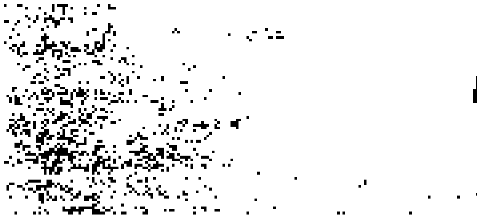

ENGINES & FUEL

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SECTION II
ENGINES & FUEL

ENGINES

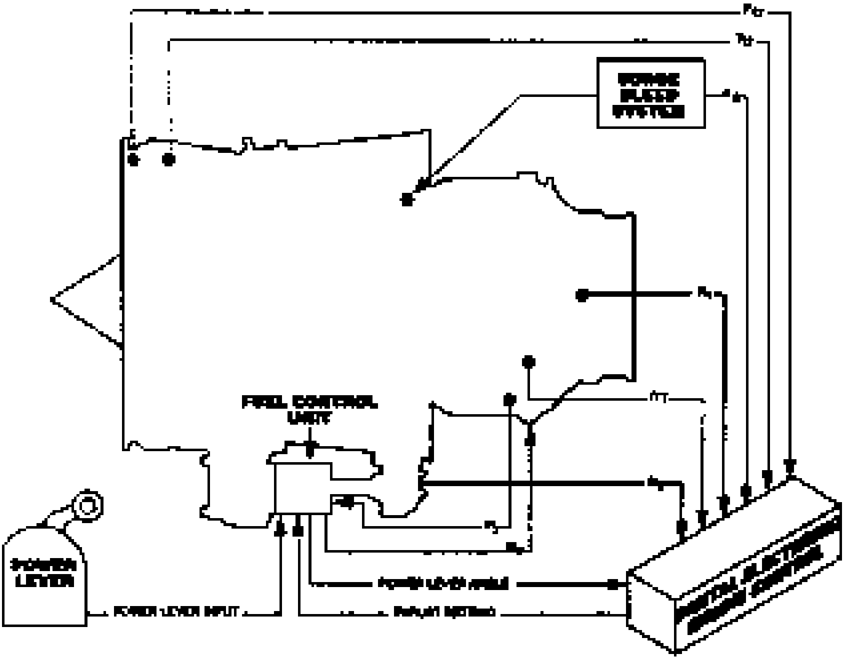
The aircraft is powered by two TFE731-2-3B turbofan engines manufactured by Garrett Division of Allied Signal. These engines are two-spool, geared transonic-stage, front-fan, jet-propulsion engines. Each engine is rated at 3500 pounds thrust at sea level.

A spinner and an axial-flow fan are located at the forward end of the engine and are gear driven by the low-pressure rotor. The fan gearbox output-to-input speed ratio is 0.556. The low-pressure rotor consists of a four-stage low-pressure axial compressor and a three-stage low-pressure axial turbine, mounted on a common shaft. The high-pressure rotor consists of a single-stage high-pressure centrifugal compressor and a single-stage high-pressure axial turbine, mounted on a common shaft. The high-pressure rotor drives the accessory gearbox through a transfer gearbox. The rotor shafts are concentric, so that the low-pressure rotor shaft passes through the high-pressure rotor shaft.

An annular duct serves to bypass fan air for direct thrust and also diverts a portion of the fan air to the low-pressure compressor. Air from the low-pressure compressor flows through the high-pressure compressor and is discharged into the annular combustor. Combustion products flow through the high- and low-pressure turbines and are discharged axially through the exhaust duct to provide additional thrust.

ENGINE FUEL AND CONTROL SYSTEM

The engine fuel and control system pressurizes fuel routed to the engine from the aircraft fuel system, meters fuel flow, and delivers atomized fuel to the combustion section of the engine. The system also supplies high-pressure motive-flow fuel to the aircraft fuel system for jet pump operation. The major components of the system are the thrust levers, the engine-driven fuel pump, the fuel control unit, the digital electronic engine control (DEEC), surge bleed control, fuel heater (optional) and the engine synchronizer.



A_B — AREA BLEED
N_1 — LOW PRESSURE ROTOR (FAN) SPEED
N_2 — HIGH PRESSURE ROTOR (TURBINE) SPEED
P_3 — COMPRESSOR DISCHARGE PRESSURE
P_{T2} — ENGINE INLET TOTAL PRESSURE
T_{T2} — ENGINE INLET TOTAL TEMPERATURE
ITT — INTERSTAGE TURBINE TEMPERATURE
W_F — FUEL FLOW

FUEL CONTROL LOGIC DIAGRAM
Figure 2-1

THRUST LEVERS

Two thrust levers, located on the upper portion of the pedestal, are operated in a conventional manner with the full forward position being maximum power. Stops at the IDLE position prevent inadvertent reduction of the thrust levers to CUTOFF. The IDLE stops can be released by lifting a finger lift on the outboard side of each thrust lever. A "floating" control, consisting of ball bearings and races enclosed in a flexible casing, connects each thrust lever to the corresponding engine's hydro-mechanical fuel control. A flight director go-around button is installed in the left thrust lever handle. A landing gear/cabin altitude warning horn mute button is installed in the right thrust lever handle. If thrust reversers are installed a thrust reverser control lever is mounted piggyback fashion on each thrust lever. Refer to ENGINE THRUST REVERSERS for a functional description of the thrust reverser levers.

ENGINE-DRIVEN FUEL PUMP

The engine-driven fuel pump provides high-pressure fuel to the engine fuel control system as well as motive-flow fuel for operation of the aircraft jet pumps. The pump consists of a low-pressure pump element, high-pressure pump element, relief valve, filter, and motive-flow provisions. *On aircraft equipped with fuel heaters*, the fuel pump also incorporates an anti-icing valve which monitors fuel temperature downstream of the filter element and mixes warm fuel from the fuel heater with the low-pressure pump element discharge flow to prevent filter element icing. In the event the fuel pressure drop across the fuel filter reaches a preset level, due to clogging, the filter bypass valve will open and fuel will bypass the filter. *On aircraft not equipped with fuel heaters*, a red bypass indicator button will pop out. The fuel bypass indicator button is visible through a small, circular, push-to-open door on the engine nacelle.

FUEL CONTROL UNIT

The fuel control unit is mounted on the fuel pump and contains the hydro-mechanical fuel metering section, thrust lever input and position potentiometer, shutoff valve, and a mechanical governor. The mechanical governor functions as an overspeed governor for the high-pressure rotor. In addition, the mechanical governor provides manual control when the digital electronic engine control (DEEC) is deactivated. When activated, the DEEC controls fuel scheduling by means of a torque motor located within the fuel control. The torque motor controls the hydro-mechanical metering section of the fuel control.

DIGITAL ELECTRONIC ENGINE CONTROL (DEEC)

An digital electronic engine control (DEEC), located in the tailcone, is provided for each engine. The DEEC is basically an engine speed governor, with provisions for fuel limits during acceleration and deceleration. The N1 DEEC (*Aircraft 31-183 and subsequent*) governs based on low-pressure rotor speed. The N2 DEEC (*Aircraft 31-035 thru 31-182*) governs based on high-pressure rotor speed. Each DEEC performs governing, limiting, and fuel scheduling functions for engine start and continuous operation. Input parameters utilized by both DEECs are: inlet pressure (PT2), inlet temperature (TT2), interstage turbine temperature (ITT), low-pressure rotor speed (N1), high-pressure rotor speed (N2), and power lever angle (PLA). The power lever angle is sensed by the DEEC from a potentiometer in the fuel control unit. An electrically heated mast in the inlet duct houses the inlet temperature sensor and inlet pressure tap for the DEEC. Interstage turbine temperature, low-pressure rotor speed and high-pressure rotor speed are sensed by the same dual-output sensors which drive the respective cockpit indicators and are discussed in ENGINE INDICATING. Output signals from the DEEC are directed to a torque motor in the fuel control unit and to the control solenoids of the surge bleed control. The N1 DEEC utilizes an additional pneumatic static pressure signal (PSO) to allow calculation of airspeed (represented by Mach number). During engine start, the DEEC provides automatic fuel enrichment up to 200° C ITT. Each DEEC incorporates a built in test capability, which detects, stores and annunciates faults. The N1 DEEC incorporates an Engine Condition Trend Monitor (ECTM) function to provide operators with engine life usage and condition trend information. Fault information and ECTM data is accessed using a personal computer and application software through a front panel mounted test connector, which supports an RS-422 serial communications data bus. The N2 DEEC's built in test capability displays fault codes in a window located on the DEEC's front panel. A function select switch and calibration switch, located on the N2 DEEC's front panel, allow for the adjustment of several operating parameters, including fuel specific gravity adjustment. The fuel computers operate on 28 VDC supplied through the 5-amp L and R FUEL CMPTR circuit breakers on the pilot's and copilot's circuit breaker panels respectively.

FUEL CMPTR SWITCHES

The DEECs are controlled by two switches on the pilot's switch panel labeled FUEL CMPTR L-MAN-OFF and R-MAN-OFF. Normally, the switches are left in the On position.

When the switch is OFF, the DEEC does not perform any engine control functions. When the switch is On, the DEEC will automatically perform the engine control functions listed above. When the switch is in the MAN position, the DEEC will revert to the manual mode and not all engine control functions will be available. The DEEC does provide engine overspeed protection during manual mode operations.

FUEL CMPTR LIGHTS

Illumination of either amber L or R FUEL CMPTR light on the annunciator panel indicates a failure in or loss of power to the corresponding DEEC. Circuits within the DEEC continuously monitor power supply, computer circuits, and inputs to the computer. A loss of input power or a computer malfunction will cause the monitor circuits to illuminate the applicable FUEL CMPTR light and switch the DEEC to manual mode. With the DEEC in the manual mode, cycling the applicable FUEL CMPTR switch will reset the DEEC if the fault clears. If operating in the manual mode, the DEEC will provide overspeed protection but will not provide automatic mode functions. Also, the lights will be illuminated whenever power is on the aircraft and the FUEL CMPTR switches are in the OFF or MAN position.

SPR SWITCH

The SPR (starting pressure regulator) switch is located on the pilot's switch panel below the FUEL CMPTR switches. The switch has three positions: L-Off-R. When the switch is positioned to L or R, the switch commands the applicable DEEC to provide increased fuel scheduling beyond the normal fuel enrichment schedule for engine start under cold ambient conditions. Use of SPR is recommended for ground starts when ambient temperature is below 0°F (-18°C) and for airstarts. SPR should not be used at ambient temperatures above 0°F (-18°C) and never used at any time other than engine start. SPR is an DEEC function and is inoperative with the DEEC off or in manual mode.

SURGE BLEED CONTROL

A surge bleed control system for each engine is installed to prevent low-pressure compressor surge. Each system consists of two solenoid control valves and a surge bleed valve. During normal operation, surge bleed valve position is controlled by the DEEC via the solenoid control valves. In the event a FUEL CMPTR light illuminates because of a failure or the computer being switched to MANUAL or OFF, the surge bleed valve will go to the 1/3-open position.

FUEL HEATER (OPTIONAL)

Each engine may be equipped with a fuel heater which warms the fuel to prevent ice formation on the engine fuel filter element. The fuel heater is of a liquid-to-liquid design utilizing the engine lubricating oil as a source of heat to warm the fuel. An anti-icing valve, in the engine-driven fuel pump, monitors fuel temperature downstream of the filter element and controls the flow of fuel into the oil-to-fuel heat exchanger. The warm fuel exits the heat exchanger and mixes with the fuel entering the fuel filter element. Operation of the fuel heater is automatic and no crew action is required.

ENGINE SYNCHRONIZER

The engine synchronizer system consists of the R ENG indicator, two ENG SYNC switches, an amber ENG SYNC light, and an engine synchronizer control box. During flight, the engine synchronizer, if selected, will maintain the right engine Fan Speed (N1) or Turbine Speed (N2) in sync with left engine Fan Speed (N1) or Turbine Speed (N2). The engine synchronizer may be used whenever both fuel computers are operating. However, the engine synchronizer must not be used during takeoff, landing, or single-engine operations. Engine synchronizer authority at takeoff and idle power settings is zero. Engine synchronizer authority at all other power settings is limited to approximately 2.5%; therefore, the engines must be manually synchronized to within approximately 2.5% before selecting SYNC. Electrical power for the engine synchronizer is 28 VDC supplied through the 5-amp R FUEL CMPTR circuit breaker on the copilot's circuit breaker panel.

ENG SYNC SWITCHES

Two ENG SYNC switches are installed on the pedestal immediately below the thrust levers. The ENG SYNC control switch is labeled SYNC-OFF and the ENG SYNC selector switch is labeled TURB-FAN. With both FUEL CMPTR switches On, the ENG SYNC control switch will activate the engine synchronizer when moved to SYNC. When SYNC is selected, Fan Speed (N1) or Turbine Speed (N2) synchronization is selected by moving the ENG SYNC selector switch to TURB or FAN as desired. When the engine synchronizer is OFF, the selector switch controls the display parameter of the R ENG indicator.

R ENG INDICATOR

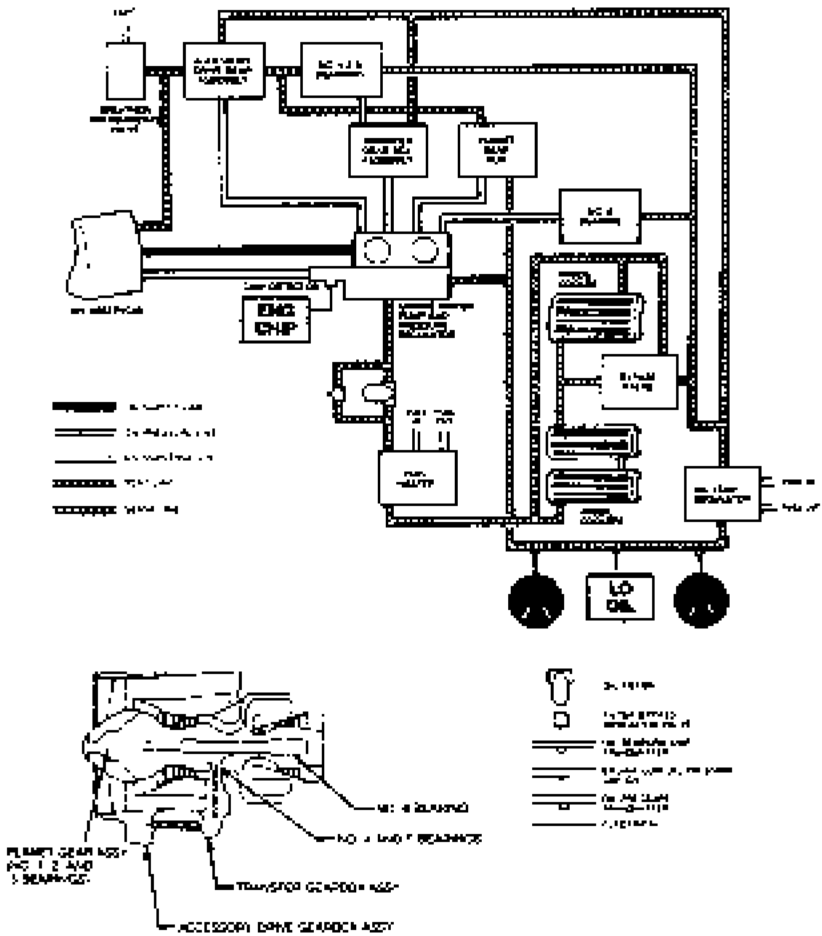
The R ENG indicator, on the instrument panel, provides an indication of right engine synchronization with left engine Fan (N_1) or Turbine (N_2) speed. The indicator indicates fan sync when the ENG SYNC selector switch is in the FAN position and turbine sync when the ENG SYNC selector switch is in the TURB position. The indicator can be used to manually synchronize the engines when the ENG SYNC control switch is OFF or to monitor the engine synchronizer when the ENG SYNC control switch is in SYNC. The R ENG indicator is operative whenever the right fuel computer is operating.

ENG SYNC LIGHT

The amber ENG SYNC light on the glareshield annunciator panel will be illuminated whenever the nose gear is down and the SYNC-OFF switch is in the SYNC position. The light will alert the pilot to deenergize the engine synchronizer for takeoff and landing.

ENGINE OIL SYSTEM

Oil for engine lubrication is drawn from the engine oil tank by the engine-driven oil pressure pump. The oil is output from the pump through a pressure-regulating valve, a filter, a fuel heater (optional), and to the oil-to-air heat exchanger. The exchanger is a three-segment, finned cooler that forms the inner surface of the fan duct. From the oil-to-air heat exchanger, the oil flow is divided so that part of the oil is directed to the accessory and transfer gearboxes and the engine shaft bearings. The remaining oil is diverted to an oil temperature regulator (oil-to-fuel heat exchanger) and then to the fan gearbox. The oil filter assembly incorporates a bypass valve and indicator to indicate when the oil filter is clogged or clogging. In the event the filter becomes clogged, the red bypass indicator button will pop out and the bypass valve will open, allowing oil to bypass the filter. Under cold oil conditions, such as engine start, the bypass indicator button is locked in; however, the bypass valve will still open. The oil bypass indicator button is visible through a small, circular, push-to-open door on the engine nacelle.



ENGINE OIL SYSTEM SCHEMATIC
Figure 2-2

ENGINE IGNITION SYSTEM

Each engine ignition system consists of an IGNITION switch, an amber ignition light, a thrust lever idle position switch, an ignition unit, two shielded cables, two igniter plugs, and associated aircraft wiring. The ignition unit is a solid-state, high-voltage, capacitor-discharge unit which provides a spark rate of 1 to 5 sparks per second at an output of 18,000 to 24,000 volts. The igniter plugs are mounted at six and seven o'clock positions in the turbine plenum. The plugs are operated by separate cables and spark when pulsed by the ignition unit. During the start cycle, the ignition system is automatically energized when the thrust levers are placed in the IDLE position and automatically deenergized at approximately 45% Turbine Speed (N_2). The ignition system may be operated continuously through the corresponding IGNITION switch. During the start cycle, the ignition system is powered by 28 VDC supplied through the 7.5-amp L and R IGN-START circuit breakers on the pilot's and copilot's circuit breaker panels respectively. During continuous operation, the ignition system is powered by 28 VDC supplied through the 7.5-amp L and R IGN circuit breakers on the pilot's and copilot's circuit breaker panels respectively. Continuous operation is available during EMER BUS mode.

IGNITION SWITCHES

The IGNITION switches, located on the pilot's switch panel, are used to obtain continuous engine ignition. The switch controlling the left engine ignition system is labeled L-OFF. The switch controlling the right engine ignition system is labeled R-OFF. When an IGNITION switch is placed in the On (L or R as applicable) position, 28 VDC from the corresponding L or R IGN circuit breaker is directly applied to the corresponding engine ignition unit.

IGNITION LIGHTS

Amber lights above each IGNITION switch are installed to indicate ignition system operation. The corresponding light will be illuminated whenever the associated ignition system is operating either continuously (IGNITION On) or automatically (Start cycle).

ENGINE INDICATING

OIL TEMPERATURE INDICATOR

The OIL TEMP indicator is a circular-scale, dual-reading instrument and is located on the center instrument panel. The instrument face is divided into two semi-circular scales marked from 30°C to 180°C in 10° increments. Each scale has a separate pointer; one for the left engine and one for the right engine. Each pointer is operated by a resistance-type temperature sensor in an oil supply line on the respective engine. Electrical power for system operation is 28 VDC supplied through the 2.0-amp L and R OIL TEMP circuit breakers on the pilot's and copilot's circuit breaker panels respectively. Refer to Airplane Flight Manual for instrument limit markings.

OIL PRESSURE INDICATOR

The OIL PRESS indicator is a circular-scale, dual-reading instrument and is located on the center instrument panel. The instrument face is divided into two semi-circular scales marked from 0 to 75 PSI in 5 PSI increments. Each scale has a separate pointer and is labeled either L or R for the respective engine. Each pointer is operated by an oil pressure transmitter located in an oil supply line on the respective engine. Electrical power for system operation is 26 VAC supplied through the 0.5-amp L and R OIL PRESS circuit breakers located on the pilot's and copilot's circuit breaker panels respectively. Refer to Airplane Flight Manual for instrument limit markings.

OIL PRESSURE LIGHTS

Red L OIL PRESS and R OIL PRESS pressure warning lights are installed in the glareshield annunciator panel. In the event that either engine's oil pressure drops below approximately 25 PSI (172 kPa), a pressure switch located in the oil supply line of the affected engine will illuminate the respective light. Also, the respective light will be illuminated whenever electrical power is on the aircraft and the affected engine is not operating. Electrical power for system operation is 28 VDC supplied through the 7.5-amp WARN LTS circuit breakers on the pilot's and copilot's circuit breaker panels.

ENGINE CHIP LIGHTS

Illumination of either amber L ENG CHIP or R ENG CHIP light on the glareshield annunciator panel indicates the presence of magnetically permeable material in the corresponding engine's oil system. The lights are activated by a magnetic chip detector installed in each engine's oil pump.

FUEL FILTER LIGHT (AIRCRAFT EQUIPPED WITH FUEL HEATERS)

Illumination of the amber FUEL FILTER light on the glareshield annunciator panel indicates that either an engine fuel filter or an aircraft fuel system fuel filter is clogged or clogging and is bypassing fuel. The aircraft fuel filter annunciator circuits are wired through the squat switch and may cause the FUEL FILTER light to illuminate only if the aircraft is on the ground. The engine fuel filter annunciator circuits are not wired through the squat switch and may cause the FUEL FILTER light to illuminate either in flight or on the ground. Refer to FUEL SYSTEM GLARESHIELD LIGHTS.

FUEL FLOW INDICATOR

The FUEL FLOW indicator is a circular scale, dual-reading instrument and is located on the center instrument panel. The indicator utilizes two pointers (one labeled "L", and the other labeled "R") to indicate fuel flow. The indicator scale reads from 0 to 2000 pounds per hour in 50 pound-per-hour increments. A fuel-flow transmitter for each engine measures fuel-flow by means of a rotor-turbine installed in the engine fuel supply line between the fuel control unit and the fuel manifold. As fuel flows through the turbine a pick-up coil in the transmitter emits pulses which are applied to a monitor. The monitor converts the pulses into a continuous DC current which is applied to the indicator. The monitor also applies a pulsating DC signal to the fuel counter on the fuel control panel. Each pulse represents one pound of fuel. The fuel flow indicating system operates on 28 VDC supplied through a 10-amp current limiter.

FAN SPEED (N_1) INDICATORS

A Fan Speed (N_1) indicator for each engine is installed on the center instrument panel. Each indicator utilizes both a four-place digital display and a circular scale with pointer to indicate Fan Speed (N_1). The circular scale is marked from 0% to 110% in 5% RPM increments. The digital display units, tens, and hundreds numerals are white numerals on a black background and the decimal digit is a black numeral on a white background. N_1 speed is measured by a monopole transducer at the aft end of the low-pressure rotor. A speed gear is attached to the low-pressure shaft rotating around a stationary transducer. As the speed gear turns within the transducer, its teeth cause the magnetic flux path in the air gap to be constantly changing. The dual output transducer monopole produces two separate and identical signals due to the changing magnetic field. The frequency of the output signal represents the rotating speed of the rotating N_1 group. One output signal is applied to the Fan Speed (N_1) indicator and the other to the DEEC. An OFF flag on the indicator will be in view whenever electrical power is not available to the instrument. Electrical power for the indicators is 28 VDC supplied through the 2-amp L and R N_1 circuit breakers on the pilot's and copilot's circuit breaker panels respectively. The N_1 indicators are powered by EMER BAT 1 during EMER BUS mode and electrical power system failures. Refer to Airplane Flight Manual for instrument limit markings.

TURBINE SPEED (N_2) INDICATORS

A Turbine Speed (N_2) indicator for each engine is installed on the center instrument panel. Each indicator utilizes both a four-place digital display and a circular scale with pointer to indicate Turbine Speed (N_2). The circular scale is marked from 0% to 110% in 5% RPM increments. The digital display units, tens, and hundreds numerals are white numerals on a black background and the decimal digit is a black numeral on a white background. N_2 speed is measured by a monopole transducer adjacent to the output gear of the transfer gearbox. The dual output transducer produces two separate and identical signals. The frequency of the output signal represents the speed of the rotating N_2 group. One output is routed to the DEEC and the other is applied to the Turbine Speed (N_2) indicator. An OFF flag on the indicator will be in view whenever electrical power is not available to the instrument. Electrical power for the indicators is 28 VDC supplied through the 2-amp L and R N_2 circuit breakers on the pilot's and copilot's circuit breaker panels respectively. Refer to Airplane Flight Manual for limit markings.

TURBINE TEMPERATURE (ITT) INDICATORS

A Turbine Temperature (ITT) indicator for each engine is installed on the center instrument panel. Each indicator utilizes both a three-place digital display and a circular scale with pointer to indicate Turbine Temperature (ITT). The scale is marked from 100°C to 1000°C in 50°C increments. The digital display numerals are white numerals on a black background. Interstage turbine temperature for each engine is sensed by Chromel-Alumel parallel wired thermocouples positioned between the high- and low-pressure turbine sections. The signal from the averaging circuit of the thermocouples is applied to the Turbine Temperature (ITT) indicator and the digital electronic engine control. An OFF flag on the indicator will be in view whenever electrical power is not available to the instrument. Electrical power for the indicators is 28 VDC supplied through the 2-amp L and R ITT circuit breakers on the pilot's and copilot's circuit breaker panels respectively. The ITT indicators are operative during EMER BUS mode. Refer to Airplane Flight Manual for instrument limit markings.

ENGINE FIRE DETECTION SYSTEM

Three heat-sensing elements connected in series are located in each engine nacelle to detect an engine fire. One element is located around the accessory gearbox; one is located around the engine tailcone; and another around the engine firewall. The fire detection system is controlled by two fire detect control boxes located in the tailcone. In the event of an engine fire, the applicable control box will sense a resistance change in the sensing elements and flash the applicable ENG FIRE PULL T-handle warning light. Warning is given if the firewall or accessory gearbox area exceeds approximately 410°F (210°C) or the engine tailcone area exceeds approximately 890°F (476°C). Electrical power for the system is 28 VDC supplied through the 7.5-amp L and R FIRE DET circuit breakers on the pilot's and copilot's circuit breaker panels respectively. Fire detect systems are operative during EMER BUS mode

SYSTEM TEST SWITCH-FIRE DETECTION FUNCTION

The rotary-type system test switch on the pilot's instrument panel is used to test the fire detection system. Rotating the switch to FIRE DET and depressing the switch PRESS TEST button will connect a resistance into both fire detect system circuits. This resistance, simulating an engine fire, will cause both ENG FIRE PULL T-handle warning lights to illuminate and flash.

ENG FIRE PULL LIGHT

A red ENG FIRE PULL warning light is installed in each T-handle on the glareshield to warn the crew of a fire in the associated engine nacelle. In the event of an engine fire, the associated ENG FIRE PULL light will illuminate and flash. Operation of the T-handle is explained under ENGINE FIRE EXTINGUISHING SYSTEM.

ENGINE FIRE EXTINGUISHING SYSTEM

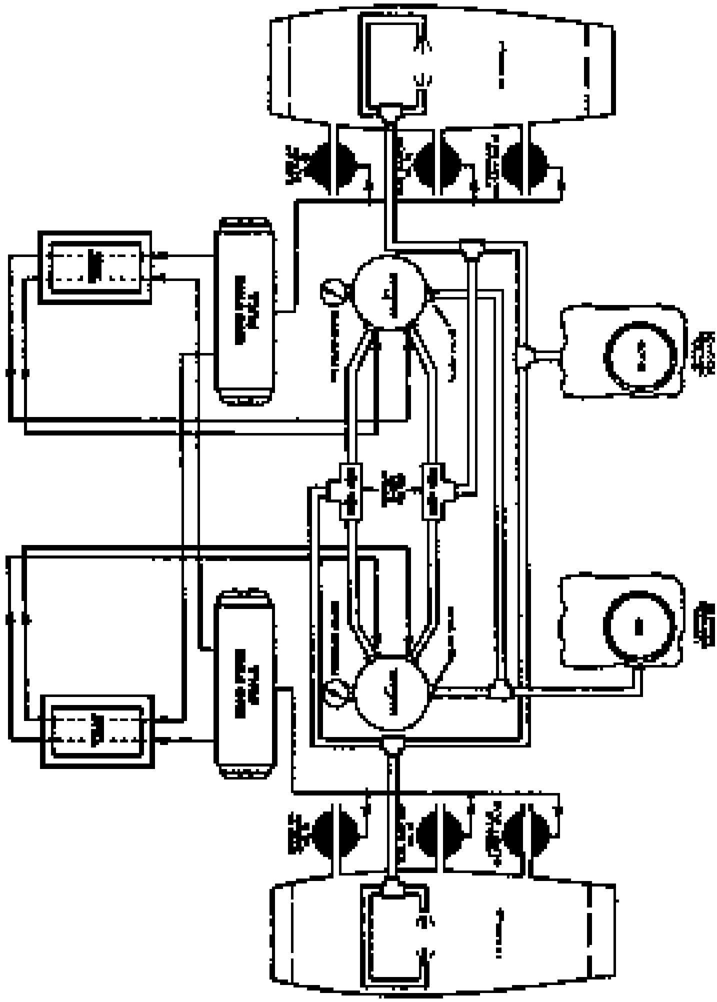
The engine fire extinguishing system components include: two spherical extinguishing agent containers, an ENG FIRE PULL T-handle for each engine, two amber ENG EXT ARMED light/switches, one hydraulic shutoff valve for each engine, one fuel shutoff valve for each engine, a thermal discharge indicator, a manual discharge indicator, and associated wiring and plumbing. The system also utilizes the pneumatic system bleed-air shutoff valves. The system is plumbed to provide the contents of either or both extinguishing agent containers to either engine nacelle. Two-way check valves are installed to prevent extinguishing agent flow between containers. The extinguishing agent, Halon 1301 (bromotrifluoromethane [CF_3Br]), is stored under pressure in the extinguisher containers and a pressure gage on each container is visible from inside the tailcone. Halon 1301 is non-toxic at normal temperatures and is non-corrosive. As Halon 1301 is non-corrosive, no special cleaning of the engine or nacelle area is required in the event the system has been used. The system operates on 28 VDC supplied through the 7.5-amp L and R FIRE EXT circuit breakers on the pilot's and copilot's circuit breaker panels respectively. Fire extinguishing systems are operative during EMER BUS mode.

ENG FIRE PULL T-HANDLE AND ENG EXT ARMED LIGHTS

The engine fire extinguishing system is operated through the ENG FIRE PULL T-handles and the ENG EXT ARMED lights on the glareshield outboard of the T-handles. The ENG EXT ARMED lights are combination light/switches. When an ENG FIRE PULL T-handle is pulled, the associated engine fuel, hydraulic, and bleed-air shutoff valves will close to isolate the affected engine and the ENG EXT ARMED lights will illuminate. Illumination of the ENG EXT ARMED lights indicates that the fire extinguishing system is armed. Depressing an illuminated ENG EXT ARMED light will discharge the contents of one extinguisher bottle into the associated nacelle. Depressing the other ENG EXT ARMED light will discharge the contents of the remaining bottle. Either or both ENG EXT ARMED lights may be depressed to extinguish the fire. Should one container control the fire, the other container is available to either engine.

FIRE EXTINGUISHER DISCHARGE INDICATORS

Two disk-type indicators are flush-mounted in the fuselage under the left engine pylon. If the contents of either or both containers have been discharged into the engine nacelles, the yellow disk will be ruptured. If the contents of either or both containers have been discharged overboard as the result of an overheat condition causing excessive pressure within the containers, the red disk will be ruptured. If both disks are intact, the system has not been discharged. The indicators are readily available for visual inspection.



20-28C

FIRE EXTINGUISHING SYSTEM
Figure 2-3

THRUST REVERSERS (OPTIONAL)

Each engine may be equipped with an independent, electrically controlled, hydraulically actuated, clamshell-type thrust reverser. The thrust reverser system consists of an engine nacelle afterbody installation on each engine, a control panel above the glareshield annunciator panel, reverse thrust levers on the main thrust levers, associated hydraulic plumbing, and associated electrical wiring. Each nacelle afterbody installation consists of an upper and lower blocker door, a door actuator, a door deployed switch, inboard and outboard latch mechanisms, latch position switches, and a throttle retard mechanism. Hydraulic power for thrust reverser operation is supplied by the aircraft hydraulic system; however, a precharged (900-1000 psi [6205-6895 kPa]) hydraulic accumulator and a check valve in the thrust reverser system provide sufficient hydraulic pressure for one deploy and stow cycle in the event the aircraft hydraulic system malfunctions. A selector valve for each thrust reverser is installed in the tailcone. The selector valves control hydraulic flow to the associated system actuators in response to electrical inputs from the associated thrust reverser lever and position switches. Electrical power for thrust reverser control and indication circuits is 28 VDC supplied through the 3-amp L and R TR PWR and the 2-amp L and R TR CONT circuit breakers on the pilot's and copilot's circuit breaker panels respectively.

In order to arm the thrust reversers, the aircraft squat switches must be in the ground mode (aircraft weight on the main gear), the main thrust levers must be in the IDLE position, and ARM must be selected on the control panel. Once armed, the thrust reversers may be deployed and stowed by operating the reverse thrust levers. When the deploy cycle is initiated, hydraulic pressure is applied to the stow side of the door actuators which move the doors into an overstowed condition. Overstowing the thrust reversers allows the latch actuators to release the latches. After the latches release, hydraulic pressure is applied to the deploy side of the door actuators which push the doors open. When stow is selected, hydraulic pressure is applied to the stow side of the door actuators and the doors move towards the overstow position. As the doors reach overstow, the spring-loaded latches close. When the latches close, the latch position switches signal the selector valve to release the stow pressure on the door actuator. Exhaust gas pressure and springs return the doors to the normal stowed position.

Each thrust reverser system incorporates an automatic emergency stow feature which will automatically stow the thrust reverser in the event the associated doors become partially unlatched or deployed. When deploy has not been selected and two latch position switches on the same side (inboard or outboard) detect an unlatched condition, the thrust reverser system will arm itself, retard engine thrust to idle, and initiate thrust reverser stow. The associated ARM annunciator will illuminate to indicate the system has self armed. The associated DEPLOY annunciator will flash to indicate automatic emergency stow has been activated. The affected engine's thrust lever will go to the IDLE position. The hydraulic stow pressure will continue until at least one of the two latches returns to the latched position or power is removed from the reverser circuits

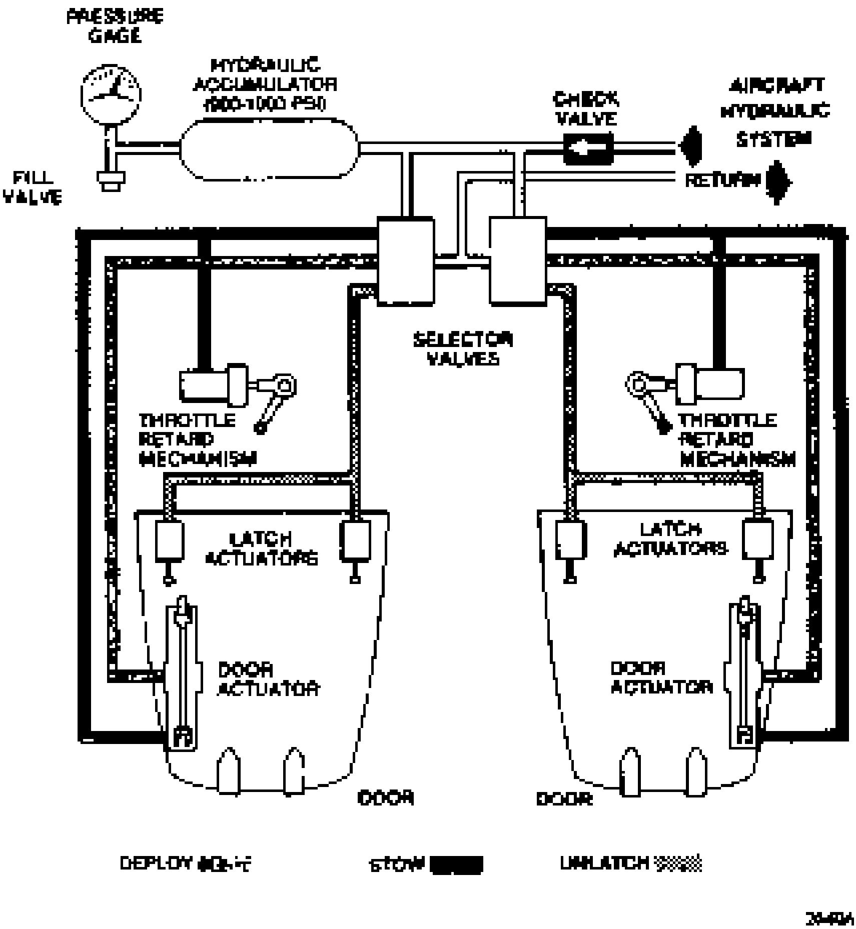
An automatic throttle retard mechanism is installed on each thrust reverser to ensure that thrust reverser stow and deploy does not occur with an engine thrust setting above idle. The throttle retard mechanism consists of an actuator, crank, and lever. Whenever hydraulic stow pressure is applied to the thrust reverser actuators, the throttle retard mechanism will position the engine fuel control input shaft to idle. When hydraulic stow pressure is removed, the mechanism will return to a neutral position. Hydraulic stow pressure is applied during the initial stage of the deploy sequence (overstow), throughout the stow sequence, and during automatic emergency stow.

THRUST REVERSER CONTROL PANEL

The thrust reverser control panel provides the thrust reverser arm, self-test, and deploy annunciation functions. The panel is located at the center of the glareshield, directly above the annunciator panel and is easily accessible from either crew position. The control panel annunciators are tested through the pilot's and copilot's light test switch and are automatically dimmed.

THRUST REVERSER CONTROL SWITCHES

A thrust reverser control switch for each thrust reverser is installed on the thrust reverser control panel. Each switch has three positions: ARM, OFF, and TEST. The TEST position is momentary. When the switch is held to TEST, the hydraulic isolation valve section of the associated selector valve will open and hydraulic pressure for the stow and deploy sections of the selector valve will be available. A pressure switch in the selector valve will illuminate the associated ARM light when pressure is applied. The successful completion of the test indicates the isolation valve is operating properly. When ARM is selected, the associated thrust reverser will be armed for operation (ARM light illuminated) if the aircraft is on the ground (squat switches depressed) and the associated main thrust lever is at IDLE. When fully armed (reverser system relays and switches are properly sequenced), the associated isolation valve is open and the system is ready for the deploy command. When the switch is in the OFF position, the associated isolation valve is closed and hydraulic pressure is not available to operate the thrust reversers.



THRUST REVERSERS
Figure 2-4

ARM LIGHTS

A green ARM annunciator light for each thrust reverser is installed on the thrust reverser control panel to annunciate a thrust reverser armed condition. Each light is operated by a pressure switch between the isolation section and deploy/stow section of the selector valves. Whenever the isolation valve is open hydraulic pressure will actuate the switch which will cause the associated ARM light to illuminate. Illumination of an ARM light with the associated thrust reverser control switch in the TEST position indicates the associated isolation valve is open. Illumination of an ARM light with the associated thrust reverser control switch in the ARM position indicates the associated thrust reverser is fully armed and will deploy when commanded. The ARM light will also illuminate during automatic emergency stow to indicate that the system has self armed.

DEPLOY LIGHTS

An amber DEPLOY annunciator light for each thrust reverser is installed on the thrust reverser control panel to annunciate thrust reverser position. A steady DEPLOY light indicates the associated thrust reverser is fully deployed following a normal deploy sequence. In this event, the light is operated by a deploy switch on the inboard side of the blocker door near the hinge point. A flashing DEPLOY light associated with a steady ARM light indicates automatic emergency stow has been activated. In this event, the light is operated by either two inboard or two outboard latch position switches.

THRUST REVERSER LEVERS

A thrust reverser control lever for each thrust reverser is mounted piggy-back fashion on each main thrust lever. The thrust reverser levers are inoperable and cannot be moved unless the associated main thrust levers are at the IDLE stop. Similarly, the main thrust levers cannot be moved from the IDLE position until the associated thrust reverser levers are in the stow (full down) position. When fully armed, a thrust reverser may be deployed by lifting the corresponding thrust reverser lever to the first (idle/deploy) stop. When the DEPLOY annunciator light illuminates, a throttle release will activate and the thrust reverser lever may be pulled beyond the idle/deploy stop to increase reverse thrust. A stop limits thrust reverser lever travel to maximum reverse at approximately 75% Fan Speed (N1). The thrust reverser is stowed by first returning the thrust reverser lever to the idle/deploy stop and then moving the lever to the stow (full down) position at engine idle speed.

AIRCRAFT FUEL SYSTEM

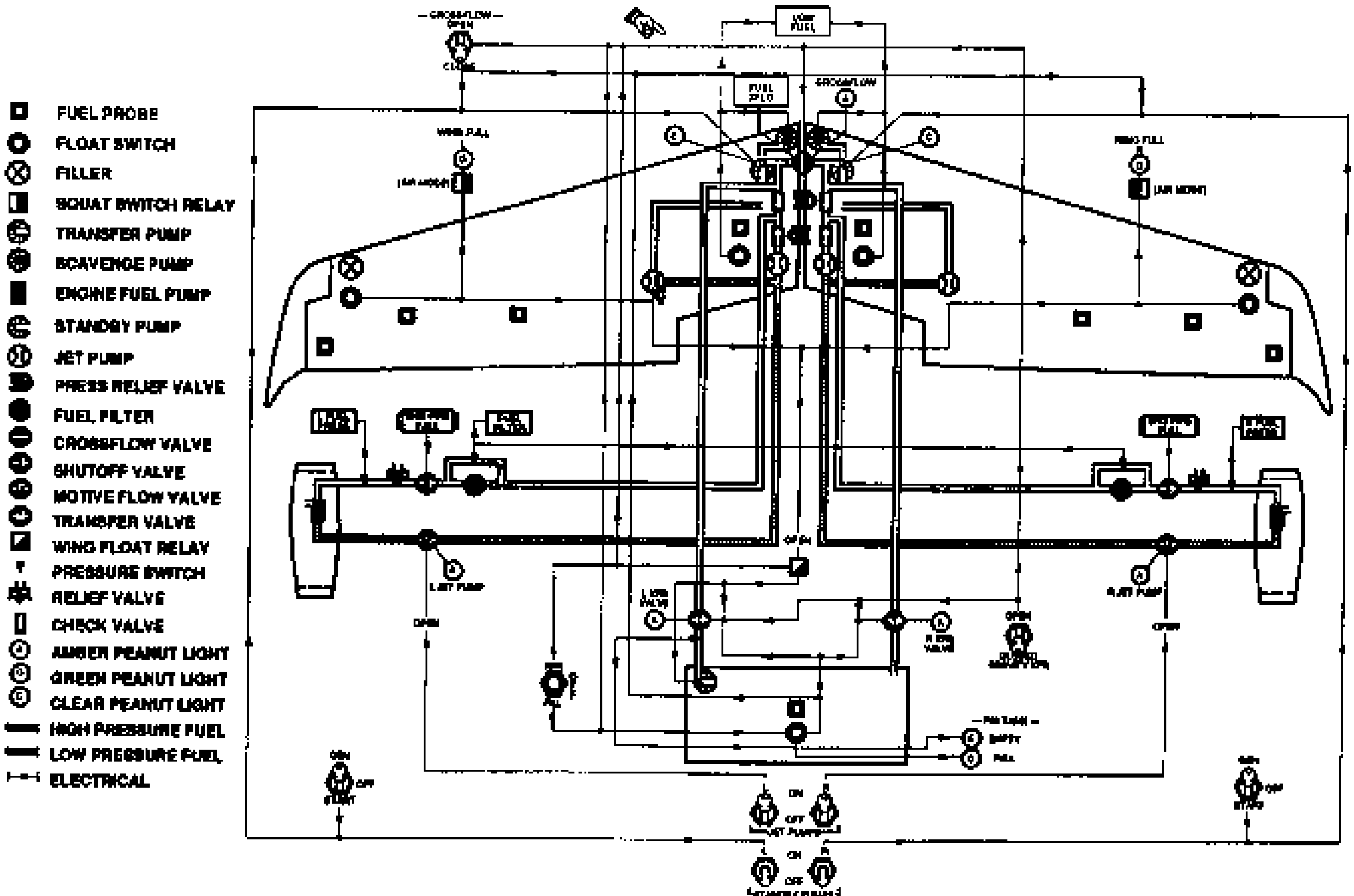
The aircraft fuel system consists of two wing tanks, a fuselage fuel tank, a fuel supply system, a fuel quantity indicating system, a fuel transfer system and a fuel vent system. Fuel fillers are located outboard near each wing tip. An optional single-point refuel (SPPR) system may also be installed.

WING TANKS

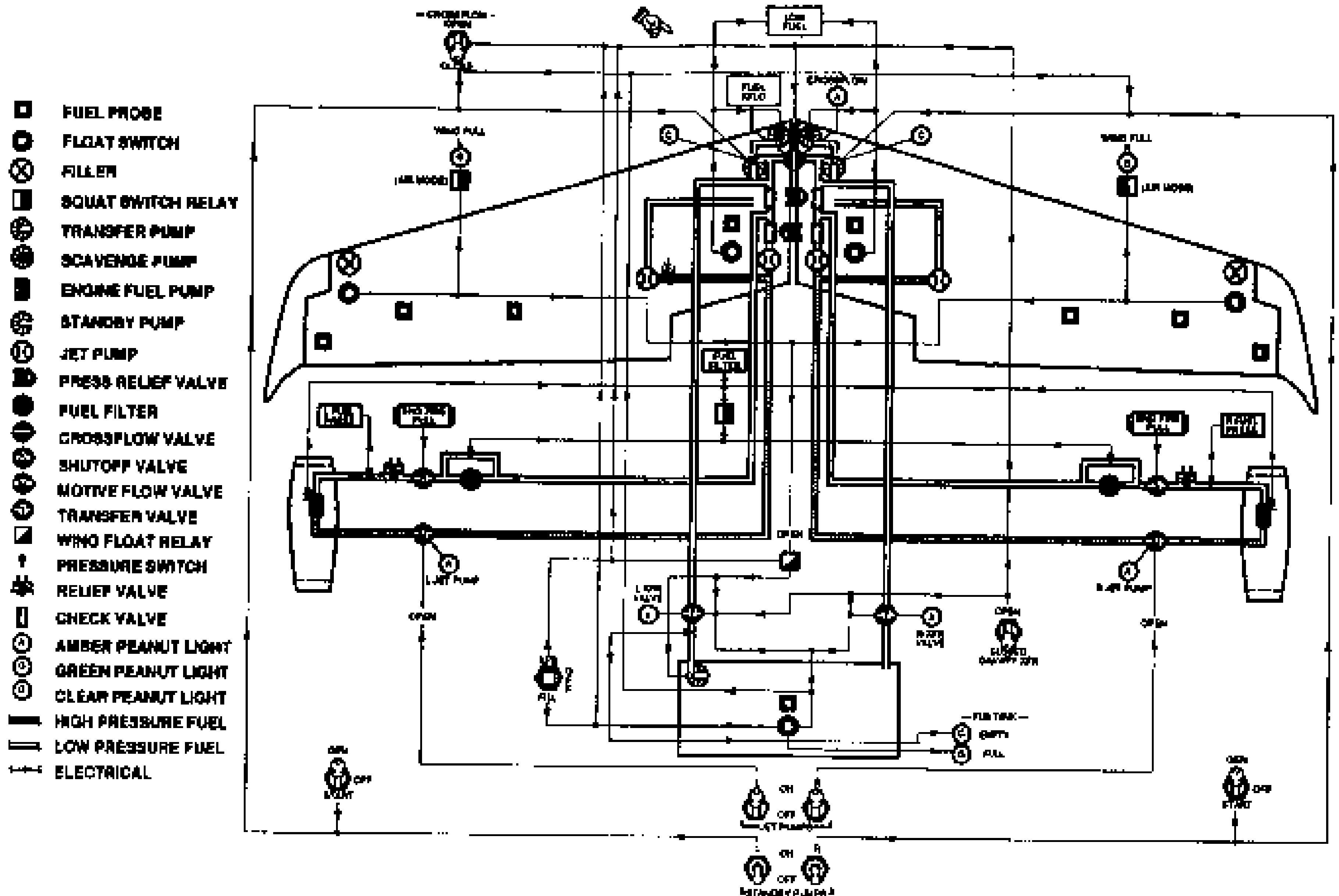
The wing is divided by a center bulkhead into two separate fuel-tight compartments which serve as fuel tanks. Each tank extends from the center bulkhead outboard to the wing tip rib, thus providing a separate fuel supply for each engine. A tank crossflow valve is installed to permit fuel transfer between wing tanks. Center bulkhead relief valves prevent wing tank over-pressurization during fuel crossflow operations. Flapper-type check valves, located in the various wing ribs, allow free fuel flow inboard but restrict outboard fuel flow. A jet pump and an electric standby pump are mounted in each wing tank near the center bulkhead to supply fuel under pressure to the respective engine fuel system. An electric scavenge pump, located in the forward inboard section of each wing tank, is used to transfer fuel to the section containing the main fuel pumps and is operated by the low-fuel float switch. A jet-type transfer pump, located outboard of the wheel well in the aft portion of each wing tank, transfers fuel to the section containing the main fuel pumps. A filler cap, located in the outer section of the wing tank, is used for fuel servicing.

FUSELAGE TANK

The standard fuselage tank consists of two bladder-type cells located in the aft fuselage. The tank is equipped with a fuel probe, float switch and transfer pump. When the airplane is being refueled through the wing fillers, the tank is filled by the wing standby pumps through two transfer lines and two transfer valves. When the tank is full, the float switch deenergizes the wing standby pumps and closes the transfer valves. Fuel can be transferred to the wing tanks by two methods: normal transfer and gravity transfer. During the normal fuel transfer, the fuselage tank transfer pump will pump fuel into both wing tanks. During gravity transfer, fuel will flow to both wing tanks through both transfer lines. The extended range (ER) fuselage tank incorporates two additional bladder-type cells installed immediately forward of the main fuselage fuel cells. Fuel from the ER fuel cells gravity flows into the main fuselage fuel cells through interconnecting tubes.



FUEL SYSTEM SCHEMATIC
(Aircraft without Fuel Heaters)
Figure 2-5



FUEL SYSTEM SCHEMATIC
(Aircraft with Fuel Heaters)
Figure 2-6

FUEL CONTROL PANEL SWITCHES AND INDICATORS

The fuel control panel (Figure 2-7) incorporates all the necessary switches and indicators to fuel the aircraft (when using the over-the-wing method) and to maintain proper fuel management.

JET PUMP SWITCHES

The JET PUMP switches, on the fuel control panel, control the motive flow valves. Setting a JET PUMP switch to ON opens the corresponding motive flow valve and allows high-pressure fuel from the corresponding engine-driven fuel pump to flow to the corresponding jet pumps. The JET PUMP switches normally remain ON at all times. During engine start, a lockout valve in the engine-driven fuel pump prevents motive flow through the motive flow valve and the applicable standby pump is energized to supply starting fuel to the engine. After the engine is running and engine-driven fuel pump output pressure is sufficient, the lockout valve will open allowing high-pressure fuel flow through the motive flow valves. Setting a JET PUMP switch Off closes the corresponding motive flow valve and prevents fuel system pressurization until the standby pumps are turned on. The motive flow valves operate on 28 VDC supplied through the 5-amp L and R JET PUMP-XFR VAL circuit breakers on the pilot's and copilot's circuit breaker panels respectively. Loss of power to the motive flow valve causes the valve to remain in its last position. Motive flow valves are operative during EMER BUS mode.

STANDBY PUMP SWITCHES

The STANDBY PUMP switches, on the fuel control panel, control the operation of the standby electric pumps. The switches normally remain OFF except in the event of a jet pump failure or during fuel crossflow. Regardless of switch position, the standby pumps are automatically deenergized during fuselage fuel transfer operations. The standby pumps are automatically energized when the FUS TANK XFER-FILL switch is set to FILL or the START-GEN switch is set to START. The standby pumps operate on 28 VDC supplied through the 15-amp L and R STBY-SCAV PUMP circuit breakers on the pilot's and copilot's circuit breaker panels respectively.

CROSSFLOW SWITCH

The CROSSFLOW switch, on the fuel control panel, controls the crossflow valve. The switch has two positions: OPEN and CLOSE. When the switch is set to OPEN, the power is applied to open the motorized crossflow valve allowing fuel to flow between the wing tanks. The crossflow valve is opened automatically when filling the fuselage tank from the wings and during fuselage fuel transfer operations. To balance wing fuel, the CROSSFLOW switch should be set to OPEN and the heavy side STANDBY PUMP switch set to ON. The standby pump on the light side should be Off. The WING FULL light, when illuminated by the wing-full float switch, is extinguished when the crossflow valve closes. The standby pump will continue to operate until the STANDBY PUMP switch is set to Off. The crossflow valve allows all usable fuel aboard the aircraft to be available to either engine. The switch should be set to Off except when correcting an out-of-balance condition. The crossflow valve operates on 28 VDC supplied through the 5-amp XFLOW VALVE circuit breaker on the copilot's circuit breaker panel. Loss of power to the crossflow valve causes the valve to remain in its last position. The crossflow valve is operative during EMER BUS mode.

FUS TANK XFR-FILL SWITCH

The FUS TANK XFR-FILL switch, on the fuel control panel, is used to operate the fuel transfer system and to fill the fuselage tank from the wing tanks. The switch has three positions: XFR, OFF and FILL. The switch incorporates a magnetic latch in the FILL position and must be held to that position a minimum of three seconds before the latch will engage. If the switch is in the FILL position and the LOW FUEL light illuminates or the squat switch goes to the air mode, the latch will disengage and the switch will go to the OFF position. The left fuel transfer valve operates on 28 VDC supplied through the 5-amp L JET PUMP XFR VAL circuit breaker on the pilot's circuit breaker panel. Loss of power to the left transfer valve causes the valve to remain in its last position. The transfer pump operates on 28 VDC supplied through the 10-amp FUS XFR PUMP circuit breaker on the pilot's circuit breaker panel.

When the switch is set to FILL, both wing tank standby pumps are energized, both left and right transfer valves are opened via the fuselage tank float switch, and the crossflow valve will open. Fuel will then be pumped into the fuselage tank from the wing tanks until the switch is turned OFF or the fuselage tank float switch actuates to close the transfer valves, shut down the standby pumps, and illuminate the green FUS TANK FULL light. Placing the switch in the OFF position will extinguish the FUS TANK FULL light and close the crossflow valve.

When the switch is set to XFR, the transfer pump is energized, the left transfer valve will open, both standby pumps will be rendered inoperative, and the crossflow valve will open. Fuel will then be pumped from the fuselage tank to the wing tanks until the wing float switches actuate to deenergize the transfer pump, close the transfer valve, and, if squat switch is in the air mode, illuminate the applicable green WING FULL light (the crossflow valve will remain open). If switch remains in the XFR position, fuel transfer will automatically reactivate when the fuel level in the wings is low enough to extinguish the WING FULL lights. If the fuselage tank should empty before the wing float switches shut down the transfer system, a pressure switch in the fuselage tank transfer line will illuminate the white FUS TANK EMPTY light. Setting the switch to OFF will extinguish the white FUS TANK EMPTY and/or WING FULL lights (if illuminated), close the left transfer valve, deenergize the transfer pump, and close the crossflow valve.

FUS TANK GRAV XFR SWITCH

The FUS TANK GRAV XFR switch on the fuel control panel can be used to transfer fuselage fuel without using the transfer pump. The switch has two positions, OPEN and CLOSE.

When the switch is set to OPEN, both transfer valves will open, the crossflow valve will open, and both standby pumps will be rendered inoperative. Fuel will then gravity flow from the fuselage tank to the wing tanks until the wings are full or the wing and fuselage tank heads are equal. When using this method to transfer fuel, approximately 150 to 300 pounds (68 to 136 kilograms) (depending upon flight attitude) of fuel will remain in the fuselage tank and the FUS TANK EMPTY light will be inoperative. To assure all possible fuel has been transferred, reference must be made to the fuel quantity indicator. The switch should be set to CLOSE when all fuel possible has been transferred and during approach and landing. Loss of power to the crossflow and transfer valves causes the valves to remain in their last position. The crossflow and transfer valves are operative during EMER BUS mode.

FUS TANK SWITCH PRIORITY

If the FUS TANK switches are positioned to contradictory positions the FUS TANK XFR-FILL switch will have priority over the FUS TANK GRAV XFR switch.

FUS TANK FULL LIGHT

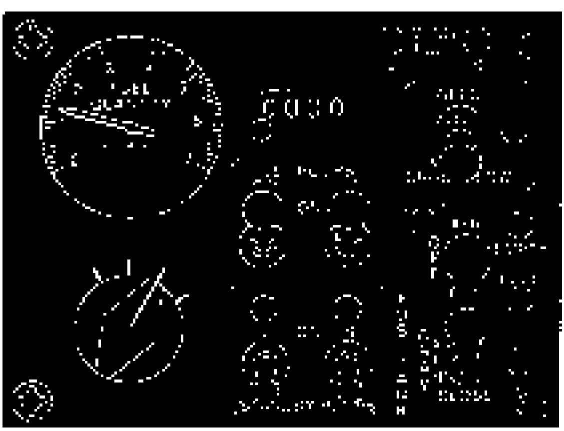
The green FUS TANK FULL light, on the fuel control panel, is installed to indicate a fuselage tank full condition during fuselage tank fill operations. The light is illuminated through actuation of the fuselage tank float switch. During normal fuselage tank fill operations, actuation of the float switch will illuminate the FUS TANK FULL light, close the transfer valves, and shut down the standby pumps. The FUS TANK XFR-FILL switch must be set to OFF to extinguish the light.

FUS TANK EMPTY LIGHT

The white FUS TANK EMPTY light, on the fuel control panel, is installed to indicate a fuselage tank empty condition during fuel transfer. The light is operated by a pressure switch in the left fuselage fuel transfer line. As the fuselage tank empties during transfer operations, the pressure switch senses a loss of pressure in the transfer line and complete circuits to illuminate the FUS TANK EMPTY light. The transfer pump will continue to operate until the FUS TANK XFR-FILL Switch is set to OFF. Setting the FUS TANK XFR-FILL to OFF will extinguish the light and deenergize the transfer pump.

STANDBY PUMP LIGHTS

On aircraft 31-037 and subsequent, a white standby pump light is installed adjacent to each STANDBY PUMP switch. Illumination of either light indicates power is applied to the corresponding standby pump.



FUEL CONTROL PANEL
Figure 2-7

FUEL VALVE LIGHTS

Steady illumination of any of the amber fuel valve lights on the fuel control panel indicates the corresponding valve is not in the position selected or a loss of power. Momentary illumination while the corresponding valve is in transit after switching modes indicates proper operation.

WING FULL LIGHTS

Two green WING FULL lights, on the fuel control panel, are installed to indicate a wing-full condition during inflight fuel transfer operations (crossflow valve open). The lights are illuminated by the wing high-level float switches if the squat switch is in the air mode. During normal fuselage fuel transfer procedures, actuation of either high-level float switch will deenergize the transfer pump, close the left transfer valve, and illuminate the appropriate WING FULL light. During gravity fuselage fuel transfer procedures, actuation of either high-level float switch will illuminate the appropriate WING FULL light but will not close the left or right transfer valves.

FUEL QUANTITY INDICATOR AND SELECTOR SWITCH

The fuel quantity indicator, on the fuel control panel, indicates fuel quantity in pounds of fuel. The indicator face is graduated in 100-pound increments from 0 to 5000 pounds on aircraft with standard fuselage tank and 0 to 7000 pounds on aircraft with extended range fuselage tank. Fuel quantity is sensed by four capacitance-type fuel quantity probes in each wing tank and a capacitance-type fuel quantity probe in the fuselage fuel tank. The left inboard fuel quantity probe incorporates a fuel temperature compensator which compensates for fuel density changes due to temperature. The selector switch permits reading of each tank separately as well as total system quantity. The usable fuel quantities for each position of the selector switch are based upon a fuel density of 6.7 pounds per gallon. The fuel quantity indicating system operates on 28 VDC supplied through the 2-amp FUEL QTY circuit breaker on the copilot's circuit breaker panel. The fuel quantity indicator is operative during EMER BUS mode.

FUEL COUNTER

The fuel counter, on the fuel control panel, indicates quantity of fuel used in pounds. The resettable digital-type counter is operated by pulsating DC signals from the fuel flow indicating system monitor.

FUEL SYSTEM GLARESHIELD LIGHTS

FUEL PRESS LIGHTS

The red L FUEL PRESS and R FUEL PRESS warning lights in the glareshield annunciator panel are installed to alert the pilot of a low fuel pressure condition. The FUEL PRESS lights are energized by a pressure switch installed in each engine fuel supply line between the aircraft fuel filter and the engine-driven fuel pump. When fuel supply pressure drops to 0.25 psi (1.72 kPa) or below, the pressure switch closes to illuminate the respective light. At 1.0 psi (6.9 kPa), the switch will re-open. Should the light illuminate, the standby pumps should be used to supply engine fuel.

FUEL FILTER LIGHT

Aircraft equipped with fuel heaters: Illumination of the amber FUEL FILTER light on the glareshield annunciator panel indicates either an engine fuel filter or an aircraft fuel system fuel filter is clogging. Engine fuel filter annunciation is discussed in ENGINE INDICATING. An aircraft fuel system filter is installed in each engine supply line. Each filter installation includes a bypass valve with an integral pressure switch. During ground operations (squat switch in ground mode) if a filter is clogging, a pressure drop across the filter element will cause the corresponding bypass valve to open and actuate the pressure switch completing a ground circuit for the light. The light will illuminate if either or both filters become clogged. Since the annunciation circuit for the aircraft fuel filters is wired through the squat switches, illumination of the FUEL FILTER light in flight indicates a clogged engine fuel filter. (Refer to ENGINE INDICATING.)

Aircraft not equipped with fuel heaters: Illumination of the amber FUEL FILTER light on the glareshield annunciator panel indicates an aircraft fuel system fuel filter is clogging. An aircraft fuel system filter is installed in each engine supply line. Each filter installation includes a bypass valve with an integral pressure switch. If a filter is clogging, a pressure drop across the filter element will cause the corresponding bypass valve to open and actuate the pressure switch completing a ground circuit for the light. The light will illuminate if either or both filters become clogged.

LOW FUEL LIGHT

The amber LOW FUEL light in the glareshield annunciator panel will illuminate when the fuel quantity in either wing tank decreases to approximately 350 pounds (158 kilograms) of fuel with the aircraft in a level attitude. The light is operated by a low wing fuel float switch installed in each wing tank. Either float switch may cause the light to illuminate. In nose-up attitudes, the light may illuminate at a lesser quantity. In nose-down attitudes the light may illuminate at a greater quantity.

FUEL XFLO LIGHT

The amber FUEL XFLO light, on the glareshield annunciator panel, will illuminate when the crossflow valve is in the open position. The crossflow valve will be open during all fuel transfer procedures.

RAM AIR FUEL VENT SYSTEM

The fuel vent system provides ram air pressure to all interconnected components of the fuel system to ensure positive pressure during all flight conditions. Flush mounted underwing scoops (inboard) admit pressure to the fuselage vent system, and a separate set of underwing scoops (outboard) admit pressure for the wing vent systems. The fuselage vent lines are connected to a common sump that has a moisture drain valve. Each wing tank vent system has a sump with a moisture drain valve located next to the wing vent underwing scoops. Overpressurization due to thermal expansion in the wing tanks is relieved through the left and right expansion lines to the fuselage tank. Overpressurization of the fuselage tank is relieved overboard through a pair of pressure relief valves.

SINGLE-POINT PRESSURE REFUEL (SPPR) SYSTEM (OPTIONAL)

The single-point pressure refueling (SPPR) system (if installed) allows the entire fuel system to be serviced through a fuel servicing adapter located on the right side of the aircraft below the engine pylon. The SPPR incorporates a precheck system which allows the operator to check the operation of the system vent and shutoff valves before commencing refuel operations. The major system components are the refuel adapter, the control panel, a vent valve, a shutoff valve and pilot valve for each tank (both wings and fuselage), solenoid valve for the fuselage tank, two precheck valves, and associated plumbing and wiring. The refuel adapter and control panel are located on the right side of the fuselage below the engine pylon. Electrical power to operate the system indicator lights and solenoid valve is 28 VDC supplied from the right battery through the remote BAT switch on the refuel control panel.

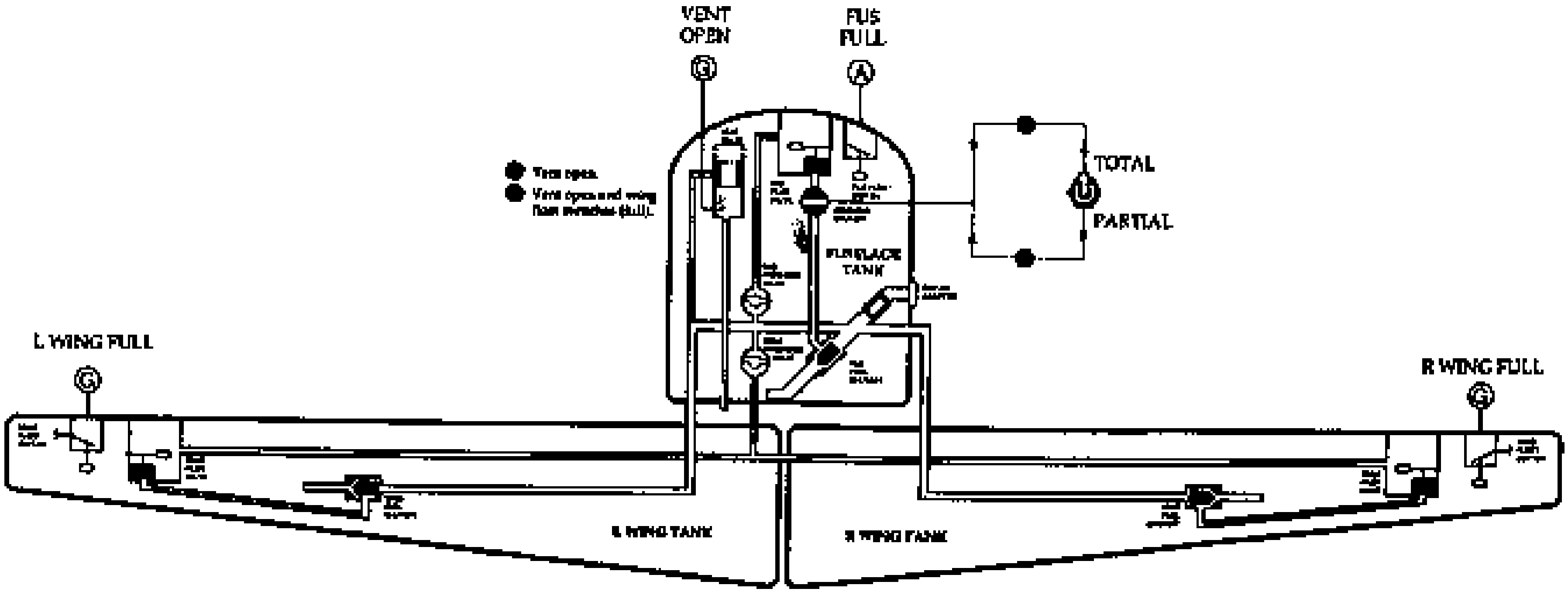
The vent valve is installed to prevent system overpressurization in the event of a shutoff valve failure. Operation of the valve is checked during the precheck sequence. The valve automatically opens whenever fuel pressure is applied to the system. When the valve reaches the full open position, a switch in the valve completes a circuit to illuminate the VENT OPEN light on the SPPR control panel.

Each shutoff valve is controlled by the associated pilot valve located at the high point in each tank. When refueling pressure is applied to the system through the refuel adapter, pressurized fuel is applied to each shutoff valve. This pressure is applied to both sides of the valve poppet. If the pilot valve is open (associated tank not full), some of the pressure acting to hold the valve closed will be vented through the pilot valve and the pressure acting to unseat the poppet will drive the valve open against the spring tension. When the tank fills, the pilot valve will close, fuel pressure on both sides of the shutoff valve poppet will equalize, and spring tension will drive the valve closed.

The solenoid valve for the fuselage tank is located in the line between the fuselage tank pilot valve and shutoff valve. This valve is normally closed and must be energized open in order to open the fuselage shutoff valve for filling the tank. The valve is used to isolate the fuselage tank if filling the fuselage tank is not desired.

WING AND FUSELAGE PRECHECK VALVES

The WING and FUSELAGE PRECHECK valves are used to check operation of the system vent valve and individual shutoff valves before full refueling procedures are commenced. System precheck is accomplished with the Refuel Selector switch set to TOTAL in order to check all shutoff valves. When the WING and FUSELAGE PRECHECK valves are set to OPEN (grips vertical) and refuel pressure is applied to the refuel adapter, fuel will be admitted to the precheck lines and to the tank fill lines. The shutoff valves will open and fuel will flow into all tanks. The fuel in the precheck lines will empty into a float basin at each pilot valve. When the basin fills the pilot valve float will close the pilot valve, which causes the associated shutoff valve to close terminating fuel flow. The vent valve should open when fuel flow is initiated. Fuel flow should stop within 30 seconds.



SINGLE-POINT REFUEL SYSTEM SCHEMATIC
Figure 2-8

REFUEL SELECTOR SWITCH

The Refuel Selector switch, on the SPPR fuel control panel, is used to select the tank(s) to be filled during refueling. The switch has two positions: TOTAL and PARTIAL.

The TOTAL position of the Refuel Selector switch is used to fill the wing and fuselage tanks simultaneously. When TOTAL is selected and refueling pressure is applied (vent valve opens), circuits are completed to open the fuselage tank solenoid valve. When the solenoid valve opens the fuselage tank shutoff valve will open to admit fuel into the fuselage tank.

The PARTIAL position of the Refuel Selector switch is used to fill the wings first and then the fuselage. This is useful when full wings and less than full fuselage fuel is desired. When PARTIAL is selected and the vent valve opens, the fuselage tank solenoid valve will be controlled by the wing high-level float switches. When the wings are full, the wing high-level float switches complete the circuit to open the fuselage tank solenoid valve. When the solenoid valve opens the fuselage tank shutoff valve will open and admit fuel to the fuselage tank.

REMOTE BATTERY SWITCH

The Remote BAT switch, on the refuel control panel, allows operation of the single-point pressure refuel system without the need to enter the cockpit in order to energize aircraft power. When the switch is set to ON, DC power from the aircraft's right battery is applied to the SPPR control circuits.

FUS FULL LIGHT

The amber FUS FULL light, on the refuel control panel, will illuminate whenever the fuselage tank float switch actuates. The light illuminates to alert the operator that refuel operations should have automatically terminated. If fuel flow continues with the light illuminated, fueling operations should be immediately terminated.

VENT OPEN LIGHT

The green VENT OPEN light, on the refuel control panel, will illuminate whenever the fuselage tank vent valve opens. The light is operated by a switch in the valve. Circuits for the fuselage tank solenoid valve are wired through this switch to prevent filling the fuselage tank until the vent valve opens.

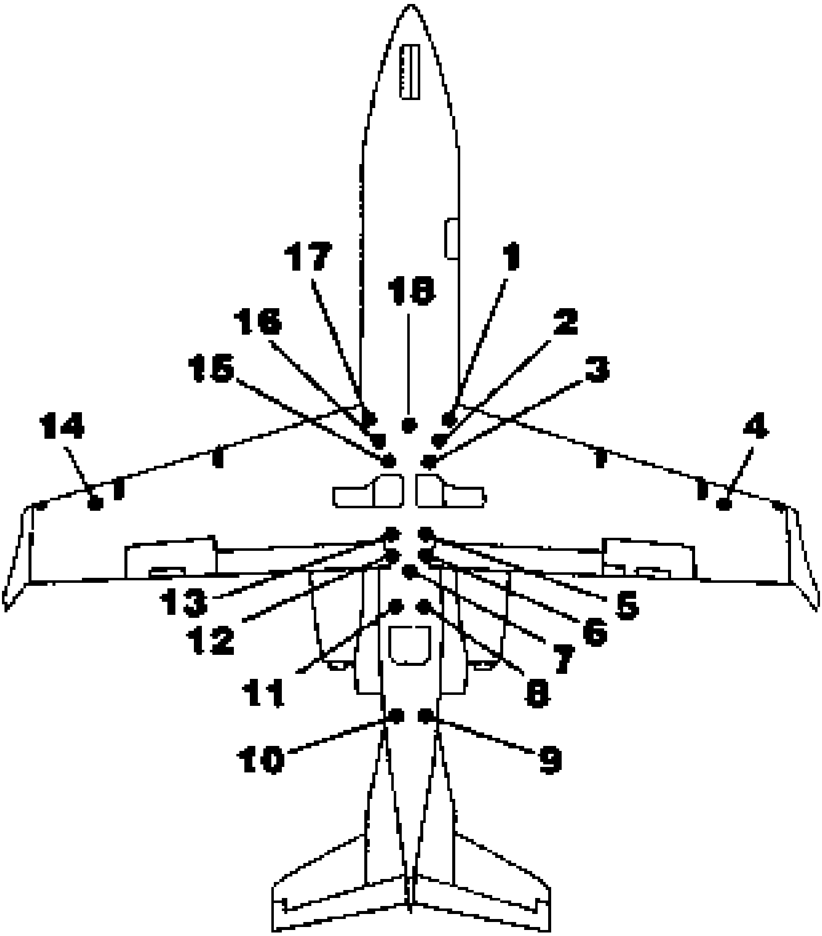
FUEL ADDITIVE

Aircraft equipped with fuel heaters: Anti-icing additive is not a requirement. However, for microbial protection, it is recommended that anti-icing additive be used at least once a week for aircraft in regular use and whenever a fueled aircraft will be out of service for a week or more. Refer to the Airplane Flight Manual for the recommended concentration and the proper method of blending anti-icing additive.

Aircraft not equipped with fuel heaters: Anti-icing additive must be blended with untreated fuel to prevent possible engine flameout due to fuel system icing. Refer to the Airplane Flight Manual for the recommended concentration and the proper method of blending anti-icing additive.

REFUELING

The aircraft is refueled through the filler caps on each wing tip or through the optional single-point pressure refuel adapter located on the right side of the fuselage below the engine pylon. Ground jacks are located on the underside of each wing near the fuel filler and just aft of the single-point pressure refuel control panel. Refer to the Fuel Servicing Addenda (Airplane Flight Manual binder) for a list of approved fuels and refueling procedures.



1. Left Wing Scavenge Pump
2. Left Wing Sump
3. Left Engine Fuel
4. Left Wing Vent (Sump)
5. Left Wing Expansion Line
6. Left Wing Transfer Line
7. Fuel Vent (Fuselage)
8. Left Fuel Filter
9. Left Fuel Computer
10. Right Fuel Computer
11. Right Fuel Filter
12. Right Wing Transfer Line
13. Right Wing Expansion Line
14. Right Wing Vent (Sump)
15. Right Engine Fuel
16. Right Wing Sump
17. Right Wing Scavenge Pump
18. Fuel Crossover

FUEL DRAINS
Figure 2-9