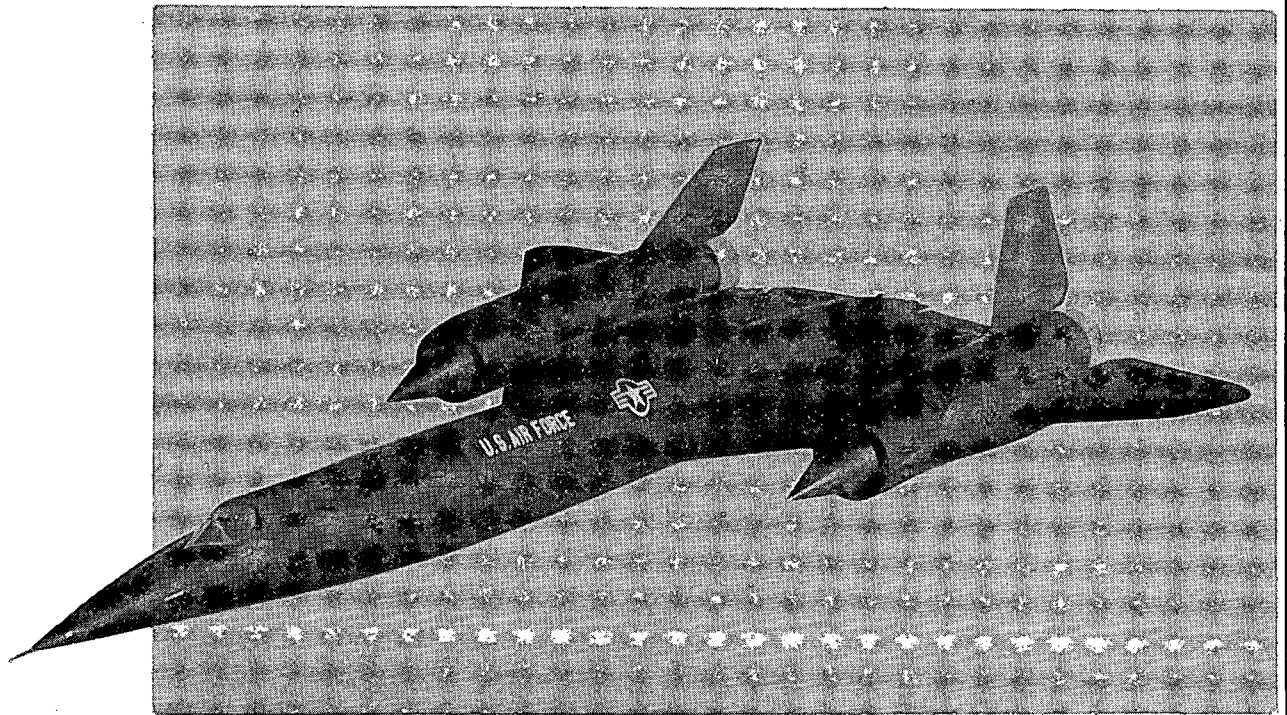


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UTILITY FLIGHT MANUAL

15 SEPTEMBER 1965

CHANGED 15 JUNE 1968

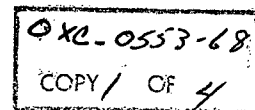
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COPY NO. 11



Page 1 of 1
TDC No. 13
15 June 1968

A-12 FLIGHT MANUAL
TECHNICAL DATA CHANGE

This TDC transmits revised pages which supersede previously furnished pages for the Flight Manual dated 15 October 1967. All previously issued TDC's are incorporated.

In addition, this TDC includes:

- a. Rapid Deployment to ARCP data.
- b. Revised presentation of normal climb performance
- c. Revised presentation of cruise performance for long range and high altitude cruise (1956 ARDC and "MEAN TROPIC" atmospheres)
- d. Revised single engine descent data for various speeds, powers, and for both 1956 ARDC and "Mean Tropic" atmospheres.
- e. Minor descriptive material.

Previously issued checklist changes conform with procedures supplied in this manual.

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TECHNICAL DATA CHANGE SUMMARY

<u>TDC</u>	<u>Date</u>	<u>Purpose</u>	<u>Status</u>
No. 1	10-16-67	Est. Tropical Atmosphere Climb Performance	Superseded by TDC 3
No. 2	1-26-68	Emergency Forward Transfer	Inc.
No. 3	2-05-68	Revised Climb and Cruise Performance (1956 ARDC Atmosphere & "MEAN TROPIC" Atmosphere)	Inc.
No. 4A	3-04-68	Time Limits & EGT Limits	Inc.
No. 5	3-01-68	Supersonic Cruise Flight Characteristics	Inc.
No. 6	3-05-68	Tire Limits	Inc.
No. 7	3-06-68	Climb and Cruise Performance	Inc.
No. 8	3-15-68	Increase Chute Deploy Limit 210 KIAS	Inc.
No. 9	3-16-68	Transmit Printed Change Dated 3-15-68	Inc.
No. 10	5-7-68	Rapid Deployment to ARCP	Inc.
No. 11	5-10-68	Normal Operation for Descent & Engine Shutdown	Inc.
No. 12	5-16-68	Normal Climb Performance Revised	Inc.
No. 13	6-15-68	Transmit Printed Change Dated 6-15-68	Inc.

Changed 15 June 1968

D/E

SECURITY INFORMATION

SPECIFIC INSTRUCTIONS FOR SAFEGUARDING THIS INFORMATION

1. This document contains information affecting the national defense of the United States within the meaning of the Espionage Laws Title 18, USC Section 793 and 794. The transmission or the revelation of its contents in any manner to unauthorized persons is prohibited by law.

The nature of this document is such that dissemination and handling will be carried out with strict adherence to the following policies:

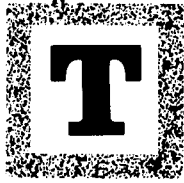
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 - b. Strict accountability of each document will be maintained.
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2. In the event this document is lost or is subject to unauthorized disclosure or other possible subjection to compromise of classified information, such fact will be promptly reported to the authority responsible for the custody of the material for appropriate action.

FOREWORD

The primary purpose of this manual is to provide systems descriptions, operating procedures, limitations, and other information necessary for operation of these aircraft. Experience and basic aircraft familiarity gained to date has been recognized in the preparation of this manual.

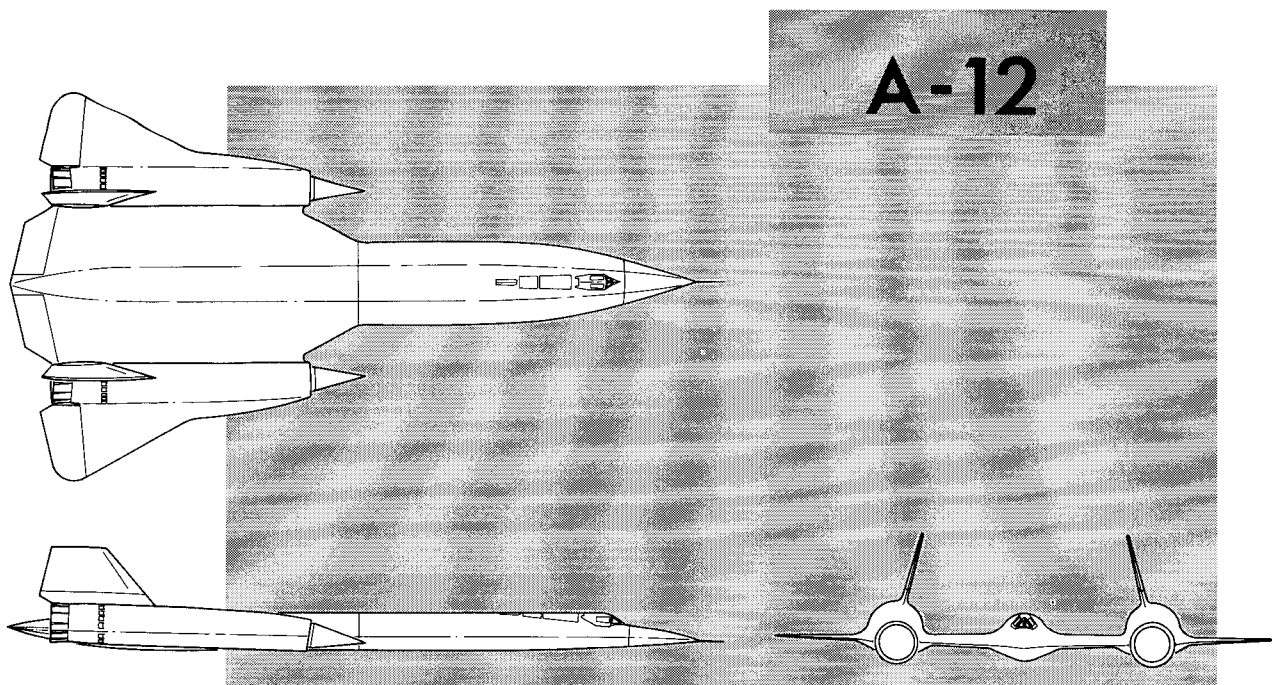
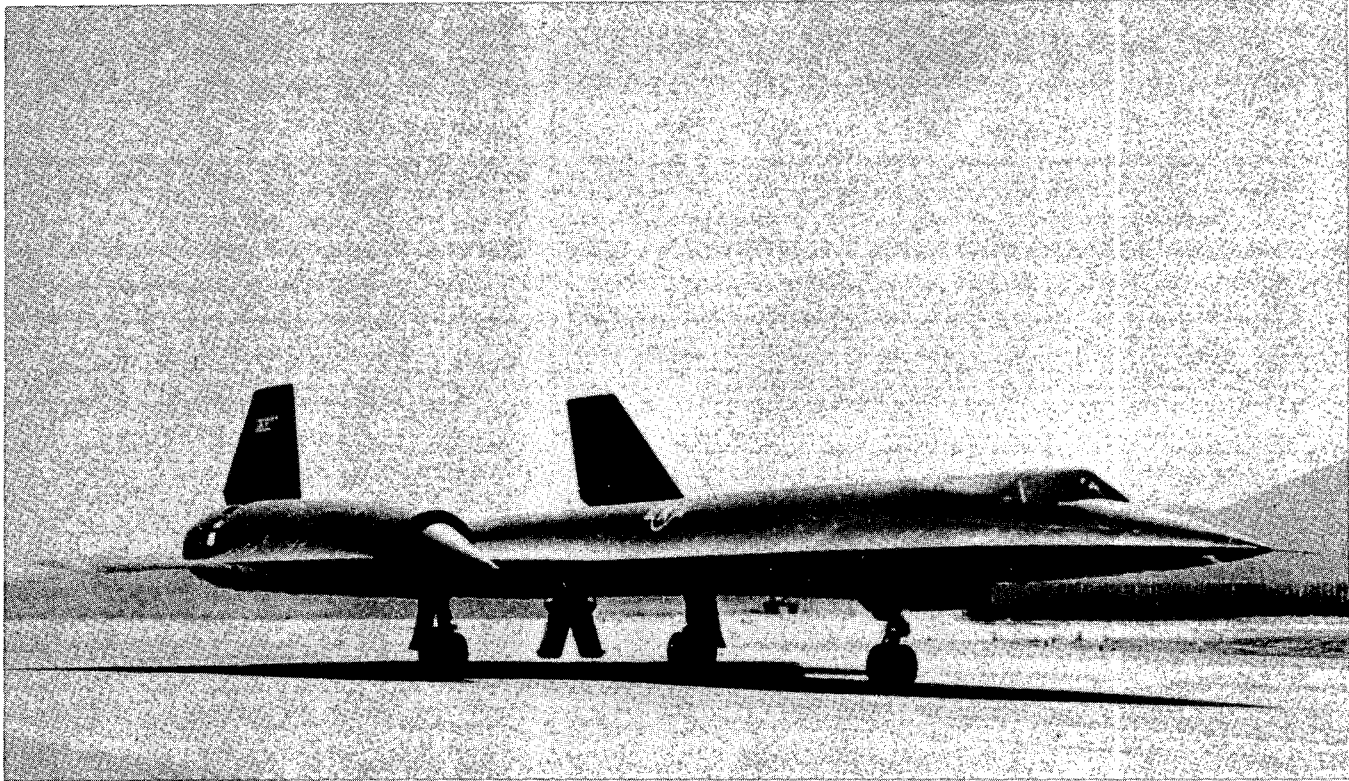
This manual will be reviewed and changed periodically by the manufacturers test organization to reflect information gained from further tests and operating experience. Comments, corrections, and/or questions regarding this manual are welcome. They may be forwarded through the manufacturer's senior engineer at the test site or directly to the flight manuals representatives.

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Section I

DESCRIPTION

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THE AIRCRAFT

The A-12 is a delta wing, single place aircraft powered by two axial flow bleed bypass turbojet engines with afterburners. The aircraft is built by the Lockheed-California Company and is designed to operate at very high altitudes and at high supersonic speeds. Some notable features of the aircraft are very thin delta wings, twin canted rudders mounted on the 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 inlet geometry. The surface controls are elevons and rudders, operated by irreversible actuators with artificial pilot control feel. A single-point pressure refueling system is installed for ground and in-flight refueling. A drag chute is provided to reduce landing roll.

AIRCRAFT DIMENSIONS

The overall aircraft dimensions are as follows:

Wing Span	55.62 ft.
Length (overall)	101.6 ft.
Height (to top of vertical stabilizer)	18.45 ft.
Tread (MLG center wheels)	16.67 ft.

AIRCRAFT GROSS WEIGHT

The ramp gross weights of these aircraft may vary from approximately 122,900 lb. to 124,600 lb. with 10,590 gallons of fuel. This is based on zero fuel weights between 54,600 lb. and 56,300 lb., fuel density of 6.45 lb. per gallon, and varying equipment loading configurations.

GENERAL ARRANGEMENT

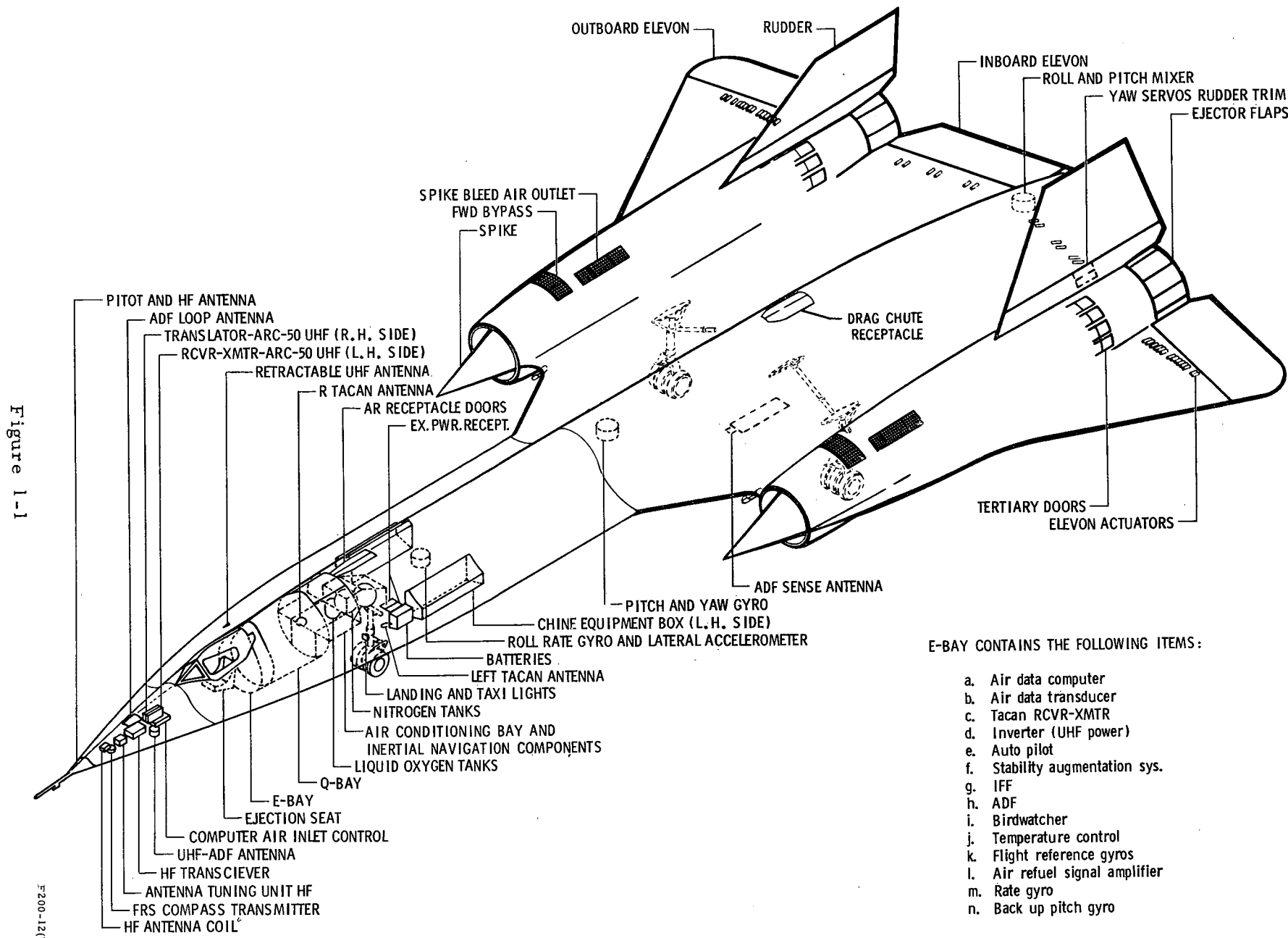


Figure 1-1

F 200-12(b)

INSTRUMENT PANEL

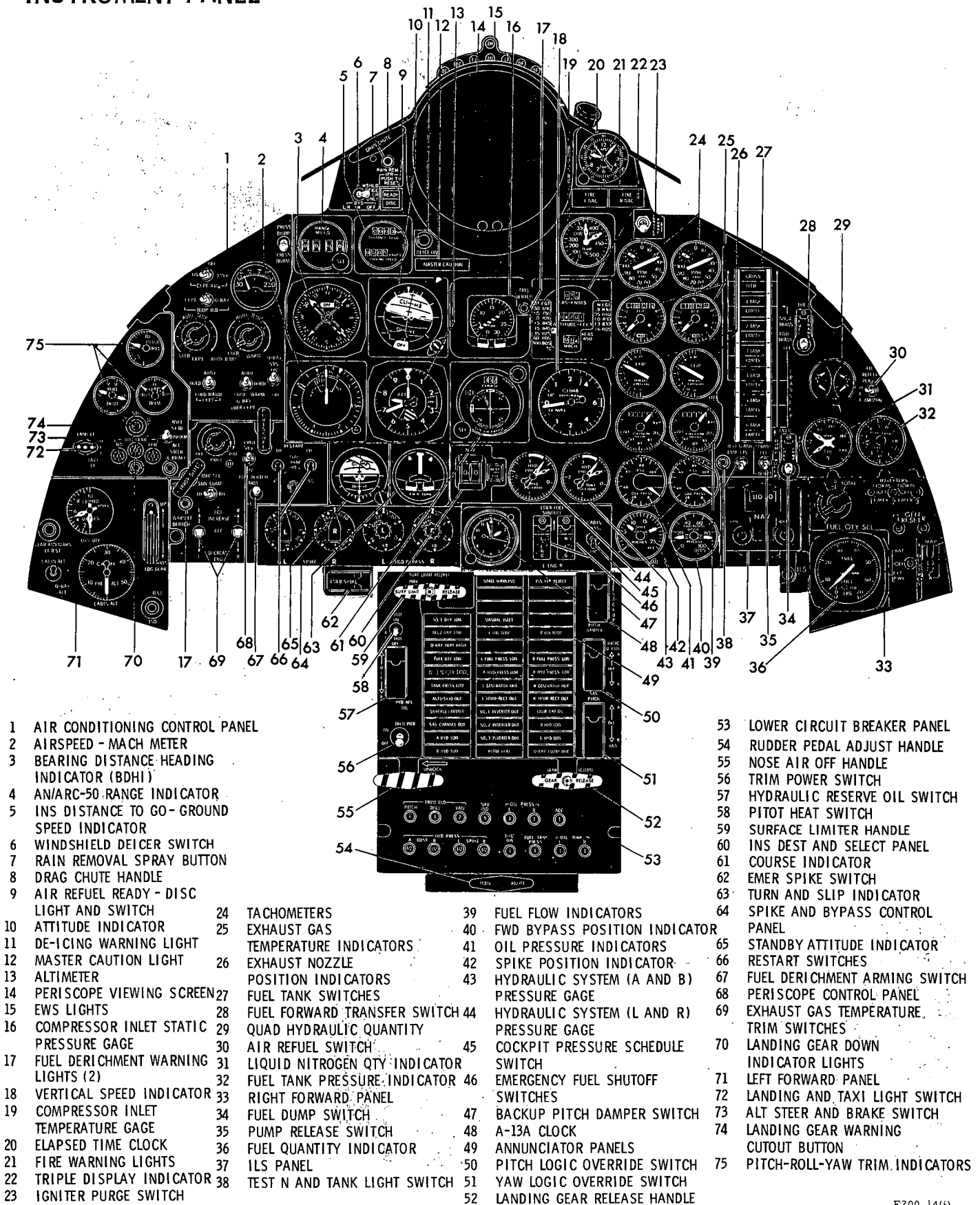


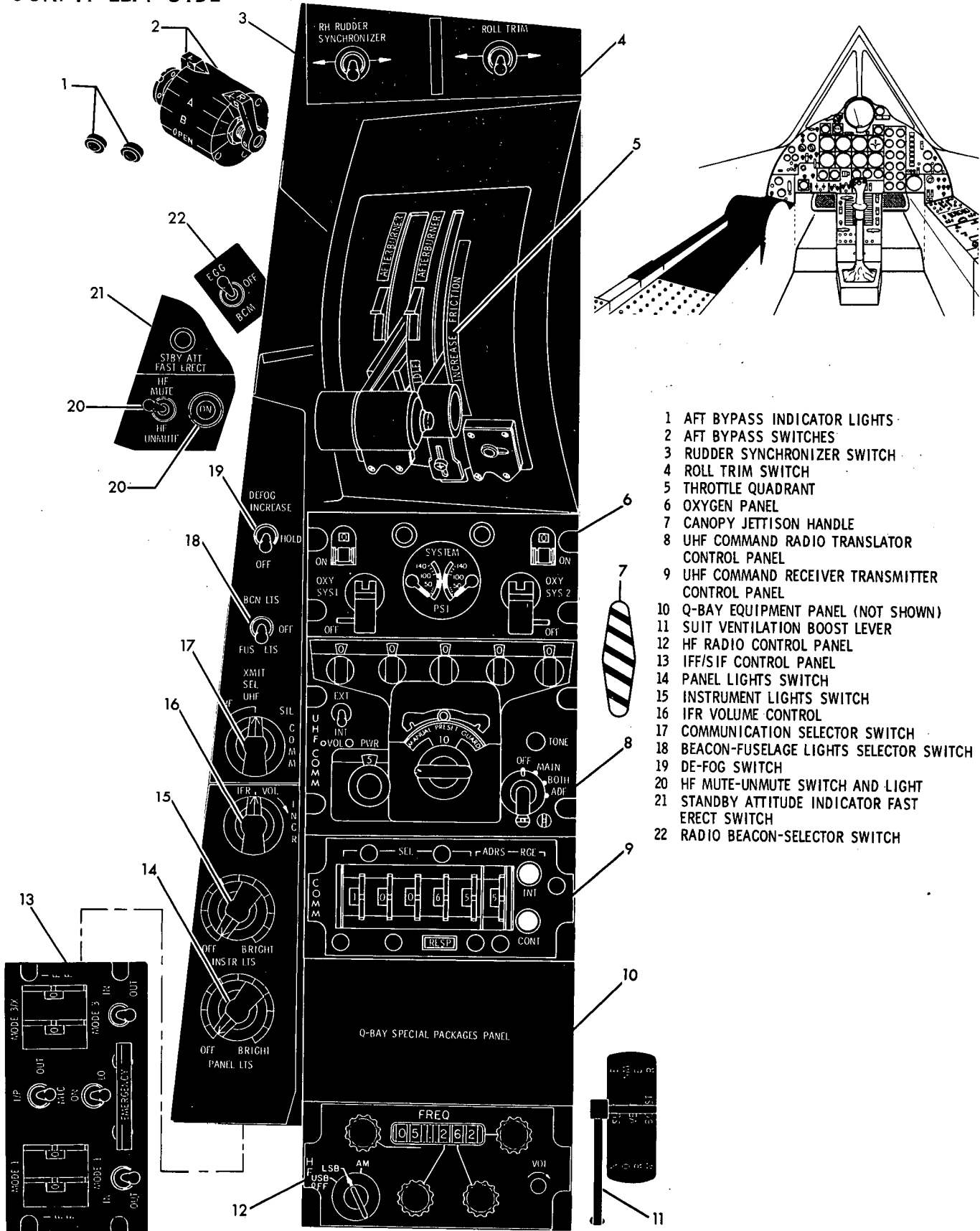
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Changed 15 March 1968

SECTION I

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COCKPIT LEFT SIDE

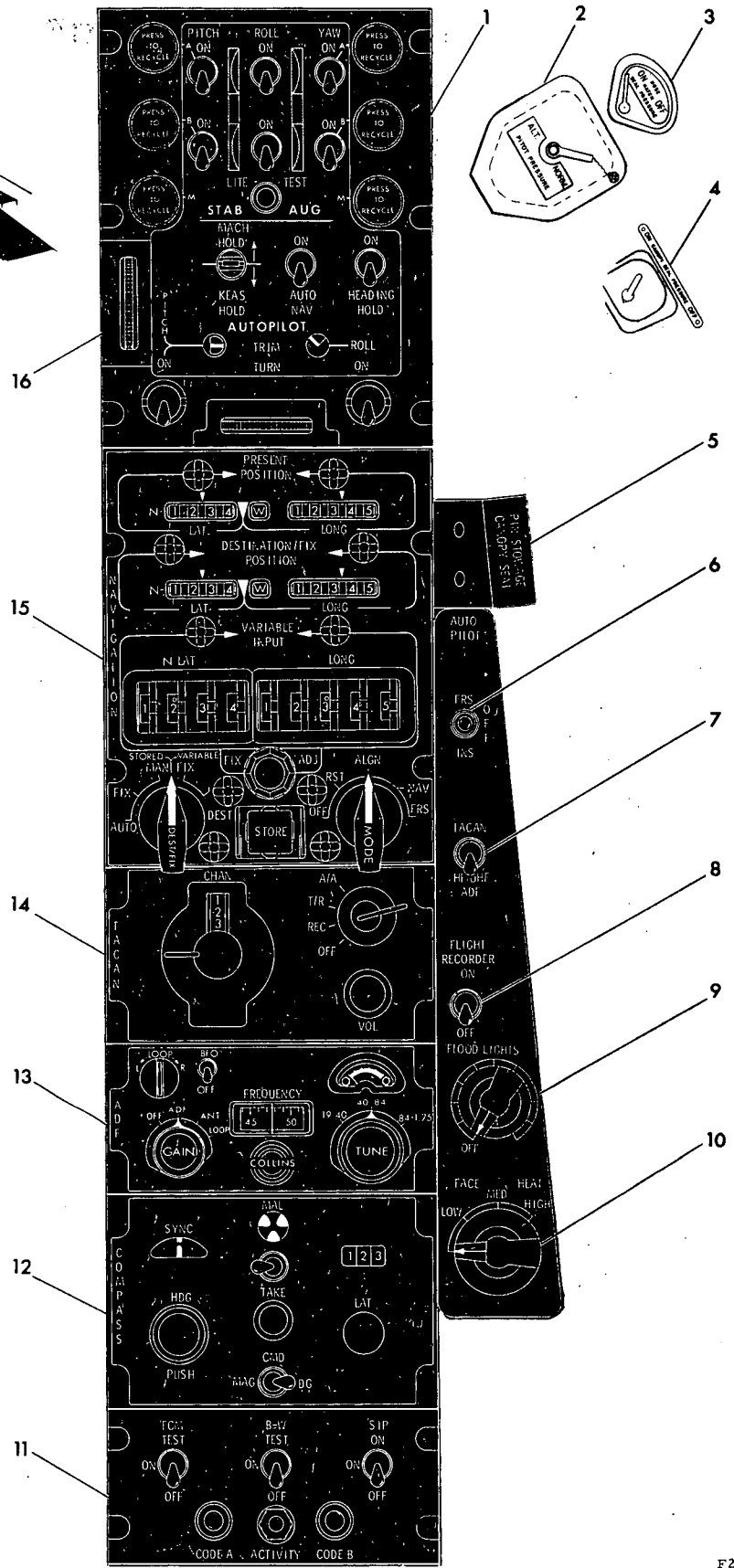
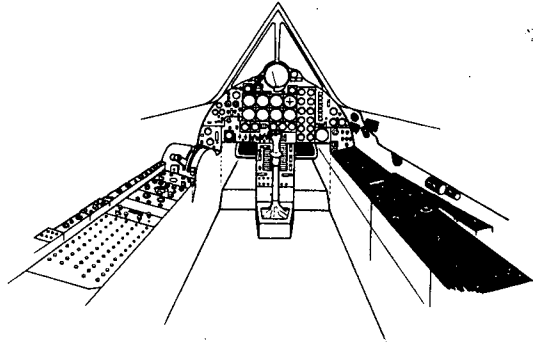


- 1 AFT BYPASS INDICATOR LIGHTS
- 2 AFT BYPASS SWITCHES
- 3 RUDDER SYNCHRONIZER SWITCH
- 4 ROLL TRIM SWITCH
- 5 THROTTLE QUADRANT
- 6 OXYGEN PANEL
- 7 CANOPY JETTISON HANDLE
- 8 UHF COMMAND RADIO TRANSLATOR CONTROL PANEL
- 9 UHF COMMAND RECEIVER TRANSMITTER CONTROL PANEL
- 10 Q-BAY EQUIPMENT PANEL (NOT SHOWN)
- 11 SUIT VENTILATION BOOST LEVER
- 12 HF RADIO CONTROL PANEL
- 13 IFF/SIF CONTROL PANEL
- 14 PANEL LIGHTS SWITCH
- 15 INSTRUMENT LIGHTS SWITCH
- 16 IFR VOLUME CONTROL
- 17 COMMUNICATION SELECTOR SWITCH
- 18 BEACON-FUSELAGE LIGHTS SELECTOR SWITCH
- 19 DE-FOG SWITCH
- 20 HF MUTE-UNMUTE SWITCH AND LIGHT
- 21 STANDBY ATTITUDE INDICATOR FAST ERECT SWITCH
- 22 RADIO BEACON-SELECTOR SWITCH

Figure 1-3

F200-17(4)

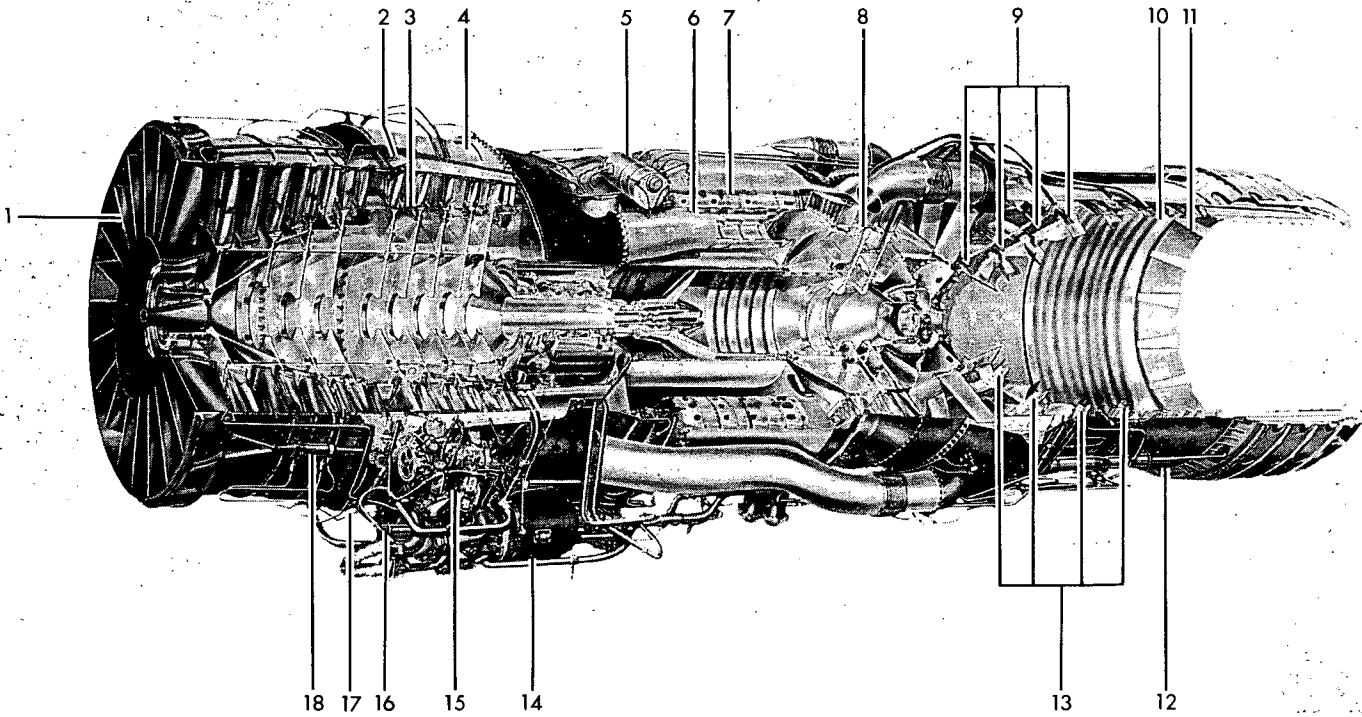
COCKPIT RIGHT SIDE



1. SAS CONTROL PANEL
2. PITOT PRESSURE SELECTOR LEVER
3. NOSE HATCH SEAL LEVER
4. CANOPY SEAL LEVER
5. SEAT AND CANOPY SAFETY PIN STOWAGE
6. AUTO PILOT SELECTOR SWITCH
7. BDHI NO. 1 NEEDLE SELECTOR SWITCH
8. FLIGHT RECORDER SWITCH
9. FLOOD LIGHT SWITCH
10. FACE PLATE HEAT SWITCH
11. B-W AND SIP CONTROL PANEL
12. FRS CONTROL PANEL
13. ADF RECEIVER CONTROL PANEL
14. TACAN CONTROL PANEL
15. INS CONTROL PANEL
16. AUTO PILOT CONTROL PANEL

Figure 1-4

F200-18(f)



- | | |
|-------------------------------------|---|
| 1 AIR INTAKE | 10 AFTERBURNER LINER |
| 2 BLEED BYPASS VALVES | 11 VARIABLE AREA EXHAUST NOZZLE |
| 3 9-STAGE COMPRESSOR SECTION | 12 EXHAUST NOZZLE ACTUATORS (4) |
| 4 STARTING BLEED VALVES | 13 FLAME HOLDERS (4) |
| 5 CHEMICAL IGNITION (TEB) RESERVOIR | 14 MAIN ENGINE GEARBOX |
| 6 BLEED BYPASS TUBES (6) | 15 ENGINE FUEL CONTROL |
| 7 BURNER CANS (8) | 16 REMOTE GEARBOX SHAFT FITTING |
| 8 2-STAGE TURBINE | 17 AFTERBURNER FUEL CONTROL (RIGHT SIDE HIDDEN) |
| 9 SPRAY BARS (4) | 18 BLEED BYPASS VALVES ACTUATOR CYLINDERS (4) |

Figure 1-5

1-29-65
FWA12-59

NOTE

See the weight and balance handbook, T.O. 1-1B-40 for information regarding specific aircraft and equipment configurations.

ENGINE AND AFTERBURNER

Thrust is supplied by two Pratt and Whitney JT11D-20A bleed bypass turbojet engines with afterburners. The interim maximum afterburning static thrust rating of each engine is 31,500 pounds at sea level with standard day conditions. The engines are designed for continuous maximum thrust operation at the high compressor inlet temperatures associated with high Mach number and high altitude operation. There is no time limit on maximum thrust operation. The engine has a single rotor, nine stage, 8:1 pressure ratio compressor utilizing a compressor bleed bypass cycle for high Mach number operation. The bypass system bleeds air from the fourth stage of the compressor, and six external tubes duct the air around the rear stages of the combustion section and the turbine. The air reenters the turbine exhaust ahead of the afterburner and is used for increased thrust augmentation. When the engine goes into bypass operation, the afterburner fuel control resets to furnish additional fuel to the afterburner. The transition to bypass operation is scheduled by the main fuel control as a function of compressor inlet temperature (CIT) and engine speed. The transition normally occurs at a CIT of approximately 150° to 190° C, corresponding to a Mach number range of 2.2 to 2.3.

Engine speed on the ground, or at low Mach numbers, varies with throttle movement from IDLE to a position slightly below MILITARY thrust. Between this throttle

position and the maximum afterburning thrust position 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 varies the afterburner fuel flow, nozzle position and thrust. At high Mach number and constant inlet conditions, the engine speed is essentially constant for all throttle positions down to and including IDLE. At a fixed throttle position, the engine speed will vary according to schedule when Mach number and CIT change.

The engine has a two stage turbine. Compressor discharge air cools the first stage and is then returned to the exhaust gas stream. Turbine discharge temperatures are monitored by indications of exhaust gas temperatures. 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 and starting bleed systems. 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. The ac generator, aircraft hydraulic pumps and fuel circulating pump are located on a remote gearbox driven by the engine power takeoff pad through a reduction gearbox.

ENGINE THRUST RATINGS

The engine thrust ratings are defined by the power lever position at the main fuel control. The power lever is mechanically linked to the throttle, providing a relationship between throttle position and thrust ratings.

SECTION I

ENGINE AND A/B FUEL SYSTEM

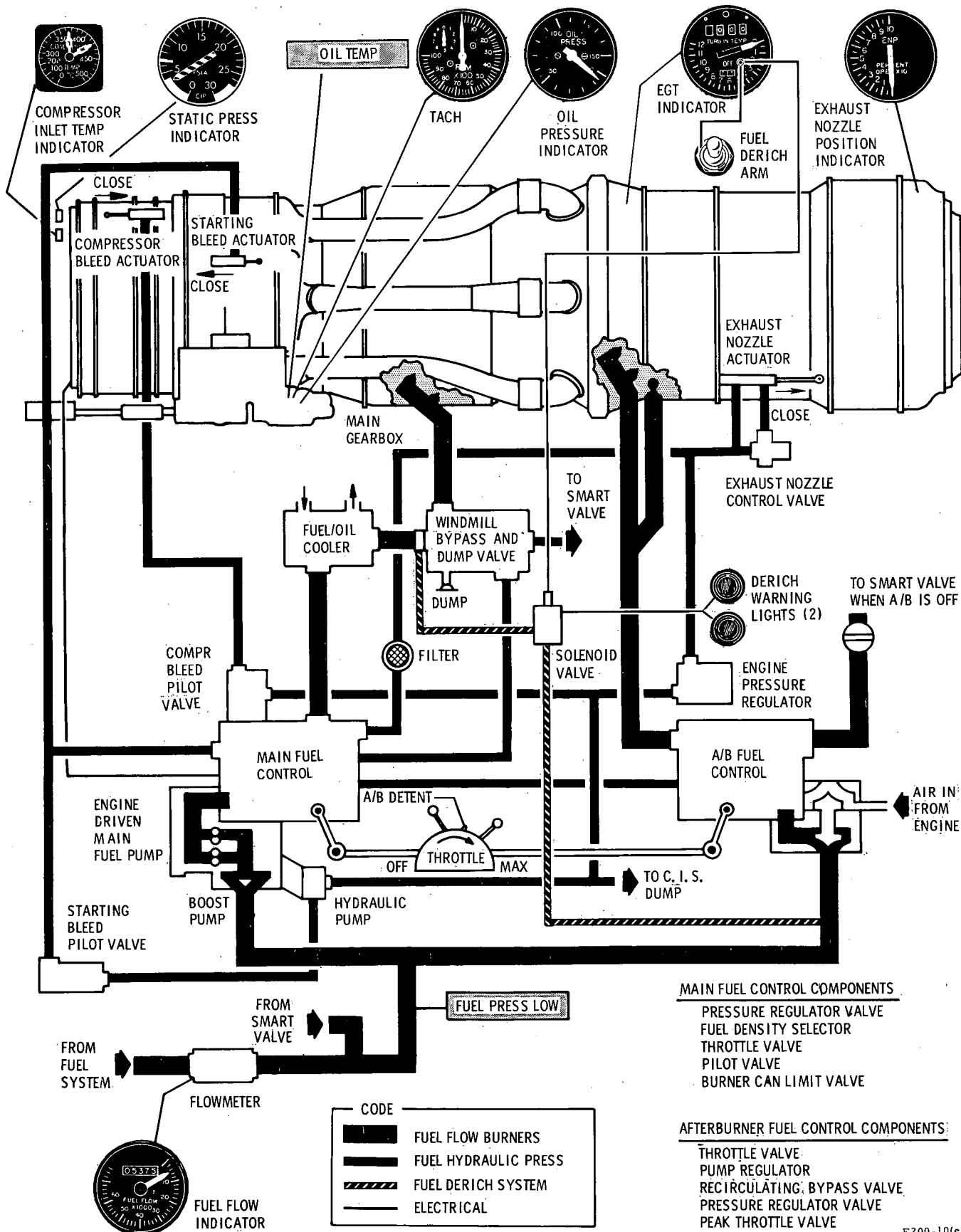


Figure 1-6

F200-10(e)

A-12

Maximum Rated Thrust

Maximum rated thrust is obtained in afterburning by placing the throttle against the quadrant forward stop.

Minimum Afterburning Thrust

MINIMUM afterburning thrust is obtained with the throttle just forward of the quadrant MILITARY thrust detent. Afterburner ignition is automatically actuated when the throttle is advanced past the detent and afterburner fuel flow is terminated when the throttle is retarded aft of the detent. Afterburning fuel flow and thrust are modulated by moving the throttle between the detent and the quadrant forward stop. Minimum afterburning is approximately 85% of maximum afterburning thrust at sea level and approximately 55% at high altitude. The basic engine operates at MILITARY rated thrust during all afterburning operation.

Military Rated Thrust

MILITARY rated thrust is the maximum non-afterburning thrust and is obtained by placing the throttle at the MILITARY thrust detent on the quadrant.

Idle

IDLE is a throttle position for minimum thrust operation. It is not an engine rating. Minimum thrust is always obtained when the throttle is at the IDLE stop on the quadrant.

Start

There is no distinct throttle position for starting. Starting is accomplished by moving the throttle from OFF to the IDLE position

as the proper engine speed is reached. This directs fuel to the engine burners by actuation of the windmill bypass valve and actuates the chemical ignition system.

Off

The aft stop on the quadrant is the engine OFF throttle position. In this position, the windmill bypass valve cuts off fuel to the burners and bypasses it back to the aircraft system. This provides engine oil, fuel pump and fuel hydraulic pump cooling when an engine is windmilling at high Mach number.

ENGINE FUEL SYSTEM

Engine fuel system components include the engine driven fuel pump, main fuel control, windmill bypass valve and variable area fuel nozzles in the main burner section.

Main Fuel Pump

The engine driven main fuel pump is a two stage unit. The first stage consists of a single centrifugal pump acting 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 one pump to operate in the event the other fails. The pump discharge pressure is determined by the regulating and metering function of the main fuel control. The maximum discharge pressure is approximately 900 psi. A relief valve is provided in the second stage discharge to prevent excessive fuel system pressure.

Main Fuel Control

The main fuel control meters main burner fuel flow, controls the bleed bypass and

SECTION I

A-12

start bleed valves and controls exhaust nozzle modulation. Thrust is regulated as a function of throttle position, compressor inlet air temperature, main burner pressure and engine speed. The bypass and start bleed valve positions are controlled as a function of engine speed biased by CIT. For steady state inlet conditions at high Mach number, the control provides essentially a constant engine speed at all throttle positions down to and including IDLE. On the ground and at lower Mach numbers, engine speed varies with throttle position from slightly below MILITARY down to IDLE. Afterburner operation is always at MILITARY rated engine speed and EGT. The fuel control is provided with a pilot operated 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 is actuated by signals from the main fuel control. The valve also opens to drain the fuel manifold when the engine is shut down.

Fuel Nozzles

The engine has eight can-annular type combustion chambers with forty-eight variable area dual orifice fuel nozzles in clusters of six nozzles per burner. The nozzles have 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 orifice pressure drop.

ENGINE FUEL DERICHMENT SYSTEM

The derichment system provides protection against severe turbine over-temperature during high altitude operation. When EGT indicates 860°C or more with the system armed, the fuel:air ratio in the engine burner cans is reduced, or deriched, below normal values. This is accomplished by a solenoid operated valve and orifice which bypasses metered engine fuel from the fuel oil cooler to the afterburner fuel pump inlet. The solenoid valve is actuated by a signal from the EGT gage when 860°C is reached. Once actuated, it remains open until the system is turned off. Two warning lights are provided to indicate when the valve is open. Power for the derich circuits is provided from the essential dc bus.

Fuel Derichment Arming Switch

A two position fuel derichment arming switch is located on the left side of the instrument panel. In the ARM (up) position the derichment circuits are armed and the respective derichment solenoid valve will open automatically and remain open if the EGT reaches 860°C. In the OFF position the derichment solenoid valve is closed and the system can not provide derichment flow. Power is furnished from the essential d. c. bus.

Fuel Derichment Warning Lights

The fuel derichment warning lights, located on the left and upper center of the instrument panel, illuminate and remain on while the derichment solenoid valve is open. The lights will be extinguished when the arming switch is placed in the OFF position and will remain extinguished when the arming switch is reset to ARM if both EGTs are below 860°C.

WARNING

- In the event of derichment the arming switch must be placed in the OFF position prior to relighting the afterburner to prevent engine speed suppression and subsequent inlet unstart. If engine flameout is experienced with inlet unstart the arming switch should also be placed to OFF prior to relighting the engine.
- Derichment at sea level will cause a thrust loss of approximately 5% if in maximum afterburning or 7% if at Military. Approximately 45% loss in thrust and 600 rpm speed suppression will occur during cruise with maximum afterburning.

AFTERBURNER FUEL SYSTEM

Afterburner fuel system components include the centrifugal afterburner fuel pump, afterburner fuel control and spray bars.

Afterburner Fuel Pump

The afterburner fuel pump is a high speed, single stage centrifugal pump. The pump is driven by an air turbine which is 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. The turbine is protected from overspeed by an aero-dynamic speed limiting air discharge venturi.

Afterburner Fuel Control

The afterburner fuel control is a hydro-mechanical fuel control which schedules metered fuel flow as a function of throttle position, main burner pressure and compressor inlet temperature. Fuel flow is

metered on a predetermined schedule to provide fuel into both zones of the afterburner spray bars simultaneously. The control incorporates a reset mechanism which increases the afterburner fuel flow when the bypass valves open and decreases the fuel flow when the valves close.

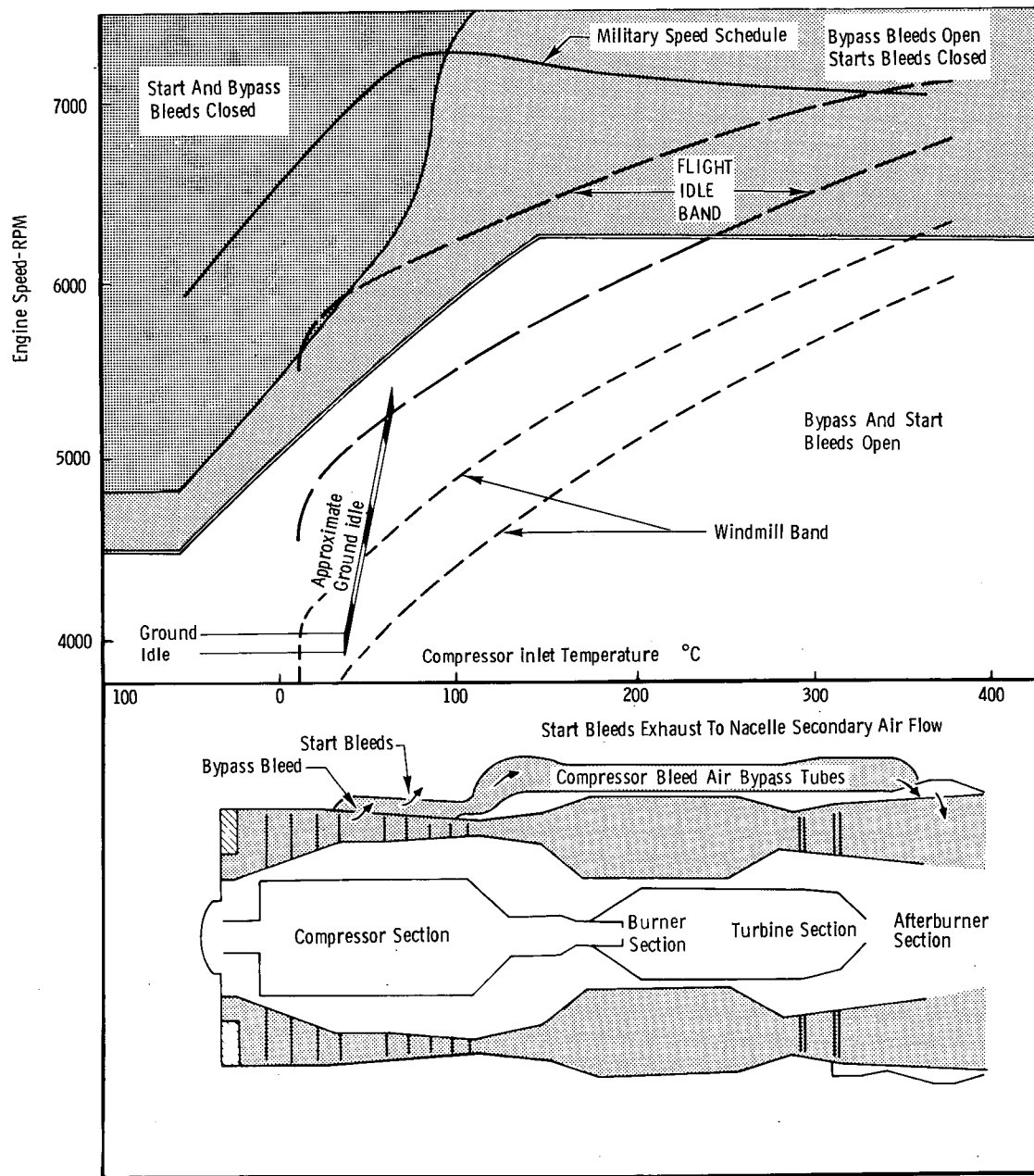
ENGINE FUEL HYDRAULIC SYSTEM

Each engine is provided with a fuel hydraulic system for actuation of the afterburner exhaust nozzle and the start and bypass bleed valves. Engine hydraulic system pressure is also required to dump the unused chemical ignition fluid. Pressure is supplied by a high temperature, engine driven, variable delivery, piston type pump. The pump maintains system pressures up to 2500 psi with a maximum flow of 50 gpm 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 non-afterburning recirculation line and the windmill bypass valve discharge line. This fuel is returned to the aircraft 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 from the main fuel pump.

Exhaust Nozzle Actuation

The exhaust nozzle control and actuation system is composed of four actuators to move the exhaust nozzle, and an exhaust nozzle control modulating the hydraulic pressure to 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. A pressure regulator is contained in a separate unit located near the exhaust nozzle control.

START BLEED AND BYPASS VALVE ACTUATION



F200-96

Figure 1-7

Start and Bypass Bleed Valve Actuation

The bypass bleed control and actuation system consists of four two-position actuators to move the bleed valves, and a pilot valve 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 compressor inlet temperature. The starting bleed control and actuation system is similar to the bypass bleed system except that three actuators are used and the pilot valve controls starting bleed valve position in response to the main fuel pump boost stage pressure rise.

EXHAUST NOZZLE AND EJECTOR SYSTEM

The variable area, iris type, exhaust 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 exhaust nozzle is enclosed by a fixed contour, convergent-divergent ejector nozzle to which free floating trailing edge flaps are attached. In flight, the inlet cowl bleed supplies secondary airflow between the engine and nacelle for cooling. During ground operation, suck in doors in the aft nacelle area provide cooling air. Free floating doors around the nacelle, just forward of the ejector, supply tertiary air to the ejector nozzle at subsonic Mach numbers. The tertiary doors and trailing edge flaps open and close with varying internal nozzle pressure, which is a function of Mach number and engine thrust.

Exhaust Nozzle Position Indicator

Each engine is provided with a nozzle position indicator located on the right side of the instrument panel. The indicators are marked from 0 to 10 and indicate the ap-

proximate percentage of open position. Additional dot markings above and below the 0 and 10 position marks are for calibration purposes. The indicators are remotely operated by electrical transducers located near the exhaust nozzles. Each transducer is cooled by fuel and is operated by the afterburner nozzle feedback link. Power for the indicators is supplied by the No. 1 inverter.

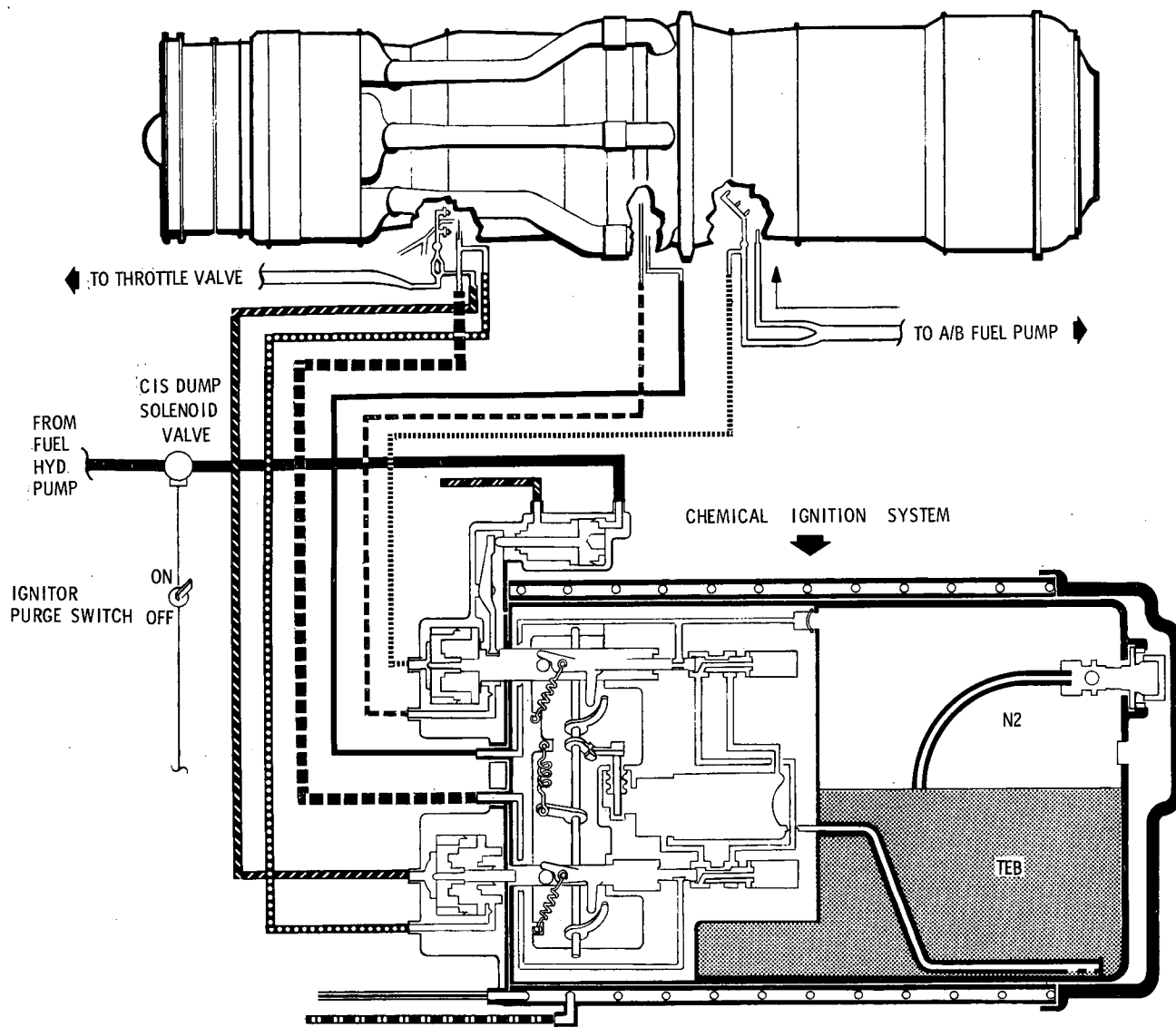
OIL SUPPLY SYSTEM

The engine and reduction gear box 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 and has a usable capacity of 4.5 gals. Total tank capacity is 6.7 gals. 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. From the fuel-oil cooler the oil is distributed to the engine bearings and gears. Oil screens are installed at the lubricating jets for additional protection. Scavenge pumps return the oil to the tank where it is deaerated. The main oil pump normally maintains an oil pressure of 40 to 55 psi. A pressure regulating valve keeps flow and pressure relatively constant at all flight conditions. Because of the high viscosity of the oil, it is diluted with trichloroethylene at lower temperatures and special cold weather shut down procedures may be required.

Main Fuel-Oil Cooler

This unit provides oil cooling by using engine fuel to absorb the heat. The oil temperature is controlled by fuel flow through the cooler. A bypass valve is incorporated to bypass fuel around the cooler when the fuel flow is greater than the cooler flow capacity of approximately 12,000 pounds per hour.

CHEMICAL IGNITION SYSTEM



CODE		
MAIN BURNER IGNITION	■■■■■■■■	COMPRESSOR DISCHARGE PRESSURE
MAIN IGNITION SIGNAL	▨▨▨▨▨▨	FUEL COOLING OUT
DUMP SIGNAL	▬▬▬▬▬▬	A/B IGNITION LINE
DUMP SIGNAL DRAIN	▨▨▨▨▨▨	TURBINE DISCH PRESS
FUEL COOLING IN	▬▬▬▬▬▬	IGNITION SIGNAL
		ELECTRICAL

F200-11(b)

Figure 1-8

Oil Quantity Low Lights

An indicator light for each engines oil system is located on the lower instrument annunciator panel. The lights are labeled L and R OIL QTY LOW and illuminate when the respective engine oil quantity is reduced to 2.25 gals. Power is furnished by the essential dc bus.

Engine Oil Temperature Light

L and R OIL TEMP lights are installed on the annunciator panel. These lights will illuminate when respective engine oil inlet temperature is less than $+15.6^{\circ} \pm 3^{\circ} \text{C}$ or greater than $282^{\circ} \text{C} + 11^{\circ} \text{C}$.

Remote Gear Box Oil System

The remote gear box contains an independent, wet sump lubricating system with its own oil supply and pressure pump. There is no scavenge pump. It is vented to the engine breather system through the remote gear box drive shaft. The oil is cooled by circulation through the remote gear box fuel-oil heat exchanger.

CHEMICAL IGNITION SYSTEM

Triethylborane (TEB) is used for ignition of main burner and afterburner fuel. Special handling procedures are required because TEB above 0°F will burn spontaneously upon exposure to air above -4°F . The TEB is contained in a 600 cc (1-1/4 pint) storage tank pressurized with nitrogen. The nitrogen provides inerting and operating pressure to supply a metered quantity of TEB to either the main burner or afterburner section. Operation is in response to a fuel pressure signal from the appropriate system. Actuation is automatic with throttle movement. A mechanical counter for each engine, located aft of the throttles, indicate TEB shots remaining. A minimum of 16

injections can be made with one full tank of TEB. The TEB tank is engine mounted and is cooled by main burner fuel to maintain the TEB temperature within safe limits. If the TEB vapor pressure exceeds a safe level, a rupture disc is provided to discharge the vaporized TEB and tank nitrogen through the afterburner section. No pilot indication of TEB tank discharge is provided. The engine is also equipped with catalytic igniters installed on the afterburner flameholders to provide improved afterburner ignition system reliability and re-light capability. Turbine exhaust temperature must be above approximately 730°C to obtain a satisfactory afterburner "light" by the catalytic igniters.

Igniter Purge Switch

A lift-lock toggle switch labeled IGNITER PURGE is installed on the upper right side of the instrument panel. When the switch is pulled out and held in the up position a solenoid operated valve supplies fuel hydraulic system pressure to the chemical ignition system dump valve. This allows the remaining TEB to be dumped into the afterburner section; while the engine is running. Approximately 40 seconds is required. Electrical power is furnished by the essential dc bus.

NOTE

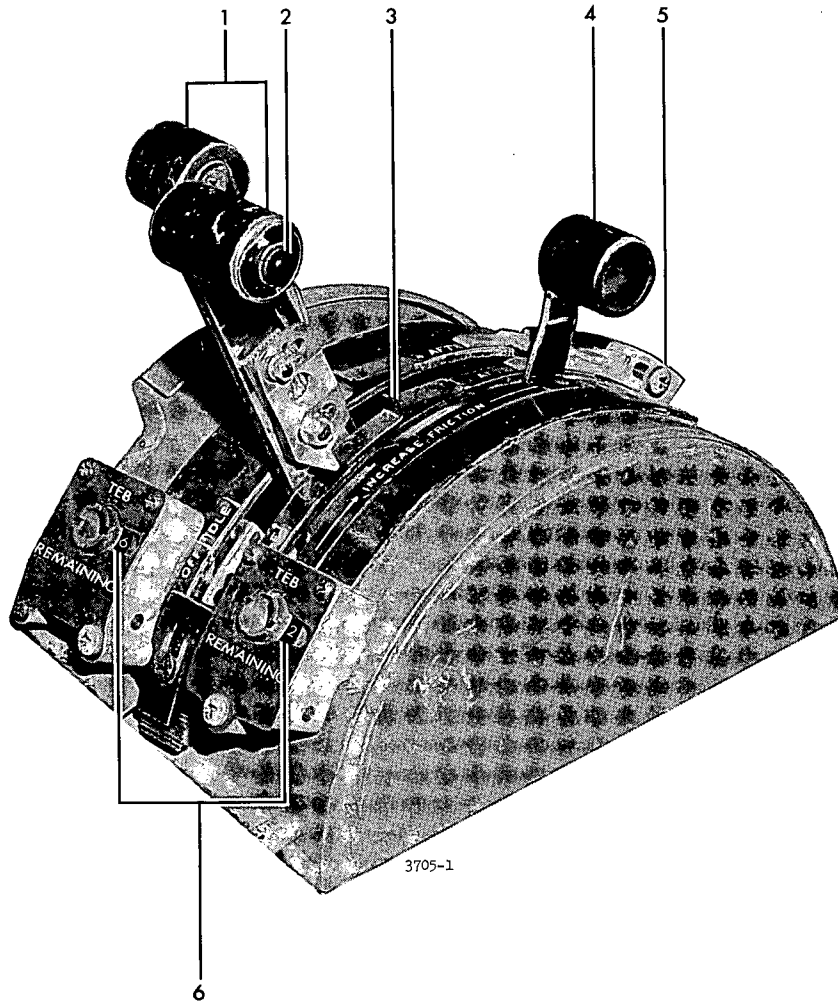
Both electrical power and engine fuel hydraulic pressure are necessary to purge the TEB system. If the engine is not rotating the system will not normally dump.

CAUTION

Do not actuate the Igniter Purge switch unless the engine is rotating in the 5000-6000 rpm range to prevent damage to the afterburner flame holders.

SECTION I

THROTTLE QUADRANT



- 1 THROTTLES
- 2 TRANSMIT BUTTON
- 3 MILITARY DETENT
- 4 THROTTLE FRICTION LEVER
- 5 MAX AFTERBURNER STOP
- 6 TEB SHOT COUNTERS

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FMA12-13-(a)

Figure 1-9

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STARTER SYSTEM

A starter cart is provided for ground starts. This may be either a self-contained gas engine cart or multiple air turbine cart. The output drive gear of either cart connects to a starter gear on the main gear box 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 signals from the pilot. Air starts are made by windmilling the engine.

THROTTLES

Two throttle levers, one for each engine, are located in a quadrant on the left forward console. The right throttle is mechanically linked to the right engine main fuel control and the left throttle to the left engine afterburner fuel control with parallelogram type linkages designed to compensate for heat expansion. The afterburner and main fuel controls are interconnected by a closed loop cable. The throttle quadrant is labeled OFF, IDLE and AFTERBURNER. When the throttles are moved forward from OFF to IDLE, they drop over a hidden ledge to the IDLE position. This ledge prevents inadvertent engine cutoff when the throttles are retarded to IDLE. When retarding the throttles from IDLE to OFF they must be lifted in order to clear the IDLE stop ledge. Forward throttle movement from IDLE to a MILITARY stop controls the non-afterburning thrust range of the engine. The throttles must be slightly raised to clear the stop before additional forward movement of the throttle can actuate the afterburner ignition. The AFTERBURNER range extends from the Military stop to the quadrant forward stop. The right throttle knob incorporates a radio transmission push-button switch. Throttle quadrants are marked to indicate 82° power lever angle (PLA) for assistance in determining the cruise power position.

Throttle Friction Lever

The throttles are prevented from creeping by a friction lever located on the inboard side of the throttle quadrant. When the lever is full aft, the throttles are free to move. Moving the lever forward as the INCREASE FRICTION label indicates, progressively increases the amount of friction to hold the throttles in the desired position.

TEB Remaining Counters

A mechanical TEB remaining counter for each engine is located aft of each throttle. The counters are spring wound and set to 12 prior to engine start. Each time a throttle is moved forward from OFF to IDLE or MILITARY to A/B the counter will reduce one number.

Exhaust Gas Temperature Trim Switches

Individual exhaust gas temperature trim switches for each engine are located on the lower left side of the instrument panel. The switches are spring loaded, momentary contact, three position switches with a center OFF position. When held in the INCREASE (up) position, a remote trim electric motor on the engine fuel control is actuated to slightly increase main burner fuel flow and turbine inlet temperature. The trim motors have a fuel flow or EGT travel range of about 150°C and a rate of change of 8°C per second. When held in the DECREASE (down) position, the trim motor reduces the fuel flow and decreases turbine inlet temperature. An increase or decrease in turbine inlet temperature will be reflected on the respective exhaust gas temperature gage. These switches are the only provision for main engine control when the throttles are in the afterburning range. They have no effect on rpm when the nozzle is modulating to provide the scheduled engine speed. Power for the trim motors is furnished by the respective ac generator bus.

SECTION I

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ENGINE INSTRUMENTS

Exhaust Gas Temperature Gages

Two exhaust gas temperature gages, one for each engine, are mounted on the right side of the instrument panel. They are calibrated in degrees centigrade from 0°C to 1200°C and indicate the temperature sensed by turbine discharge thermocouples. The four digit windows at the top of the gages indicate the exhaust gas temperature to the nearest degree. An OFF window at the bottom of each dial when visible indicates instrument power failure. A small red light on the dial will light when EGT reaches 860°C. This will activate the respective derichment system if armed. The indicating system receives power from the No. 1 inverter.

Fuel Flow Indicators

A fuel flow indicator for each engine is mounted on the instrument panel and displays total fuel flow (engine and afterburner) in pounds per hour. The dial is calibrated in 2000 pound per hour increments to 76,000 pph. The five digit center window indicates the fuel flow to the nearest 100 pph. The indicator is not compensated for return flow and indicates total fuel flow to engine, afterburner and heat sink system. A positive indication is normal during windmill operation and the indicator will read high when the mixer and temperature control valve is diverting cooling loop fuel back to tank 4. During descent after high speed cruise both high and low fuel flows and flow oscillations may be indicated. Power for the indicators is supplied by the No. 1 inverter.

Tachometers

A tachometer for each engine is mounted on the right side of the instrument panel. The tachometers indicate engine speed in revolutions per minute. The main pointer is cali-

brated up to 10,000 rpm and the subpointer makes one complete revolution for each 1000 rpm. The tachometers are self-energized and operate independently of the aircraft electrical system.

Engine Oil Pressure Gages

An oil pressure gage is provided for each engine on the right side of the instrument panel. The gages indicate output pressure of the respective engine oil pump in pounds per square inch. The gages are calibrated from 0 to 100 psi in increments of 5 psi. Power for the gages is furnished by the No. 1 inverter bus through the 26-volt autotransformer.

Compressor Inlet Temperature Gage

A dual indicating compressor inlet temperature gage is mounted on the upper right side of the instrument panel. It is calibrated in 50° increments from 0°C to 300°C and 10° increments from 300°C to 500°C. The needles indicate the air temperature forward of the first compressor stage of each nacelle. The system uses platinum resistance sensors and power is furnished by the No. 1 inverter.

Compressor Inlet Air Static Pressure Gage

A dual indicating compressor inlet air static pressure gage located on the upper center of the instrument panel, measures absolute pressure at the engine compressor inlet. The gage is calibrated in one psi increments and has marked red ranges from 0 to 4 psi and 27 to 30 psi and a green radial mark at 7 psi. A white striped third pointer on the CIP gage indicates pressure to be expected when the inlets are operating normally if over Mach 1.8 and 250 KEAS. The L and R labeled pointers indicate actual inlet static pressures. Power is furnished from the No. 1 inverter.

AIR INLET SYSTEM

The air inlets for each nacelle are canted inboard and down to align with the local air-flow pattern. The inlet system consists of the cowl structure, a moving spike to help provide optimum internal airflow characteristics, a spike porous centerbody bleed and an internal cowl shock trap bleed for internal shock wave position and boundary layer flow control, forward and aft bypass doors for control of airflow in the inlet and to the engine, cowl exhaust louvers, system controls, sensors, actuators and instrumentation. Suck-in doors are also provided in the aft nacelle area for ground cooling. Nacelle cooling air is provided in flight by air from the cowl shock trap bleed and aft bypass. Normally, the spike and forward bypass are operated automatically by the air inlet control system. Inlet airflow is controlled so that the proper amount of air is supplied to the engine and, at supersonic airspeeds, the positions of shock waves ahead of the inlet and in the inlet throat are controlled so as to provide maximum practical ram pressure recovery at the engine face. Controls are provided in the cockpit for incremental control of the aft bypass for those conditions where additional bypass area is required or where a reduction in forward bypass flow is desired. Manual controls are provided to override the automatic spike and forward bypass control systems.

INLET SPIKE

The spike is locked in the forward position for ground operation and flight below 30,000 feet. It is unlocked above this altitude and is programmed during automatic operation to move 1/4 inch off the forward stop at Mach 1.4. Above Mach 1.6, the spike retracts so as to increase the nacelle inlet area and decrease the area of the throat or narrowest portion of the duct. Spike position is scheduled primarily as a function of

Mach number as sensed by the Rosemount boom pitot static ports with biasing for angle of attack and yaw angle. The spike moves aft approximately 26 inches during transition between Mach 1.6 and 3.2. The inlet control also includes a shock expulsion sensor (SES) and restart feature which can operate automatically when speeds for inlet scheduling are reached. It is effective above approximately Mach 2.0. If an inlet becomes unstable and expels the internal shock, the shock expulsion sensor for that inlet overrides the automatic spike and forward bypass schedule. It causes the forward bypass to open fully and the spike to move forward as much as 15 inches. Spike retraction is started automatically 3.75 seconds after the expulsion is sensed and, when schedule position is reached, the forward bypass is returned to automatic operation. The SES reference pressure is CIP, and the system is triggered when a momentary decrease of CIP is 23% or more. This is a characteristic CIP indication of inlet unstart occurrence. However, it may also operate as a result of pressure fluctuations if CIP decreases rapidly below the previous normal condition during compressor stalls. The SES feature does not override a manually operated spike or forward bypass control. Manual operation of a restart switch overrides the SES operation for that inlet. Spike and forward bypass door position changes may be observed during SES operation on the spike and forward bypass position indicators. Local pitch attitude and yaw angle are sensed by a pressure probe mounted on the Rosemount pitot boom. The spike porous centerbody bleeds boundary layer air from the inlet throat to prevent flow separation. This air is ducted overboard through the supporting struts and nacelle louvers. The spikes can be fully controlled by use of cockpit controls when hydraulic pressure is available.

Emergency spike forward

SECTION I

A-12

switches provide pneumatic pressure to move and lock the spikes forward in the event of hydraulic system failure.

INLET FORWARD BYPASS

The forward bypass provides an exhaust for inlet air which is not required by the engine, and controls the inlet diffuser pressure so as to properly position the inlet shock. It consists of a rotating basket which opens duct exhaust ports located a short distance aft of the inlet throat. When the landing gear is down, the forward bypass doors are held open by an electrical override signal from a landing gear door switch. The switch is positioned to allow manual or automatic control of the bypass when the landing gear retracts. In automatic operation, the forward bypass remains closed until a low supersonic speed is reached, then it modulates in accordance with air inlet control system Mach and pressure schedules. The inlet usually "starts" at Mach 1.4, that is, the inlet shock is positioned near the cowl shock trap bleed in the inlet throat area. As speed is increased, the forward bypass schedules as required to maintain the inlet shock at the throat position.

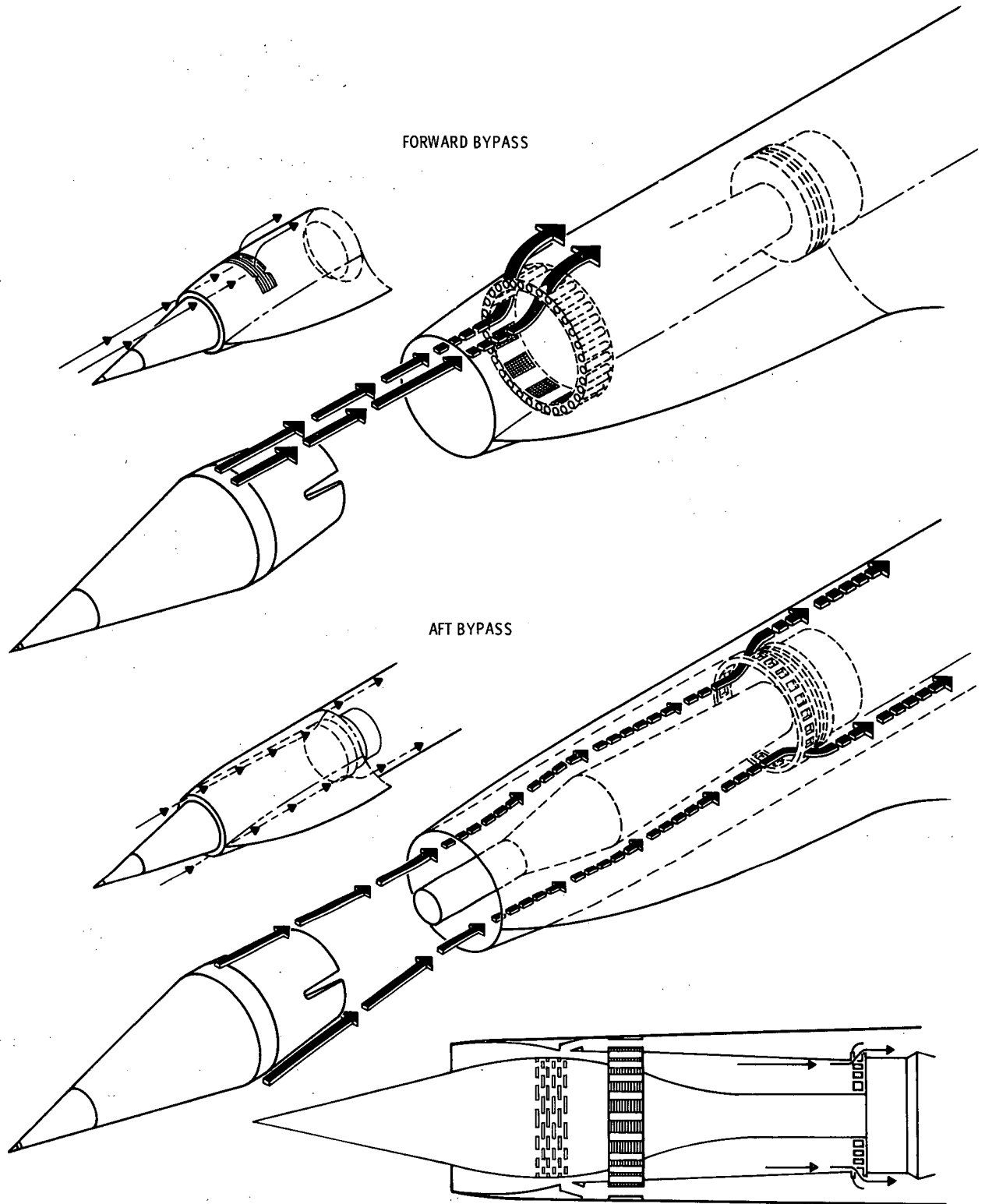
The forward bypass position is controlled by the ratio of inlet duct static pressure to a reference total pressure. The inlet duct static pressure is sensed by taps located aft of the shock trap bleed.

The reference total pressure is sensed by two external probes one located on the lower inboard side of the nacelle and the other at the top of the nacelle. The forward bypass control also senses an unstart as a result of the sudden decrease in pressure at the engine face and controls the inlet through a timed sequence. The minimum Mach number at which the automatic re-start actuates varies with the intensity of the unstart but is generally in the vicinity of Mach 2.0. An overriding switch holds the forward bypass closed at speeds lower than Mach 1.4.

INLET AFT BYPASS

The aft bypass consists of a ring of adjustable peripheral openings allowing a maximum mass flow of approximately 3/4 of that available from the forward bypass. The ring is located just forward of the engine face. These openings allow excess inlet air to be bypassed around the engine. The bypassed air joins cowl shock trap bleed air and passes between the outside of the engine and afterburner and the inside of the nacelle. This flow augments the exhaust gas in the ejector area. Each aft bypass ring is positioned by a hydraulic actuator which is powered by the respective L or R hydraulic system and is controlled by the cockpit switch. The bypass is held closed during takeoff and landing by an electrical signal from the nose gear downlock. It is also closed during subsonic operation. Position in flight is set manually in accordance with determined Mach number and engine operating requirements.

INLET AIRFLOWS

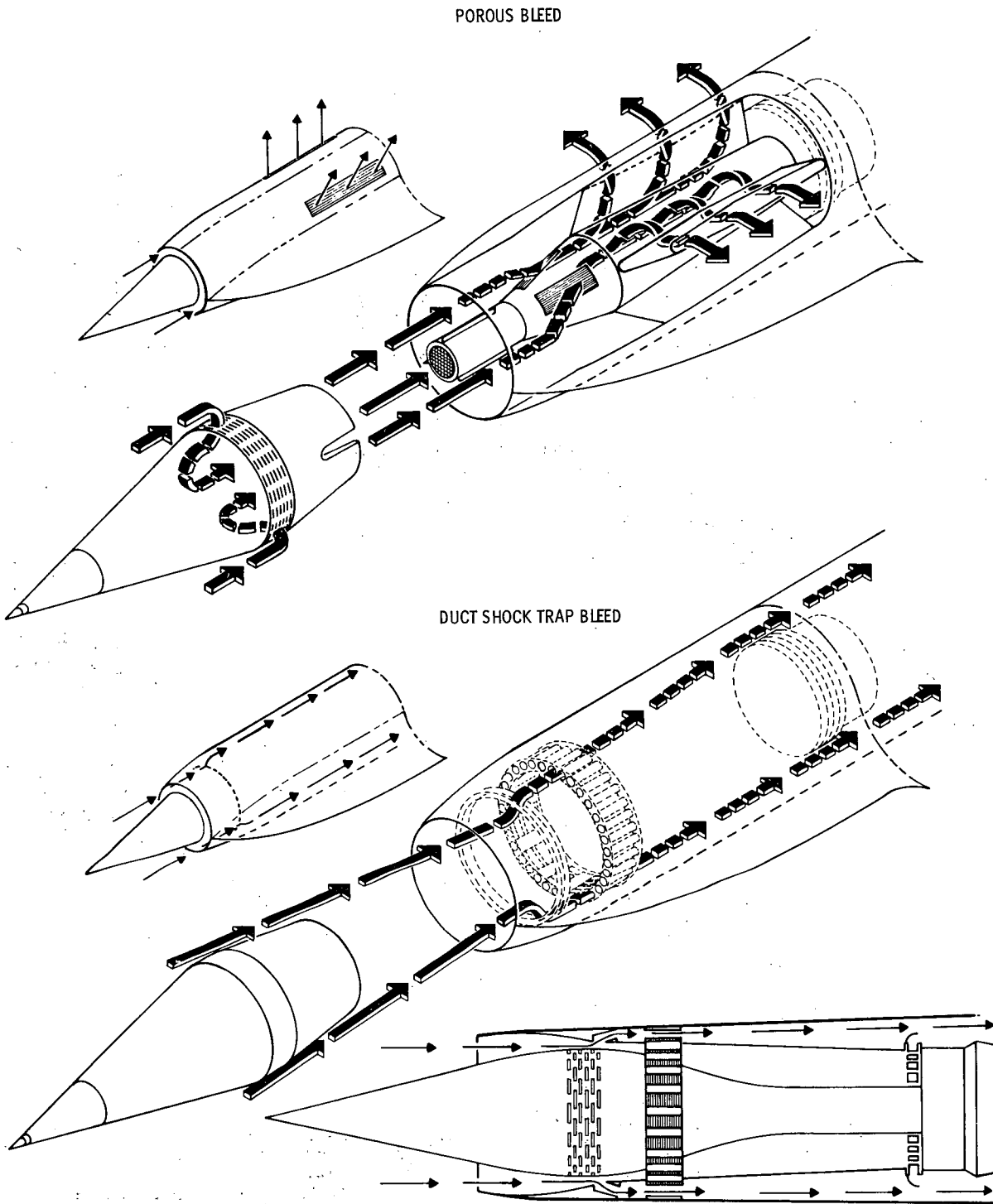


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Figure 1-10 (Sheet 1 of 2)

SECTION I

INLET AIRFLOWS



POROUS BLEED

DUCT SHOCK TRAP BLEED

NOTE

DUCT SHOCK TRAP BLEED AIR FLOWING THROUGH THESE TUBES REACHES NACELLE SECONDARY AREA AND EXHAUSTS THROUGH EJECTOR.

10-4-65
F200-71(2)(a)

Figure 1-10 (Sheet 2 of 2)

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AIR INLET CONTROL SYSTEM

The air inlet control system incorporates a computer which utilizes electrically transmitted pneumatic pressure signals to automatically schedule and reposition the spikes and forward bypass. The computer also serves as a calibrated path for the manual spike and manual forward bypass control. Major components for each inlet control are the computer, pressure transducer, angle transducer and two pressure ratio transducers. The spike and forward bypass controls consist of four rheostat type knobs and two inlet restart switches and an emergency spike switch. Aft bypass control is by means of two rotary type switches located above the throttle. Three annunciator panel lights are pertinent to the inlet control system.

Nine different pressures are sensed for inlet control. The Rosemount airspeed boom provides pitot total and static pressures to the pitot pressure transducer. The pitch and yaw attitude probe on the left side of the boom provides angle of attack and yaw angle pressures for conversion to electrical signals by the attitude transducer. At each nacelle local pitot pressure and two inlet duct static pressures are sensed to enable two sensors within the pressure ratio transducer to convert pressure ratios to electrical signals which (1) direct forward bypass control, and (2) cause an automatic restart following shock expulsion. Some control functions are also accomplished within the pressure transducer. Most of the electrical outputs of the pitot pressure transducer, attitude transducer, and both pressure ratio transducers are transmitted to the computer. The computer also receives a signal from the main landing gear doors to assure that the forward bypass will be open whenever the main gear is down.

Spike Controls

The L and R spike controls are located on the lower left side of the instrument panel. The controls are labeled AUTO, FWD, and have labeled marks for 1.4, 1.8, 2.2, 2.6, 3.0 and 3.2 Mach numbers. Intermediate marks for 0.1 Mach increments allow the knobs to be positioned manually at any setting from 1.4 to 3.2 Mach number. In the detented AUTO position, spike position is scheduled automatically by the inlet control system. In the detented FWD position, the spike will move to the full forward position. The Mach numbered positions are used in manual operation. Use of settings corresponding to aircraft flight Mach number moves the spike aft to the correct position for proper inlet performance. The spike control also biases the forward bypass as a function of control knob position when the bypass is being manually controlled. The forward bypass position indicator and bypass control knob will not be in agreement by the amount of bias. Control power for the left spike is from the No. 2 inverter and for the right spike the No. 3 inverter.

Forward Bypass Controls

The L & R BYPASS controls are located just inboard of the spike controls. When a control is turned full counterclockwise to the detented AUTO position, operation of the respective forward bypass is automatically controlled by the inlet computer. As the control is turned clockwise the first detented position will position the forward bypass to the full open. As the control is turned further clockwise the forward bypass will incrementally move towards the closed position and will be fully closed in the full clockwise position. Markings from 0 to 100 in increments of 10 percent allow the control to be positioned at any percentage of full open. Power for the circuits is from the essential dc bus and No. 2 and No. 3 inverters.

SECTION I

AIR INLET CONTROLS AND INDICATORS

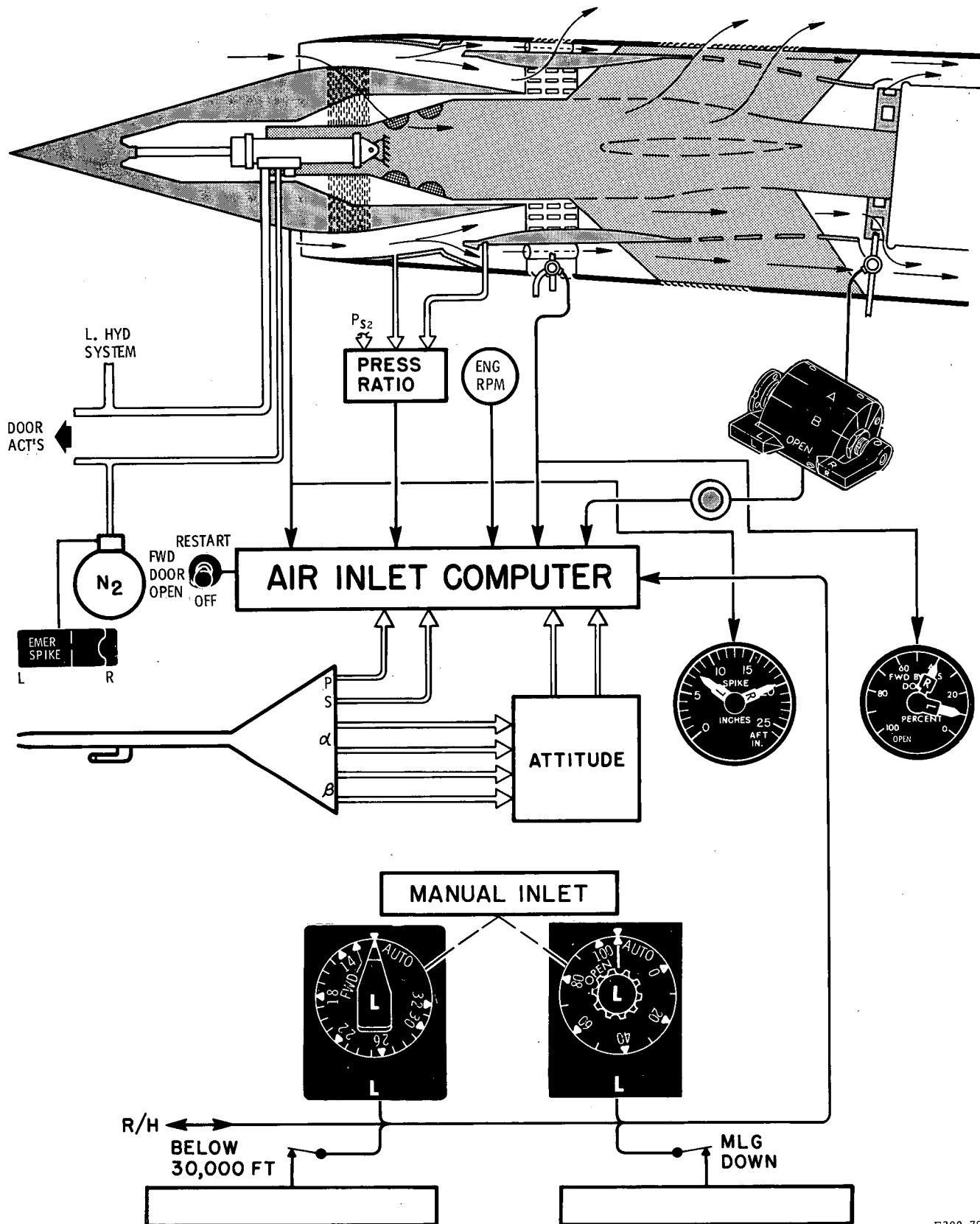


Figure 1-11

CAUTION

Manual operation of the forward bypass is permissible with the spike operating on its automatic schedule; however, when the spike is operated manually, the forward bypass must be operated manually or the bypass will open fully and will not schedule.

Inlet Restart Switches

Two 3-position toggle switches are located on the left side of the instrument panel. The L & R switches are labeled RESTART (up), FWD DOOR OPEN (center) and OFF (down). In the RESTART position the spike and bypass control settings are overridden, the forward bypass is opened and the spike is moved forward. In the center FWD DOOR OPEN position the forward door is moved to/or held open but the spike position responds to its control knob. In the OFF position both the spike and forward bypass are controlled by their respective controls. Power for the restart circuit is supplied by the essential dc bus.

Emergency Spike Switch

A single 3-position guarded switch, labeled EMER SPIKE, is provided below the instrument panel. The switch is guarded in the center OFF position. After the guard is opened the switch may be positioned in either L or R positions as necessary. In the event of L or R hydraulic failure, the one shot emergency pneumatic bottle in the respective nacelle is activated to drive and lock the spike in the full forward position. Power for the emergency spike circuit is from the essential dc bus.

Inlet Aft Bypass Switches and Indicator Lights

The inlet aft bypass switches and indicator lights are located above the throttle quadrant. They are four-position rotary type switches equipped with concentric lever handles. The switch positions from top to bottom are labeled CLOSED, A (15% open), B (50% open), OPEN (100%). Left and right amber lights, located near the switch levers illuminate to indicate when an aft bypass position and the switch setting do not correspond. A light should illuminate each time its switch is moved, then extinguish as the bypass reaches the required position. Approximately 5 seconds is required for the aft bypass to move from full closed to full open. The aft bypass actuator control circuits are powered by the essential dc bus.

Spike Position Indicator

A dual spike position indicator is located on the lower right side of the instrument panel. The L & R labeled pointers indicate the position of the respective spike in inches aft of the forward position. It is calibrated in inches from 0 to 26 with 5, 10, 15, 20, and 25 inch labeling. Power is furnished from the No. 2 inverter for the left spike and the No. 3 inverter for the right spike.

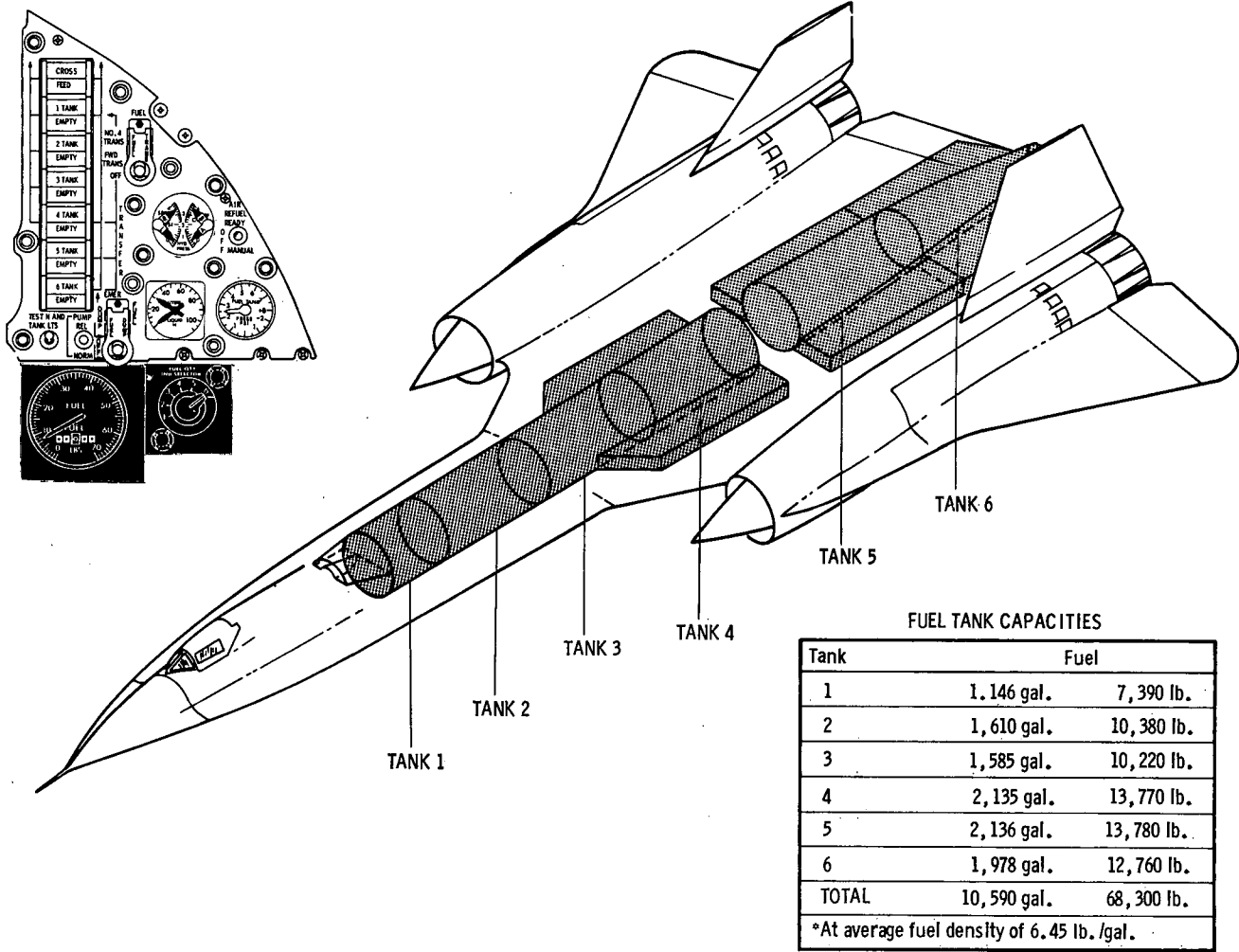
Forward Bypass Position Indicator

A dual forward bypass position indicator is located on the lower right side of the instrument panel. The L & R labeled pointers indicate the opening of the respective forward bypass in 10% increments. Labeled positions are 20, 40, 60, 80 and 100 OPEN. Power is furnished from the No. 2 inverter for the left bypass and the No. 3 inverter for the right bypass.

SECTION I

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FUEL QUANTITY DATA



F200-61(c)

Figure 1-12

Manual Inlet Indicator Light

The annunciator panel MANUAL INLET light, when illuminated, indicates that one or more of the four rotary spike and forward bypass controls is not in the AUTO position or that an inlet restart switch is not in the OFF position. Power for the light is furnished by the essential dc bus.

FUEL SUPPLY SYSTEM

There are six individual fuel tanks, identified from forward to aft as tanks 1, 2, 3, 4, 5, and 6. 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, a single-point refueling receptacle, and a fuel quantity indicating system. In addition to furnishing fuel to the engines, automatic fuel manage-

A-12

ment provides center of gravity and trim drag control. The fuel is also used to cool cockpit air, engine oil, accessory drive system oil, and hydraulic fluid by means of the fuel heat sink system.

FUEL TANKS

The integral, internally sealed, fuel tanks are contained in the fuselage and wing root. The tanks are interconnected by right and left fuel manifolds and a single vent line. Submerged boost pumps supply fuel through the manifolds and transfer fuel for c. g. control. Forward transfer is accomplished by manual control of the right manifold. Aft transfer is accomplished automatically through the left manifold. A fuel dump valve is installed in each fuel manifold. Normal sequence of tank usage is controlled by float switches to automatically maintain an optimum c. g. for cruise. The left engine is normally sequenced from tanks 1, 2, 3, and 4, the right engine is sequenced from tanks 1, 6, 5, and 4. Normal automatic tank sequencing is as follows:

L Engine	R Engine
Tank 1 and 2	Tanks 1 and 6
Tank 2	Tank 6
Tank 3	Tank 6
Tank 3	Tank 5
Tank 4	Tank 5
Tank 4	Tank 4

The fuel manifolds can be connected by depressing the crossfeed switch. This operates a motor operated valve between the fuel manifolds and is mainly used during single engine operation.

REFUELING AND DEFUELING

A single point refueling receptacle installed on top of the fuselage aft of the air conditioning bay is used for both ground and in-

flight refueling. Ground refueling is accomplished by use of an in-flight refueling probe specially modified to utilize a hand operated locking device so that refueling may be done without hydraulic power. Fuel from the receptacle flows through the fueling manifold to each tank. The use of a different size orifice for each tank allows all tanks to be filled simultaneously in approximately 15 minutes with a nozzle pressure of 50 psi. Dual shutoff valves in each tank terminate refueling flow when the tank is full. A defueling fitting is installed on the right fuel feed manifold in the lower right side of tank 3. Tanks 2 and 3, which feed the left manifold, are defueled by opening the crossfeed valve.

CAUTION

Any fuel in tanks 5 and 6 must be balanced with a like amount of fuel in the other tanks during ground fueling or defueling to prevent the aircraft from rocking down on the tail.

FUEL TANK CAPACITIES

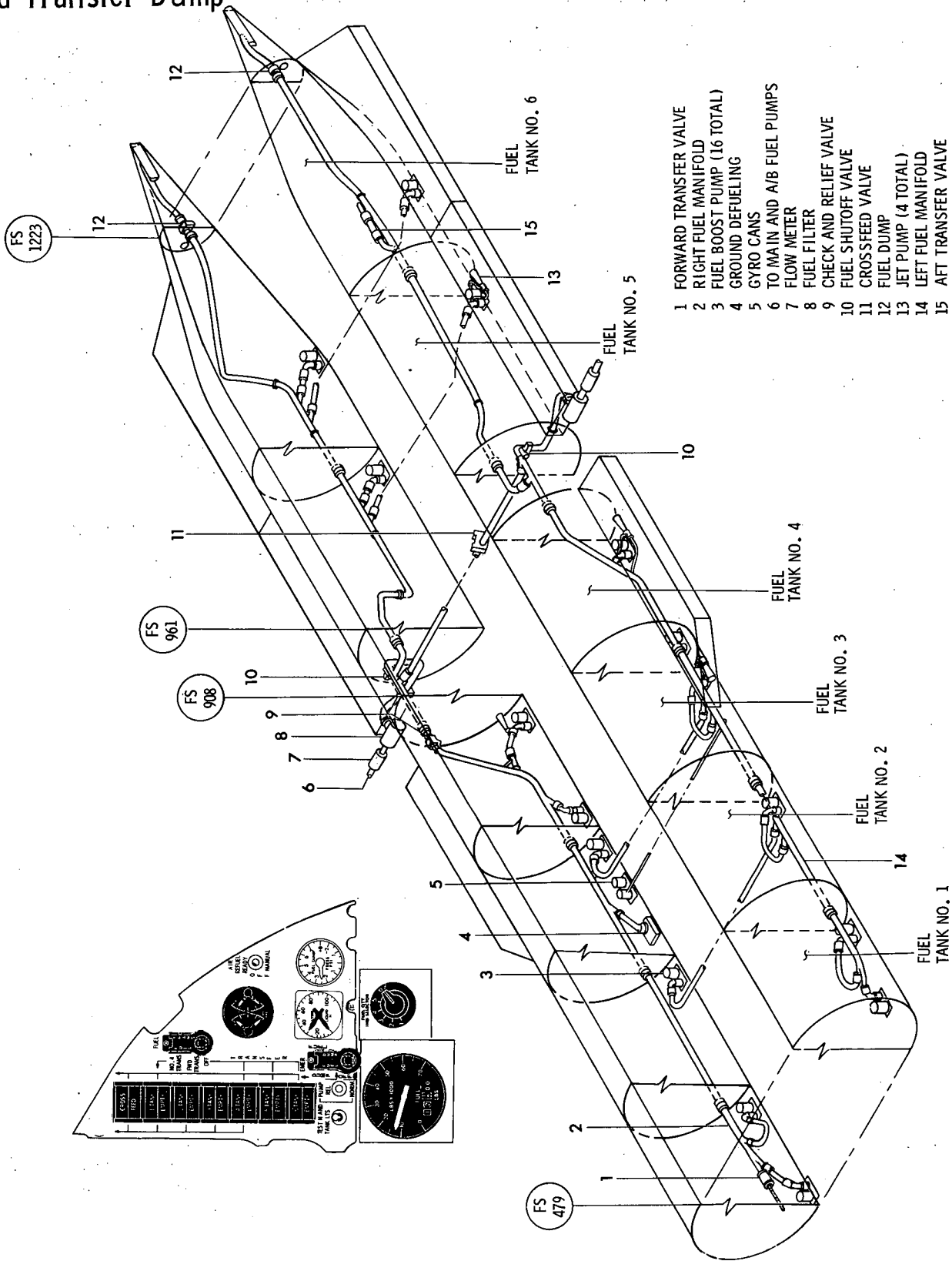
See figure 1-12.

FUEL BOOST PUMPS

Sixteen single stage centrifugal ac powered boost pumps are used to supply the fuel manifolds. Tanks 1 and 4, which normally feed both engines, are equipped with four pumps and tanks 2, 3, 5 and 6 have two pumps each. Either pump of a pair is capable of supplying fuel to its manifold at a rate sufficient for normal engine operation in the event of a failure of the other pump. The pumps in each tank may be operated out of the normal sequence by use of the individual tank boost pump control switches located on the right side of the instrument panel. These switches supplement auto-

SECTION I

FUEL SYSTEM Feed-Transfer-Dump



- 1 FORWARD TRANSFER VALVE
- 2 RIGHT FUEL MANIFOLD
- 3 FUEL BOOST PUMP (16 TOTAL)
- 4 GROUNDFUELING
- 5 GYRO CANS
- 6 TO MAIN AND A/B FUEL PUMPS
- 7 FLOW METER
- 8 FUEL FILTER
- 9 CHECK AND RELIEF VALVE
- 10 FUEL SHUTOFF VALVE
- 11 CROSSFEED VALVE
- 12 FUEL DUMP
- 13 JET PUMP (4 TOTAL)
- 14 LEFT FUEL MANIFOLD
- 15 AFT TRANSFER VALVE

Figure 1-13

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matic tank sequencing if a tank fails to feed in the proper sequence. It is necessary to actuate the pump release switch to terminate the manually actuated pumps when the tank is empty. Normally, each pump (except pumps 1-1 and 1-2 which are protected by a common float switch) is protected by a float switch that deactivates the pump when the tank is empty. One of the float switches in each tank illuminates the yellow tank empty light contained in the respective boost pump tank switch. For example, the float switch for the number 4 pump in tank 4 is used to indicate that tank 4 is empty and pump 4-4 is off. (The tank 4 light indicates green when pumps 4-3 and/or 4-4 are on. When pump 4-4 is on and in automatic sequencing, the green light may not indicate the status of other tank 4 pumps whose operation is affected by automatic features of the ullage and refueling systems.) The boost pumps that feed the left hand manifold are normally powered from the left generator bus and the pumps that feed the right hand manifold are normally powered from the right generator bus. Individual circuit breakers for each pump are located in the compartment behind the cockpit and are not accessible in flight.

Emergency Fuel Shutoff Switches

A guarded fuel shutoff switch for each engine is installed on the lower right side of the instrument panel. Each switch is guarded in the down (fuel on) position. Fuel is shut off in the OFF (up) position. Movement of a switch causes a motor operated valve in the respective engine feed line to operate. Motor power is supplied from the corresponding ac generator bus.

Fuel Boost Pump Switches and Indicator Lights

Six pushbutton type fuel boost pump switches with green and yellow indicator lights are installed in a vertical row on the right side of the instrument panel. These switches are provided for manual control of the fuel boost pumps.

NOTE

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

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. This feature is intended to prevent more than eight boost pumps from operating simultaneously.

NOTE

It is possible to operate more than eight boost pumps at once by a combination of automatic sequencing and manual actuation; however, this condition will not overload the electrical system except when operating on a single generator.

When a set of boost pumps is actuated, either automatically or manually, a green light will illuminate in the pushbutton. When a tank is empty, a yellow EMPTY light in the pushbutton illuminates. When depressed, the boost pump switch will hold down electrically until released by the pump release switch. Power for the boost pump switch circuit and lights is furnished by the essential dc bus.

Pump Release Switch

A momentary pump release switch is installed on the instrument panel below the fuel boost pump switches. The switch has two positions, PUMP REL (up) and NORM (down). When placed in the momentary PUMP REL position, any boost pump switch that has been depressed during

SECTION I

A-12

manual boost pump selection will be released and automatic sequencing of the fuel tanks is continued. Power for the circuit is furnished by the essential dc bus.

CAUTION

A manually selected boost pump should be released when a tank indicates empty so that the pumps in that tank will be shutoff; otherwise, damage to the pump may occur.

Crossfeed Switch

A pushbutton type crossfeed switch is located above the boost pump switches on the instrument panel. When depressed, it illuminates a green light in the switch, opens a motor operated valve between the left and right fuel manifolds, allowing operating boost pumps to pressurize both fuel manifolds. The switch must be depressed a second time to terminate crossfeeding. Power for the circuit is furnished by the essential dc bus.

Fuel Transfer Switch

A guarded three-position fuel transfer switch is located on the right side of the instrument panel. The switch is guarded in the OFF position. When the guard is raised and the switch is moved to the FWD TRANS position, the pumps in tank 1 are inactivated, a valve at the forward end of the right fuel manifold opens into tank 1 if fuel manifold pressure is above approximately 8 psi and fuel will transfer forward through the right side fuel manifold as long as automatic or manual pump sequencing continues. Transfer will be automatically terminated by closing of the forward transfer valve when the tank 1 fuel level reaches 4000 pounds. Tank 1 boost pumps will remain inactivated until either tank 4 has approximately 800 lbs remaining or the transfer switch is moved to the OFF (down) position. Tank 1 pumps will also start when the tank 1 pump switch is pressed. The forward transfer valve is not closed by manual selection of tank 1 but right side boost pump pressure

makes forward transfer ineffective. The lift-lock forward transfer switch may also be pulled out and placed in the NO. 4 TRANS position. In this position, tank 1 pumps are inactivated, the right side pumps in tank 4 are turned on, and tank 5 is turned off if operative. The transfer is only from tank 4, which prevents the accumulation of hot fuel in tank 4 and puts the warmer fuel into tank 1 where it will be used immediately after an air refueling.

NOTE

Forward transfer should be discontinued before refueling is started to restore normal tank sequencing.

Transfer is automatically terminated when the tank 1 4000 pound float switch operates, and the tank 1 pumps remain off until either tank 4 has 800 pounds remaining or the transfer switch is moved to the OFF position. Power for the transfer control circuits is furnished by the essential dc bus.

Those aircraft incorporating S/B 1141 are modified to replace the Tank 4 Forward Transfer position with an EMER forward transfer position on these aircraft. When the lift-loc switch is pulled out and replaced in the EMER position, tank 1 pumps are inactivated and the dual 4000 lb stop transfer float switches in tank 1 are replaced by dual 7400# float switches. This allows forward transfer to continue until tank 1 is full.

WARNING

The EMER position is to be used only in case of an aft c. g. emergency.

Fuel Dump Switch

A guarded 3-position lift-lock fuel dump switch is located on the right side of the instrument panel. The switch is guarded in the OFF (down) position. In the DUMP (center) position dual type solenoid dump valves in each manifold are opened and the pumps in tank 1 are inactivated unless selected manually. If fuel pressure is above 10 psi, all other tanks dump in normal usage sequence until tank 4 is down to a 8000 pound remaining level. Dumping nor-

SECTION I

Add the following text to Fuel Transfer Switch

Those aircraft incorporating S/B 1141 are modified to replace the Tank 4 Forward Transfer position with an EMER forward transfer position. When the lift-loc switch is pulled out and placed in the EMER position, tank 1 pumps are inactivated and the dual 4000 lb stop transfer float switches in tank 1 are replaced by dual 7400 # float switches. This allows forward transfer to continue until tank 1 is full.

WARNING

The EMER position is to be used only in case of an aft c. g. emergency.

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mally stops at this point and, if fuel is in tank 1, the tank 1 pumps will start unless the forward transfer switch is in either the FWD TRANS or NO. 4 TRANS position. The switch knob must be pulled out to put the switch through the DUMP position either to the EMER or OFF position. In the EMER position, the 8000 pound stop dump switch in tank 4 is bypassed and fuel dumping will continue from all tanks except tank 1. If tank 4 is to be completely dumped, tank 1 should be pressed on before tank 4 empties in order to avoid fuel pressure fluctuation as tank 4 empties. Power for the circuit is furnished by the essential dc bus.

WARNING

Emergency fuel dumping must be terminated by placing the dump switch to DUMP or OFF. All fuel can be dumped with EMER dump on and tank 1 selected manually.

Fuel Quantity Selector Switch and Quantity Indicator

A fuel quantity indicator and a rotary seven-position fuel quantity selector switch is installed on the lower right side of the instrument panel. Positions on the selector switch are marked for TOTAL and each of the six tanks positions. The switch is rotated to the individual tank or TOTAL position to select the desired reading on the fuel quantity indicator. The dial is calibrated in 5000 pound increments from zero to 70,000 pounds. The indicator has a digital read-out window indicating to the nearest 100 pounds. Power for the circuit is furnished by the No. 1 inverter.

Fuel Quantity Low Light

A FUEL QTY LOW light on the annunciator panel will illuminate when total fuel remaining in tank 4 is 5000 pounds or less. Power for the light is furnished by the essential dc bus.

Fuel Pressure Low Warning Lights

Fuel pressure warning lights, labeled L and R FUEL PRESS LOW are located on the annunciator panel. Illumination indicates that engine fuel inlet pressure has fallen below approximately 7 ± 0.5 psi. The light is extinguished by restoring fuel pressure above approximately 10 psi. Power is furnished by the essential dc bus.

NOTE

It is possible for a fuel pressure low warning light to illuminate when only two fuel pumps are feeding an engine during high fuel flows, especially with forward transfer and/or fuel dump selected.

Test N and Tank Lights Switch

A test N and tank lights switch is installed below the boost pump switches on the instrument panel. The switch has two positions, up and down (spring loaded down) and is used to test the operation of the liquid nitrogen indicators, nitrogen system annunciator light, derichment light and fuel boost pump lights. When the switch is moved to the up position, the liquid nitrogen indications will move down-scale toward zero and the N QTY LOW annunciator light, fuel boost pump lights and derichment light will illuminate. Power for the circuit is furnished by the essential dc bus.

**FUEL SYSTEM
Pressurization**

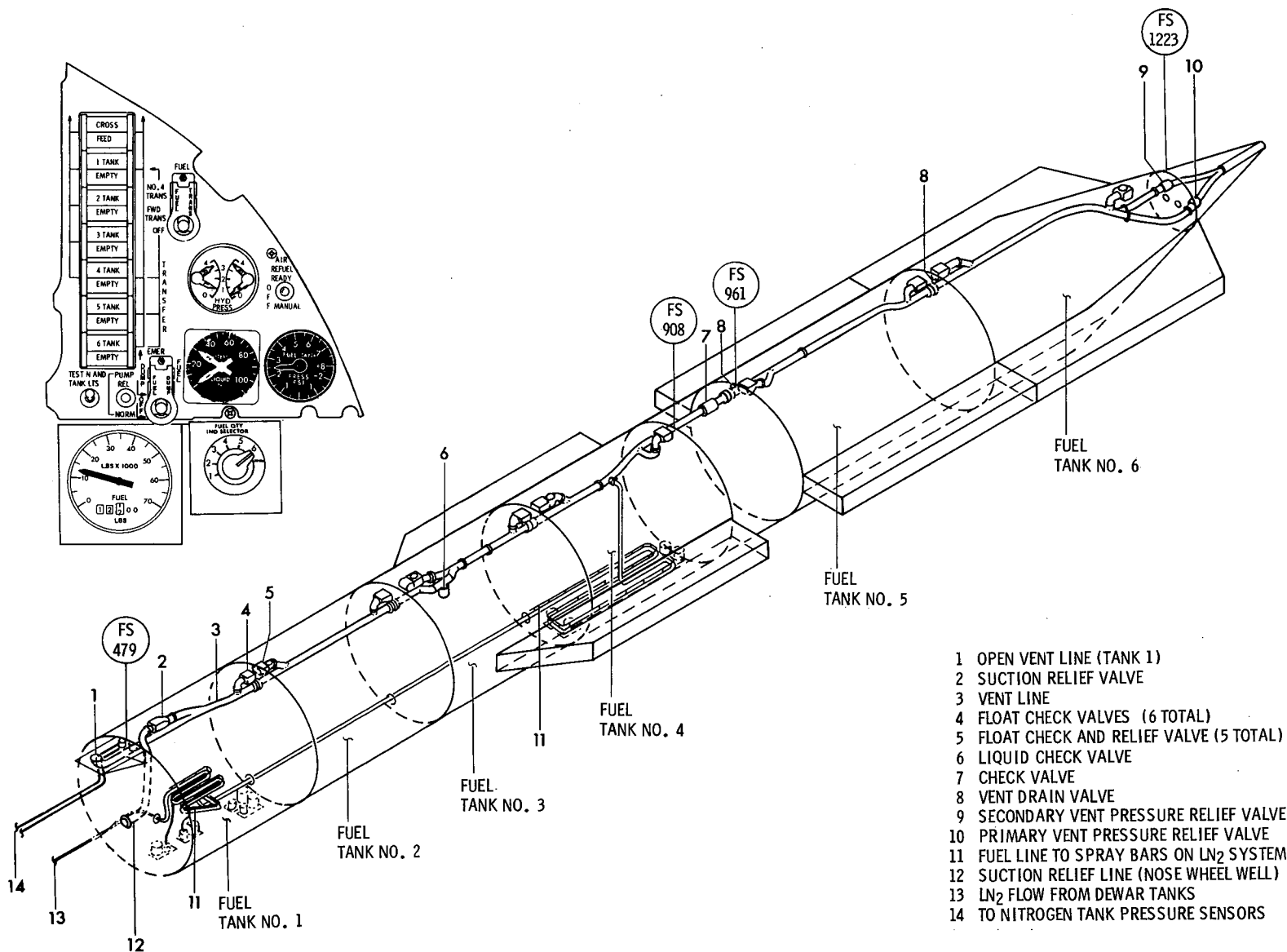


Figure 1-14

- 1 OPEN VENT LINE (TANK 1)
- 2 SUCTION RELIEF VALVE
- 3 VENT LINE
- 4 FLOAT CHECK VALVES (6 TOTAL)
- 5 FLOAT CHECK AND RELIEF VALVE (5 TOTAL)
- 6 LIQUID CHECK VALVE
- 7 CHECK VALVE
- 8 VENT DRAIN VALVE
- 9 SECONDARY VENT PRESSURE RELIEF VALVE
- 10 PRIMARY VENT PRESSURE RELIEF VALVE
- 11 FUEL LINE TO SPRAY BARS ON LN₂ SYSTEM
- 12 SUCTION RELIEF LINE (NOSE WHEEL WELL)
- 13 LN₂ FLOW FROM DEWAR TANKS
- 14 TO NITROGEN TANK PRESSURE SENSORS

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FUEL PRESSURIZATION AND VENT SYSTEM

The fuel pressurization system consists of two Dewar flasks, located in the nosewheel well, and associated valves and plumbing to the fuel tanks and indicators. These flasks are equipped with automatic ac powered heaters and contain liquid nitrogen. The forward flask contains 75 liters and the aft flask contains 106 liters. They supply nitrogen gas to the fuel tanks at $1.5 \pm .25$ psi above ambient pressure. This inerts the ullage space above the fuel and will produce some fuel flow to the engine-driven pump in case of boost pump failure. The liquid nitrogen from the bottom of the flasks is routed through submerged heat exchangers in tanks 1 and 4 to ensure that the nitrogen has become gaseous. The nitrogen gas is then ported to the common vent line and to the top of all tanks.

The venting system consists of a common vent line through all tanks with two vent valves in each tank except the No. 1 tank which has only one vent valve and the open forward end of the vent line. The forward vent valve in tanks 2 through 6 is equipped with a relief valve to relieve tank pressure at 1.5 psi, and a float valve that closes the vent valve when the tank is full. The float shutoff is provided to keep fuel from entering the vent line. The aft vent valve is similar to the forward except it has no relief valve. The common vent line tees into two lines in tank 6 and both go through the rear bulkhead. In the tail cone area there is a relief valve in each line with the left valve set to relieve pressure at $3.25 \pm .25$ psi above ambient pressure. In the event of failure of this valve, the right valve will relieve pressure at 3.85 to 4.15 psi. A suction relief line and valve connects to the common vent line in tank 1 and terminates in a bell mouth fitting in the aft end of the nosewheel well.

Two valves are provided in the vent system to prevent fuel from surging forward in the vent line during aircraft deceleration. A check valve prevents fuel that is coming forward from tank 6 from going farther than tank 5. A python valve located in tank 3 prevents fuel coming from tank 4 from going any farther than tank 3. This float actuated valve closes the vent when fuel is moving forward in the vent line and diverts it into tank 3. Acceleration presents no problem of fuel shift between tanks.

Liquid Nitrogen Quantity Indicator

A dual liquid nitrogen quantity indicator is installed on the right side of the instrument panel. The indicator displays the quantity of liquid nitrogen remaining in each of the two dewar flasks. The indicator is marked in 5 liter increments from 0 to 110 liters. Power for the indicator is furnished by the No. 1 inverter bus.

N2 Quantity Low Light

An indicator light labeled N QTY LOW is provided on the annunciator panel. The light will illuminate when either hand on the liquid nitrogen quantity gage reaches 1 liter remaining. Power for the light is furnished by the essential dc bus.

Fuel Tank Pressure Indicator

A fuel tank pressure indicator is installed on the right side of the instrument panel. The gage indicates the pressure existing in the No. 1 fuel tank, and is marked from -2 to +8 in increments of 1/2 pound per square inch. Power for the indicator is furnished by the 26-volt instrument transformer.

SECTION I

FUEL HEAT SINK SYSTEM

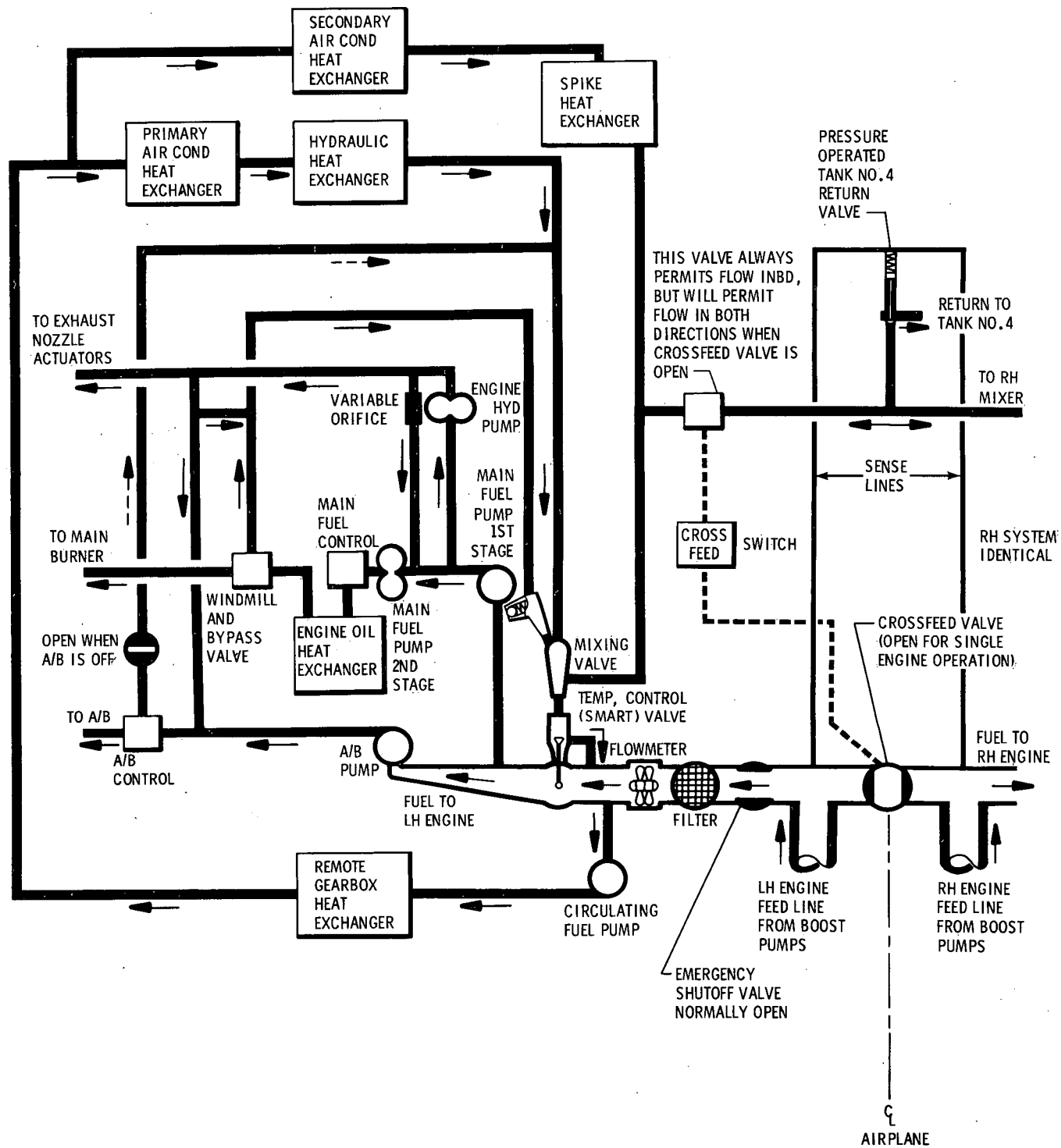


Figure 1-15

Tank Pressure Low Light

A TANK PRESSURE LOW warning light is located on the annunciator panel and will illuminate when the tank pressure reduces to +.25 to +.10 psi. Power for the light is furnished by the essential dc bus.

FUEL HEAT SINK SYSTEM

Fuel from the fuel manifolds is used as a cooling medium for the air conditioning systems, the aircraft hydraulic fluid, and the engine and remote gear box oil. Circulated fuel from the engine fuel hydraulic system is also used to cool 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. Hot fuel returning from the remote gear box 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 (smart valve) and returned to the main engine and afterburner fuel manifold. If the mixed fuel temperature is below 265° F, all of the hot fuel will be burned by the operating engine and afterburner. If the temperature of the mixed cooling loop and incoming engine fuel exceeds 265° F, the smart valve starts to close and a portion 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 smart valve is completely closed at 295° F and all cooling loop fuel is returned to tank 4. If tank 4 is full, the hot 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 cross-feed valve also allows the hot recirculated fuel from the windmilling engine to crossover and mix with the cooling loop and incoming fuel for the operating engine.

AIR REFUELING SYSTEM

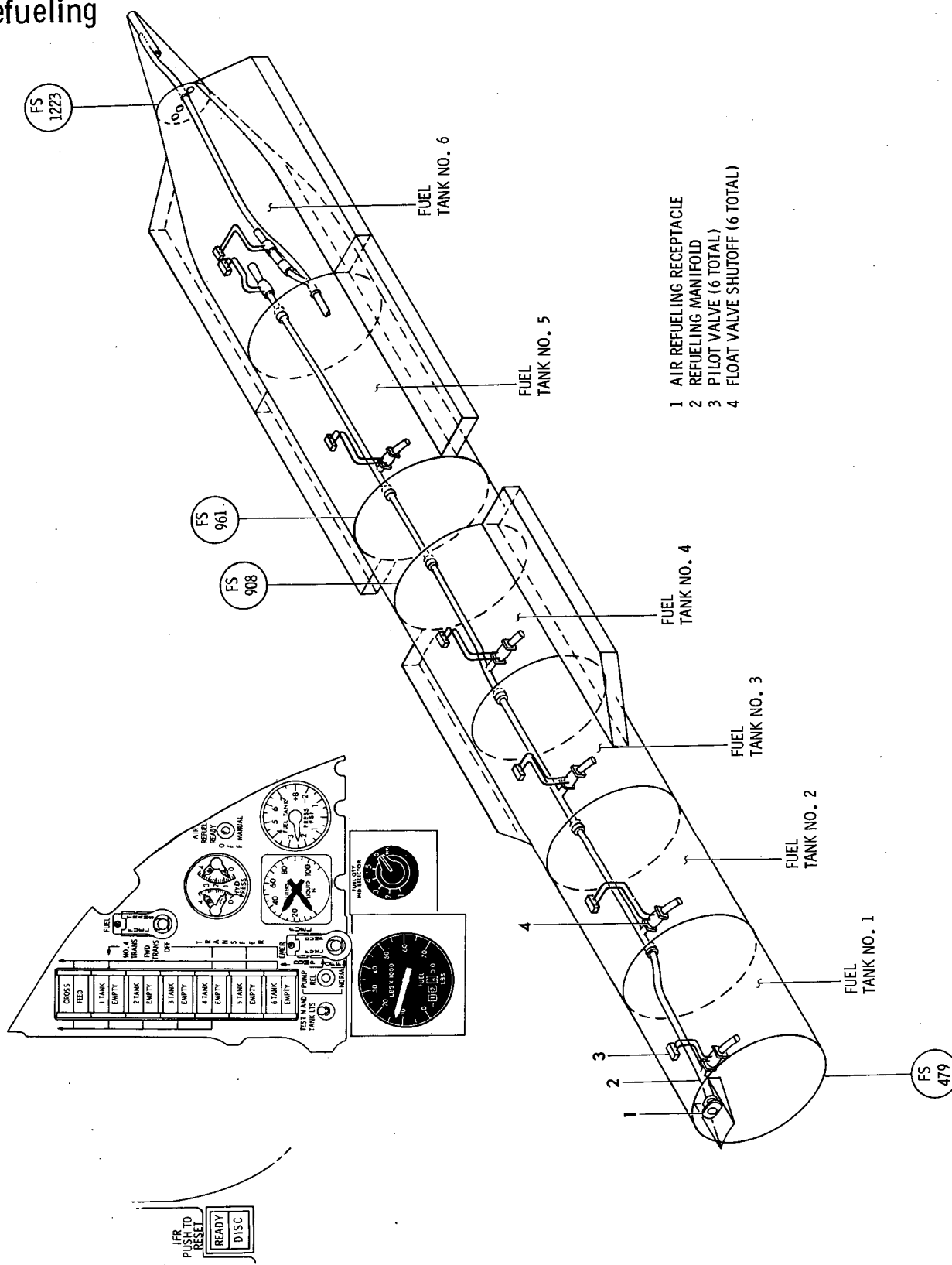
The aircraft is equipped with an air refueling system capable of receiving fuel at a flow rate of approximately 5000 pounds per minute from a KC-135 boom type tanker aircraft. The system consists of a boom receptacle, doors, hydraulic valves, hydraulic actuators, a signal amplifier and control switches and indicator light. Hydraulic power for the system is normally supplied from the L hydraulic system. If the L hydraulic system is inoperative the refuel system can operate from R hydraulic pressure by selecting alternate steering and brakes. Electrical power is supplied by the essential dc bus.

Air Refuel Switch

An air refuel switch is installed on the right side of the instrument panel. The switch has three positions; READY, OFF and MANUAL. When the switch is placed in the READY (up) position hydraulic actuators open the refueling doors, the boom latches are armed, the receptacle lights illuminate and the green READY light illuminates. The receptacle doors are opened by spring action if hydraulic pressure is not available. In the MANUAL (down) position the latching dogs in the receptacle are closed. They may be opened by holding the disconnect (trigger) switch on the control stick until the boom is seated. When the disconnect switch is released the latches

SECTION I

FUEL SYSTEM
Refueling



F200-63(a)

Figure 1-16

will close and hold the boom. The latches will open to release the boom when the disconnect switch is depressed. This position is used in the event of a malfunctioning amplifier. A 3 second time delay is incorporated to prevent nozzle damage if the manual position is selected during refueling contact.

Air Refuel Reset Switch and Indicator Lights

A square dual indicator light and reset button, labeled IFR PUSH TO RESET, is located at the top left side of the instrument panel. The top half is labeled READY and will illuminate green when the air refuel switch is in the READY or MANUAL position, and the refueling receptacle is open and ready to accept the refueling boom. The light will extinguish after the boom is engaged. If the boom disconnects from the fueling receptacle the lower half of the switch will illuminate amber and show DISC. If the air refuel switch is in the READY position the light button is then pressed to reset the system amplifier for another engagement. If the air refuel switch is in the MANUAL position the READY light will be illuminated and manual engagement and disconnect are controlled by the disconnect switch on the control stick. Power for the switch and light is supplied by the essential dc bus.

Disconnect Switch

A momentary contact trigger type switch is installed on the forward side of the control stick. Depressing the trigger switch will normally initiate a disconnect. The disconnect switch is also depressed to open the receptacle latches when the air refuel switch is in the MANUAL position. Releasing the disconnect switch will close the latches.

Disconnect

A disconnect may be accomplished in four ways:

1. Automatically, if boom envelope limits are exceeded (except when using manual boom latching).
2. Automatically, when manifold pressures reach 85-90 psi.
3. Manually, by the boom operator.
4. Manually, by actuating the disconnect switch on the control stick.

Pilot Director Lights (Tanker)

Pilot director lights are located on the bottom of the tanker fuselage between the nose gear and the main gear. They consist of two rows of lights; the left row for elevation and the right row for boom telescoping. The elevation lights consist of five colored panels with strip green, triangular green and triangular red colors and two illuminated letters, D and U, for down and up respectively. Background lights are located behind the panels. The colored panels are illuminated by lights that are controlled by boom elevation during contact. The colored panels that indicate boom telescoping are not illuminated by background lights. An illuminated white panel between each colored panel serves as a reference. The letters A for aft and F for forward are visible at the ends of the boom telescoping panel. The Air Refueling Director Lights Profile (Figure 2-5) shows the panel illumination at various boom nozzle positions within the boom envelope. There are no lights to indicate azimuth; however, a yellow line is visible on the tanker to indicate the centered position. When contact is made, the panels automatically reflect the correction the pilot must make to maintain position.

SECTION I

ELECTRICAL POWER DISTRIBUTION

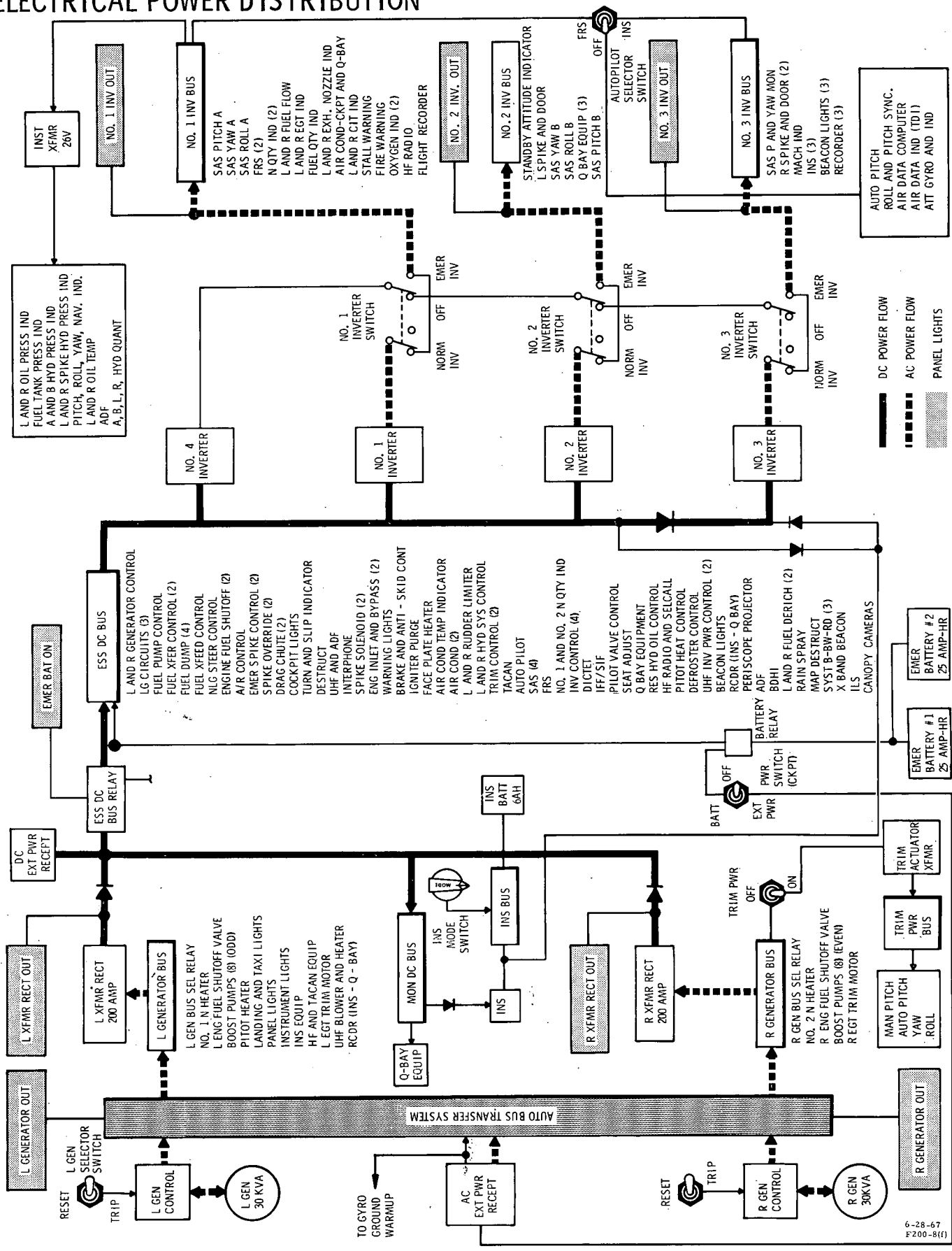


Figure 1-17

ELECTRICAL POWER SUPPLY SYSTEM

Three phase 115/220 volt ac power is provided by two engine driven generators rated at 26 to 32 KVA depending on the installation. Each generator supplies a separate ac bus and a 200 ampere transformer rectifier. Output of the transformer rectifiers is paralleled and furnishes 28-volt ac power to an essential dc bus and a monitored dc bus and to a system of four 600VA inverters. In the event of a single generator failure, a bus transfer and protection system connects the two generator buses. Two 25-amp hour batteries are furnished to supply emergency power to the essential dc bus in the event of complete power failure and a smaller battery provides emergency power to the INS and the No. 3 inverter.

AC ELECTRICAL POWER SUPPLY

Each engine drives an ac generator through its remote gear box to supply 115/200 volt 3-phase power. There are no constant speed drive units, so the ac frequency varies directly with engine rpm; however, the frequency is essentially constant at scheduled engine speed during climb and cruise. When the output of either generator drops below 200 + 5 cps, it is automatically tripped and the other generator automatically provides power through the bus transfer system. Generator cutout occurs at an engine speed of approximately 2800 rpm. Conventional switches are provided for manual control of the generators.

EXTERNAL POWER SUPPLY

The aircraft is equipped with two receptacles for connecting ac and dc external power sources to the aircraft electrical system. These receptacles are located in the nose-wheel well. When external power is con-

nected to the aircraft and the power switch is in the EXT PWR position, the ac generators are automatically disconnected from their respective buses and the buses receive power from the ground power unit. External dc power is paralleled with the dc output of the two aircraft transformer rectifiers. External dc power and inverter cooling air must be connected in order for the external ac power to be available.

DC ELECTRICAL POWER SUPPLY

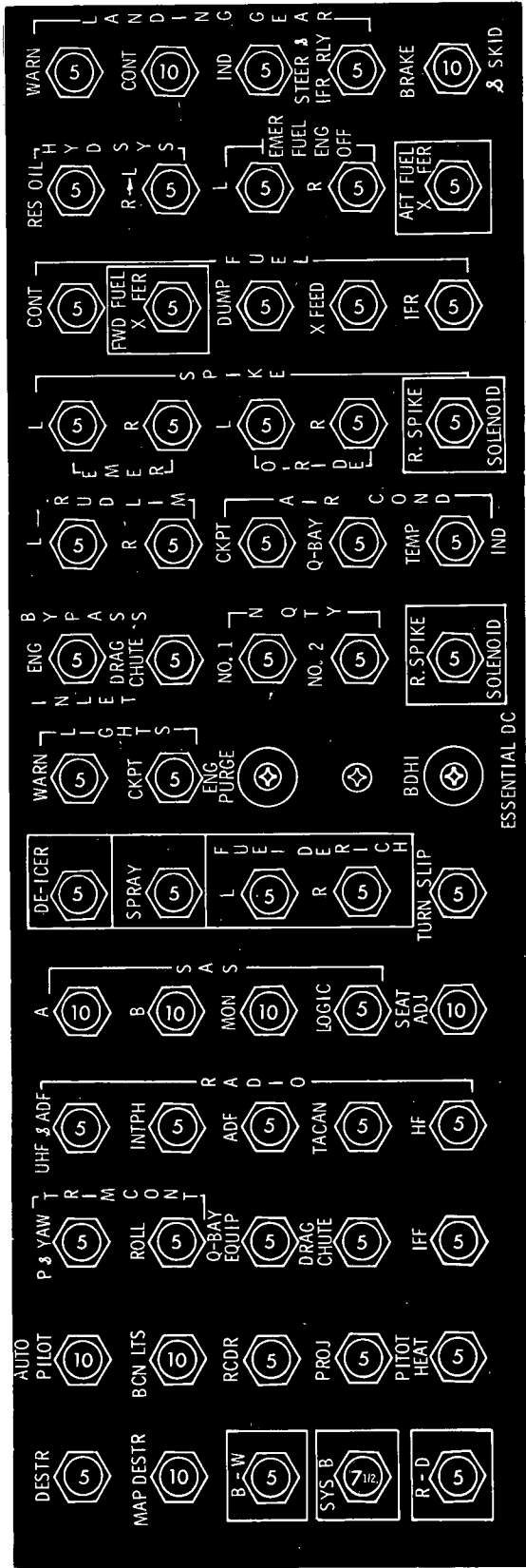
Electrical power for the essential and monitored dc buses is normally supplied by the paralleled output of two 200-amp transformer rectifiers which are powered individually by the ac buses. The two 25 ampere-hour emergency batteries are furnished to supply the essential dc bus with power for a limited time when both transformer rectifiers or both generators are inoperative.

AC INVERTER POWER SYSTEM

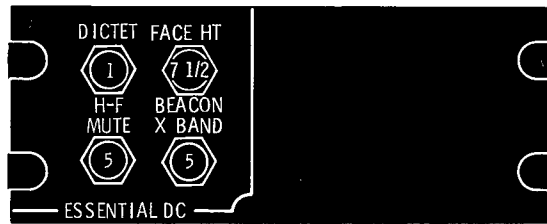
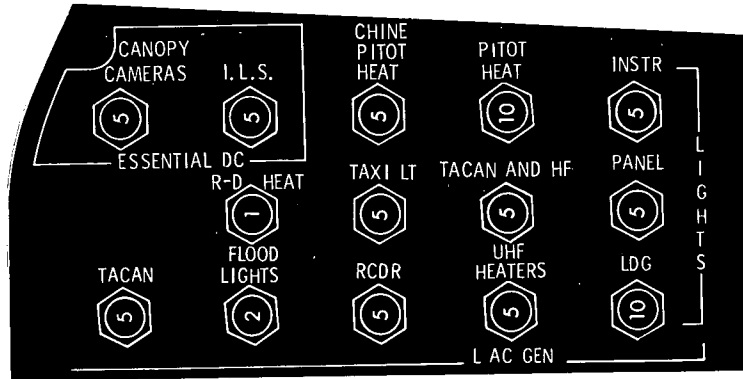
Fixed frequency ac power is supplied by four 600 VA solid state air cooled inverters. These inverters, located in the cheeks of the nosewheel well, are controlled by cockpit switches and powered by the essential dc bus. The No. 3 inverter is also connected to the INS battery whenever the INS mode switch is on. Normally the No. 1, No. 2 and No. 3 inverters furnish power to their respective buses. The No. 4 inverter is normally off. Inverter power distribution is so arranged that the No. 1 inverter bus and its 26-volt instrument transformer powers most of the flight and engine instruments. The No. 3 inverter bus furnishes ac power for the INS. In the event of inverter failure or other electrical system malfunction, any one of the three inverter buses may be operated from the No. 4 in-

SECTION I

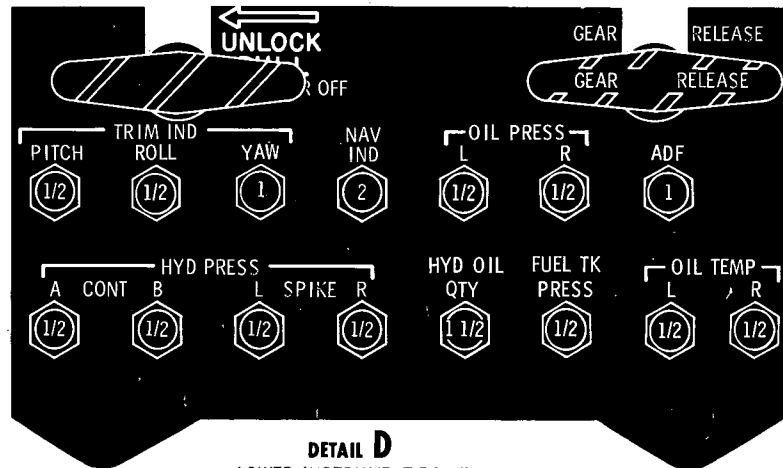
CIRCUIT BREAKER PANELS (Typical)



DETAIL C LEFT CONSOLE



DETAIL B UPPER LEFT

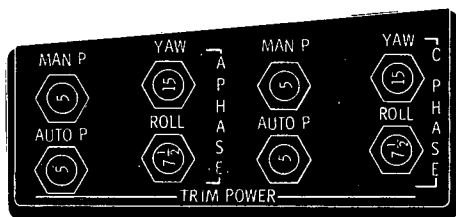
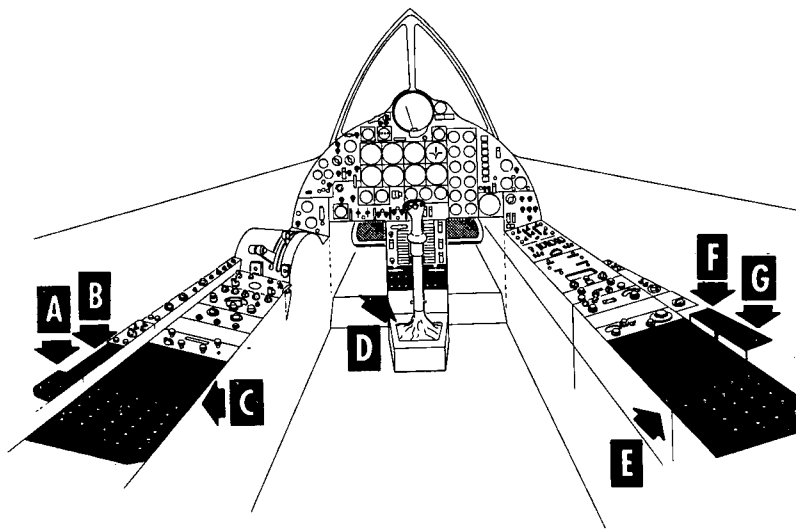


DETAIL D LOWER INSTRUMENT PANEL

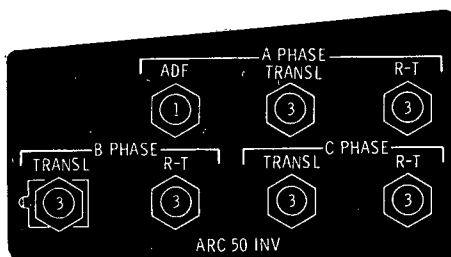
F200-35(1)(g)

Figure 1-18 (Sheet 1 of 2)

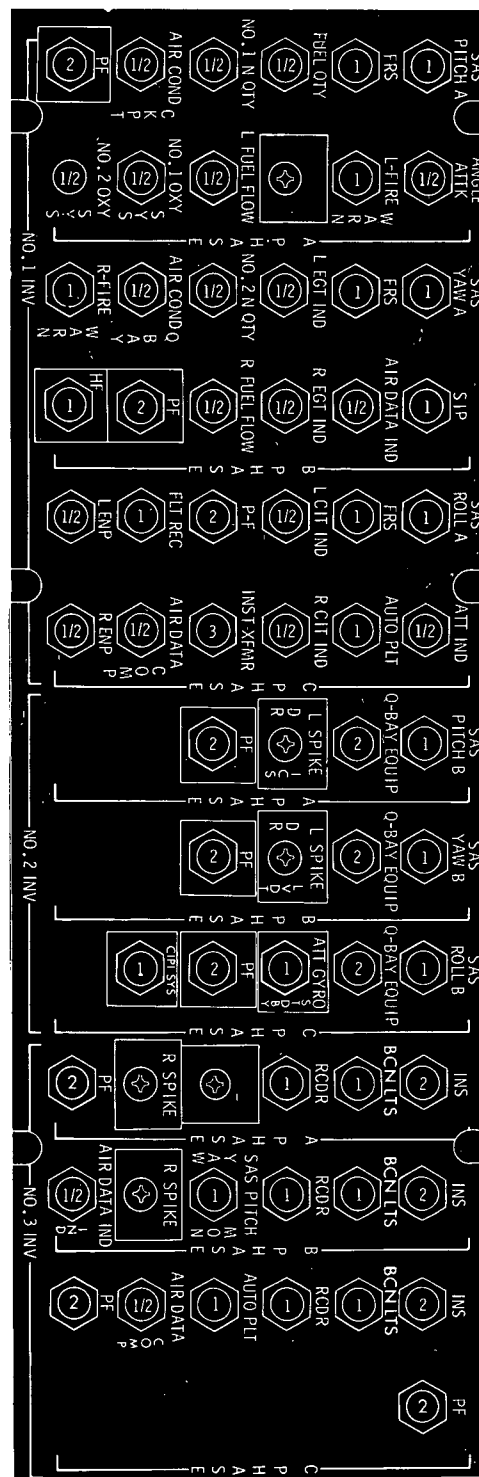
CIRCUIT BREAKER PANELS (Typical)



DETAIL F
UPPER RIGHT



DETAIL G
UPPER RIGHT



DETAIL E
RIGHT CONSOLE

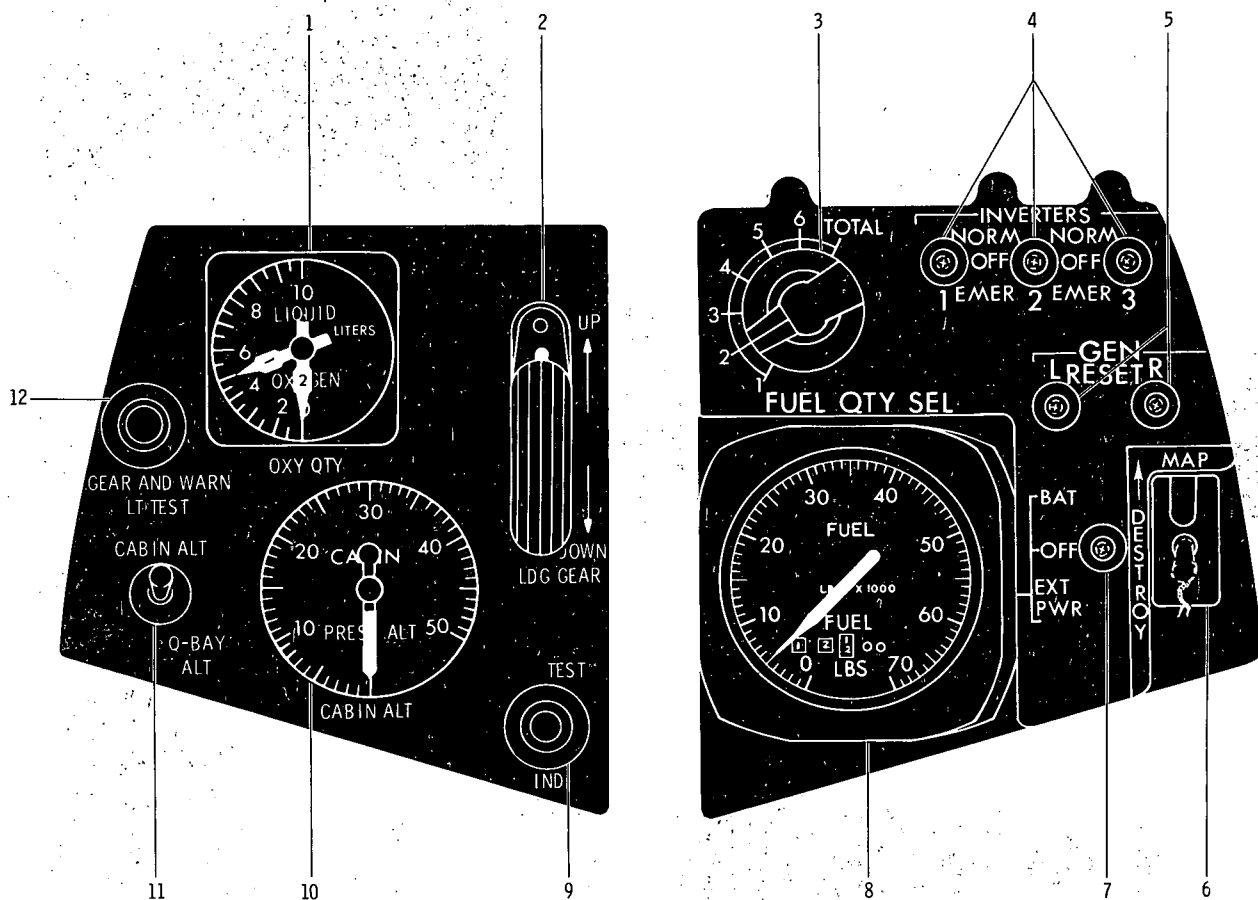
Figure 1-18 (Sheet 2 of 2)

SECTION I

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LEFT AND RIGHT FORWARD PANELS

LEFT AND RIGHT FORWARD PANELS



- 1 OXYGEN QUANTITY GAGE
- 2 LANDING GEAR LEVER
- 3 FUEL QUANTITY INDICATOR SELECTOR SWITCH
- 4 INVERTER SWITCHES
- 5 GENERATOR SWITCHES
- 6 MAP DESTROY SWITCH
- 7 BATTERY SWITCH
- 8 FUEL QUANTITY INDICATOR
- 9 CIP AND OXYGEN TEST SWITCH
- 10 CABIN ALTIMETER
- 11 CABIN ALTIMETER SELECTOR LEVER
- 12 GEAR AND WARNING LIGHTS TEST BUTTON

F200-54(f)

Figure 1-19

verter power supply. Certain related equipment is transferred from the No. 1 and No. 3 inverter by operation of the autopilot selector switch to maintain the proper power phase relationships. The AN/ARC 50 UHF radio has its own rotary inverter supply. Refer to Electrical Power Distribution diagram this section.

ELECTRICAL SYSTEM CONTROLS AND INDICATORS

Circuit Breakers

The cockpit circuit breaker panels are located on the right and left consoles and below the annunciator panel. The circuit breakers are push to reset, pullout type breakers for certain ac and dc circuits as listed on the electrical power distribution chart, figure 1-17. Circuit breaker panels which are not accessible during flight, but which should be inspected before flight, are located in the air conditioning bay (just forward of the refueling receptacle) and electrical load center (left hand side of nose-wheel well).

Generator Switches

A switch for each generator is located on the right side of the instrument panel and is powered by the essential dc bus. Each switch has three positions; GEN RESET (up), TRIP (down) and center (neutral). The switches are spring loaded to the center neutral position. Holding the switch in the GEN RESET (up) position will return the respective generator to normal operation if it has been removed from the bus for any reason other than complete generator failure. In the TRIP (down) position, the generator output will be removed from the generator bus and the auto bus transfer system will supply that bus from the other generator if it is operating.

NOTE

The generators must be reset and connected to the bus after the engines are started and before the ac ground power is removed.

Power Switch

A three-position battery-external power switch is located on the right side of the instrument panel. When in flight or on the ground with ground power disconnected, placing the power switch in the BAT (up) position causes the emergency batteries to supply power to the essential dc bus. In the EXT PWR (down) position, the external power sources furnish power for the electrical systems. In the center OFF position, external ac power is disconnected but power from the dc external receptacle will continue to supply the essential and monitored dc buses and dc power will not be interrupted by moving the power switch from the EXT PWR to OFF positions.

Inverter Switches

Switches for No. 1, No. 2 and No. 3 inverters are located on the right side of the instrument panel below the generator switches. In the NORM (up) position, the respective inverter is energized and supplies power to its individual bus. In the OFF (center) position the inverter is disconnected from the essential dc bus. In the EMERG (down) position the No. 4 inverter is activated and connected to that inverter bus. In the event of multiple inverter failure, the lowest numbered inverter switch that is placed in the EMERG position receives power from the No. 4 inverter. Under this condition, a higher numbered inverter can not receive power even if its inverter switch is in the EMERG

SECTION I

A-12

position. No. 3 inverter also may receive dc power from the small INS battery if the INS mode switch is not in the OFF position.

Generator Out Indicator Lights

The L and R GENERATOR OUT indicator lights, located on the annunciator panel, illuminate when a generator is not furnishing power to its ac bus.

Transformer-Rectifier Out Indicator Lights

The L and R XFMR-RECT OUT indicator lights, located on the annunciator panel, illuminate to indicate that the respective transformer-rectifier is not furnishing power to the dc buses.

Inverter Out Indicator Lights

Three INVERTER OUT indicator lights are located on the annunciator panel. When illuminated, the numbered light indicates that the respective inverter bus voltage is too low. An inverter switch must be placed in the OFF position to disconnect that inverter from the bus. When a disconnected inverter is switched to the EMERG position, the No. 4 inverter is activated and will furnish power to the respective inverter bus and the light will be extinguished unless a lower numbered inverter switch has already been turned to EMERG.

Emergency Battery On Indicator Light

The EMER BAT ON light located on the annunciator panel illuminates when the emergency batteries are furnishing power to the essential dc bus.

HYDRAULIC POWER SUPPLY SYSTEMS

Four separate hydraulic systems are installed on the aircraft, each with its own pressurized reservoir and engine-driven pump. The pumps for the A and L system are driven from the left engine remote gear box and the B and R system pumps are driven from the right engine remote gear box. Hydraulic fluid is cooled by fuel-oil exchangers, using the aircraft fuel supply as the cooling agent. The A and B hydraulic systems provide power for operating the flight controls. The L and R systems provide power for all other hydraulic requirements of the aircraft. Under normal operating conditions, the systems are independent of one another. The L hydraulic system provides hydraulic power to the left engine air inlet control, the landing gear (including uplocks and door cylinders), normal brakes, in-flight refueling door, UHF retractable antenna, and normal nose-wheel steering. The R hydraulic system provides hydraulic power to the right air inlet control and also to the alternate brakes, nosewheel steering, refueling door and landing gear (emergency retraction only) when the L hydraulic system has failed. When the R hydraulic system supplies power to the brakes, the anti-skid feature is inoperative.

Hydraulic System Pressures Gages

Two dual indicating hydraulic gages are installed on the lower center portion of the instrument panel. The right hand gage indicates hydraulic pressure of the A and B (flight controls) systems, and the left hand gage indicates hydraulic pressure of the L and R systems. The gages are calibrated in 100 psi increments from 0 to 4000 psi. Pressure indication on the gages is accomplished by means of remote transmitters in the individual systems. Twenty-six volt ac power is furnished by the instrument transformer and the No. 1 inverter.

A AND B HYDRAULIC POWER SUPPLY SYSTEM

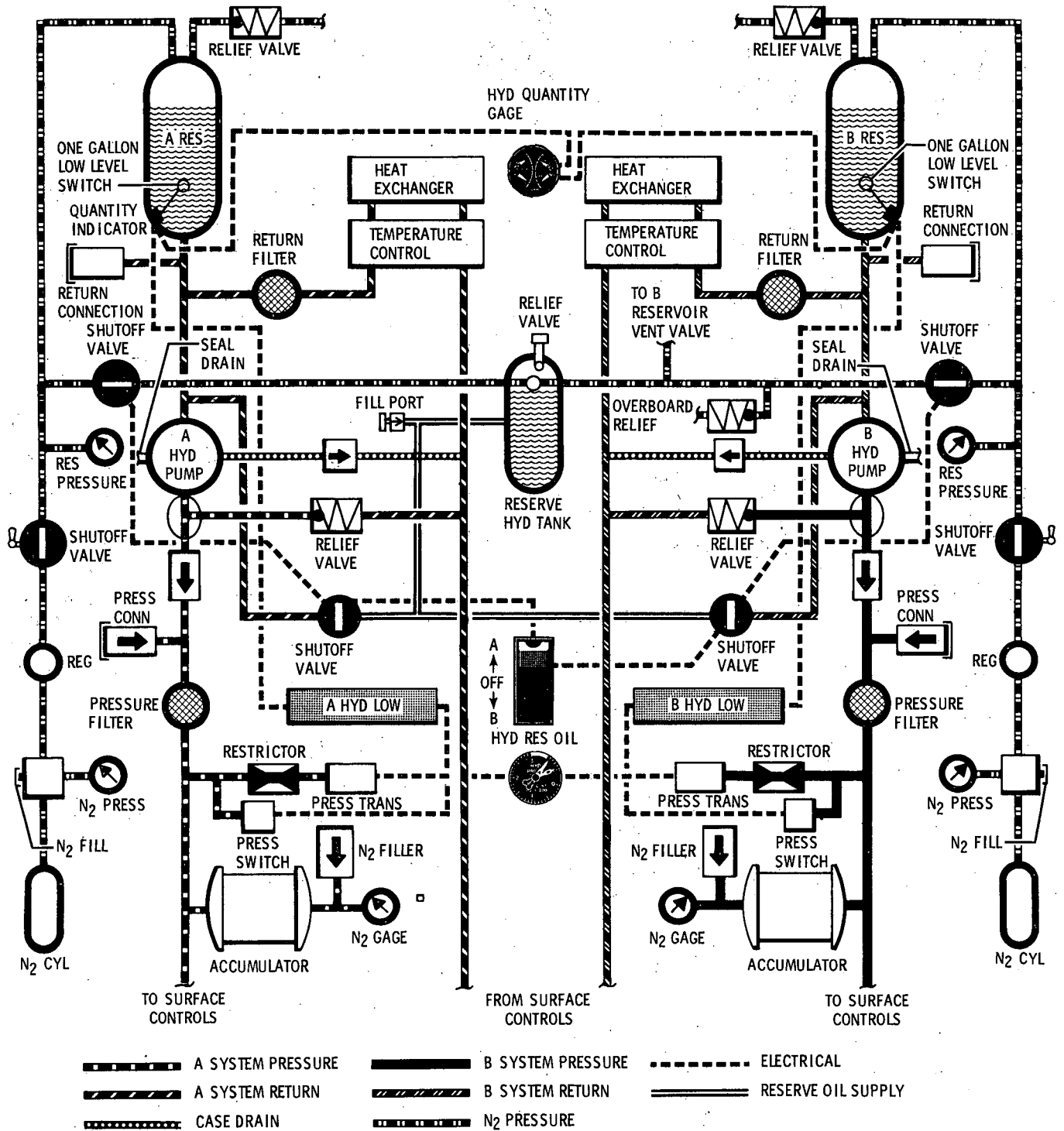


Figure 1-20

F200-20(e)

L AND R HYDRAULIC POWER SUPPLY SYSTEM

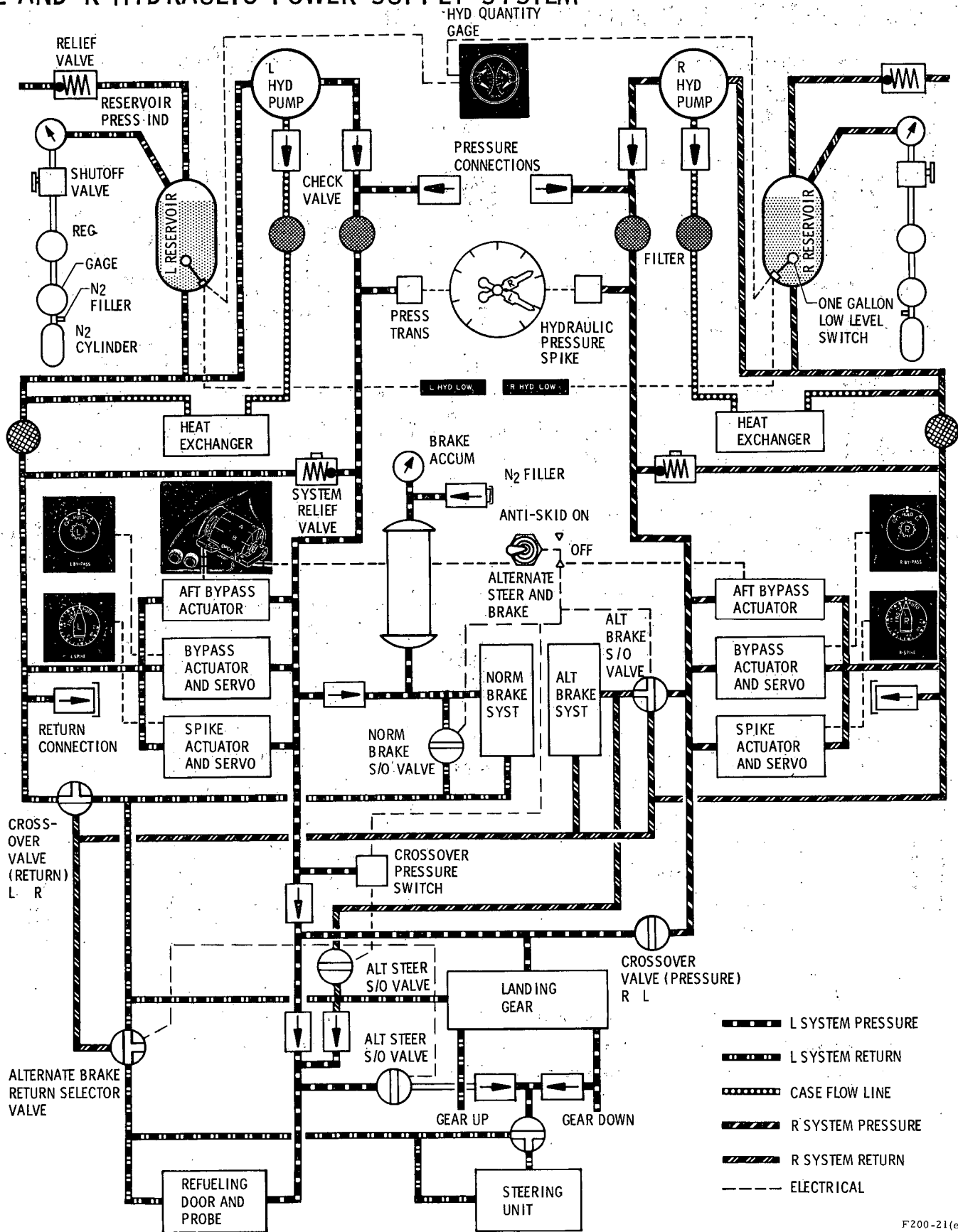


Figure 1-21

F200-21(e)

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Hydraulic Warning Lights

Six hydraulic warning lights are located on the annunciator panel. The A and B HYD PRESS LOW lights will illuminate when the pressure in the respective system drops below 2200 +0 -150 psi. The A and B HYD LOW light will illuminate when the quantity is less than $1\text{-}1/4 + 1/8$ gallons. The L and R HYD LOW light will illuminate when the respective reservoir quantity is less than $1\text{-}1/4 + 1/8$ gallons. Power for the lights is furnished by the essential dc bus.

Hydraulic System Quantity Gage

A quadruple hydraulic fluid quantity indicator installed on the right side of the instrument panel. The L and R concentric needles are on the left side of the gage and the A & B concentric needles are on the right side of the gage. The dials are marked in gallons. Power is furnished from the 26 V ac instrument transformer.

HYDRAULIC RESERVE OIL SYSTEM

A reserve oil supply for the A and B hydraulic systems is contained in an 8.5 gallon reserve tank mounted in the No. 4 fuel tank. The reserve hydraulic oil is transferred by gravity flow and nitrogen pressure through solenoid operated shutoff valves to either the A or B hydraulic system.

Hydraulic Reserve Oil Switch

The hydraulic reserve oil switch is mounted on the left side of the annunciator panel. It is a three position switch, guarded in the center OFF position. In the A (up) position, solenoid operated shutoff valves are opened to the A hydraulic system suction and tank vent lines. This allows the reserve hydraulic fluid to supply the A system as needed up to approximately 0.3 gallon per

minute. In the B (down) position the solenoid valves to the B system are opened and the reserve fluid will supply the B system. Power for the valves is furnished by the essential dc bus.

WARNING

Reserve hydraulic fluid is to be used only to supply the operative A or B system in the event of malfunction of the other system.

FLIGHT CONTROL SYSTEM

The cockpit flight controls consist of a conventional control stick and rudder pedals. The delta wing configuration utilizes elevons instead of separate aileron and elevator control surfaces. The elevons, moving together in the same direction, function as elevators and when moving in opposite directions, function as ailerons. Each elevon consists of an inboard and outboard panel with the inboard panel located between the fuselage and the nacelle and the outboard panel outboard of the nacelle. Both panels on one side function as a single unit with the servo input to the outboard elevon connected directly to the inboard elevon surface. The dual canted rudders are full moving, one piece, pivoting surfaces with a small fixed stub at the junction of the vertical surface and the nacelle. Deflection and control of the elevons and rudders is by means of dual, full hydraulic, irreversible actuating systems.

Control surface travel limits are as follows:

	Elevons	Rudders
Pitch	10° Down 24° Up	-
Pitch plus Roll	20° Down 35° Up	-
Yaw	-	20° Left 20° Right
Roll	12° Down 12° Up	-

SECTION I

FLIGHT CONTROL SYSTEM (Rudders)

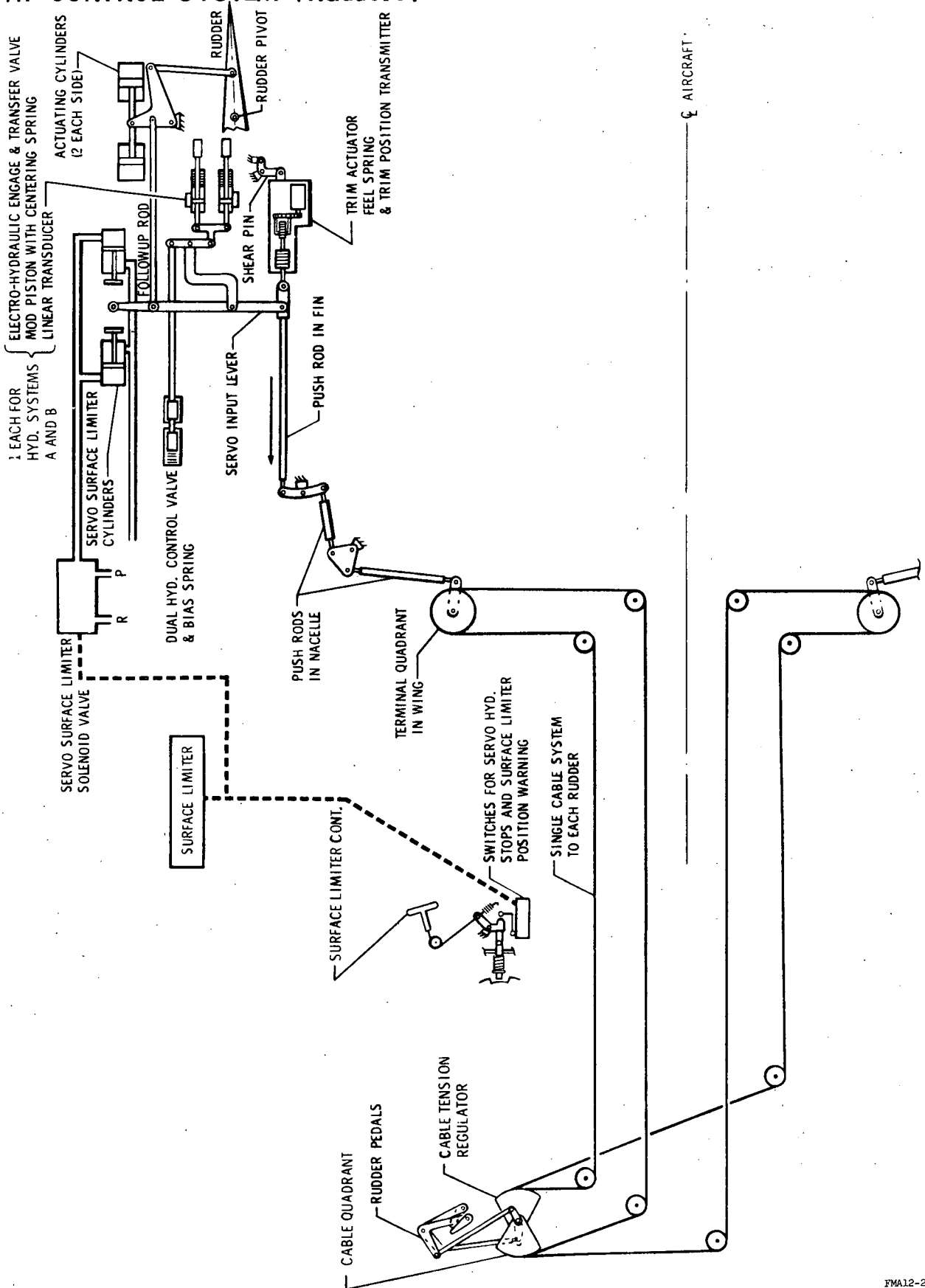


Figure 1-22

FMA12-22

Manually operated mechanical stops are incorporated in the cockpit mechanism to limit the surface movement at high speed. Elevon travel in roll is limited to 7° up, 7° down and rudder travel is limited to 10° right, 10° left. An additional stop is installed in each rudder servo package to limit the rudder travel. These stops are electrically controlled and hydraulically operated by separate electrical and hydraulic systems. If no electrical power is available, the rudders will be limited to approximately 10° L and R travel. If electrical power is available to one stop, that rudder only will have the full 20° L and R travel available. The rudder cable must be stretched to obtain this travel, causing a noticeable increase in rudder pedal force.

CABLE SYSTEM

Cable systems are utilized to transfer control movements from the control stick and rudder pedals to the flight control mechanisms. The pitch and roll axis cable systems are duplicated from the cockpit to the mixing mechanism in the aft fuselage. The rudder system has two separate closed loop single cable systems, one to each rudder. Cable tension regulators and slack absorbers are incorporated in the cable systems.

TRIM CONTROL SYSTEM

Flight control trim is accomplished by deflecting the control surfaces through the use of electrical trim actuators. The roll and pitch trim actuators are located downstream of the feel springs so that stick position remains neutral, irrespective of the amount of trim. The trim actuator and feel spring location is combined in the rudder mechanism and yaw trim is reflected by rudder pedal position.

Travel limits of the trim system are $3-1/2^{\circ}$ down to $6-1/2^{\circ}$ up in pitch, 4.5° up and down (each side) in roll, and 10° left to 10° right in yaw. Trim position indicators are provided for each axis. Trim rates are as follows:

	Pitch	Roll	Yaw
Max.	$1.5^{\circ}/\text{sec}$	$.95^{\circ}/\text{sec}$ Total Diff.	$1.5^{\circ}/\text{sec}$
Min.	$0.67^{\circ}/\text{sec}$	$.47^{\circ}/\text{sec}$ Total Diff.	$0.67^{\circ}/\text{sec}$

Automatic pitch trim uses a separate, slow speed motor for autopilot synchronization. The automatic pitch trim rate is $0.15^{\circ}/\text{sec}$ maximum and $0.067^{\circ}/\text{sec}$ minimum. Trim power is normally furnished by the R generator bus.

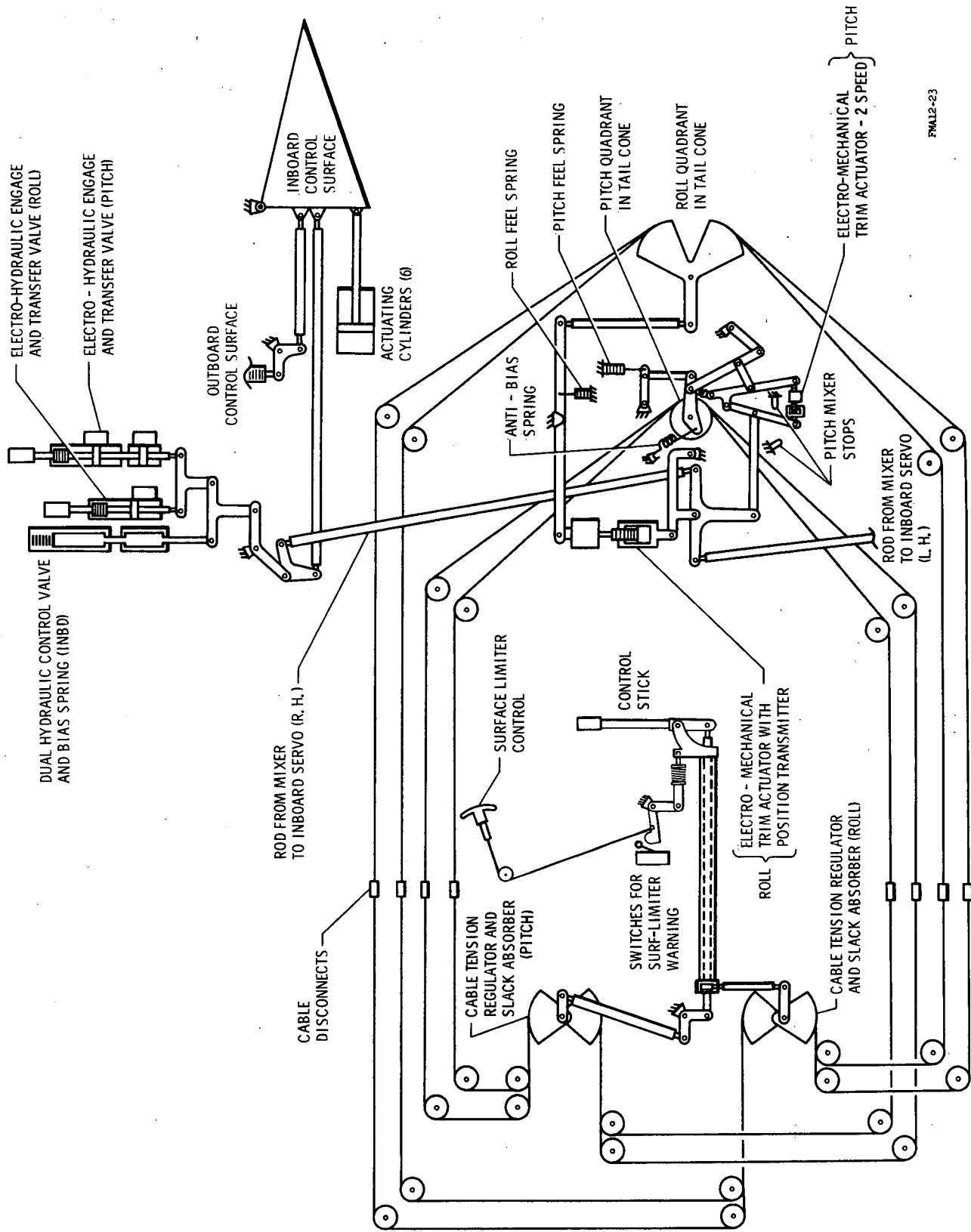
RUDDER PEDALS

Primary control for the rudders consists of conventional rudder pedals mechanically connected by cables, bell cranks and push-rods to hydraulic control valves at the rudder hydraulic actuators. The rudder pedals are released for adjustment by pulling the T-handle labeled PEDAL ADJ located below the annunciator panel. Wheel brakes are controlled conventionally by toe action on the rudder pedals; refer to Wheel Brake System, this section. Rudder pedal movement also controls nosewheel steering; refer to Nosewheel Steering System, this section. The pedals are hinged to fold inboard and upward, providing foot space on the cockpit floor.

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FLIGHT CONTROL SYSTEM (Inboard Elevons)



FWA12-23

Figure 1-23 (Sheet 1 of 2)

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FLIGHT CONTROL SYSTEMS (OUTBOARD ELEVONS)

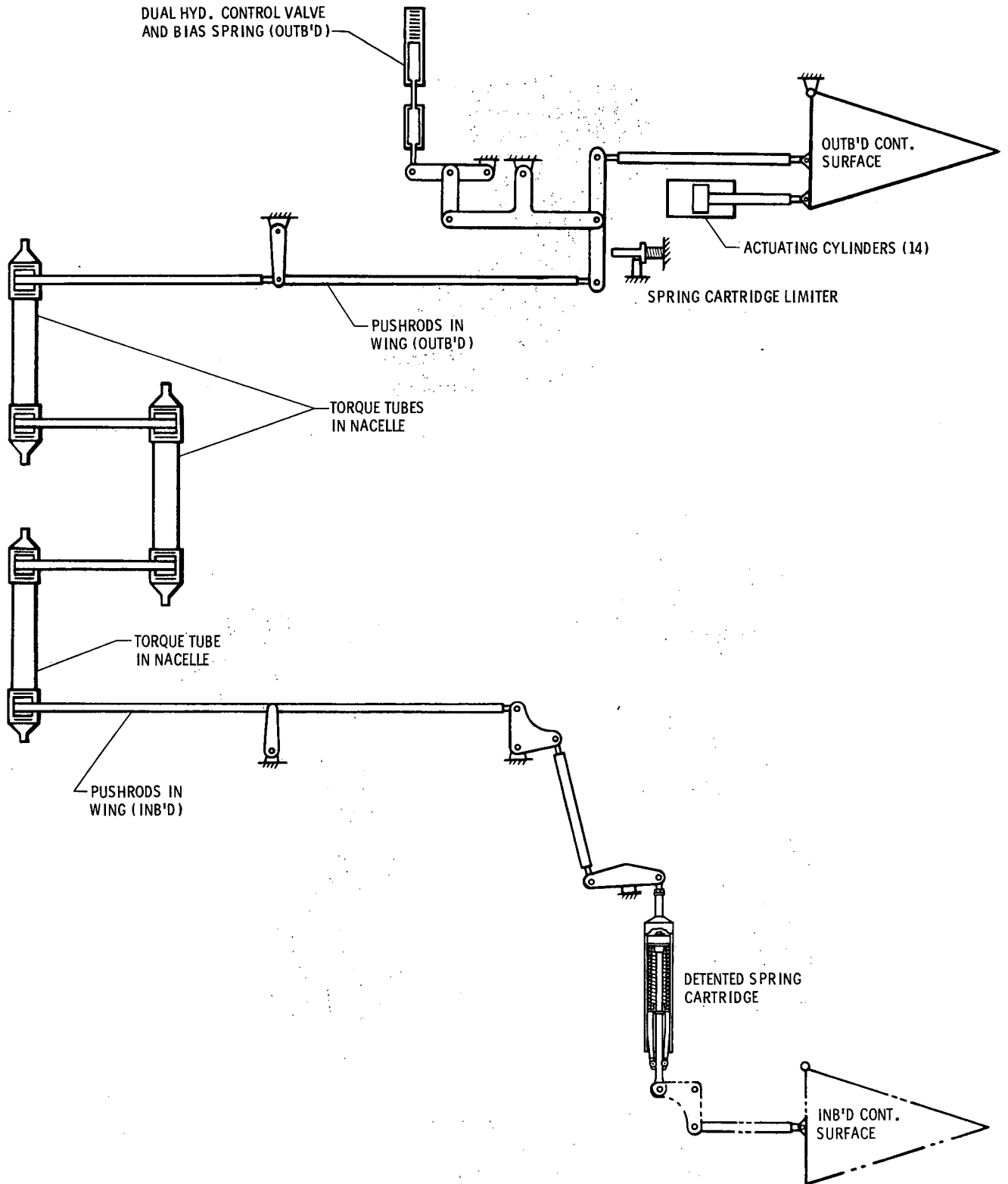
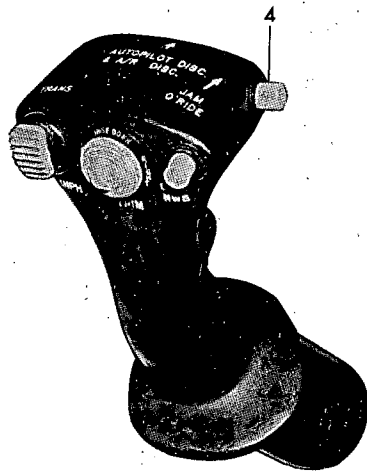


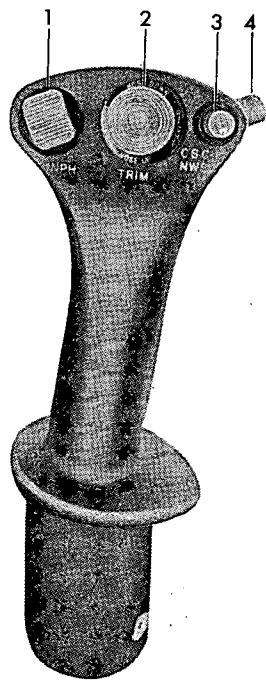
Figure: 1-23 (Sheet 2 of 2)

3-30-66
F200-3(a)

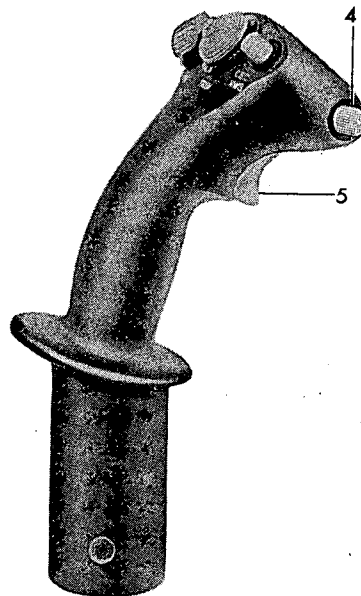
CONTROL STICK GRIP



TOP VIEW



FRONT VIEW



SIDE VIEW

- 1 TRANSMITTER-INTERPHONE CONTROL SWITCH
- 2 PITCH AND YAW TRIM SWITCH
- 3 CONTROL STICK COMMAND-NOSEWHEEL
- 4 JAM O'RIDE SWITCH
- 5 EMERGENCY AUTOP ILOT DISENGAGE SWITCH AND AIR REFUEL DISCONNECT

F200-24(c)

Figure 1-24

ARTIFICIAL FEEL SYSTEM

The use of a full power irreversible control system for actuation of the surfaces prevents air loads and resulting "feel" from reaching the cockpit controls. Therefore, feel springs are installed in each of the pitch, roll and yaw axis mechanisms to provide an artificial sense of control feel. The springs apply loads to the pilot controls in proportion to the degree of control deflection.

CONTROL STICK

The control stick is mechanically connected by a torque tube, push rods and bell cranks to the dual cable system which operates the roll and pitch quadrants in the aft fuselage tail cone. Mechanical push rod linkages mix the control movements and position dual hydraulic control valves. These valves direct both A and B system hydraulic pressure to the inboard elevon actuating cylinders.

Push rods, bell cranks and torque tubes transfer inboard elevon deflection to position the outboard dual hydraulic control valves. These valves direct both A and B system hydraulic pressure to the outboard elevon actuating cylinders. A push rod followup system closes off the flow of hydraulic fluid to the actuators when the desired elevon deflection is obtained. Located on the control stick grip is a combination pitch and yaw trim switch, an autopilot control stick command, a nosewheel steering button, a microphone switch for both interphone and radio transmission, a combination autopilot disconnect and in-flight refueling disconnect switch and a jam override pushbutton.

Control Stick Command Switch (CSC)

Refer to Autopilot System, Section IV.

Pitch and Yaw Trim Switch

Pitch and yaw trim control is provided by a spring-loaded, four position thumb actuated switch installed on the control stick grip with a center OFF position. The switch positions are LEFT, RIGHT, NOSE UP and NOSE DOWN. The switch controls trim motors powered by the right generator bus through the 28-volt ac trim actuator transformer and trim power bus.

NOTE

The trim power switch must be in the ON position before the pitch, roll and yaw trim switches will operate.

Lateral movement of the switch to the left corrects for right yaw and lateral movement to the right corrects for left yaw. Forward movement of the switch produces down elevon operation of the trim motors and actuators (aircraft nose down). Aft movement moves the elevons up (aircraft nose up).

Trim Power Switch

A trim power ON-OFF switch is located on the annunciator panel. It enables the pilot, if necessary, to disconnect power to all trim motors quickly as the main trim power ac circuit breaker is not available to the pilot. To prevent inadvertent movement the switch must first be pulled out before it can be moved from the ON to the OFF position. In the ON position 200 volt 3 phase ac power from the right generator bus is applied to the primary side of the trim actuator transformer. Individual 28

SECTION I

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ac circuit breakers for A and C phases of the Manual Pitch, Auto Pitch, Roll and Yaw trim circuits are located on the right console.

Roll Trim Switch

A three-position roll trim switch is located just forward of the throttle quadrant. The switch positions are indicated by L (left) and R (right) arrows. The switch is spring-loaded to the center off position. When the switch is held in the R position, the roll trim motor actuates to move the right elevons up and the left elevons down. Actuation of the switch to the L position moves the right elevons down and left elevons up. 28-volt ac power is furnished from the trim power bus.

Rudder-Synchronization Switch

A three-position rudder synchronization switch is installed just forward of the throttle quadrant. The switch positions are indicated by L (left), R (right) arrows. It is springloaded to the center off position. In the L and R positions the switch provides electrical power to the right rudder trim motor which moves the right rudder to agree with the position of the left. Rudder synchronization is obtained by superimposing the L and R needles on the yaw trim gage. 28-volt ac power is furnished by the trim power bus.

Roll, Pitch and Yaw Trim Indicators

Separate roll, pitch and yaw trim indicators are located on the left side of the instrument panel. The roll trim indicator uses a double ended needle and displays the amount of roll trim from 0° to 9° differential. The pitch trim indicator displays the amount of pitch trim from 5° nose down to 10° nose up, al-

though only 3-1/2° nose down and 6-1/2° nose up trim is available. The yaw trim indicator displays the amount of yaw trim from 10° left to 10° right for both rudders. Rudder synchronization is obtained by superimposing the L and R needles on the yaw trim gage. 26-volt ac power for the indicators is furnished by the instrument transformer and the No. 1 inverter.

Surface Limiter Control Handle

A T-handle, labeled SURF LIMIT RELEASE, is located on the left side of the annunciator panel. When the handle is turned 90° counterclockwise and released, the mechanical stops in the roll and yaw axis of the cockpit control system are activated. This action also opens an electrical switch which de-energizes a solenoid operated valve in each rudder servo package and activates the servo package rudder stops. When the handle is pulled out and rotated 90° clockwise, the mechanical stops in the cockpit are released and the solenoid is energized, releasing the servo package stops. Power for the rudder limiting circuit is furnished by the essential dc bus.

Surface Limiter Indicator Light

When speed exceeds Mach 0.5, an indicator light on the annunciator panel will illuminate until the surface limiter handle is released. If the speed is below Mach 0.5 and the surface limiters are on, the indicator light will illuminate until the surface limiter handle is pulled out. Power for the lights is furnished by the essential dc bus.

AUTOMATIC FLIGHT CONTROL SYSTEM

The automatic flight control system includes stability augmentation, autopilot, and air data systems, plus additional subsystems furnishing attitude and navigational course inputs for the autopilot. The air data sys-

tem furnishes signals to the autopilot and inertial navigation systems. The stability augmentation system supplies signals to the hydraulic servos that operate the control surfaces. The inertial navigation system supplies attitude and navigational course inputs for the autopilot. Heading and attitude reference signals for the autopilot are also supplied by the Flight Reference System. The autopilot moves the aircraft hydraulic servos through the SAS. For further information on the autopilot and inertial navigation systems, refer to Section IV.

STABILITY AUGMENTATION SYSTEM

The three axis stability augmentation system is a combination of electronic and hydraulic equipment which augments the natural stability of the aircraft. It is designed for optimum performance at the basic mission cruise speed and altitude, but also provides improved stability for in-flight refueling, landing and takeoff. The SAS is part of the aircraft's basic control system and is normally used for all flight conditions.

Dual electronic channels are provided for all axes and a third monitor channel is provided for both the pitch and yaw axis. Logic circuits compare the functioning of each pitch and yaw channel and automatically delete a failed channel. The pilot is also provided with a visual warning of a failed channel.

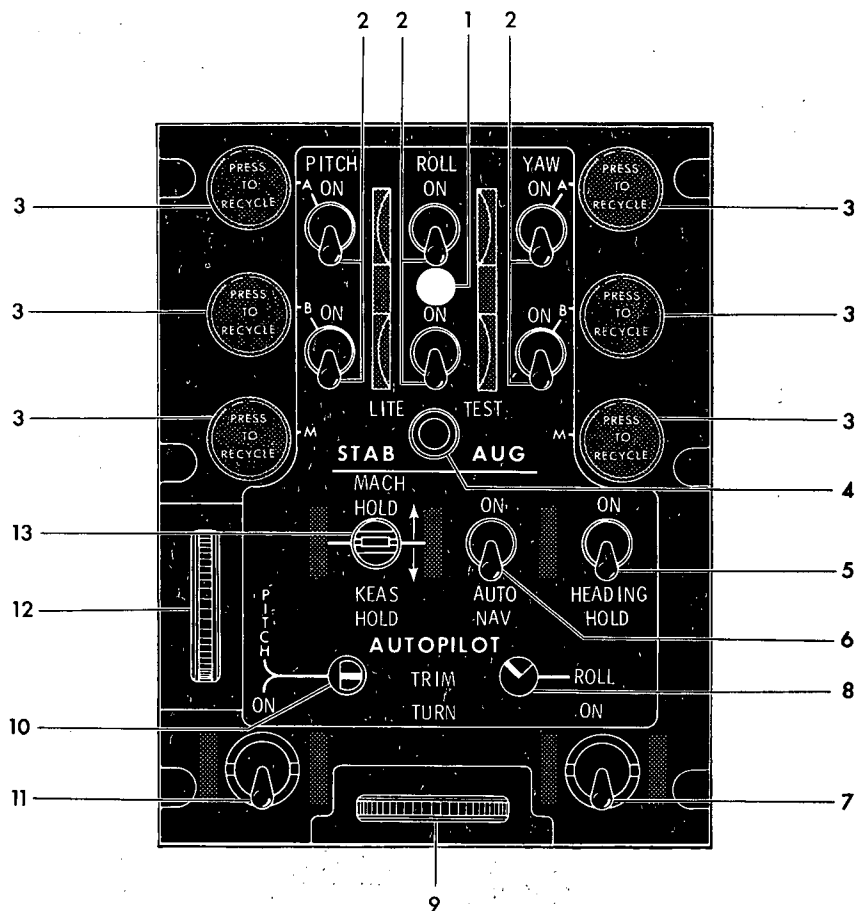
In the roll axis, each channel controls the elevons on only one side of the aircraft. The pilot may select a single channel if desired. Reliability is provided through dual hydraulic and inverter supplies. Each active channel in each axis is powered by separate supplies so that the two halves of each system are operated independently. A separate gyro system is provided for each channel in each axis. The design is such that no single failure except overheating of a complete gyro package can cause loss of all channels in one axis. Even if this oc-

curred, it is unlikely that all of the gyros in the package would fail simultaneously. The SAS system compares the 3 electronic systems and disengages a malfunctioning A, B or M channel. Automatic gain increase is applied to the remaining channels so that control response remains the same. A malfunctioning electronics channel is indicated by illumination of the A or B and/or M light.

STABILITY AUGMENTATION PITCH AXIS

The pitch axis SAS consists of two independent active channels A and B and a third monitor M channel. The two independent active channels A and B provide the desired control through two pairs of tandem servos. There is one pair of servos on each side of the aircraft. The servos are in series with the autopilot and the pilot's control movements. Damping signals to the elevons do not move the control stick. Each A and B channel drives one servo on the left side of the aircraft and one on the right side. A channel uses A hydraulic system and B channel uses the B hydraulic system. This avoids loss of both channels in case of failure of either the A or B hydraulic systems. The sensors for the pitch axis are rate gyros located in tank No. 3. The gyros provide signals in proportion to the rate of pitch attitude change of the aircraft. Above 50,000 feet a "lagged" pitch rate gain is programmed into the pitch SAS electronic circuits. This pitch rate signal changeover may be felt as an abrupt pitch transient during a turn while climbing or descending through the 50,000 foot level. Refer to Section VII, Pitch Axis Characteristics due to Lagged Pitch Rate Switching. Phasing of the gyro signals is such that an angular pitch motion produces elevon movement to oppose and restrict attitude change. The system will take corrective action rapidly in the event of a gust disturbance. Pilot inputs are also opposed; however, the elevon motion produced by the SAS is designed to aid the pilot in avoiding overcontrol and improve the handling qualities of the aircraft.

SAS AND AUTO PILOT CONTROL PANEL



TYPE "A" PANEL

- | | | | |
|---|------------------------------|----|--|
| 1 | ROLL CHANNEL DISENGAGE LIGHT | 8 | A/P ROLL TRIM SYNCHRONIZATION INDICATOR |
| 2 | SAS CHANNEL SWITCHES | 9 | A/P TURN CONTROL WHEEL |
| 3 | SAS RECYCLE INDICATOR LIGHTS | 10 | A/P PITCH TRIM SYNCHRONIZATION INDICATOR |
| 4 | SAS LIGHT TEST SWITCH | 11 | A/P PITCH ENGAGE SWITCH |
| 5 | A/P HEADING HOLD SWITCH | 12 | A/P PITCH CONTROL WHEEL |
| 6 | A/P AUTO NAV SWITCH | 13 | A/P MACH/KEAS HOLD SWITCH |
| 7 | A/P ROLL ENGAGE SWITCH | | |

F200-25(c)

Figure 1-25

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The logic circuit is able to isolate a SAS failure in either the electronics or the servos. When a malfunction is isolated, the failed active channel will disengage and the system continues in operation on a single channel. Malfunctioning and disengaging of channels is indicated by indicator lights. The pitch axis can command a maximum elevon surface travel of 2.5° up to 6.5° down. Dual or single channel operation produces the same corrective action of the elevon surface. Power for A channel is from the A phase of No. 1 inverter bus. Power for B channel is from the A phase of No. 2 inverter. Monitor channel power is from the B phase of the No. 3 inverter. Each power source is protected by individual circuit breakers in the cockpit.

STABILITY AUGMENTATION YAW AXIS

The yaw axis of the SAS is very similar to the pitch axis, using two independent A and B channels and a monitor channel. There is one pair of hydraulic servos for each rudder, each pair mounted in a whiffletree arrangement. Damping signals to the rudder do not move the rudder pedals. Each A and B channel drives one servo on each side of the aircraft. The A hydraulic system is connected to A channel and the B hydraulic system to B channel. The rate gyro sensors for the three channels are identical to the pitch rate gyros, except for the physical orientation to sense yawing motions. A "Hi Pass" filter circuit is installed to allow passage of normal short term damping signals, but will stop the signals when a deliberate turn is made. A lateral accelerometer sensor is also used in each channel of the yaw axis. This sensor provides an input for high gain lateral acceleration function to provide a more rapid rudder response during engine failure conditions. However, this function will oppose the pilot when he is purposely trying to sideslip.

The logic circuit is identical to the pitch axis and functions in the same manner. The yaw axis can produce a maximum rudder travel of 8° left to 8° right. Corrective

surface motion is the same regardless of one or two channel operation due to automatic gain doubling if only one channel is operative. Power for A channel is from the B phase of the No. 1 inverter, B channel from the B phase of the No. 2 inverter and the monitor channel from the B phase of the No. 3 inverter. The circuitry from each power source is protected by individual circuit breakers.

STABILITY AUGMENTATION ROLL AXIS

Roll axis reliability requirements are not as severe as pitch and yaw; therefore, less complicated circuitry and components are used. The roll axis has two independent channels, each operating the elevons on one side of the aircraft. A channel positions the left elevon surfaces and operates from the A hydraulic system. B channel positions the right elevon surfaces and operates from the B hydraulic system. There is no monitor channel. Each channel can be operated individually. Although the system gain is the same as two channel operation, roll control is not symmetrical. Coupling into the yaw and pitch axes is possible, but the systems operating in those axes minimize undesirable aircraft motion. Maximum elevon travel in the roll axis is 2° up to 2° down (each side), for a total of 4° differential with both systems operating. Power for A channel is from C phase of the No. 1 inverter and B channel from C phase of the No. 2 inverter.

STABILITY AUGMENTATION SYSTEM (SAS) CONTROL PANEL

The SAS control panel on the right console contains six channel switches, for A and B channels of the pitch, roll and yaw axis. The panel also contains a press-to-test switch and six indicator lights for the A, B and MON channels in the pitch and yaw axis. Three guarded switches for the backup pitch damper, pitch logic override and yaw logic override are located on the right side of the annunciator panel. A roll channel disengage light is located between the roll channel switches. Individual circuit breakers are located on both right and left consoles.

SECTION I

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Channel Switches

There are six toggle switches located on the SAS control panel. There is one pair for each axis; pitch, roll and yaw. The forward switch of each pair is A channel and the rear switch is B channel. The switches have two positions; ON (forward) and OFF (aft). When electrical power is on the aircraft and the channel switches are OFF, the SAS electronics are powered, but the channel servos are not engaged into the control system. Moving the switches to the ON position engages the SAS servos providing the recycle light is extinguished. If the recycle light is not extinguished it must be depressed for engagement.

Recycle Indicator Lights

Six indicator lights are located on the SAS control panel adjacent to the pitch and yaw channel engage switches. One light is provided for each A, B and MON channel in the pitch and yaw axes. When the channel switch is on and the light is not illuminated, the channel is functioning properly. If the light is illuminated, it indicates that the channel has disengaged and the light may be pressed to recycle the channel. In the event the failure was momentary, this will reengage the channel. If the light reilluminates, the channel is malfunctioning, but it is not necessary to turn the channel engage switch off because the light indicates that automatic disengagement has occurred.

NOTE

The lighted recycle indicator light should be pressed down firmly and released. A control surface transient will occur if a hardover servo exists in that channel. Refer to Section III.

The six recycle lights will be illuminated when electrical power is applied to the aircraft. The channel switches must be on and the recycle lights must be pressed to engage the channel electronics to the servos. When engaged and operating, the channel lights will be out.

Roll Channel Disengage Light

A single roll channel disengage light is located between the two roll channel switches. When illuminated it indicates that both roll channels have disengaged. Disengagement results when the roll servo channel outputs differ by more than an amount equivalent to 0.6° surface deflection. When operating on a single roll channel the light will not be illuminated and disengagement in the event of a failure is not provided. The switch must be ON for the active channel and OFF for the malfunctioning channel.

Light Test Switch

A pushbutton light test switch is located in the center of the SAS control panel. Pressing the button illuminates all SAS lights for test.

Backup Pitch Damper Switch

A guarded BUPD switch is located on the right side of the annunciator panel. It is guarded in the OFF position. It is used in case the SAS pitch channels are unusable due to electronic malfunctions or overheating of the pitch gyro package. In the ON position the backup gyro, located in the electronic compartment, supplies pitch rate signals through an independent electronic channel to either the A or B servos. The pitch logic override switch must be used to select their A or B servo operation.

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NOTE

The primary purpose of the BUPD is to provide an emergency system for pitch stability augmentation during refueling and landing approach. The system is optimized for use at light weight, aft center of gravity and subsonic speeds. It is not intended as an emergency backup system during cruise. Refer to Section III, Emergency Procedures.

SAS Pitch Logic Override Switch

A guarded, three-position SAS pitch logic switch is located on the right side of the annunciator panel. It is OFF in the center guarded position and the logic circuit is operative. Placing the switch in the A (up) position deletes the logic circuit and selects A channel operation. In the B (down) position, the logic circuit is deleted and B channel is selected. The switch must be placed in either the A or B position when the BUPD is used. This selects operation of either the A or B servos.

NOTE

The override switch is only used as an emergency procedure. Refer to Section III.

SAS Yaw Logic Override Switch

A guarded, three-position SAS yaw logic switch is located on the right side of the annunciator panel below the pitch logic override switch. It is guarded in the OFF position. The A (up) position deletes the logic circuit and selects A channel operation. The B (down) position deletes the logic circuit and selects B channel operation.

NOTE

The override switch is only used as an emergency procedure. Refer to Section III.

PITOT-STATIC SYSTEMS

The pitot-static system supplies the total and static pressure necessary to operate the basic flight instruments and air data system components. The pressures are sensed by an electrically heated probe mounted on the nose of the aircraft. The probe and forward nose also serves as an antenna for the high frequency radio. The pitot orifice of the probe is divided inside the head to provide two separate pressure sources. It also has two circumferential sets of four static pressure ports each. One pitot and the aft set of static ports supply pressure signals to the air data computer and inlet air control systems. The other set of pickups supply pitot and static pressure directly to the speed sensors on the ejection seats, the altimeter, the rate of climb and airspeed indicators. An offset head on the left side of the probe provides yaw and pitch pressure signals to the inlet spike controls and to the stall warning light sensor.

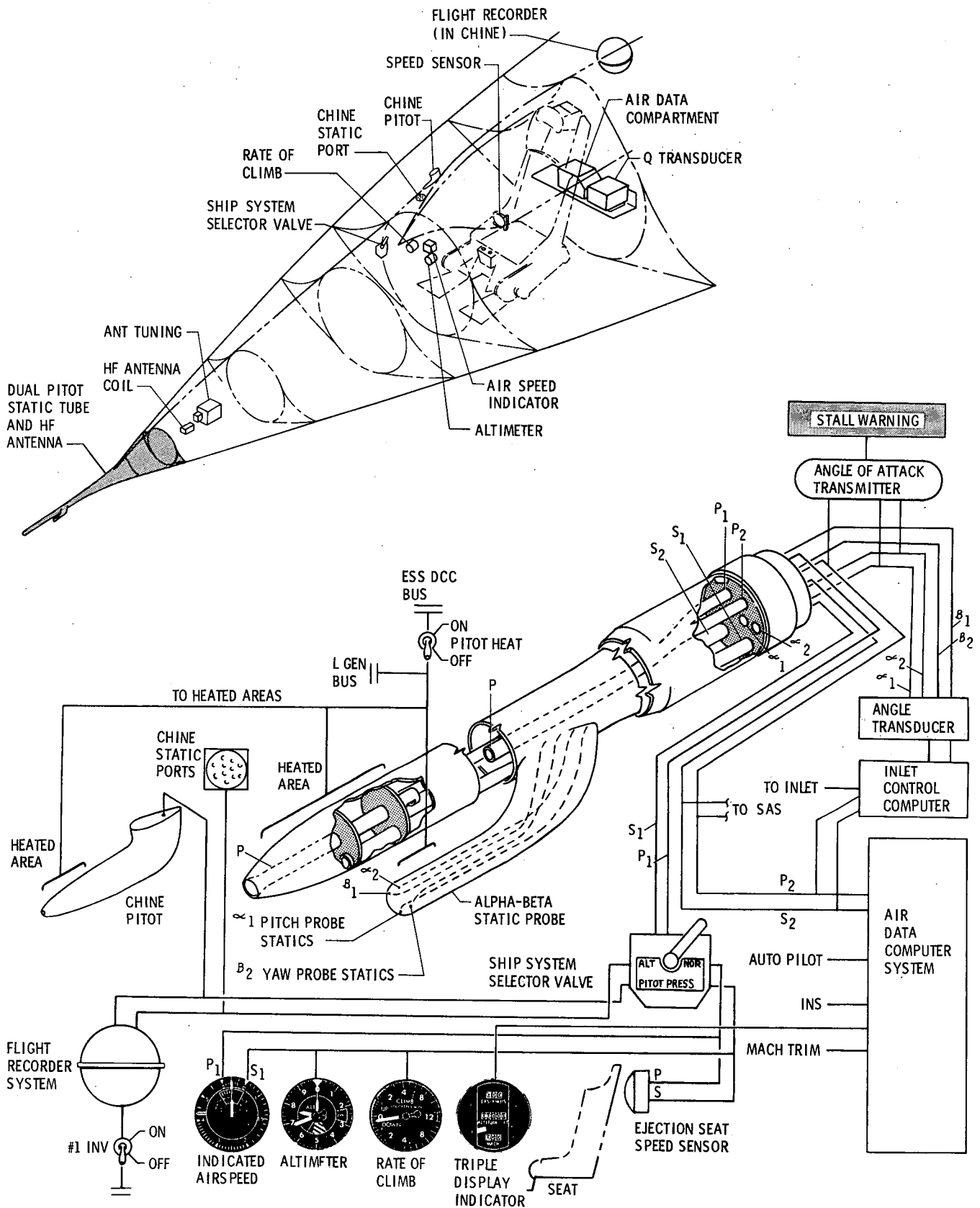
The heating elements of the probe are controlled by the pitot heat switch located on the left side of the annunciator panel. Power is furnished by the left ac generator bus.

An alternate heated pitot static source is available from the Flight Recorder System. Refer to Flight Recorder, Section IV.

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PITOT STATIC SYSTEM



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Figure 1-26

Pitot-Heat Switch and Indicator Light

A two-position toggle switch is located on the left side of the annunciator panel. In the ON (up) position ac power is applied to the heating elements of the pitot-static probe. The probe is grounded to the airframe in a manner which permits the HF radio to be operated while pitot heat is on. In the OFF (down) position ac power is disconnected from the probe heating elements.

The circuitry also incorporates an altitude switch and a PITOT HEAT light located on the annunciator panel. The pitot heat light will be on when the switch is in the ON position and the altitude is above 65,000 feet, and also when the switch is in the OFF position and the altitude is below 50,000 feet. The light will be OFF if when below 50,000 feet and pitot heat is ON, and when above 65,000 feet with the switch in the OFF position.

AIR DATA COMPUTER

The air data computer performs two functions, computation and display. The total and static pressures from the pitot-static probe are converted to electrical signals required for the pilot's triple display indicator, compressor inlet pressure indicator system, the automatic flight control and inertial navigation systems. The ports which supply pressure to the air data computer are separate from those that furnish pressure to the basic flight instruments. Therefore, failure of the air data computer pressure source will not leave the pilot without the altitude, vertical velocity or airspeed information. The air data computer converts pitot-static pressures into proportional rotary shaft positions which are equivalent to pressure altitude and dynamic

pressure. These shaft positions are combined in a mechanical analog computer made up of cams, gears and differentials to drive the output functions. Outputs of the air data computer and the using equipment are listed below:

OUTPUT SIGNALS	USING EQUIPMENT
Pressure Altitude	Triple Display Indicator
Equivalent Airspeed	
Mach	
KEAS + MACH	Compressor Inlet Pressure Indicator
KEAS Mach	Autopilot
Mach Rate	
Altitude	
Dynamic Pressure	
Pressure Altitude	Inertial Navigator Computer

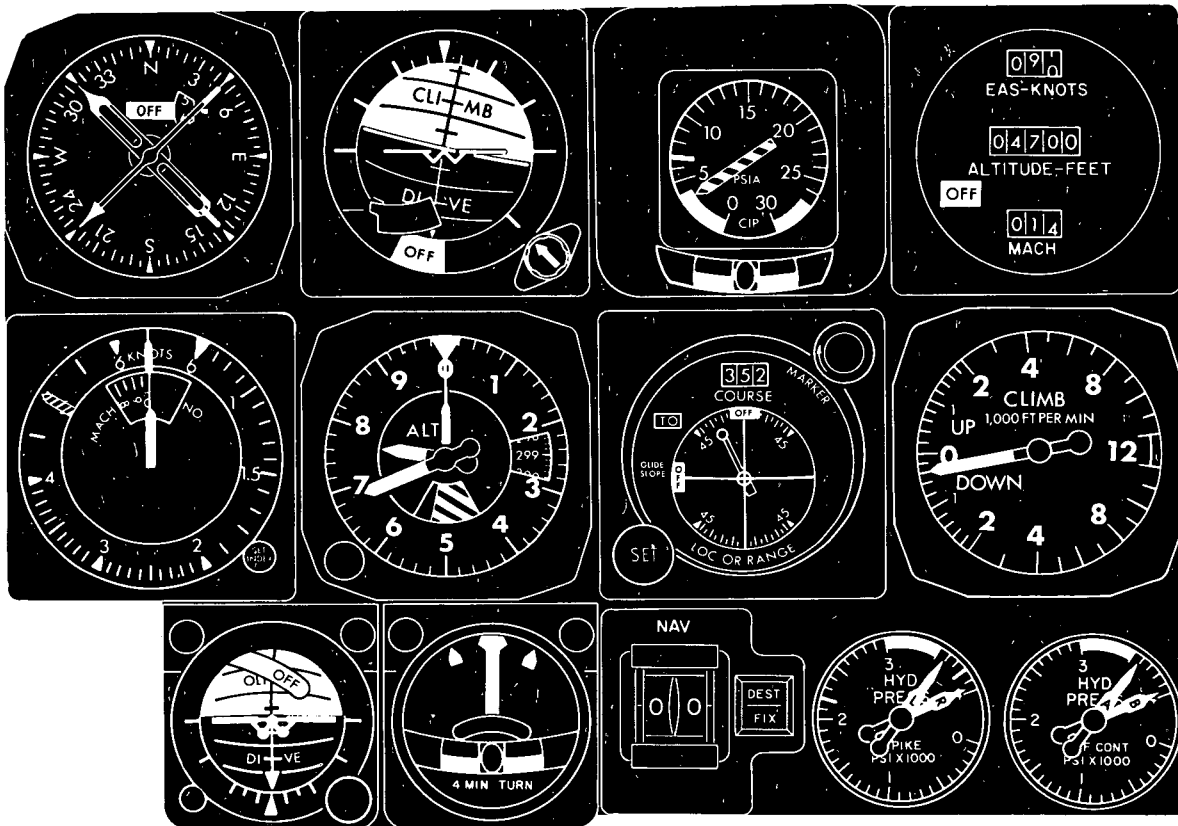
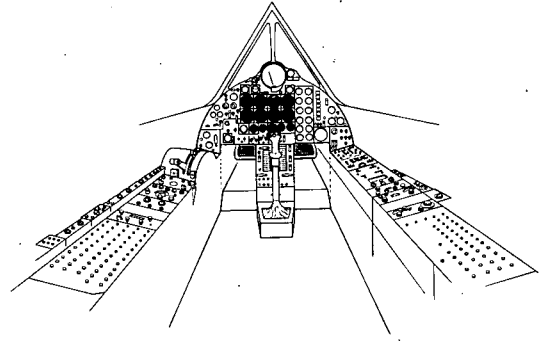
Power for the air data computer is furnished either by the No. 1 or No. 3 inverter depending on the position of the autopilot selector switch.

Triple Display Indicators

A triple display indicator is located on the instrument panel to provide digital displays of airspeed, altitude, and Mach number as computed by the Air Data Computer. The altitude indication range of the TDI is from

SECTION I

FLIGHT INSTRUMENTS



F200-70(d)

Figure 1-27

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0 to 99,950 feet. At 100,000 feet the first digit is dropped, indicating 09,950 feet at 109,950 feet pressure altitude, the maximum limit of the ADC signal to the instrument. The Mach number display capability range of each instrument is 0 to 3.99; however, the minimum indication at static conditions normally ranges from 0.1 to 0.2 Mach number and the maximum indication would be Mach 3.5 for a normally functioning instrument. This range corresponds to the range of signals which the ADC is capable of providing. The TDI displays airspeed in knots equivalent airspeed (KEAS) within an instrument capability from 0 to 599 KEAS; however, the minimum indication is normally 75 to 110 KEAS to correspond with the minimum ADC signal provided. The maximum signal provided by the ADC results in an airspeed indication which decreases from 599 KEAS at sea level to 523 KEAS at 66,800 feet and Mach 3.5, and then decreases further at high altitudes to show the KEAS corresponding to Mach 3.5 and the existing pressure altitude. An off flag appears on the face of the instrument if the ADC loses power. Power for the instrument is from the No. 1 or No. 3 inverter.

NOTE

- . Indications of the triple display indicator and the basic pitot-static flight instruments should be periodically cross checked to confirm proper system operation. Refer to figure A1-2, Appendix I.
- . The triple display indicator is primarily used for aircraft control above FL 180 and to maintain proper airspeed control during climbs to FL 180. Basic pitot-static operated flight instruments shall be used in the landing pattern, during takeoff until proper climb schedule is established on the TDI, and during all simulated or actual instrument flight below FL 180.

- . If KEAS indications oscillate between two values on the high end of the range, it is an indication that the indicator limit is being approached.

INSTRUMENTS

For information regarding instruments that are an integral part of a particular system, refer to applicable paragraphs in this section and Section IV.

Airspeed-Mach Meter

A combination airspeed and Mach meter operating directly from pitot-static pressure is located in the flight instrument group. This is a special instrument with airspeed and Mach number ranges compatible with aircraft performance capabilities. Mach number and indicated airspeed are read simultaneously on the window and outer index respectively. A limit airspeed needle (white barred) shows the airspeed limit of the aircraft. The actual airspeed limit is in equivalent airspeed; however, the needle varies with altitude to read the indicated airspeed that converts to equivalent airspeed.

Altimeter

A sensitive pressure altimeter is located on the instrument panel. In addition to the 1000 foot and 100 foot pointers, it also has a 10,000 foot pointer. This pointer extends to the edge of the dial with a triangular marker at its extremity. The center disc has a cutout through which yellow and black warning stripes appear at altitudes below 16,000 feet. The barometric pressure scale is in a cutout at the right side and is set by a knob located at the lower left side of the instrument.

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Attitude Indicator (MM-3)

The attitude indicator is located in the basic flight instrument group on the instrument panel. It provides constant visual indication of nose and wing position in relation to the earth's surface. Attitude indications are presented by a spherical graduated dial, a W reference line, a bank pointer, and a marked outer ring. A horizontal line is formed on the spherical dial by the meeting of a gray, upper climb section and a black lower dive section. The instrument shows attitude in climb or dive up to 85 degrees.

NOTE

At approximately 85 degrees climb or dive, the attitude indicator will flip but will not tumble. The 180 degree flip in roll will be very rapid and the instrument will accurately indicate pitch and roll attitudes immediately thereafter. Some small inaccuracies may develop after a series of maneuvers beyond the 85 degree climb or dive attitude. These inaccuracies will automatically be cancelled out at the erection rate of $.8^{\circ}$ to 1.8° per minute.

The W reference line remains fixed with the marked outer ring and represents the aircraft in miniature. The spherical dial moves up or down, or the whole spherical dial assembly rotates within the instrument case behind the W reference line and outer ring to indicate aircraft attitudes. As the dial assembly rotates, the bank pointer moves with it to indicate degrees of bank on the outer ring. The outer ring indicates 0° - 90° bank. The spherical dial and pointer are capable of rotating a full 360 degrees of roll with the aircraft. Pitch attitude of the aircraft is indicated by the position of the horizon line in relation to the miniature aircraft. A pitch adjustment knob on the lower right side is used to change the position of the spherical dial as desired. During initial gyro erection, and when power is off or is insufficient to keep

the gyro stabilized, a warning OFF flag appears at the bottom of the indicator. The autopilot and attitude reference selector switch is used to select pitch and roll attitude signals from either the INS or FRS stable platforms.

CAUTION

To avoid gross pitch attitude errors the pitch adjustment knob of the attitude indicator should be adjusted to align the index marks before the autopilot and attitude reference selector switch is changed in flight.

NOTE

To determine a possible malfunction of the attitude indicator, an occasional accuracy check should be made by comparing it against the standby attitude indicator and other basic flight instruments.

The system is powered by the No. 1 and No. 3 inverter depending on the position of the autopilot selector switch.

Standby Attitude Indicator

The standby attitude indicator located on the lower left side of the instrument panel provides the pilot with an independent attitude reference. It contains a sphere inscribed with an artificial horizon and calibrated in degrees of aircraft angle of pitch. The globe is detailed to represent the sky and earth areas, and is capable of rotating to indicate pitch angles of ± 82 degrees and roll angles of 360 degrees. The bank angle scale is marked on the lower periphery. A pitch reference adjustment knob is provided on the lower right corner of the instrument for positioning the reference bar as desired. A fast erect pushbutton is provided on a small panel above the throttles.

ANNUNCIATOR PANELS

	STALL WARNING	INS FIX REJECT
NO. 1 OXY LOW	MANUAL INLET	
NO. 2 OXY LOW	L OIL TEMP	R OIL TEMP
Q-BAY HEAT HIGH		
FUEL QTY LOW	L FUEL PRESS LOW	R FUEL PRESS LOW
N QTY LOW	A HYD PRESS LOW	B HYD PRESS LOW
TANK PRESSURE LOW	L GENERATOR OUT	R GENERATOR OUT
ANTI-SKID OUT	L XFMR-RECT OUT	R XFMR-RECT OUT
SURFACE LIMITER	NO. 1 INVERTER OUT	EMER BAT ON
SAS CHANNEL OUT	NO. 2 INVERTER OUT	R HYD LOW
A HYD LOW	NO. 3 INVERTER OUT	L HYD LOW
B HYD LOW	PITOT HEAT	Q-BAY EQUIP OUT

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Figure 1-28

CAUTION

Do not hold fast erect button for more than 45 seconds to prevent overheating of fast erect motor.

This instrument has its own self-contained gyro and is not dependent on another reference source. The OFF flag will be visible whenever power to the indicator is interrupted. Power is provided by the C phase of the No. 2 inverter.

Vertical Velocity Indicator

A vertical velocity indicator is located on the instrument panel and shows the rate of change of altitude in feet per minute. Changes in pressure due to changes in alti-

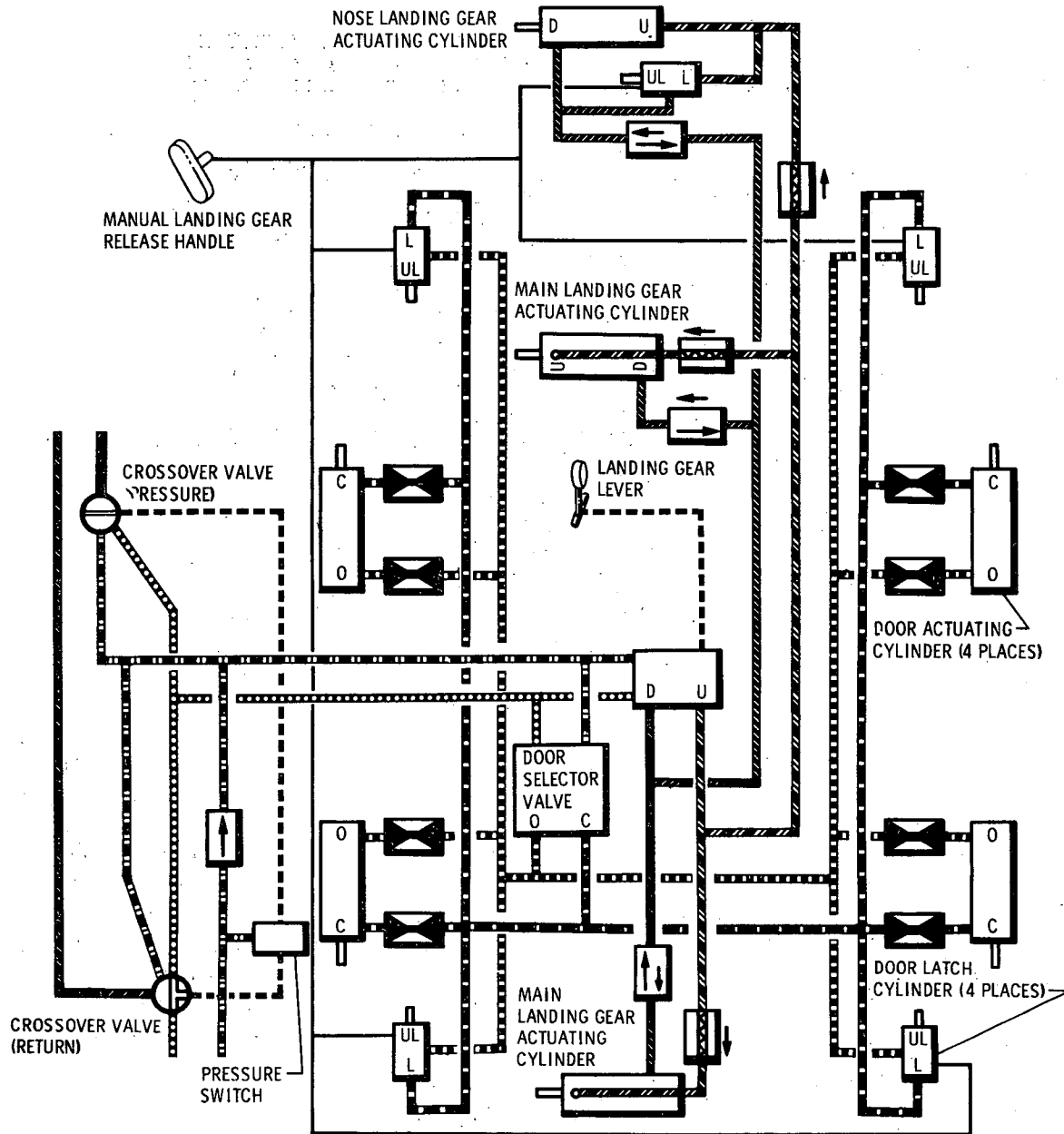
tude are sensed by the static system and transmitted to the indicator. Depending on the instrument installed the instrument is capable of indicating vertical speed of 0 to + 12,000 feet per minute or 0 to 6,000 feet per minute. An over-pressure diaphragm and valve prevent excessive rates of climb or descent from damaging the instrument.















Turn and Slip Indicator

A turn and slip indicator is installed on the instrument panel. The indicator is calibrated for either a two or four minute turn. The indicator is powered by the essential dc bus. An additional larger slip indicator is mounted on the upper center instrument panel beneath the CIP indicator.

SECTION I

LANDING GEAR SYSTEM



- | | | | |
|---|---|--|-------------------|
|  | CABLE |  | R SYSTEM PRESSURE |
|  | ELECTRICAL CONNECTION |  | R SYSTEM RETURN |
|  | CHECK VALVE |  | L SYSTEM PRESSURE |
|  | RESTRICTOR VALVE (SMALL ARROW INDICATES DIRECTION OF RESTRICTED FLOW) |  | L SYSTEM RETURN |
|  | FLOW REGULATOR |  | MLG DOORS CLOSED |
|  | RESTRICTOR VALVE (RESTRICTED FLOW IN BOTH DIRECTIONS) |  | MLG DOORS OPEN |
| | |  | LANDING GEAR DOWN |
| | |  | LANDING GEAR UP |

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Figure 1-29

Clocks

An elapsed time clock is located on the instrument panel. It contains an elapsed time mechanism that is started and stopped by pushing the winding knob. An A13A clock is also installed in the panel. The second hand is started and stopped by the small button on the upper right corner. The third hand serves as a 60 minute recorder.

EMERGENCY EQUIPMENTMASTER WARNING SYSTEM

An annunciator panel is mounted on the lower instrument panel. The panel contains individual warning lights that indicate malfunctions or failures of equipment and systems. Illumination of any individual light also illuminates an amber master caution light on the upper portion of the instrument panel. Once illuminated, the master caution light can be extinguished (reset) by depressing the light. The individual annunciator panel light will remain illuminated. Another malfunction again illuminates the master caution light. Warning lights are automatically dimmed when the instrument panel lights are on. The master warning system does not include the fire warning and landing gear unsafe lights. Power is furnished by the essential dc bus.

NACELLE FIRE WARNING SYSTEM

A fire warning system detects and indicates the presence of a fire in the engine nacelles. A hot spot anywhere along the length of the detection circuit will illuminate the light of that particular nacelle. The lights are located on the pilot's instrument panel above the respective column of instruments pertaining to each engine.

Nacelle Fire Warning Lights

Left and right nacelle FIRE warning lights located on the top right side of the instrument panel, illuminate when nacelle temperature at the turbine or at the afterburner exceeds $1050^{\circ}\text{F} \pm 50^{\circ}$. Flip down glare shields are provided for night flying. Power for the circuit is furnished by the No. 1 inverter.

STALL WARNING LIGHT

A STALL WARNING light is located on the annunciator panel which is illuminated when the aircraft angle of attack reaches ± 14 degrees and the nose landing gear scissor switch is open. Pressure differences between the α 1 and 2 inlets on the pitch and yaw probe are sensed by a pitch transmitter unit to actuate this light. A steady tone warning signal is also produced in the pilot's earphone. Power for the stall warning light is furnished by the essential dc bus.

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. Down locks inside the actuating cylinders hold the gear in place in the extended position. Hydraulic pressure is also on the gear in the extended position when L system pressure is available. The landing gear cylinders and doors are actuated in the proper order by two sequencing valves. Normal gear operation is

SECTION I

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powered by the L hydraulic pump on the left engine. Should pressure drop to 2000-2200 psi during retraction, the power source automatically becomes the R hydraulic pump. R hydraulic pressure will not extend the gear in the event of an L system failure and the manual landing gear release must be used. Normal gear extension time is 12-16 seconds.

LANDING GEAR LEVER

A wheel shaped landing gear lever is installed on the lower left side of the instrument panel just forward of the throttle quadrant. The lever has two positions; UP and DOWN. A locking mechanism is provided to prevent the gear lever from being inadvertently placed in the DOWN position. A button which extends upward from the top of the lever must be pressed forward in order to release the lock mechanism. An override button is installed just above the gear lever and may be used to override the ground safety switch should it become necessary to raise the gear when the weight of the aircraft is on the landing gear. Once energized, the gear lever must be recycled to the DOWN position in order to bring the ground safety switch back into the circuit. A red light installed in the transparent wheel illuminates during cycling, or when the gear is in an unsafe condition. Power for the circuit is furnished by the essential dc bus.

Manual Landing Gear Release Handle

A manual landing gear release handle labeled GEAR RELEASE is installed on the annunciator panel. If the L hydraulic system has failed but R hydraulic pressure is available, the landing gear lever must be in the DOWN position or the landing gear CONT circuit breaker must be pulled out before pulling the GEAR RELEASE handle. Otherwise, the R system will retract the

gear. The gear extends by gravity force. Approximately 9 inches of pull on the handle is required since the uplocks are released at different positions along the cable length. The nose gear uplock is released first followed by the right gear then the left. Gear retraction is possible after being lowered by the manual gear release handle, provided L or R hydraulic system pressure is available.

Gear and Warning Light Test Button

A gear and warning light pushbutton switch is located on the left forward panel. When depressed it illuminates the landing gear lever red light, all annunciator panel lights, the right and left nacelle fire warning lights, and actuates the gear warning tone in the headset. It is also used to test the three green landing gear position lights when airborne.

Landing Gear Position Lights

Three green lights, located on the left side of the instrument panel indicate the down and locked condition of the landing gear. The location of each light corresponds to the respective wheel it monitors. Power is from the essential dc bus.

Landing Gear Warning Light and Audible Warning

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

1. Gear is cycling.
2. Gear system is not locked in the UP or DOWN position.
3. Gear is UP and throttle settings are below MILITARY and altitude is below 10,000 feet.

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A pulsed tone warning signal is also produced in the pilot's earphones when the throttles are retarded below approximately 1/3 the distance between the IDLE and MIL throttle settings, the landing gear is not in the down and locked position and aircraft altitude is below 10,000 feet \pm 500 feet. Power for the light and pulsed tone warning is furnished by the essential dc bus.

Landing Gear Warning Cutout Button

The audio gear warning signal can be eliminated by pressing the GR SIG REL push-button switch on the instrument panel. The circuit is reactivated when the throttles are advanced above the minimum cruise setting. Power is supplied by the essential dc bus.

Land Gear Ground Safety Pins

Removable ground safety pins are installed in the landing gear assemblies to prevent inadvertent retraction of the gear while the aircraft is on the ground. Warning streamers direct attention to their removal before flight. An additional set of ground safety pins is provided in a container behind the seat.

LANDING GEAR STRUT DAMPER

A landing gear strut damper system is installed to control gear "walking" during brake operation. The system is sensitive to less than one 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.

NOSEWHEEL STEERING SYSTEM

The nosewheel steering system provides power steering for directional control when the aircraft weight is on any one gear. The nosewheel is steerable 30 degrees either side of center. Steering is accomplished by a hydraulic steer-damper unit controlled through a cable system by the rudder pedals. L hydraulic system pressure from the nose landing gear down line is routed to the steering system through a shutoff valve, which is controlled by the nosewheel steering (NWS) button on the control stick grip. Steering is engaged by depressing the NWS button and matching pedal position with nosewheel angle. A holding relay circuit allows the NWS button to be released after it is once depressed and steering will stay engaged. It is disengaged when the NWS button is again pressed and released. Steering is engaged at any time the NWS button is held depressed. Nosewheel steering radius is approximately 75 feet. A mechanically operated centering cam automatically centers the nosewheel when it retracts. Power for the system is furnished by the essential dc bus.

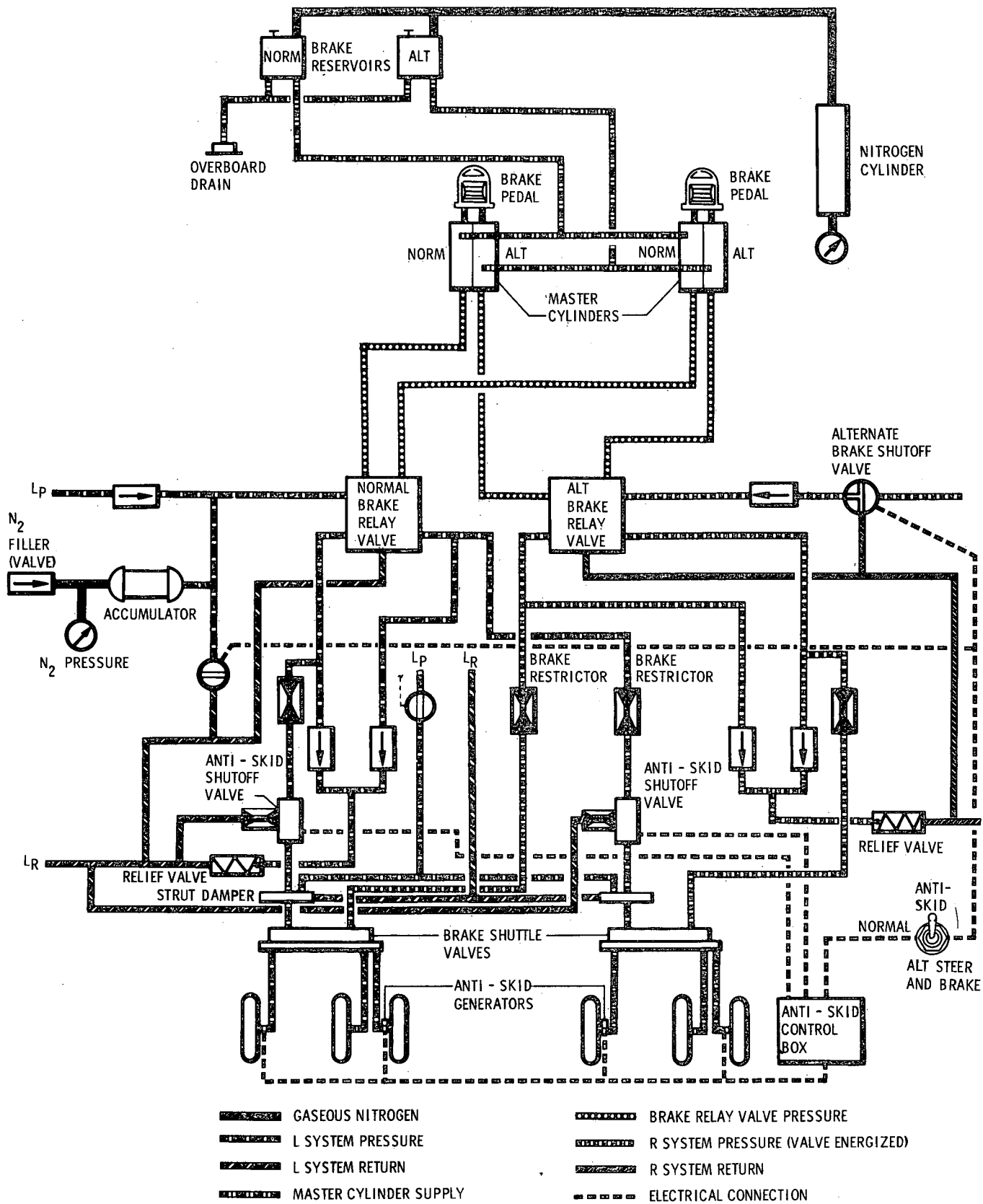
NOTE

Nosewheel steering is operable only if essential dc bus power is available and weight of the aircraft is on any one gear. If the L system pressure should drop below 2000-2200 psi alternate nosewheel steering may be obtained by placing the brake switch to ALT STEER & BRAKE position.

WARNING

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

BRAKE SYSTEM



F200-27(b)

Figure 1-30

WHEEL BRAKE SYSTEM

The aircraft is equipped with artificial feel hydraulically operated power brakes. Depressing the rudder pedals actuates the four rotor brakes on each of the six main wheels. The L hydraulic system furnishes brake pressure with optional antiskid operation. The hydraulic pressure to the brakes is approximately 1200 psi. Should the L hydraulic system fail, alternate brakes are available. The alternate brakes operate from an independent system using R hydraulic pressure with no antiskid provision.

A small accumulator is incorporated in the normal brake system which should provide up to five brake applications after L and R hydraulic failure provided accumulator pressure has not been dumped by selecting alternate brakes. Certain types of hydraulic system failures such as a broken line could deplete the system fluid. Normal or antiskid brakes are usable if left hydraulic pressure is steady and above 2200 psi. Alternate brakes are used if left hydraulic system pressure is below this pressure.

Brake Switch

A three-position brake switch is located on the left side of the instrument panel. In the NORM (center) position, brake pressure from the L hydraulic system is available, but the antiskid system is not operative. In the ANTISKID (up) position, the antiskid system is operative. In the ALT STEER & BRAKE (down) position, the brakes, NWS and air refueling system are powered by the R hydraulic system if left system pressure is below 1250 psi. Power for the circuit is furnished by the essential dc bus.

WARNING

Do not switch to alternate brakes unless normal left hydraulic pressure is unavailable or normal brakes are inoperative. Pressure may be trapped in the brakes after the pedals are released, causing grabbing or locking.

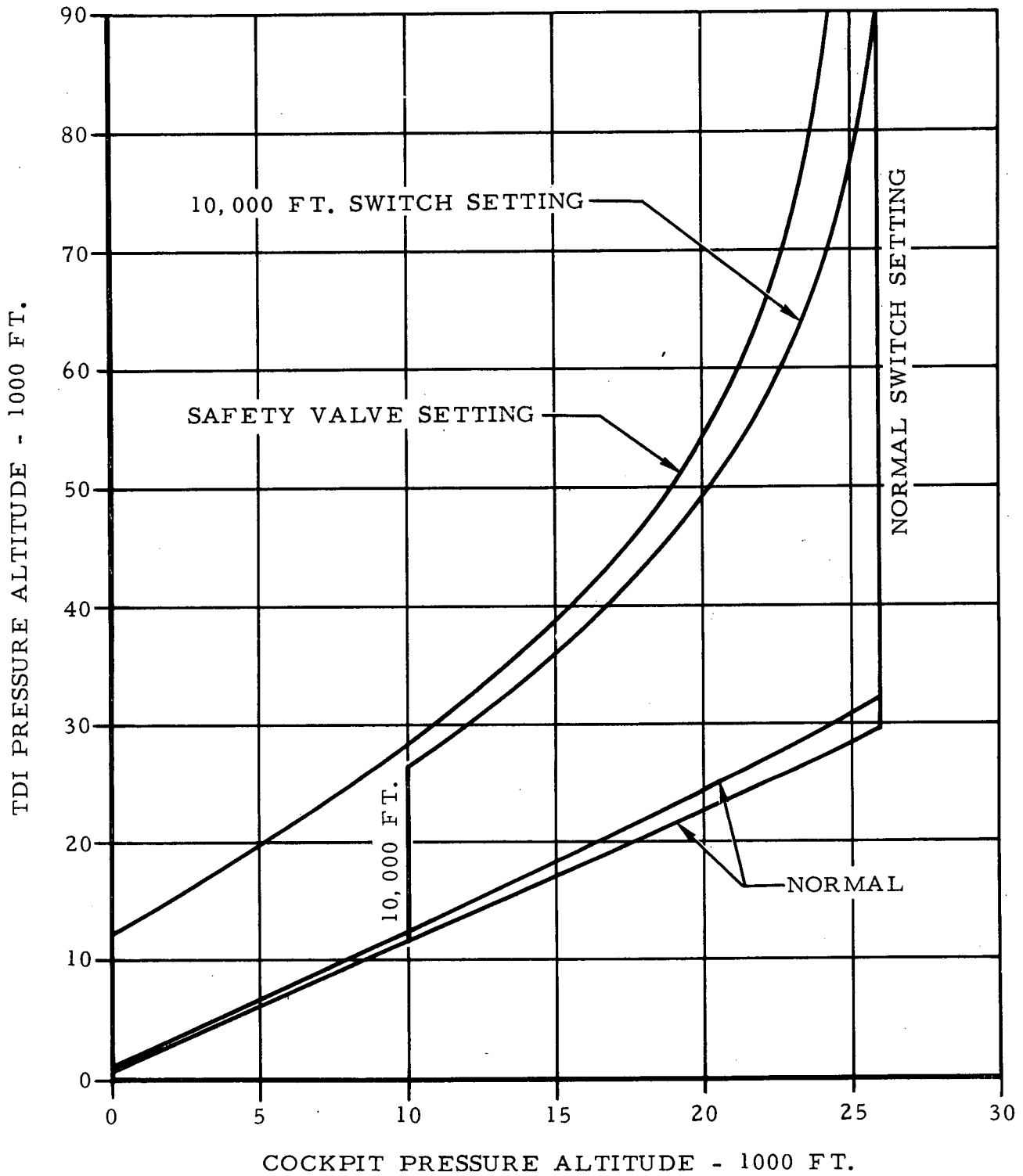
Anti-skid Out Indicator Light

Illumination of the ANTI-SKID OUT indicator light on the annunciator panel indicates that the anti-skid system is inoperative. When the aircraft is on the ground, the light will be illuminated when the brake switch is in the NORM or ALT STEER & BRAKE position. The light will be off when the switch is in the ANTI-SKID position, if the anti-skid control box and wheel generators are operative. If the fail safe circuit within the anti-skid control box is tripped and power from the essential dc bus is on the system, the light will illuminate. The light is off at all times when the weight of the aircraft is not on the gear.

DRAG CHUTE SYSTEM

The drag chute system is provided to reduce landing roll and aborted takeoff roll out distance. The 45-foot ribbon type parachute is packed in a deployment bag and stowed in the upper aft end of the fuselage. It rides free in the compartment and is locked onto the airplane at the initial stage of its deployment action. The neck of the drag chute link incorporates a breakaway section to protect against aircraft structural damage if the chute is deployed at too fast a speed. The chute deployment is actuated electrically and power is furnished by the essential dc bus.

COCKPIT PRESSURIZATION SCHEDULE



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Figure 1-31

Drag Chute Handle

The drag chute deploy and jettison handle is located on the left edge of the instrument glare shield. When pulled the handle activates micro switches which deploys the drag chute. When turned 90 degrees counter-clockwise and pushed in, the drag chute is jettisoned. Power for the circuit is furnished by the essential dc bus.

AIR CONDITIONING AND PRESSURIZATION SYSTEM

Similar left and right hand air conditioning and pressurization systems utilize high pressure ninth stage compressor air from each engine to pressurize and cool the cockpit and equipment compartments. System shutoff valves allow compressor air to flow when the engines are running and the system switches are ON. Cooling is accomplished by ducting the bleed air through a ram air heat exchanger, primary and secondary fuel/air heat exchangers, and through an air cycle refrigerator. Temperature of the air supplied by each system is modulated by temperature control bypass valves located upstream from the air cycle refrigerators. The bypass valves are positioned by control switches located in the cockpit.

A water separator is installed in each air conditioning system downstream of the air-cycle refrigeration units. Below an altitude of approximately 36,000 feet a pressure switch in the automatic temperature control circuit limits the minimum outlet temperature of the air from the air-cycle refrigeration to 35° F to prevent freezing of water in the separator. Using the manual temperature controls will allow lower temperature air to come from the refrigerator but icing of the water separator may occur if humidity is high. Above 36,000 feet the altitude pressure switch opens the water separator bypass valve and air does not flow through the separator.

The left engine normally furnishes air for the cockpit, nose compartment, ventilated flying suit, inverters and INS platform. The right engine normally furnishes air to the E-bay where it mixes with cockpit discharge air for ventilation of the E-bay, Q-bay, and the aft equipment compartments. A fixed orifice restriction and a duct dividing into two outlets provide for a portion of the right system air to flow to the upper part of the cockpit. A crossover system is provided to supply right engine system air to the cockpit and equipment normally supplied by the left engine system. The operation of the crossover system will not depressurize the Q-bay since the cockpit air exhausts into the Q-bay; however, a rise in temperature will occur in the Q-bay. High pressure canopy and hatch seal air and windshield defog air is furnished from both right and left engine systems by ducts connected downstream from the primary fuel/air heat exchangers.

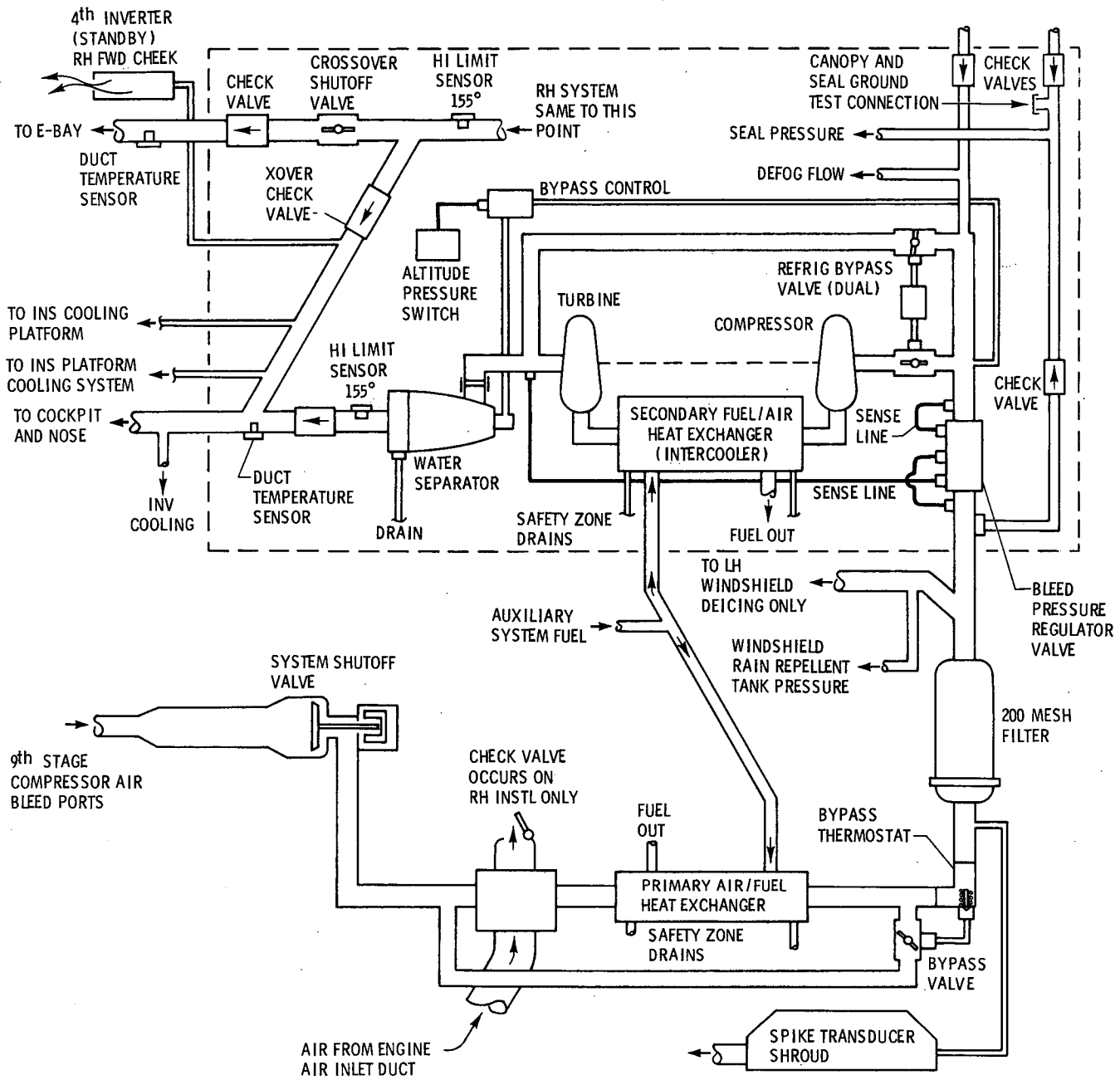
COCKPIT COOLING AND PRESSURIZATION

When the aircraft is at high altitude, the pressurization systems maintain a constant altitude of approximately 26,000 feet in the cockpit and nose and 28,000 feet in the Q-bay.

Cabin Pressure Schedule Switch

The cockpit pressure schedule switch is a two position toggle switch labeled CABIN PRESS located on the lower center of the instrument panel. In the NORMAL (down) position, the cockpit and Q-bay pressurization systems provide the normal pressure schedule and will maintain constant altitudes of 26,000 and 28,000 feet when the aircraft is above 32,000 feet. In the 10,000 feet (up) position, the cockpit pressure is regulated to a 5 psi maximum differential and will maintain a 10,000 foot cockpit altitude up to

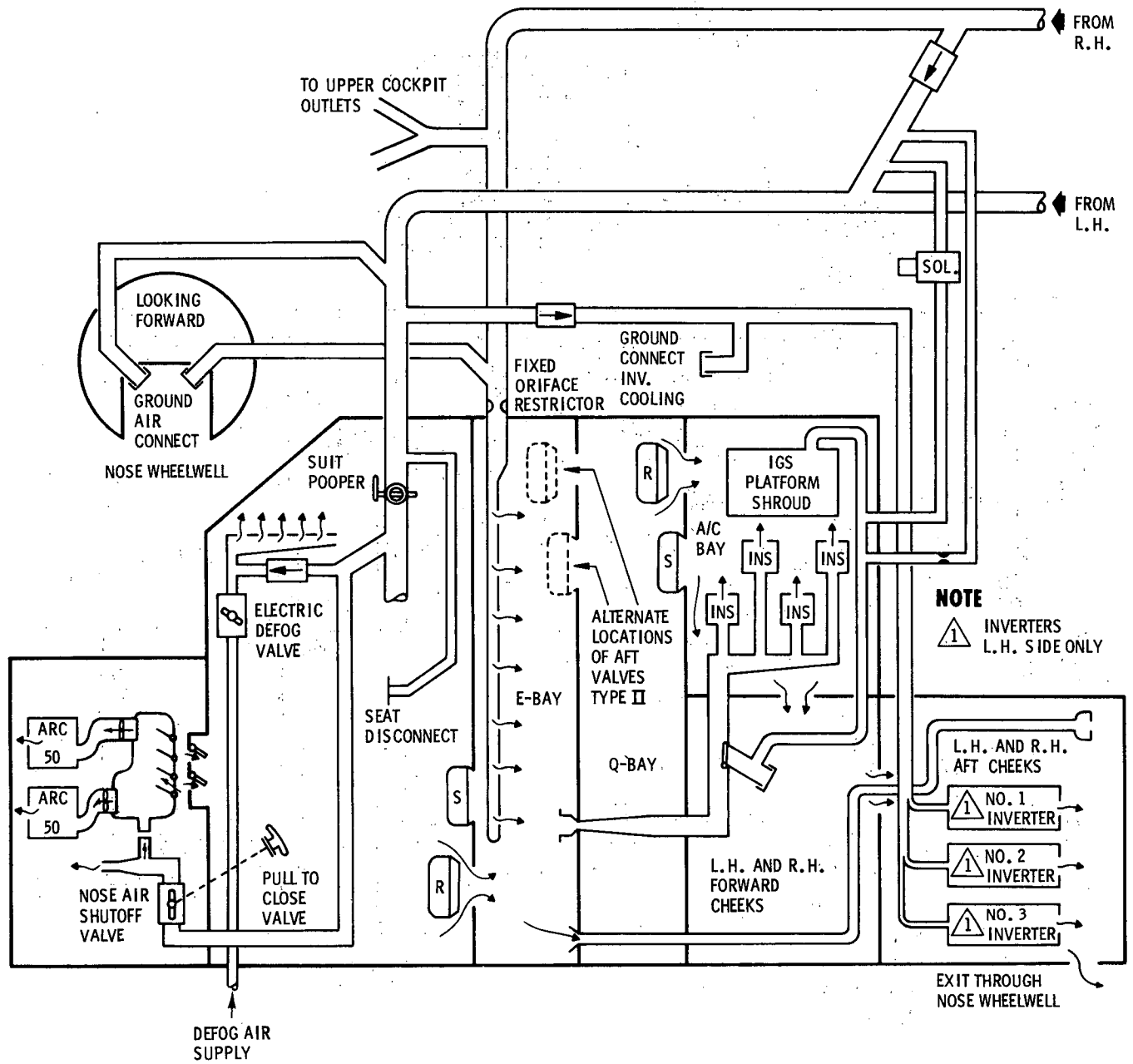
AIR CONDITIONING AND PRESSURIZATION SYSTEM



F200-7(1)(d)

Figure 1-32 (Sheet 1 of 3)

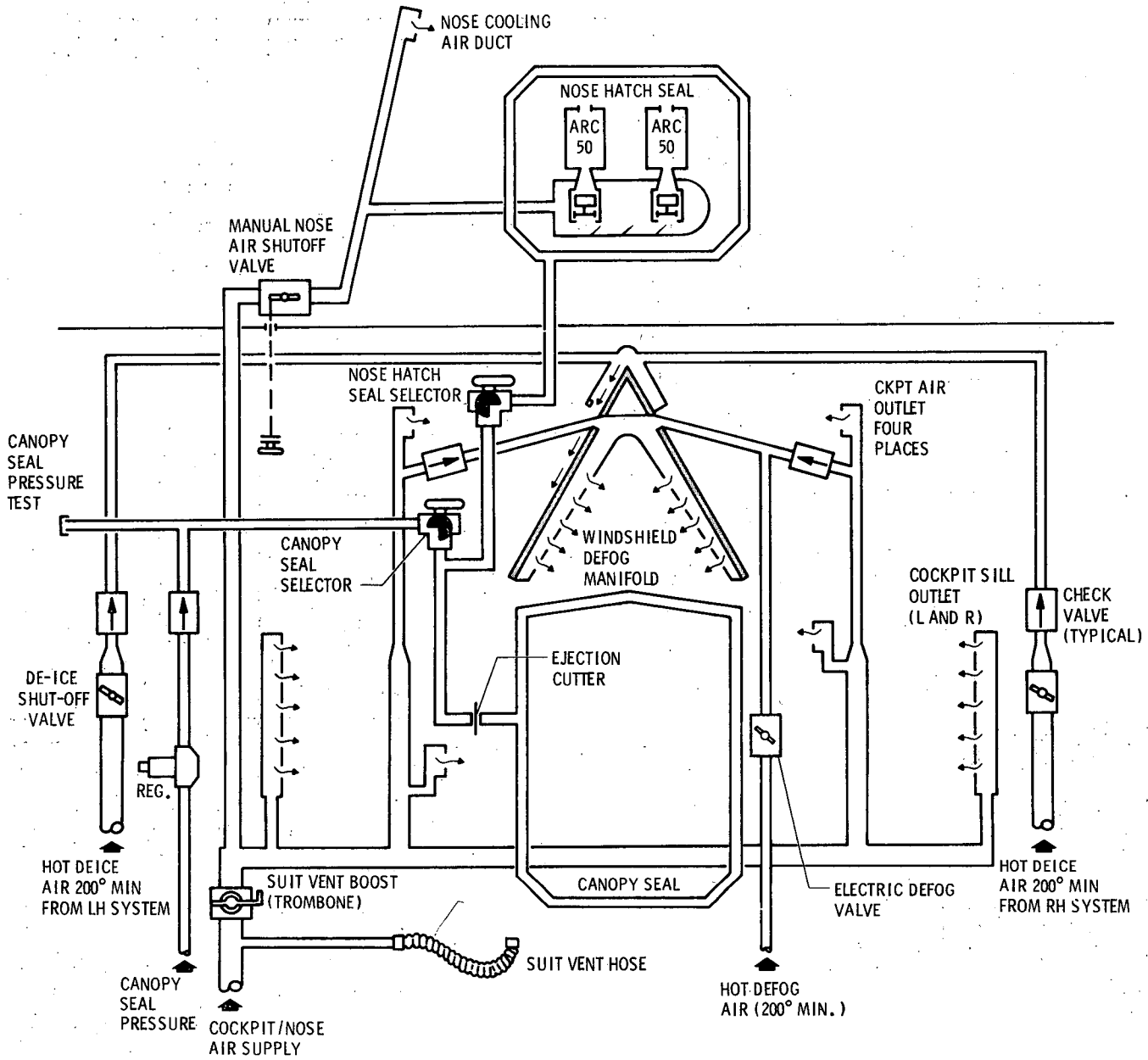
AIR CONDITIONING AND PRESSURIZATION SYSTEM



F200-7(2)(d)

Figure 1-32 (Sheet 2 of 3)

AIR CONDITIONING AND PRESSURIZATION SYSTEM



F200-7(3)(d)

Figure 1-32 (Sheet 3 of 3)

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26,500 feet. The 10,000 foot position is intended for use during subsonic low altitude ferry flights but is not restricted for use as desired during climbs, descents and high altitude cruise. A rate control is incorporated which limits the pressure change to 2500 ft/min when changing schedules.

NOTE

During descents from high altitude, only the normal cockpit pressure schedule will provide optimum cockpit cooling. The 10,000 foot schedule will not cool the cockpit in descents as well as the normal schedule due to increased turbine back pressure.

Cockpit Air Switch

The cockpit air switch is a three position switch with labeled positions of NORM (left) OFF (center) and EMER (right). In the NORM position the left system shutoff valve is deenergized to open and the left engine system furnishes air to the cockpit. In the OFF position the left shutoff valve is energized to closed, shutting off the normal cockpit air. In the EMER position left system air is shutoff, the crossover valve in the right system is energized closed and the right system shutoff valve is deenergized to open and right system air is furnished to the cockpit. The circuit is powered by the essential dc bus.

NOTE

In the EMER position the Q-bay system switch OFF position is ineffective and right system air must be shut off by moving the cockpit air switch to the NORM position.

Q-Bay System Switch

The Q-bay system switch has two positions and is located on the upper left side of the instrument panel. In the ON (up) position the right engine system's shutoff valve is deenergized to open so that right engine air can flow to the E-bay. If the cockpit air switch is in the crossover or EMER position this air will be ducted to the cockpit and will enter the E-bay through the cockpit regulator valving. In the OFF position the shutoff valve is energized to off and Q-bay system air is shutoff if the cockpit air switch is in NORM position. The circuit is powered by the essential dc bus.

Temperature Control Selector Switches

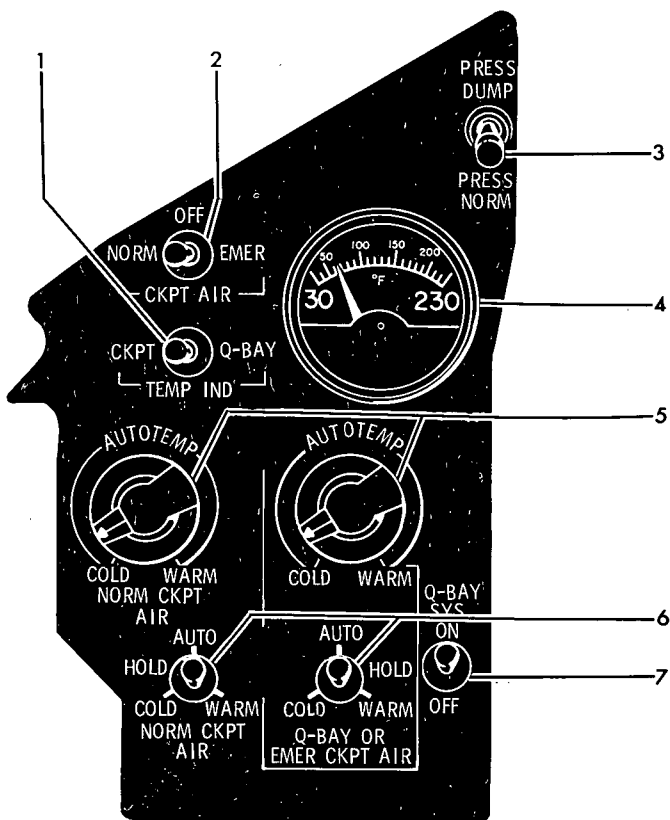
Two selector switches, one for the cockpit and one for the Q-bay and/or emergency cockpit air, are installed on the upper left instrument panel. Each switch has four positions; AUTO (up), COLD (down left), WARM (down right) and HOLD (center). The switches are spring loaded to HOLD from the COLD and WARM positions. The switches will normally be in the AUTO position; however, the pilot can manually override the automatic feature by moving the switch to either the momentary COLD or WARM position. The manual COLD control will provide colder air, if required, than the automatic control. The No. 1 inverter powers the cockpit temperature control system. The No. 2 inverter powers the Q-bay and/or emergency cockpit air temperature control system.

Temperature Indicator Selector Switch

A temperature indicator selector switch located on the upper left instrument panel allows the pilot to monitor cockpit or Q-bay temperature. Cockpit temperature is indicated when the switch is placed in the CKPT (left) position and Q-bay temperature when

SECTION I

AIR CONDITIONING CONTROL PANEL



- 1 COCKPIT TEMPERATURE MONITOR SELECTOR SWITCH
- 2 COCKPIT AIR SWITCH
- 3 DEPRESSURIZATION SWITCH (DUMP)
- 4 TEMPERATURE INDICATOR
- 5 TEMPERATURE CONTROL KNOBS
- 6 TEMPERATURE CONTROL SELECTOR SWITCHES
- 7 Q-BAY SYSTEM SWITCH
- 8 CABIN PRESSURE SCHEDULE SELECTOR SWITCH
- 9 CABIN ALTITUDE GAGE
- 10 ALTITUDE INDICATOR SELECTOR LEVER

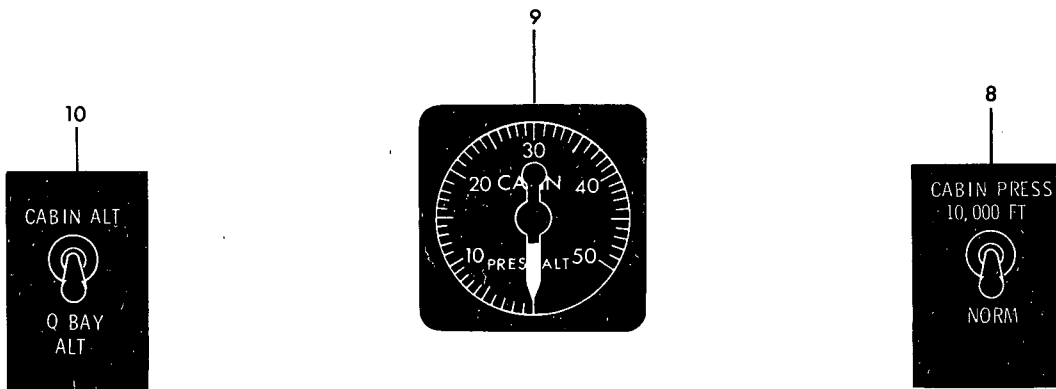


Figure 1-33

the switch is placed in the Q-BAY (right) position. Power for the indicator is furnished by the essential dc bus.

NOTE

Up to a point, the insulation and ventilation of the pressure suit will keep the pilot comfortable in a cockpit environment that is too warm. The temperature indicator is provided so as to allow anticipation of a temperature condition that might eventually become too hot for comfort. If the cockpit temperature approaches 140°, the suit will not keep the pilot comfortable.

Temperature Control Rheostats

Two temperature control rheostats, one for the cockpit and one for the Q-bay and/or emergency cockpit air are installed on the upper left instrument panel. Arrows indicate the direction of rotation necessary to increase temperature. Generally, it is necessary to periodically rotate the respective temperature control rheostat toward the COLD position to maintain a comfortable temperature in the ventilated flying suit and keep the Q-bay temperature in tolerance. Electrical power for the cockpit temperature control circuits is from the No. 1 inverter. Q-bay and/or cockpit emergency air control is powered by the No. 2 inverter.

Pressure Altitude Gage

A cockpit and Q-bay pressure altitude gage is located on the left forward panel and indicates either cockpit or Q-bay altitude as selected by the cabin-Q-bay selector.

Altitude Selector Lever

This switch type lever is located on the left forward panel. It is labeled CABIN ALT in the up position and Q-BAY ALT in the down position and selects the respective pressure altitude to be indicated on the gage.

Depressurization (Dump) Switch

A two position lift-lock depressurization switch labeled PRESS DUMP and PRESS NORM is located on the upper left instrument panel. When the switch is pulled out and moved to the PRESS DUMP position, both the cockpit and Q-bay will be depressurized by the opening of the safety valves. When moved to the PRESS NORM position the safety valves will close and the cockpit and Q-bay will repressurize.

WARNING

Depressurization and repressurization will occur at an extremely rapid rate.

Nose Hatch Seal Shutoff Lever

A nose hatch seal shutoff lever, located on the forward right side of the cockpit, operates the nose hatch seal shutoff valve. It is normally in the ON position to allow canopy seal pressure to inflate the nose hatch seal. In the OFF position the nose hatch seal is isolated from the canopy seal system. This prevents the deflation of the cockpit canopy seal in the event of excessive nose hatch seal leakage.

Nose Air Shutoff Handle

A nose air shutoff T-handle is located at the bottom of the annunciator panel. It is normally in the locked ON position. The handle is turned counterclockwise to unlock and then pulled out to shut off airflow to the pressurized nose compartment.

SECTION I

LIQUID OXYGEN SYSTEM

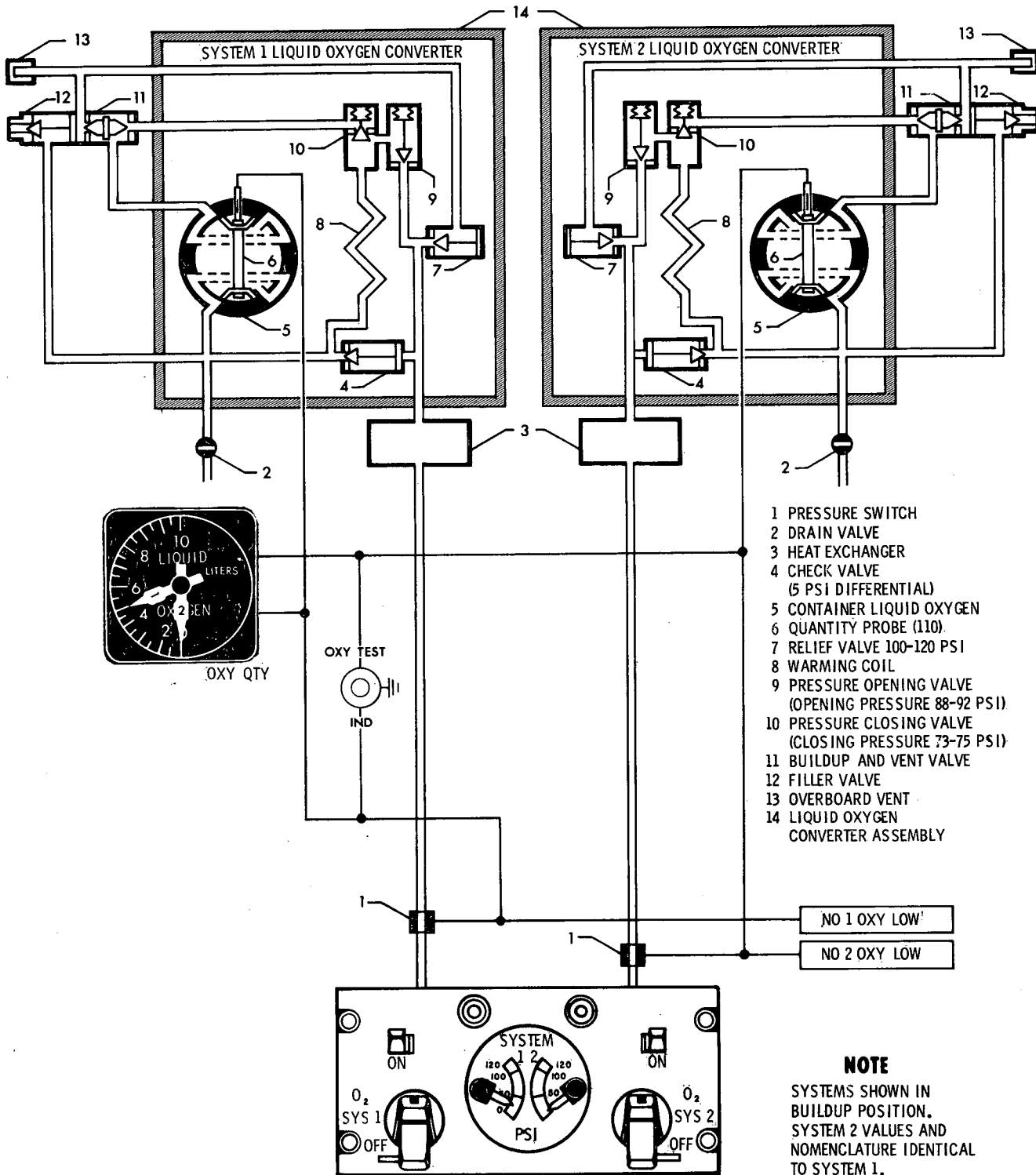


Figure 1-34

F200-55(b)

OXYGEN SYSTEM AND PERSONAL EQUIPMENT

The aircraft is equipped with dual liquid oxygen systems. Two liquid oxygen converters located in the right side of the nose-wheel well have a capacity of ten liters (2.6 gallons) each. The liquid oxygen flows, by gravity, into the pressure buildup coil and vaporizes because of exposure to ambient temperature surrounding the coils. The gas flows through the pressure closing portion of the pressure control valve and the buildup and gas ports of the fill valve and then back into the top of the container where it collects and develops into a higher pressure. This cycle continues until the system operating pressure is reached (80 ± 2 psi) at which time the pressure closing valve closes and stops the flow of liquid oxygen through the pressure buildup coil. The liquid oxygen will now flow through the check valve and out the converter supply port to the aircraft heat exchanger. During periods of shut down system pressure will continue to rise because of normal liquid boil off. The increase in pressure is sensed at the pressure opening valve. At 90 ± 2 psi this valve opens dumping the gas back into the converter. The pressure will continue to slowly rise, due to boil off, until it reaches relief valve opening pressure of 100 to 120 psi. The excess pressure is vented overboard through the relief valve. Two ON-OFF levers for the two systems are located on the oxygen control installed on the left console. The needles on the pressure gage will fluctuate, indicating oxygen flow when the pilot inhales. Liquid oxygen is warmed and converted to gas for breathing by passing through a heat exchanger which consists of additional length of tubing in the supply line. The low pressure gage on the oxygen control panel indicates a normal pressure of 50-100 psi.

Liquid Oxygen Quantity Gage

The liquid oxygen quantity gage is located on the left side of the instrument panel. It is calibrated in 1/2 liter increments from 0 to 10. The quantity gage is a double needle type and indicates the quantity of

liquid oxygen remaining in the No. 1 or No. 2 systems. When visible, a red OFF indicator at the bottom of the gage indicates the gage is not receiving power from the No. 1 inverter.

Indicator Test Switch

A red test button labeled IND TEST is located on the left side of the instrument panel. When this button is pressed the oxygen quantity gage needles will reduce indications. As the oxygen needles approach the 1 liter mark the OXY LOW warning light will illuminate. When the button is released the gage needles will resume their original position. The CIT and spike and forward bypass position indicators are also tested by this button.

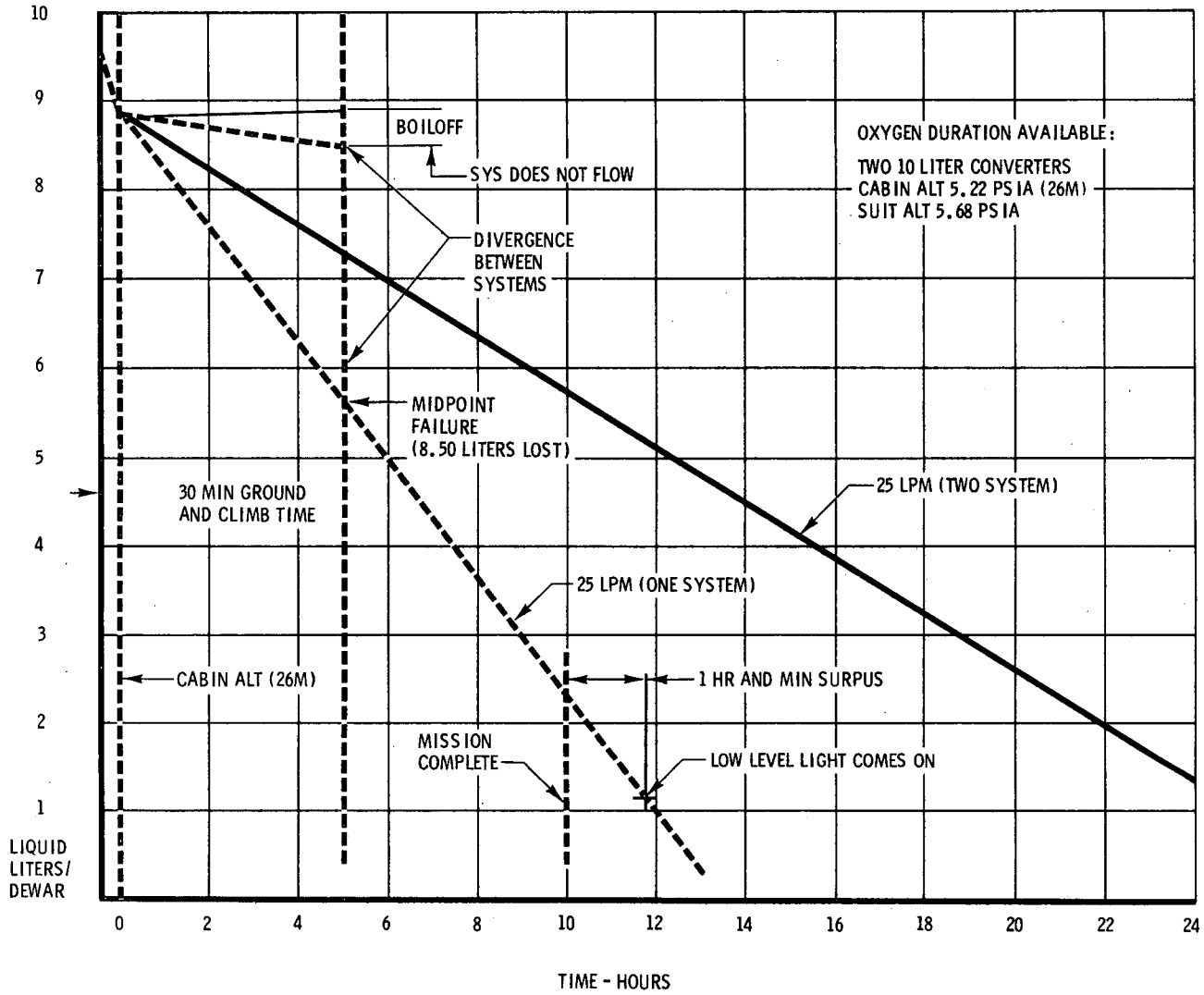
Oxygen Low Indicating Lights

Two oxygen low warning lights are located on the pilot's annunciator panel. The lights are labeled NO. 1 OXY LOW and NO. 2 OXY LOW. Each light will illuminate when oxygen pressure drops to 58 ± 3 psi or when 1 liter or less remains in the system.

EMERGENCY OXYGEN SYSTEM

Two independent emergency oxygen systems are installed in the pilot's parachute pack. Each system consists of a 45 cubic inch, 2100 psi cylinder. The systems will supply oxygen simultaneously during bailout and when the aircraft oxygen systems fail. An oxygen line is routed around each side of the pilot's waist and connects to the suit controller valve. Emergency oxygen flow pressure is slightly lower than aircraft system pressure. Oxygen duration of each emergency system is approximately 15 minutes.

OXYGEN DURATION CHART



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F200-84

Figure 1-35

The emergency oxygen system is actuated either manually by pulling the conventional green apple, or automatically by the upward motion of the seat during ejection. The emergency oxygen system should be actuated if the aircraft systems are not delivering the desired amount of oxygen or hypoxia or noxious fumes are suspected.

FULL PRESSURE SUIT

A full pressure suit is provided which is capable of furnishing the pilot with a safe environment regardless of pressure conditions in the cockpit. The suit consists of four layers, ventilation manifold, bladder, link net, and heat-reflective outer garment. The ventilation manifold layer allows vent air to circulate between the pilot's underwear and the bladder layer. The bladder provides an air-tight seal to hold pressurized air in the suit. The link net is a mesh which holds suit configuration in conformance with the pilot's body. The outer layer of heat-reflecting cloth provides some protection from a hot environment. Air pressure to the suit is regulated by a suit controller valve, located on the front of the suit just above the waist.

Pressure Suit Ventilation Air

Air for suit ventilation is provided by the cockpit air-conditioning system. Temperature of the ventilation air cannot be varied except by changing cockpit inlet air temperature. Ventilation airflow rate may be regulated by a suit flow control valve installed at the hose connection point on the suit. Ventilation air and exhaled breathing air are exhausted from the suit.

Suit Ventilation Boost Valve Lever

The suit ventilation boost valve lever, labeled SUIT VENTIL BOOST, is located on the left console. The lever is marked NORMAL (aft) and EMERG (forward). Operating the lever positions a butterfly valve in the cockpit air-conditioning air supply line in such a way as to vary the pressure of the air available to the suit system. Increased pressure results in more air to the suit. Moving the lever toward EMERG position progressively results in more pressure to the suit system by constricting the air-conditioning airflow to the cockpit; in the NORMAL position (used when engine rpm is high) the cockpit air-conditioning line requires no constriction to provide sufficient airflow to the suit. At IDLE engine rpm the ventilation boost valve lever must be kept at 2/3 of the way from NORMAL to EMERG in order to provide sufficient air for conditioning the suit and cooling the INS platform and inverters in the A/C bay. During takeoff and normal flight the valve lever is kept in the NORMAL position. If the pilot suffers discomfort, such as might happen with a gradual climb to an extreme altitude or during low-rpm descents, the valve lever is gradually moved toward the EMERG position until a comfortable pressure and ventilation condition is attained. The valve lever should not be moved toward EMERG more than necessary to provide pilot comfort; excessive suit system pressure will unduly reduce the available refrigeration.

Suit Controller Valve

All four aircraft and emergency oxygen system lines enter the controller valve at the front waist of the pressure suit. The controller valve contains a sensor that programs airflow and oxygen to keep internal

SECTION I

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suit pressure at 3.5 psi (equivalent to pressure at 35,000 ft) in the event of cockpit depressurization. A press-to-test button for each oxygen system is installed on the controller valve, which allows the pilot to check suit inflation.

Face Plate Heat Switch

A face plate heat switch is installed on the right console of the cockpit. The switch has four positions; OFF, LOW, MED and HIGH. Heat may be regulated to defog the face plate as required. Defogging is accomplished by the combination of face plate heat and oxygen flow. The face plate heater circuit is powered by the essential dc bus.

HELMET

The helmet head area is divided into two separate sections by a rubberized cloth face seal. The front area between the face and the face seal receives oxygen from either the aircraft or emergency oxygen system through regulators built into the helmet. Oxygen flows across the face plate from the inhalation valves inside the helmet and accomplishes some face plate defogging before it is inhaled. The rear area receives vent air for helmet interior temperature regulation. The face seal is not positive; however, the pressure of the oxygen in the front area is slightly higher to prevent vent air from leaking forward. An external crank on the helmet is provided for head band adjustment. Buttons on each side of the helmet, when actuated, will lower the face plate and visor. The face plate is opened by moving the buttons and dumping the pressure, allowing the face plate to be rotated upward. If the aircraft or emergency oxygen supply to the helmet is interrupted or exhausted, the regulators in the helmet sense the drop in pressure and the face plate seal deflates, allowing ambient air to enter the helmet so the pilot will not suffocate.

GLOVES

Leather gloves fasten onto the suit at the wrist rings. The inner liner of the glove is similar to the suit inner liner and will retain pressure.

BOOTS

The sock or boot liner fastens onto the suit at the thigh by means of a zipper. The boots are made of white leather for heat reflection and fit snugly over the socks. A spur that fastens to the seat is attached to each boot.

OXYGEN MASK AND REGULATOR

When permitted by appropriate regulations a substitute oxygen mask assembly may be used in place of a pressure suit for flights at low or intermediate altitudes. The assembly consists of a specially designed oxygen mask and F2700 oxygen regulator, anti-suffocation valve and two oxygen personal leads with connectors for both aircraft and emergency oxygen systems. In the event that the regulator should malfunction or the oxygen supply is exhausted, an anti-suffocation valve installed between the regulator and the mask will sense the drop in oxygen pressure and allow ambient air to enter the mask to prevent suffocation.

SURVIVAL KIT

A reinforced fiberglass survival kit container fits into the seat bucket and attaches to the parachute by snap attachments on each side. A door on the top provides access to the survival items stored inside. The kit contains standard survival items such as radio, flares, mirror, whistle, knife, matches, rations, water, compass and first aid kit. Various additional items depending on the terrain and season may be

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provided. The kit is packed in a water proof bag attached to a 20 foot retention lanyard. If an overwater flight is anticipated, a life raft may be stowed on top of the plastic bag and attached to the lanyard. During ejection the life raft inflating device is armed. Following ejection, the survival kit release handle should be pulled before reaching the ground. This action separates the survival gear from the pilot and inflates the life raft. The survival gear and life raft remain attached to the parachute harness by the retention lanyard. During a rapid abandonment of the aircraft on the ground, the survival kit release handle may be used to free the pilot of the survival kit (including the lanyard) without inflating the life raft.

PARACHUTE

A special parachute with a 35 foot canopy is used. The large canopy provides a normal descent rate with the bulky personal equipment required for high altitude flight. A small diameter, ribbon type stabilizing drogue chute is also provided. Above 17,000 feet altitude, the drogue chute is deployed first in order to stabilize free fall of the pilot. The drogue is automatically jettisoned at 15,000 (+ 400) feet after an aneroid controlled opener deploys the main chute. Below 16,200 feet the main chute only deploys immediately. A manual T-handle is also available for opening the main chute. The chute pack is equipped with conventional quick release buckles. The emergency oxygen bottles are located between the chute canopy and the pilot's back. A combination hand squeezed bulb and manually operated pressure relief valve located adjacent to the suit controller is used to adjust cushion pressure as desired. A red knob located on the left harness strap is connected to the parachute timer arming cable and is used to actuate the timer when bailout is made.

WINDSHIELD

The windshield is composed of two glass assemblies secured and sealed in a V-shaped titanium frame. The glass surfaces are coated with low reflective magnesium fluoride. A collapsible vision splitter is also installed on the windshield center line to minimize reflections.

DEFOG SYSTEM

The windshield defog system delivers hot air from both right and left air systems through check valves to defog the windshield and canopy. A plastic V-shaped air duct runs along the lower edge of the windshield. Hot defog air is supplied through this duct when selected by a switch that is located on the upper left console. The air is directed to the windshield through a series of holes on the upper surface of the duct. Holes are also provided at the aft ends of the duct to direct air toward the canopy glass.

Defog Switch

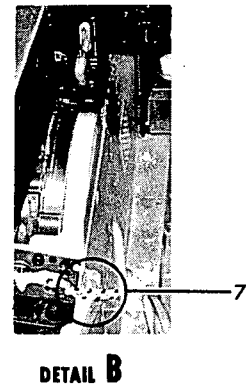
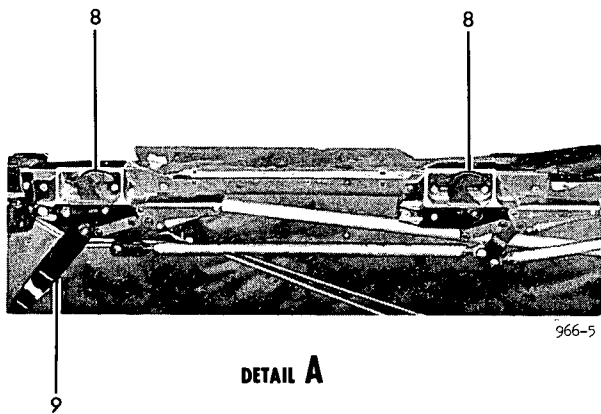
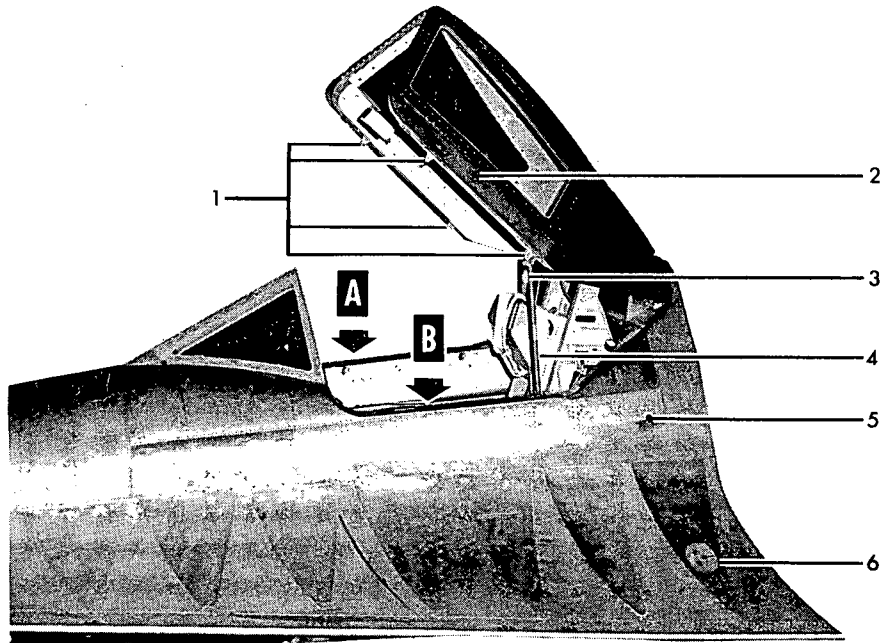
A three position defog switch is located at the forward end of the upper left console. When held in the momentary DEFOG INCREASE (forward) position the motor driven defog valve will open. Time of travel to full open is approximately 7 to 13 seconds. In the HOLD (center) position the valve will stop at any desired partial open position; in the OFF position the valve will completely close. The circuit is powered by the essential dc bus.

LEFT WINDSHIELD HOT AIR DEICING SYSTEM

Hot air is ducted from the L & R pressurization supply downstream of the fuel air heat exchanger and upstream of the pressure regulator and air cycle refrigerator,

SECTION I

CANOPY AND CONTROLS



- | | |
|-----------------------------------|-----------------------------------|
| 1 CANOPY LATCH ROLLER BRACKETS | 6 CANOPY EXTERNAL JETTISON HANDLE |
| 2 CANOPY LIFTING HOLE | 7 CANOPY INTERNAL JETTISON HANDLE |
| 3 CANOPY PROP ASSEMBLY AND UPLOCK | 8 CANOPY LATCH HOOKS |
| 4 CANOPY PROP (GROUND HANDLING) | 9 CANOPY LATCH HANDLE |
| 5 CANOPY EXTERNAL LATCH CONTROL | |

Figure 1-36

FMA12-36

to a series of orifices located on the left side of the outside center windshield support. The system includes left and right solenoid shutoff valves controlled by a switch in the cockpit. Power is furnished by the essential dc bus.

CAUTION

Do not apply rain repellent on a dry windshield as prolonged obscuration may result.

Windshield Deice Switch and Indicator Light

The 3-position windshield deice switch is located on the upper left instrument panel. In the OFF (right) position the shutoff valves are closed and no deicing air is supplied. In the R ON (center) position the hot air is furnished by the right pressurization system and 1/2 flow is available for deicing. In the LR ON (left) position both L & R shutoff valves are opened and full flow is available to the windshield orifices. Power for the switch and lights is furnished by the dc essential bus.

NOTE

- . A considerable amount of air is used when operating the deicing system in the L/R ON position. This may reduce the cockpit and Q-bay air supply when operating in the lower ranges of engine rpm.
- . The deicer indicator light, located above the switch, will be illuminated at any time the deice switch is not in the OFF position.

WINDSHIELD RAIN REMOVAL SYSTEM

A rain removal system is provided for clearing the windshield when operating the aircraft in rain. It has a tank that is pressurized by air from the windshield deicer system and the tank is connected to spray tubes located on each side of the windshield center divider. A pushbutton switch, located on the upper instrument panel, is used to spray the rain removal fluid onto the windshield. Power is furnished by the essential dc bus.

CANOPY

The canopy consists of two high temperature resistant glass windows secured within a reinforced titanium frame which is hinged at the aft end of two hinge pins. Operation of the canopy is completely manual. Small holes in each side of the canopy are provided as lifting points from the outside. No handles are provided on the inside of the canopy for moving it up or down. A prop assembly locks the canopy in the full open position. The canopy is secured in the closed and locked position by a four hook interconnected latching mechanism. A nitrogen boost counterbalancing system is provided to aid in the manual opening and closing of the canopy. This nitrogen is also used to force water into the map case when the destruct system is actuated.

NOTE

Actuation of the destruct system tends to deplete the nitrogen boost counterbalance system and increase the manual force needed to open the canopy. Canopy jettisoning may be necessary for rapid egress.

An internal latching handle is installed below the right canopy sill, allowing the canopy to be latched from the inside. An external fitting located on the left side of the aircraft can be used to operate the latches from the outside.

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CAUTION

The canopy should be opened or closed only when the aircraft is completely stopped. Maximum taxi speed with the canopy open is 40 knots. Gusts or severe wind condition should be considered as a portion of the 40 knot limit.

Canopy Latch Handle

A canopy latch handle is located under the right sill in the cockpit and rotates forward to lock. The sill trim is cutout to expose the action of the locking lugs and pins as the handle is rotated forward. A cam over center action allows the handle to remain only in the latched or unlatched position. No canopy unsafe warning light is provided.

Canopy External Latch Control

A flush mounted external latch fitting is located on the left side of the aircraft and permits the canopy to be opened from the outside. The fitting accepts a 1/2 inch square bar extension. Once the canopy is unlocked, it may be raised manually until the prop locks it in the open position.

Canopy External Jettison Handle

The canopy external jettison handle, located beneath an access panel on top of the left chine, permits ground rescue personnel to jettison the canopy. Sufficient cable length is provided to allow the operator to stand clear of the fuselage during the jettisoning procedure.

Canopy Internal Jettison Handle

A canopy jettison T-handle is located on the left console wall adjacent to the pilot's leg. The handle can be used to jettison the canopy without initiating the seat ejection system. The handle is held in the stowed position by a lockwire and a ground safety

pin. Storage for the canopy jettison and seat safety pins is provided at the forward end of the upper right console. Cable travel is approximately six inches.

CANOPY SEAL

An inflatable rubber seal is installed in the edge of the canopy frame. The seal seats against the mating surfaces of the canopy sill and windshield to provide sealing for cockpit pressurization. The canopy seal pressurization lever above the forward right console operates the seal inflation valve. A nose hatch seal shutoff lever is also provided to prevent deflation of the canopy seal in the event of nose hatch seal leakage.

CANOPY JETTISON SEQUENCE

The canopy jettison system is designed to unlatch and jettison the canopy from the aircraft by means of explosive initiators and thrusters. The system consists of two initiators which are independently actuated by either the ejection seat D-ring or the canopy jettison handle, a canopy unlatch thruster, a canopy removal thruster, a canopy seal hose cutter, cable linkage and gas pressure lines. Either the D-ring initiator or the canopy initiator or the canopy initiator will fire the unlatch thruster which unlocks the canopy. This thruster then activates the canopy seal hose cutter and fires the canopy removal thruster which jettisons the canopy. Whenever the canopy is jettisoned by use of the canopy jettison handle, the canopy jettison initiator gas pressure positions a seat jettison safety valve to prevent initiating the seat ejection sequence until the D-ring is pulled. Pulling the D-ring jettisons the canopy as the initial step in the ejection sequence.

REAR VIEW PERISCOPE

A manually extended rear view periscope is mounted in the top of the canopy to enable the pilot to see the engine nacelles and rear fuselage and rudder area. The periscope,

normally is locked in a fully retracted position. It is moved by using the white nylon pad, mounted on the aft side of the viewing tube, as a handle. Pushing the handle to the left unlocks the tube, allowing the periscope to be extended. Then, pushing the tube upward to a spring-detented position makes the rear view available. Cockpit pressure tends to assist extension, and resists retraction. The diameter of the instantaneous cone of view is approximately 10° ; however, head movement extends the viewing cone to approximately 30° total angle. When extended, the periscope can be rotated horizontally to move the center of the viewing arc up to 10° from the aft centerline. The de-magnification ratio of the lens system is 1 to 0.5.

EJECTION SEAT

The ejection seat system utilizes an upward catapult and rocket thrust to provide minimum risk ejection capability at ground level when airspeed is at least 65 KLAS. The seat incorporates an ejection ring, headrest, knee guards, automatic foot retractors, automatic foot retention separation, a pilot-seat separation device, shoulder harness, inertia reel lock assembly, and an automatic opening seat belt. A speed sensor mounted on the fuselage behind the seat automatically selects one of two seat separation delays, depending upon airspeed at ejection. (Refer to Ejection Sequence this section.) Quick disconnect fittings installed on the seat rails and the floor of the aircraft permit disconnection of the oxygen, ventilated suit and electrical lines.

Seat Vertical Adjustment Switch

The seat may be adjusted vertically by means of an electric actuator mounted on the lower end of the catapult. The three-position switch is located on the right side of the seat bucket. The seat moves in the direction the switch is moved. Power for seat adjustment is furnished by the essential dc bus.

Shoulder Harness Inertia Reel Lock Lever

A shoulder harness inertia reel lock lever installed on the left side of the seat bucket is provided for locking and unlocking the shoulder harness. The lever has two positions, LOCK and UNLOCK. Each position is spring loaded to hold the lever in the selected position. An inertia reel located on the back of the seat will maintain a constant tension on the shoulder straps to keep them from becoming slack during backward movement. The reel also incorporates a locking mechanism which will lock the shoulder harness when a 2 to 3 g force has been exerted in a forward direction. When the reel is locked in this manner, it will remain locked until the lever is moved to the LOCK position and then returned to the UNLOCK position.

Ejection (D) Ring

An ejection ring, located on the front of the seat bucket, is the primary control for ejection. An ejection safety pin is installed in the ejection ring housing bracket.

Ejection T-Handle

The aircraft are equipped with a backup secondary seat ejection system. The T-handle for this seat ejection system is unlocked and made accessible only by first pulling the ejection D-ring.

WARNING

The ejection seat must not be fired by pulling the T-handle while the canopy is still in place. The pilot can not eject through the metal canopy.

When the secondary ejection T-handle is pulled a separate initiator fires the seat catapult and seat separation and belt opening initiator.

SECTION I

EJECTION SEAT

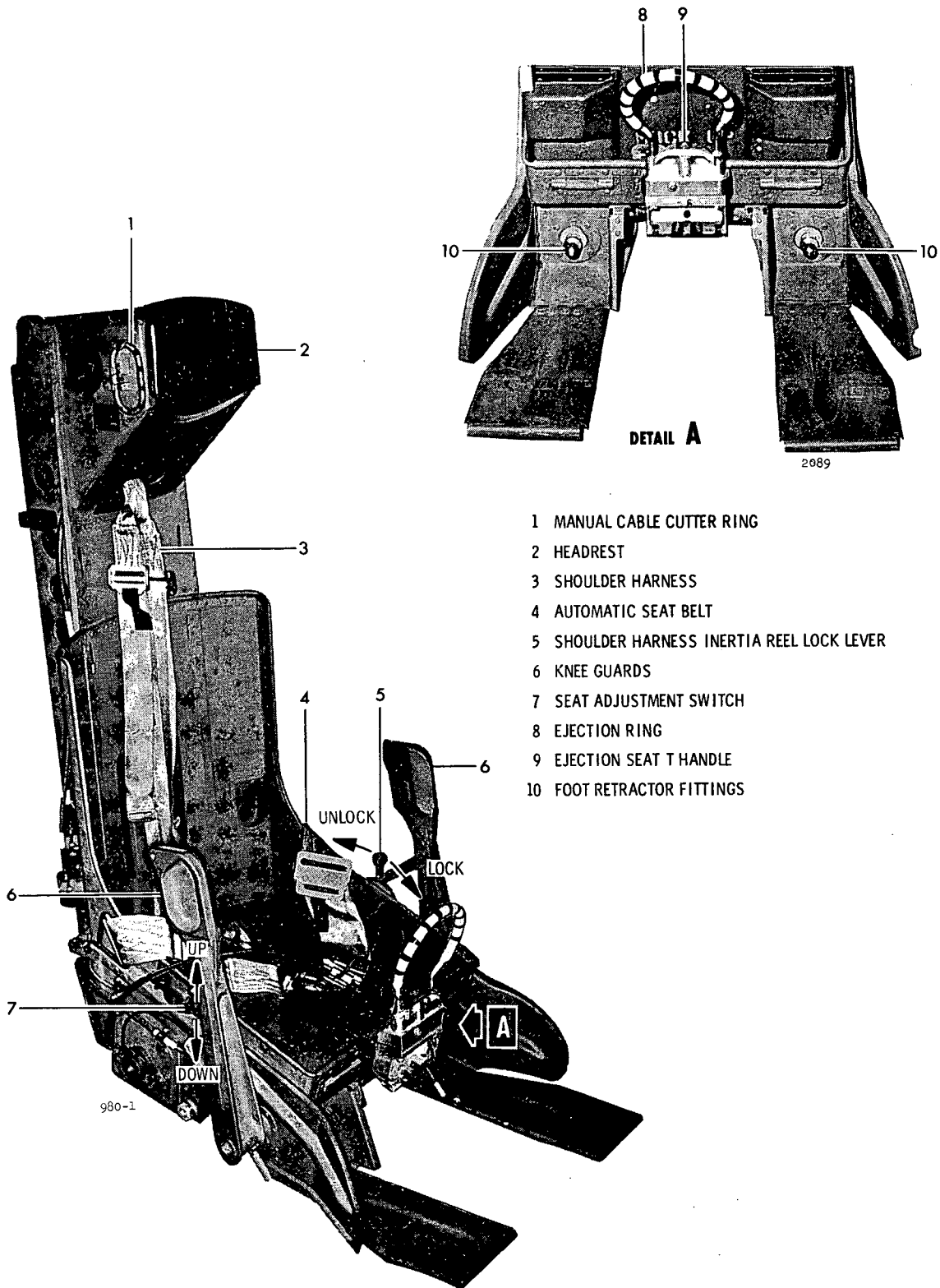


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Foot Spurs

Foot spurs, attached to the pilot's shoes, are attached to the ejection seat by cables. Normal foot movement is in no way restricted since the cables, under a slight spring tension, reel in and out freely. When the ejection ring is pulled, the knee guards rotate from their stowed position, the cables to the foot spurs are reeled in and the pilot's feet are retracted into the foot rests. The foot cables are automatically severed by a set of cutters as part of the ejection sequence.

Manual Cable Cutter Ring

The ejection seat incorporates an emergency means for cutting the foot retractor cables. A D-ring, located to the right of the seat headrest, will actuate the cable cutters initiator if the automatic cable cutter system fails or rapid abandonment of the aircraft is required on the ground.

PILOT-SEAT SEPARATION SYSTEM

The ejection seat is provided with a pilot-seat separation system which operates in conjunction with the automatic seat belt release system. A windup reel is mounted behind the headrest, and a single nylon web is routed from the reel to halfway down the forward face of the seat back. From this point two separate nylon straps continue down, pass under the survival kit, and are secured to the forward seat bucket lip. After ejection, as the seat belt is released, an initiator actuates the windup reel which winds the webbing onto a cross-shaft, pulls the webbing taut, and causes the pilot to be separated from the seat with a sling shot action.

AUTOMATIC SEAT BELT

The ejection seat is equipped with an automatic opening seat belt which facilitates pilot separation from the seat following ejection. Belt opening is accomplished automatically as part of the ejection sequence and requires no additional effort on the part of the pilot.

SEAT BELT-PARACHUTE ATTACHMENT

If the pilot is wearing an automatic opening aneroid type parachute, the parachute lanyard anchor from the parachute aneroid must be attached to the swivel link. As the pilot separates from the seat, the lanyard, which is anchored to the belt, serves as a static line to arm the parachute aneroid. The parachute aneroid preset altitude is approximately 15,000 feet.

EJECTION SEQUENCE

Pulling the D-ring is normally the only action required to initiate pilot ejection and results in firing both the canopy jettison and ejection seat systems. All resultant actions will occur automatically and in a specific sequence as explained below.

The D-ring cable fires the ejection sequence initiator, actuating the canopy jettison system and the leg guard thruster. The leg guard thruster rotates the knee guards, retracts the pilot's feet, activates the cable cutter backup initiator and locks the shoulder harness. Movement of the canopy jettison thruster (final step in canopy jettison sequence) actuates an initiator which fires a 0.3 second delay catapult initiator and arms the speed sensor. The 0.3 second delay assures complete canopy separation prior to seat ejection. Gas pressure from the catapult initiator fires the rocket-catapult,

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the 4-second seat separation delay initiator, and enters the speed sensor. If airspeed is below 295 KIAS, the gas pressure passes through the speed sensor and fires the 1.0 second delay seat separation initiator. If airspeed is above 302 KIAS, the pressure is blocked by the speed sensor.

Initial seat movement upward on the rails disconnects normal oxygen, ventilated suit and electrical lines, and activates the emergency oxygen supply. Between 295 and 302 KIAS either the 1 or 4 second delay may be experienced because of the speed sensor tolerance.

Either the 1.0 second delay initiator (below 295 KIAS) or the 4-second delay initiator (above 302 KIAS) actuates the cable cutters, releases the pilot's feet, opens the seat belt and fires the seat separation system.

A static line attached to the seat belt is pulled as the pilot separates from the seat and activates the automatic parachute sequence.

If the normal D-ring ejection sequence was not accomplished, the canopy must be jettisoned either by use of the canopy jettison system or manually. Pulling the T-handle initiates the secondary seat ejection sequence.

CAUTION

The T-handle backup ejection sequence does not rotate the knee guards nor retract the foot cables. Seat separation delay time will be 4 seconds regardless of airspeed.