Switch

A map drive switch on the viewsight control panel provides selection of either manual map rate control (MAN-TAS), or automatic map rate (AUTO) synchronized to true airspeed.

Map Rate Control and Indicator

The map rate control on the viewsight control panel is a two-turn potentiometer, calibrated to true airspeed, which permits synchronization between map film speed and aircraft groundspeed when the map DRIVE switch is in MAN - TAS. The control has a dial face with 10 peripheral divisions labeled 1 through 0 (10), a long and a short pointer, and a movable knurled bezel to move the pointers. The short pointer indicates thousands of knots and the long pointer indicates hundreds of knots. The inner O marking with + and - arrows provides a reference to bias the map rate when the DRIVE switch is in AUTO. For normal automatic operation both the short and long pointers should be on the O inner scale position (1 on outer scale) to provide zero bias. Counterclockwise rotation decrases map film speed; clockwise rotation increases map film speed.

Slew Control Switch

A rotary control switch, labeled SLEW, is located on the viewsight control panel. It controls the direction and speed of the film strip that has been selected by the film-select switch. Counterclockwise rotation of the switch toward REV (REW with video viewsight) causes reverse slew and clockwise rotation toward FWD causes forward slew. Increasing switch rotation causes an increasing rate of film movement in the indicated direction up to 0.6 ips for map film and 0.9 ips for data film. At the end of switch travel an electrical contact is made to cause rapid slewing of 3.5 ips for map film and 6 ips for data film. (Image speed across projector screen is nine times the film speeds.)

NOTE

> To return to a specific position, reverse slew, then slew forward and stop on the point of interest. This eliminates any delay in automatic film drive advance.

Film Select Switch

A two-position film-select toggle switch, labeled SELECT, is located on the viewsight control panel. The DATA and MAP positions select the type of film to be projected.

Illumination Control Switch

A three-position illumination control switch, labeled ILLUM, is located underneath the right end of the optical viewsight control panel and on the front of the video viewsight panel. The switch is springloaded to the center off position. A motor closes or opens an iris in the projector to change the degree of illumination; when held in the forward (DIM) position or the aft (BRT) position on the optical viewsight panel, or in the down (DIM) position or up (BRT) position on the video viewsight panel.

Dimming Filter

A manually positioned dimming filter can be placed over the projector lens to reduce brightness during night operation.

RSO MAP PROJECTOR OPERATION

1. PROJ power switch on PWR & SENSOR panel - Press to illuminate ON.
2. ILLUM switch - BRT or DIM, as desired.
3. SELECT switch - DATA or MAP, as desired.
4. DRIVE switch - AUTO or MAN-TAS as desired.
5. MAP rate knob - as desired.

If no display:

1. ILLUM switch - BRT as required.
2. Lens dimming filter - Check open.
3. Projector slew control - Slew to assure dark leader is not blocking light.
4. Lamp change control - Rotate to new position.

If image is out of focus:

1. Focus adjustment - Set.

LIGHTING EQUIPMENT

EXTERIOR LIGHTING

Landing/Taxi lights and Switch

A 1000 watt landing light and a 450 watt taxi light are mounted on the nose gear strut. The lights are controlled by a luminous three-position landing and taxi lights switch on the pilot's left instrument side panel. The switch positions are LAND LT (up), TAXI LT (down), and OFF (center). The switch is ineffective when the nose gear is retracted, as the uplock switch prevents power being applied to the lights. Power for the lights is obtained from the essential ac bus through the LDG LT and TAXI LT circuit breakers on the pilot's right console.

CAUTION

Use the taxi light if lighting is required during ground operation or after landing. The landing light burns out without airstream cooling.

Anti-Collision/Fuselage Lights and Switch

Two combination retractable anti-collision and fuselage lights are located at the top and bottom of the fuselage, near the middle. The lights are controlled by a three-position toggle switch on the pilot's lighting panel. The switch positions are: ANTI COLLISION (forward), FUS (aft), and OFF (center). In ANTI COLLISION, the lights extend, illuminate red, and rotate at 45 rpm (which produces 90 flashes per minute). In FUS (fuselage), the lights are retracted and illuminate white. The lights are retracted and off when the switch is in OFF. Three-phase essential ac powers the lights through the ANTI COLL LTS circuit breaker on the pilot's right console.

Tail Lights and Switch

A white tail light is located on the top and bottom of the tail cone. The lights are controlled by a three-position toggle switch, labeled TAIL LT, on the pilot's lighting panel. In STEADY (aft), the lights illuminate continuously. In FLASH (forward), the lights flash 85 times per minute. The center position is off. Power is furnished by the essential ac bus through the TAIL light circuit breaker on the pilot's right console.

Fuselage and Tail Lights Intensity Switch

A FUS & TAIL lights intensity switch is located on the pilot's lighting panel. The positions are BRT (forward) and DIM (aft). The switch controls the light intensity of the tail lights and also the anti-collision lights when they are operated as fuselage lights.

FORWARD COCKPIT LIGHTING

Integral and edge lighting for the instruments and consoles is controlled by the lighting panel on the left console. The four instrument lights circuit breakers on the lighting panel (LH, RH, FUEL CONT, and FUEL CONT TEST) are only operative if the ac hot bus INSTR light circuit breaker on the right console is closed. Front cockpit interior lighting also includes: one floodlight located above each side console, and one floodlight located above the circuit breaker panel on each side; a thunderstorm light located on the canopy on each side; a movable spotlight secured on the console on each side; flex point lights mounted on the canopy sill on each side; two emergency instrument lights located on the instrument panel glareshield; and one combination floodlight/emergency instrument light, above the right console, which illuminates the right instrument side panel.

ADI Light Rheostat

The ATT DIR IND rheostat on the lighting panel, controls light intensity of the attitude director indicator only. Power is from the ac hot bus through the FUEL CONT TEST instrument lights circuit breaker.

Instrument Lights Rheostat

The INSTR LTS rheostat on the lighting panel, controls instrument panel light intensity. Power is from the ac hot bus through the LH and RH instrument lights circuit breakers.

Console Lights Rheostat

The CONSOLE LTS rheostat on the lighting panel, controls light intensity of the right and left consoles. When the rheostat is not OFF, all caution and warning lights (including annunciator panel lights) are dimmed, except the nacelle fire warning lights and the landing gear handle warning light. Power is from the essential ac bus through the PANEL lights circuit breaker on the right console.

Floodlights Rheostat

The FLOOD LTS rheostat on the lighting panel, controls light intensity of the floodlights. Power is from the essential ac bus through the FLOOD lights circuit breaker on the right console.

Thunderstorm Lights Switch

The THUNDERSTORM lights switch on the lighting panel has two positions: ON (forward) and OFF (aft). Power is from the essential dc bus through the SPOT light circuit breaker on the left console.

Spotlights

The spotlights (utility lights) incorporate a rheostat on the aft end to control light intensity. A push-button switch bypasses the rheostat to provide maximum intensity. Power is from the essential dc bus through the SPOT light circuit breaker on the left console.

Flex Point Lights

The bodies of the two flex point lights are fixed to the left and right canopy sills. A flexible shaft, attached to each light body, contains a hood used to vary the size of the light beam. The hood is covered with velcro pile for stowing the flexible shaft under the canopy sill when not in use. A rheostat at the aft end controls light intensity. A push-button switch bypasses the rheostat to provide maximum intensity. Power is from the essential dc bus through the SPOT light circuit breaker on the left console.

Emergency Instrument Lights

The emergency instrument lights are automatically energized if ac hot bus power is lost or the ac hot bus INSTR light circuit breaker is open. Power is from the essential dc bus through the EMER INSTR light circuit breaker on the left console.

AFT COCKPIT LIGHTING

Integral and edge lighting for the instruments and consoles are controlled from the lighting panel on the left console. The left console PNL and LGD lighting circuit breakers on the lighting panel are only operative if the essential ac PNL L circuit breaker on the right console is closed. The right console TEST & BRT, PNL, and LGD lighting circuit breakers on the lighting panel are only operative if the essential ac PNL R circuit breaker on the right console is closed. Aft cockpit interior lighting also includes: one floodlight located above the left console; two floodlights above the right console; two floodlights above the circuit breaker panel on the right side; a floodlight on the canopy on each side; a spotlight secured on the left console; a spotlight secured on the right console; and flex point lights mounted on the canopy sill on each side.

Instrument Lights Rheostat

The INSTR rheostat on the lighting panel controls instrument panel light intensity. Power is from the essential ac bus through the INST lights circuit breaker on the right console.

L and R Console Lights Rheostats

The L CONSOLE and R CONSOLE rheostats on the lighting panel control legend light intensity for the left and right consoles, respectively. Power is from the essential ac bus through the respective PNL and LGD circuit breakers.

Floodlights Rheostat

The FLOOD rheostat on the lighting panel controls light intensity of the floodlights. Power is from the essential ac bus through the FLD lights circuit breaker on the right console.

Spotlights

The spotlights (utility lights) are identical to the spotlights in the forward cockpit. Power is from the essential dc bus through the SPOT LTS circuit breaker on the right console.

Flex Point Lights

The flex point lights are identical to the flex point lights in the forward cockpit. Power is from the essential dc bus through the SPOT LTS circuit breaker on the right console.

ENVIRONMENTAL CONTROL SYSTEM

These aircraft operate in an extremely adverse speed and altitude environment. Ram air temperatures may exceed 400°C at design airspeed, and ambient static air pressure can be less than 1/3 psi near the limit altitude. The external skin surfaces are painted black to radiate heat. Special insulating materials are used extensively to minimize temperature build-up within the aircraft. Pressurized and conditioned air must be supplied to the crew and equipment to provide a suitable environment. Both crewmembers wear full pressure suits for protection against cockpit depressurization or bailout at high altitude.

Air-conditioning includes two identical and parallel air-cycle compressor turbine refrigeration systems. Each is supplied by ninth-stage bleed air from one engine. The bleed air is regulated to no more than 26 psi, and is cooled by air-to-air and air-to-fuel heat exchangers before it is ducted to the air-cycle units. A temperature of about 200°F is maintained while below 44,000 feet. At higher altitudes, the air receives the maximum possible cooling. The bleed air supply is shut off if pressure can not be maintained above 5 psi. Cold air manifolds from the refrigeration turbines supply the chine and mission bays and the electrical, radio, and navigation bays. Conditioned air, which is temperature-controlled by the pilot, is used to pressurize and ventilate the cockpits and pressure suits. Cockpit exhaust air cools the nose and radar equipment and supplements cooling of the chine and equipment bays. Regulated bleed air supplies windshield deicing and, when mixed with conditioned air, windshield defogging. Regulated bleed air is also used to pressurize the canopy seals, some of the nose radar equipment, and the windshield rain removal system reservoir. The inlet control pressure ratio transducers and the inlet forward bypass actuator LVDT's, which are located in high temperature environments in the nacelles, are also cooled by regulated bleed air. Engine inlet ram air and engine fuel are used as heat sinks for the air-conditioning system. The ram air is exhausted within the nacelles after passing through the air-to-air heat exchangers. Heat sink fuel is either returned to the engine fuel supply system or diverted to tank 4 after use in the air-to-fuel heat exchangers. Refer to Figure 1-80, Environmental Control System, and Figure 1-39, Fuel Heat Sink System, for schematic diagrams.

Either of the two refrigeration systems can maintain a suitable cockpit environment and cool the ANS and essential radio and flight equipment in the E and R Bays. However, if one of the refrigeration systems fails at high supersonic speeds, cooling air for the nose, chine, and mission bays must be turned off and use of most of the equipment in these spaces should be discontinued.

The air pressure available at the chine and bay equipment boxes is substantially the same as that provided in the cockpits; however, pressure is reduced by orifices in individual equipment supply or bay vent manifolds to regulate flow to individual equipment. Normally, the cooling air temperature is automatically controlled to -30°F when at supersonic flight altitudes. Selecting FULL COLD on the manifold temperature switch can result in much lower supply temperatures in some cases. With the manifold temperature switch in AUTO, equipment cooling air temperature is approximately +37°F when below 41,000 to 44,000 feet, to prevent ice formation in the cold air manifold.

Water separators are provided in the supply manifolds to the ANS and bay areas. Although they have sufficient capacity for normal flight operations, they may not be adequate for sustained periods of ground operation when high humidity conditions exist. Suitable ground support equipment can be connected when prolonged ground operation is anticipated. The cockpit air supply is not dehumidified.

COCKPIT PRESSURIZATION SCHEDULES AND TEMPERATURE SELECTION

The crewmembers operate within a sealed and specially insulated compartment which contains the two cockpits. The compartment can be pressurized at either a 10,000 FT (foot) or 26,000 FT schedule. Each of the schedules can be used without restriction; however, the 26,000 FT schedule is usually preferred since it allows more airflow through the cockpits and enhances cockpit and bay cooling. An automatic control restricts the maximum rate of pressure change to approximately 5000 fpm when the selection is altered. Refer to Environmental Control Systems Controls, Cabin Pressure Switch, this section.

The cockpits remain essentially unpressurized while below 26,000 to 28,000 feet pressure altitude with the 26,000 FT schedule. Cockpit pressure is then maintained at 26,000 feet at all higher flight altitudes. With the 10,000 FT schedule selected, the cockpits remain substantially unpressurized while below 10,000 feet. They maintain approximately 10,000 feet until aircraft altitude exceeds

28,000 feet, then a pressure 5 psi greater than ambient at higher altitudes. Refer to Figure 1-81. Note that at 25,000 feet aircraft altitude, for example, cockpit pressure altitude would be only slightly less than the flight altitude with the 26,000 FT pressure schedule, and would equal 10,000 feet with the 10,000 FT schedule. The 10,000 FT schedule can be used at subsonic flight altitudes if a crewmember wishes to open the helmet faceplate temporarily.

Cooling capability of the air-conditioning system will be reduced somewhat during descents from supersonic cruise if the 10,000 FT schedule is used. This is due to increased back pressure at the cooling turbines and the resultant decrease in cooling capability. To minimize the effect of reduced cooling, the cockpit AUTOTEMP control should be turned toward COLD before descending from high supersonic cruise.

NOTE

> Use of the 26,000 FT schedule is recommended if air-conditioning system difficulties result in high cockpit temperatures.

A safety relief valve opens automatically to maintain 5.4 psi differential pressure if the normal pressure regulator valve malfunctions. The safety valve can also be fully opened by the PRESS DUMP switch, on the pilot's left instrument side panel. Operation of this solenoid dumps cockpit pressure to ambient static pressure very rapidly.

The pilot controls cockpit air temperature by adjustment of his automatic controls or by manually selecting the proportion of regulated hot air that is mixed with the cockpit cold air supply. Suit ventilation air temperature is also controlled by the pilot, but suit vent flow is controlled individually. Individual controls are provided for helmet faceplate heat settings and for oxygen system selection. The pilot controls the refrigeration system shutoff valves and the cockpit pressure dump valve. The RSO controls the cockpit air shutoff valve.

ENVIRONMENTAL CONTROL SYSTEM CONTROLS

Refer to the following discussions in this section: Windshield, for a description of the Windshield Hot Air De-Icing/Rain Removal System; Canopy, for the canopy latching system and the canopy seal; and Life Support Systems, for the suit and helmet protective systems (including suit and faceplate heat control).

Refrigeration Switches

The two-position REFRIG switches, on the pilot's left instrument side panel, control the engine bleed air supply valves for each refrigeration system. The valves are located downstream from the air-to-air and air-to-fuel heat exchangers in the "heat sink packages" in each wing, inboard of the nacelles. See Figure 1-80.

The ON (up) position of either the L (left) or R (right) refrigeration switch deenergizes the respective control circuit and allows the corresponding system shutoff valve and pressure regulator to open.

Each valve is spring-loaded closed and when its control circuit is deenergized, the valve opens fully if at least 5 psi bleed air pressure is available from its engine. The OFF (down) position of either switch energizes a solenoid in the corresponding shutoff valve and closes the valve, stopping all of the air supplied for that pressurization and air conditioning system. The solenoid is also energized automatically by action of either of two thermal switches if temperatures exceed $365 \pm 15^{\circ}F$ at the compressor inlet or intercooler outlet positions of the systems air cycle refrigeration machine. The shutoff valve reopens automatically when lower temperatures are sensed. An L or R AIR SYST OUT caution light on the annunciator panel illuminates when the corresponding shutoff valve is closed.

The shutoff valve solenoids are powered by the essential dc bus through the AIR SOV CONT circuit breaker on the pilot's left console.

ENVIRONMENTAL CONTROL SYSTEM

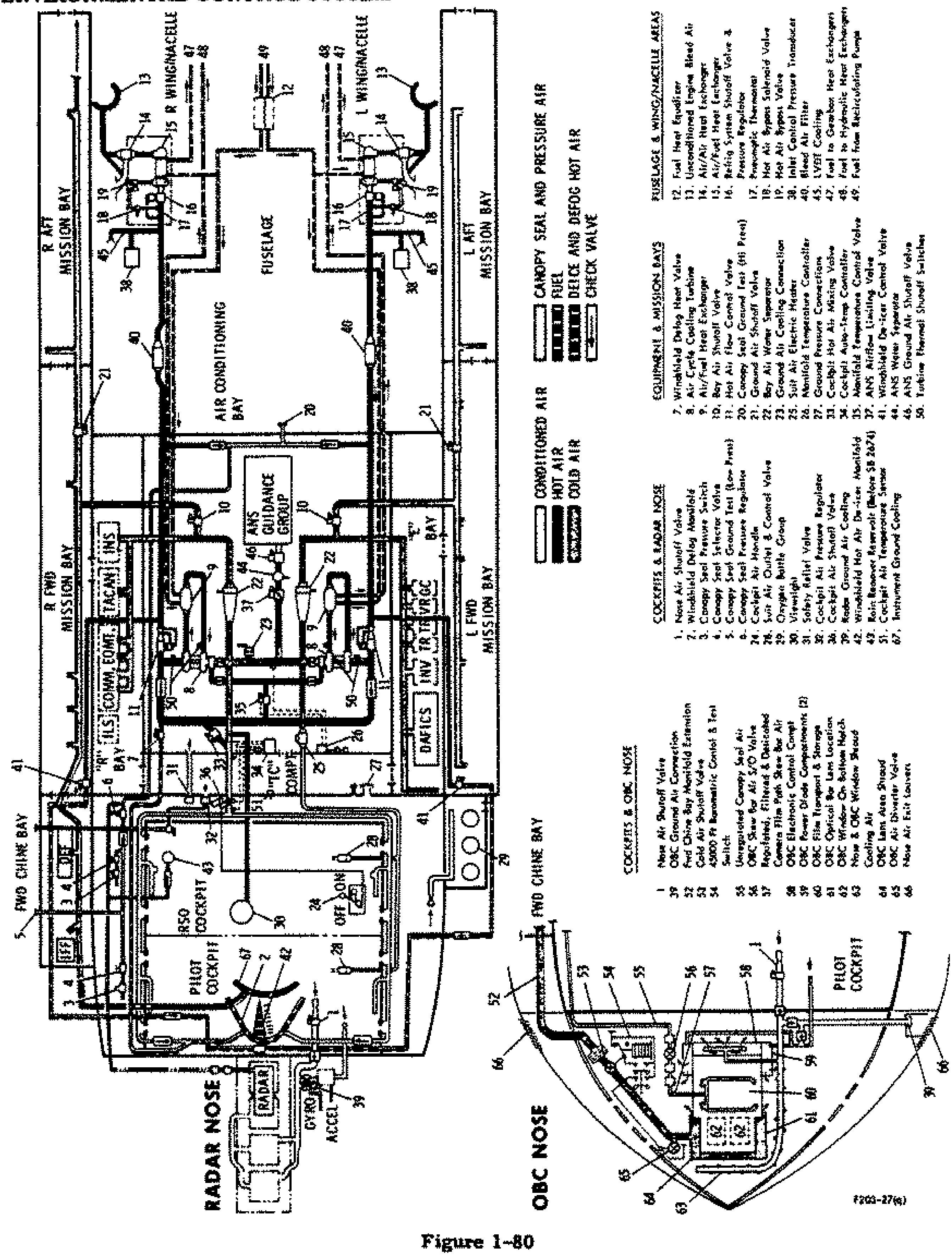
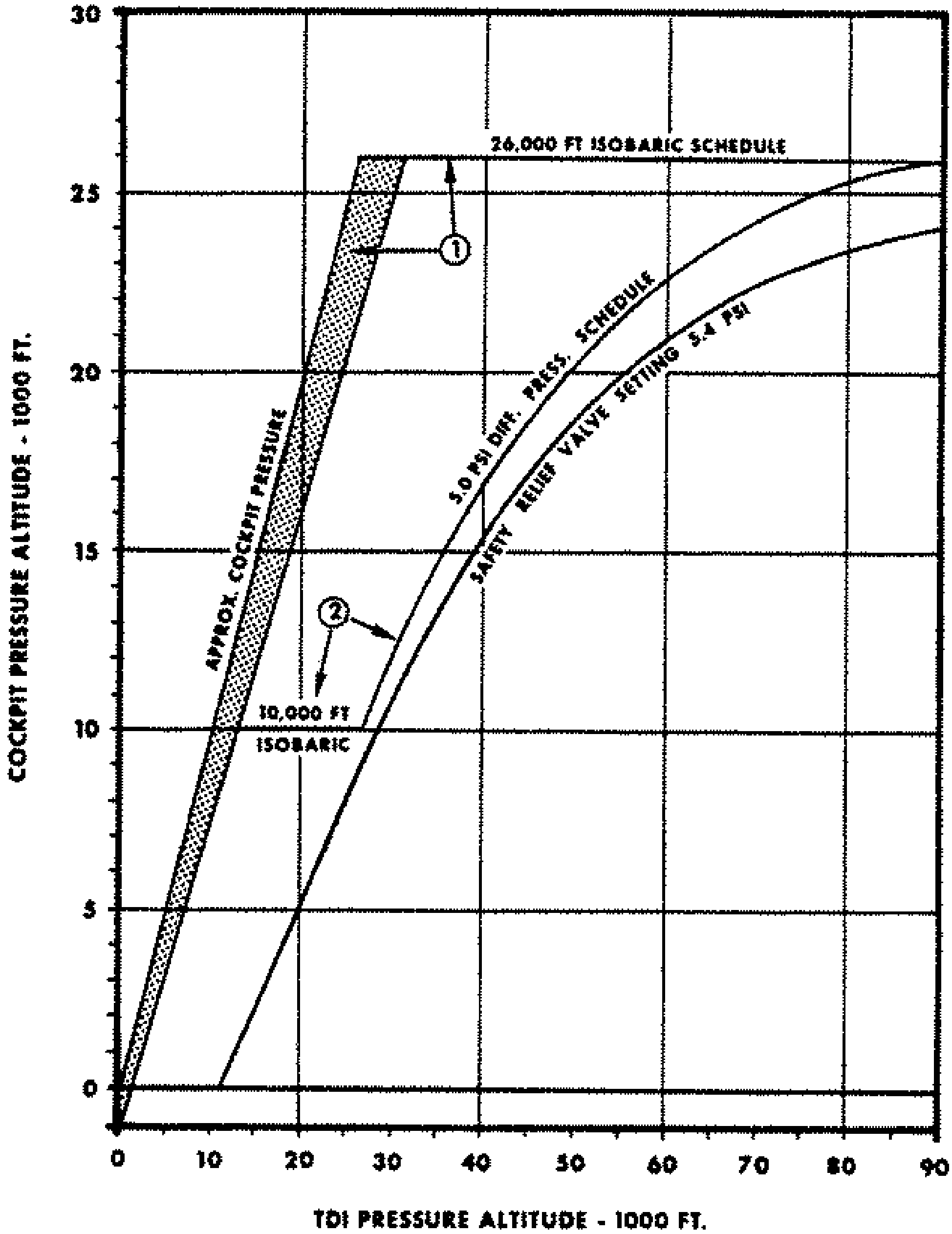


Figure 1-80

COCKPIT PRESSURIZATION SCHEDULE



① 26,000 FT PRESSURE SCHEDULE

② 10,000 FT PRESSURE SCHEDULE

Figure 1-81

Bay air may be off or on, to suit requirements for cockpit cooling, if one refrigeration system is off while CIT is low (subsonic). Bay air should not be turned off when cockpit air is shut off unless one REFRIG switch is also turned off, as cold air manifold temperatures could increase due to back pressure at the refrigeration turbines, and all of the cooling flow would then be directed toward and exceed the supply requirements for the ANS and E-Bay and R-Bay equipment boxes.

Power for the bay air shutoff valves and for opening the nose air shutoff valve is from the essential dc bus through the AIR SOV CONT circuit breaker on the pilot's left console.

CAUTION

With the Bay Air switch off, the chine bay equipment will overheat during subsonic or supersonic flight if electrical power to the equipment is not turned off. If supersonic, the E and/or R BAY OVERHEAT caution lights may illuminate even with the equipment off.

Cabin Pressure Switch

A two-position lift-lock CABIN PRESS switch is on the pilot's right console and on the RSO's instrument panel. The 26,000 FT position is normally preferred on flights involving high altitudes and air refueling as cockpit pressure remains relatively constant and cockpit and bay cooling is more effective. The switches in both cockpits must be in the 26,000 FT position to select the 26,000 FT pressure schedule. If either crewmember selects the 10,000 FT position, the 10,000 FT pressure schedule is selected. Figure 1-101 depicts the pressure schedules.

NOTE

- At low power settings, cockpit altitude may exceed 10,000 feet while below 28,000 feet altitude with the 10,000 FT schedule selected due to reduced bleed air supply pressure.
- Suit and cockpit cooling may be reduced due to increased back pressure with the 10,000 FT schedule selected.

Cockpit Air Valve

A cockpit air shutoff valve is located in the duct which supplies conditioned air to the cockpits. It is operated manually, either by the cockpit air handle under the RSO's left canopy sill, or by the RSO's canopy latch handle. The valve is open and air is furnished to the cockpits when the cockpit air handle is aft <u>and</u> the canopy latch handle is forward (locked). The valve is closed and conditioned air flow to the cockpits is shut off when the cockpit air handle is positioned forward to its OFF detent <u>or</u> when the canopy latch handle is aft (unlatched). Cockpit air is normally shut off for landing to prevent fog entering the cockpits from the air-conditioning system. The pilot's CKPT AIR OFF annunciator panel caution light illuminates when the cockpit air valve is closed.

WARNING

In high humidity conditions, do not resume use of cockpit air while at low altitude or after landing until certain that onset of fog will not endanger the aircraft.

When the cockpit air valve (36, Figure 1-80) is closed, the hot air mixing valve (33, Figure 1-80) is also closed automatically and the cockpit air supply temperature becomes full cold. If the cockpit temperature control switch is in AUTO TEMP, the hot air mixing valve reopens when the flow of cockpit air is resumed, and modulation at the selected automatic temperature level continues after a short delay. If the air temperature control is in MAN HOLD, the air supply temperature remains full cold when flow of cockpit air is resumed. The pilot must reset the temperature if a warmer temperature is desired.

Movement of the RSO's cockpit air handle has no effect when the canopy latch handle is in the aft (unlatched) position. Movement of the RSO's canopy latch handle has no effect on the cockpit air valve when the RSO's cockpit air handle is in the forward (off) position. In both cases, the cockpit air valve remains closed.

CAUTION

If either canopy is open, the aft canopy latch handle must be in the aft position or the cockpit air handle must be in the forward (off) position for adequate equipment cooling. Otherwise, most of the cooling air would exit through the cockpit openings instead of the bays.

Unless operating with a ground air supply connected, closing the cockpit air valve may increase the cold air manifold temperature slightly due to an increase in back pressure at the refrigeration turbines.

CAUTION

Do not operate with the cockpit air and bay air shutoff valves closed while above 10,000 feet unless one refrigeration system is also shut off. The increase in back pressure can result in decreased refrigeration turbine efficiency, higher than normal cold air manifold temperatures, and ANS equipment overheating.

The cockpit air temperature controls are not effective with cockpit air shut off, but hot air from the windshield defog system may be used to raise cockpit air temperature.

Cockpit Pressure Dump Switch

A two-position PRESS DUMP switch is on the pilot's left instrument side panel. It allows the pilot to dump cockpit pressure to ambient. In OFF (down), the cockpit pressure safety relief valve is closed and the cockpits are pressurized. The switch must be pulled out before it can be moved to ON (up). With ON selected, a solenoid in the safety relief valve fully opens the valve and the cockpit compartment depressurizes rapidly to air conditioning bay pressure (approximately ambient). The pressure suits inflate if cabin altitude exceeds 35,000 feet. As pressure in the cold air manifold is not affected substantially by the loss of cockpit pressure, airflow to equipment boxes in the bays and chines is not materially reduced by depressurizing the cockpits. Hot air remains available for defogging and windshield deicing. Power for the dump valve solenoid is from the essential dc bus through the PRESS DUMP circuit breaker on the pilot's left console.

WARNING

Cockpit pressure dump and/or repressurization occurs very rapidly.

Cabin Altimeter

A single-revolution cabin altimeter is located on the pilot's left instrument side panel. It is vented to cockpit static pressure and indicates cockpit altitude in increments of 1,000 feet from sea level to 50,000 feet.

Air System Caution Lights

The following caution lights on the annunciator panels are associated with the environmental control system.

Forward cockpit:
- L AIR SYS OUT
- R AIR SYS OUT
- E BAY OVERHEAT
- R BAY OVERHEAT
- CKPT AIR OFF
- WINDSHIELD DEICE ON
- CANOPY UNSAFE
- ANS REF
- BAY AIR OFF

Aft cockpit:
- ANS FAIL

The L or R AIR SYS OUT light illuminates when its corresponding refrigeration system shutoff valve is closed, either manually (using the refrigeration switches) or automatically because of high air temperature at the refrigerator compressor inlet or intercooler outlet.

The E or R BAY OVERHEAT light illuminates when air temperature exceeds 150°F at the aft outboard wall of the corresponding compartment.

The CKPT AIR OFF light illuminates when the cockpit air valve is closed either by aft (unlocked) positioning of the RSO's canopy latch handle or by setting the RSO's cockpit air handle to OFF (forward).

The WINDSHIELD DEICE ON light illuminates when the windshield deicing switch is in ON DE-ICE. The need for hot air deicing should be monitored, as the quantity of air available to the air contitioning and pressurization systems is reduced by deicing. Temperature of the air supply is approximately 200°F when below 44,000 feet.

The CANOPY UNSAFE light illuminates if either canopy is not closed properly on its sill, if either canopy latch handle is not full forward (latched), or if the canopy seals in either cockpit are not fully inflated.

The forward cockpit ANS REF and aft cockpit ANS FAIL lights illuminate if cooling air flow to the ANS is inadequate (less than 2.5 pounds per minute), if ANS component temperatures are too high or too low, or if the ANS is not operating normally. The RSO's ANS MAL and TEMP LIMIT/TEMP TOLR lights indicate abnormal air temperature and /or quantity. (Refer to Astroinertial Navigation System, Section IV.)

The BAY AIR OFF light illuminates if the BAY AIR switch is OFF or fails, or if the cap for the ground air receptacle is improperly installed, or if the microswitch in the cap for the ground air receptacle fails.

LIFE SUPPORT SYSTEMS

AIRCRAFT OXYGEN SYSTEM

One standby and two normal oxygen systems are provided. The normal system has two 10-liter liquid-oxygen-converter containers, and the standby system has one. Each of the three containers, located in the left chine opposite the aft cockpit, have a buildup circuit which maintains pressure at 65-100 psi until the oxygen supply is virtually depleted. Pressure above 120 psi opens a relief valve and vents overboard. The liquid oxygen from each converter is warmed and converted into a gas by passing through the supply line tubing (See Figure 1-82.)

Oxygen Control Panels

Control panels for both the normal and standby systems are located on the pilot's left console. A control panel for the normal system only is located on the RSO's left console. Two ON-OFF levers, labeled SYS 1 (system 1) and SYS 2 (system 2), located on each panel, manually control oxygen shutoff valves. When a lever is ON (full forward) a mechanical latch prevents moving the lever to OFF (aft). The normal control panels have a dual-reading pressure gage (0 to 140 psi) and the standby control panel has a single pointer pressure gage (0 to 140 psi). The normal oxygen gages indicate pressure only when the individual levers are ON, while the standby system pressure gage indicates pressure regardless of the position of the levers on the standby panel. Moving any lever to

ON opens a valve and permits oxygen to flow to the respective pressure suit helmet or oxygen mask of the crewmember. Check valves are provided at the seat disconnect for each of the supply lines, to prevent free flow of oxygen if a system valve is turned ON while the suit connections are disengaged.

NOTE

> If standby system pressure is higher than the normal system pressure, standby system oxygen pressure is displayed on the normal oxygen control panel gage(s) when the system(s) (1 and/or 2) levers are ON for both the normal and standby control panels.

<u>Liquid Oxygen Quantity Gage</u>

A LIQUID OXYGEN quantity gage is on the pilot's left instrument side panel and the RSO's instrument panel. The gages have dual pointers that indicate the volume in liters of the normal system or standby system, depending on the position of the liquid oxygen system quantity switch. The gages are calibrated in 1/2-liter increments from 0 to 10 liters.

<u>Liquid Oxygen System Quantity Switch</u>

A two-position switch, labeled LOX QTY, is on the pilot's left console. In SYS 1 IND 1 (aft), the liquid oxygen quantity indicators in both cockpits display the volume of liquid oxygen in systems 1 and 2. In STANDBY IND 1 (forward), needle 1 of both cockpit liquid oxygen quantity indicators display the volume of liquid oxygen in the standby system.

NOTE

> The volume of liquid oxygen in system 2 is always displayed by pointer 2 regardless of the position of the switch.

<u>Oxygen System Caution Lights</u>

Four oxygen system caution lights are installed on the annunciator panel in each cockpit. The SYS 1 OXY QTY LOW and SYS 2 OXY QTY LOW lights illuminate when the quantity of the respective system is less than 1 liter. If the No. 1 pointer is selected to indicate the standby system quantity, the SYS 1 OXY QTY LOW light indicates that the standby system quantity is less than 1 liter. The SYS 1 OXY PRESS LOW and SYS 2 OXY PRESS LOW lights illuminate when the pressure in the supply lines of the respective system is less than 50 (± 3) psi.

EMERGENCY OXYGEN SYSTEM

Each crewmember has two independent emergency oxygen bottles in the ejection seat survival kit. Each 45 cubic-inch capacity cylinder is pressurized to 2000 psi. Both emergency oxygen bottles are supplied automatically to the helmet upon ejection. Both oxygen systems can be activated manually by pulling the "green apple". Once the emergency oxygen system is actuated, it cannot be shut off. The emergency oxygen system should be actuated if the aircraft is not delivering the desired amount of oxygen from the ship systems, or hypoxia or noxious fumes are suspected. When actuated, check-valves prevent oxygen flow into the aircraft system. To prevent emergency oxygen flowing if aircraft system pressure is available, emergency oxygen system regulated pressure is slightly lower than the aircraft system pressure. Because emergency oxygen system regulated pressure is lower than the aircraft system pressure, turn the normal oxygen system supply levers to OFF after actuating emergency oxygen if contamination of the aircraft system is suspected. Pressure in the emergency system containers is indicated by gages on the forward left side of the survival kit (Figure 1-84) container.

<u>Oxygen Consumption</u>

The rate of oxygen consumption varies inversely with cockpit altitude. For a normal mission profile with the cockpit pressure switch in 26,000 FT, the average rate of oxygen consumption for two persons is 1 liquid liter per hour; with the switch in 10,000 FT, this average rate increases to 1.3

OXYGEN SYSTEM

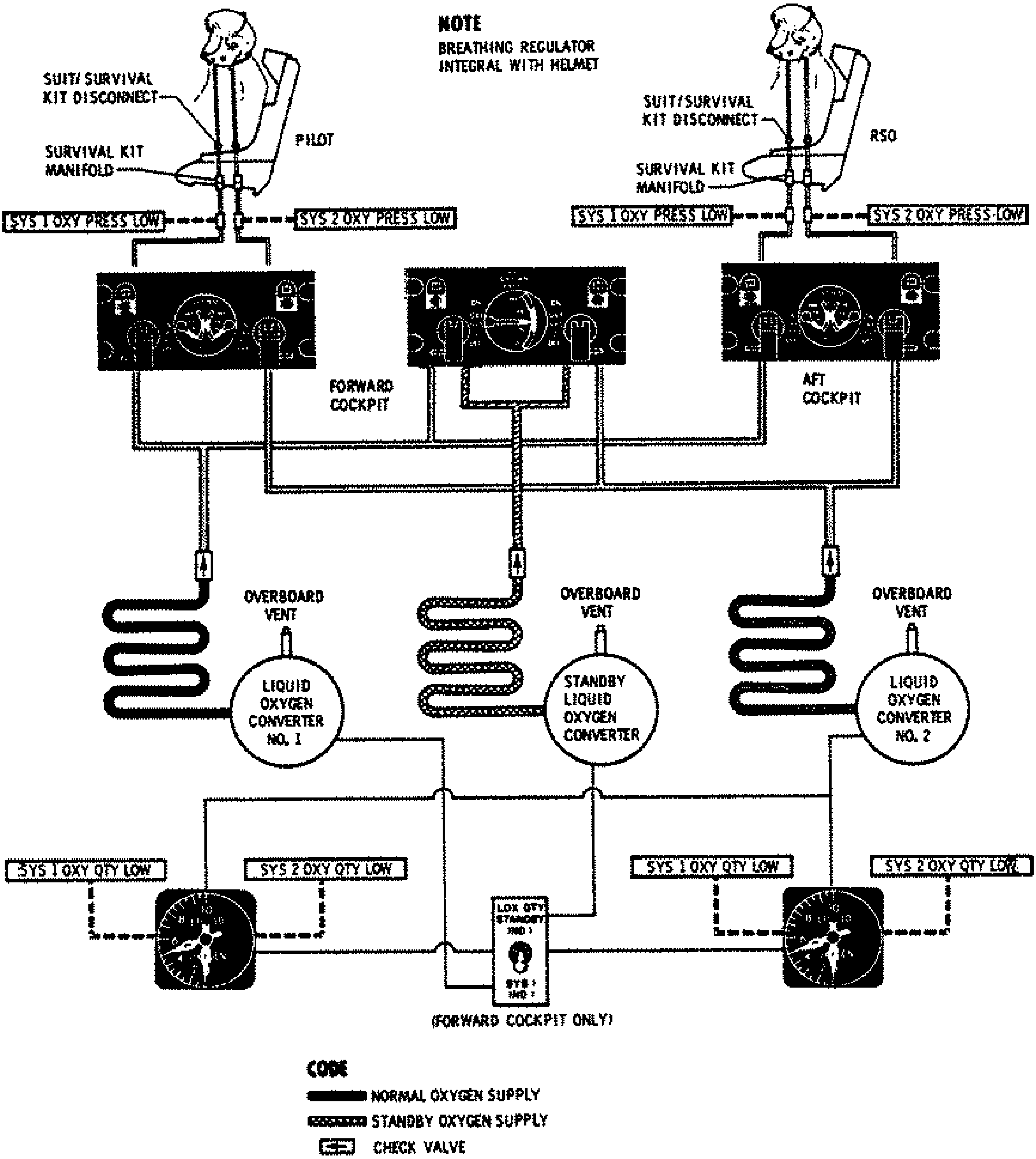


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Figure 1-82

liquid liters per hour. The rate of consumption for individual crewmembers may vary between 50% and 150% of average consumption rates. Emergency oxygen duration is approximately 15 minutes.

FULL-PRESSURE SUIT

The model 1030 full-pressure suit provides the crewmember a safe environment, regardless of cockpit pressure. The suit has six layers: internal comfort liner, vent duct, bladder, exposure garment, link net restraint, and exterior cover. The ventilation layer allows vent air to circulate between the crewmember's cotton underwear and the bladder layer. The airtight bladder holds pressurized air. The link net is a mesh used to shape the suit to the crewmember's body. The outer garmet provides some protection from high temperatures and wind. A vertical entry zipper is locate in the back.

Thermal Protective Garment

The double-walled, dual-function gas container/exposure garment contains the pressurized gas. The double-wall provides an interspace which can be orally inflated by a tube stored behind the left thigh pocket, to create an air-space, thermal-barrier for exposure protection. This double-walled bladder extends over most of the torso, arms, and legs (except for the hands and feet).

Pressure Suit Vent Air

Air for suit ventilation comes directly from the air conditioning cold air manifold. In cruise, the temperature of this air may be as low as -30°F with suit heat OFF and the MANIFOLD TEMP switch in AUTO. The air temperature can be substantially lower with FULL COLD selected. The air can be warmed by positioning the suit heat rheostat. Vent air and exhaled breathing air exhausts through the suit controller valve which controls suit pressure and vent flow rate.

Suit Heat Rheostat

A suit heat rheostat, on the pilot's left instrument side panel, controls an electric heater for the air supply to the pilot's and RSO's pressure suits. Individual comfort adjustment is accomplished by varying suit airflow through the ventilated air valve on each suit.

Suit Controller Valve

Air pressure in the suit is regulated by dual suit controller valves on the front of the suit just above the waist. Each controller valve meters airflow to keep internal suit pressure at 3.5 psi if the cockpit depressurizes. A press-to-test button on each controller valve bypasses the suit controller by closing the outflow valve, allowing the suit to inflate. A knurled knob may be rotated to check suit inflation or to partially inflate the suit for comfort.

Helmet

The helmet head area is divided into two sections by a rubber face seal. The front area receives oxygen from either the aircraft or emergency oxygen system through two independent regulators built into the helmet. Inhalation causes oxygen to flow across the visor and accomplishes some visor defogging before it is inhaled. The rear area receives ventilating air. The face seal may not be positive; however, oxygen pressure in the face area is slightly higher than in the rear area and prevents air from leaking forward. If the oxygen supply is interrupted or exhausted, an antisuffocation valve in the helmet senses the drop in pressure and allows ambient air to enter the helmet to prevent suffocation.

A transverse "Baylor" bar encircles half of the helmet. It pivots at each side attachment point. When raised, it lifts the visor and closes the helmet oxygen regulator valves. When lowered, it opens the helmet oxygen regulator valves and closes the visor. A latch below the face opening must be engaged by the bar and a lever on the bar rotated 90 degrees to lock and hold the visor sealed.

A microphone adjustment knob, below the helmet face opening, adjusts the fore-and-aft position of the helmet microphone.

CAUTION

> Do not force the microphone adjustment knob after the microphone has reached full travel; otherwise, the helmet shell may be damaged.

The visor is raised and lowered by the Baylor bar. With the bar raised, the visor is held up by a spring, uplock plunger, and cam on each side of the helmet. A transparent dark sunshade also pivots at each side connection. A friction clutch holds the sunshade in any position from fully open to closed. An external knob at the rear of the helmet adjusts the face seal. External helmet connections include two oxygen supply hoses and an electrical lead which provides power for visor heat, microphone, and earphone connections.

Face Heat Switches

A FACE HEAT rheostat switch is on the pilot's left instrument side panel and the RSO's instrument panel. The discrete switch positions are numbered 1 thru 7 with additional labels: OFF, LOW, MED and HIGH. Both oxygen flow and face heat defog the visor. Power for the switches is from the essential dc bus through individual FACE HTR circuit breakers in each cockpit.

CAUTION

- Do not use the HIGH face heat position when equipped with the PPG (glass) visor except for emergency heating. Continuous use of the HIGH heat position may delaminate the visor.
- The face heat switch should not be set above 5 with the visor raised, or the faceplate may be damaged.

Boots

A boot bladder is fastened to the suit by a zipper near the calf. The boot bladder will retain pressure. Flying boots are worn over the pressure suit boot bladder.

Gloves

Gloves fasten onto the suit at the wrist rings. The inner bladder of the glove is similar to the suit inner bladder and will retain pressure. The outer glove palm is leather.

TORSO HARNESS

The parachute harness is part of a torso harness worn over the pressure suit. The torso harness contains a built-in, dual-cell flotation vest with inflators to provide bouyancy in water after ejection. The two flotation sections are contained within removable velcro patches.

Each of the two cells is inflated by a water sensitive inflator which actuates within 0.5 second after immersion. Each automatic immersion inflator is a battery powered, electrically fired explosive squib which pierces the CO_2 bottle and inflates the vest. Each cell may also be manually actuated by a lanyard. Each cell may be inflated (or deflated) by an oral inflation tube.

The torso harness is also used for "shirt-sleeve" flights (flights below 50,000 ft. without a pressure suit). For shirt-sleeve flight, extra attachments are fastened on the harness to restrain the oxygen regulator. The left attachment (located where suit vent would be if a pressure suit were worn) has a pocket to hold the oxygen regulator. The right attachment (where the suit controller valve would be if a pressure suit were worn) restrains the oxygen line.

OXYGEN MASK AND REGULATOR

An oxygen mask assembly is used for "shirt-sleeve" flights below 50,000 ft. The mask assembly consists of a standard oxygen mask, a special oxygen regulator, an antisuffocation valve, and oxygen personal leads with connectors for both aircraft and emergency oxygen systems. If the regulator malfunctions or the oxygen supply is exhausted, the antisuffocation valve between the regulator and the mask senses a drop in pressure and permits ambient air to enter the mask to prevent suffocation.

EMERGENCY ESCAPE SYSTEM

The emergency escape system is comprised of an SR-1 stabilized ejection seat, canopy jettison system, and the operating controls and indicators in both cockpits. (See Figure 1-83.)

EJECTION SEAT

The SR-1 seat is usable from zero speed and altitude to the extremities of the flight envelope. The ejection seat is a rocket-propelled, upward-ejecting unit, which uses a drogue chute to stabilize seat descent until man-seat separation occurs automatically at approximately 15,000 feet. The personnel (main) chute is deployed automatically immediately after separation from the seat.

In the normal ejection sequence using the D-ring, the canopy is automatically jettisoned prior to seat ejection. If using the secondary ejection method (initiated by pulling the seat T-handle), the canopy must be jettisoned by separate action prior to pulling the T-handle.

WARNING

Do not pull the secondary ejection T-handle with the canopy still in place.

The survival kit, with emergency oxygen supply, and parachute are installed in the seat before the crewmember enters the cockpit. Hookup to the crewmember is by means of five quick-disconnect attachments.

Seat Vertical Adjustment

An electric motor, controlled by a 3-position switch on the right side of each seat bucket, moves the seat up or down. Seat movement is in the same direction as switch movement. The spring-loaded center position de-energizes the motor. Maximum vertical seat travel is 9 inches for the pilot's seat, and 6.75 inches for the RSO's seat. Power is from the essential dc bus through a SEAT ADJ circuit breaker in each cockpit.

Emergency Face Heat

A battery on the left side of the seat automatically provides face heat after ejection. The battery is hose-connected to the D-ring ballistic line, and a power cable is connected (by pull-to-release plug) to the helmet. During ejection, a squib in the battery fires to rupture diaphragms at both ends of the battery and forces electrolyte into the battery cells, providing a charged battery for visor heat. The battery cable plug is pulled loose from the helmet or seat during man-seat separation.

Inertia Reel

An inertia reel in the seat headrest maintains constant pressure on the shoulder harness straps to keep them taut and permit unrestricted movement of the crewmember. Placing the inertia reel control knob (on the left side of each seat) in the LOCK (forward) position manually locks the shoulder straps. When the inertia reel control knob is in the normal (UNLOCK) position, the reel will automatically lock with an instantaneous forward load of 2 to 3 g, and remains locked until released by moving the control knob to LOCK then back to UNLOCK.

Early in the ejection sequence, an initiator fires to roll up and lock the shoulder straps. Later in the ejection sequence, the shoulder straps are automatically severed when strap cutters fire. If the straps are not cut, they pull free of the inertia reel during man-seat separation. The strap cutters are fired manually by pulling the manual release (scramble) handle on the right side of the seat.

D-Ring Assembly

A D-ring, located on the forward part of the seat, is pulled to initiate ejection and also is a handhold for protection of the arms during ejection. A compression spring within the D-ring assembly acts as a shock absorber. During man-seat separation, the D-ring cable is automatically severed to release the D-ring. The D-ring cable cutter may be fired manually by pulling the scramble handle. A safety pin is inserted in a hole at the base of the D-ring (when on the ground) to prevent accidental firing of the ejection system.

T-Handle Assembly

A T-handle backup ejection control, is used if the D-ring initiator does not fire. The T-handle is located on the left side of the seat, under a cover shield which must be pushed away to gain access to the handle. A safety pin is inserted in a hole in the cover shield to prevent accidental firing of the ejection system when on the ground. Pulling the T-handle initiates only seat ejection; the canopy must be jettisoned by separate action prior to pulling the T-handle. The D-ring must be pulled before the T-handle can be actuated, since pulling the D-ring arms the T-handle system (secondary ejection sequence).

Scramble Handle

The manual release (scramble) handle on the right side of each seat is a quick release from the seat for bailout with the ejection seat inoperative, or for ground emergency egress. When a button at the forward end of the handle is depressed and the handle is pulled, the handle mechanically rotates the seat torque tube to release the lap belt and the parachute arming cable (the lap belt and cable remain attached to the parachute) and fires a no-delay initiator to release the crewmember from the seat through the following actions:

1. Foot retraction cables are cut.
2. Shoulder harness straps are severed.
3. Inertia reel is released (so shoulder straps pull free of reel if cutters fail).
4. D-ring cable is cut.
5. Main (personnel) chute lanyard is released from the seat.

NOTE

When pulling the scramble handle, expect a loud report from the initiator firing.

After a manual bailout using the scramble handle, the parachute must be manually deployed, using the shoulder D-ring.

Foot Retention System

The crewmember connects shoe spurs to foot retention fittings at the rear of the footrest. The fittings are cable-connected to reel assemblies which maintain the cables under slight spring tension to permit unrestricted foot movement. When ejection is initiated, the foot spur cables automatically retract to hold the crewmember's feet in the footrests. The spur cables are severed automatically during man-seat separation, or may be severed by pulling the scramble handle.

The spur is normally released from the cable by moving the foot aft (against the seat footrest) and raising the heel.

Parachute Beacon

A battery powered radio beacon is installed in the parachute. The beacon has a minimum operating life of 15 hours, and transmits an automatic signal on a frequency of 243.0 megacycles. During chute deployment, the beacon automatically turns on when a lanyard, attached to a chute riser, pulls a plastic plug from between the beacon control switch contacts. The beacon has a 22-inch telescoping antenna and a flexible removable antenna. Pulling a push-pull control knob, attached to the chute harness at the right front shoulder, disables the automatic beacon activation feature.

Seat Catapult

During ejection, the catapult gas charge is pressure-fired to initiate seat ejection. The catapult gas charge has a duration of 0.15 seconds, sufficient to raise the seat above the canopy sills, at which point the seat rocket motor automatically ignites. The seat rocket motor provides sufficient thrust and duration (0.5 second) to provide a seat elevation (relative to the aircraft) of approximately 300 feet.

When ejection is initiated by the D-ring, the seat catapult fires through a 0.3-second-delay initiator to provide time for canopy jettison prior to seat ejection. When the T-handle is used, the seat catapult fires immediately and no canopy ballistics are fired. The canopy must be jettisoned by separate action prior to pulling the T-handle.

Ejection Seat Drogue Parachute

A 6.5-foot-diameter drogue chute (stowed in the headrest) is deployed by a ballistic drogue gun, which fires 0.2 second after the seat catapult fires. The drogue (stabilizing) chute is connected to the seat at four points by a bridle and 10 feet of webbing. Ten seconds after drogue chute deployment (to permit seat deceleration), the lower two bridle lines are automatically severed to stabilize the seat, and the seat continues its descent upright. The drogue chute controls rate of descent and attitude until man-seat separation at approximately 15,000 feet.

Man-Seat Separation

Dual aneroid-actuated initiators located on the upper left and right sides of the seat, delay man-seat separation until 15,000 feet if ejection occurs at a higher altitude. Each unit contains a small dial, colored red and green. The dial is a leak indicator for the aneroid bellows and will indicate in the red band if the bellows leaks. When man-seat separation is initiated:

a. Foot retention cables are severed.

b. Drogue chute upper risers are severed 0.3 seconds after seat separation (to release drogue chute).

c. Lap belt is released.

d. Shoulder harness is severed and inertia reel is unlocked.

e. D-ring cable is severed.

f. Rotary actuator (butt-snapper) is fired.

When the rotary actuator is fired, the crewmember is forcibly separated from the seat as seat webbing is rapidly retracted by reel rotation. The seat webbing is secured to the front of the seat and passes under the seat pack, up to the rotary actuator in the headrest. Rapid tightening (retraction) of the seat webbing as the actuator fires, snaps the crewmember from the seat with a sling-shot action. The main chute is automatically deployed by a lanyard after man-seat separation.

Main Parachute

A 35-foot diameter main parachute in a backpack is deployed automatically after man-seat separation. The chute can also be deployed manually, after separation from the seat.

Automatic deployment is initiated by a drogue gun in the upper left corner of the pack. A lanyard connected to the seat fires the drogue gun as the man and seat separate. Accidental deployment is prevented by a lanyard housing which disconnects from the seat at separation.

The chute can be manually deployed by pulling the D-ring which is held in place by a Velcro patch on the suit, near the left shoulder harness. See Figure 1-83. The drogue gun is not fired when the chute is manually deployed.

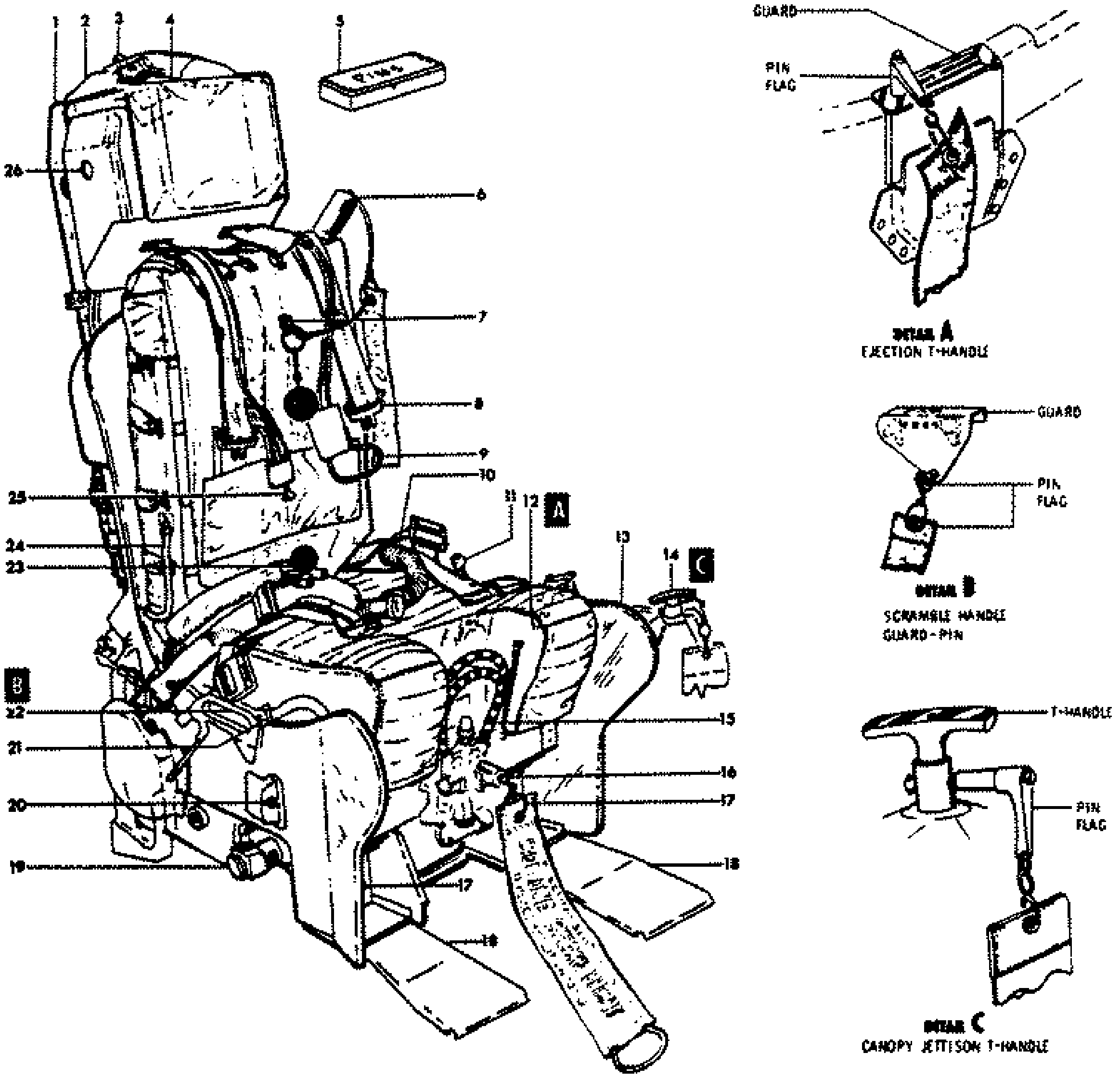
NOTE

> The main chute must be deployed manually if the crewmember has used the scramble handle for manual bail-out, as the main chute lanyard is released from the seat when the scramble handle is pulled.

The chute has a lanyard with a 7-inch loop tacked with breakaway thread on each rear riser strap. After deployment, if the parachute is not damaged, each loop should be pulled downward with a sharp tug approximately 1-1/2 feet. This releases three pairs of suspension lines on each side (24 pairs

SECTION I

EJECTION SEAT



1. AIRCRAFT GUIDE RAIL
2. DROGUE PARACHUTE
3. DROGUE PARACHUTE GUN
4. DROGUE PARACHUTE PIN
5. PIN STOWAGE - FWD COCKPIT ONLY
6. PARACHUTE GUN SAFETY CAP
7. PARACHUTE GUN SAFETY PIN
8. SHOULDER HARNESS
9. MANUAL PARACHUTE DEPLOY RING
10. SUIT VENT AIR HOSE
11. INERTIA REEL LOCK
12. EJECTION TEE HANDLE
13. LEG GUARD
14. CANOPY JETTISON HANDLE
15. EJECTION D RING
16. SAFETY PIN
17. FOOT RETENTION CABLES
18. FOOT RAMPS
19. FOOT RETENTION CABLE CUTTER (2)
20. SEAT VERTICAL ADJUSTMENT SWITCH
21. SURVIVAL KIT RELEASE
22. SCRAMBLE HANDLE
23. EMERGENCY OXYGEN HANDLE (GREEN APPLE)
24. MAIN PARACHUTE ARMING CABLE
25. RADIO BEACON CONTROL
26. LEAK DETECTOR ANEROID ACTUATED INITIATOR

Figure 1-83

remain), provides steering capability, and imparts a three to four ft/second forward speed to the chute.

WARNING

Do not pull either loop if the chute has sustained damage.

NOTE

Pull both loops. The chute will revolve continuously if only one set of lines is released.

For maximum effectiveness, start steering soon after chute deployment. A 180° turn requires 20 to 30 seconds.

The parachute may be equipped with a back cushion and/or a lowering device for descending from a tree. The lowering device consists of a let-down hook assembly, riser line section (colored blue), and 150 feet of nylon lead line packed in a special container in the parachute backpad. The last 25 feet of the lead line is colored differently (yellow, red, black, or white). See Figure 3-3.

Survival Kit

The survival kit is a hard-shell seat pack containing an emergency oxygen supply and survival equipment. Emergency oxygen quick disconnects are on the aft right corner of the kit, and two small oxygen quantity gages are at the forward left corner. (See Figure 1-84.) During ejection, emergency oxygen is turned on by a lanyard attached to the seat rail, or may be manually turned on by pulling the green apple. The kit may be released from the crewmember harness by pulling the kit release handle at the right side of the kit. The kit release handle should be pulled while descending in the main parachute, except when a tree landing is anticipated. When the release handle is pulled, the kit falls to the end of a 25-foot lanyard. The lanyard is disconnected from the harness if the release handle is pulled while seated.

WARNING

Pulling the kit release handle while seated disconnects both normal and emergency oxygen supplies.

PRIMARY EJECTION SEQUENCE

The D-ring is pulled to initiate the primary ejection sequence, which includes automatic canopy jettison. Pulling the D-ring fires an initiator on the front apron of the seat. Gas pressure from the D-ring initiator is ported directly to the foot retractor, the shoulder harness reel, and the canopy unlatch thruster. Pulling the D-ring also arms the secondary ejection sequence.

When the canopy unlatch thruster reaches full throw, gas pressure is ported to the canopy seal cutter and canopy removal thruster. Another initiator (pin-fired as the canopy is raised) ports pressure through the jettison valve to a 0.3-second delay initiator, which allows time for complete canopy jettison prior to firing the seat catapult initiator.

Gas pressure from the delay initiator provides pressure to fire the catapult initiator and arm the 0.2 and 1.4-second delay initiators in the drogue chute. The seat rocket motor ignites as the catapult raises the seat above the canopy sills.

The 0.2-second-delay initiator fires the drogue gun to deploy the drogue chute. The 1.4-second-delay initiator fires to arm the drogue chute lower riser cutters and the dual aneroid actuators. The lower riser cutters are armed through a 10-second delay, which provides time for seat deceleration before severing the lower riser lines. When the lower risers are severed, the seat is stabilized upright by the drogue chute. The dual aneroid actuators delay man-seat separation until 15,000 feet altitude if ejection occurs at a higher altitude. At completion of man-seat separation, the main chute is automatically deployed. See Figure 1-85.

SURVIVAL KIT

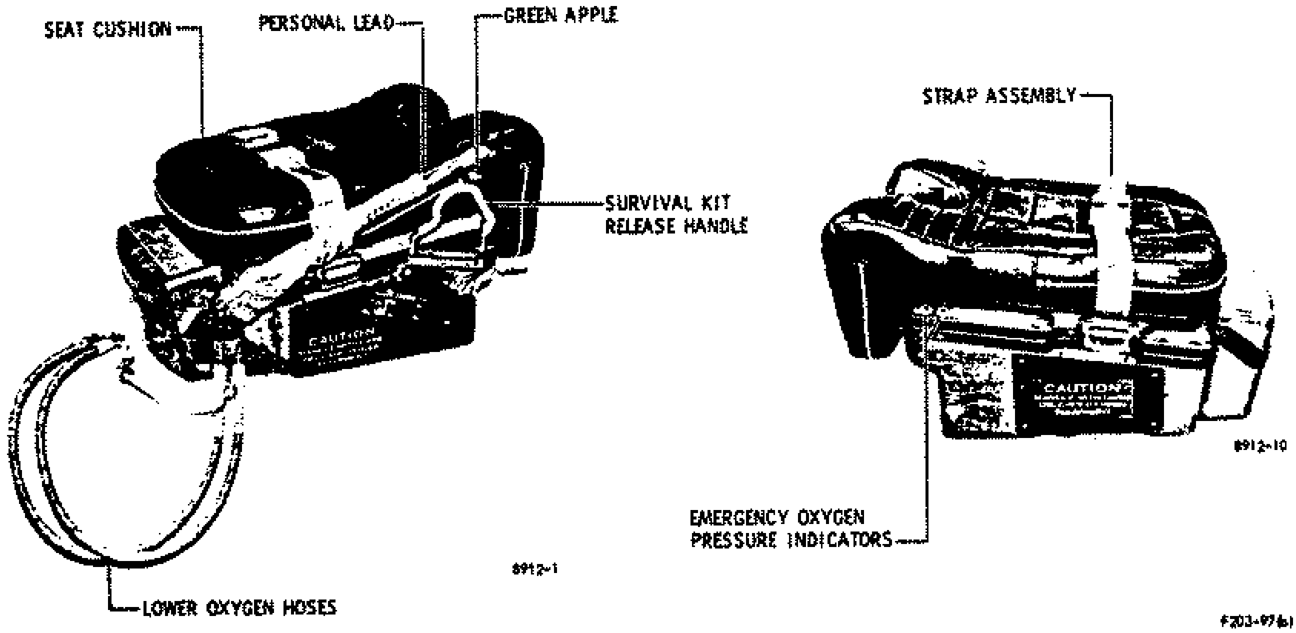


Figure 1-84

SECONDARY EJECTION SEQUENCE

If the primary ejection sequence malfunctions, the canopy must be jettisoned and then the ejection seat T-handle is pulled to initiate the secondary ejection sequence. The secondary sequence does not automatically jettison the canopy.

WARNING

Do not pull the secondary ejection T-handle with the canopy still in place.

NOTE

Secondary ejection sequence can not be initiated before the D-ring is pulled.

When the T-handle is pulled, the catapult initiator fires immediately and gas is ported to the two delay initiators in the drogue chute. Action downstream of the 1.4-second-delay initiator in the drogue chute is identical to the primary ejection sequence.

EGRESS COORDINATION SYSTEM

An egress coordination system supplements interphone communication. With this system, the pilot can issue and check compliance with an electrical bailout signal.

Egress Lights and Switches

A guarded RSO BAILOUT switch on the pilot's instrument panel has three positions: ALERT (down), OFF (center), and GO (up). Above this switch is a red RSO EJECTED warning light. The RSO instrument panel has an ALERT caution light, and PILOT EJECTED and BAILOUT warning lights. When the pilot actuates the RSO BAILOUT switch to ALERT or BAILOUT position, the corresponding light on the RSO panel illuminates. The ALERT light is a flashing amber light and the BAILOUT light is a steady red light. The forward cockpit RSO EJECTED light is operated by a switch on the aft cockpit ejection seat track. The aft cockpit PILOT EJECTED light is operated by a switch on the forward cockpit ejection seat

EJECTION TRAJECTORIES

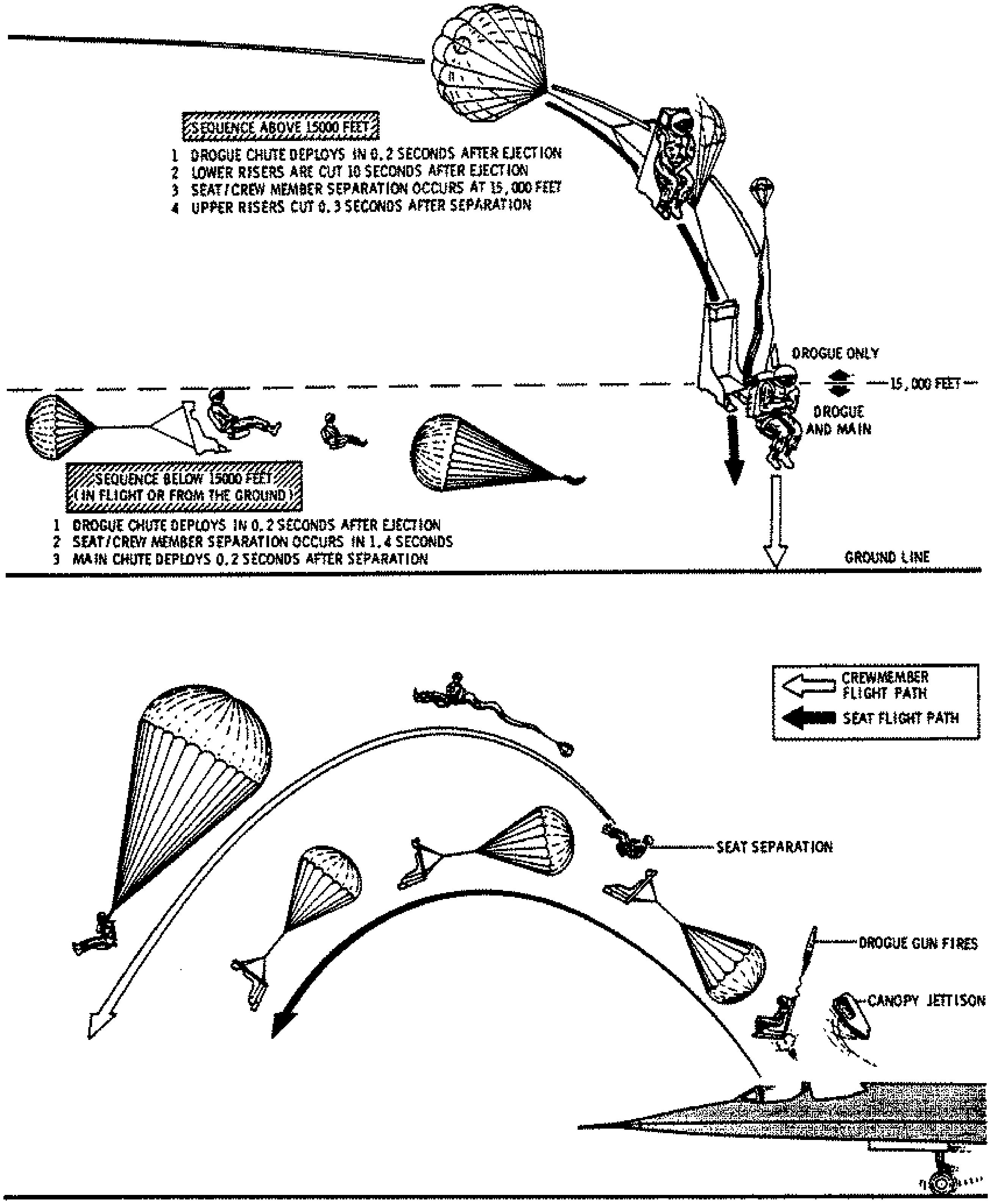


Figure 1-85

track. When a seat ejects, the respective light illuminates in the opposite cockpit. The aft cockpit BAILOUT light illuminates automatically when the PILOT EJECTED light illuminates. Power for the lights is provided directly from the battery through a BAILOUT LT circuit breaker in the E-bay.

EMERGENCY WARNING EQUIPMENT

MASTER WARNING SYSTEM

Forward Cockpit

An annunciator panel, located below the instrument panel, contains amber caution and red warning lights, with engraved legends, to indicate abnormalities of certain systems.

An amber master CAUTION light and a red master WARNING light are located at the top of the instrument panel. Annunciator lights flash when initially activated. A flashing amber or red annunciator light will cause the corresponding master CAUTION or master WARNING light to illuminate steady. Depressing the master light (amber or red) extinguishes the master light and causes the associated annunciator light to illuminate steady. Any subsequent annunciator light that illuminates will flash until the associated master light is pressed.

The annunciator lights do not give an indication of multiple failures such as could occur in the A or B HYD system. For example: The loss of A system hydraulic fluid illuminates the A HYD light and the master CAUTION light. The A HYD light will flash until the master CAUTION Light is reset, then will remain on steady. If the A HYD pressure subsequently fails, the A HYD light will remain on steady and the master CAUTION will remain off.

The master caution and warning lights do not illuminate with fire, inlet unstart, or landing gear unsafe warning lights.

All caution and warning lights, except the nacelle fire warning lights and landing gear handle warning light, are dimmed when the CONSOLE LTS switch is not OFF. Power for the KEAS warning, RSO ejected, A/P OFF, both IGV, and both derich lights is furnished by the essential dc bus through the WARN 2 circuit breaker on the pilot's left console. Power for all annunciator caution and warning lights and all other caution and warning lights except the nacelle fire warning lights, the inlet unstart lights, the landing gear handle warning light and lights on the WARN 2 circuit breaker is furnished by the essential dc bus through the WARN 1 circuit breaker on the pilot's left console.

NOTE

> The L and R OIL TEMP annunciator lights are not functional. The OIL TEMP lights only illuminate when the IND & LT TEST switch is pressed.

Aft Cockpit

An annunciator panel and an amber master CAUTION light are located on the instrument panel. The annunciator lights and CAUTION light function the same as those in the forward cockpit except that the annunciator lights illuminate steady instead of flashing. The annunciator lights and master CAUTION light cannot be dimmed. There are no red annunciator lights or master warning light in the aft cockpit.

An amber PILOTS CAUTION light illuminates when the pilot's master WARNING and/or master CAUTION light illuminates. The PILOTS CAUTION light remains illuminated until the master light(s) in the forward cockpit are pressed off. This permits the RSO to alert the pilot if the forward lights are not immediately noticed by the pilot. Power for aft cockpit caution lights is furnished by the essential dc bus through the WARN LTS circuit breaker on the RSO's right console.

Indicators and Warning Lights Test Button

Forward Cockpit

A push-button indicator and warning lights test switch, labeled IND & LT TEST, is located on the left instrument side panel. When the IND & LT TEST switch is depressed:

The following indicators move to their full counter-clockwise positions:

- Liquid nitrogen quantity
- Liquid oxygen quantity
- Fuel quantity
- CIP
- CG (forward and aft cockpit)
- Spike and forward bypass

The following lights illuminate:

- Annunciator panel caution and warning
- Master CAUTION and master WARNING
- L & R UNST
- Air refueling READY and DISC
- Fuel derich
- IGV
- KEAS
- SHAKER
- Landing gear indication (3 green lights)
- Nacelle fire warning. Illumination of each fire warning light indicates that the respective fire warning loop is functioning.
- Landing gear handle warning. (landing gear warning tone also heard in both headsets if one or both throttles are in idle or OFF).

All annunciator lights illuminate steady 1-1/2 seconds after the IND & LT TEST switch is depressed. Depressing the switch also illuminates the following aft cockpit lights: CG, master CAUTION, and PILOTS CAUTION.

Aft Cockpit

An indicator and warning light push-button test switch, labeled LAMP TEST, is located on the PWR & SENSOR control panel. Power for the switch is furnished by the essential dc bus. When the LAMP TEST switch is depressed:

Oxygen quantity indicator needle moves toward zero.

Fuel quantity needle moves toward zero.

The following lights illuminate:

- Annunciator panel
- UHF TRANS
- Master CAUTION
- ANS control panel
- PWR & SENSOR control
- Radar control, test and display panel
- DEF control and DEF warning panel

NACELLE FIRE WARNING SYSTEM

A fire warning system indicates the presence of a fire or hot spot in the engine nacelles. A high temperature, anywhere along the length of the detection circuits, illuminates the warning light for the corresponding nacelle.

There are two pairs of fire warning loops in each nacelle. One forward and aft pair is on the outboard side of the nacelle and the other forward and aft pair is on the inboard side. Both wires of a pair must sense a hot condition before a warning is given. Pressing the pilot's IND & LT TEST switch illuminates the FIRE LEFT NAC and FIRE RIGHT NAC warning lights if the fire warning loops are intact.

Nacelle Fire Warning Lights

FIRE LEFT NAC and FIRE RIGHT NAC warning lights, on the pilot's instrument panel, illuminate when nacelle temperature near the turbine or afterburner exceeds 1200° (± 50°) F. Covers are provided for reducing the brightness of the lights at night. Power for the lights is furnished by the emergency ac bus through the L and R FIRE WARN circuit breakers on the pilot's right console.

Fire Warning Loop Test Switches

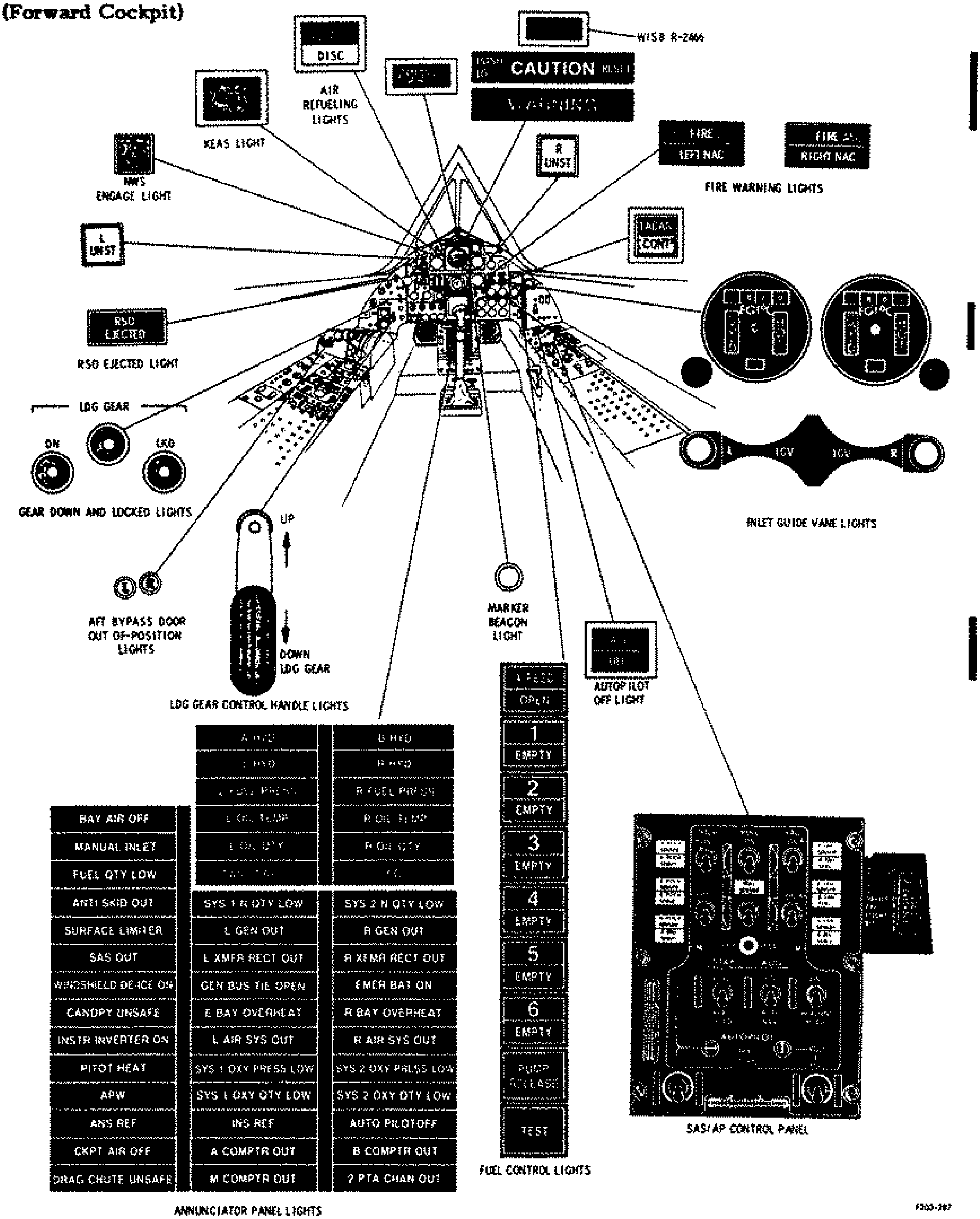
Two switches, one for each engine, are installed above and aft of the left console in the forward cockpit. The switches, labeled L LOOP SEL SW and R LOOP SEL SW, are used by maintenance personnel to checkout the fire warning system.

MISCELLANEOUS EQUIPMENT

Dinghy Stabber

A dinghy stabber, on the pilot's glareshield and on the RSO's right console, is provided to deflate the dinghy if it accidentally inflates in the cockpit.

WARNING, CAUTION AND CONDITION LIGHTS (Forward Cockpit)



WARNING, CAUTION AND CONDITION LIGHTS
(Aft Cockpit Instrument Panel)

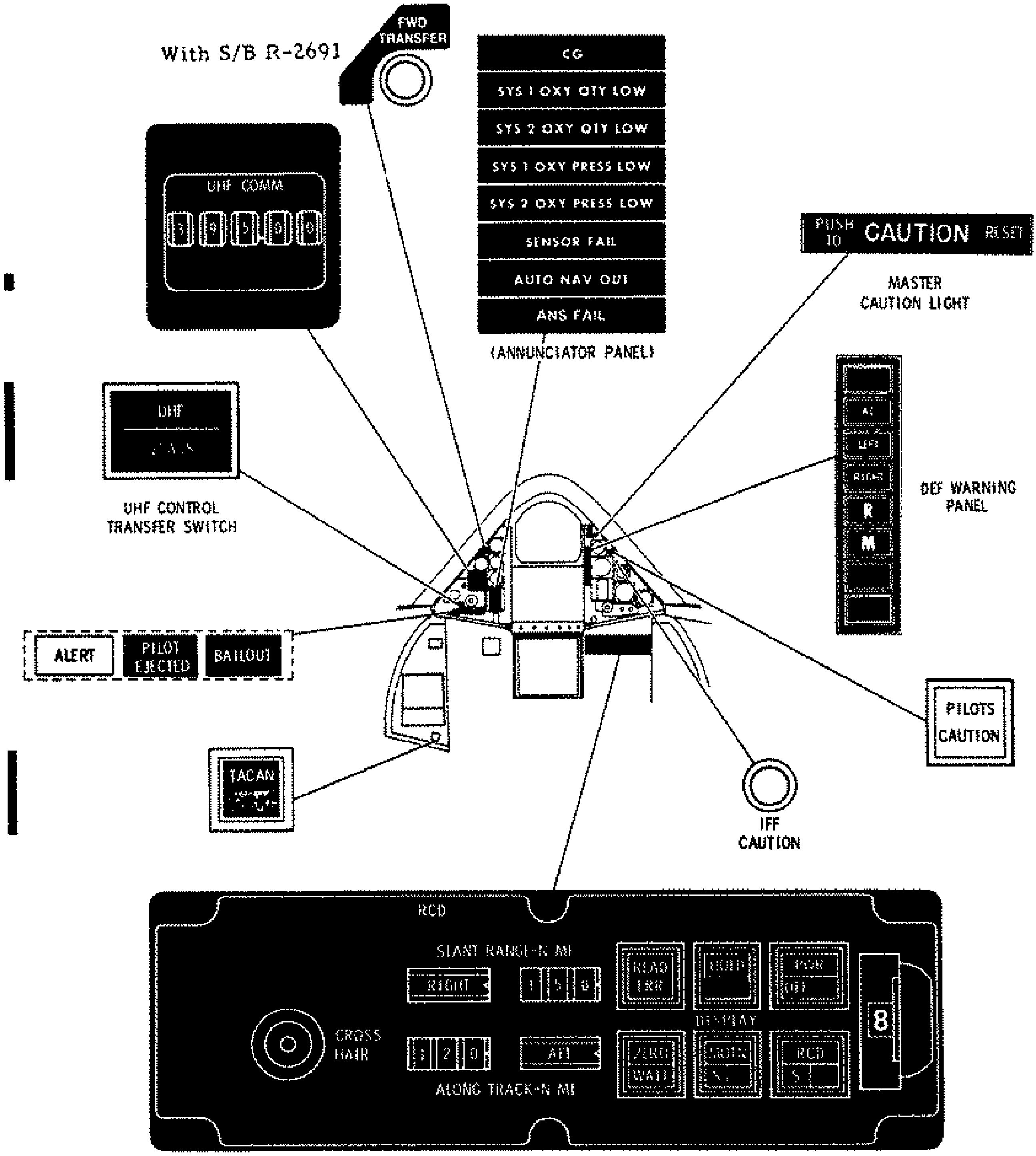


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Figure 1-87