AIR CONDITIONING

Air conditioning packs are used to provide the airflow into the cockpit and passenger cabin areas. This conditioned air input is used for ventilation and is also used to maintain cabin pressurization. The source of the air used to operate the packs is from the pneumatic system. The primary source of hot air for pressurizing the pneumatic system is from the bleed air system on each of the 4 engines. The APU will also supply bleed air for operating the packs on the ground or inflight up to an altitude of 15,000 feet. Ground carts can also be used as a source of air for operating the packs.

Each pack consists of an air cycle machine (ACM), an electronic pack controller and the controls and indicators on the Flight Engineer's panel. The function of the ACM is to cool the hot pressurized bleed air being supplied to the packs.

The air cycle machines utilize heat exchangers to cool the hot bleed air to a temperature suitable for use as a conditioned air supply to the airplane. Outside air is used as the cooling agent for the heat exchangers. The outside air enters the airplane through a large inlet door forward of each pack. The air is then routed through two heat exchangers and then out through a larger exit door. The ACM drives a fan in the coolant air ducting to ensure that coolant air will circulate during ground operations. In flight ram air pressure ensures coolant air circulation. The air cycle machine utilizes a turbine driven by the supply air to drive a compressor which increases the pressure and temperature of the supply air prior to being routed through the second heat exchanger. By increasing the difference between the temperature of the supply air and the coolant air the efficiency of the heat exchanger is increased. The pressurized supply air is allowed to expand prior to leaving the ACM. The temperature of the supply air decreases during this expansion.

A pack controller is utilized to automatically obtain the required ACM outlet temperature. To accomplish this the controller regulates a bypass valve which increases or decreases the air being forced through the ACM turbine. The resultant change in turbine speed also changes the temperature of the supply air leaving the compressor and being routed through the second heat exchanger. During ground operation the INLET and EXIT doors for the air coolant system are fixed in the open positions. The maximum available coolant airflow through the heat exchangers is necessary for all ground operations.

For inflight operations, the pack controller also regulates the position of the INLET and EXIT doors to vary the flow of coolant air through the heat exchangers. When inflight the coolant air system becomes more effective and the bypass valve will open decreasing the speed of the turbine-compressor unit. When the bypass valve reaches the maximum open position the INLET and EXIT doors are programmed towards the closed position as required to maintain the required ACM outlet temperature. When more cooling of the supply air is needed the INLET and EXIT doors will first move to their maximum open position, then the bypass valve will move towards the close position. Whenever the engine thrust level is changed, the temperature of the bleed air will also change. The pack controller automatically repositions the BYPASS VALVE and/or INLET DOOR and EXIT DOOR to maintain the required ACM outlet temperature.

The Pack Valve Switches are on the Flight Engineer's pneumatic system control panel. A pack will operate anytime supply air is routed to the pack. A pack valve operated by the Pack Valve Switch controls the flow of air. The valve closes when the pressure of the supply air is below approximately 8 to 12 psi. The pack valve will continue to modulate airflow when electrical power to the pack controller is lost and pneumatic pressure is available as long as the aircraft altitude remains at 7,000 ft MSI or above.

AUTO PACK OPERATION

The controls for the packs are on the pack temperature control panel. A Pack Control Switch with AUTO and MAN positions is provided to control mode of operation. For all normal operations, these switches are left in the AUTO position. The pack controller then automatically controls the position of the bypass valve, inlet door and the exit door. Indicators are provided to monitor the operation of the packs. Pressing the Pack Selector Switch will illuminate the ON portion of the switch. The indicators will then monitor the operation of that pack.

The indicators include bypass valve position, inlet door position, exit door position, ACM outlet temperature, and compressor discharge temperature. The ACM outlet discharge temperature is controlled within the limits of 35° to 135°F (2° to 57°C). Each ACM temperature controller compares ACM outlet temperature to the desired compartment temperature. The controller positions the pack turbine bypass valve and inlet door so the ACM outlet temperature is correct. The actual ACM outlet temperature is limited by the zone temperature controllers to 160°F (71,C).

When the pointer on the bypass valve position indicator is at the full COOL Position the ACM turbine-compressor is at its maximum speed. For this condition the compressor discharge temperature will be high. When the pointer is at full HEAT position the ACM turbine-compressor is at its minimum speed; the compressor discharge temperature will be low.

An "on the ground" signal from the landing gear TILT indicating system is used to preposition both the INLET and EXIT doors to the full open (COOL) position. They remain in this position for all ground operations. After takeoff, the bypass valve, inlet door, and exit door prepositions to full HEAT, full COOL, and mid range respectively. When a pack is shut down either manually by closing the pack valve or automatically by the pack protection circuits, the bypass valve, inlet door, and exit door will automatically move to a position suitable for pack starting. These are called the "PREPOSITIONS".

All pack controllers receive the same temperature command signal from one zone temperature controller. The ACM outlet temperatures and all other indications for all packs will be approximately the same. The zone that requires the coolest air to achieve the selected zone temperature will control all the operating packs.

MANUAL PACK OPERATION

The pack can be operated in the manual mode by positioning the Pack Control Switch to MAN and pressing the appropriate pack selector switch illuminating the ON portion of the switch. In this configuration the MANUAL Temperature Switch controls the position of the INLET DOOR and the BYPASS VALVE. In MAN operation the EXIT DOOR is always in the full COOL Position. By holding the MANUAL Temperature Switch in the HEAT or COOL position the ACM outlet temperature can be increased or decreased.

When operating with the Pack Control Switch in the MAN position the sequencing of the BYPASS VALVE and the INLET DOOR is the same as in AUTO operations. The EXIT

DOOR will not move from the full COOL position which is the maximum open position. This is true for either ground or inflight operations. When a pack is shut down the BYPASS VALVE and INLET DOOR will remain in their last position.

PACK TRIP LIGHTS

There is an amber PACK TRIP Light for each pack on the Flight Engineer's panel. Protection circuits are provided that will automatically close the pack valve and illuminate the PACK TRIP Light for the following conditions:
High ACM outlet temperature 185°F
High compressor discharge temperature 425°F
Out of sequence of bypass valve vs inlet or exit door.
On some aircraft: the No. 2 pack will shut down when a PRESS RELIEF valve open light or lights illuminate (3 pack airplanes only).
The first three conditions will only occur if there is a malfunction or a failure in the pack itself or in the control circuits. Improper operation in the MAN mode could cause a pack trip because of either or both over temperature conditions. A failure in the pack controller or in one of the door-actuation systems could result in the BYPASS VALVE or INLET or EXIT doors not moving in sequence. The last condition would result from improper operation of the pressurization system.

PACK TRIP RESET SWITCH

A Pack Trip Reset Switch allows the protection circuit to be reset and the pack to be restarted.

CONDITIONED AIR DISTRIBUTION

Air from the cooling packs is ducted into the pressurized compartments, circulates freely, passes through grilles in the sidewalls, and is guided through channels formed by structures down the lower sidewalls and aft to the pressurization control outflow valves. Air cooled by the air cycle packs is collected in a main distribution manifold below the passenger compartment floor. Air is dispersed from the manifold into conditioned air supply ducts for the flight compartment, passenger compartment, and gasper air distribution. Recirculation fans augment the flow rate through the passenger conditioned air system. Most air leaves the passenger compartment through floor level grilles in the sidewalls. Ground service conditioned air can be connected to pack output ducts leading to the main manifold.
The airplane is divided into zones for temperature control. Conditioned air distribution ducts are isolated into separate manifolds.

RECIRCULATION FANS

To provide high airflow rates for rapid temperature control response without bleeding excessive air from the engines, some of the main cabin compartment conditioned air is recirculated by means of fans. Main cabin compartments have recirculating fans above the ceiling which draw in air that has passed through the compartment and up around trim panels. Recirculation air is directed into the zone header where it mixes with fresh air from the main distribution plenum. The configuration of the fan outlet and the ducts makes the introduction of fan discharge air tend to induce flow up the risers from the plenum to assure the proper ratio of fresh and recirculated air.

The three fans are independent of one another. Each is controlled by a separate switch and its own relay. Fan motors operate on 115 volt AC. Control circuits use 28 volt DC. Fans can be operated without any conditioned air supply. Check valves prevent reverse flow when fans are not operating.

All operating packs will provide air at the same ACM outlet temperature to the conditioned air manifold. Hot pneumatic system air is then used to increase the temperature of the conditioned air going to a zone. Each zone temperature controller automatically determines the temperature of the conditioned air needed for its zone. The zone that requires the coolest conditioned air automatically controls the operating packs. The packs will operate to provide an ACM outlet temperature suitable for use by that zone. All other zones will automatically add hot pneumatic trim air to the conditioned air as necessary to satisfy each zones requirements. The zone controlling the packs will automatically shift when conditions change and another zone controller requires cooler air. The trim air system routes the hot pneumatic trim air from the cross body pneumatic duct to the ducting going to each zone. A TRIM AIR Switch when positioned to OPEN will open a master trim air valve when a pack is operating. With this valve open hot trim air is available for use by the zone temperature controllers for increasing the conditioned air temperature. Each zone controller automatically adjusts its trim air valve to regulate the amount of hot trim air being added to the conditioned air going to its zone. Trim air indicators are provided to display position of the trim air valves for each zone and indicate the relative amount of hot trim air being added to the conditioned air going to that zone.

MANUAL ZONE TEMPERATURE CONTROL

A MANUAL mode of operation for zone temperature control is provided. This would be utilized when directed by an alternate operation. This usually would follow failure of a zone temperature controller, trim air valve or loss of trim air (ducting failure). Rotating a Zone Temperature Switch to the MANUAL range will bypass the zone controllers automatic control of its trim air valve. In MANUAL the position of the trim air valve is controlled by the zone temperature switch. Moving and holding the switch towards COOL will close the associated trim air valve. This is indicated on the trim air indicator by the pointer moving towards COOL.

NOTE: Moving all zone temperature switches to the MANUAL range will result in no zone maintaining control of the packs. The operating packs will automatically operate to provide an ACM outlet temperature of 35°F.

ZONE OVERHEAT PROTECTION

When operating a zone temperature control in auto and an overheat occurs, the zone OVERHEAT light will illuminate and position the zone trim air valve to full COOL.

If a zone overheat occurs while operating in manual, the zone OVERHEAT light will illuminate; however, the zone trim air valve will not reposition automatically.
A Zone Overheat Reset Switch allows the automatic zone temperature control to be activated after the duct temperature has decreased.

PRESSURIZATION GENERAL

The conditioned air being supplied to the main cabin and cockpit zones is also used for maintaining airplane pressurization. This is accomplished by controlling the quantity of air being vented overboard. Two outflow valves located in the lower airplane surface aft of the lower aft cargo compartment are used to control the airflow leaving the pressurized cabin areas. A cabin pressurization controller is used to automatically control the position of the two outflow valves. Both valves are repositioned at the same time by the signal from the controller.

AUTOMATIC PRESSURIZATION CONTROL

For all flight conditions the pressurization controller is operated in the AUTO mode. The AUTO mode is selected with the Pressurization Mode Switch. This switch and the other controls and indicators for the pressurization system are on the Flight Engineer's panel. A Pressurization Rate Switch is provided to set the rate at which the cabin altitude will climb or descent. An index mark on the panel is used as a standard setting. With the Rate Switch set on the index mark the cabin will climb at a rate no higher than 500 feet per minute or descend at a rate no higher than 300 feet per minute. The available range of the Rate Switch is 150 to 2500 feet per minute climb or 100 to 1500 feet per minute descent.
To initiate operation in the AUTO pressurization mode, the pressurization controller needs an input for the desired cruise airplane altitude and the associated barometric pressure. With these inputs, pressurization control will be maintained during takeoff, climb and cruise. For control during descent, approach and landing phases the landing field elevation and existing field barometric pressure must be provided to the pressurization controller. A Flight/Cabin Altitude Selector and a Barometric Setting Selector are provided to insert these altitudes and pressures.
For normal operation the pressurization controller will control the position of the outflow valves to control the cabin rate of climb or descent to the selected cabin cruise altitude and pressure or the selected field elevation and pressure. The pressurization controller has its own limit for the cabin differential pressure, which is 8.5 pounds per square inch. The indicator tape controlled by the Flight / Cabin Altitude Selector shows the relationship between the airplane cruise altitude and the cabin altitude based on the 8.5 psi limit for the controller. The controller uses an input from the No. 2 Auxiliary static system for determining the actual airplane altitude.
Indicators are provided to monitor system operation. The pointers on the two OUTFLOW VALVE indicators should move together and indicate approximately the same position. Indicators are provided for cabin altitude, cabin differential pressure and cabin vertical speed.
The controller will automatically reposition the two outflow valves to the full open position on landing. The valves will remain in the full open position during all ground operations. An "on

the ground" signal from the landing gear TILT indicating system initiates this action. The controller opens the outflow valves to the full open position but also keeps the rate within the selected pressurization rate.

MANUAL PRESSURIZATION CONTROL

In the AUTO pressurization mode the outflow valves are controlled by AC electric motors. As a backup the valves can also be repositioned by DC electric motors. To move the outflow valves with the DC motors, the Pressurization Mode Switch must be placed in the MAN position and the outflow Valve Manual Control Switches held in either the OPEN or CLOSED position. The Pressurization Mode Switch also has MAN L and MAN R positions which allows the designated outflow valve to be manually controlled while the other is being automatically repositioned by the controller. If one of the outflow valves fails to move during auto mode operation, this could indicate that the AC motor has failed. By using the appropriate Manual Control Switch, reposition the outflow valve to be in agreement with the valve that is being automatically controlled.

PRESSURIZATION CONTROLLER FAILURE PROTECTION FEATURES

The pressurization controller has an auto fail protection feature. When operating in the AUTO mode, an amber AUTO FAIL (RATE LIMIT) light will illuminate any time the cabin rate climb or rate of descent is excessive. These rates are beyond the range that can be selected with the Pressurization Rate Switch. When one of these conditions is sensed the controller automatically illuminates the AUTO FAIL (RATE LIMIT) light, automatically shifts to the DC motors for controlling the outflow valves and automatically repositions the outflow valves to maintain the cabin altitude existing at the time the failure condition was detected. The AUTO FAIL (RATE LIMIT) system is inoperative when cabin altitude is above approximately 10,000 feet. A second auto pressurization protection feature will automatically close the outflow valves if the cabin altitude climbs to approximately 12,500 feet.

A horn will sound intermittently in the cockpit if the cabin altitude is allowed to exceed 10,000 feet. A Horn Cutout Switch is provided to silence the horn.

A Rate Limit Switch allows the flight crew to check that the auto fail feature of the controller is operative. This check is performed during preflight.

AIRPLANE OVER PRESSURIZATION PROTECTION

Two safety pressure relief valves are located in the lower forward cargo compartment area. These valves will open any time the cabin differential pressure is allowed to reach 9.25 psi. These valves are pneumatically operated and utilize their own internal and external static sources for determining the differential pressure when the valve should open. Each valve also has a second differential pressure sensing system, which operates at a differential pressure of 9.7 psi.

There are negative relief doors located in the lower forward, aft cargo, and side cargo compartment doors that will open to prevent the cabin pressure from going below the ambient pressure.

EQUIPMENT COOLING SYSTEM

The equipment cooling system provides additional ventilation for the electrical/electronic equipment in the cockpit area and for the equipment in the lower electrical/electronic compartment.

The equipment cooling system controls and indicators are on the Flight Engineer's panel. For normal operations with the SMOKE Detector and NO AIRFLOW lights extinguished, the system will automatically operate when the main electric AC bus No. 3 is powered. A blower in the equipment cooling ducting coming from the cockpit areas automatically operates to move the warm air from the cockpit equipment to the lower forward cargo compartment. This warm air is used to provide heating for the lower forward cargo compartment. Two blowers are in the equipment cooling ducting coming from the lower electrical/electronics compartment. One or both of these blowers automatically operate to move warm air from the lower electrical/electronic equipment compartment either overboard or into the lower forward cargo compartment. The Blower Selector Switch and an "on ground" signal from the landing gear TILT indicating system controls which blower or blowers will operate. With the switch in the NORM position, both blowers operate when the airplane is on the ground but only one of the blowers operates inflight. With the Blower Selector Switch in the ALT position only one blower operates during ground operations. A valve in the ducting for discharging the warm air overboard will automatically close as the cabin differential pressure increases. All the warm air then is discharged into the lower forward cargo compartment.

EQUIPMENT COOLING SMOKE AND FAILURE MODES

A photo electric type smoke detector is in a connecting line between the equipment cooling ducting coming from the cockpit area and the ducting coming from the lower electrical/electronic equipment compartment. On detecting smoke from either area the equipment cooling system will automatically initiate operation in the smoke mode and the amber SMOKE detector light will be illuminated. In the smoke mode, a valve in the equipment cooling ducting from the cockpit areas is automatically opened and the blower operation is automatically stopped. Cabin differential pressure will keep the equipment cooling system operating with all the warm air being discharged overboard. The blower in the equipment cooling ducting from the lower electrical/electronic equipment compartment continues to operate in the smoke mode but a valve in the ducting going to the lower forward cargo compartment is automatically closed and the overboard discharge valve is automatically opened. All the air then is discharged overboard. In the smoke mode the smoke detector light could extinguish even though the smoke condition still exists.

There is an airflow detector in the equipment cooling ducting coming from the cockpit area and in the ducting coming from the lower electrical/electronic equipment compartment. When the airflow in either ducting stops, the amber NO AIR FLOW Light will illuminate and the equipment cooling system will automatically operate in the smoke mode. The ground crew call horn will also sound for this condition. It is not recommended to operate electronic equipment without the equipment cooling system operating.

The detector element is a heated thermoswitch element. The airflow has to be restored for up to 30 seconds before no airflow light can be extinguished and the circuit reset with the

No Air Flow Reset Switch. If the NO AIR FLOW light remains illuminated the procedure limits the T/R units load to 54 amps.

There is a Valve Control Switch which is normally left in the guarded NORM position. For certain smoke or fire conditions the procedure will direct the flight crew to place this switch to the SMOKE position. The equipment cooling system will then operate in the smoke mode. In the ditching procedure the flight crew is directed to place the switch in the DITCH position to close the lower overboard discharge valve.

LOWER CARGO COMPARTMENT HEAT

Hot air from the pneumatic system is used to provide heating for the lower aft cargo compartment. The supply ducting comes from the pneumatic system ducting between the two wing isolation valves.

The Lower Aft Cargo Heat Switch in the NORMAL position arms the system. The position of the override valve and control valve are then controlled by thermoswitches in the cargo compartment. A temperature control thermoswitch will open and close the control valve to maintain the correct compartment temperature. An overheat thermoswitch will close the override valve and illuminate the amber OVERHEAT light when the compartment reaches the overheat temperature of 110°F. When both valves are open the green ON light will be illuminated. If the override valve closed due to an overheat condition the ON Light will be extinguished. If the Aft Cargo Heat Switch is left in the NORMAL position with an overheat indication the compartment temperature will be maintained at a higher than normal range, the OVERHEAT and ON Lights illuminating alternately.

During ground operations with high ambient temperatures the overheat light may be illuminate with the Aft Cargo Heat Switch in the OFF position.

AUXILIARY POWER UNIT

The Auxiliary Power Unit (APU) is a self-contained gas turbine engine, installed in the tail cone of the airplane, isolated from flight critical structures and control surfaces by a firewall. The APU provides electrical and pneumatic power for systems operation on the ground. The operational certification decal located over the APU module states if the APU can be started or operated inflight. The APU is not designed to provide electrical power in-flight because generator breakers will trip as the APU generator breakers are selected to close and the aircraft will move to the ground mode electrically.

FUEL SYSTEM

Fuel is normally supplied to the APU from the airplane No.2 main tank. Fuel can be supplied from any tank through the crossfeed system. A DC operated pump, powered by the battery bus and controlled by the APU master switch, will supply fuel from the No. 2 main tank to the APU when the 115 Volt AC ground service bus is not powered. With AC power available to the ground service bus, the No. 2 aft main boost pump will operate automatically. The DC pump is shut down by a pressure switch normally actuated by fuel pressure from the No. 2 main tank aft boost pump. Any other source of fuel for the APU will also turn the DC pump off.

OIL SYSTEM

The APU oil system is a self-contained system consisting of independent supply, pumps, regulator, cooler, filters, and indicator. Tank capacity is four quarts. A quantity indicator is provided on the APU module.

ELECTRICAL SYSTEM

The APU drives two generators identical to the engine driven generators but rated at 90 KVA due to superior cooling. The APU generators alone can normally supply the entire electrical load.

BLEED AIR SYSTEM

The APU provides pressure regulated bleed air to the airplane pneumatic manifold for operation of pneumatic components. When electrical load and bleed air extraction combine to raise EGT above normal operating temperature, a load control valve will restrict bleed air extraction. Therefore, when large amounts of pneumatic power are required, as during engine start, electrical loads should be reduced.

APU BATTERY

The APU battery is identical to the aircraft battery. The main airplane battery switch must be on in order to start the APU.
The APU battery is provided with a battery charger which will be available for n charging the battery when the 115 volt AC ground service bus is powered. The battery charger is disconnected during APU starter engagement.

REMOTE CONTROL MODULE

A remote control module located in the right body wheel well provides the ground crew with a means for:
APU Stop
Fire Warning
Fire Bottle Discharge
NOTE: The APU FAULT light will not illuminate when the remote switch is used for shut down.

STARTING SEQUENCE

7%-- APU fuel solenoid valve opens and ignition starts.
50%-- Starter cutout. DC ammeter and DC voltmeter normal.
95%-- Ignition cutout. APU starts governing. Load enabling switch closes.
Bleed air valve can be opened and generators can be connected to the sync bus. Hour meter starts recording.
100%- Normal operation with APU governing :t: 1.25% RPM.

PULSATING EGT INDICATION

The electronic turbine controls monitors each of the two EGT thermocouple harness circuits of the APU. If one of the two circuits is open or is presenting a significantly lower temperature the fault will be shown on the APU EGT indicator. The EGT reading will pulse every 2 1/2 seconds from a stabilized (valid) reading to a reading approximately 100C above or below the stabilized reading. If indicator pointer is pulsing UP the right harness is malfunctioning, DOWN left harness is malfunctioning. This will not cause an automatic shutdown and the APU is operative if this condition exists. However, corrective maintenance action should be requested as soon as practical to replace the faulty thermocouple harness.

AUTO SHUTDOWN

The auto shutdown circuits will cause APU shutdown by closing the APU fuel shutoff valve, fuel solenoid valve, and deactivating ignition (if operating). The following will cause APU auto shutdown:

- Overspeed (110% RPM)
- Fire signal (either from the fire detection system or by operation of one of the fire switches). Fault light will not illuminate for these conditions.

- High EGT (when EGT reaches maximum allowable for the RPM range).
- Low oil pressure.
- High oil temperature.
- Loss of DC voltages (fault light will not illuminate).
- Cooling air shutoff valve closes.
- APU bleed air duct leakage.
- Loss of RPM input to electronic turbine control.
- Air intake screen blocked.
- Fuel starvation (fault light will not illuminate).

AUTOPILOT/FLIGHT DIRECTOR SYSTEM

The Auto Flight System consists of three independent systems: autopilot / flight director (AP/FD) system, yaw damper systems and auto throttle system. The systems provide automatic airplane stabilization about the pitch, roll and yaw axes and control the airplane with selective guidance from radio, compass, GPS navigation, and air data command inputs.

The autopilot system is a two-axis (pitch and roll) system which operates the elevators and ailerons to automatically maintain altitude, airspeed and/or guide the airplane to designated locations and make automatic landings. Control functions are also translated into flight director commands for display on the pilots' attitude director indicators (ADI's), thereby providing the pilots flight attitude commands during manual operation or allowing the pilots to monitor autopilot operation. Automatic stabilizer trimming relieves sustained elevator loads which might be incurred due to fuel burnoff.

The yaw damper systems operate the rudders to correct any periodic yaw oscillations (dutch roll) and assist in making coordinated turns.

The autothrottle system automatically maintains selected airspeeds and assists the autopilot when making automatic landings by adjusting engine thrust levers.

AUTOPILOT/ FLIGHT DIRECTOR SYSTEM

The AP/FD system incorporates three separate channels (A, B and C). Each channel controls the pitch and roll axes. Channel A, B and C flight director command signals and flag logic (for the pilots' ADI's) are selected with computer select switches. Flight director commands and flag logic are connected to the Captain's ADI or the First Officer's ADI with two flight director switches on the AP/FD mode select panel.

Autopilot control of the airplane is enabled by engaging channels A or B switches on the AP/FD mode select panel. Channels A and B can be engaged simultaneously only when making automatic landings. Channel C is a flight director channel and cannot be engaged as an autopilot channel.

Each AP/FD channel consists of pitch and roll computers. Accessory boxes and a monitor and logic unit interconnect the auto flight systems. An AP/FD mode select panel controls autopilot/flight director systems and autothrottle system operation.

The automatic stabilizer trim system is a dual channel system with two computation sections (channel A and B) in one unit. The system is operational only when channel A or channel B autopilot system is engaged. A stabilizer trim interface unit contains relays which connect channel A or B stabilizer trim discrete control signals to the stabilizer trim control module and associated hydraulic actuators when channel A or B autopilot system is engaged. The stabilizer is then trimmed by hydraulic actuators.

Autopilot/flight director disengage warning consists of autopilot warning lights and wailer operation when certain autopilot functions are not correct and the autopilot is disengaged. ADI pitch and/or roll bars are biased out of view and/or the flight director flag (or computer flag) is biased into view as an indication of certain flight director malfunctions.

The Sperry SPZ-1 fail operational autopilot/flight director (AP/FD) system is an integrated autopilot and flight director system using common computational components. The AP/FD system provides three independent F/D channels and two A/P channels. Any one or any combination of the two A/P channels as selected on an

AP/FD mode select panel can control the airplane roll and pitch axes control surfaces to selected automatic path guidance commands. A flight controller with pitch and turn knobs is provided for manual autopilot guidance. Any one of the three F/D channels can be selected by the Captain or First Officer for flight director guidance commands as indicated on Attitude Director Indicators (ADI's).

Two independent Bendix yaw damper systems provide full yaw axis stability control. The upper yaw damper system controls the upper rudder and the lower yaw damper system controls the lower rudder. Series yaw damping provides for yaw damper units independent of pilot input to the rudder.

Independence of each AP/FD channel and yaw damper system is assured by isolation of power supplies, sensors, computers, aircraft wire bundles and shelf wire harnesses.

Two hydraulic systems supply pressure for the elevator power control units (PCU's) and aileron central control actuators (CCA's) which are controlled by the autopilot channels. Two hydraulic systems supply the rudder PCU's controlled by the yaw damper systems.

Output of the autopilot systems control electrohydraulic transfer valves on autopilot modules which are part of the PCU's and CCA's. Output of the modules mechanically control the hydraulic actuators through linkage which is balanced against spring-loaded detents in each module. During dual channel approaches, control surface commands generated by each A/P control module are force-summed into a non-jammable mechanical least value voter. The mechanical voter linkage is balanced against the feel unit; failure of one channel is force-summed by the linkage to provide a least value movement of the control surface. A failure of any component is balanced out by the mechanical voter thereby preventing unwanted control surface movement.

During single channel or dual channel A/P operation, the pilot can override any unwanted commands without disengaging the A/P system(s) by moving his control wheel/column with sufficient pressure to cam-out the A/P module(s). Pressure transmitted through the control cables and feel unit causes the spring-loaded detents to unlock the mechanical linkage of the A/P module for as long as the pressure is held above the detent range. Releasing control wheel/column pressure allows the linkage to be locked by the spring-loaded detent and operation returns to normal. This same cam-out action is what allows the two A/P modules to operate together and operate as a least value voter during dual channel operation.

During dual channel operation, electronic circuits detect cam-out action and automatically illuminate the faulty channel warning light. Additional detection circuits in each system monitor system sensors and system internal circuits. Additional warnings are provided for specific failures. Specific logic valid failures may disengage any of the channels at any time (such as loss of attitude valid or electrical power). In-flight failures during dual channel operation are indicated by latching type maintenance monitor indicators on the front panel of the monitor and logic unit (MLU).

One channel of a dual channel stabilizer trim system automatically trims the horizontal stabilizer whenever at least one A/P channel is engaged. The second channel remains in standby and is only connected in the event of a failure of the first channel when both A/P channels are engaged. One channel is an automatic stabilizer trim unit (ASTU) drives the stabilizer hydraulic motors (through a stabilizer trim control module) with discrete control voltages whenever sensing circuits on the feel pressure unit and

elevator PCU's exceed a predetermined level for a given level of time. Repositioning the stabilizer relieves sustained elevator loads which may have been caused by fuel burnoff, etc. Failure detection circuits within the ASTU disconnect the faulty ASTU channel discrete drive circuits if the stabilizer mistrims (or fails to trim) and provides a warning light indication. The second channel is automatically connected to the system thereby providing fail operational capability during dual channel operation.
Each of the Sperry components contain built-in test equipment (BITE) which aid in testing each box during ground maintenance. Operation of a box under test is indicated by pass or fail lights on the front panel. The Bendix yaw damper computers also contain self-test circuits which illuminate a pass light on the front of the computer. Each yaw damper system can also be tested with a confidence test switch on the yaw damper systems control panel. The confidence test is checked by observing the rudder position indicators.

AP/FD AND YAW DAMPER COMPONENTS

The following components are the major units within the system:

1. One AP/FD mode select panel (MSP)
2. One flight controller
3. Two flight mode annunciators
4. Three pitch computers
5. Three roll computers
6. One monitor and logic unit (MLU)
7. Two autopilot accessory boxes
8. Two normal accelerometers
9. One automatic stabilizer trim unit (ASTU)
10. One stabilizer trim interface unit (STIU)
11. Two F/D computer select switches
12. Two F/D computer select relays
13. One yaw damper control panel
14. Two yaw damper computers

AP/FD MODE SELECTOR PANEL

The AP/FD mode selector panel (MSP) is centered on the lightshield section above the instrument panels. The MSP contains all switches required for autopilot and flight director mode selection, flight director operation and autopilot engagement.
The MSP contains two solenoid held-three position A/P engage switches, two flight director switches, two F/D pitch trim wheels, one auto throttle switch with speed control and indicator, two course select controls with indicators, a three-positioned solenoid-held course select switch, a rotary five-position mode select switch, a solenoid-held back beam switch, an altitude select control with indicator, a three-position solenoid-held altitude switch, and a three-position solenoid-held TURB/SPEED switch. Associated green indicator lights illuminate when the auto throttle switch is on, back beam is on, or the altitude switch is positioned to ALT SEL or ALT HOLD.

Each course select control has two control synchros (referenced from the two compass systems) which provide course error data to each AP/FD channel, a resolver connected to an associated VHF system for VOR reference and one differential sychro which provides course data to the HSI's. The heading control has two control synchros (referenced from the two compass systems) which provide heading error data to each AP/FD and two differential synchros which provide heading data to the HSI's. The indicators read from 000 to 359. The course select switch controls logic circuits in the roll and pitch computers which connect either VHF navigation receivers No. 1 (No. 3) or No. 2 (No. 3) VOR/LOC deviation signals and course error signals into the system control circuits. The course select switch is solenoid-held in the No. 1 and No. 2 positions and returns to the dual position when ILS or LAND is selected.

Each autopilot engage switch is spring loaded and locked in the OFF position until basic interlock requirements are satisfied. Associated channel power and attitude references present. The switch is solenoid-held in the MAN and COMMAND positions. Certain interlock requirements must be satisfied to keep the switch in either position during selected operational modes. In all modes, except when LAND is selected on the mode select switch, each engage switch energizes roll and pitch hydraulic solenoids in the hydraulic control packages whenever engaged to MAN or COMMAND. The control surfaces are then controlled by the engaged channel. When LAND is selected, the switch engaged first is the control (even though both switches may be in the COMMAND position) until specific self-tests and interlocks have been satisfied, then the remaining channel hydraulics are engaged and both hydraulic control packages are operational.

The mode select switch has five positions: GPS, HDG, VOR/LOC, ILS and LAND. The GPS, HDG and VOR/LOC positions provide navigation input control to the roll computers. When GPS is selected, the AP/FD channels provide guidance to capture and track the airplane over a course selected on the associated GPS system. When HDG is selected, the AP/FD channels provide guidance referenced to a selected heading on the mode select panel. When VOR/LOC is selected, the AP/FD channels provide guidance referenced to a VOR radial if a VOR frequency is selected or to a localizer course if a localizer frequency is selected. The ILS and LAND positions control localizer and glide slope signals and inter-lock logic to the pitch and roll computers and the monitor and logic unit. Only one channel can be engaged at a time in all of the positions except land. More than one channel can be engaged after LAND is selected to enable a fail-passive automatic landing approach with flare or F/D go-around.

The back beam switch is solenoid-held and can be used only for flight director guidance with manual control of the airplane when VOR/LOC is selected. The back beam switch reverses summed course and navigation receiver localizer deviation signals in each roll computer to provide the correct roll bar indications on the ADI when flying a back beam approach.

The TURB/SPEED switch is spring loaded to the OFF position and solenoid-held in the TURB and IAS positions. Selections of IAS causes indicated airspeed to be used as the reference to pitch attitude control in the pitch computers. When TURB is selected, gains in the pitch and roll computers are reduced approximately 50% to provide smoother airplane guidance.

The altitude switch is spring loaded to the OFF position and solenoid-held in the ALT HOLD and ALT SEL positions. When in ALT HOLD, a clutch in the CADC is engaged on an altitude reference synchro and the pitch channels are controlled by an altitude error reference. When in ALT SEL, the altitude selected with the altitude select control is used as a reference. The pitch channels are controlled by indicated airspeed until within the altitude select capture range at which time the airplane is pitched over or flared onto the selected altitude and an altitude select hold mode is assumed.
The altitude select control consists of two differential transformers (one using coarse altitude, the other, fine altitude references from the CADC's) and a five-digit indicator (with the last two digits fixed at zero) which indicates in hundreds of feet.
The flight director switch is either ON or OFF with no holding devices. When positioned to ON, pitch and roll commands and F/D flag warning circuits are completed from the selected channel to the associated ADI. Any of the modes selected on the nav mode select switch, altitude switch, TURB/SPEED switch and back beam switch may be used for flight reference. The pitch trim wheel adjacent to the FLT DIR switch operates a potentiometer which establishes a pitch attitude hold reference on the associated ADI when the flight director is on and an autopilot engage switch is not positioned to COMMAND.

FLIGHT CONTROLLER

The flight controller is installed on the aft electronic section of the control stand. A turn knob and two pitch wheels on the controller provide attitude commands proportional to their position during manual control of the engaged A/P channel. The controller is normally used when one A/P channel is in MAN. The turn knob is mechanically detented in the center position and drives potentiometers when rotated out of detent. Detent switches complete A/P engage interlock circuits. The pitch wheels are on a common shaft and are clutch engaged to two potentiometers when an A/P channel is engaged to MAN or (if a pitch mode is not selected) to COMMAND. When a pitch mode such as ALT HOLD is selected, the clutch is disengaged and the potentiometers clamped in position with a brake; rotation of a pitch wheel does not affect the system. One potentiometer on the turn knob and one on the pitch wheel are connected to channel B. The remaining two turn and pitch potentiometers are connected to channel A.

FLIGHT PROGRESS DISPLAY PANEL

One AP/FD Flight Progress Display panel is installed on P1 panel and one on P3. Each panel is composed of two sections: one section provides F/D flight progress annunciation, the other A/P warning and flight progress annunciation. An auto throttle warning light is on the left side of the annunciator panel.
The flight director annunciators on the left side are: ALT SEL, NAV, G/S, FLARE and GO-AROUND. The autopilot annunciators on the right side are: ALT SEL, NAV, G!S, and FLARE. One A/P warning light is on the right. One AUTO THROT warning light is on the left. The warning lights can be pressed to reset the associated system warning circuits and extinguish the light. Each warning light illuminates either amber or red depending upon the seriousness of the condition. Each annunciator light

illuminates amber when the associated mode is armed. At capture or engage, the annunciator illuminates green. Pressing the left F/D annunciator panel test and illuminates all amber lights. Pressing the right A/P annunciator panel tests and illuminates all red and green lights.
Two photocells provide variable illumination in proportion to ambient light conditions. The photocells are connected to dimming circuits in the accessory boxes. Each annunciator is controlled by grounding one side of an annunciator bulb at the roll and/or pitch computers and/or MLU. The bulbs can be replaced by pulling the cap assembly straight out from the base, thereby exposing the bulbs and filter assembly. The filter can then be removed and placed on a new bulb, the bulb inserted and the cap assembly pressed back into position.

YAW DAMPER SYSTEM

Two identical yaw damper systems control the upper and lower rudders. Each system monitors airplane yaw rate and positions the rudder to compensate for periodic yaw oscillations (dutch roll). Correction signals are applied to the rudder package during manual and autopilot controlled flight to displace the upper and lower rudders sufficiently to damp out any yaw oscillations of the airplane. Rudder displacement is limited to 3.6 degrees. The yaw damper system also provides a turn coordination feature which improves airplane response during turn maneuvers when the flaps are down at least 1 degree. System gains are also changed as a function of flap position. When the flaps are down, the roll attitude signal from the DG/VG is introduced to provide rudder displacement proportional to roll rate. The roll attitude signal is not used when the flaps are up. The yaw damper system is normally engaged for all flight modes and operates full time.
Each system uses the following components: one engage switch and a test switch (on yaw damper control panel), yaw damper computer, and a transfer valve, engage solenoid valve and LVDT (linear variable differential transducer) installed as part of the rudder power control unit.
Control and test of each yaw damper system is accomplished from the flight compartment by means of two engage and two confidence test switches (one for each channel) located on the yaw damper control panel. The Captain's and First Officer's rate-of-turn indicators use signals provided by a rate gyro in the yaw damper computer.
The upper rudder yaw damper system receives 115-volt AC and 28-volt DC power from the Essential Flight Instrument bus; the lower rudder yaw damper system receives power from number 2 Flight Instrument bus on circuit breaker panel P7.

YAW DAMPER CONTROL PANEL

Yaw Dampers automatically position rudders to damp out any yaw tendency and improve directional stability. the yaw damper incorporates an additional feature (turn coordination) which deflects the rudders an amount proportional to the roll rate to improve roll control response in a "flaps down" configuration. The turn coordination feature is locked out when the flaps are up.

The yaw damper control panel is on the pilots' overhead panel. The control panel contains two confidence test switches and two guarded engage switches. The confidence test switches are three-position toggle switches spring loaded to the center position. The engage switches are two-position toggle switches guarded to the ENGAGE position.

AUTOTHROTTLE SYSTEM

The autothrottle system automatically moves all four thrust levers, in response to preselected airspeed commands, thereby causing the airplane to acquire and maintain the selected airspeed. The system is normally used during approach maneuvering and the landing phases of flight. If two autopilot channels are engaged, the autothrottle automatically retards the thrust levers as the airplane descends through the final 30 feet of the flare maneuver. The pilot can manually override autothrottle thrust lever positioning. Clutches between the thrust levers and autothrottle drive mechanism drive the thrust levers when pressure is applied from the drive mechanism. The clutches slip when a small amount of pressure is applied directly to the thrust lever handles.

The autothrottle system is engaged with a switch on the autopilot/flight director (AP/FD) mode select panel. Reference airspeeds are selected with a rotary control on the AP/FD mode select panel. A digital readout ranging from 101 to 259 knots is presented in a window above the control and by a servo-driven bug on the ASI's. Small overspeed and underspeed indications are provided by a fast-slow pointer on the Captain's and First Officer's attitude director indicators (ADI's). Pitch attitude signals used by the autothrottle computer are obtained from DG/VG No. 2. Flare logic is obtained from the AP/FD systems. Altitude data is obtained from low range radio altimeter No. 2. Limit switches installed on a microswitch assembly on the forward end of the control stand (P8) monitor thrust lever position and automatically stop autothrottle operation. The switches prevent the autothrottle system from driving more than one thrust lever to the maximum limit or more than two levers to the minimum limit. The autothrottle system drives the thrust levers out of the limit when the selected airspeed is correspondingly decreased or increased. The autothrottle system is disengaged by any of the following methods: positioning the engage switch to OFF, pressing one of the disconnect levers (switches) on thrust lever No. 1 or 4 or pressing one of the go-around levers (switches) on thrust lever No. 2 or 3, or when autothrottle flare logic or radio altimeter valid logic is lost. An autothrottle warning light on the Captain's and First Officer's flight progress displays flash red whenever the system is disengaged. The autothrottle warning lights also illuminate amber if the indicated airspeed exceeds 10 knots from the selected reference airspeed. Pressing the annunciator extinguishes the lights and resets the warning circuits.

Autothrottle system 115-volt AC and 28-volt DC power is from two AUTO THROT AC, DC circuit breakers on circuit breaker panel P7. Power is from Flight Instrument bus No. 2. Battery bus power for the autothrottle red warning lights is from the 28-volt AFC WARN DC circuit breaker on circuit breaker panel P6.

ANNUNCIATOR TEST

The flight mode annunciators (FMA) and warning lights are tested by pressing the PTT 1 and 2 switches. Pressing the PTT 1 switch tests all armed annunciators or amber lights (as installed). Pressing the PTT 2 switch tests the associated captured annunciators or red lights (as installed). Photoelectric cells automatically adjust the progress annunciator light brightness for the surrounding light conditions. This adjustment occurs only when the indicator light dim/test switch on the pilots' overhead panel is in the DIM position

Electrical

The 747 electrical power system is basically a 3-phase, 400-Hz, 115/200-volt, generating and distribution system. Single-phase transformers are used to reduce portion of this power to 28-volts AC. Transformer rectifiers, fed from the 3-phase system are used to furnish 28 volts DC. A battery is installed to supply emergency DC power to certain critical loads when the basic source is de-energized. A DC driven static inventor is provided to supply 115-volt AC single-phase power for critical flight items when the normal supply is not available.

GENERATOR

Four brushless generators, mounted one on each engine, provide the primary electrical power supply which is a 115l200 volt, 3-phase, 400 Hz constant frequency alternating current system. Each generator is rated at 60 KVA. Air taken from the engine fan exhaust, is used as the cooling medium, which is then exhausted overboard. A fan on the generator rotor shaft helps to force the cooling air through the generator. The generators are made up of 3 separate units, a permanent magnet generator (PMG), an exciter generator, and the main AC generator, all contained in the same housing. Output of the PMG is used as field current for the exciter and operating current for the generator control panel. The battery bus provides a backup source of current for operation of the generator control panel only. The generators, whether mounted on the engine or APU, are identical. The APU-driven generators which have a greater supply of cooling air are rated at 90 kva. The output of a generator is connected to its bus system by a generator breaker. Bus tie breakers join each generator system to the synchronization bus. Under normal operating conditions the generators are paralleled, thus sharing the total electrical load. This is accomplished by the closing of the generator breakers. A generator and bus system can be unparalleled by opening the bus tie breaker. A split system breaker, when open, cuts the sync bus into two parts allowing isolation when malfunctions occur.

CONSTANT SPEED DRIVE (CSD)

Generator speed is maintained at 8000 RPM by means of a hydro-mechanical constant speed drive unit, driven by the N, compressor via the accessory gear box, providing a constant frequency of 400 Hz, regardless of engine speed between ground idle and maximum thrust. In case of malfunctions in the drive unit or generator, the input drive shaft can be disconnected from the accessory drive. The drive unit has an integral reservoir, a sight gauge, a low oil pressure switch, temperature bulbs, speed switches, and an external oil cooler which is affected by engine fan exhaust air. Oil level in the sight gauge should be in the green area. The amber low oil pressure light will be illuminated when the pressure drops below a predetermined value. Temperature normally illustrated by the gauge is the outlet temperature, which is the temperature of the oil as it leaves the hydraulic package and is on its way to the oil cooler. To read the "Temperature Rise", a button to the right of #4 CSD oil temperature gauge must be pressed. The gauges will then indicate the difference between the inlet and outlet oil temperatures. After disconnecting a CSD from its drive, it cannot be reconnected until the aircraft is on the ground and the engine is stationary. An overfrequency will cause a trip of the generator field relay, and an under frequency will trip only the generator breaker.

GENERATOR CONTROL PANEL

Four combined voltage regulator and control panels are located in the lower equipment center and contain circuits to control the output voltage of the generators, fault protection and generator control. Power to operate the control circuits is obtained from the respective PMG via a integral TRU. Backup power from the battery bus is available when the PMG is not available.

Protection is provided for the following:

1. Underfrequency - Trips the generator breaker.
2. Overfrequency - Trips the generator field relay.
3. Undervoltage - Trips the generator field relay.
4. Overvoltage - Trips the generator field relay.
5. Feeder Faults (open phases and short circuits) - Trips the generator field relays.
6. Sync Bus Short Circuits - Trips the SSB and 2 respective bus tie breakers.
7. Imbalance between generators - Trips the respective bus tie breaker.

An engine fire handle, when pulled, will trip the respective generator field relay.

LOAD CONTROLLER

The load controller controls the constant speed drives so that the real loads will be evenly distributed between paralleled generators. The load controller also provides a CSD output overspeed signal to the generator control unit and a CSD output speed signal to an indicator on the Flight Engineer's instrument panel. Also contained in the load controllers are speed jogging circuits for paralleling generators. There are four load controllers, one for each engine-driven generator.

FIELD RELAY

The Generator Field Relay is a latch-type relay and, when open, the amber "FIELD OFF" light will be illuminated. When closed, field current is applied to the exciter portion of the generator and the amber light will be extinguished. Output of the generator is zero when the field relay is open.

Control of the field relay is accomplished by means of the three-position switch located just to the right of the "FIELD OFF" tight. The switch is of the momentary type, center being the normal position. To close the relay, regardless of why it is open, momentarily hold the switch in the "Close" position. If a fault exists, the relay will trip again after satisfying the required time delay. When a field relay is tripped open, a circuit will be completed to trip the associated generator breaker. Manual tripping is done by momentarily holding the switch in the "Trip" position. Pulling an engine fire handle will also trip the respective field relay. Various faults will trip a field relay automatically after time delays have been satisfied.

The APU generator field relays are identical in operation to those used by the engine driven generators. Pulling the APU fire switch will trip both APU generator field relays.

Providing external power is not connected, the closing of #1 APU generator field relay will apply power to the ground handling bus, and the #2 will supply power to the main deck cargo handling bus. The field relays normally remain closed when the APU is shut down.

GENERATOR BREAKER

The generator breaker (GB) is a latch-type relay and, when open, disconnects generator output from its bus system. A "GEN OFF" light , amber in color, will be illuminated when the breaker is open. A three position switch controls the opening and closing of the GB. Paralleling is accomplished by momentarily holding the generator breaker switch in the "Close" position. Auto parallel protection is built into the close circuits, preventing a generator being paralleled when phase conditions are unfavorable.

The generator breaker will trip open when its generator field relay is opened. The breaker switch must be momentarily held in the "Close" position before the breaker will close (it will not close automatically). The generator breakers will trip when an auxiliary power breaker is closed, and conversely, the closing of a generator breaker will cause the auxiliary power breakers to trip.

BUS TIE BREAKER

The bus tie breaker (BTB) is a latch-type relay which is used to connect the output of a generator to the sync bus. It will be tripped open by generator imbalance and short circuit type faults. An amber "BUS TIE OPEN" light will illuminate when the BTB is open. The bus tie breaker is normally closed by momentarily holding the switch in the "Close" position, but under an abnormal condition where the BTB has been tripped, followed by a GB trip, the BTB will close automatically to ensure that power is available from the sync bus for the would-be dead load busses (providing no other malfunctions are sensed). Manual tripping is accomplished by momentarily holding the switch in the "Trip" position.

DISTRIBUTION

AC busses one through four are located on upper P6, aft of the Flight Engineer. These busses are supplied by the respective generator via its GB, or from the sync bus through the respective BTB.

ESS AC BUS

The ESS AC bus is supplied under normal operating conditions from AC bus four. A rotary type switch allows Ess AC to be taken from any one of the three generators, #1, 2, and 3 from a point between the generator and the generator breaker. Thus it can be seen that if Ess AC is selected to generators #1, 2 and 3, the respective generator breaker does not have to be closed. There are protective circuits to prevent undesirable power being connected to the Ess AC bus. The amber "ESS BUS OFF" light will be illuminated whenever the Ess AC bus is dead. The Ess AC bus is located on upper P6, aft of the Flight Engineer's station.

STANDBY AC

Standby AC is normally supplied from the Ess AC bus, but under emergency conditions (loss of essential AC) it can be obtained from the standby invertor which is powered from the battery bus. The standby AC bus is located on upper P6.
The standby busses supply the following systems:

ADI (Attitude Excitation)	Interphone
HIS No. 1	PA
RMI No. 2	Standby Ignition
VHF No. 1	EGT
VOR No. 1	Ni & Nz
LOC No. 1	Nacelle Anti-Ice
GS No. 1	Standby Attitude Indicator

BATTERIES

The main battery and the APU battery are nickel cadmium and are identical and interchangeable. The main battery is located on the floor behind the Flight Engineer's instrument panel. The APU battery is located in the tail section of the airplane between the aft pressurized bulkhead and the APU firewall.

BATTERY CHARGER

Independent battery chargers maintain the charge in the two batteries at all times that the 115V AC ground service bus is powered. The battery chargers are capable of charging a completely discharged battery to a full charge in one hour. Initial charge will produce a large positive deflection of the needle in the DC ammeter which will decrease as the battery approaches a full charge. After the battery reaches full charge, the charger will maintain a constant potential mode of sufficient amplitude to sustain the state of charge.
Integral to the batteries is a thermal switch to protect the battery from overheating during charging. The thermal switch will disconnect the charger from the ground service bus when overheat is sensed.

TRANSFORMER RECTIFIER

All power on the airplane is generated as alternating current. A portion of this is converted to direct current by means of transformer rectifiers. Seven transformer rectifier units are used. The transformers are identified as the essential TR, TR #1, TR #2, TR #3, ground handling TR, main deck cargo handling TR, and external power TR. Each of the units receives AC power from a similarly identified 115 volt AC bus and delivers 28 volts DC to a similarly identified DC bus.

ESSENTIAL DC BUS

The Essential DC Bus is powered by the Essential AC Bus through the Essential TR or by DC Buses through the DC Bus Isolation.

DC BUS ISOLATION

Isolation relays are installed to provide for isolation of a DC bus or busses, as desired. DC busses 1, 2, 3, and essential may be operated in parallel. Automatic isolation of busses 1 & 2 from bus 3 and essential is provided when the essential AC power switch is moved from the NORMAL position to the 3, 2, or 1 position. In this situation, the No. 3 DC bus isolation relays opens (the switch does not move) and the associated OPEN light illuminates.

STANDBY DC

The DC standby system is used to supply 28 volts DC to systems required to maintain safe flight. Under normal operation the 28 volt DC standby bus receives its power from the essential bus and the battery bus is powered from the essential DC bus. If airplane power fails, the essential DC bus is disconnected from the battery bus and the hot battery bus is connected to the battery bus. The 28 volt DC standby bus is then powered from the battery bus. By this arrangement the hot battery bus, the battery bus and the 28 volt DC standby bus are all powered from the main battery.

RADIO BUS

The radio master busses located on P7 (above the O'HD switch panel) supply power to all radio equipment, except those listed on the standby bus. The ESS radio bus will power the remaining #1 radio equipment, the radio bus #2 will power all #2 radio equipment. Power to the radio busses is controlled by switches at the top right-hand corner of the overhead panel.

GROUND SERVICE BUS

The ground service bus is normally powered from AC bus #1, but can be selected to a #1 auxiliary power source if it is available. Systems powered are such things as service outlets, ground service lighting and the battery chargers. The ground service bus is located in the lower equipment center.

GROUND HANDLING BUS

A ground handling bus supplies power to such things as the #4 electric hydraulic pump, the flight deck fan, service lighting, refuel valves and the lower compartments cargo loading equipment. A #1 auxiliary power source must be available and the aircraft must be on the ground. External power #1 has priority to power this bus when it is available. No switch selections have to be made on the flight deck. A 20 amp TRU is installed to provide 28 volts DC to the DC ground handling buss. The TRU will be operating providing the AC ground handling buss is powered. The CB's are located in the lower equipment center.

MAIN DECK CARGO HANDLING BUS

The main deck cargo handling bus is provided to power cargo handling equipment on the main deck plus the main cargo door. A #2 auxiliary power source must be available and the aircraft must be on the ground. External power #2 has priority, providing it is plugged in. No switch selections have to be made on the flight deck. A 20 amp TRU powered from the main deck cargo handling AC bus supplies power to the main deck cargo handling DC bus.

GALLEY POWER BUSSES

The galley busses are supplied power from the main airplane AC busses. The galley feeder circuit protection system is designed to trip off the galleys and illuminate appropriate galley TRIP OFF lights when:

The current demand from an APU generator or an external power source exceeds a safe value by:

- There is an electrical fault in the feeder circuit.
- The galley feeder current exceeds a safe value.

, NOTE: The protective system is reset by cycling the appropriate galley power switches.

During APU starter engagement with APU start selector switch in TRU position, power to the aft galley (bus 3) will be transferred to the APU start T-R unit. The No. 3 galley feeder circuit protection system then protects the APU start feeders and a fault will be indicated by steady illumination of No. 3 galley power TRIP OFF light.

AUXILIARY POWER

Auxiliary power is supplied by external power and an APU. Two sources of external power can be connected to the sync bus simultaneously, providing the split system breaker is open. One external power source can serve the entire sync bus when the split system breaker (SSB) is closed. The two generators on the APU are rated 90 KVA each and can be connected to the sync bus in the same manner as external power.

Fire Protection

The fire, smoke, or overheat detection circuits give the flight crew visual and/or aural indications of abnormal conditions in the engine, APU, main deck cargo compartment, lower cargo compartments, wing leading edge, or wheel wells. All systems are operative when aircraft power is available.

ENGINE FIRE DETECTION

Fire detection includes four dual-element sensors (in series) on each engine, an engine fire detection control panel on P4, fire detection card file aft side of P6, fire warning bell under the flight engineer's table, four fire switches on the overhead panel P5, two master warning lights on the glareshield, a repeater fault warning light on the pilot's annunciator panel on the P2 panel. The 28 volt DC battery bus supplies power for system operation.

Four dual-element, series-connected, heat-sensitive sensors are placed around each engine. This provides two parallel detection loops, loop A and loop B. Sensor resistance to ground is monitored by fire detection control circuits indicating system status to the flight crew. Under normal conditions both loops are employed and signals from either or both loops will produce a fire signal. However, if a loop is faulty it can be disabled by switches on the flight engineer's panel and a single loop used to operate the system.

A fire condition causes the fire switch light and the master fire warning lights to illuminate, the fire warning bell to sound, and both nacelle temperature indicators should be in the red band area. The fire warning bell and master fire warning lights may be turned off by pressing the bell reset switch (P4) or by pressing either master fire warning lights. A fault (short or sudden low resistance to ground) causes the nacelle fire detector fault light (P4) and the repeater fault light (P2) to illuminate. Individual loops including fire and fault circuitry can be tested by operation of switches on the flight engineer's fire detection panel.

SENSING ELEMENTS

Four dual-element sensor assemblies are placed around each engine so that if an engine fire did occur, the heat from the fire would impinge on at least one sensing element assembly. The dual-element sensor assemblies are located on the upper forward, upper aft, lower forward and lower aft areas of each engine.

ENGINE NACELLE TEMPERATURE INDICATORS

The engine nacelle temperature indicators are installed above the fire detection control panel on the flight engineer's instrument panel. The indicators receive sensing-element outputs and continuously display the temperature trend of the engine nacelles. Each of the four indicators has two independent channels designated channel A and channel B. Each channel is completely independent of the other in its operation. A malfunction of one channel does not affect the operation of the other channel.

The indicators have fault indication capability. With the engine fire detection system normally on, the pointers will move from their stops and indicate the temperature (within limitations of the scale) existing in the engine nacelle. Abnormal indications for an engine is indicative of either an abnormal temperature condition in the engine or a fault in the detection system. A short circuit will cause the pointer to move to the top extremity of the scale. An open circuit condition will cause the pointer to go to bottom of the scale. Any faulty loop in the system, as indicated by the master fault light (P4) and repeater fire detection light (P2), is easily detected by viewing the temperature indicators.

FIRE DETECTION CONTROL PANEL

The fire detection control panel is located on the flight engineer's upper instrument panel (P4). This panel consists of control switches, a fault indicator light, and test switches. These components provide for the testing of the A and B loops for fire or fault operation. This panel also allows for the disconnection of any faulty loop from the system. The fault indicator lights on the P2 and P4 instrument panels illuminate when a fault condition, operational or test, is encountered.

ENGINE FIRE SWITCH

The engine fire switches are located on the pilots' overhead panel. A fire switch is provided for each engine. Each fire switch contains four red lamps. The fire switch illuminates when a fire condition is detected.

FIRE WARNING BELL AND MASTER FIRE WARNING LIGHT

A fire warning bell is located under the shelf of the flight engineer's console. This bell will sound when any engine, APU, wheel well, or compartment fire (GE aircraft) is detected.

The captain's and first officer's master fire warning lights are located on the glareshield above their individual main instrument panels. These lights are illuminated when any of the engine, APU, wheel well, main deck cargo, or lower cargo compartment fire detection systems detect a fire condition.

The fire warning bell and master fire warning lights may be reset by pressing the BELL-RESET switch on the flight engineer's engine fire detection control panel or by pressing the captain's or first officer's master fire warning light. Reduction in temperature of the sensor will also turn off the fire warning bell and master lights.

GRAVINER FIRE DETECTION SYSTEM (Kidde Basic Airplane)

The engine fire detection system (Kidde) has been designed to accommodate engine pooling. Engines equipped with a Graviner fire detection system may be used at any nacelle position after changing the Loop A and Loop B control cards and ensuring that engine and strut fire detection elements and mating wire bundles are compatible. The card change will make the engine fire detection and warning system operative. However, the Graviner card output to the temperature meter on the P4 panel will drive the temperature indicator to a higher scale position than will the Kidde card outputs. One or more engines equipped with a Graviner fire detection system may be used.

CAUTION: A characteristic of Graviner fire detector elements can result in afire to be sensed and reported as a fault. Therefore, when an engine with Graviner elements is installed on an airplane with Kidde fire detection

system, a double fault (or single fault if one Loop is inoperative) on the Graviner equipped engine "should be treated as a fire."
When the aircraft is configured as mentioned above, the associated engine nacelle temperature indicator(s) will be placarded to inform the crew of fire detection system intermix. The fire test procedure remains unchanged. Fire warning test needle deflection will be slightly lower than that seen while testing the Kidde system. Fault test needle deflection will be above top of scale or "peg out" when tested.

NOTE: The indicator placards are informative and are not indicative of what you see during system test. They simply state how you must interpret a single loop fault (1 loop inoperative) or dual loop fault in flight.
When a pool engine with Graviner fire detection sensors is installed, and the Fire Warning Module (P-4 panel) has been modified by SB 26-2050, the fault circuit for that engine is disconnected from the FAULT light and connected to the fire warning, since the Graviner sensor has a failure mode that could make a fire appear as a fault. The NACELLE TEMPERATURE gauge for that engine is placarded INOP or placarded "Graviner System Installed" because of a different response.
During fault test, the engine fire switch for the engine with Graviner sensors illuminates, the fire bell sounds, both pilots' master FIRE lights illuminate, the FAULT light illuminates, and the lights in the other fire switches are out.

GRAVINER FIRE DETECTION SYSTEM (GE Aircraft)

N487EV, N488EV and N489EV have Graviner Fire Detection systems installed and are considered Graviner basic airplanes.
With the loop selector switch set to both a fire warning will occur if both loops sense a fire signal, both loops sense a fault signal, or one loop senses a fire signal and the other loop senses a fault signal.
If only one loop senses a fire or fault condition the fault light will come on. The nacelle temperature indicator will show which loop has the fire or fault condition.

SUMMARY OF FIRE AND FAULT LOGIC

A double fault, or single fault with A or B selected, will result in a fire warning. EIA procedure is to treat a double fault, or single fault with A or B selected as a FIRE.
Nacelle Temperature Indicators - Indicate relative temperature levels in the engine nacelle area. Normal indication will be in the 0-4 range (green band); overheat in the 4-8 range (amber band); fire in the 8-10 range (red band).
With a fire condition existing and fire warning initiated, a detector electrical short will cause the warnings from the affected detector to be locked on. Nacelle temperature indication for the failed detector will be in the 8-10 range (red band) and the nacelle fire detector fault light will remain extinguished.
A single detector electrical open will not affect fire warnings or temperature indications. An open will cause temperature display to be at bottom of the scale.
If selector switch is in both and one has an open and the other senses a fire condition the display would only be fault on P4 and fire detection on P2 panel -

no fire indication.
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Master Fire Warning Lights (Red) (Captain's and First Officer's)

Illuminate and fire warning bell sounds (as indicated) for a fire condition, or during test, for the following areas:

- Engine nacelle. (Bell)
- Main landing gear wheel wells. (Bell on some aircraft)
- APU. (Bell)
- Lower forward and aft/bulk cargo compartments.

Push to extinguish master fire warning lights, silence fire warning bell and reset warning circuits.

Engine Fire Extinguisher Discharge Switches - With engine fire switch pulled, press to discharge indicated fire extinguisher bottle. Switch illuminates amber and remains illuminated to indicate extinguisher bottle discharged. DISCHARGE light does not illuminate when overpressure relief occurs.

Engine Fire Switches (Red) - Illuminate for a nacelle fire condition or during test. With detectors operating fire switches will remain illuminated as long as fire condition exists. Pulling switch:

- Arms fire extinguisher.
- Closes fuel shutoff valve.
- Depressurizes engine driven hydraulic pump and shuts off hydraulic fluid supply.
- Trips generator field after a time delay.
- Closes bleed air pylon valve.

Fire Switch Flag - When fire switch is pulled yellow flag rotates into view and locks switch in pulled position.

ENGINE FIRE EXTINGUISHING

The engine fire extinguishing system is a gaseous smothering system designed to flood the strut area and engine cowling space with an inert gas in case of fire. This system is electrically controlled by engine fire extinguisher discharge switches on the overhead panel (P5), and may be tested by the flight engineer through the use of the squib test switch located on the squib test panel (upper P4). Two identical bottles are installed for each engine, and these may be independently fired and monitored. Freon gas is the extinguishing agent, and nitrogen gas is used to pressurize the bottles.

Two fire extinguishing bottles for each engine are located in the engine nacelle above the firewall. Each bottle contains 6.5 pounds of Freon. The discharge tubes are connected in a y-fitting and a single line then carries the Freon to the fire area. Each engine may receive two applications of Freon if required.

The engine fire switches on the overhead panel (P5) when pulled will arm the engine fire extinguisher discharge switches for discharge of either left or right bottle for each engine as selected by the discharge switch. Operation of the engine fire switch also disconnects fuel and all combustibles from the engine area. A pressure switch in the fire extinguisher line detects the release of Freon pressure and turns on the discharge light (amber).

FIRE EXTINGUISHING BOTTLES AND MANIFOLD ASSEMBLIES

The fire extinguishing bottles are high-rate discharge, nitrogen pressurized cylinder bottles. The fire bottles contain 6.5 pounds of Freon (bromotrifluoromethane). The firing mechanism is a single discharge valve. A diaphragm is welded over the discharge port in the bottle. An electrical squib is provided to ignite a powder charge which in turn ruptures the diaphragm. A pressure switch mounted after the Y-connection from the two bottles senses the discharge of a fire bottle and provides a ground signal which illuminates the amber DISCHARGE light.

Excess pressure in a fire extinguisher bottle causes the opening of a fusible discharge plug, discharging the Freon contents overboard through an overboard vent line. There is no cockpit indication of this occurring. Red overpressure discharge indicators installed on the external surface of the engine strut, two per strut, are blown out in the process. When intact, these are readily seen at ground level.

The fire extinguishing system is powered by the hot battery bus.

SQUIB TEST PANEL

A squib test panel is located on the flight engineer's panel to allow testing of all engine squibs. The panel contains the switch (momentary) for testing the engine left and right squibs and a SQUIB OK light (green) for each engine.

The continuity check of the squib firing circuit includes an electrical check of the squib itself. Approximately 30 ma of current are passed from the switch through the wiring to the squib to ground in order to illuminate the green SQUIB OK light.

APU FIRE DETECTION

The APU fire detection system provides for fire detection in the APU compartment. Dual-Element continuous sensors are used, each capable of independently operating the fire detection system. The flight engineer is provided with fault indicator lights and test switches for fire and fault circuits. The captain and first officer are notified of a fire condition by the master fire warning and APU fire warning light on overhead panel P-5. A fire warning bell alerts the flight crew of a fire condition.

Continuous loop dual-element sensors are placed in the upper and lower areas of the APU compartment. Each sensor has an A and a B loop; that is, two identical sensing elements mounted in parallel. A low-resistance or short circuit fault in a sensor or in the overall circuit will result in a fault indication. A fire condition will cause aural and visual indication of the fire to the cockpit. Additional visual and aural fire warning will occur on the APU remote control panel in the right body gear wheel well.

A fire condition causes the APU fire switch to illuminate on the APU control panel, an APU fire indicator light on the fire warning module of the overhead panel (P-5) illuminates, and the captain's and first officer's master fire warning lights illuminate.

A fire warning bell under the flight engineer's panel provides aural warning and may be reset at the flight engineer's panel by pressing the bell reset switch on the engine fire protection panel, or by pressing either master fire warning light. Fault signals from either loop A or loop B will illuminate the appropriate APU FIRE DIET light.

Testing is comprised of fault tests and fire tests for loops A or B. In fault testing, a fault (short circuit) is simulated to the Loop A or B circuits, as selected, and the respective fault light illuminates. Should the loop A be inoperative, loop B may be used by selecting B.

A fire test introduces a simulated fire signal in the detector system. The master FIRE warning lights, APU fire warning and APU fire switch will illuminate and fire bell will sound. The APU remote fire warning horn will sound and APU remote fire warning light will illuminate.

A detected fire condition causes the APU shutdown relay to be energized: shutting down the APU and isolating it by shutoff of fuel supply. A fire indication also causes a red FIRE light to illuminate on the APU fire shutdown module (wheel well). If the fire switch is not pulled, and the landing gear is extended, a warning horn sounds in the wheel well area.

SENSING ELEMENTS

Two dual-element sensor assemblies are placed in the APU compartment so that if an APU fire did occur, the heat from the fire would impinge on at least one sensing element assembly. The dual-element sensor assemblies are located on the APU compartment access door and near the upper aft APU mount.

APU CONTROL PANEL

The APU Control panel is located on the Flight Engineer's panel. This panel has control switches and indicators to allow testing of the A and B loop for fault or fire operation. The fault indicators on this panel will light amber when a fault condition, operational or test, is encountered.

A FIRE TEST - FAULT TEST switch is provided for both loops A and B. These switches condition the dual-element sensors for either test. APU FIRE DETECTOR selector switch allows loop A, B or both to be selected.

The APU fire switch is located on the APU control panel and contains four red light bulbs. Two of these bulbs, one upper and one lower, are connected to loop A. The other two are connected to loop B. Selection of A or B loops on the APU fire panel connects all lights in the APU fire switch to the selected loop.

A relay, located in the control panel, isolates fire test signals from APU Autofire shutdown circuitry. Actuation of either test switch will energize the relay and the relay will remain energized as long as the fire test signal is present.

The Flight Engineer's APU control module has a fire switch to arm the extinguisher for discharge of the bottle. Operating the fire switch also disconnects fuel and all combustibles from the APU, and causes shutdown of the APU. A discharge switch causes discharge of the fire bottle. A pressure switch detects firing of the bottle and causes the discharge pushbutton switch/indicator to illuminate providing visual indication to the flight engineer.

A squib test switch allows testing of the squib circuit for continuity. An operative squib circuit results in a green SQUIB OK light.

APU REMOTE CONTROL PANEL

An APU remote control panel is mounted in the right body wheel well. This panel

allows ground personnel to be alerted to a fire condition in the APU. This panel has a red fire light, a fire warning horn, remote APU stop switch, fire switch, and a switch to discharge the APU fire extinguisher.
A relay in the landing gear area disables the wheel well horn circuit when the landing gear is not extended. The wheel well horn is disabled when the APU control panel fire switch is pulled, or by operating the fire switch on the APU remote control panel.

APU FIRE EXTINGUISHING

The APU fire extinguishing system is a gaseous smothering system designed to flood the auxiliary power unit with an inert gas in case of fire. This system is electrically controlled by switches located on the APU control panel on the engineer panel and the APU remote control panel in the right hand body gear wheel well, and may be tested by the use of squib test switch on the APU control panel. Freon gas is the extinguishing agent, and nitrogen gas is used to pressurize the bottle.
The fire extinguishing bottle is located on the forward side of the APU fire wall. The bottle contains 18 pounds of Freon. A single line carries the Freon to the APU shroud.
Excess pressure in the fire extinguishing bottle causes the opening of a fusible discharge plug discharging the Freon overboard through a vent tine. There is no cockpit indication of this occurring. A red over pressure discharge indicator located under the bottle on the lower surface of the fuselage is blown out in the process. When intact the red discharge indicator is readily seen at ground level.
The fire extinguishing system is powered by the hot battery bus.

LOWER CARGO COMPARTMENT SMOKE DETECTION

A smoke detection system monitors the pressurized forward and aft cargo compartments for smoke, which is indicative of a fire condition. If smoke is detected by one or several detectors, the Flight Engineer is notified by warning lights. The Captain and First Officer have visual indication of any fire condition at their master fire lights, and of a cargo compartment fire on the fire warning panel of the pilots' overhead panel (P5).
The lower cargo smoke detector system may be tested at the flight engineer's instrument panel. A test switch is provided for each smoke detector. Operating a test switch causes a smoke condition to be simulated if all circuits are operative.
The lower cargo smoke detector system is comprised of six smoke detector modules, two in the forward and four in the aft compartment and are powered by the essential DC bus.

SMOKE DETECTORS

Each lower cargo compartment smoke detector is mounted in a tray which collects the air for sampling. The smoke detector is mounted in a protective case. The detector consists of a photoelectric cell, a beacon lamp, a test lamp, and a light trap. When the air reaches 10 percent saturation of smoke, light is reflected from the beacon lamp to the photoelectric cell, resulting in a smoke detection signal.
The test lamp is in series with the beacon lamp and mounted opposite the photoelectric cell. When testing, both lamps, the photoelectric cell, and all other

circuits must be operational to get a favorable test result.

LOWER CARGO FIRE PROTECTION PANEL

Control and monitoring of the lower cargo smoke detector system is accomplished at the lower cargo smoke detector panel on the Flight Engineer's instrument panel. This panel has a test switch to activate each smoke detector, and indicator lights to notify the flight engineer of a smoke condition in the forward or aft or bulk (N477, N478, and GE aircraft) cargo compartment.

A cargo compartment fire indicator light is mounted on the fire warning panel of the pilots' overhead panel. This indicator illuminates when a smoke condition exists in a lower cargo compartment.

The detected forward, aft or bulk (N477, N478, and GE aircraft) smoke condition causes activation of the common fire warning circuits, which consist of the Captain's and First Officer's master fire warning lights. This circuit is reset by reduction in smoke in the area or by the Flight Engineer's bell reset switch, or by pressing the Captain's or First Officer's master fire warning light. On GE aircraft, the fire warning bell also sounds.

The lower cargo compartment smoke detection system is turned on automatically when 28-volt DC is available from the essential DC bus. Smoke detection will turn on the CARGO indicator light on the fire warning panel (P5), and both master fire warning indicators.

Testing is accomplished by pressing a smoke detector test switch and observing a normal fire warning. This procedure should be accomplished for each detector.

WHEEL WELL FIRE DETECTION

One fire detector loop is installed with detector elements in each of the main landing gear wheel wells. A fire or overheat condition in one of the wheel wells will initiate a fire warning. The detector system operates similar to the nacelle and APU fire detection systems except there is no discriminator. With an electrical short (detector failure) or a fire condition causing a short the warning will be locked on. With a fire condition causing an open the wheel well fire warning light will remain illuminated but will extinguish when the overheat condition is over. With an open in the detector loop the system will not respond to test. The brake temperature monitor module on the flight engineer's panel provides an alternate means of monitoring the wheel well area for an overheat condition.

Wheel Well Fire Detection Test Switch - Checks wheel well fire detector circuits. Master FIRE warning lights and WHEEL WELL fire warning light will illuminate while switch is pressed. Bell will sound on some aircraft.

Wheel Well Fire Warning Light (Red) - Illuminates for a wheel well fire condition or during test. Remains illuminated as long as fire condition exists.

NOTE: brake temperature module will display individual wheel temperatures.

MAIN DECK CARGO COMPARTMENT SMOKE DETECTION

A smoke detection system monitors main deck cabin air for the presence of smoke, which is indicative of a fire condition. If smoke is detected by one or several detectors, the Flight Engineer is notified by warning lights on the main deck cargo smoke detection panel on the upper P-4. The pilot's have visual indications of any fire conditions on their master fire warning indicator's lights on the glareshield, and CARGO fire warning light on the overhead panel (P-5).

On N479EV, N480EV, N481 EV, and N482EV, only the warning lights on the Main Deck Cargo Smoke Detection panel illuminate. The Cargo Fire Light (P-5) and master Fire Warning Lights will not illuminate.

On GE aircraft the warning lights on the Main Deck Cargo Smoke Detection module (FE Panel) Cargo Fire Light and master Fire Warning Lights illuminate and the fire warning bell sounds.

The main deck cargo smoke detection system, operating from DC essential bus, is comprised of five smoke detector modules which effectively supply smoke detection coverage to the entire length of the main cargo deck. Each smoke detector module consists of a control amplifier and an A and B flow through type smoke detector.

Both detectors are connected between a smoke sampling vacuum manifold with six to ten sampling orifices located in the ceiling area and a common main vacuum manifold. The vacuum for the main vacuum manifold is created by a venturi ejector which is pneumatically supplied from the pneumatic crossover manifold. When the crossover manifold is pressurized, air is drawn from the cabin ceiling into the sampling orifices through the tubing to the smoke detectors and out the ejector into the forward lower cargo compartment.

The main deck cargo smoke detection system may be tested at the Flight Engineer's upper P-4 panel.

Test switches provided to test all smoke detectors. Operating a Detector Test momentary switch simulates a smoke condition at the smoke detectors causing a fire signal if all circuits are operative.

No air flow lights will illuminate as follows with the loss of crossover manifold pneumatic air pressure at the ejector.

N471 will illuminate with the loss of vacuum in the ejector or vacuum manifold.

With the no air flow light illuminated the main deck smoke detection system is inoperative.

The master fire warning lights may be reset by pressing the BELL-RESET switch on the Flight Engineer's engine fire detection panel or by pressing the Captain or First Officer's master FIRE PUSH TO RESET LIGHT on the glareshield.

WING LEADING EDGE OVERHEAT DETECTION SYSTEM

The wing leading edge overheat detection system provides overheat protection to the fiberglass upper skin panels of both wing leading edges of the airplane and to the wing strut areas. This system consists of thermal switches and a wing leading edge overheat detection panel.

Thermal switches are installed along the pneumatic ducts in each wing leading edge and above the pneumatic manifold in each nacelle strut. The thermal switches are wired in parallel and closure of any one will provide overheat warning light for that wing.

The switches are set to close at three different temperature settings:

In the strut area 300°F (may not be installed in some aircraft).

In the wing/strut area 250°F.

In the wing area 200°F.
The wing leading edge overheat detection module contains the right wing and left wing overheat indicators and the test switches for the system. This module allows for the testing and monitoring of the system.
At normal condition, all thermal switches on both wing leading edges are in the open position. An overheat condition caused by a leakage or rupture in the pneumatic ducts causes the thermal switch or switches in that area to close, illuminating a WING OVHT indicator light on the Flight Engineer's instrument panel.
The wing leading edge overheat detection system receives essential DC power from a master dim and test (IND/WARN LTS-F/E) circuit breaker located on the electrical circuit breaker panel, P2.
The system is tested by actuating the test switch on the wing leading edge overheat detection module to A and then to B positions. This test checks out the continuity of the loops in both wings.
The wing leading edge overheat detection panel is located on the Flight Engineer's instrument panel. This panel contains the LEFT and RIGHT overheat indicators and the test switch for checking the circuit continuity of the system.

Flight Controls

Primary control of airplane flight attitude is provided by ailerons, elevators and rudders. The control surfaces are positioned by hydraulic power packages served by four independent hydraulic systems. Control of the surfaces is accomplished by conventional aileron control wheels, control columns, and rudder pedals located in the flight compartment.

Secondary controls consist of trailing edge flaps, leading edge flaps, spoilers and an adjustable horizontal stabilizer. Trailing edge flaps are hydraulically powered and controlled by a flap control lever in the pilots' control stand. Leading edge flaps are primarily powered by pneumatic motors which are controlled by an electrical output from the trailing edge flap system. Secondary power to the leading edge and trailing edge flaps is provided by electric motors which are controlled by switches in the flight compartment. The spoilers are hydraulically powered from different hydraulic systems. When used for lateral control, the spoilers are positioned by an output from the aileron control system. When used as speed brakes, the spoilers are controlled by a speed brake control lever. The horizontal stabilizer is positioned by hydraulic motors controlled primarily by switches in the aileron control wheel. Levers on the pilots' control stand provide the alternate method which overrides all other command signals.

Lateral trim is accomplished by shifting the neutral position of the aileron control system. The repositioning is done by an electric motor controlled by a switch on the pilots' control stand. Rudder trim is also provided by shifting the neutral position of the rudder control system but is accomplished mechanically by a rudder trim control knob located on the pilots' control stand. Repositioning of the horizontal stabilizer provides pitch trim of the airplane.

In case of malfunction, portions of the flight control hydraulic systems can be isolated. This is accomplished by eight flight control shutoff valves and two stabilizer control modules containing shutoff valves in the hydraulic system. The valves are controlled by switches located in the flight compartment.

AILERONS

Two ailerons in each wing operating with the spoilers provide lateral control of the airplane. The ailerons are hydraulically powered with the control system utilizing all four airplane hydraulic supply systems. The inboard ailerons are operable at any airspeed; the outboard ailerons are used exclusively during low speed flight. Extension of the trailing edge flaps activates the outboard ailerons. Aileron trim is accomplished by use of the aileron arming and trim control momentary switches on the pilots' control stand.

Each aileron is positioned by a single hydraulic power package connected directly to the aileron. The inboard ailerons are designed for a full travel of 20 degrees up and 20 degrees down, available at all times. The outboard ailerons are designed to operate only during slow flight, and become operable by extending the trailing edge flaps. At flap extension greater than 1 degree, full outboard aileron travel of 25 degrees up through 15 degrees down is available.

AILERON CONTROL LOAD LIMITER

The aileron control load limiter performs two functions. It provides an alternate system for moving the ailerons in the event either body cable system should jam and it incorporates a lost motion feature, which prevents undesired feedback of cable system motion into the aileron control wheels.

In normal operation, rotational movement of either control wheel is transmitted to the ailerons through the left body control cables. The right body control cables constitute a standby system and, normally, will not impart motion to ailerons.

The aileron control load limiter is located at the base of the first officer's control column. If the left body control cables should become jammed, the Captain's control wheel will become inoperable and the first officer's control wheel must be used to maintain lateral flight control. Additionally, each attempt to rotate this wheel out of its neutral position will require application of substantially greater force than normal. This force (approximately 26 pounds) will overcome the coupling of the two control wheels and allow the First Officer's wheel to operate the ailerons through the right body control cables.

The lost motion feature is provided by a device in the load limiter, which permits motion of the control wheel bus drum with respect to the load limiter drum within certain mechanical stop limitations. The lost motion feature is required because of system lag through the cable system and power packages, which causes the right body cable to be slightly out of phase with the control wheels. Up to ±4 degree system lag is permitted without causing any input to the aileron control wheels.

AILERON TRIM AND FEEL MECHANISM

The aileron trim and feel mechanism provides artificial feel at the aileron control wheels, and centering and trim of the lateral control system. The mechanism receives its input from the left body control cables and provides an output which operates the central control actuators.

Trim input to the mechanism is provided by a trim actuator attached to the adjacent structure. The actuator consists of an electric motor and jackscrew which provides linear motion of the output shaft. The actuator is controlled by switches on the pilots' control stand. The actuator output shaft is connected to the mechanism support assembly. Output of the actuator repositions the entire trim and feel mechanism, including the input quadrant, to establish a new control system neutral position. This rotation provides an input to the central control actuators which position the lateral flight control surfaces to give the desired trim correction. Rotation of the input quadrant due to trim input also repositions the aileron control wheels away from the normal neutral positions. Positioning of the trim actuator is not affected by reverse force resulting from normal control and operational feedback of the system.

CENTRAL CONTROL ACTUATORS

Two central control actuators, mechanically connected, receive the lateral control input and provide a hydraulically powered output which operates the aileron programmers and spoiler differentials. Hydraulic pressure of 3000 psi is provided to the left actuator from main hydraulic supply system No. 1 and 2. Hydraulic supply systems No. 3 and 4 provide power for the right actuator. The actuator also includes provisions for input from the airplane autopilot system.

AILERON AND SPOILER HYDRAULIC SUPPLY SHUTOFF

An aileron and spoiler hydraulic supply shutoff valve is provided in each of the four airplane hydraulic systems to shut off hydraulic pressure to the aileron and spoiler power control packages and the central control actuators. Each shutoff valve is controlled by an ON-OFF switch on the pilots' overhead panel. A valve position indicator light is located below each switch. The shutoff valve switches are part of the FLIGHT CONTROLS HYD POWER panel and are identified as SPOILERS-AILERON-CCA.

SPOILERS

The spoiler control system supplements the ailerons in providing lateral control of the airplane about the roll axis. There are a total of 12 spoilers of which 10 are flight spoilers and two are ground spoilers. Only the flight spoilers are used with the spoiler control system. However, the flight spoilers are used with the ground spoilers in the speed brake control system. The spoilers are numbered 1 through 12 from left to right. The four outermost spoilers on each wing are identified as outboard flight spoilers. The next spoiler on each side No. 5 and 8, are identified as inboard flight spoilers. The two remaining spoilers, No. 6 and 7, are ground spoilers.

FLIGHT SPOILERS

The flight spoilers are positioned hydraulically and controlled directly by the aileron control system. Actuation of the spoilers is accomplished by a flight spoiler power control package for each spoiler. The airplane main hydraulic systems No. 2, 3, and 4 supply pressure to the flight spoiler control package. The hydraulic pressure for operation of spoilers No. 1, 4, 9, and 12 comes from hydraulic system No. 3. Pressure for operation of spoilers 5 and 8 comes from hydraulic system No. 4.

The control input for operation of the flight spoilers comes from the central control actuators through the spoiler control differential mechanisms and then by cables to the control valves of the flight spoiler power control packages. The central control actuators are part of the lateral control system operated by the aileron control system.

GROUND SPOILERS

The ground spoiler panels are located on the upper surface of each wing trailing edge as are the flight spoilers. The ground spoilers are installed at the most inboard position. In the retracted position, they are flush with the upper surface. Spoilers No. 6 and 7 are identified as ground spoilers.

SPEED BRAKES

The speed brake control system is used to increase drag and reduce lift both in flight and on the ground. The system consists of 12 spoiler panels, numbered 1 through 12, a power control package for each spoiler, a speed brake control lever, drum mechanism, sequence mechanism, ground spoiler control valve, automatic speed brake actuator and the necessary cables and pulleys to operate the system. The five outermost spoilers on each wing, are identified as flight spoilers.

When using spoilers in flight as speed brakes, spoilers No. 3 through 10 are used. Moving the speed brake control lever to FLIGHT DETENT positions spoilers 3, 4, 9, and 10 to 45 degrees and spoilers 5 through 8 to 20 degrees from faired position. A solenoid-operated stop at the speed brake crank under the pilots' floor stops the speed

brake control lever at FLIGHT position during flight.
The automatic speed brake system provides automatic extension of all flight and ground spoilers at touchdown and after a refused takeoff. The system also provides automatic retraction of all flight and ground spoilers when a go-around is initiated after touchdown. When the speed brake control lever is placed in the ARMED position, automatic speed brake operation will be initiated by electrical signals which control operation of the automatic speed brake actuator at touchdown.

AUTO-SPOILER WARNING

An amber AUTO SPOILERS light is located on the pilots' center instrument panel. This light monitors the automatic speed brake control system for failure and, when illuminated, serves to warn the flight crew that the speed brakes should only be operated manually. The AUTO SPOILERS light will illuminate when any one of the following sets of conditions exist.

a) Fault monitor senses fault in landing gear tilt logic and/or hydraulic pressure logic.

b) Speed brake control lever is in DOWN and locked position and automatic speed brake actuator is not in fully retracted position.

TRAILING EDGE FLAPS

The trailing edge flap system provides additional lift during takeoff and landing by increasing the camber of the wing. This is accomplished with four triple slotted trailing edge flaps operating in conjunction with 26 leading edge flaps.
The inboard flaps are powered by hydraulic system 1, and the outboard flaps by hydraulic system 4. Electrical power provides an alternate method of operation. Flap asymmetry protection is provided for the normal hydraulic extension system. The affected hydraulic flap drive will automatically shut down when an asymmetry condition exists on the inboard flaps or on the outboard flaps.
The Flap Protection System will shut off hydraulic trailing edge flap power to all the flaps or to symmetrical sections (both inboard or both outboard) when any or all flap sections move without the flap lever being repositioned. The system is armed when the flaps are either in the UP position or in the TAKEOFF position (flaps 10 or 20) and the flaps and flap lever are both in agreement.
Should the system sense movement of only the inboard or outboard sections during extension or retraction while the flaps are in the aforementioned range, only those sections will lock out and the flaps will "split". To illustrate: while extending the flaps from 10 to 20 on approach, the inboard sections stop at 10, while the outboards extend to 20. Should this happen, consult the Abnormal Procedures section of the Flight Operations Manual. Also, avoid rapid reversals of the flap lever when operating the flaps to insure that flap stoppages do not occur.

ALTERNATE TRAILING EDGE FLAPS

Alternate flap operation is provided by an electric motor installed on each of the flap power packages. The motors are controlled by switches on the pilots overhead panel P5. Guarded arming switches are installed in the circuit to prevent inadvertent operation of the alternate system.

LEADING EDGE FLAPS

The leading edge flaps are normally powered by the pneumatic system. Leading edge flap groups 2 and 4 are programmed by the outboard flaps. Leading edge flap groups 1 and 3 are programmed by the inboard flaps. Extension or retraction using the alternate electrical system will override the pneumatic operation. The leading edge directional switches must be left in the UP or DOWN position and the leading edge alternate flaps arm switch in the ARM position to deactivate the pneumatic system.

ALTERNATE LEADING EDGE FLAPS

Alternate operation of the leading edge flaps is accomplished by an electric motor installed on each of the flap drive units. These motors are controlled by switches on the pilots overhead panel P5. A guarded arming switch in the switch group activates the control circuit. Each of the control switches disengage the normal pneumatic system and operates the electric motor in one drive unit in the left wing and its corresponding drive unit in the right wing. The flaps are extended using the same drive components as used with the pneumatic system.

ELEVATORS

The elevator control system provides primary control of the airplane about its pitch axis. Hydraulic power drives the elevators in response to control column inputs. Two elevators are provided on each side of the airplane.

Airplane hydraulic systems No. 1, 2, 3 and 4 operate the elevators in response to manual or electrical control inputs. The operator applies manual inputs to either control column. The autopilot system applies electrical inputs directly to the inboard elevator power control packages.

In the manual mode, fore and aft displacement of the control column transmits motion from the forward control quadrant and crank to the common aft quadrant through two pairs of cables. As the control column is displaced progressively from neutral, the elevator feel unit imparts a progressively increasing centering force to the control system.

The feel force is controlled by the feel computer.

HORIZONTAL STABILIZER

The stabilizer trim control system trims the airplane longitudinally by varying the horizontal stabilizer angle of attack. The system pivots the stabilizer about its rear attachments to the empennage structure by driving a hydraulically powered trim drive mechanism.

The system provides four modes of stabilizer trim control. Three electrical modes, and an emergency manual-mechanical mode. The electrical modes permit manual-electric actuation by either the Captain or the First Officer, or automatic actuation by the autopilot systems. In all modes, hydraulic pressure drives the trim drive mechanism to position the stabilizer.

The autopilot provides two independent channels of stabilizer trim operation. One channel connected to the left modular control package and the other to the right package. The two channels do not function simultaneously; however, failure of the active channel results in automatic transfer of control to the other channel.

Hydraulic power for the stabilizer trim system is obtained from No. 2 and 3 airplane hydraulic systems. Each system supplies power to an independently functioning group of components. Each group of components consists of a modular control package, a hydraulic motor, and a hydraulic brake. No. 3 hydraulic system supplies the right component group (A system); and the No. 2 hydraulic systems supplies the left component group (B system). Two normally ON, guarded switches on the control stand control hydraulic power to the stabilizer system. These are labeled STAB TRIM. The switches are labeled NO 2 HYD and NO 3 HYD.

STABILIZER TRIM GREEN BAND SYSTEM

The multiple position stabilizer trim green band system reduces the possibility of mistrim for takeoff. The allowable green band takeoff range for a particular airplane gross weight and C.G. is reduced by the selected green band. The actual airplane C.G. is obtained from a sensor on the nose gear. The amber GREEN BAND lights illuminate if the selected green band disagrees with the C.G. The system is operative only on the ground when landing flaps are not selected.

STALL WARNING SYSTEM

The stall warning system receives inputs from left inboard trailing flap, the angle of attack vane, and activates the stick shaker on the Captain's control column to warn of an approaching stall. The stall warning system activates the stick shaker at 107% stall speed, based on the airplane configuration, and the nosegear strut extended (flight mode). The angle of attack vane is electrically heated. However, the heater is deactivated when the APU or an external power source is powering the plane.

OVER ROTATION WARNING SYSTEM

The over rotation warning system is provided to alert the pilots if tail contact is imminent just prior to lift-off. The control column shaker will be activated during takeoff when the rate or angle of rotation is excessive. The system consists of an over rotation warning computer, two lift-off switches, and a control column shaker that is shared with the stall warning system. The system computer utilizes body geometry and pitch attitude from the gyro platform system, in conjunction with Lift-off Logic signals from the nose and body gear switches, to predict an over rotation rate or angle based on a function of time.

RUDDER

Two independently supported and operated rudders without tabs provide yaw control of the airplane. Each rudder is positioned by a hydraulically operated power control package. Hydraulic power for the two power control packages is provided by No. 1, 2, 3, and 4 airplane hydraulic systems. Systems No. 1 and 3 supply the upper rudder power control packages; systems No. 2 and 4 supply the lower rudder power control packages.

As airplane speed increases, rudder movement is decreased by ratio changers that sense airspeed through the auxiliary pitot-static system.

Rudder trim is provided by means of a rudder trim control knob located in the rear portion of the control stand. The rudder control knob, which is attached to a forward control drum drives a rudder trim actuator by means of cables. The rudder trim actuator controls the feel and centering mechanism null position. Linkage common to the rudder control system connects the rudder trim system to the power control packages.

YAW DAMPER

Each hydraulic power control unit receives a separate yaw damper signal that is in series with pilot inputs, but is not felt at the rudder pedals. The yaw damper has approximately four degrees of authority. Hydraulic system 2 is the power source for the lower rudder and system 3 provides power for the upper rudder. Yaw damper control switches are located on the pilots' overhead panel.

Rate of turn indication is supplied by the yaw damper computers. The needle on the captain's ADI is supplied by the upper yaw damper computer and the needle on the first officer's ADI by the lower yaw damper computer.

TURN COORDINATOR

The yaw damper system provides turn coordination when the flaps are out of the up position. Roll angle from DG/VG 1 (upper rudder control) and DG/VG 2 (lower rudder control) is transmitted to the yaw damper computers. Each hydraulic power control unit receives a signal from the computer to move the rudder in proportion to the roll angle signal. In addition, yaw damper inputs are integrated with the turn coordination inputs. The movement is not felt at the rudder pedals

YAW DAMPER WARNING LIGHTS (TURN COORDINATION)

Two lights located on the pilot's caution panel labeled YAW DAMPER UPPER and YAW DAMPER LOWER indicate turn coordination out of sequence with flap position. With flaps up and the light(s) illuminated, the turn coordination is still operating. With flaps down and the light(s) illuminated, the turn coordination is not operating. The YAW DAMPER UPPER light will normally come on as the flaps are extended if STANDBY POWER switch is in the OFF POSITION or if the standby power light is ON. This is to prevent rudder oscillations if a failure permits STBY AC and ESS AC to be on simultaneously. Turning the yaw damper switch off will not extinguish the light.

Fuel

The fuel tanks provide storage for the fuel. The tanks are located in the interspar area of each wing and in the wing center section in the fuselage. The tanks are designated No. 1, 2, 3 and 4 main tanks, No. 1 and No. 4 reserve tanks, and center wing tank. Each wing tank is an integral portion of the wing structure, sealed to provide a fluid-tight compartment. Sealing is accomplished primarily by the close metal-to-metal fit of all parts, by use of sealed fasteners, and by the application of sealing compound to all applicable joints in the structure. This design is called a wet wing. The wing ribs in the tanks act as baffles to prevent excessive fuel sloshing. Baffle check valves are installed in the ribs at various locations to prevent fuel from flowing away from the boost pumps and to prevent a rapid shift in C.G. due to airplane attitude changes (roll).

The center tank is comprised of interconnected bladder cells. The -100 aircraft has 4 cells, and the -200 aircraft has 5 cells.

A vent system connects the tanks to overboard vent near each wing tip, on the lower wing surface.

Fuel sump drain valves are installed in low points in the tanks to permit draining accumulated moisture from the tanks and for draining remaining fuel when the tank has been defueled.

Located within the fuel tanks are the fuel lines, pumps, valves, vents, drains, and sensing equipment required for operation and monitoring of the fuel system.

The electric motors that operate the fuel boost pumps and fuel valves are located on forward and aft wing spars and in dry bays. The valves and pumps are inside the tanks. Fuel valves and their motors are connected with drive shafts called adapter shafts. Monitoring the position of the valves is accomplished at its' respective drive motor.

Fuel quantity is accomplished with a digital indicating system with gages on the flight engineers panel and at the fueling station under the left wing.

Fueling and defueling is accomplished at stations under the left and right wings, with 2 standard single point receptacles under each wing. All controls and indicators are under the left wing.

Fuel jettison is accomplished with 6 pumps and associated valves. The fuel is dumped overboard through a nozzle valve near each wing tip, on the trailing edge of the wing.

All 4 engine fuel feed systems are connected to a crossfeed manifold to enable engine fuel feed from the center tank or any main wing tank, to any engine.

A refuel/defuel and dump manifold is used for refueling, defueling and fuel jettison.

The 2 manifolds can be connected for fuel transfer between tanks on the ground, and for fuel jettison inflight using center wing jettison valves.

APU fuel is supplied from main tank number 2.

Residual fuel in the center tank is scavenged into main tank number 2.

The fuel vent system provides positive venting of the fuel tanks to atmosphere during all attitudes of the airplane. Venting of the fuel tank is accomplished by sealing hat-shaped upper wing skin stringers interconnected with drain and vent tubing. The ducts are vented into a vent surge tank and overboard through a vent outlet near the wing tips.

The expansion space above the fuel in the tank shifts according to attitude changes of the airplane. To provide adequate venting to atmosphere under any operating condition, inboard and outboard vent ports are provided in each wing tank and a left and right vent port in the center wing tank. The ports are located so that at least one port is open to the expansion area at all times. Float actuated vent valves are installed on vent ports at various locations to prevent fuel from entering the vent system, when the airplane is in a wing-low attitude. The float valves are located on the rear vent ports in the center wing tank to prevent fuel from entering the vent system when the airplane is nose up. Float actuated drain valves are installed at low points in the vent system to allow fuel trapped in the vent lines to drain back into the tank when normal flying attitude is established. Fuel surges too large to be trapped in the vent ducts are directed to surge tanks located near the wing tips. Fuel in the surge tanks will drain back into the inboard main tanks through two drain lines and check valves located between the lower wing stringers.

The vent outlet located in the surge tank vent line is designed to provide tank pressure close to atmosphere for all flight conditions.

When the airplane is in flight, the vent outlet provides atmospheric pressure in the vent surge tank. The vent ducts connecting the surge tank with each fuel tank distribute the pressure to each fuel tank. When the airplane is in normal flight all vent ports are open to atmosphere. In a wing low attitude, the inboard vent ports are open and the outboard ports may be covered by fuel. Float actuated vent valves connected to the outboard ports prevent fuel from entering the vent system and flowing into the surge tank. In a wing high attitude the outboard ports will be open. Fuel entering the inboard ports will drain back into the tank through float actuated drain valves when the airplane returns to a normal attitude. Any fuel escaping through the vent system into the surge tank will drain back into the inboard main tanks through two drain lines connected to the bottom of the surge tank.

Each wing tip's outboard fuel tank bay is utilized as a vent surge tank. The surge tank will collect any overfill from any fuel tank. If the overfill exceeds approximately 40 U.S. Gallons, a fueling overfill float switch (front spar mounted) will cause all fueling valves to close. If more than 125 U.S. Gallons is collected, the fuel will spill overboard through the vent opening in the surge tank (standpipe), the vent duct (via Flame Arrester), and the opening to atmosphere on the Ram Air Scoop (underside of wing). The Surge Tank is equipped with both positive and negative pressure relief valves, which are designed to provide tank pressure relief in the case of a blocked vent system.

SUMP DRAIN VALVES

Fuel sump drain valves are installed at the low point of each fuel tank for draining accumulated moisture and for draining fuel trapped in the sumps when the tanks are defueled. The drain valves for the center wing tank, No. 2 and No. 3 mains, and surge tanks are located in the bottom of their respective tanks. The drain valves for No. 1 and No. 4 reserve and No. 1 and No. 4 main tank are located inboard of the tanks and are connected to the tanks by tubing.

BAFFLE CHECK VALVES

Baffle check valves are installed in various ribs in the main fuel tanks to reduce the rate of fuel flow toward the wing tips during airplane attitude changes and to maintain fuel in the boost pump area of the tank.

OVERWING FILL PORTS

Overwing fill ports are located in the four main tanks only. Each fill port consists of a filler cap adapter, a seal ring and clamping nut fitted into a hole in the upper wing panel. A flush fitting cap assembly is installed in each fill port.

RESERVE AND MAIN TANK INTERCONNECTION

The reserve and main tanks are interconnected to facilitate fuel transfer during normal fuel feed operation, defueling or jettison operation. Fuel transfer and tank interconnection is limited to the transfer of fuel by gravity flow from No. 1 reserve to No. 1 main tank, No. 4 reserve to No. 4 main, No. 1 main to No. 2 main and No. 4 main to No. 3 main. Tank interconnection is accomplished through the use of electric motor-driven shutoff valves located in the transfer lines between the tanks. The reserve tank transfer valves are used to transfer fuel from the reserve to the outboard main tanks during a normal fuel feed operation in order to move the fuel to a usable location. The valve is also used during a defueling or jettison operation. The main tank transfer valves are used only during a defueling or jettison operation.

The reserve tank transfer valves are controlled from the fuel control panel while the main tank transfer valves are controlled from the fuel jettison panel. Both panels are located at the Flight Engineer's station. Power to the valves is 28 volt DC supplied from the P12 electrical circuit breaker panel.

RESERVE TANK TRANSFER VALVE

The reserve tank transfer valves control the transfer of fuel from the reserve tanks to the respective outboard main tanks. No. 1 reserve transfer valve is located in the transfer line between No. 1 reserve tank and No. 1 main tank. No. 4 reserve transfer valve is located in the line between No. 4 reserve and No. 4 main. The valves are controlled by individual rotary type switches on the fuel control panel.

RESERVE TANK TRANSFER SWITCH

Two rotary-type switches are located on the fuel control panel to supply 28 volt DC power to the reserve tank transfer valves. The knob on the switch is marked to correspond with the flow diagram on the fuel control panel. When the mark on the knob is aligned with he flow diagram, the valve is open. When the knob is rotated until the mark is across the low patch of the diagram, the valve is closed.

RESERVE TANK TRANSFER VALVE INDICATOR LIGHT

The reserve tank transfer valve indicator lights are installed on the fuel control panel to indicate position of the valves. The lights are blue with white letters that read RES VALVE. The lights are illuminated when the valves are in transit and are extinguished when the valves are open or closed. Although the master dim circuit provides power to the indicator light, master dim is not used.

RESERVE TANK TO OUTBOARD MAIN TANK FUEL TRANSFER

Fuel from the reserve tank can be transferred into the outboard main tank when 28 volt DC power is available to the reserve tank transfer valve. Fuel transfer is initiated by rotating the switch control knob on the fuel control panel until the knob is aligned with the flow diagram. When the fuel in the outboard main tank is below the level in the reserve tank, fuel will gravity flow from the reserve tank into the main tank. Fuel flow is indicate on the fuel control panel by an increasing fuel quantity indication for the outboard main tank and decreasing fuel quantity indication for the reserve tank. To close valve, the control knob is turned perpendicular to flow diagram

MAIN TANK TRANSFER VALVES

The main tank transfer valves are used to transfer fuel from No. 1 and No. 4 main tank to No. 2 and No. 3 main tanks when defueling the airplane or when dumping fuel through the jettison system. The valves are located in the outboard end of the inboard main tanks. The valves are controlled by switches on the fuel jettison panel at the Flight Engineer's station.

MAIN TANK TRANSFER VALVE INDICATOR LIGHT

The gravity transfer valve indicator lights are installed on the fuel jettison panel to indicate position of the valves. The lights are illuminated when the valves are in transit and extinguished when the valves are open or closed. Although the indicator lights receive power from the master dim circuit, master dim is not used on these lights.

OUTBOARD MAIN TO INBOARD MAIN TANK FUEL TRANSFER

Fuel from the outboard main tank can be transferred into the inboard main tank when 28 volt DC power is available to the main tank transfer valve. Fuel transfer is initiated by placing the No. 1 or No. 4 main tank transfer switches to OPEN. These switches, which are located on the jettison control panel at the Flight Engineer's station, will allow fuel to gravity flow from the outboard to the inboard main tank. Fuel flow is indicated in the fuel control panel by an increasing fuel quantity indication for the inboard main tank and decreasing fuel quantity indication for the outboard main tank. To close main tank transfer valve, the valve switch is positioned to CLOSE.

CENTER WING TANK SCAVENGE SYSTEM

The center wing tank scavenge system provides a means of removing residual fuel remaining in the center wing tank that cannot be pumped out by the override/jettison pumps. The scavenge pump transfers this fuel into No. 2 main tank. Power for the scavenge pump is 3-phase, 115 volts AC, supplied from AC bus No. 1 on the P6 main power circuit breaker panel.
The scavenge system is controlled by a switch on the Flight Engineer's panel, through a relay in the flight deck equipment panel. Power for the relay is 28 volts DC, supplied from DC bus No. 1 in the P12 electrical circuit breaker panel.

CENTER WING TANK SCAVENGE PUMP

The scavenge pump assembly is mounted in the aft rear spar wall of the center wing tank. The pump has a capacity of approximately 1700 pounds per hour at 7 PSI.

The motor is provided with three non-resettable thermal fuses, which are integrally installed. The fuse senses over temperature and at a preset value will actuate causing pump operation to stop. This will necessitate replacement of the motor. The motor is cooled and lubricated by the circulation of fuel through the motor housing.

SCAVENGE PUMP SWITCH

The scavenge pump switch is a lever lock type switch mounted on the fuel control panel at the flight engineer's station. The switch through the scavenge pump relay, controls power to the scavenge pump.

The center wing tank scavenge system becomes operative when the scavenge pump relay is energized by 28 volt DC power and 115 AC power supplied to the scavenge pump.

To remove remaining fuel in the center wing tank, the scavenge system is operated by placing the scavenge pump switch to the ON position. This energizes the scavenge pump relay which allows power to be supplied to the scavenge pump. With the pump operating, fuel in the tank is drawn in through the pump inlet and screen into the suction line and pump housing. Fuel in the pump is discharged into main tank No. 2 through a check valve in the pump housing and another check valve installed in the discharge line.

When pressure on the discharge side of the pump reaches 4.0 to 7.0 PSI, the low pressure indicator light adjacent to the scavenge pump switch extinguishes. When the pump discharge pressure falls below the range of pressure switch actuation, the indicator light will illuminate, indicating that fuel has stopped flowing. Toward the end of the scavenging operation, the pressure indicator light will be on and off continually until the pump can suck in no more fuel.

FUEL PRESSURE INDICATING SYSTEM

The fuel pressure indicating system provides a means of monitoring the operation of various fuel pumps in the airplane fuel tanks. Indicator lights at the Flight Engineer's panel provide the necessary indication in the flight compartment.

The fuel pressure indicating system provides both low and high pressure indications at the Flight Engineer's panel. The low pressure indicating system contains amber lights which illuminate when the discharge pressure of any main boost pump override/jettison pump, main jettison pump or scavenge pump falls below a preset value, indicating no fuel flow. The high pressure indicating system contains a green light, which illuminates when the APU pump discharge pressure rises above a preset pressure, indicating fuel flow. Power for both systems is 28 volts DC supplied through the master dim and test circuit.

The fuel pressure indicating system consists of 16 indicator lights and 16 pressure switches, one for each pump. The main boost pump, override/jettison pump, and scavenge pump lights are located on the fuel control panel. The main jettison pump lights are on the jettison panel. The APU pump light is on the auxiliary power panel. The lights are controlled by the master dim and test switch on the Flight Engineer's auxiliary panel.

PRESSURE SWITCHES

The pressure switches control the pressure indicating lights.

The switches for the main boost pumps are located next to the boost pump motors and are connected to a sensor line from the boost pump discharge port. Switches for the override/jettison, scavenge, APU and jettison pumps are mounted directly on the pumps and are connected to the discharge side of the pumps through an internal passage.

To operate the low pressure indicating system, both 28 volt DC and 115 volt AC power must be available on the airplane. The power supply for the indicating systems is from the master dim and test circuit.

For the main jettison pumps and scavenge pump indicator lights to illuminate, the pump must be operating with no flow. Placing the pump switch in the ON position energizes the pump control relay and connects the indicating light circuit to ground through the pressure switch. When the pump starts to operate, the pressure switch actuates on an increasing pressure of 4.0 to 7.0 PSI and opens the circuit. When the pump switch is positioned to ON, the indictor light may flash momentarily until the above pressure is reached. With the circuit open, the indictor light will remain extinguished until the output pressure of the pump decreases to a pressure of 7.0 to 4.0 PSI. At this point, the indicator light will illuminate indicating low pressure or no flow from the pump. When the pump switch is positioned OFF, the pump control relay will de-energize and open the circuit between the light and the pressure switch.

The pressure indicator lights for the main boost pumps and override/jettison pumps are on anytime power is applied to the master dim and test system and the pumps are not operating. When boost pump output pressure increases, the pressure switch actuates between 4.0 and 7.0 PSI and opens the ground to the circuit. The indicator lights will remain extinguished until pump pressure decreases to a pressure 7.0 to 4.0 PSI. At this point the indicator light will illuminate indicating low pressure or no flow from the pump.

HIGH PRESSURE INDICATING SYSTEM

The high pressure indicating system is used only for the APU boost pump. When output pressure of the APU pump reaches between 4.0 and 7.0 PSI, the pressure switch is actuated and connects the indicating light circuit to ground. With the circuit closed, power from the master dim and test circuit will illuminate the indicator light. The light will remain illuminated until the pump output decreases to a pressure between 7.0 and 4.0 PSI, at which time the light will extinguish, indicating a very low pressure or no flow from the pump.

ENGINE FUEL FEED SYSTEM

The engine fuel feed system provides a means of delivering fuel from the tanks to the engines. The system is designed to supply fuel to the engines at all airplane attitudes. The system consists of four main tank-to-engine subsystems, which are interconnected by crossfeed valves and crossfeed lines, such that fuel may be delivered from any main tank or center wing tank to any or all engines.

Each main tank-to-engine subsystem delivers fuel from one main tank to its respective engine. Each subsystem is equipped with an engine fuel shutoff valve, and engine

crossfeed valve, and two AC powered fuel boost pumps. The pumps in each subsystem are controlled by separate switches and power sources so that fuel supply to the engine will not be disrupted by failure of a single power source or failure of one pump. In addition, each engine fuel feed subsystem is provided with a boost pump bypass valve to allow suction feed to the engine in case of boost pump failure.
The center wing tank pumps, known as the override/jettison pumps, will override the main tank boost pumps and supply fuel from the center wing tank to any or all engines when the applicable crossfeed valves are open.
The reserve tanks are not connected to a fuel feed manifold. Fuel from the reserve tanks is fed by gravity to main tanks No. 1 and No. 4 by means of transfer valves.
The engine fuel feed system is controlled and monitored at the flight engineer's panel. Power to the system is 115 volts AC and 28 volts DC for the pumps and valves respectively.

ENGINE FUEL FEED MANIFOLD

The engine fuel feed manifold is used to distribute fuel from the boost pumps to the engines. The manifold is installed inside the fuel tanks and is secured in place with clamps and support brackets. The fuel feed manifolds in the left and right wings are interconnected by a crossfeed manifold installed along the rear spar inside the center wing tank. Fuel can be directed through the use of engine fuel crossfeed valves, to any or all engines from any main tank or tanks, or center wing tank.

ENGINE FUEL SHUTOFF VALVES

Four engine fuel shutoff valves provide a means of shutting off the supply of fuel to the engines. The shutoff valves are individually controlled by engine fuel shutoff valve switches on the fuel control panel at the Flight Engineer's station, by the fire switches on the pilots' overhead panel, and by the engine start levers on the pilots' control stand.

ENGINE FUEL SHUTOFF VALVE CONTROL SWITCHES

The engine fuel shutoff valve switches control 28 volt DC hot battery bus power to the engine fuel shutoff valves. The switch for each engine is located on the fuel control panel. Each switch is a single-pole, double-throw switch that is guarded to the OPEN position.

ENGINE FUEL SHUTOFF VALVE INDICATOR LIGHTS

The engine fuel shutoff valve indicator lights are installed on the fuel control panel to indicate position of the valves. The lights are white with black letters that read ENG VALVE. The lights are illuminated bright when the valves are in transit and are extinguished when the valves are open. The lights are dim when the valves are closed.

FUEL BOOST PUMPS

The fuel boost pumps deliver fuel under pressure from main tanks 1,2,3 and 4, to the engines. 2 pumps are installed in each main wing tank, FWD and AFT. Each pump is a centrifugal type pump driven by a 3-phase AC electric motor.

A non-resettable thermal fuse is installed in each phase winding of the motor. The fuse senses over temperature and is set to actuate at a preset temperature. Actuation of a fuse will necessitate replacement of the motor. The motor is cooled and lubricated by the circulation of fuel through the motor housing.

FUEL BOOST PUMP CONTROL SWITCHES

The fuel boost pump switches control power to the main boost pumps. The switches are located on the fuel control panel. The switches provide 28-volt DC power to each of the boost pump relays, which enable power to be supplied to the pumps.

FUEL BOOST PUMP RELAYS

The fuel boost pump relays enable 115-volt 3-phase AC power to be supplied to the boost pumps. The relays are controlled by 28 volt DC power from the boost pump control switches on the fuel control panel. Relays for boost pumps No. 1 main aft, No. 2 main aft, No. 3 main forward and No. 4 main forward are located in the main power center - left (P14) panel. Relays for boost pumps No. 1 main forward, No. 2 main forward, No. 3 main aft and No. 4 main aft are located in the main power center - right (P15) panel.

FUEL BOOST PUMP CHECK VALVES

The fuel boost pump check valve closes the boost pump discharge line when the pressure drops below a preset amount, thus preventing reverse fuel flow through the pump. The valves are located at the discharge port of each boost pump inside the fuel tank.

FUEL BOOST PUMP BYPASS VALVES

A boost pump bypass valve is provided for each engine to allow suction feed when the boost pumps fail. The valve consists of a screen and a check valve body with a hinged flapper. The valve is attached to the suction feed line for each engine.

ENGINE FUEL CROSSFEED VALVES

Four engine fuel cross feed valves provide a means of directing fuel flow in the fuel manifold. The valves are the same as the engine fuel shutoff valves except for length of the adapter shafts. The valves are located in the fuel feed manifold with valve actuators installed on front spar. The valves are controlled by individual fuel crossfeed valve switches on the fuel control panel.
Valve operation may be verified by monitoring manual position lever on valve actuator.

ENGINE FUEL CROSSFEED VALVE CONTROL SWITCHES

Four engine fuel crossfeed valve control switches located on the fuel control panel supply 28-volt DC power to the crossfeed valves. The switches are a rotary type with a knob that is marked to correspond with the flow diagram on the fuel control panel. When the mark on the knob is aligned with the flow diagram, the valve is open. When the knob is rotated until the mark is across the diagram flow path, the valve is closed.

ENGINE FUEL CROSSFEED VALVE INDICATOR LIGHTS

The engine fuel crossfeed valve indicator lights are installed on the fuel control panel to indicate position of the valves. The lights are blue with white letters that read CROSSFEED VALVE. The lights are illuminated when valves are in transit and are extinguished when the valves are open or closed.

The engine fuel feed system becomes operative when 28 volt DC power is supplied to the pump relays and valves and the pumps are energized by 115 volt AC power.

The engine fuel feed system supplies fuel to the engines by direct tank-to-engine feed or by crossfeed from any main tank or center wing tank. To obtain a direct tank-to-engine fuel feed, the engine fuel crossfeed valves are closed and the engine shutoff valves are opened. The main boost pump switches on the fuel control panel are then placed to ON. This energizes the boost pump relays, which enable power to be supplied to the pumps. When the pumps operate, fuel is drawn from the tanks through screened inlets, and fed to the engines through the fuel feed manifold and the engine shutoff valves. When the boost pumps become inoperative, fuel is drawn to the engines through the boost bypass valve by suction from the engine-driven fuel pumps.

Fuel from the center wing tank can be delivered to the engine by fuel crossfeed through the fuel crossfeed manifold. This is accomplished by placing the override/jettison pump switches to ON which energize the pump relays and enables power to be supplied to the pumps. When the pumps operate fuel in the center wing tank is drawn in through screened inlets and discharged into the fuel crossfeed manifold. Rotating the applicable crossfeed valve switch, or switches, to align with the fuel flow diagram on the fuel control panel allows fuel flow in the crossfeed manifold to the applicable engines.

Fuel crossfeed from one main tank to the adjacent engines is accomplished by manipulating the applicable crossfeed valve switches, provided the fuel flow diagram is satisfied. When fuel crossfeed is not desired, the crossfeed valve switches are rotated such that the aligning marks on the knobs are perpendicular to the fuel flow diagram.

Fuel remaining in the center wing tank, which cannot be pumped out by the override/jettison pumps, can be removed with the scavenge pump. This is accomplished by placing the scavenge pump switch on the fuel control panel too ON. When the pump operates, fuel in the center wing tank is transferred into main tank No. 2 for engine feed.

To terminate engine fuel feed system operation, the boost pump and override/jettison pump switches are placed to OFF.

FUEL TEMPERATURE INDICATOR (JT9D)

1. Fuel Temperature Indicator- Indicates fuel temperature in the No. 1 main tank or fuel temperature downstream of the fuel heater, as selected by fuel temperature switches.
2. Fuel Temperature Indicator Switches - Depress desired switch for readout on fuel temperature indicator. Depressed switch illuminates white.

APU FUEL FEED SYSTEM

The APU fuel feed system supplies fuel from the No. 2 main tank to the auxiliary power unit in the aft end of the fuselage. The system consists of a fuel supply line extending from No. 2 main fuel tank to the APU, a drain tine with a flame arrester connected to the supply line shroud, a DC motor-operated pump, a fuel valve and a pressure switch. The system is connected to the No. 2 main aft boost pump, which automatically takes over from the DC pump when AC power is available. Control for the system is from the APU master control switch on the Flight Engineer's panel.

APU FUEL SUPPLY LINE

The APU fuel supply line supplies fuel from the APU fuel valve to the auxiliary power unit. The supply line consists of an aluminum tube from the fuel valve, through the center wing tank to the rear spar. From the rear spar aft to the APU firewall, the line consists of a series of flexible hoses coupled together and encased in a sealed shroud.

The fuel line shroud consists of several sections of aluminum tubing connected with sealed couplings. The shroud connects to a flexible metal bellows that attaches to the APU firewall. The line is installed on an incline so that any fuel in the shroud that might leak from the flexible hose will drain forward to the shroud drain line.

APU FUEL LINE SHROUD DRAIN

The APU fuel shroud drain is connected to the APU fuel shroud to drain residual fuel that could collect in the shroud from a leak in the flexible hose.

APU FUEL BOOST PUMP

The APU fuel boost pump provides fuel to the auxiliary power unit when AC power is not available.

The 28-volt DC electric powered motor incorporates a non-resettable thermal fuse, integrally installed in the motor winding. The fuse senses temperature only and activates below 400°F. Actuation of the fuse will necessitate replacement of the motor. A port on the motor flange is provided for the installation of a pressure switch that senses pump discharge pressure and causes a DC PUMP ON light to illuminate. The motor is cooled and lubricated by the circulation of fuel through the motor housing.

APU FUEL SHUTOFF VALVE

The APU fuel shutoff valve controls fuel flow to the auxiliary power unit. The valve assembly consists of two replaceable subassemblies consisting of an electric actuator located on the aft side of No. 2 main tank rear spar and a valve body located inside the tank. The valve is controlled by the APU master control switch on the Flight Engineer's instrument panel.

APU FUEL BOOST PUMP CONTROL PRESSURE SWITCH

The APU fuel boost pump pressure switch provides automatic changeover from the APU DC boost pump to the No. 2 main aft boost pump after the APU has started and AC power is available.

With the APU master control switch in the ON position, 28 volt DC power from the main battery opens the fuel valve, actuates No. 2 main aft boost pump relay and actuates the APU fuel boost pump control relay. When the APU fuel boost pump control relay closes, power is supplied to the APU DC boost pump from the 28 volt DC battery bus on the main power circuit breaker panel (P6). When the DC boost pump is operating, a green light is illuminated on the auxiliary power panel. After the APU has started and the AC ground service transfer bus is energized, the No. 2 main aft boost pump starts. Fuel pressure from this pump actuates the APU fuel boost pump control pressure switch, breaking the ground to the APU fuel boost pump relay and removing power from the DC pump.

PRESSURE FUELING SYSTEM

The pressure fueling system provides a rapid means of filling the fuel tanks in the airplane. The system distributes fuel under pressure from a fueling station in each wing to the tanks through manifolds and refuel valves. The refuel valves allow fuel to flow from the main distribution manifold into the tanks. The refuel valves are installed in the system; one each in the reserve and outboard main tanks; two in the inboard mains; and two in the center wing tank.

Power for the refuel valves is 28 volts DC supplied from the ground handling bus on P14 left main power center or the battery bus on P6 main power circuit breaker panel.

Each fueling station is equipped with two fueling receptacles coupled together to a manifold that extends through the outboard main tank to the main distribution manifold. Each receptacle incorporates a manual shutoff valve. The fueling station in the left wing is provided with a control panel consisting of fueling quantity (repeater) indicator, refuel valve control switches, valve position indicator lights, a refuel power switch and a test switch for the volumetric shutoff control unit, and indicators. A proximity switch is actuated by the door to cut off all power to the panel when the door is closed.

The fueling receptacles provide a means of connecting ground refueling hose nozzles to the pressure fueling system. Two receptacles at each fueling station are mounted on the forward face of the front spar.

During pressure fueling, the check valve opens when nozzle fuel pressure is greater than receptacle manifold pressure. For defueling, the check valve is held open by the nozzle after positioning a lever in the receptacle marked LIFT TO DEFUEL.

REFUELING MANIFOLD

The refueling manifold distributes fuel to all tanks during a pressure fueling operation. The manifold consists of two crossover manifolds and a main distribution manifold; the main distribution manifold is also utilized as the fuel jettison manifold. The crossover manifolds extend from the fueling receptacles at the front spar through the outboard main tanks to the main distribution manifold. The main distribution manifold is routed inside the fuel tank area with solenoid-controlled refuel valves installed in each tank. Multiple outlet tube manifolds are installed at No. 1, No. 2, No. 3 and No. 4 main tank refuel valves to provide equal distribution of fuel in each tank.

REFUELING CONTROL PANEL

The refueling control panel contains all the controls required for operation of the ground refueling system. The panel is located in the lower leading edge of the left wing, outboard of the fueling receptacles. Components mounted on the panel include seven fueling quantity (repeater) indicators, 10 refuel valve switches, 10 refuel valve position indicator lights, a refuel power switch, and overfill reset switch and a test switch. Three floodlights provide illumination for the area.

The fueling quantity (repeater) indicators provide an indication of the fuel quantity in each individual tank. The indicators are connected to corresponding primary indicators on the Flight Engineer's lower instrument panel and obtain their signals form the fuel quantity indicating system tank units.

The test switch provides a means of checking the indicators for operation and also checks the operation of the volumetric shutoff control unit causing all refuel valves to close with fuel pressure in the manifold. The overfill reset switch resets the fueling shutoff relay to allow continued fueling provided a surge tank float switch is not closed.

The four FUELING RECEPTACLE assemblies contain an international-standard self-sealing hose connection and cap, and a manual rotary gate valve.

The outboard receptacles contain a port connecting to a line from the DEFUELING VALVE.

The poppet of the fueling receptacle functions as a check valve. For defueling, a tab must be raised to allow the hose nozzle poppet to hold the receptacle poppet open to obtain reverse flow.

The DEFUEL VALVES, when open, connect the crossfeed manifold to the outboard fueling receptacles. If the outboard receptacle manual valve is open, the crossfeed manifold is connected to the fueling and jettison manifold. The defueling valves can be used for defueling and transferring fuel.

The handles (levers) for the defueling valves and fueling receptacle valves are not interlocked with the fueling doors. These handles must be placed in the closed position before flight.

SURGE TANK FLOAT SWITCHES

A float switch in each surge tank closes all refuel valves when a tank is overfilled during pressure fueling. Fuel from the overfilled tank flows through the tank vent into the surge tank. The float switch closes and energizes the fueling shutoff relay. When the relay is energized, it removes power to the volumetric shutoff control unit and closes all refuel valves. When the fuel drains from the surge tank, the float switch opens. However, the fueling shutoff relay will remain energized until the overfill reset switch is depressed.

Older aircraft that do not have the RESET button on the fueling panel require the circuit breaker on the P15 panel pulled and reset.

The pressure fueling system is operative when 28 volt DC power is supplied to the refuel valves and 115 volt AC power is supplied to the volumetric shutoff control unit.

After the caps are removed from the fueling receptacles, the fueling hose nozzles are connected and the manual shutoff valves at each receptacle are opened. The airplane is ready to receive fuel.

Prior to starting the fueling operation, the refuel valve position indicator lights may be checked by press-to-test. The quantity indicators may be checked by pressing the TEST GAGES switch. If a tank has an inoperative indicator, the fuel quantity can be measured by using the fuel measuring sticks.

The TEST GAGES switch on the refueling control panel permits testing both the volumetric shutoff system and the fuel quantity indicating system components on the refueling control panel during the refueling operation. Pressing the switch causes the volumetric shutoff system to close all open refuel valves so that the corresponding position indicator lights extinguish, indicating that the volumetric shutoff system is operative. Releasing the switch causes the position indicator lights, and repeater indicators to return to their original state. Use of the TEST GAGES switch to test position indicator light operation is possible only during the refueling operation because pressurization of the fuel manifold is necessary for operation of the refuel valves.

DEFUELING SYSTEM

The fuel system is designed to permit complete or partial defueling of any or all tanks. To accomplish the defueling operation, portions of the fuel jettison, pressure fueling, center wing tank scavenge, and engine fuel feed systems are used in combination. Rapid defueling of the fuel tanks is accomplished by utilizing the fuel jettison system to bring the fuel in the tanks below the jettisonable level. However, instead of discharging fuel through the jettison nozzles at the wing tips, fuel is discharged into fuel truck or trucks through manual shutoff valves at the fueling receptacles. Either or both fueling stations may be used to defuel the tanks.

Fuel remaining in the main tanks which is below the jettisonable level can be pumped out by operating the main tank boost pumps to pressurize the fuel feed manifold. Fuel in the fuel feed manifold is discharged into the outboard fueling receptacles through the manual defueling valves. Remaining fuel in the center wing tank can be removed by operating the scavenge pump to transfer fuel into the No. 2 main tank. Fuel in the tanks, which cannot be drawn in by the pumps, can be removed through the sump drain valves.

MANUAL DEFUELING VALVE

The manual defueling valve provides a means of defueling the fuel tanks by using the main tank boost pumps. The valves, one at each fueling station, connect the fuel feed manifold with a pressure fueling receptacle. The valve is located on the front spar between the fueling receptacles. The valve consists of a manually controlled shutoff element, a valve body, and provisions for attaching the valve to the spar and to the fuel manifold. An elbow attaches the forward side of the valve to the fueling receptacle.

DEFUELING CHECK VALVE

A defueling check valve is attached to the manual defueling valve inlet to prevent air from entering the fuel feed manifold when the engines are in suction feed, and the manual defueling valve has been inadvertently left open. The valve consists of the valve body and a hinged flapper, which is spring-loaded to the closed position. The valve is installed on the wet side of the front spar opposite the manual defueling valve.

FUELING/DEFUELING RECEPTACLE

The fueling receptacles connect the defueling hoses to the airplane fuel system. When the hose nozzle is coupled to the receptacle adapter, a spring-loaded check valve is lifted to a position that allows the hose nozzle to lock into place. For pressure fueling, the check valve opens only when nozzle fuel pressure is greater than receptacle manifold pressure. For defueling, the check valve is held open by the nozzle after positioning the receptacle adapter lever marked LIFT TO DEFUEL.

FUEL JETTISON SYSTEM

The fuel jettison system provides a means for dumping fuel overboard during an inflight emergency to reduce the weight of the airplane to an allowable landing weight. Fuel from the tanks is pumped into the jettison manifold and then discharged through a nozzle located on both wig tips of the airplane. Power to the system is 28 volts DC supplied from the P12 electrical circuit breaker panel and 115 volts AC from the P14 and P15 main power center.

The jettison system consists of two wing tip nozzles, two nozzle valves, two jettison pumps is each inboard main tank, two override/jettison pumps in the center wing tank, two center wing tank jettison valves, and a fuel jettison manifold. Also used during the jettison operation are two main tank transfer valves and two reserve tank transfer valves. All controls used in the jettison system are located on the flight engineer's panel.

Fuel jettison system components, such as pumps, valves and controls, are also used for pressure fueling and defueling while the airplane is on the ground.

FUEL JETTISON MANIFOLD

The fuel jettison manifold provides a passage for fuel to be dumped overboard during a jettison operation. The jettison manifold is located inside the fuel tanks and extends from wing tip to wing tip of the airplane. Located at each end of the manifold is a fixed fuel jettison nozzle. The continuous manifold permits dumping of fuel from one side of the wing to the other wing tip should the jettison nozzle valve on that side fail to open. The manifold is also used for pressure fueling to the tanks.

FUEL JETTISON NOZZLE

The fuel jettison nozzle is provided in the jettison system to discharge fuel at the wig tips of the airplane. The fuel jettison nozzle is a 3-inch diameter aluminum tube extending approximately 4 inches past the trailing edge of the wing.

MAIN FUEL TANK JETTISON PUMPS

The main jettison pumps deliver fuel from the main fuel tanks to the jettison manifold during the jettison operation. The pumps are identified as No. 2 outboard, No. 2 inboard, No. 3 inboard and No. 3 outboard. Each pump assembly consists of an AC electric motor with an impeller attached to the shaft.

The motor is powered by three-phase AC current. A non-resettable thermal fuse is integrally installed in each phase winding of the motor. At a predetermined temperature, the fuse will actuate. Actuation of a fuse will necessitate replacement of the motor. The motor is cooled and lubricated by circulation of fuel through the motor housing.

MAIN JETTISON PUMP CONTROL SWITCHES

The main jettison pump switches control power to the jettison pump relays. The switches are located on the fuel jettison panel at the Flight Engineer's station.

MAIN JETTISON PUMP RELAYS

The main jettison pump relays enable three-phase AC electrical power to be supplied the main jettison pumps. The relays are controlled by 28 volt DC power from the main jettison pump control switches on the fuel jettison panel. Relays from tanks No. 2 outboard and No. 3 outboard jettison pumps are located in the left main power center (P14). Relays for tanks No. 2 inboard and No. 3 inboard jettison pumps are located in the right main power center (P15).

JETTISON SYSTEM STANDPIPES

Jettison pumps 2 and 3 inlets are connected to standpipes set at the 7,000 pound level to prevent inadvertently draining the tanks. Outboard to Inboard main tank transfer valves are also connected to standpipes at the 7,000 pound level.
This assures 28,000 pounds will remain in the main wing tanks for engine operation.

OVERRIDE /JETTISON PUMPS

The override / jettison pumps through the jettison valves deliver fuel from the center wing tank to the fuel jettison manifold during the jettison operation. Each pump is controlled by a separate switch on the fuel control panel.
The override / jettison pumps can lose their prime when operated dry. Fuel from the discharge line of each refuel valve will prime the override / jettison pumps whenever the center wing tank is pressure fueled.
The override / jettison pumps are identical to the main jettison pumps. The pumps are also used during normal operation to deliver fuel from the center wing tank to the engine fuel feed system.

OVERRIDE/ JETTISON PUMP SWITCHES

The override / jettison pump switches control 28 volt DC power to the override / jettison pump relays. The switches are located on the fuel control panel.

OVERRIDE / JETTISON PUMP RELAYS

The override / jettison pump relays enable three-phase AC electrical power to be supplied to the override / jettison pumps. The relays are controlled by override / jettison pump switches on the fuel control panel. The left pump relays is located in the left main power center (P14) and the right pump relay is located in the right main power center (P15).

FUEL JETTISON NOZZLE VALVES

Two fuel jettison nozzle valves provide a means of controlling the fuel flow to the fuel jettison nozzles. The valves are located in the fuel jettison manifold in the wing tips, outboard of the surge tanks. The valve assembly consists of two separate subassemblies, a valve and an electric actuator.

FUEL JETTISON NOZZLE VALVE CONTROL SWITCHES

The nozzle valve switches control 28 volt DC power to the jettison nozzle valves. The switches are located on the fuel jettison panel and are identified as NOZZLE VALVES, LEFT and RIGHT.

FUEL JETTISON NOZZLE INDICATOR LIGHT

The fuel jettison nozzle indicator lights are installed on the fuel jettison panel to indicate position of the valves. The lights are located between the nozzle valve switches. When the nozzle valves are in transit the lights are illuminated and are extinguished when the valves are open or closed.

CENTER WING TANK JETTISON VALVES

Two center wing tank jettison valves direct the flow of fuel from the override / jettison pumps to the jettison manifold.

CENTER WING TANK JETTISON VALVE CONTROL SWITCHES

The jettison valve switches control 28 volt DC power to the jettison valves. The switches are located on the fuel jettison panel and are identified as CENTER WING JETTISON valves.

CENTER WING TANK JETTISON VALVE INDICATOR LIGHTS

The jettison valve indicator lights are installed on the fuel jettison panel to indicate position of the valves. When the valves are in transit the lights are illuminated and are extinguished when the valves are open or closed.

The fuel jettison system becomes operative when 28 volt DC power is supplied to he pump relays and jettison valves, and the jettison pumps are energized by 115 volt AC power.

The fuel jettison system is operated by placing the main jettison pump and the override/jettison pump switches to ON position. Fuel discharged from the main jettison pumps will pressurize the fuel jettison manifold. Placing the jettison transfer valve switches to the OPEN position will allow fuel under pressure from the override/jettison pumps to enter the fuel jettison manifold. The jettison nozzle valve switches are then placed to the OPEN position to allow fuel in the jettison manifold to be discharged at the wing tips through the jettison nozzle.

Immediately after placing the jettison nozzle valves to OPEN position, the reserve tank and main tank transfer valve switches should also be placed to the open position. This would allow fuel from the reserve and outboard main tanks to flow into the inboard main tanks.

The fuel jettison operation can be terminated at the discretion of the operator by closing valves and shutting off the pumps. If the operation is not terminated, fuel will continue to be dumped overboard until a required quantity of unjettisonable reserve fuel remains in the tanks.

To operate the fuel jettison system, 28 volt DC and 115-volt AC power must be available on the airplane. It is assumed that all circuit breakers are closed and that power is available to all jettison system components.

FUEL MEASURING STICK ASSEMBLY

The fuel measuring stick assemblies are provided as a manual means of measuring the quantity of fuel in the tanks. The measuring sticks can also assist the ground crew in fueling operations by allowing a comparison between the measuring sticks and the fuel quantity indicators in the airplane. The measuring sticks are installed at 15 different locations. One stick is located in each reserve tank, three sticks in each inboard and outboard main tank, and one stick in the center wing tank. The lengths and configurations of the sticks vary, depending on the tank and location within the tank in which they are installed.

MEASURING STICK

The fuel measuring stick is a calibrated stick, which provides a manual means of determining fuel quantity in the fuel tanks. The measuring sticks are installed in the lower surface of the wing except the measuring stick in the center wing tank. The measuring stick in the center wing tank is installed in the tank bottom and can be reached by opening an access panel on the wing-to-body fairing.

To operate the fuel level measuring stick, the stick is unlocked from the base by applying a slight upward pressure and a 90 degree counterclockwise turn to the end of the stick. When the stick is unlocked, it is lowered to the fully extended position. Depending on fuel level, a bearing located on top of the stick will magnetically latch onto a magnet in the float before full extension. At the fully extended position, the stick stops against a cushion giving a soft firm feel.

To determine the fuel level, the stick is gently raised until the magnetic latch is felt to engage. There is a positive feel in this position since the magnetic latch is strong and will jerk the stick into position. The stick is now read to determine fuel level in the tank.

If the measuring stick is being monitored during a refueling or defueling operation, the float and stick will move with the fuel level. However, since the magnetic attachment between the magnets of the stick and float causes a small friction against the housing, the stick and float will not move up the float to overcome the static friction. On declining fuel levels when defueling, friction will occur in the opposite direction. Calibration marks on the stick are set to read on rising fuel levels and the most accurate reading will be immediately after a jump of the stick.

The stick is returned to its original position by raising it into the base and with a slight upward pressure, the head is turned 90 degrees clockwise to lock the stick in place.

HYDRAULIC POWER

Hydraulic power is supplied by four independent systems. Each system incorporates an engine driven hydraulic pump and an air driven hydraulic pump. The pumps, reservoir, and associated components are located in the engine and nacelle areas. An electrically driven hydraulic pump in system 4 is provided primarily for operation of the brake system during towing operations. The electric pump is powered by the ground handling bus (APU or External Power only).

Hydraulic power is used to operate the landing gear and flight controls. The distribution of hydraulic power is designed so that each primary flight control axis receives power from all four hydraulic systems. Each primary control surface is powered by a dual hydraulic actuator supplied by two hydraulic systems except for the outboard elevators which are powered by a single source of pressure.

System 1 is basically assigned to the inboard systems such as the inboard flaps and the inboard gear (nose and body gear operation and steering). System 4 is basically assigned to the outboard flaps and gear (wing gear). System 2 powers flight control surfaces controlled by autopilot channel "B", and system 3 powers flight control surfaces controlled by autopilot channel "A".

The only selectability between hydraulic systems is in the brake system which uses system 4 as a primary source and system 1 as a secondary source. A reserve brake system, when activated, uses system 2.

Hydraulic system assignment for the ailerons, spoilers, CCA's, elevators, and rudders is placarded adjacent to the flight control Hydraulic Power Switches on the pilots' overhead panel.

Hydraulic System assignment for the stabilizer trim is placarded adjacent to the Stabilizer Hydraulic Shutoff Switches on the pedestal.

When a control surface is redundantly powered, only one system is required to operate the surface.

Trim movement and braking of the horizontal stabilizer is provided by two independent hydraulic motors and brakes; one pressurized from system 2, the other from system 3.

ENGINE DRIVEN HYDRAULIC PUMP (EDP)

The engine driven hydraulic pump is a variable displacement type pump with the output pressure regulated to approximately 3000 PSI. The pump is capable of operating at a reduced output with engines windmilling. The engine pump is controlled by the engine driven hydraulic pump switch on the Flight Engineer's panel.

The pump is equipped with a solenoid valve which functions to close the pump outlet port. The valve is normally open (de-energized), and when energized, will close and shut off all delivery flow from the pump and reduce pump output pressure to 800-1000 PSI. Control of the solenoid valve is accomplished by placing the ENG PUMP switch on the Flight Engineer's hydraulic control panel to the NORMAL position to de-energize the valve and to DEPR or SUPPLY OFF positions to energize and close the valve. The EDP is mounted on the right side of the engine accessory gear box.

AIR DRIVEN HYDRAULIC PUMP (ADP)

The air driven hydraulic pump operates to supplement the engine driven hydraulic pump during high demand periods. The air pump is capable of supplying the normal

system demands in the event of engine driven pump failure or after Fire Switch actuation. The air pump is driven by bleed air from the pneumatic system. It can be supplied from engine or APU bleed air, or a ground air source. With the switch in the AUTO position, the pump will run as required to maintain system pressure.
The air pump has an overspeed trip circuit that will automatically close off the pneumatic supply to the air pump. This condition will be reflected by failure of the pump to operate normally in any switch position.
The ADP is physically identical and interchangeable with the EDP. Operation and control are different.

HEAT EXCHANGER (HYDRAULIC FLUID COOLING)

The hydraulic fluid heat exchanger is used to cool pump case drain fluid before it enters the system reservoir. The heat exchanger for each system is installed in its respective main fuel tank and cools the case drain module is routed to the heat exchanger through the coils of the heat exchanger and to the return module. The heat exchanger consists of a coiled tube assembly having approximately 6 112 coils of tubing.
The tube end fittings of the heat exchanger coil assembly project through sealed holes at the rear of the fuel cell (rear wing spar). The length of coiled tubing in each of the four main system heat exchangers is the same. However, the position of the tube end fittings in respect to the coils is different for each heat exchanger and the unit for one system cannot be interchanged with one for any for the other systems.
RESERVOIR
The hydraulic reservoir for each main system supplies hydraulic fluid to accommodate the pressure flow demands of the various hydraulically actuated components of the system. The design and operation of the reservoirs are the same and differ only in capacity. The reservoirs for supply systems No. 1 and 4 are each serviced to 9.5 gallons while the systems No. 2 and 3 reservoirs are each serviced to 5.5 gallons. Reservoir pressurization is provided to each reservoir. Each reservoir includes a hydraulic fluid level transmitter, a sight glass level indicator, a manual drain valve, a negative -G trap, and an air pressurization relief valve.
HYDRAULIC SYSTEM CHECK VALVES
Check valves are installed in the hydraulic power systems to prevent reverse flow of fluid, to direct fluid flow, prevent fluid loss, prevent drainage during maintenance and to isolate subsystems. Flow direction through a check valve is shown by an arrow on the valve body. Normally no maintenance other than removal and installation is required of a check valve.

HYDRAULIC FLUID SHUTOFF VALVE

A hydraulic fluid shutoff valve located in the nacelle area controls hydraulic fluid supply to the engine driven pump only. Operation of this valve is controlled by either the engine fire switch or the Flight Engineer's EDP switch. When the fire switch is pulled, the EDP is depressurized first, then the supply shutoff valve closes. In order to place the Flight Engineer's EDP switch to the supply shutoff position, it must first pass through the "DEPRESSURIZE" position. In this manner, at least some fluid will remain trapped in the EDP for cooling and lubrication. No such valve is needed for the ADP because the ADP can be stopped by turning it off.

CONTROL

Each system is displayed at the Flight Engineer's panel, with a quantity gauge, a low quantity light, 2-three position switches for the ADP & EDP, a Blue "Run" light for the ADP, a low pressure light for each pump, a pressure gauge for each system, an overtemperature light with temperature gauge and a quantity test button.
To the right of the dual temperature gauges is the guarded No. 4 A.C. hydraulic pump switch, powered by the Ground Handling bus.
To the left of the dual temperature gauges is the Normal Brake Source Select panel. It contains a two position switch marked PRIMARY SYS. 4 and SECONDARY SYS. 1. It is guarded to PRIMARY SYS. 4. An amber low pressure light and a green, "SEC. SYS. 1" light are also incorporated.
The pilots' center panel annunciator contains 4 amber system low pressure lights, classed as summary lights. They illuminate only when both ADP's and EDP's are inoperative.
At the Captain's lower right instrument panel, there is a Reserve Brake Panel. It has a RESERVE BRAKE valve switch (Guarded closed), a Green "VALVE OPEN" light and -an amber "BRAKE SOURCE" low pressure light.
At the bottom left of the Co-Pilots' instrument panel is a brake pressure gauge. It reads accumulator air charge or hydraulic pressure in the normal brake system only. The air pre-load is 750 PSI.

HYDRAULIC PRESSURE INDICATION

Four hydraulic pressure indicating systems, one for each of the main hydraulic supply systems, provide remote indication of fluid pressure in the main hydraulic systems. Each system consists of a pressure transmitter and indicator for remote indication of pressure.

HYDRAULIC LOW PRESSURE WARNING

The low pressure warning system operates whenever the engine-driven pump output pressure is below the minimum operational level. The low pressure warning switches are set to open at 1200 (+250) PSI, on pressure increase, and close at least 100 PSI below the opening pressure, but no lower than 700 PSI on decreasing pressure.
When the engine-driven pump pressure switch closes, the control relay closes the contacts, switching the control of the HYD SYS PRESS light on the pilots' center instrument panel to the air-driven pump pressure switch. When air-driven pump pressure is normal the HYD SYS light remains extinguished. If the air-driven pump pressure is low, the pressure switch closes and illuminates the HYD SYS PRESS light.
HYDRAULIC FLUID OVERHEAT WARNING
The hydraulic fluid overheat warning system provides visual indication of an overheat condition of case drain fluid in each of the four main hydraulic supply systems. Each system consists of a cartridge-type overheat switch located in the case drain module and an amber warning light located on the Flight Engineer's panel.
An overheat switch is located in each case drain module to monitor the case drain flow from the pumps. The switch closes at a temperature of 104°C (±3) and opens at

79.5°C (±4.5).

When the case drain flow of either system pump exceeds a temperature of 104°C, the overheat light will illuminate. When the temperature drops to 79.5°C, the overheat light will extinguish. The system is energized by 28 volt DC power provided by the FLT ENGR WARN LTS circuit breaker on main power circuit breaker panel P6.

Ice and Rain Protection

The Ice and Rain Protection systems are designed to orotect the airplane and aid the pilots when operating under ice and rain conditions

The following systems are used on the B-747

1. Engine Nacelle Anti-Ice
2. Wing Anti-Ice
3. Window Heat and Defog
4. Pitot Static Heat
5. Rain Removal and Window Washer

The engine inlet cowl and PT2 EPR probe are anti-iced by engine bleed air Engine bleed air from the pneumatic system is routed to the inlet cowl and PT2 EPR probe areas through the nacelle anti-ice valve which limits the maximum pressure of the bleed air. At low duct pressures the nacelle anti-ice valve will close and the NACELLE VALVE OPEN LIGHT will extinguish. In the event of a valve malfunction allowing higher pressure air into a nacelle anti-ice system, the NAC TAI (Thermal Anti-Ice) VALVE light will illuminate as a warning. Icing of the PT2 EPR probe will be indicated by abnormal EPR indications.

NACELLE ANTI-ICING CONTROL VALVES

An anti-icing control valve is provided for each engine. The valve is a 3-inch solenoid-operated valve. When energized, the valve regulates the pressure into the ducts within 20 to 25 psi. When duct pressure drops to 4 ps! the valve will be closed

The valve is normally closed and operated by energizing the solenoid. When the solenoid is energized and the butterfly is open for air flow, a pilot regulator senses the downstream (outlet) pressure and modulates the inlet pressure admitted to the actuator piston. The piston thus positions the butterfly to maintain outlet pressure between preset limits.

NACELLE ANTI-ICE CONTROL MODULE

The nacelle anti-ice control module is located on the pilots' overhead panel. This module has a switch and an indicator for each engine nacelle TAI system. Placing a switch ON energizes the valve for that engine, and the corresponding NACELLE VALVE OPEN light illuminates.

NACELLE ANTI-ICE OVERPRESSURE WARNING SWITCH

The overpressure warning switch is mounted on the nacelle structure and is connected to the duct with a flexible line. The switch is connected downstream of the anti-ice control valve and will indicate an overpressure malfunction of the system. The overpressure switch contacts close at 26 to 35 psi with an increasing pressure and open at 26 psi with decreasing pressure in the ducts.

WING ANTI-ICE

The leading edge of the wings from the wing tip to the inboard engine is protected from ice accumulation by hot air from the pneumatic manifold. There is no anti-ice protection provided for the wing root or the empennage. Tests have shown this is not

required.
The flow into the wing anti-icing system is controlled by two shut-off valves, one for each wing. Both valves are actuated by a single three-position switch, ON-OFF-GRD Test, located on the cockpit overhead panel. The system is deactivated when the airplane is on the ground by the landing gear tilt switches. A Ground Test switch position is provided to check valve operation on the ground. During the ground test, the valve will automatically close when wing leading edge temperatures become excessive.

WING ANTI-ICE CONTROL VALVES

The flow of air into the wing TAI system from the engine pneumatic system is controlled by a 3-1/2 inch shutoff valve in each wing. The valves are of the butterfly type, driven open and closed by an electric motor within the valve assembly. The valves have limit switches to prevent over travel, and valve position switches which are used in the valve actuation. Each valve has an external position indicator which shows the position of the valve, and a handle which allows the valve to be opened manually.
The control switch allows the system to be turned on, off, or ground tested. During the ground tests, the overheat relay and the overheat switch prevents the system from heating the leading edge over 200°F. The VALVE indicators are off when the valves are in the fully opened position, brightly lighted when the valves are opening or closing, and off when the valves are closed. The VALVE lights will also illuminate if the valve position does not agree with the switch positions.

PROBE HEAT

The left and right main pitot-static probes, left and right auxiliary pitot-static probes, and the No. 1 and No. 2 total air temperature probes are heated to prevent ice formation which would affect sensing accuracy.

OPERATION

Probe heat is controlled by two three-position switches, one for the left side and one for -the right. These switches are located on the pilots' overhead panel.
GROUND: With PROBE HEATERS switches OFF, pitots and probes are not being heated and PITOT and TAT lights are on.
With PROBE HEATERS switches ON, PITOT and TAT lights are out and pitots are being heated. TAT probes are armed and will automatically be turned on by nose gear air/ground logic circuits when airborne.
FLIGHT: With PROBE HEATERS switches ON, all pitot and TAT probes are heated; PITOT and TAT lights are out.
TAT TEST: When the PROBE HEATERS switch is held to the TAT TEST position, nose gear air/ground logic is bypassed and 115V AC is applied to the associated temperature probe. TAT PROBE light should be out.

ANNUNCIATOR LIGHTS:

Probe heater annunciator lights (6) on the overhead control are provided to show that heating current is being supplied to the probes. If current is not being supplied when it is supposed to be, the associated light(s) will be illuminated.

Whenever a master probe heater annunciator light is illuminated, the PROBE HEAT light on the forward caution panel will also be illuminated.

STALL WARNING SENSOR HEAT

The stall warning sensor is a vane located on the left side of the nose and is electrically heated for anti-icing.

Vane heat is controlled by a three-position STALL WARNING switch located on the pilots' overhead panel.

A power off (PWR OFF) light indicates status of both system power and heater current through the sensor vane.

Window Overheat Test Switch (Must be held in during test)

CAUTION: DO NOT TEST IN FLIGHT.

Simulates an overheat condition for both No. 1 windows with the Window Heat Switches On and the Power on lights illuminated (window temperature below control temperature). No. 1 POWER on lights will extinguish. WINDOW 1 OVHT Light (on F/E's Annunciator Panel) will illuminate.

NOTE: After test, switch the No. 1 Window Heat Switches to OFF position momentarily to reset overheat trip circuits.

Window Heat Power ON Lights (Green) -- Illuminated when window is being heated (Window Heat Switches ON, but window temperature below control temperature). Lights cycle on and off as power is being applied and removed from window. Lights can be deactivated by Power Lights Switch.

WINDOW HEAT

The cockpit windows are electrically heated to provide anti-icing and defogging for the Number 1 left and right windows. The Number 2 and 3 L and R windows are provided with defogging only. The controls and indicator lights for normal system operation are located on the cockpit overhead panel. An overheat warning light is installed on the Second Officer's annunciator panel. Power to each window will automatically cycle to maintain correct operating temperature. An overheat thermostat in L-1 and R-1 windows will cut off power to that window should it overheat. In addition to window heat, conditioned air may be used to defog the interior surface of the Number 1 windows. A knob located at the base of the flight instrument panel controls the operation of this system.

WINDSHIELD WASHER AND WIPERS

A windshield washer system for the Number 1 windows is installed for ground use. The windshields may be washed at any time with a system very similar to that used on modern automobiles except that separate controls are provided for the left and right side. The washer fluid container is located behind the Captain's side panel with a view port for inspection purposes. A fluid level mark on the side panel indicates the refill level. The system is controlled from the windshield wiper control module by two ON-OFF switches, springloaded to the OFF position.

To operate the windshield washer, hold washer toggle switch on for approximately two seconds. Set wiper switch to low and operate washer for two more seconds. Allow windshield wiper to operate until area is nearly dry. This procedure will clean the most bug encrusted windshield.

The Number 1 windows are provided with windshield wipers to maintain a clear area during takeoff, approach and landing in rain or snow.

Two wiper speeds may be selected by the control switch at the panel; HIGH or LOW. The OFF position is the "PARK" position. In case the wiper motor overheats, thermal overload protection will interrupt power to the motor. After the motor cools, operation is automatically resumed.

CAUTION: DO NOT OPERATE WINDSHIELD WIPER ON DRY WINDSHIELD AS THE WINDSHIELD MAY BE DAMAGED.

RAIN REPELLENT SYSTEM

A rain repellent system is provided to work in conjunction with the windshield wipers to improve visibility during heavy rain.

The system provides independent control for each windshield by separate control switches.

The maximum quantity of fluid sprayed per system activation is predetermined by the setting on a timer and cannot be exceeded by extended duration of switch actuation.

CAUTION: THE RAIN REPELLENT SYSTEM SHOULD NOT BE OPERATED ON A DRY WINDOW.

The rain repellent system consists of a pressurized container of rain repellent fluid, a visual reservoir and two nozzles which are shared wit the windshield washer system.

Instruments

The flight instruments are the primary source of flight reference information for the crew. This section will describe the following:

1. The Air Data System Instruments which relate the airplane to the local air mass in terms of speed, altitude and temperature.
2. The Flight Guidance Instruments which relate the airplane to its path over the earth in terms of attitude, turn rate, heading, radio altitude, and airplane position relative to a selected radio course or GPS navigation track.
3. The Central Instrument Warning System which monitors the flight guidance instruments to bring discrepancies to the crew's attention.
4. The Standby Attitude Indicator and clocks.
5. The Flight Data Recorder which records specific flight information.

Air Data Instruments include:
Airspeed Indicators
Altimeters
Vertical Speed Indicators
~ Total Air Temperature Indicator (TAT)
Static Air Temperature Indicator (SAT)
True Airspeed Indicator (TAS)
MACH Indicators (MACH)
Calibrated Airspeed Indicator (CAS)

Pitot Static probes and Total Air Temperature probes collect air pressure and temperature data. This data is provided directly to some instruments. This data is also provided to 2 Central Air Data Computers, which process the information and provide additional information to display on the instruments.

Data collected by the probes includes:
Total or Pitot pressure (Pt)
Static Air pressure (Ps)
Total Air Temperature (TAT)

Flight Guidance Instruments include:
Attitude Direction Indicators (ADI)
Horizontal Situation Indicators (HSI)
Radio Magnetic Indicators (RMI)

The flight guidance instruments receive data from several sources including the Compass system, Attitude Heading Sensing System, VHF navigation radios, ADF navigation radios, and Global Positioning System computers.

The flight guidance instruments and data displayed on the instruments, is monitored by the Central Instrument Warning System, which will advise the crew of failures and/or invalid data being displayed. Warning and caution lights are located on each pilots instrument panel.

PITOT-STATIC SYSTEMS

The pitot-static system senses the dynamic or total pressure (pitot) and ambient pressure (static) outside the airplane to determine and display the vertical speed, airspeed and altitude.

The pitot-static system comprises the following components:

- Four independent pitot systems
- Four independent static systems
- One alternate static system.

Pitot Probes

The pressures of the four independent pitot and static systems are sensed at four probes (one upper and one lower on each side of the forward fuselage). The upper probes are the primary probes and the lower probes are the auxiliary probes. Each probe has three sensing ports including one pitot and two static.

The alternate static pressure system has two flush mounted static ports, one on each side of the fuselage.

The pitot probes are electrically heated by two elements in each probe, to prevent ice formation. The alternate static ports are not heated.

Total Air Temperature probes (TAT)

Two Total Air Temperature probes are installed, one in front of each pilots number 1 window. These probes sense outside air temperature subject to ram rise. This temperature is displayed directly on the TAT indicator on the P-2 panel. This temperature is also sent to the Central Air Data Computer, where it is corrected, and displayed on the Static Air Temperature indicator on the P-3 panel.

The probes are electrically heated, when NOT on the ground.

Pitot System

The captains pitot pressure is supplied by the upper left pitot probe.

The first officers pitot pressure is supplied by the upper right pitot probe.

Each probe supplies pressure to drive the corresponding Indicated Airspeed pointer and the central air data computer (CADC).

The lower left and lower right probes supply pressure to the rudder ratio changers, elevator feel computers, and the stabilizer rate controllers. In addition, the first officers aux pitot supplies pressure to the Mach-Airspeed Warning and the airspeed switches.

Static System

Each static system is balanced by sensing pressure on each side of the fuselage into a common line. This assures accurate static pressure regardless of airplane attitude.

The Captains static pressure is supplied from the upper left and lower right probes.

The First Officers' static pressure is supplied from the upper right and lower left probes.

Auxiliary static pressure No. 1 is supplied from ports in the upper left and lower right pitot probes. This is supplied to the lower rudder ratio control.

Auxiliary static pressure No. 2 is supplied from ports in the upper right and lower left pitot probes. This is supplied to galley/lav fan sensing, Mach/Airspeed Warning, airspeed switches, upper rudder ratio control, Cabin Pressure Auto Controller and Cabin Differential Pressure Gauge.

Alternate Static System

The alternate static system is supplied from flush mounted ports on each side of the fuselage. This is provided as a backup system if the Captains or First Officers

primary static system becomes unreliable.
Static Source Selector
Each pilot has a Static Source Selector, located on the outboard side console, near the floor. The selector lever is guarded to the NORMAL (UP) position.
This is a manual selector valve.
When NORMAL is selected, the corresponding pilots primary static pressure is routed to his airspeed indicator, altimeter, vertical speed indicator and CADC.
When ALTERNATE is selected, alternate static pressure is supplied to the corresponding instruments and CADC.

CENTRAL AIR DATA SYSTEM

The Central Air Data System uses 2 computers (CADC), to provide corrected data to the following components:
Air Data Instruments
GPS Navigation Processing Unit (NPU)
Autopilot and flight director computers
Ground proximity warning system (including TAWS)
ATC transponders
Flight data recorder
Each of the Central Air Data Computers (CADC) uses the same pitot and static pressures as the flight instruments of the corresponding pilot.
To provide a reference temperature for the direct reading Total Air Temp gage and for the CADC to compute Static Air Temp and True Airspeed, a temperature sensing probe is mounted just forward of each pilots number 1 window.
Air Data Instruments:
Data provided to the flight instruments includes:
Mach number(MACH), computed by the respective CADC.
Calibrated airspeed (CAS), which is IAS corrected for the static source position error.
Altitude (ALT), which is corrected for static source position error by the CADC and displayed when the switch on the altimeter is positioned to CADC.
Total Air Temperature (TAT), which is sensed directly at the probe and not corrected by the CADC.
Static Air Temperature (SAT), which is TAT corrected for compressibility (temperature rise).
True Airspeed (TAS), which is computed by the CADC #1 but is displayed on the First Officers panel.
Autopilot and flight director computers:
These computers are supplied altitude data, TAS and CAS.
Altitude is supplied from CADC #1 for altitude HOLD, and CADC #2 for altitude SELECT.
Vertical speed is supplied from CADC #1 when both autopilots are OFF, or when Autopilot A is engaged. It is supplied from CADC #2 when autopilot B is engaged.
TAS is required to limit the maximum bank angle when flying on autopilot in the HDG mode.
CAS is required when operating in the IAS mode.

GPS Navigation Processing Unit (NPU):
The GPS NPU needs TAS and Magnetic Heading information, to compare with track and ground speed, in order to compute the wind. The CADC provides the TAS information. The Attitude Heading Sensing Unit (AHSU) (DG and compass coupler) provides gyro stabilized Magnetic Heading.
If TAS or Mag Heading are not received, the GPS notifies the crew with a message, and requests that TAS be manually entered.
If data which was previously entered is generating incorrect wind data, delete the manual data by using the CLEAR and ENTER keys. After the manual data is zeroed out, the CADC
data will automatically be used.
Ground Proximity Warning System:
Altitude rate of change data is supplied from CADC #1.
Flight Data Recorder:
CAS and altitude data are supplied to the flight recorder from CADC #1.
Altitude Alert:
Altitude information is supplied from CADC #2 (non switchable)
Altitude Hold:
Altitude information is supplied from CADC #1. (non switchable)
Altitude Select:
Altitude information is supplied from CADC #2. (non switchable)
Transponders:
Altitude information for both transponders is supplied from CADC #1 or CADC #2, as selected by the ALT switch on the transponder controller.

COMPASS SYSTEM

Two compass systems are installed (No. 1 and No. 2), to provide magnetic heading information.
A sensor called a Flux Valve is installed near each wing tip, in a dry bay area. Errors caused by installation position and interference from aircraft structure are corrected for each system by compensators.
A compass controller for each system is installed on the P-5 overhead panel.
Two operating modes can be selected on the controller, SLAVED and DG.
In the normal SLAVED position the heading provided to the compass coupler is slaved to magnetic heading . This heading is stabilized by the directional gyro in the AHSU.
In the DG or free position, the heading is selected with the HDG SEL knob and deviations are detected by the directional gyro in the AHSU. This is used in areas of magnetic unreliability, for Grid navigation.
The AHSU directional gyro connected to the compass system coupler changes when the associated pilots' ATTITUDE switch is repositioned.
Captain's ATTITUDE switch:
NORM - AHSU No. 1
STBY - AHSU No. 3
F.O.'s ATTITUDE switch:
NORM - AHSU No. 2
STBY-AHSU No. 3

With the RADIO/GPS switches positioned to RADIO, and both pilots ATTITUDE switches set to NORM;
#1 compass system Magnetic Heading is displayed on the Captain's HSI and the F.O.s RMl.
#2 compass system Magnetic Heading is displayed on the F.O.s HSI and the Captain's RMI.
When the RADIO/GPS switches are positioned to GPS, Magnetic Heading is still provided to the RMIs. The HSIs will display MAG (default) or True heading, as selected in the GPS CDU Aux page.
Each pilot has a compass system transfer switch on the lower P-1 and P-3 panels.
The Captain normally uses system #1, but can select #2 if required.
The First Officer normally uses system #2, but can select #1 if required_
Magnetic Heading is also provided to the Autopilot and Flight Director roll computers, GPS Nav Processing Units, and the flight recorder.
Magnetic Headings displayed on the HSIs are monitored by the Central Instrument Warning System, discussed later in this chapter.

ATTITUDE HEADING SENSING SYSTEM

The Attitude Heading Sensing System provides gyro stabilized attitude and heading information.
This is provided by three individual Attitude Heading Sensing Units (AHSU). Each AHSU contains a vertical gyro (VG) and a directional gyro (DG).
The AHSUs are located in the E/E compartment.
The AHSU operates as soon as power is applied to the airplane.
The #1 AHSU is powered by Standby AC, , The #2 AHSU is powered by Radio bus 2 and AHSU number 3 is powered by either the Essential Flight Instrument bus or Radio bus 2, depending on the position of the F.O.'s Attitude select switch.
The AHSU supplies attitude, heading, comparison and validity information to the ADI, HSI and RMI. Additionally it supplies pitch and roll data for the autopilot, digital flight data recorder, and weather radar antenna stabilization.
The AHSU - VG and DG outputs are controlled by the ATTITUDE switch on the Captain's.
P-1 panel and the First Officer's P-3 panel. These switches operate the respective attitude transfer relays.
The AHSU VG and DG output signals are controlled as follows:
AHSU #1 VG, with Captain's attitude switch in NORMAL, supplies:

- Pitch and Roll to Captain's AD] (switchable)
- Pitch and Roll To WX Radar #1 (hard wired)
- Pitch and roll to autopilot and flight director A (hard wired)
- Pitch and Roll to flight director C, if installed, (hard wired)

AHSU #1 DG, with Captain's attitude switch in NORMAL, supplies:

- Heading to the #1 compass coupler

AHSU #2 VG, with First Officer's attitude switch in NORMAL, supplies:

- Pitch and Roll to the First Officer's ADI (switchable)
- Pitch and Roll to WX Radar #2 (hard wired)
- Pitch and Roll to Autopilot and flight director B (hard wired)

AHSU #2 DG, with First Officer's attitude switch in NORMAL, supplies:
- Heading to the #2 compass coupler

AHSU #3, the alternate VG, supplies:
- Pitch and Roll to the Captain's ADI, it ALT selected, or
- Pitch and Roll to the First Officer's ADI, if ALT Selected

AHSU #3, the alternate DG, supplies:
- Heading to the #1 compass coupler, if captain selects ALT, or
- Heading to the #2 compass coupler, if first officer selects ALT.

COMPASS SWITCH -Used to select which compass system will be referenced to HSI and RMI indicators.
NORM (guard down) -
The Captain's switch is normally guarded down to No. 1.
Compass system #1 provides magnetic heading information to the captains HSI and the first officers RMI.
The First Officer's switch is normally guarded down to No. 2.
Compass system #2 provides magnetic heading information to the first officers HSI and the captains RMI.
ALT (guard open) -
If the Captain's switch is positioned to ALT, compass system #2 provides magnetic heading information to the captains HSI and the first officers RMI.
If the First Officer's switch is positioned to ALT, compass system #1 provides magnetic heading information to the first officers HSI and the captains RMI.
ATTITUDE SWITCH- Used to select which AHSU source will provide pitch and roll information to the respective ADI and heading stabilization to the respective compass coupler.
NORM (guard down)-

With the Captain's switch in NORM, his AD[receives pitch and roll information from VG #1 and the No. 1 compass coupler receives heading stabilization from DG #1.
With the First Officer's switch in NORM, his ADI receives pitch and roll information from VG #2 and the No. 2 compass coupler receives heading stabilization from DG #2.
ALT (guard open) -
With the Captain's or First Officer's attitude switch in ALT, his ADI receives pitch and roll information from VG #3 and the respective compass coupler receives heading stabilization info from DG #3.
Note: The Captain and First Officer cannot select ALT at the same time. The first switch positioned to ALT will transfer to AHSU No. 3.
ATTITUDE DIRECTOR INDICATOR (ADI)
An Attitude Director Indicator (ADI) is installed on each pilot's panel. Conventional pitch and roll attitude information, supplied by the AHSU gyros, is displayed by the horizon sphere. The sphere is driven by a pitch servo and a roll servo. Pitch and roll command bars on the sphere are positioned by a Flight Director Computer. The ADI also displays:
- Glideslope deviation.
- Expanded localizer course deviation.

- Turn and slip indications.
- Runway closure rate below 200-ft radio altitude.
- Speed deviations relative to command airspeed (red) bug.

Horizon Sphere

The horizon sphere of the ADI receives pitch and roll information from primary and alternate sources as selected by the Captain's and First Officer's attitude transfer switches.

Pitch and Roll Command Bars

The pitch and roll command bars are positioned by information received from the selected Flight Director Computer. The AP/FD computers receive pitch and roll signals from the respective AHSU.

AP/FD computer A is "hard-wired" to AHSU No 1

AP/FD computer B is "hard-wired" to AHSU No. 2

HORIZONTAL SITUATION INDICATOR (HSI)

The Horizontal Situation Indicator (HSI) displays airplane position in relation to a selected VOR course or GPS track. Each pilot has a RADIO-GPS switch on the glareshield panel to select a radio or GPS display for his HSI.

A window in the face of each HSI indicates the type of information displayed; either GPS 1, GPS 2, RAID 1 or RAID 2. When a GPS display is selected, the word TRACK appears under the lubber line. When a radio display is selected, MAG appears under the lubber line.

The HIS's also display:

- MILES NO. 1, MILES NO. 2 distance to the next waypoint on the respective GPS. The miles display will function with either GPS or RADIO display selected.
- A steady ALERT light when within two minutes of the next GPS waypoint. At the waypoint, the ALERT light will go out as the GPS shifts to the next leg.

The ALERT light indications will only function when a GPS display is selected.

HSI Indications With GPS Selected

These indications are automatically supplied from the GPS.

When a GPS display is selected, the HSI indicates:

- Present track on the azimuth card at the lubber line.

NOTE: True or Mag heading may be displayed in the GPS mode. If true heading is being presented, TRUE will be displayed on the HSI instead of TRACK.

- True heading in the HEADING window, and by the heading bug on the azimuth card.
- Desired track in the COURSE window and by the course indicator on the azimuth card.
- Cross-track deviation by the course deviation bar on the dot scale. A full-scale displacement of two dots is equal to 8 nautical miles cross-track deviation. When the course deviation bar is centered with the course indicator on the lubber line, the airplane is on the desired track.
- Drift angle is indicated by the angular displacement between the heading bug and the lubber line.

- To the waypoint by the triangular flag used for VOR.

HSI Indications With RADIO Selected

When a RADIO display is selected, the HSI indicates:

- Airplane magnetic heading on the azimuth card at the lubber line,
- VOR/LOC deviation by the course deviation bar.
- Glideslope deviation on the glideslope deviation scale.
- Course and heading information as manually selected by the pilot.

RADIO MAGNETIC INDICATOR (RMI)

Gyro stabilized magnetic heading is indicated on the radio magnetic indicators (RMIs). The azimuth card on the captain's RMI is normally driven by the No. 2 compass system, but can be driven by the No. 1 system if the Captain's COMPASS transfer switch is placed in the No. 2 position. The converse is true for the First Officer's RMI.

CENTRAL INSTRUMENT WARNING SYSTEM

The Central Instrument Warning System (CIWS) monitors data inputs to the flight instruments and illuminates the master WARN light(s) and appropriate annunciator lights when the data input is invalid.

NOTE: Some CIWS self-cancel the master WARN light(s) when a warning lasts less than 30-seconds.

The system performs the following functions:

- Monitors warning flag signals from HSI, ADI, GPS, AHSU, VOR/ILS system, and low range radio altimeter.
- Compares roll and pitch attitude displays in the ADIs.
- Monitors its own operation.
- Provides a manual reset to the master WARN light signals.
- Provides a dimming feature for the attitude, heading and monitor lights.

STANDBY ATTITUDE INDICATOR

The standby attitude indicator is located on the pilots' center instrument panel. It provides a visual display of basic airplane pitch and roll attitude. Power to operate the indicator is from the STANDBY BUS. If DC power is lost, the indicator may be considered reliable for up to nine minutes, due to its long spin-down characteristics. This indicator receives no pitch erection cutoff (PECO), unlike the three primary vertical gyros; therefore, the standby attitude indicator is more vulnerable to precession due to aircraft acceleration forces.

AIRCRAFT CLOCKS

Electronic clocks are installed on the Captain's, First Officer's and Flight Engineer's instrument panels, and are intended as the flight crew's primary time reference.

A single-pulse generator for the Captain's and First Officer's clocks is installed in the Flight Engineer's console. It provides time signals to both clocks. A spare pulse generator is installed adjacent to the operating unit to provide a quick-change backup in the event of a failure.

A separate time-base generator, located in Lower 41, is provided for the Flight Engineer's clock. The generator also provides a GMT signal to the flight recorder.

All the system components receive operating power from the hot battery bus.
Flight Engineer's Clock
The Flight Engineers clock is identical to the Pilots clock on most airplanes, with the exception that it does not have a remote button to operate the sweep hand.
On some airplanes this has been replaced with a basic clock with no extra functions.

DIGITAL FLIGHT DATA RECORDER

The expanded flight recorder system records over 30 selected items of flight data obtained during the latest 25 hours of operation.
Some of the parameters stored on the continuous tape loop are obtained from:

- AHSU #1 (attitude)
- CADC #1 (speed and altitude)
- Flight control position
- All hydraulic systems (low pressure)

The flight recorder system consists of a flight data acquisition unit (FDAU) in Lower 41, a digital flight data recorder (DFDR) on the aft equipment rack (aft of L-5 door), a flight data entry panel and interconnecting wiring.
The FDAU processes all input signals to prepare them for recording. The DFDR records the output data stream from the FDAU on six-track mylar magnetic tape. Primary operating power is obtained from the essential flight instrument bus. The system is turned on automatically when engine generator power is applied and the parking brake is released.

ALTITUDE ALERT SYSTEM

The Altitude Alert System (AAS) provides aural and visual signals to the pilots when the airplane approaches or deviates from a selected altitude. Selected altitude is set by the ALT SEL control on the autopilot-flight director mode select panel.
The altitude alert system includes an altitude alert computer located behind the flight engineer's panel, an altitude alert test switch located on the center instrument panel and an altitude alert annunciator located on the captain's and first officer's instrument panels.
The altitude alert computer receives 28-volt dc power from flight instrument bus No. 2 and 26-volt ac power from flight instrument bus No. 2 autotransformer No. 6.
The altitude alert computer receives altitude data from CADC No. 2, altitude select data from the autopilot-flight director mode select panel, barometric pressure correction from the First Officer's altimeter barometric setting, and an inhibit deviation alerting signal from the main landing gear DOWN sensor.
The Captain's and First Officer's altitude alert annunciators illuminate when approaching the selected altitude in either climb or descent. The altitude alert computer automatically resets to the approach mode whenever the difference between the selected altitude and the actual altitude exceeds 900 feet because of airplane altitude change or new altitude selection on ALT SEL control.
Altitude Alert Computer
The altitude alert computer evaluates data from the airplane central air data computer; autopilot mode select panel (altitude select function) and First Officer's barometric altimeter, and computes the deviation from a preselected altitude.

The altitude alert computer is located behind the Flight Engineer's instrument panel.

Altitude Select Panel

The altitude select panel enables the pilots' to preselect an altitude in 100-foot increments from 0 to 50,000 feet. The selector panel is located on the autopilot-flight director mode select panel on the pilots' lightshield.

Annunciators

The altitude alert annunciators are located on the Captain's and First Officer's instrument panels. Steady alert annunciators indicate that the airplane is approaching the selected altitude and flashing alert lights indicate that the airplane is deviating from the selected altitude.

Altitude Alert Light (Amber)

(Above each pilots ADI)

ILLUMINATES STEADY (and 2 second aural sounds) when approaching (900 feet above or below) selected altitude; remains illuminated until 300 feet above or below altitude.

EXTINGUISHES when 300 feet above or below selected altitude and remains extinguished while within that 300 feet above or below range.

ILLUMINATES FLASHING (and 2 second aural tone sounds) when deviating 300 feet above or below selected altitude; light continues to flash until 900 feet above or below, at which time the light extinguishes and the system is automatically reset for subsequent altitude alerting.

"Deviation from altitude" alerting is inoperative when landing gear is extended.

Altitude alert is referenced to CADC #2.

Altitude Alert Test Switch (Under glareshield)

The ALTITUDE ALERT TEST switch on the center instrument panel allows a confidence test of the altitude alert system. Pressing the TEST switch disables the inhibit circuit so that the system will respond to the simulated altitude variation. A system self-test may be performed by pressing the TEST switch and slowly adjusting the selected altitude on the ALT SEL indicator from 1000 feet above the airplane altitude to the airplane altitude and back to 1000 feet above the airplane altitude. The corresponding aural and visual indications will occur when each level is crossed as described previously for normal system operation.

Landing Gear

The B-747 landing gear consist of four main gear struts, each carrying a 4 wheel truck, and a nose gear strut carrying 2 wheels. Two of the main gear are wing mounted and two are body mounted. All 5 gear retract forward and up.

In normal operation, landing gear extension and retraction is accomplished by using 2 separate hydraulic sources: system 1 for the nose and body gear and system 4 for the wing gear.

An alternate gear extension system is provided, which utilizes electric motors to unlock all 5 gear and door uplocks. Landing gear weight and airloads will extend and lock the gear after the doors fall open.

Hydraulic actuators tilt all the main gear as soon as the airplane is airborne, so that they can fit into the wheel wells. The wing gear incorporate a mechanical lock to keep the gear fully tilted while retracted, thus preventing them from scrubbing the wheel wells and possibly jamming on extension. The body gear do not require this protection due to design differences.

Automatic wheel braking is provided during gear retraction.

MAIN GEAR DOORS

The body gear doors consist of inboard and outboard mechanically actuated strut doors plus a hydraulically actuated wheel well door. The strut doors are mechanically actuated by the body gear.

The wing gear doors consist of a mechanically actuated strut door and a hydraulically actuated wheel well door.

The nose doors are clam-shell type that fair with the fuselage when closed. There are 4 doors. The 2 largest are the forward main doors, while the remaining two smaller doors are classified as "Strut" doors, which move only when the strut moves up or down.

The ground door release handle is located on the keel beam in each body wheel well. Once open, the doors can only be closed by hydraulic pressure from Systems 1 and 4.

WARNING

BEFORE OPENING THE DOORS, MAKE SURE NO ONE IS WORKING IN THE GEAR WELLS OR ADJACENT TO THE WING AREA.

LANDING GEAR POSITION SENSORS

Two separate systems consisting of Primary and Alternate sensors are used for Landing Gear position indications and warnings.

The primary (PRIM) system has sensors to determine the following:

a. Gear down and locked
b. Gear up and locked
c. Gear tilted
d. Gear doors closed

The alternate (ALT) system has sensors to determine the following:

a. Gear down and locked
b. Nose near up and locked
c. Gear tilted

d. Gear doors closed

NOTE: There are no alternate sensors provided for main gear up and locked.

PRIMARY AND ALTERNATE SYSTEM ANNUNCIATOR LIGHTS

The Flight Engineer's Landing Gear Annunciator module contains the annunciator lights used to display the indications from the primary or the alternate system. There are no annunciator lights provided for the Gear Up and Locked sensors. However, if the sensors fail to agree with the gear handle position, the red gear light on the Landing Gear Handle module will remain illuminated. In this event, follow instructions in the appropriate Alternate Procedure. The Landing Gear and doors can be considered safe if either the Primary or the Alternate sensors confirm its position.

LANDING GEAR DOOR GROUND RELEASE HANDLES

All Main and Nose Gear doors can be opened by Ground Release Handles located in the Body Gear Wells and the Nose Wheel Well. The handles in the body gear wells will open both doors (Wing Body Gear) on the respective side. If the handles are left in the DOWN position (doors open), gear retraction will re-position the handle to NORMAL (door closed) position.

GROUND SAFETY RELAY

Ground or inflight functions or various airplane equipment is controlled by the ground safety relay. Failure of the ground safety relay to operate in flight is indicated by illumination of the Ground Safety Relay light on the Flight Engineer's annunciator panel. Normal inflight operation of the affected equipment can be achieved by pulling the Ground Safety Relay circuit breaker.

BRAKES

The main gear landing wheels have hydraulic braking. The normal brake system is powered by Hydraulic System #4, with Hydraulic System #1 as an alternate source. A reserve brake system powered by Hydraulic system #2 is also provided. Braking with either the normal or reserve system is modulated by the anti-skid system.

NOTE: When using the reserve brake system, parking brakes and touchdown locked
wheel protection are not available.

ANTI-SKID

The anti-skid system modulates the brake pressure to prevent wheel skidding and damage to the tires. The system also provides protection against inadvertent brake application on touchdown.

With the anti-skid switch ON, the system is activated after all of the following conditions are met:

1. One main landing gear down and locked.
2. Aircraft is in the ground mode. (One gear on each side of the airplane is not tilted either PRIM or ALT.)
3. Wheel rotation is sensed.

Touchdown protection signals a full release of each anti-skid valve until all the above

conditions are met.
An anti-skid Tilt Input Test Switch provides a means of verifying (on the ground only) that the required tilt input signals are operative and available to the anti-skid control unit.
The system has annunciator lights on both the Pilot's and Flight Engineer's annunciator panels. The Pilot's annunciator is a "Summary" panel and contains two lights relating to the anti-skid system. One is the "Anti-Skid" light, which indicates that an electrical problem exists in the circuitry to one or more wheels. The other is the Anti-Skid Hydraulic light, which indicates that the Anti-Skid return line valve is not fully open.
The Flight Engineer's annunciator has a light for each of the 16 anti-skid valves in the normal brake system. When an electrical problem exists, the anti-skid valves are designed to put that wheel in manual braking and its respective light will be illuminated.
The reserve brake system is also protected by the anti-skid system. The difference is that the reserve brake system utilizes only eight anti-skid valves (one for each lateral pair of wheel(s) and illustrates only four reserve brake valves (one per truck). If circumstances require that the reserve brake switch on the Captain's lower right instrument panel be opened and Hydraulic System No. 2 be used to brake to a stop, simply check that all four reserve anti-skid lights are extinguished. This means that all 16 wheels will have anti-skid protection.

NOSE GEAR STEERING

Nose wheel steering is powered by the No. 1 Hydraulic System. Steering (up to 70° left and right of center) is accomplished using either pilots' steering tiller.

BODY GEAR STEERING

Normal nose wheel steering is supplemented by body gear steering. The use of body gear steering will reduce excessive engine power needed for turns at low speeds and prevent tire scuffing. When armed, body gear steering is actuated automatically when the nose wheel is turned in excess of 20°. Maximum body gear steering angle is 13°.
NOTE: In the event of the failure of Hydraulic System No. 1, which powers both the nose and body gear steering, directional control can be maintained at slow speeds (under 40 KTS) by careful use of differential braking and at higher speeds with the rudder.

RUDDER PEDAL STEERING

DIFFERENCES- N470EV, N471 EV, N477EV
Rudder pedal steering (7° left and right of center) is available from either set of rudder pedals and can be overridden by tillers. The rudder pedal steering is primarily intended for use during takeoff and landing.

NOSE LANDING GEAR MANUAL EXTENSION

The alternate nose gear extension system is driven manually by a manual crank. The electric actuator is removed and the crank is inserted into the gearbox. The crank (stowed on an adjacent bulkhead) and gearbox is located in the pressurized area on the left-hand side of the nose gear wheel well forward of the main electronic bay. The crank releases the nose doors through the cable system in the same manner as the doors are released by the electrical alternate extension system. The doors and gear are designed to free fall fully open after unlocking without any additional force.

Pneumatics

The pneumatic system provides compressed air for:

- engine starting,
- airconditioning pack operation,
- pressurizing fuselage compartments, hydraulic reservoirs, and potable water tanks,
- leading edge flap positioning,
- air-driven hydraulic pump opertion,
- wing anti-icing, and
- aft cargo compartment heat.

The pneumatic manifold can be pressurized with engine bleed air, by the APU, or from ground carts.

GENERAL

The pneumatic system supplies compressed air from the 8th and 15th-stage engine compressor bleeds, the APU, or a ground pneumatic source, to the pneumatic manifold. The pneumatic manifold distributes the air for use in the air conditioning and pressurization systems, engine starting, thermal anti-icing systems, leading edge flaps, air-driven hydraulic pumps, aft cargo compartment heating, potable water system pressurization, and thrust reverser operation.

ENGINE BLEED AIR DISTRIBUTION SYSTEM

The engine bleed air distribution system consists of a manifold which carries hot compressed air from the engine compressor to the air conditioning packs and to the air-driven units in other systems. Ducts from the engines connect to respective left and right wing manifolds. A crossover duct, forward and below the center wing section rear spar, joins the left and right wing manifold sections. The air control components connecting the pneumatic manifold to related systems are covered as part of the systems they serve.

The primary supply of pneumatic air is from the 8th-stage compressor of each engine, though a check valve. When 8th-stage bleed air pressure is not high enough to supply system demands, 15th-stage bleed air is used. Switching from 8th to 15th-stage bleed on each engine is controlled automatically by the high stage bleed air valve, which also regulates the pressure of the air it supplies to the pneumatic manifold. Air from either stage is routed through a precooler and a pylon valve on each engine before entering the pneumatic manifold. The pylon valve limits manifold pressure to 45(±1) psig and controls manifold air temperature by controlling the amount of airflow through the precooler. A pressure relief valve upstream of the precooler on each engine protects the ducts from overpressure.

The pneumatic manifold is separated into left, right and center sections by two wing isolation valves. The center section contains the APU duct connection and the ground service pneumatic connectors. Flow of APU air into the center section is controlled by the APU bleed air valve. Ground pneumatic sources bypass the pylon valves and must have pressure regulation equipment.

Switches on the Flight Engineer's and Pilots' overhead panels control solenoids in the pneumatically operated pylon valves and high stage bleed air valves. The engine fire switches, when actuated, cause the pylon valves to close. Each engine ignition switch, when paced to GRID START, causes the respective high stage bleed air valve to be held closed and the respective pylon valve to be held open for reverse flow through the pylon valve during engine start.

PNEUMATIC MANIFOLD

On each engine, pneumatic air is extracted primarily from two 8th-stage ports, ducted together before it passes through the precooler heat exchanger and the pylon valve, and discharges into the cross-over manifold in the wing. Four 15th-stage bleed ports are used to augment the 8th-stage pressure whenever 8th-stage pressure is not high enough. Air-driven hydraulic pump supply is taken from the manifold before it enters the wing. The nose cowl anti-ice air is tapped off upstream of precooler temperature sensor and control. Duct pressure and temperature sensing elements, and hydraulic reservoir pressurization lines attach to the pylon duct.

The APU bleed air duct runs under the main deck floor and attaches to the crossover duct. Two check valves, the APU bleed air valve, and two negative pressure relief valves are in the APU duct for control, to prevent reverse flow and to protect the duct from damaging negative pressure.

8TH-STAGE CHECK VALVE AND PRESSURE RELIEF VALVE

Two intermediate (8th-stage) check valves are upstream of the precooler. A spring loaded relief valve installed in the duct upstream of the precooler, relieves excessive pressure into the engine nacelle. The valve opens at approximately 95 psi and reseats at 85 psi.

PYLON SHUTOFF AND PRESSURE REGULATING VALVE

The pylon valve consists of a 6-inch diameter butterfly valve, a pneumatic actuator, a temperature sensor, and pneumatic control. The assembly also includes two 28-volt DC solenoids that allow manual closing of the valve to isolate the engine from the pneumatic system, and to open the valve during engine starting.

The valve has a manual override lever which allows it to be opened for engine start, or closed to isolate the bleed air from the pneumatic system, if the valve should fail.

The pneumatic pre-cooler controls discharge temperature to approximately 350°F. If this temperature control malfunctions the pylon valve will start restricting air flow when temperature exceeds approximately 365°F. The pylon valve will be fully closed by 450°F for overheat protection.

HIGH STAGE BLEED AIR VALVE AND VALVE CONTROL

The high stage bleed air valve is located on the left side of the engine at the junction of the four 15th-stage bleed air ducts. The valve controller is located on the upper right portion of the fan exit diaphragm.

The high stage bleed air valve is a 6-inch diameter, high-temperature butterfly valve. A 28-volt DC solenoid mounted on the control allows the valve to be closed by the engine bleed valve switch. The valve has provisions for manually overriding the valve control to lock the valve in the closed position.

The pressure regulator consists of a poppet attached to a hinged arm which is controlled by two diaphragms subjected to downstream duct pressure. The regulator is designed to prevent sudden increases of pressure in the ducts.
The valve closed solenoid, when energized, moves its poppet to release servo pressure and close the valve.
The overpressure shutoff is a backup device for the control regulator. It senses high stage bleed pressure, and should the pressure become high enough to endanger the integrity of the ducts downstream of the valve, the diaphragm will move a poppet to release servo pressure and close the valve.
One of four lights on the Flight Engineer's panel illuminates whenever the respective high stage bleed air valve is not fully closed. The lights are labeled HIGH STAGE.

HIGH STAGE BLEED AIR CHECK VALVE

A check valve in the duct downstream of the high stage bleed valve prevents reverse air flow when the bleed manifold is pressurized for engine start.

WING ISOLATION VALVES

Isolation of the cross-ship manifold into left, right, and center sections is accomplished by two 115-volt, singe-phase, motor-operated butterfly valves. The left wing isolation valve is in the air conditioning bay. The right-wing isolation valve is on the right end of the crossover duct.
Control switches for both valves are on the Flight Engineer's Instrument panel.

GROUND AIR CONNECTORS

Two connectors are provided to allow pressurization of the pneumatic manifold by ground service carts. The connectors are accessible through a hinged panel on the underside of the fuselage. Air enters the crossover duct between the wing isolation valves.

APU BLEED AIR (SHUTOFF) VALVE

Flow of air from the APU to the pneumatic manifold is controlled by a motor-operated butterfly valve in the APU manifold between the two check valves. The 115-volt AC motor-operated valve is controlled by a switch on the P4 Flight Engineer's APU module.

BLEED AIR PRECOOLER

To prevent bleed air temperature from becoming excessive at the higher engine power settings, an air-to-air heat exchanger is installed on each engine. Air from the fan section is used as the cooling agent to precool the bleed air extracted from the engine to a nominal temperature of approximately 350° (±30°)F before it enters the pneumatic duct. The flow of cooling air across the heat exchanger is controlled by a temperature sensor and pressure anticipator, which automatically positions the cooling air valves on the fan sections, according to the temperature and pressure of the bleed air.

BLEED AIR PRESSURE INDICATING

Pneumatic manifold pressure is displayed on the Flight Engineer's instrument panel as a means of monitoring engine bleed air systems performance. Pressure information

also provides guidance in management of bleed air source and the systems using air. Two pressure transmitters send 28 volt AC signals to the dual pointers of the duct pressure indicator on the Flight Engineer's instrument panel.
Left wing manifold pressure is sensed by a transmitter located immediately forward of the left wing isolation valve.
The right wing manifold pressure transmitter is on the opposite side of the airplane with its tap in the portion of manifold running aft along the air conditioning bay. The transmitter is attached to structure near the right wing manifold supply to pack No. 3. Both transmitters are upstream of the wing isolation valve on their side of the airplane. An indicator graduated from 0 to 100 psi is on the Flight Engineer's instrument panel below the isolation valve switches. Separated pointers respond to electrical signals from the left and right pressure transmitters.

BLEED AIR TEMPERATURE INDICATING

Temperature of air in the engine pneumatic ducts is sensed by overtemperature switches. Separate lights for each engine will indicate OVERHEAT if bleed air temperature reaches 490°F. The overtemperature switches are in the pylon ducts downstream from the pylon shutoff valves. Indicator lights are on the P4 Flight Engineer's instrument panel.
Power for the lights is through the master dimming circuit for control panel lights. The master test circuit includes these lights. Since there are controls in the bleed air system designed to keep pneumatic air temperature below the actuation point of the overtemperature switches, operation of the OVERHEAT light is a signal of pneumatic component malfunction.

WING LEADING EDGE OVERHEAT DETECTION

The wing leading edge overheat detection system uses 10 thermal switches in the leading edge of each wing and two thermal switches in each nacelle strut to detect an overheat condition. The thermal switches are wired in parallel and closure of any one will provide overheat warning on the Flight Engineer's instrument panel. The two thermal switches in each nacelle strut are set to close at 300°F. The four thermal switches located in the nacelle strut areas of the wing are set to close at 250°F and the 15 thermal switches located away from the nacelle struts are set to close at 200°F.

OPERATION USING THE APU

The pneumatic manifold can be pressurized when the APU is operating and the APU bleed air valve is open. Pressures of 30 to 45 psig can be obtained in the manifold without systems operating. There are no means of regulating the pressure.
OPERATION USING ONE OR MORE PNEUMATIC GROUND SOURCES
One or more pneumatic ground sources may be connected to the pneumatic ground service connectors. Number of sources connected would depend on system demands. Ground sources should have pressure regulating equipment.
OPERATION DURING ENGINE STARTING
When the manifold is pressurized in order to start an engine, flow through the pylon valve is reversed. Circuits controlled by the engine start switches cause the pylon to be locked open and the high stage bleed air valve to be held close. Air passes in

reverse direction through the precooler to the engine start valve and to the starter. The 8th-stage check valve and the high stage bleed air valve prevent air from entering the engine.

OPERATION WITH ENGINES RUNNING

The pneumatic manifold is pressurized to 45 (±1) psig by 8th-stage or high-stage bleed air from whichever engine pylon valves are open. When high stage pressure drops to about 90 psi at sea level, sensing lines will signal the high stage bleed air valves to open.

From the precooler bleed air enters the pylons, where ducts branch off to serve nose cowl thermal anti-icing, thrust reverser 3-way valve, hydraulic reservoir, and the air-driven hydraulic pumps. The main pneumatic duct then enters the wing where connections are made to wing thermal anti-icing and leading edge flaps. The water tank pressurization line, aft cargo heating duct, and air conditioning packs are connected to the duct center section.

If manifold pressure should drop below about 12 psig, the air conditioning pack flow control and shutoff valves will close to give priority to wing leading edge flaps for remaining air supply.

Powerplant CF6-50

The General Electric CF6-50E2 engine is a dual-rotor, axial flow, high bypass ratio turbofan. The takeoff thrust rating at sea level in temperatures up to 30°C is 51,800 pounds. The fan delivers approximately 75% of the thrust.

The low-pressure compressor unit (Ni) consists of a single stage fan and a three-stage compressor connected to a four-stage turbine. The high-pressure compressor unit (N2) consists of a fourteen-stage compressor unit connected to a two-stage turbine through concentric shafting. The first six stages of the high stage compressor consist of variable stators, which are positioned by the main engine control in response to thrust lever movement.

FAN SECTION

The fan section consists of the rotor, a stator and a fan frame. The fan frame is the major structural component of the fan section. In addition to housing the inlet gearbox and variable bypass valves, it supports the front of the high-pressure compressor, the fan rotor, the fan stator, the transfer gearbox and the accessory gearbox.

COMPRESSOR SECTION

The inlet guide vanes and first six stator stages of the 14 stage compressor have variable angle vanes. Major components of the compressor are the compressor rotor, compressor front stator, compressor rear stator, and the compressor rear frame.

TURBINE SECTION

The turbine section consists of a high-pressure, low-pressure, and a rear frame section. The high-pressure turbine elements are cooled by a continuous flow of compressor discharge air. The major components of the low-pressure turbine section are the turbine mid frame, a four-stage turbine rotor, first stage low-pressure turbine nozzle and a !ow-pressure turbine stator assembly. The turbine mid frame houses the No. 5 and No. 6 main bearings which provide support for the aft end of the high pressure turbine rotor and the forward end of the low pressure turbine rotor. The rear frame houses the No. 7 main bearing which supports the rear of the low pressure turbine rotor.

ACCESSORY GEARBOXES

The accessory drive section consists of the inlet gearbox, radial drive shaft, transfer gearbox, horizontal drive shaft, and accessory gearbox.

The inlet gearbox is located in the forward sump of the engine. The gearbox extracts energy from the core engine rotor and transmits the energy to the radial drive shaft. The radial drive shaft is located in a housing aft of the bottom vertical strut of the fan frame. The shaft transmits power from the inlet gearbox to the transfer gearbox. The transfer gearbox, mounted on the bottom of the fan frame, consists of an enclosed 90-degree bevel gear train. The transfer gearbox functions to change the direction of the energy from vertical to horizontal, or from radial drive shaft to the horizontal drive shaft which powers the accessory gearbox.

The accessory gearbox is mounted under the fan frame. Engine and aircraft accessory mounting and drive pads are provided on both the forward and the rear faces to the gearbox. The engine accessories mounted on the gearbox are the starter, fuel pump, main engine control, lube and scavenge pumps, and Nz tachometer generator. Pads are also provided for mounting the aircraft hydraulic pumps, constant-speed drive, and alternator.

ENGINE CONTROL SYSTEM

The engine control system consists of the start control system and the thrust control system. The start control system consists of a start lever assembly and a fuel shutoff valve. The thrust control system consists of a thrust lever and clutch pack assembly, control cables, and a quadrant and thrust reverser interlock mechanism.

The control system, in conjunction with components of other systems, also provides appropriate electrical signals to the flight/ground idle solenoid on the main engine controls (MEC).

START LEVER

Both positions of the start lever (CUTOFF and IDLE) are electrically connected to the fuel condition actuator and pylon fuel shutoff valve. The fuel condition actuator is located on the lower right side of the engine. When the start lever is in IDLE, it positions the shutoff lever on the MEC to a corresponding position for fuel delivery for engine start. With the start lever in CUTOFF, the motor drives the actuator and MEC to the CUTOFF position.

GROUND IDLE LIGHT (Amber)

The flight/ground idle solenoid installed on the MEC provides two discrete engine idle speeds: ground idle (solenoid energized) and flight idle (solenoid de-energized, which is approximately 14% Nz higher than ground idle). Ground idle is used during ground operation and all flight operations except approach and landing. The engines shift to flight idle during approach and landing to facilitate quicker response and acceleration for a Go-Around. The shift to flight idle occurs when the aircraft is in the air and the flaps are positioned to 25 or 30. Flight idle is maintained until 5 seconds after the landing gear tilt sensors sense the aircraft has landed. The engines then shift to ground idle.

The GRID IDLE light on the control stand illuminates in-flight whenever the flaps are extended to 25 or more and the flight/ground idle solenoid on any engine is not in the flight mode. On the ground, the light will be illuminated as long as the flaps are extended greater than 25.

TACHOMETER

The engine tachometer system measures the speed of the low speed Ni rotor and high speed Nz rotor and provides the speed to indicators in the cockpit. Ni speed is a means of monitoring engine power output, engine condition, Ni rotor integrity, and Ni rotor overspeed. Nz rotor speed is a means of monitoring engine starting, engine condition, Nz rotor integrity, and Nz rotor overspeed.
The system consists of an Ni tachometer transmitter (fan speed sensor), an Ni indicator, an Nz transmitter (core speed sensor) and an Nz indicator. The two transmitters are AC generators whose frequency is directly proportional to rotor speed. The Ni transmitter is mounted on the fan case at the two o'clock position. The Nz transmitter is located on, and driven by, the oil system lube and scavenge pump.

STARTING SYSTEM

The engine starting system provides the means of rotating the engine Nz compressor on the ground or in flight to a RPM range at which engine start can be attained. For in-flight start, the system can be energized to supplement a windmilling engine if required.
CONTROL
With 28V DC power supplied to the engine ignition switch, the start system is armed. When the pneumatic duct is pressurized, and the engine ignition switch is moved and held to GRD START, electrical power is supplied to the corresponding engine bleed valve, high stage valve and start valve. The bleed valve and the start valve open and the high stage valve is driven closed. The start valve position indicator switch actuates and illuminates the start VALVE OPEN light when the start valve opens.
When the start valve opens, compressed air is admitted to the starter air inlet. The air passes through the starter inlet nozzle and is directed axially through a turbine rotor, imparting high-speed rotation to the rotor. The starter rotation is transmitted through the accessory gearbox, transfer gearbox and inlet gearbox to the Nz compressor. The Nz compressor begins to rotate and establishes airflow through the engine. When the Nz compressor speed reaches 15%, fuel and ignition are applied by advancing the start lever to IDLE.
The starter continues to assist the engine until the ignition switch is released to the OFF position closing the start valve. Closing of the start valve is indicated by the start VALVE OPEN light extinguishing. If the Nz compressor speed exceeds the starter speed, a clutch disengages the starter.

ENGINE STARTER

The engine starter converts high-energy compressed air into starter shaft torque sufficient to accelerate the engine to starting speed. The start valve opens to permit the flow of air to the starter inlet.

START VALVE

The start valve is a spring and pressure loaded closed, pneumatically operated, electrically controlled shutoff valve. A regulator bleed orifice is incorporated to provide a controlled opening time. The lower end of the valve shaft is equipped with a handle to allow manual opening of the valve in the event the solenoid fails to operate. Markings are provided on the valve body, which align with the manual override handle, to give an external indication of the valve position.

IGNITION SYSTEM

The ignition system for each engine incorporates two 115V, 400Hz powered, capacitor discharge type circuits, which deliver a minimum of 2.0 joules per spark. The circuits are electrically and physically independent.

CONTROL

Two toggle type ignition switches and a start lever ignition switch for each engine provide control of ignition. When the start switch is in the Flight Start or Ground Start position and the start lever is in the Idle position, electrical power is supplied to the ignition exciters.

IGNITER PLUGS

Two igniter plugs are provided for each engine. The plugs are mounted in the compressor rear frame at approximately the four o'clock and five o'clock positions.

STANDBY IGNITION

In the event of a loss of all generators, ignition can be supplied to all engines simultaneously by utilizing the standby ignition switch. When the standby ignition switch is positioned to IGN 1 or IGN 2, the ignition switches on the ignition control modules are bypassed. A battery powered inverter then supplies power to the igniters.

ENGINE FUEL SYSTEM

The fuel control system in conjunction with the variable stator vane and variable bypass valve systems regulates the thrust output of the engine under all conditions. The main engine control takes inputs from the thrust lever, compressor inlet temperature (CIT), compressor discharge pressure (CDP), variable stator vane feedback position, high-pressure compressor speed (N2), and variable bypass valve feedback position. A flight/ground idle electrical signal is used to establish the desired idle level. The system controls transient and steady state fuel flows, and sets variable stator vane and variable bypass valve positions, to maintain steady state speeds and acceleration/deceleration transients within stall and temperature limits.

PRESSURIZATION AND DRAIN VALVE (P&D)

The P&D valve maintains system pressure to ensure adequate operating pressure for servo operation and engine motoring. When the fuel shutoff valve on the main engine control is mechanically closed, the P&D valve opens and allows fuel remaining in the fuel manifold and fuel nozzles to drain into the drain can. An ecological drain system removes drainage fuel from the drain can and returns it to the fuel system during subsequent engine operation.

FUEL PUMP

The fuel pump consists of a centrifugal boost element and a high-pressure gear element. Excess fuel, supplied by the fuel pump to the main engine control, is returned to the pump downstream of its low-pressure element. Downstream the fuel is directed through the pump mounted fuel/oil heat exchanger. The heat exchanger maintains the fuel temperature above 20°C. Fuel is then passed through the fuel filter before entering the main engine control.

MAIN ENGINE CONTROL (MEC)

The MEC is basically a speed governor which senses engine speed, compressor discharge pressure (CDP), compressor inlet temperature (CIT), and thrust lever position and adjusts the fuel flow as necessary to maintain the desired RPM. The control is a hydromechanical device that operates by use of fuel operated servo valves. The main engine control also schedules variable stator vane (VSV) and variable bypass valve (VBV) positions and directs high-pressure fuel to the VSV and VBV actuators to position the vanes as required.

ENGINE AIRFLOW CONTROL COOLING

Fan discharge air, admitted through the forward and aft compartment cooling tubes, pressurizes and cools the nacelle compartments. Fan discharge air is admitted through a port in the left side cowl and ducted through a manifold to cool the low-pressure turbine case.

The generators are cooled by low-pressure fan air forced through the cooling air inlet duct. Cooling air leaves through the openings around the generator housing, passes through the exhaust duct, and is exhausted overboard through the exhaust port in the right fan cowl panel.

Ignition leads and igniters are cooled by fan discharge air collected by a scoop on the fan frame. The air is ducted through the lower pylon fire seal and to a conduit around each igniter lead. The air discharges from the opposite end of the conduit onto the igniters.

Low-pressure compressor discharge air pressurizes the bearing sump seals. The air that flows through the seals is vented into the exhaust stream through a tube located in the center shaft of the engine.

COMPRESSOR CONTROL VARIABLE STATOR VANE SYSTEM (VSV)

The VSV system maintains satisfactory compressor performance over a wide range of operating conditions. The system varies the angle of the inlet guide vanes and the six stages of variable vanes to aerodynamically match the low-pressure stages of compression with the high-pressure stages. By varying the variable vane position in accordance with a predetermined schedule the critical low-pressure stages are automatically realigned to maintain compressor performance during all engine-operating conditions.

VARIABLE STATOR VANE FEEDBACK SYSTEM

The Variable Stator Vane feedback system is a mechanical system, which connects the variable stator system to the MEC. The system transmits the position of the variable vanes to the main engine control reset actuator for the purpose of reducing overshoot during throttle transients to takeoff power.

VARIABLE BYPASS VALVE SYSTEM (VBV)

The variable bypass valves open at scheduled conditions to permit some of the low-pressure compressor airflow to pass into the fan stream, thereby improving the matching of the high-pressure compressor with the low pressure compressor. The bypass valves are closed during takeoff and cruise operation and operate during other scheduled flight operations.

ENGINE FUEL SYSTEM

The fuel control system in conjunction with the variable stator vane and variable bypass valve systems regulates the thrust output of the engine under all conditions. The main engine control takes inputs from the thrust lever, compressor inlet temperature (CIT), compressor discharge pressure (CDP), variable stator vane feedback position, high-pressure compressor speed (Nz), and variable bypass valve feedback position. A flight/ground idle electrical signal is used to establish the desired idle level. The system controls transient and steady state fuel flows, and sets variable stator vane and variable bypass valve positions, to maintain steady state speeds and acceleration/deceleration transients within stall and temperature limits.

PRESSURIZATION AND DRAIN VALVE (P8D)

The P&D valve maintains system pressure to ensure adequate operating pressure for servo operation and engine motoring. When the fuel shutoff valve on the main engine control is mechanically closed, the P&D valve opens and allows fuel remaining in the fuel manifold and fuel nozzles to drain into the drain can. An ecological drain system removes drainage fuel from the drain can and returns it to the fuel system during subsequent engine operation.
FUEL PUMP

The fuel pump consists of a centrifugal boost element and a high-pressure gear element. Excess fuel, supplied by the fuel pump to the main engine control, is returned to the pump downstream of its low-pressure element. Downstream the fuel is directed through the pump mounted fuel/oil heat exchanger. The heat exchanger maintains the fuel temperature above 20°C. Fuel is then passed through the fuel filter before entering the main engine control.

MAIN ENGINE CONTROL (MEC)

The MEC is basically a speed governor which senses engine speed, compressor discharge pressure (CDP), compressor inlet temperature (CIT), and thrust lever position and adjusts the fuel flow as necessary to maintain the desired RPM. The control is a hydromechanical device that operates by use of fuel operated servo valves. The main engine control also schedules variable stator vane (VSV) and variable bypass valve (VBV) positions and directs high-pressure fuel to the VSV and VBV actuators to position the vanes as required.

ENGINE OIL SYSTEM

Each engine contains an independent oil system to provide lubrication and cooling it)[the engine main bearings, radial drive shaft bearings, and gearbox.
Oil flows, at a positive pressure, from the oil tank for distribution and is returned to the tank by the recirculating oil system. The air is removed from the oil by the tank de-aerator and the air is vented out of the tank through the tank pressurization valve.
The oil level within the tank decreases from full level during engine operation At takeoff power, the level may be up to (9 quarts) lower than the full level. Whenever the engine is motored but not operating, oil will flow into the engine and approximately 1.9 liters ;2 quarts) will not return to the tank unlil the engine is started

TANK PRESSURIZING VALVE

The tank-pressurizing valve vents excess air in the engine oil tank to the transfer gearbox, which is vented overboard.

THRUST REVERSER SYSTEM

The fan reverser on each engine is stowed and deployed by a single pneumatic drive motor. A translating fan cowl is deployed aft and blocker doors are rotated to direct the normal exhaust flow forward through vaned deflectors. The maximum available reverse thrust limit is 95% Ni.

FAN REVERSER

When reverse thrust is selected, the pneumatic drive motor rotates flexible shafts, which are connected to actuators.
The actuators convert the rotational motion to linear motion and drive the translating

cowls aft. As the cowls deploy, the blocker doors are rotated by the door links into the fan exhaust flow path.
Movement of the translating cowls also actuate cams on the feedback brackets. The cams actuate switches and valves controlling the thrust reverser position indicating lights, the operating speed of the pneumatic drive motor, and operation of the variable bleed valves.

PNEUMATIC DRIVE MOTOR

The pneumatic drive motor is the power source for stowing and deploying of the fan reverser. Source air is from the respective engine pneumatic system.
The motor operates at two regulated speeds: high speed is used during the initial 90% of deploy and stow travel; low speed is used during the final 10% of deploy and stow travel.

PRESSURE REGULATING AND SHUTOFF VALVE

The pressure regulating and shutoff valve limits engine or airplane bleed system pneumatic pressure as required for operation of the pneumatic drive motor and other components of the reverser control system.

DIRECTION PILOT VALVE

The direction pilot valve provides stow or deploy signals to the pneumatic drive motor in response to reverse thrust lever commands.

PNEUMATIC LIMIT SWITCH

The pneumatic limit switch signals the pneumatic drive motor to operate at a high or low speed during the stow and deploy cycles.

OVERPRESSURE SHUTOFF VALVE

The overpressure valve is energized by stow and deploy signals from the reverser control switch in the control stand. When the solenoid valve is energized the valve opens. In the event of a failure of the pressure regulating function of the pressure regulating and shutoff valve, the overpressure shutoff valve will close and latch. The valva mr rst than hp manuallv reset to oermit further use of the svstem.

FOLLOW UP DRIVE MECHANISM AND PUSH PULL CABLES

The drive mechanisms and push pull cables, in conjunction with the interlock cams, prevent application of forward or reverse thrust beyond idle until the fan reverser has nearly reached the fully deployed or stowed position. In the event of an in-flight reversal of the fan reverser, the throttle for the affected engine will be forced to the idle position.

ENGINE VIBRATION INDICATIONS

The airborne vibration monitoring (AVM) system continuously indicates the vibration level of each engine. Indicated AVM values vary between engines. The installed vibration characteristics are unique to each engine, installation, and instrumentation. The system is used for trend monitoring purposes.

Crew procedures associated with engine operation are based primarily on maintaining engine indications within operating limits. There are no operating limitations for the AVM system; therefore there are no flight crew actions (or procedures) based solely on a response to engine vibration indications.

Certain engine malfunctions result in airframe vibrations from a windmilling engine. As the airplane transitions from cruise to landing, there can be multiple, narrow regions of altitude and airspeeds where the vibration level can become severe. In general, airframe vibrations can best be reduced by descending and reducing airspeed. However, if after descending and reducing airspeed the existing vibration level is unacceptable and if it is impractical to further reduce airspeed, the vibration level may be reduced to a previous, lower level by a slight increase in airspeed.

Powerplant JT9D

The Pratt and Whitney model JT9D-7 is a dual axial compressor turbofan engine. Each engine is equipped with a thrust brake system and the necessary controls and indicators for operations of the engine.

The engine has a high bypass ratio (5 to 1) which results in the fan delivering approximately 75% of the thrust. The low pressure compressor unit (N,) consists of a single-stage fan and three-stage compressor connected by a through shaft to a four-stage turbine. The high pressure compressor unit (N2) consists of an eleven-stage compressor unit connected to a two-stage turbine through concentric shafting. A compressor bleed control system is automatically positioned to provide an adequate stall margin for engine starting, acceleration, deceleration, reverse thrust, and lower power operation. The fuel control unit schedules fuel to provide the thrust called for by the throttle setting.

The JT9D is a somewhat surge-prone engine. If a surge occurs, it is important to immediately check the EGT. If the stall occurs under high power steady state conditions, a cruise altitude of FL 350 or lower may be necessary to prevent additional stalls.

There are two engine idle speeds; low (ground) idle is used during ground operation and during all flight operations except the approach and landing when the engines shift to high (flight) idle to facilitate engine acceleration for a go-around. In the air the idle speed is determined by the position of the trailing edge flaps. Until the flaps are positioned to 25 or 30 the engines remain in low idle. High idle is then programmed and maintained until 5 seconds after touchdown. Landing gear tilt sensors then dictate a shift to low idle. The engine inlet is designed to provide optimum cruise performance. The nacelle inlets have thermal anti-icing.

Accessories are driven by the N2 compressor through an angled gearbox. A complete self-contained oil system provides lubrication and cooling for internal parts. The thrust reverser system provides means of reversing fan exhaust air.

FAN STAGE

The fan case assembly consists of the fan case (forward portion of the assembly), fan exit case and vane assembly (intermediate portion), and fan exit rear case. The assembly incorporates sound-absorbing liners and anti-icing tubes. Stator vanes, located between the inner and outer casings of the fan exit case and vane assembly, straighten the secondary airflow.

The fan rotor blades are dovetailed into matching grooves in the hub and are retained by four compressor blade locks and four lock retainers. This blade lock arrangement will prevent forward and rearward movement of the blade. Stator vanes in the fan exit case straighten discharge airflow, before it enters the fan exit rear case. A mounting pad is provided on the fan exit case and vane assembly for an electric sensor to determine low pressure (NZ) rotor speed.

N, COMPRESSOR

The axial flow front (low pressure) compressor partially compresses the air that passes through the primary (inner) airstream of the engine, then delivers this air to the rear

(high pressure) compressor.
The front compressor (N,) section consists of a rotor and stator assembly made up of the 2nd through 4th-stage rotors and the 1 st through 3rd-stage stators. The front compressor is driven by the 3rd, 4th, 5th and 6th turbine stages.
Rotation of the compressor is indicated in percent RPM. N, is displayed on the pilots' center instrument panel (P-2).

N2 COMPRESSOR

The purpose of the rear (N2) compressor is to further compress the air delivered by the front (N,) compressor and to then feed this air into the diffuser case and combustion section.
The rear compressor is driven by the 1 st and 2nd turbine stages. The 2nd-stage turbine disk is splined onto the (short) shaft which is in turn bolted to the 15th stage rear compressor disk.
External parts attached on the rear compressor case include the engine vane control, rear compressor stator vane actuator and mechanical linkages, regulated fine-filtered servo supply (PR) filter, pressurizing and dump (P&D) valve, angle gearbox supporting brackets, compressor bleed control, compressor bleed valves, stator vane anti-icing system, ignition exciters, fuel/oil cooler, fuel flow transmitter, oil tubes, fuel tubes, and bearing cooling tubes.
Rotation of the compressor is indicated in percent RPM. NZ is displayed on the pilots' center instrument panel (P-2).

TURBINE SECTION

The turbine section consists of the rear compressor drive turbine, front compressor drive turbine, and turbine exhaust section. Reaction to the combustion gases passing through the turbines causes the turbines to rotate and drive their respective compressors.
The two-stage rear compressor drive turbine utilizes a simple air cooling system for the 1 st-stage rotor blades in both inlet guide vane stages to permit and maintain optimum turbine inlet temperatures.
The rear compressor drive turbine stators consist of the 1st-stage nozzle guide vanes and the 2nd-stage inlet stator.
The 4-stage front compressor drive turbine is designed to maintain high efficiency and provide increased turbine work required at the relatively low speed and high bypass ratio of the fan.
The front compressor drive turbine rotor consists of the 3rd, 4th, 5th and 6th-stage rotor, disks, blades, airseals, turbine shaft, cap, turbine shaft coupling, and spacers. The turbine assembly is supported by the No.1 and No. 4 bearings.
The exhaust gas temperature (Tt 6) thermocouple probes are mounted on the turbine rear case and extend into the turbine discharge passage.

ACCESSORY GEARBOXES

The main gearbox is connected to the NZ rotor through a power train which includes the angle gearbox. The main gearbox provides a drive pad for the fuel pump and the fuel control, which is driven by the fuel pump. The fuel/air heater and filter, also on the

pump, complete the fuel system module. Drive pads are furnished for the starter, the constant speed drive, the hydraulic pump. Provisions are made for mounting the CSD and generator on the forward side of the gearbox. A drive pad is also provided for an NZ tachometer generator. In addition, the oil pressure and scavenge pump, oil relief valve, and an oil strainer are incorporated in the gearbox.
The angle gearbox transmits power from the engine to the main gearbox and from the starter on the main gearbox to the engine. An oil scavenge pump scavenges the oil draining into the angle gearbox from the No.1 and No. 2 bearing compartments and returns it to the oil tank. Power is transmitted from the high pressure rotor (N2) shaft to the angle gearbox by means of an inner drive shaft.

ENGINE INDICATING DISPLAY SYSTEM

The engine instruments on the forward, center instrument panel consist of two LCD screens mounted in a "smart chassis" (Ref. Figure 1). The chassis has controls and switches mounted on the back side that are used to set the aircraft engine type, select any intermixed engines, and provide other maintenance functions. These are accessible only when the full engine instrument display unit is pulled forward and they are accessed only by maintenance personnel.
The unit is "smart" in that it has memory functions that record engine exceedances and it has the ability to record snapshots. Is has enough memory to record a total of 16 exceedance events and 24 snapshots. When either of these total number of exceedances or snapshots is exceeded, the earliest recorded one is erased and replaced by the latest event.

BUILT-IN-TEST (BIT)

The preflight BIT check is actuated automatically when power is applied. It may also be actuated on the ground manually (if the N2 % RPM is less than 5%) by pressing the RST button. That function is not available in flight or when the engines are running. A manual or power-on BIT test (without error messages) is shown for a maximum of eight (8) seconds (five seconds of B & D logo followed by three seconds of EIA logo), or until the RST button has been pushed, where as the display units shall immediately display current engine parameters.
In addition, there is a "continuously running" BIT check that checks internal components of the display. If BIT errors exist, they will be displayed in the upper left-hand corner of the MENU screen. Major errors will be displayed in a red box and minor errors will be displayed in a yellow box. When the BIT check is running on the ground and no engines are running (below 5% Nz) and a major error is detected, the display unit will "freeze" on BIT start-up and not function further until appropriate maintenance actions are completed. Error codes will be displayed.
There will be no MEL relief for major errors. The existing (old technology) MEL relief for individual instrument indicators (EPR, N1, Nz, EGT, and Fuel Flow) will remain in effect for the "flat panel LCD" screens on an individual indicator basis.
The EIDS has a Flight Time counter and a Total Time counter to "time stamp" a snapshot or exceedance. The System Time counter is shown on the start-up screen. The time counters are designed in increments of one second and are defined as follows:

System Time Time EIDS Display Unit has been active.
* Flight Time Time since takeoff. WOW active (Weight On Wheels indicated by the Ground Safety Relay).
Total Time Time from first engine start when Nz > 5% RPM to last engine stop when Nz < 5%.
* The Flight Time is reset when the WOW goes from Active to Inactive signifying a new flight leg. Once a new flight leg has started, all Snapshot Flight legs are incremented. If an erroneous WOW signal is received, a new flight leg is started. Normal operations will not be affected.

CONTROLS

The switches and controls on the bottom left face of the panel provide for setting control EPR indexes in the AUTO mode and Manual mode (Ref. Figure 1). If AUTO is selected (with the three position EPR command switch), a message pops up in the lower left corner of the EPR screen that states, "INVLD POS". Additionally, there is an IMIX (engine intermix) position that allows setting the EPR index for an engine intermix if an intermix is installed. If an intermix is not installed, an amber "INVLD POS" message appears if the [MIX position is selected. The EPR indexes are set with the rotational CMD knob. There are three "push type" buttons in the center of the panel that allow selection of different screen functions: Reset (RST), Recall (RCL), and Snapshot (S/S). There is an ambient light sensor that requires no crew input or adjustment. It senses cockpit lighting and adjusts panel light balance. Brightness is controlled by the CENTER FWD PANEL rotary light switch, located on the center control stand.
On the lower right side of the panel, there is a three position (LEFT, NORM, and RIGHT) rotary switch that selects the left, right or normal screen displays. In the NORM position, the engine instruments are displayed normally with the EPR and Ni instruments on the left LCD display and the Nz, EGT and Fuel Flow instruments on the right display. If the LEFT position is selected, all of the engine instruments (EPR, Ni, N2, EFT, and Fuel Flow) are compressed and displayed on the left LCD screen (ref. Figure 2). If the RIGHT position is selected, all of the engine instruments are compressed and displayed on the right screen.
NORMAL DISPLAY
The normal engine instrument display consists of EPR and Ni instruments displayed in the left screen display. The right screen will display the Nz, EGT, and Fuel Flow instruments. (Ref. Figure 1)

EPR

Engine Pressure Ratio (EPR) indicates the ratio of exhaust gas pressure (Pt7) to inlet air pressure (Pt2).
The EPR instruments have a white "A", "F", or "J" displayed above each engine display. These designate which type of engine is installed. An intermix is indicated with the engine letter designation in green instead of white.
Below each engine identifier letter is a digital readout of EPR. A vertical scale of indicated EPR is below the digital readout. The associated EPR bug is white. If an intermix is present, the EPR bug for the intermixed engine will be green. The command EPR window is located at the bottom center of the display. If an intermix is present,

(Ref. Figure 6) a smaller command EPR window and intermix engine EPR bug (green in color) can be set with the command set knob. If an intermix is not installed, "INVALID POS" will appear in yellow when IMIX is selected.
When MAN (manual) is selected, the command EPR window for non-intermix engines can be set along with associated EPR bugs.
"OFF" flags will appear if there is a loss of signal or if the system is powered by standby power. (Ref. Figure 6)

N1 INSTRUMENT DISPLAY

Ni display indicates the speed of the Ni compressor rotation (RPM) and displayed in percent (Ref. Figure 6). Fan blades passing a sensor located in the fan case provides the Ni indication signal.
The Ni display has tick marks at 70% and 50% RPM. The 70% mark is a target for reverse thrust and the 50% is a reference for engine anti-ice.
The red and yellow limitation ranges will remain aligned horizontally even if an engine intermix is installed. The 70% and 50% tick marks will also remain at the appropriate RPM. This intermix alignment is made possible by automatic re-scaling. (Ref. Figure 6)
The vertical tape indicator will remain in white until reaching the caution range and will then turn yellow. It will remain yellow until leaving the caution range. The "ball" light exceedance indicator above the instrument will illuminate yellow and remain yellow until the yellow tape turns white or red.
When the vertical tape reaches the red limitation range, it will turn red and remain red until leaving the red range. The "ball" light exceedance indicator will turn red and remain red until the MAX IND RESET button is pushed. The red "ball" light exceedance indicator will then turn blue and remain illuminated until reset by maintenance personnel.
There are no "OFF" flags for the Nt indicator.

N2 INSTRUMENT DISPLAY

N2 display indicates the speed of the Nz compressor rotation (RPM) and is displayed in percent (Ref. Figure 6). An Nz tachometer generator, driven by the accessory gearbox, provides the N2 indication signal.
The red and yellow limitation ranges will remain aligned horizontally even if an engine intermix is installed. This intermix alignment is made possible by automatic re-scaling. (Ref. Figure 6)
The vertical tape indicator will remain in white until reaching the caution range and will then turn yellow (Ref. Figure 6). It will remain yellow until leaving the caution range.
The "ball" light exceedance indicator above the instrument will illuminate yellow and remain yellow until the yellow tape turns white or red.
When the vertical tape reaches the red limitation range, it will turn red and remain red until leaving the red range (Ref. Figure 6). The "ball" light exceedance indicator will turn red and remain red until the MAX IND RESET button is pushed. The red "ball" light exceedance indicator will then turn blue and remain illuminated until reset by maintenance personnel.
At max motoring RPM, approximately 18 to 22%, the vertical tape will turn blue and will remain blue until reaching 50% RPM, which is starter disengagement speed (Ref.

Figure 6). This color change is provided as an aid during engine start.
There are no "OFF" flags for the Nz indicator.

EGT INSTRUMENT DISPLAY

Exhaust gas temperature (EGT) indicates turbine exhaust temperature in degrees centigrade. This is measured by probes located between the second and third stage turbines (Ref. Figure 6),
There is a white line provided at the engine start EGT limit appropriate for the type of engines installed.
The red and yellow limitation ranges will remain aligned horizontally even if an engine intermix is installed (Ref. Figure 6). The engine start EGT limitation tick mark will also remain at the appropriate temperature. This intermix alignment is made possible by automatic re-scaling.
The vertical tape indicator will remain white until reaching the caution range and will then turn yellow (Ref. Figure 6). It will remain yellow until leaving the caution range.
The "ball" light exceedance indicator above the instrument will illuminate yellow and remain yellow until the yellow tape turns white or red.
When the vertical tape reaches the red limitation range, it will turn and remain red until leaving the red range (Ref. Figure 6). The red "ball" light exceedance indicator will turn red and remain red until the MAX IND REST button is pushed. The red "ball" light exceedance indicator will then turn blue and remain illuminated until reset by maintenance personnel.
An "OFF" flag will appear on each tape to indicate instrument failure or loss of signal.

FUEL FLOW INSTRUMENT DISPLAY

Fuel Flow (FF) indicates fuel flow to each engine as measured by a fuel flow meter mounted on the Fuel / Oil coolers. The Fuel Flow meter also sends a signal to the FUEL USED gauge on the flight engineers panel.
Fuel flow is displayed vertically in thousands of pounds per hour for each engine. There is a digital readout at the top that displays the sum total fuel flow for all operating engines. The Fuel Flow indicators may give erroneous indications when electrical power is applied to the aircraft with the engines shutdown. This indicator does not display "OFF" flags. When "0" fuel flow is displayed, the engine is shutdown or there is a loss of signal.

MENU SCREEN

With either LEFT or RIGHT selected with the display knob, the engine instruments are displayed on the selected screen and the other screen will initially display a "MENU". The menu gives five choices (Ref. Figure 2):
1) "RCL" Exceedance Review
2) "RST" Eng Tune / Return
3) "S/S" Record Snapshot
4) "RCL+S/S" Snapshot Review
5) "RST+S/S" Erase Snapshots

Additionally, there is a message at the bottom of the screen stating whether weight is on the wheels or not. Finally, in the upper left-hand corner of the screen, the results of a continuously running BIT error test are displayed. A description of all minor BIT errors follows. The five choices on the MENU screen provide direction as to which of the buttons to press to select a specific function:

1) Pressing RCL provides a review of any recorded engine limitation exceedances.
2) Pressing RST provides the Engine Tune and Engine Digital screen (used by maintenance to trim the engines and by the flight crew to perform engine starts and recording trend information.
3) Pressing S/S records "snapshots" of engine performance.
4) Pressing RCL+S/S review any snapshots that have been recorded.
5) Pressing RST+S/S erases any unneeded snapshots.

COMPACT SCREEN

When the LEFT or RIGHT position is selected with the control knob on the right side of the display console, the left or right display screen will display all engine instruments in a compacted mode (Ref. Figure 2). This provides continuous monitoring of engine parameters when reviewing other screen display functions.

This compacted mode is also displayed when all power sources have been lost except STANDBY POWER. When on STANDBY POWER, the left display screen displays the compacted mode with Ni, Nz, and EGT engine parameters active. Also, when on STANDBY POWER, there will be flags visible in the EPR indicators and the fuel flow will indicate zero fuel flow.

SNAPSHOT SCREEN

This screen is used for reviewing any snapshots taken by the flight crew (Ref. Figure 3). It provides a "frozen-in-time" shot of all engine parameters. This is useful when performing trend reading or recording any other desired information regarding engine parameters.

The snapshot process is initiated automatically within 2.5 minutes of takeoff at the time of maximum EGT for each engine, or by the press of the "S/S" push button switch. The flight crew may view snapshots by the simultaneous press of the RCL and S/S push button. The most recent snapshot is numbered "#1" and the oldest is "#24". Multiple snapshots may be reviewed as directed by the menu line at the bottom of the screen.

EXCEEDANCE SCREEN

This screen, when selected, displays any exceedances and all other engine parameters in a digital readout (Ref. Figure 4). It also displays the START TIME of the exceedance, which individual parameter caused the exceedance to be recorded, PEAK TIME of the exceedance, DURATION of the exceedance, and CAT (category) of the exceedance. This category designation is important to maintenance personnel. It indicates what action the maintenance publications direct.

The engine parameter exceedance process is automatically initiated by the Ni, Nz, and EGT engine parameters exceeding the engine manufacturer's specifications for normal operation. No pilot input is required.

Review of the exceedances may be performed after the compacted display mode has been selected and the RCL push button for each exceedance to reviewed has been pushed. There is a menu choice at the bottom of the screen that allows reviewing previously recorded exceedances (if any exist) and returning to the current one. Erasure of the Exceedance Events is performed by the maintenance crew and is a restricted procedure.

ENGINE DIGITAL SCREEN

This screen displays all engine parameters in an active digital display rather than a tape display (Ref. Figure 5). This enables the flight crew and maintenance personnel to accurately ascertain the reading of any engine parameter. It is useful to maintenance
personnel for "trimming" the engines and to the flight crew for performing starts and trend recordings. If LEFT or RIGHT is selected, a press of the RST push button causes the maintenance mode to be displayed.

LIMITATIONS

1) The system requires two fully working display units, with no Major BIT errors occurring. If any Major BIT errors do occur prior to takeoff, takeoff will not be initiated and maintenance will be notified.
2) Takeoff will not be initiated if any engine exceedances are displayed by the engine instruments. If any "blue", "yellow", or red exceedances are displayed. maintenance will be notified.
3) The "compacted" display mode will not be used for takeoff.

PREFLIGHT PROCEDURES

1) Verify that no BIT errors are displayed. If any are displayed, notify maintenance
2) Verify that no exceedances are displayed by the engine instruments. If any recorded exceedances are evident by a "red', yellow". or blue exceedance display, notify maintenance.

POWER BUSSES

Each display screen is powered by a Main AC buss. If the Main AC buss fails power automatically transfers to Standby AC. If any two of three power sources (Main AC buss 2, Main AC buss 3, or Standby AC buss) are available, both screens will function. If only one of the sources available, only one screen (compacted mode) will function.

BIT ERROR MESSAGES (MINOR ERRORS)

Minor BIT errors will be displayed in a yellow box.

BIT ERRORS (MAJOR ERRORS)

Major BIT errors will not have any MEL relief and will be displayed in a red box. Additionally, if a major error occurs on the ground and the engines are not running (below 5% Nz), the affected display will cease to function and the error code will be displayed. If the error occurs in flight, the unit will continue to function with the error code displayed.

START AND IGNITION SYSTEM

The engine starting system provides means of rotating the NZ compressor to establish airflow through the engine. Air pressure to start the engines is normally obtained from the APU, but can be supplied by ground equipment or an operating engine. Dual, physically and electrically independent 4-joule ignitions systems are 115 Volt AC

powered. The Standby Ignition System is powered on the Standby AC Buss. The Battery Bus provides operating power for the standby inverter which powers the Standby AC Buss.
The engine start system provides a means of rotating the Nz compressor to an RPM at which a successful start can occur when ignition and fuel are supplied. The system is also used to provide an assist for in flight starts when required. The ignition system initiates fuel combustion during ground and in flight starting and is used for flameout protection when required.

START SYSTEM

The start system consists of the following:

- An air-driven starter.
- An electrically controlled, pneumatically operated starter valve.
- A pressure switch to operate a starter VALVE OPEN light.

Ignition switches on the pilot's overhead panel control the operation of the start system.
On the ground, the engines can be started from any one of the following three separate air sources:

- APU
- Ground carts
- Crossbleed from an operating engine.

Placing an ignition switch to the GND START position supplies electrical power to:

- Open the pylon bleed air valve.
- Open the starter valve.
- Close the high stage bleed air valve.

Electrical power for all four engine start systems is 28V DC through a single ENG START AIR CONT CB on the BAT BUS section of the P6 panel. Starter valve operation is indicated by the starter VALVE OPEN light coming on when the valve opens, then going out when the valve closes.
NOTE: An increase in duct pressure does not indicate the starter valve has closed. Closure of the pylon bleed air valve will increase duct pressure.

IGNITION SYSTEM

The normal ignition system consists of the following for each engine:

- Two ignition switches
- A start-lever operated ignition micro switch
- Two ignition exciter boxes
- Two igniters

There is also a single STANDBY IGNITION switch located on the pilot's overhead panel, which permits energizing either igniter in all engines at the same time.
Electrical power for the normal ignition system is 115V AC through the IGNITION 1 CBs on the AC BUS 1 section of P6 and the IGNITION 2 CBs on AC BUS 3. The standby ignition system receives its power through the STBY IGN CBs on the AC STBY section of P6.
IGNITION SWITCHES
There are two engine ignition switches for each engine. They are labeled SYS 1 and SYS 2. Each switch has the following three positions:

- GND START, which is momentary and must be held.
- OFF
- FLT START, which is self-holding.

With the switch(es) in GRID START, in addition to opening the starter valve, the start lever operated ignition micro switch is armed. Moving the start lever out of CUTOFF closes the switch, providing power to the ignition exciter(s) and energizing the igniter(s).

With the switch(es) in FLT START, the ignition system operates the same as for the GND START switch position, however, the starter circuit is not energized. The FLT START position is selected when continuous ignition is required.

STANDBY IGNITION SWITCH

This switch is a three-position switch, labeled IGN 1, NORM and IGN 2. In the NORM position ignition is controlled through the respective normal ENGINE IGNITION system switches. The other positions of the STANDBY IGNITION switch are used in the event of a loss of normal ac power, The selection of IGN 1 or IGN 2 powers the respective igniter system on all engines from the standby ac bus, provided start levers are in other than the CUTOFF position.

NOTE: Anytime the STANDBY IGNITION switch is out of the NORM position, or STBY IGN 2 CB is out, the captain's compass system is deactivated, because of the system design features to conserve standby electrical power.

4. START LEVER - Electrically controls the fuel control unit (condition actuator), engine fuel shutoff valve, and normal and standby ignition systems using micro switches in the pedestal.

CUTOFF - Closes engine fuel shutoff valve and signals fuel control unit to shut off. Disarms normal and standby ignition.

RICH - Selection is valid only when selected from the cutoff position. Opens engine fuel shutoff valve and signals fuel control unit to open. Increases normal start fuel schedule about 100 pph, reverts to normal as the engine speed reaches idle. Arms normal and standby ignition.

IDLE - Opens engine fuel shutoff valve and signals fuel control unit to open. Arms normal and standby ignition. Arms flight idle circuits.

ENGINE FUEL SYSTEM

The engine fuel system delivers fuel to the engine at the flow rates required to obtain the desired engine thrust output.

FUEL SHUTOFF VALVE

An electrically-operated fuel shutoff valve for each engine is located where the fuel manifold exits the wing leading edge and enters the pylon. The location of these valves protects against fuel loss from pylon engine fuel leaks. The valve is actuated by the respective:

- ENGINE VALVE switch on the Flight Engineer's panel.
- Start lever.
- Engine fire switch.

A "close" signal from any of these sources overrides any "open" signal.

ENGINE FUEL PUMP

The engine fuel pump has a centrifugal first stage and two parallel gear stages. The fuel heater and filter units are between the first stage and the two gear stages. If the first stage fails, fuel bypasses this stage, as well as the heater and filter to maintain engine operation.

The main gear stage supplies the fuel control unit; the smaller gear stage (hydraulic stage) provides hydraulic fuel pressure for actuating the variable stator vanes and the Ni 3.0 bleed valve. In the event the hydraulic stage fails, the main stage will supply all the actuators through a transfer valve. It is only effective at high fuel flow; if the fuel flow is reduced, all bleeds will open and may remain open; the engine will produce some thrust, but N2 rpm and EGT will be high. If fuel flow is reduced to near idle, the stator vanes will close and remain closed.

FUEL FILTER

The fuel filter traps ice crystals or other foreign matter as fuel flows through it. If the filter becomes clogged, a bypass valve opens to allow continued operation with unfiltered fuel.

When a pressure differential exists across the filter as a result of clogging, a pressure switch illuminates an ICING light on the Flight Engineer's panel.

FUEL HEATER

The fuel heater is used to melt ice crystals trapped in the fuel filter. Turning on the associated FUEL HEAT switch opens a solenoid controlled shutoff valve. Opening the valve permits 15th-stage air to flow through the heater, raising the temperature of the fuel entering the filter, which in turn, causes the ice crystals to melt.

The fuel heat system uses HP bleed air as the heat source. Immediately after the system is turned on, EPR will drop slightly due to the increased bleed offtake. As the engine fuel supply heats up, a secondary effect of fuel temperature on the operations of the hydromechanical fuel control unit will cause an increase in EPR stabilizing at approximately 0.03 above the fuel heat off value (assuming thrust lever is held constant). N,, N2, and EGT will increase correspondingly.

FUEL CONTROL UNIT

To accomplish its functions, the fuel control unit uses the following inputs;

- Thrust lever position - Pilot command input. (increasing thrust level increases fuel flow.)
- Burner pressure (Pb) - A reference of air mass to establish fuel flow requirements. (Increasing burner pressure increases fuel flow.)
- N2 rpm - Reference NZ to throttle command signal. (Increasing NZ rpm decreases fuel flow until steady state exists.)
- Engine inlet temperature (Tt2) - Adjust acceleration and deceleration fuel schedule to TAT. Cold TAT increases fuel schedule.

A condition actuator mounted on the fuel control unit is electrically operated by the start lever and accomplishes the following:

- Provides opening and closing command to fuel control outlet shutoff valve.
- Provides initial starting fuel flow of:

Idle selection = 950 t 100 pph

Rich selection =1,050 t 100 pph

- Provides a change to idle fuel flow.

Ground idle = approx. 28% N,
Flight idle = approx. 55% N,
Changes from ground idle to flight idle if, airborne and landing flaps selected.
Changes from flight idle to ground idle when TILT sensors indicate on ground after a five second time delay.

FUEL FLOWMETER

The fuel flow meter measures the rate of fuel flow in pounds per hour (pph). The flow rate is transmitted to the respective fuel flow indicator on the center instrument panel and to the FUEL USED gage on the Flight Engineer's panel.

OIL COOLER

The oil cooler uses a heat exchanger principle. The fuel being delivered for combustion flows through the cooler and absorbs the oil's heat.

PRESSURIZING VALVE

The pressurizing valve distributes fuel flow to the primary and secondary manifolds according to the rate of fuel flow. Low fuel flow utilizes only the primary system.

FUEL NOZZLES

The fuel nozzles are externally mounted and extend into the combustion chamber. The nozzles have primary and secondary fuel spray outlets.

OIL SYSTEM

The engine has a self-contained oil system to cool and lubricate engine bearings and gears.
Oil flows by gravity from an oil tank to an oil pump. The oil is pumped through a filter, a fuel oil cooler, and then to the engine bearings and gears. The hot oil from these components is collected, and scavenge pumps return the oil to the tank.

OIL TANK

The oil tank is located on the left side of the engine. Because of wing dihedral, left-wing oil tanks hold about three quarts more than right-wing tanks. Quantity is displayed on a gage on the Flight Engineer's panel.
Oil quantity drops approximately one gallon on shutdown. Oil does not drain into the engine while it is shut down.

OIL FILTER

Oil, under pressure from the pump, is delivered to the oil filter. If the filter becomes clogged, a bypass valve opens to permit unrestricted flow to the system. A filter differential pressure switch will detect a clogged filter and illuminate an engine oil pressure light on the forward caution panel.

FUEL OIL COOLER

The fuel oil cooler cools the oil by heat exchange; fuel flowing through the cooler absorbs the oil's heat. An oil temperature regulator, controlled by a thermostat, allows oil to by pass the cooler as necessary to maintain the preset oil temperature. Oil temperature leaving the cooler is read on a gage on the Flight Engineer's panel.

PRESSURE REGULATOR

The pressure regulator maintains oil system pressure by relieving excess pressure to the inlet side of the oil pump.

COMPRESSOR BLEED CONTROL SYSTEM

The compressor bleed control system provides compressor stall prevention to overcome unstable airflow conditions within the compressor. Unstable airflow conditions are corrected by bleeding air from various compressor sections to reduce the pressure ratio across the compressor for a given airflow. These bleed systems, acting with the variable stator system, permit optimum compressor performance to achieve desired thrust output and maintain desired fuel consumption while retaining adequate compressor stall margin. The compressor bleed control system is comprised of a 3.0 bleed valve (ring), 3.5 bleed valves, start bleed control system, tandem bleed control system and a reverser actuated bleed system (RABS). A variable stator system is also included to permit optimum Nz compressor operation.

3.0 BLEED VALVE (RING)

The 3.0 bleed ring is located around the engine case between the Ni and Nz compressors. When the ring is opened air from the rear of the Ni compressor is exhausted into the fan exhaust through multiple openings. The 3.0 bleed ring actuator is powered hydraulically using fuel pressure from the engine fuel system. The 3.0 bleed ring is controlled by the tandem bleed system.

3.5 BLEED VALVES

There are three, 3.5 bleed valves located on the upper right, lower right and upper left of the engine case. When open, these valves dump air from the 9th stage of compression. The valves are spring loaded open and are commanded closed by a fuel signal from the fuel control. 15th stage bleed air pressure closes the valves.

VARIABLE STATOR SYSTEM

The engine N2 compressor incorporates variable stator vanes at the inlet of the Nz compressor and the 5th through 7th stages of compression. A engine vane control and vane actuator, utilizing engine fuel pressure, positions the stator vanes to provide the most efficient air flow through the N2 compressor.

START BLEED CONTROL SYSTEM

The start bleed control system utilizes the upper right and upper left, 3.5 bleed valves to decrease air drag on the Nz compressor during engine start. Springs hold the bleed valves open during the starting sequence. The start bleed control system closes them at approximately 52% Nz rpm. The start bleed control system uses a fuel pressure signal from the engine fuel control to signal the 3.5 bleed valves closed. 15th stage bleed air pressure closes the 3.5 bleed valves.

TANDEM BLEED SYSTEM

The tandem bleed system provides increased compressor stall margin during rapid deceleration at altitude and improved stall margins during ground operations. A Pressure Ratio Bleed Control schedules the tandem bleed system either to full open or full closed. The tandem bleed system controls the 3.0 bleed ring and the lower right 3.5 bleed valve. On the ground the tandem bleed system will close at approximately 1.20 EPR. The system will open when decelerating the engine through the same EPR range. Inflight, the tandem bleed system will open or close at various EPR settings depending on airspeed and altitude (see Estimated Tandem Bleed Operating Schedule, Supplemental Procedures).

REVERSER ACTUATED BLEED SYSTEM

The reverser actuated bleed system utilizes the tandem bleed system (3.0 bleed ring and the lower right 3.5 bleed valve) and the upper left 3.5 bleed valve to improve compressor stall margins during reverse thrust operations. When the fan reverser sleeve moves out of the stowed position, two electrical switches are actuated. One switch signals the tandem bleed system to open and the other signals the upper left 3.5 bleed valve to open. These valves will remain open as long as the fan reverser sleeve remains unstowed.