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AIR CONDITIONING

The pneumatics system provides compressed air at a constant flow rate to the two air conditioning packs. In these units the air temperature is modified to keep the cabin and cockpit comfortable. In normal operation some of the air from the left pack provides conditioned air to the cockpit. The rest of the air from the left pack mixes with the air from the right pack in a distribution duct and provides conditioned air to the passenger cabin. The air is exhausted through the pressurisation system at a flow rate that allows the cabin to be pressurised.

The air conditioning packs are located beneath the floor in the centre fuselage area. An air conditioning pack valve controls the flow of air from the pneumatics system into each pack. The pack valves are controlled by two switches on the Flight Engineers panel.

In each pack the air is split into three paths.

In one path the air passes through a refrigeration unit, then to a set of mixing valves. The mixing valves mix the refrigerated air with air from the other two paths. This allows the air to be delivered to the cabin at the proper temperature.

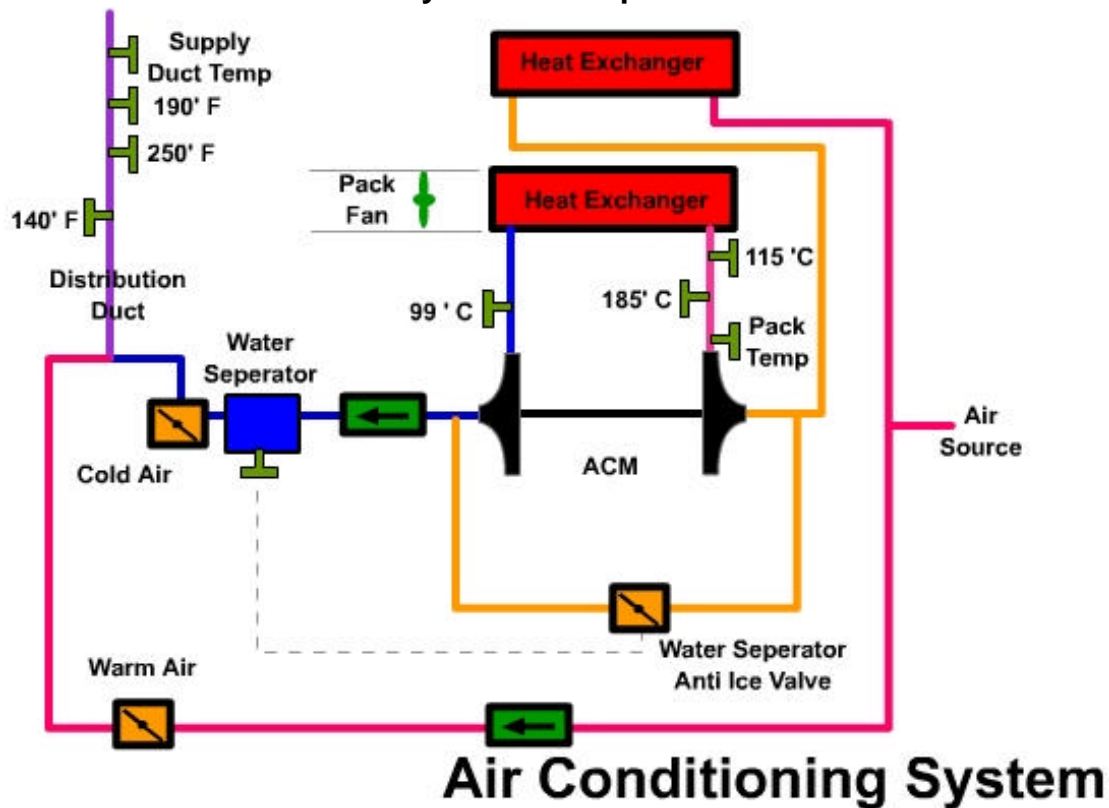
The second path to the mixing valves delivers hot air directly.

The third path is through only a portion of the refrigeration unit, and it reaches the mixing valves at a moderate temperature. The refrigeration unit is called an air cycle machine. It operates on the same principle as any other refrigeration device, except that it uses air instead of freon for refrigeration.

The usual compression cooling and expansion seen in any refrigeration cycle is accomplished in the air cycle machine by a compressor, the secondary heat exchanger and an expansion turbine. The work extracted by the turbine is transmitted by a shaft to the compressor. A primary heat exchanger cools the air before it reaches the compressor, and thus increases the efficiency of the air cycle machine.



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The primary and secondary heat exchangers are normally cooled by air picked up by two inlets on the bottom of the fuselage. The air passes through the primary and secondary heat exchangers and out through a set of louvers at each heat exchanger. Doors at the inlets control the airflow through the heat exchangers. The cooling door and louvers on each pack are interconnected and driven by a single motor. Pack temperature is most vitally affected by the position of the cooling doors. The pack cooling doors are controlled switches on the Flight Engineer's panel. On some aircraft the cooling door switches have positions to open and close, and are spring loaded to a centre off position. Some are on open, off and close with no spring loading. Others are equipped with automatic operated pack cooling doors which will modulate to keep the pack at the proper temperature. These doors have the open and close positions, but the centre position is auto. The centre position is not spring loaded. When the cooling door switch is left in the auto position the cooling doors will remain open while the airplane is on the ground or the flaps are not up. Once flaps are retracted, the associated pack temperature will be automatically regulated to a temperature schedule bias altitude. Below 10,000 feet the temperature is kept at 125 degrees C. From 10,000 feet to 30,000 feet the temperature decreases linearly to 45 degrees C, and remains at 45 degrees C as altitude increases further. This schedule should be used if the doors must be controlled manually. To provide additional cooling for low speed flight and ground operation, an electric fan for each pack is used to force air through the heat exchangers.



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This fan will operate when the pack is on and the inboard flaps are not fully retracted, or when a pack is on and the airplane is on the ground. The ground cooling fan has its own motor driven air inlet door on the side of the fuselage that remains open when the fan is in operation. When a pack fan is started. It draws a very heavy load from the electrical system, and when stabilised, fan consumes about ten kilowatts of power. These are the highest loads on the electrical system.

To monitor the operation of the air cycle machines, each has a temperature transmitter at the outlet off its compressor. The temperature sensed is displayed on a pack temperature gauge for each pack. As more air is directed through cycle machine to provide more cooling, more compression is required from the compressor. This results in a higher compressor outlet temperature. Therefore, the pack temperature gauge monitors air cycle machine workload. To protect the compressor from excessively high temperature an over temperature sensor at the outlet of the compressor will cause the pack to shut down if the temperature reaches the limiting value. Another temperature limiting sensor located at the inlet to the turbine. This uses the temperature of the air as an indication of the energy in the air. If the temperature of the air, and thus the energy entering the turbine becomes too high, the pack will shut down to prevent an overspeed.

In order to return the pack to operation after the temperature in the pack has reduced, a reset button on the pack control panel is provided. The pack cannot be returned to operation until the button has been pressed.

If the pack fan is operating when an air conditioning pack trips off, the fan will continue to operate. The fan will stop when the pack temperature drops, the pack switch is turned off, and the reset button is pressed.

To allow unattended ground operation of the air conditioning system in the 727, the pack trip off sensing and the pack valves are powered from the battery transfer bus. Should the AC electrical power fail, the pack cooling fans will stop. Hot air from the APU will overheat the pack and a pack trip will occur, providing the battery transfer bus is powered. This is one reason for leaving the battery switch on.

As air is cooled it will hold loss moisture. To remove this condensation a water separator is installed downstream of the air cycle machine turbine. The water separator swirls the air over an impingement surface causing the moisture to drop out. This water can be seen coming from the lower fuselage on humid days. The air cycle machine is capable of lowering air temperatures below freezing, which would cause the moisture in the water separator to freeze. To prevent ice accumulation from blocking the water separator, a sensor monitors the temperature. If the temperature gets too low, a water separator anti-ice valve is opened which allows warm air to bypass the air cycle machine and keep the temperature above freezing. 35F.

The air conditioning units are controlled by switches on the Flight engineer's panel. Each switch opens and closes its pack valve at a rate that will not overload the air cycle machine. The pack valves are powered from the battery transfer bus.

Each air mix valve set is actually three valves ganged together, one hot, one



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intermediate, and one cold. These valves operate together to provide the proper mixing of hot, cool, and cold air. There is a set of three valves for each air conditioning pack.

As the outside air temperature drops, the temperature of the cooling air passing through the heat exchangers is low enough to provide sufficient temperature drop in the conditioned air. To compensate, the intermediate valve opens, allowing air to bypass the turbine and flow directly from the secondary heat exchanger into the cabin or cockpit. The turbine slows as a result of this bypassing action causing the compressor to be driven at a slower speed. This allows some of the compressed air to bypass the compressor, flowing directly from the primary heat exchanger to the secondary heat exchanger. The restriction to airflow caused by the air cycle machine is reduced as a result of this bypassing, reducing the need for high-pressure bleed air. Reducing the need for high stage bleed air improves engine efficiency, reducing the amount of fuel being used by the engine.

The temperatures in the cabin and cockpit are normally controlled by automatic temperature regulators. Each regulator provides signals to a motor which drives the associated air mix valve. Each temperature regulator receives inputs from a temperature sensor in the cockpit or cabin and a temperature selector on the flight engineer's panel. The temperature sensor in the forward cabin provides temperature signals to the automatic temperature regulator for the right pack, and the cockpit temperature and left temperature selector position are sent to the temperature regulator for the left pack. The position of each air mix valve is shown on an indicator next to the associated temperature selector on the Flight Engineer's panel. The air mix valve moves to the full cold position automatically when the associated pack valve is closed.

Conditioned air flowing from the air mix valves enters a common distribution duct. From this ducting a small portion of the air is directed into the cockpit. The remainder going to the passenger cabin. Both packs supply the distribution ducting; therefore the same distribution ratio of air to the cabin and cockpit would result whether one or both packs are in operation.

The air to the passenger cabin flows through risers between the windows to keep the cabin walls warm. The air in the cabin eventually flows out through a grill along the floor line into the lower fuselage where it is exhausted through the pressurisation valves (outflow).

Duct temperature is automatically restricted when the temperature control is operating in the automatic range. A temperature sensor in the duct downstream of each air mix valve signals the associated automatic temperature regulator if the temperature reaches a limiting value. When this limiting temperature is reached, a circuit called the topping circuit, prevents the mixing valve from moving toward a higher temperature position.

If an automatic temperature regulator fails to control the temperature of the air satisfactorily, the associated air mix valve can be controlled manually. - To operate the air mix valve manually, spring tension must be overcome and the



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selector rotated to the manual position. In this position the automatic temperature regulator is cut out. Holding the selector lightly against spring tension to the cool position will cause the air mix valve to move towards cold. In the warm position, the valve will move towards hot.

To prevent the air from a pack getting too hot, should the automatic temperature regulator fail, a second temperature sensor is installed downstream of each mixing valve. When the limiting temperature is reached, the associated air mix valve will move to the full cold position, and the duct overheat light next to the associated temperature controller will illuminate. If the automatic temperature regulator and the duct overheat protection both fail, to prevent the duct temperature from rising, a third temperature sensor will cause the pack to trip. If the overheat and pack trip are on the left portion of the systems the location of the supply duct temperature transmitter near the right air mix valve will prevent the temperature indication from reaching the trip off temperature.

To regain control of the temperature regulating networks and turn off the trip lights after an overheat has occurred, a reset button is installed on the temperature control panel. Once the temperature has reduced, pressing the button will return the temperature control system to normal operation.

A temperature gauge on the flight engineers panel is used to monitor the temperature of the air being supplied to the cabin at two locations. The air temperature selector can be used to select the temperature in the forward and aft supply ducts, the main supply distribution duct, and in the forward and aft cabins.

Air is tapped off at the cold side of the left air conditioning pack and delivered to the individually controlled outlets above the passengers, the lavatories, and the cockpit. This is referred to as the gasper system. To increase the flow of gasper air, a fan is installed in the gasper ducting. A switch on the flight engineers air conditioning control panel turns the gasper fan on or off.

If the left air conditioning pack is not operating when the gasper fan is on, cabin air is recirculated through the gasper system.

Conditioned air flows through the airplane and exhausts through three principal exit systems. First of these is the normal pressurisation outflow valve. Operation of this valve will be covered under pressurisation.

Some air flows into the electronic equipment compartment and circulates through the various electronic components; it passes through electronic equipment and circuit breaker panels in the cockpit, the electronic equipment bay, and the weather radar compartment. This air absorbs the heat generated by these units and carries it overboard through an exhaust system on the forward right side of the fuselage.

In normal flight, cabin differential pressure provides necessary airflow through this system. Since the electronic equipment operates continuously, a means of inducing airflow on the ground and at low cabin pressure differential pressure is required. To provide this flow, an electric fan has been installed in the exhaust duct. This fan comes on automatically at low cabin differential pressure. The exhaust to this system has a large and small outlet. So that unrestricted flow can be achieved at low cabin differentials, both outlets are



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used. As the flow rate increases, a flow rate sensitive valve closes preventing excessive loss of air at high differential pressures.

A warning light on the lower right corner of the flight engineers panel will alert the crew to inadequate cooling of electronic equipment. A sensor in the cooling air outlet monitors airflow through the cooling system. If cooling airflow becomes inadequate the "no equipment cooling" light will come on. The cargo compartments on the 727 are class D cargo compartments, which are designed to confine a fire without endangering the safety of the airplane or the occupants. No air circulates through them although a small amount of air flows through the equalisation valves to maintain equal pressure between the cargo compartment and the surrounding cavities, should cabin pressure vary. If a fire develops, it will smother itself as the oxygen in the compartment is consumed. To maintain temperature in the forward cargo compartment, conditioned air from the cabin flows around an airtight inner shell then is discharged through the cargo heat outflow valve. Approximately 30% of the air in the aircraft will exit through this valve.

A switch on the flight engineers panel controls the cargo heat outflow valve. In the normal position the valve is open, permitting air circulation around the forward cargo compartment. If a pressurisation problem should occur, closing the switch can stop the flow of air through this exit. Without airflow around the forward cargo compartment the temperature within the compartment will drop rapidly to a much lower value.

The air that passes from the cabin to the pressurisation outflow valve in the aft fuselage of the airplane heats the aft cargo compartment.

An automatic pack trip system is incorporated in the 727 200 series aircraft. With the system armed before takeoff, loss of thrust on any engine will trip off both packs. This allows the engines to develop somewhat higher thrust for the remainder of the takeoff and initial climb. In addition, both pack fans will stop, thereby reducing the electrical load. To arm the auto pack trip system, the airplane must be on the ground, the flaps must be out of the up position, the auto pack trip switch must be in the normal position, and all engines must be above 1.5 EPR.

When the flaps reach the up position after takeoff the auto pack trip system will be deactivated. After takeoff, and when clear of obstacles the auto pack trip switch should be returned to the cool position. This will deactivate the auto pack trip system. Should any engine lose power below 1.3 EPR both packs will trip off, both pack valves will close, both pack fans will stop and both pack trip lights will illuminate. In addition, an engine fail light will illuminate on each side of the pilot's glare shield. These engine fail lights can be extinguished by pressing on either light cap. When a substantial power reduction is anticipated, such as a noise abatement takeoff. The flight engineers should anticipate the thrust reduction and place the auto pack trip switch to cut out prior to reducing thrust to remove the possibility of an inadvertent auto pack trip.

The airplane is equipped with a means of controlling the temperature in the aft cabin without affecting the temperature of the forward cabin. This is done through the aft cabin zone temperature system. A single switch operates two



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valves in this system. This allows warm air from the right air conditioning pack to enter the forward or aft cabin bringing about the requested change in aft cabin temperature. Should the aft cabin ducting overheat an amber light on the panel will illuminate. Both zone control valves will close and the needle will centre. The flight engineer can monitor the use of this system with the air temperature selector.

APU

The auxiliary power unit, or APU in the Boeing 727 is a small turbine engine mounted between the main wheel wells. It draws air from the wheel well area for combustion and cooling and exhausts through louvers in the top of the right wing root. Most of the accessories are mounted on the left hand side of the unit. The components of most interest are the APU starter (electric), hour meter, fuel control unit, 3-speed switch (3 psi oil, 35%, 95%) Generator, tachometer (RPM) note that these last two components are interchangeable with their brothers on the engine. Power for starting the APU comes directly from the airplane battery. The battery switch must be on when operating the APU, if the switch is turned off the APU will shutdown.

The APU uses fuel from the number 2 tank. The APU fuel shutoff valve is located at the tank. It is opened or closed by the APU master switch.

Operation of either APU fire switch or activation of the auto fire shutdown circuitry will also close this valve

The controls for the unit may be found in the flight deck at the flight engineers auxiliary panel and in the left hand wheel well. It contains controls for starting and stopping the APU, fire detection and protection, generator operation, and gauges for monitoring electrical load and exhaust temperature.

There is a three-position control switch marked Off, ON and Start.

It remains in the OFF position when the APU is shut down and in the ON position when the APU is operating. It must be held in the START position against spring pressure when starting the APU. ON is the normal operating position. Selecting this position prior to starting will open the APU fuel valve. After the fuel valve is opened, positioning the master switch to START will initiate the automatic starting sequence. When the APU crank light comes on, the automatic starting sequence has begun. Once the light is on, the master switch may be released to the ON position. During the start sequence, if the EGT does not rise within 15 seconds or there is no frequency on the AC meter within 30 seconds, the APU fire shutoff handle should be pulled to interrupt the start sequence. The APU crank light goes out when the starter releases. [Click here to view starting sequence.](#)

Self-contained lubrication system requires no crew monitoring. The APU will not operate if the oil pressure fails.

An exhaust temperature gauge located in the lower right corner of the APU control panel shows APU turbine exhaust temperature in degrees C. The temperature will vary widely depending on bleed air loads. The green band is the normal operating range, and the red radial is the maximum operating temperature.



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The APU is governor controlled to maintain 100% RPM, thus it can be geared to drive an AC generator directly. This generator can supply electrical power to all airplane systems for ground operation.

Bleed air is extracted from the APU compressor to be used for airplane air conditioning and for engine starting on the ground. An APU bleed air valve is installed on the APU to control flow of bleed air from the APU to the bleed air distribution system. The APU bleed air valve will open if the APU has reached operating RPM and either or both Engine No.2/APU BLEED switches are in the OPEN position.

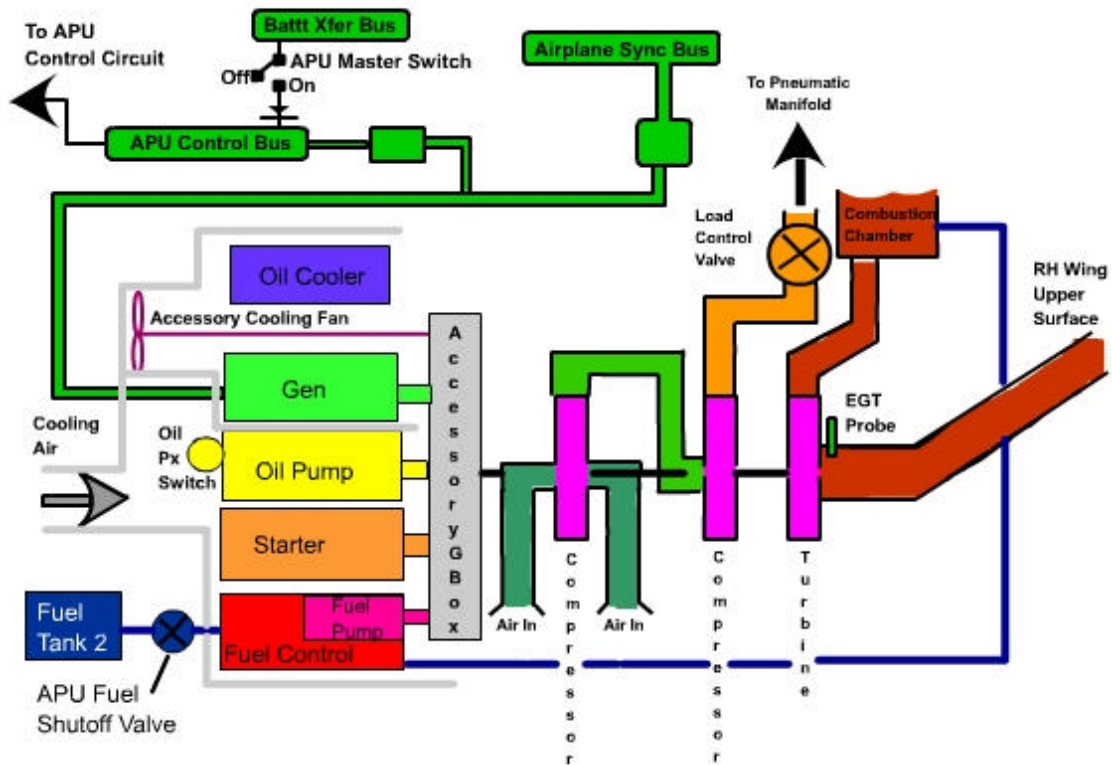
An APU light is located on the door warning annunciator at the flight engineers panel. This light will come on any time the fuel valve is open if the number 1 DC electrical bus is powered.

For maintenance personnel external control of the APU, a second APU control panel is located in the left wheel well between the fuselage and the gear strut. It is not used for normal operation. The start switch on this panel will start the APU if the battery switch and APU master switch in the cockpit are on. The stop switch will shut the APU down. Also the panel contains a fire switch, a fire warning light, and a bottle discharge button. These controls permit fighting an APU fire without going to the cockpit.

The APU will shut down automatically for the following reasons; loss of oil pressure or overspeed will cause the fuel to be cut off at the fuel control in the APU. The fire detection loop, if it reaches the warning temperature, will close the fuel shutoff valve in the number 2 tank and at the fuel control and the APU will stop. Heat sensitive probes in the turbine exhaust provide other protection. First, these probes cause the APU to be unloaded by modulating the APU bleed air valve toward close; if closing the bleed air valve fails to solve the high exhaust temperature, the probes will cause the fuel control to reduce fuel flow until the temperature is lowered sufficiently or the APU flames out.



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On the 200 series aircraft a turbine and compressor assembly called a flow multiplier is used to improve the airflow from the APU. It draws in additional air from the right wheel well adding it to the APU compressed air output. This higher volume of air makes it possible for the APU to supply air to both air conditioning packs. During single pack operation however, the flow multiplier shut off valve remains closed and the turbine is bypassed.

A FLOW MULTIPLIER OVERHEAT light on some airplanes and a BLEED AIR light on others warns of an overheat in the output of the flow multiplier compressor. The overheat will cause the APU bleed air valve to close. Cycling the number two engine bleed switches will reset the APU bleed air valve. Further protection is provided by a fusible plug, which should it melt, it will close the flow multiplier shutoff valve, preventing compressed air from reaching the flow multiplier turbine.

Pressures in the pneumatic ducts can be read on the duct pressure gauge under most conditions. The duct pressures will read zero, when either air conditioning pack is turned on, if the APU is the only source of air in the pneumatic ducting (200 Series). Turning either pack switch on prevents APU bleed air from reading on the duct pressure transmitters.

The APU on the Boeing 727 can be used for ground operation only. Electrical loads and EGT limits must be observed for all operations. The EGT limits are red radial for maximum and the green band for continuous operation. The electrical load limit is 165 amps, the higher rating is due to the improved



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cooling of the APU installation. The APU should be operated without pneumatic load for at least one minute after start or prior to shut down. A pretty good piece of kit, most of the problems you experience are to do with starting (3 speed switch or over-speed switch) occasionally the shorting link on the exhaust disconnect link. Extensive troubleshooting will require the test set, (not a great deal you can do down route). It's manufactured by Allied Signal (Garrett).

APU Starter is limited to 1 min on 4 min off

Max time for APU fire test AC Busses powered 30 sec- 45 sec. Battery power 60 sec. Max

APU generator load 165 amps

One pack on for cooling (100 Series)

Two packs on for heating

Two packs on for cooling (If flow multiplier installed)

Normal operating EGT Green Band (Marked @ 700)

Max Operating Red Radial line (Marked 750 - 790)

APU EGT Operating	GTCP85-98 and 98C	98CK
Maximum	760 °C	710 °C
Continuous	710 °C	663 °C

AUTOPILOT

The Autopilot (A/P) can control the aircraft in a climb, cruise decent and approach phases of flight or as directed manually by the pilot via the control knob. It may also be directed by signals from the VHF, GPS, and INS navigation systems. It can also find and maintain a pre selected heading, altitude, pitch attitude or operate in a split axis configuration.

It requires 115V AC for operation from the aircraft generators or an external source. If using the latter you need to operate the ground test switch.

Electrical interlocks prevent selection or operation unless all the proper conditions for correct functioning are satisfied.

INTERLOCKS

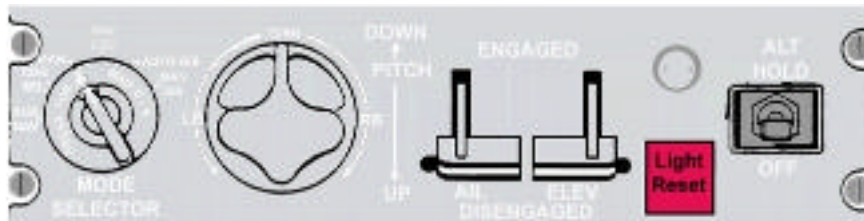
Mandatory interlocks are at least one yaw damper switched on and it's disengage flag out of view, turn and pitch controller in neutral detent, operating vertical gyro if these conditions are correct you will be able to engage the aileron switch. To engage the elevator switch the aileron switch must be engaged and the cruise stabilizer trim switch must be in the normal position.



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The autopilot will disengage if any of the following actions or conditions occur. Yaw damper positioned to off, capt or co pilot press autopilot release switch, power to vertical gyro is lost, switching compass source or power to A/P aileron (roll) is lost. Elevator channel disengage will occur if the stabilizer trim switch is activated, pitch channel selector is switched (A or B) cruise trim switch is activated, A/P and cruise trim cut out switches positioned to cut out, electrical power to the A/P elevator (pitch) is lost.

A/P mode will return to manual (turn and pitch knob) if any of the following occur reference VHF/GPS/INS is switched, turn and pitch controller moved out of detent, ILS frequency is switched while in approach mode.



ELECTRICS

AC POWER

Modern transport aircraft use 400-hertz alternating current to power much of their electrical equipment for several reasons. Voltages are easily converted from high to low or low to high. The higher frequencies used in aircraft electrical systems allow components to be smaller, but develop the same power as the 60 hertz devices normally found in the home and industry.

Power for the electrical system is supplied by the three engine driven generators, and as a backup on the ground, APU generator, or an external power source. Normally all of the electrical power in the airplane is produced by the engine driven AC generators.

An AC generator must be rotated at a constant speed throughout the operating RPM range of the engine. This is necessary to maintain the appropriate frequency output of the generator. A generator drive unit, called a constant speed drive, or CSD, which is a hydro mechanical device between the engine drive pad and the generator, accomplishes this function for each generator. Each generator drive unit contains its own integral oil supply and pumps. So that the oil pressure within the unit can be monitored, a low oil pressure light for each unit is included on the flight engineers electrical panel. The amber light will come on if the oil pressure in a unit is too low. To cool the generator drive oil an air-cooled heat exchanger is installed in each drive system. Engine fan stage bleed air continuously provides the required cooling airflow for the heat exchangers. There is an oil temperature gauge for each generator drive unit. These gauges have two scales. Oil "IN" temperature is indicated on the lower scale, which is calibrated from 40 degrees to 160



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degrees Celsius. The upper scale is calibrated from 0 degrees to 30 degrees Celsius, and indicates the rise in the temperature of the oil as it passes through the generator drive unit. Above each temperature gauge is a toggle switch for selecting either the "IN" or "RISE" temperature scale. The "IN" temperature is sensed downstream of the oil cooler as it enters the generator drive. This gives an indication of oil cooler efficiency. The rise temperature is the difference between oil in and oil out temperatures and is an indication of heat generated within the drive unit in rotating the generator, or the workload of the generator drive unit. The rise temperature caution range is 20 degrees to 30 degrees C. If the "IN" temperature reads between 127 degrees and 140 degrees C the operating time limit is two hours. With the "IN" temperature between 140 degrees and 160 degrees C. Operating time limit is 50 minutes.

To the left of each low pressure light is the disconnect switch for the associated generator drive unit. These switches are red guarded and safety wired. The switch under the guard has two positions, Normal and Disconnect. The Disconnect position is momentary contact and spring loaded to the normal position. The generator drive unit can be disconnected by opening the guard and moving the disconnect switch to the disconnect position. This action disengages the mechanical coupling for the generator drive unit from the engine drive pad. Also this action trips the associated generator breaker, breaking the electrical connection between the generator and its load bus. The generator drive can only be reconnected on the ground by maintenance personnel.

The generators each produce three phase, 400 hertz, 115 volt AC electrical power. The voltage produced by a generator depends on there being a magnetic field in the generator. Current flowing from the voltage regulator produces this field. The generator's output voltage is sampled by the voltage regulator which adjusts the current to the field so that the generator's output, when not in parallel, will be 115 volts, plus or minus 5 volts.

Under certain abnormal or emergency conditions it is necessary to reduce the voltage output of the generator to a minimum. The field relay performs this function by interrupting the current flow from the voltage regulator to the generator. With the field relay open, only residual voltage of 10 to 17 volts will be produced if the generator is rotating. Each generator has a field relay controlled by the field switch on the electrical panel. The light next to the field switch will be on when the field relay is open.

When the generator is producing full voltage, it may be connected to its load bus by closing the generator breaker. The load bus is a distribution point for the power produced by the generator. Heavy load items, such as air conditioning pack fans, galleys, and hydraulic "B" pumps are powered directly from the load busses. Power is also sent to the various circuit breaker panels to power electrical equipment throughout the airplane. Each generator breaker is closed with a switch labelled GEN.



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When a generator is connected to its load bus through the generator breakers the generator breaker light next to that generator switch will be out. When the generator is not connected to its load bus the light will be on.

To equalise the loads on the generators, and to protect the electrical system if one generator should fail, the three load buses are connected together by a third set of relays. These bus tie breakers are connected to a common circuit referred to variously as the "tie bus" or the synch bus. The tie bus shuttles power among the three AC load buses as power requirements change, but only when the bus tie breakers are closed. If a generator should fail the tie bus will power the AC load bus associated with that generator through its bus tie breaker.

Since we are dealing with alternating current, we must be certain that the voltages of the various sources we are joining in parallel are "in phase". By this we mean that the positive and negative portions of the two voltages that we are connecting occur at the same time. If we joined the voltage sources when they were not "in phase", serious damage could be done to a generator. In practice, the bus tie breakers are left closed so a single power source could power all three AC load buses. To protect against connecting a generator out of phase, automatic protective circuits prevent the generator breakers from being closed unless the associated generator is in phase with the other generators already powering the system. The bus tie breakers do not have this protective feature.

Some AC powered items are considered to be more critical to safe flight. These are powered through the essential AC bus, which can be supplied by any of the three generators directly without the necessity of its generator breaker being closed. The selected generator's field relay must be closed so that the generator will be able to supply electrical power. The essential power selector switch on the upper right side of the electrical panel controls the selection of a power source for the essential AC bus. Normally generator three supplies the essential power, with the other two generators available. The essential AC bus is also powered when external power or the APU is supplying the airplane. Preferences for essential sources are eng 3, 1, 2 in that order. It's due to the loads on each bus, 3 being the lightest load, 2 the heaviest

Red warning lights show failure of the selected essential AC power source. There is a steady red light on the essential power selector panel and a flashing red light, labelled "WARN - PUSH TO RESET", on the pilot's centre instrument panel. The flashing red light can be extinguished by pressing the light cap, but the steady red light will not go out until the essential AC bus is powered from another source.

Certain of the captain's instruments are protected even further. They are powered from the standby AC bus. As long as the essential AC bus is



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powered, it powers the standby AC bus. In the event of a failure of essential AC power in flight the standby bus will automatically be powered by a static inverter. The static inverter is powered from the battery bus.

On the ground the standby AC bus may be powered from the static inverter, however, it is necessary to select "STANDBY" with the essential power selector to do so. This is normally done when standby AC power is required, but AC power is not available from the airplane's generators or an external power source. The essential power selector must be depressed before it can be rotated to the "STANDBY" position. When the selector is moved to the standby position on the ground or in flight the essential AC bus will no longer be powered even if it were powered previously.

The last major AC bus is the AC transfer bus. Normally this bus is powered from the number 3 AC load bus, but under certain conditions can be powered from an external power source. Distribution of electrical power is through the various circuit breaker panels. The lower portion of the P 6 panel is divided into three sections. P6-11, P6-12, and P6-13. These sections are associated with the three AC load buses. 1, 2, and 3 respectively. On each panel is a power light, which glows continuously when the associated bus is powered. The rest of the P6 panel and the P18 panel contain the systems sections with the circuit breakers for those systems.

The upper portion of the P 6 panel is divided from top to bottom into four main sections, P6-1, P6-2, P6-3, and P6-4. The other main circuit breaker panel is P-18, which is located on the left sidewall above the first observer's seat. The P18 panel is further subdivided into four main sections numbered from bottom to top. In general these panels are: P 18-1. Radio equipment; P18-2, light instruments, autopilot, and interphone; P18-3. Passenger accommodation and P18-4. Cockpit lighting, service lights, and exterior lighting. Isolated groups at circuit breakers related to lighting are installed in several other cockpit locations. There are also some circuit breakers, which are inaccessible to the crew located in the electronic equipment compartment.

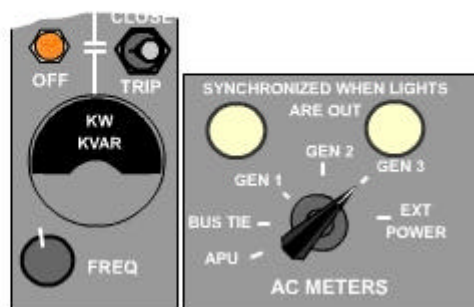
On the right side of the flight engineers panel is an AC meters selector. Each of the three engine driven generators, the APU generator, or the external power can be sampled as well as the voltage and frequency on the synch bus. When a generator is rotating with its field relay open, it will produce 10 to 17 volts residual voltage. This voltage can be read on the voltmeter lower scale by selecting that generator and pushing the residual volts button. When the generator field relay is closed the generator field is energised by the voltage regulator. Now normal voltage can be read on the top scale of the voltmeter. It should read 115 volts plus or minus 5 volts. Above the AC meters selector is a frequency meter. This meter will indicate the frequency of the power source selected by the AC meters selector. When selected to generators 1, 2, or 3. the frequency desired is 400 hertz plus or minus 9 hertz. If the frequency is not 400 hertz, it can be adjusted using the frequency knob on the left panel. The knob for each generator allows adjustment of the



System Descriptions

frequency within about a 15-hertz spread. Frequency will be indicated only when the generator field relay is closed.

Just above the meters selector are two white lights labelled "SYNCHRONIZED WHEN LIGHTS ARE OUT". These lights are used when connecting generators in parallel with the bus tie breakers. Automatic paralleling protection is provided when the generators are brought online normally since the generator breakers are used. If the generator breakers cannot be used because an abnormal or emergency procedure requires the bus tie breakers to be used, the manual paralleling procedure outlined here must be used. After one generator is connected to the synch bus. Selection of another with the AC meters selector will cause the synch lights to flash in unison. They are indicating the synchronisation of the selected generator in comparison to the synch bus. Before closing the bus tie breaker, which places that generator in parallel with any other on the synch bus, the frequency of that generator is adjusted with its frequency knob to 400 hertz so that the lights are flashing slowly. When the lights are out. The generator is synchronized and can be safely paralleled.



When an electrical load is sustained by an engine driven generator, the load is indicated in kilowatts on its kilowatt meter. There is one for each engine driven generator. The maximum continuous load for a single generator that is not operating in parallel is 36 kilowatts. It can sustain an overload of 54 kilowatts for 5 minutes. Two generators operating in parallel are limited to 54 kilowatts total load, and 3 generators in parallel may be operated continuously with a total load of 102.5 kilowatts.

The loads on paralleled generators should be nearly equal, indicating that the generators are sharing the loads equally. Another electrical quantity which can be read on the meters is kilovolt-amperes reactive, or KVARs. The KVAR button is shown in yellow. When the KVAR button is pushed and held, it changes the three meters to read kilovolt amperes reactive. A measure of reactive power. All three meters should show the same readings for reactive power.



System Descriptions

When certain electrical system faults occur, lights on the electrical fault annunciator panel on the flight engineer's auxiliary panel will indicate the type of fault and the system involved. The reset button on the panel is used to put out the annunciator lights when required. The test button is used to test the annunciator lights.

OTHER POWER SOURCES

The system may be powered by the APU generator which is identical to the engine driven generators but is geared directly to the APU accessory drive. When the APU is operating at 100% RPM, the APU generator will be providing 400 hertz power. The controls for the APU generator are located on the flight engineer's auxiliary panel. There is a field switch and a generator breaker switch. Both of these switches are three position lever lock switches. The amber light associated with the field switches is a field off light. The amber light associated with the generator breaker is a generator circuit open light.

The AC ammeter located on the APU control panel indicates the AC load on the APU generator in amps. This ammeter will also indicate external power load in amps if an external power unit is being used. The maximum electrical load when using the APU generator is limited to 165 amps. APU voltage and frequency can be read on the AC meters with APU selected. When the APU is running at normal speed and its generator field relay is closed, closing its generator breaker will connect the APU generator directly to the airplane's synch bus. The individual load buses will be powered from the synch bus if the bus tiebreakers are closed. In normal operation the bus tie breakers are left closed and the transfer of power sources is done with the generator breakers, or in the case of external power, the external power contactor. With all bus tie breakers closed, AC buses 1,2, and 3 are now powered by the APU generator.

An external power unit may be used to provide electrical power to the airplane systems. An AC connected light on the flight engineer's electrical panel comes on when external power is plugged into the nose of the airplane. This light signifies that power is available but it does not show whether the power is actually energizing the airplane's AC buses. By selecting external power with the AC meters selector, the voltage and frequency of the external power can be monitored. Approximately 115 volts and 400 hertz should be indicated before external power is accepted. The external power source can be connected to the airplane electrical system by means of the external power switch. The switch is held in the ON position by a solenoid. A temporary loss of AC power will allow the switch to return to the centre OFF position. Moving the switch to the ON position will connect the external power to the synch bus. As with APU power, the bus tie breakers must be closed for the power from the synch bus to reach the three AC load buses.

ESSENTIAL POWER

Essential power can be supplied on the ground when either the APU or an



System Descriptions

external power source is powering the AC load buses. Power from the number 3 AC load bus is tapped off and supplied to a pair of relays which, when the proper conditions are met, will allow the essential power selector to supply AC power to the essential AC bus. There are certain requirements, which must be met before the essential AC bus can be powered by selecting APU or external power on the essential power selector. When the number 3 AC bus is powered from any source. The essential AC bus will be energized from that bus with APU selected, only if the APU is running and its field relay is closed.

To power the essential AC bus from the external power position. External power must be powering the buses. With the external power switch off, the external power position of the essential power selector is not powered. The bus tie breaker between the synch bus and number 3 load bus must be closed to power the number 3 load bus and feed the external power position. The external power switch has a third position which allows certain outlets and lights in the cabin to be powered without energising any other buses in the airplane. This position of the external power switch is labelled GROUND SERVICE. It was mentioned earlier that the AC transfer bus is normally powered from number 3 AC load bus. The AC transfer bus provides the power for the outlets and lights for the passenger cabin.

If it is desired to energise these circuits without the necessity of powering any other buses in the airplane. The ground service switch position is used. With external power plugged in. moving the external power switch to ground service powers the AC transfer bus without supplying power to any other AC bus.

To prevent damage to electrically powered airplane components, automatic protective features are incorporated into the electrical system. Control of the AC electrical system is provided by latching relays. Which require electrical power for opening or closing. This power is supplied by three units called generator control panels. These control panels must be powered at all times to provide indicator lights, remote control from the second officer's panel, and protective circuits for its generator. There are two power sources for each control panel. If its generator is not operating, the battery powers the control panel with 24 volts DC. The battery switch must be on to provide this power. If the generator associated with the control panel is operating with its field relay closed. It will power its own control panel through a rectifier. Changing 115 volt AC to 28 volt DC.

One feature of the protective circuitry is that not more than one source of power may be connected to the airplane electrical system at the same time. For examples if the airplane generators are powering the airplane, the APU generator breaker open. If the APU generator breaker switch is moved to close. The airplane generator breakers open before the APU generator breaker closes. A similar sequence occurs when connecting an external power source. Conversely if an airplane generator breaker switch is moved to the close position, the APU generator breaker or the external power contactor will open and then the engine driven generator breaker will close. This



System Descriptions

feature is commonly referred to as the "break before make" protection. The generator control panel contains regulating control, and protection circuits to insure that proper power is delivered to the system. The panel maintains constant voltage output from an isolated generator under varying loads and maintains equal sharing of the kilowatt and KVAR loads when the generators are operating in parallel. Protective sensing circuits will cause the bus tie breakers, a generator field relay and generator breaker together. Or a generator breaker alone to open due to system or generator malfunction. In general, the bus tie breaker may be opened by its control switch; it will trip automatically for an excitation fault or a phase unbalance. An excitation fault is usually caused by a failure of the voltage regulator when the voltage regulator calls for too much or too little current to the field windings of a paralleled generator, or if the current is unstable. When a high / low or unstable excitation current is sensed on a generator, the protective system opens the bus tie breakers on that generator to isolate that generator. The synch bus is also monitored by the protective circuits. A short or ground on one of the three phases on the synch bus will show an imbalance in loads on the three phases. This fault, called a phase unbalance, will cause all three bus tie breakers to open. Isolating the generators from the faulty synch bus. A generator field relay will open due to tripping the generator field switch, pulling the engine fire switch, voltage faults, or a differential fault. An excitation fault may be followed immediately by a voltage fault. Once the generator is no longer paralleled, a voltage regulator fault can then be sensed as a high or low voltage. The generator field will trip along with the generator breaker.

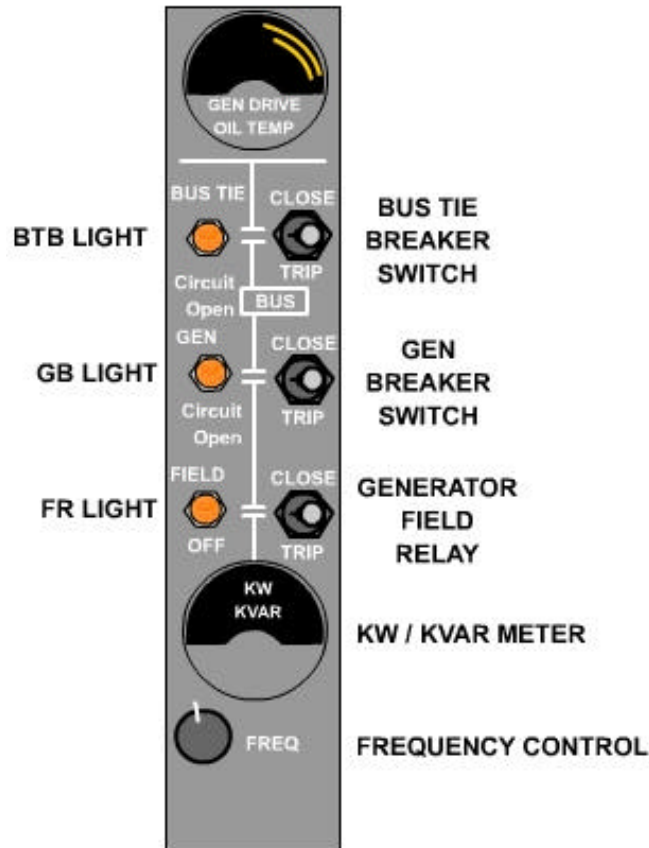
The output of each generator is sensed at the generator and also at the load bus. If a difference exists between the two. Then there is a ground or short in the feeder lines between the generator and the load bus. This fault, called a differential fault or sometimes a feeder fault. It is serious enough so that when it trips the field relay and generator breaker, the field relay is locked out and may not be reset without resorting to an abnormal procedure to defeat the lockout.

A generator breaker will open due to tripping of the generator field relay by any means. Tripping of the generator control switch, or disconnecting the generator drive unit. The generator breaker will also open due to a generator drive underspeed or overspeed, closing the APU generator breaker. Or by turning the external power switch to ON. A tendency for the constant speed drive on a generator to overspeed or under speed, if not corrected by that generator's load controller, will be indicated by a high or low kilowatt load if the generators are paralleled. The KW loads should be monitored to determine if a load controller is failing. If the CSD overspeeds, the generator will assume a greater and greater portion of the airplane's electrical load until the generator is tripped automatically by its overload protection circuits. This will isolate the generator since its bus tie breaker will open. An unparalleled or isolated generator is protected from over or underspeed by a speed switch,



System Descriptions

which opens the generator breaker and prevents essential power from being supplied by that generator.



An electrical fault annunciator is located at the flight engineers auxiliary panel. The series of five lights in the upper part of the first column will indicate an electrical fault in generator one's system. Column two's upper section for generator two, and Column three's upper section for generator three. The bottom centre light, phase unbalance, indicates a fault, which affects all three systems. Under the annunciator panel is a test button for testing its lights. On the left is a reset button which will put out any light remaining after a fault has been corrected, with the exception of a differential fault.

DC POWER

The main DC electrical system is powered by transformer rectifiers. These TR units convert 115 volts AC to 28 volts DC. The number 1 TR unit converts AC power from the number 1 AC load bus. Likewise the number 2 TR converts power from the number 2 AC load bus. The essential TR is powered from the essential AC load bus. This DC power is delivered from the TR units through circuit breakers to the DC buses. Then the DC buses feed the circuit breakers on the P 6 and P 19 panels to power equipment throughout the airplane. The number 1 and number 2 DC buses are connected by a current



System Descriptions

limiter to provide backup for loss of either TR 1 or TR 2. Either TR 1 or TR 2 is capable of carrying the load on both DC buses.

To provide a backup source of power for the essential DC bus, there is a connection to the output from the number 1 TR. Should the essential TR fail; either TR 1 or TR 2 could carry the load on all DC buses. A blocking rectifier in the circuit prevents reverse current flow from the essential DC bus to the normal DC buses. The DC volt and ammeters and the DC meter selector are located at the lower right side of the flight engineers electrical panel. With the DC meter selector in a TR position the voltmeter indicates voltage on the associated DC bus. Amperage is sensed just downstream of the TR unit, so the ammeter indicates current flow through the TR unit that has been selected. Thus, the amperage readout on the gauges is from the TR and the voltage from the DC bus.

BATTERY BUSES

The battery and battery transfer buses are normally powered from the essential DC bus. If the essential DC bus is not powered, and the battery switch is on, the battery and battery transfer buses will be powered from the hot battery bus. The battery transfer bus is powered any time the battery bus is powered. The "transfer" designation implies that the source for that bus will change under certain circumstances. When external power is plugged in, the battery transfer bus will be powered from a TR in the airplane's external power circuitry instead of the battery bus. The hot battery bus is powered by the airplane battery at all times that a serviceable battery is installed. The hot battery transfer bus is powered any time the hot battery bus is powered. When the airplane's electrical system is not energised, the hot battery transfer bus is powered from the hot battery bus. When the essential DC bus is powered, the power source for the hot battery transfer bus becomes the battery bus.

BATTERY

The Boeing 727 has a 24-volt nickel cadmium battery. The battery provides an emergency power source for certain radio and instrument systems, and is required for starting the APU. With the battery switch either on or off, battery voltage is indicated on the voltmeter with the battery selected. The ammeter will show current flow from the battery as negative amperage and charging current as positive amperage. The battery charger is powered from the 115 volt AC transfer bus and is connected to the hot battery bus to charge the battery. Whenever the AC transfer bus is powered, the battery charger will be operating. Charging the airplane's battery regardless of battery switch position. After a high load, such as an APU start, the battery will accept a high charging current. The charger will supply this high current until the battery becomes sufficiently charged so that the current drops below a threshold value. At which time the charger will go into a pulsing mode for two



System Descriptions

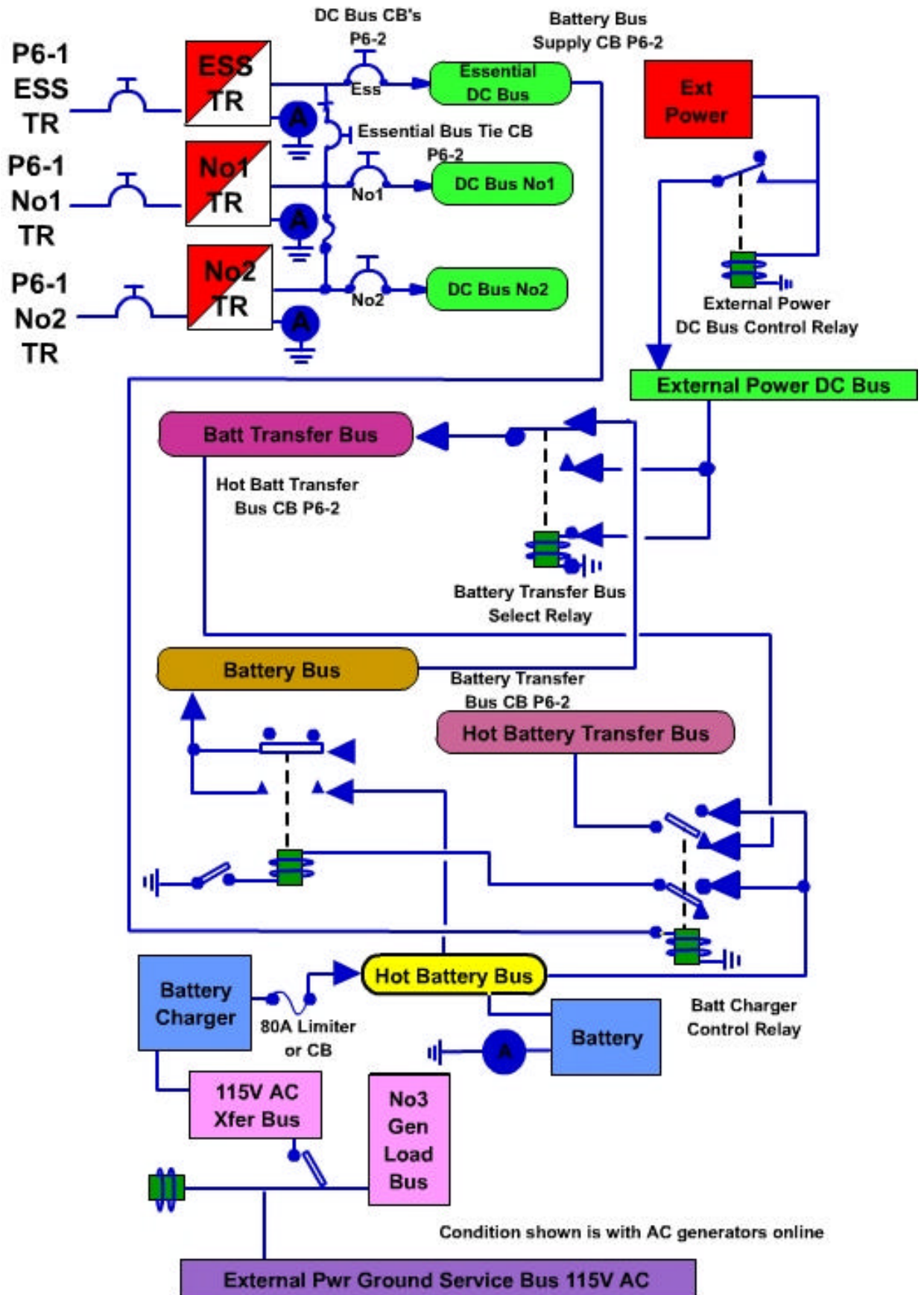
minutes. After the two-minute period the charger will drop to a low steady charge that is barely discernible on the ammeter.

SUMMARY

The standby AC bus is powered whenever the essential AC bus is powered and can be powered from the battery bus through an inverter. The standby DC bus is powered whenever the essential DC bus is powered. When the essential AC bus loses power in flight, or the essential power selector is moved to standby. Both the standby AC and standby DC buses will be powered from the battery bus.



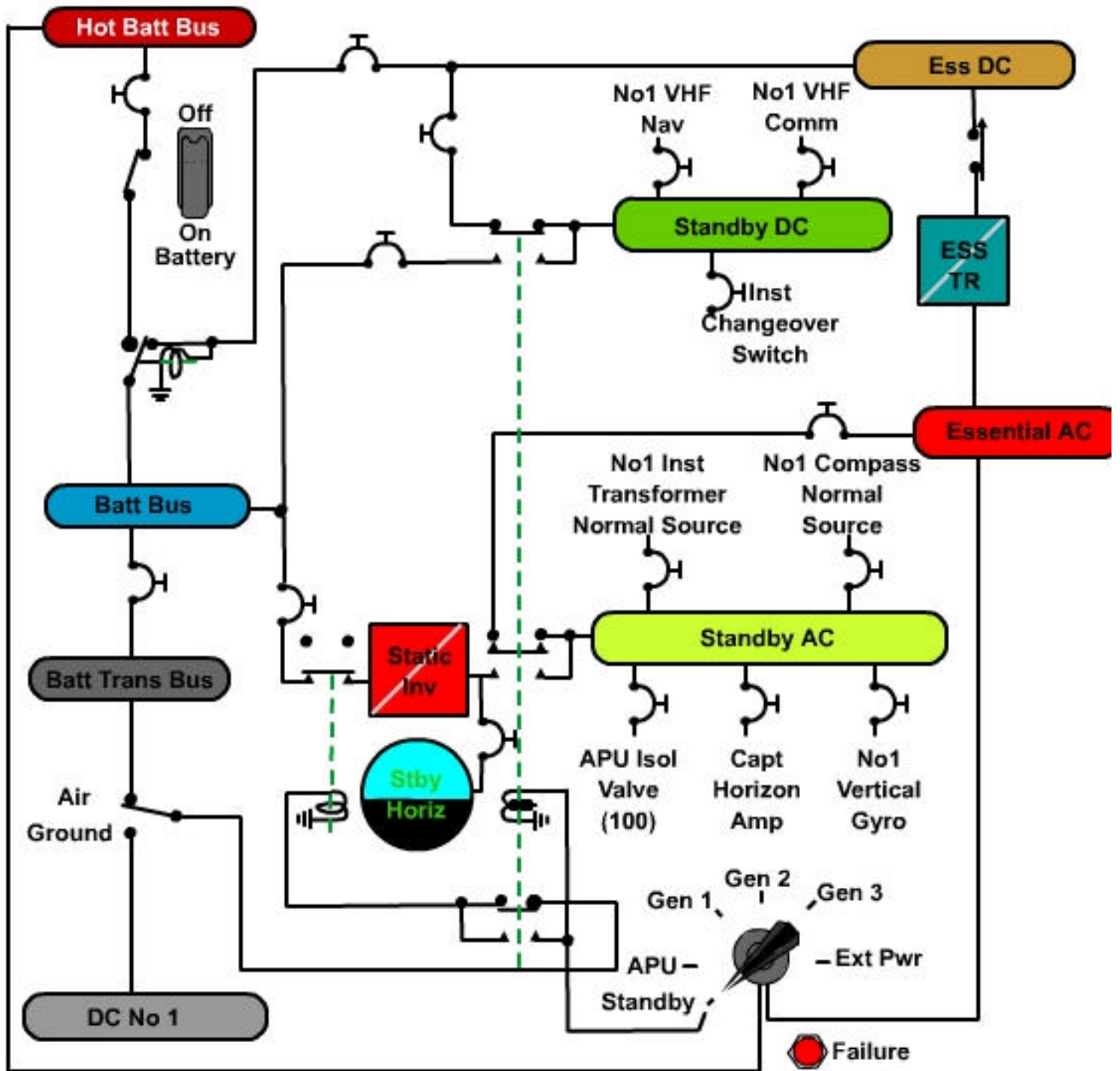
System Descriptions





System Descriptions

Standby Electrical Power (Without Auto Standby Transfer)





System Descriptions

FIRE PROTECTION

The fire protection system is separated into two categories: detection and extinguishing. Detection systems are installed in the engine; wheel well, APU, and lavatory areas. Fire extinguishing capability is available for all engines, the APU, and the lavatories.

A fire detection sensor is wrapped around each engine case. Another fire detection sensor is installed on the engine firewall. The firewalls for #1 and #3 engines are in the engine mounting struts. Due to the mounting of #2 engine in the aft fuselage, there is a single overheat sensor serving both the vertical firewall forward of the engine and the horizontal firewall above.

The electrical power source required for activation of the engine fire detection circuits is essential AC.

The fire shutoff handles are located in the pilot's overhead panel. Upon activation of an engine fire detection systems a red warning light in the fire shutoff handle comes on, a master FIRE WARN light on the glareshield in front of each pilot comes on and a bell sounds in the cockpit. Pressing the bell cut-out button near the fire shutoff handles can silence the bell. On some aircraft also by pressing either FIRE WARN cap on the glareshield, or using the RESET switch on the FE's auxiliary panel. The lights in the fire shutoff handles remain illuminated until the high temperature condition no longer exists or the fire detection system is destroyed.

Two fire extinguisher bottles charged with Freon are available for combating an engine fire. These bottles, with associated plumbing are mounted on the right side of the aft airstair's area. Each bottle has a pressure gage. Required engine fire extinguisher pressure is around 575 pounds per square inch at a bottle temperature of 70 degrees Fahrenheit. Bottle pressure will decrease approximately four PSI for every one-degree drop in bottle temperature.

A selector valve is installed for each engine to control the direction of flow of the extinguishing agent. Pulling a fire shutoff handle arms the associated selector valve. When the discharge button is pushed. The extinguishing agent flows through the selector valve to the appropriate engine.

Fire shutoff handles contain the warning lights for the respective engines. In addition, on some aircraft there is a master "FIRE WARN" light located on the glareshield in front of each pilot. It comes on and the bell sounds along with the illumination of the respective fire detection system light. Pressing on either master FIRE WARN light cap will silence the bell and the FIRE WARN light goes out. The fire shutoff handles are used to actuate the fire switches.

When a fire shutoff handle is pulled the following things occur:

It arms the: Bottle discharge circuit and Engine selector valve. It closes the following Fuel shutoff valve, Engine bleed air (eng.#1 & #3), Bleed air valves, (eng. #2), Wing anti-ice shutoff (eng.#1 & #3), Cowl anti-ice (eng.#2), Hydraulic supply shutoff (eng.#1 & #2), Disarms the associated "A" hydraulic pump low pressure light and trips the generator field relay after a 5 - 10 second delay to allow the valves to close.



System Descriptions

After the fire shutoff handle is pulled, pressing the bottle discharge button releases the extinguishing agent and opens the proper engine selector valve. Once it is open and pressure has built up at the valve, the discharge button may be released. It takes about two seconds for the pressure to build up at the valve. The engine selector valve closes when the extinguishing agent is depleted. The left or right bottle is selected with the bottle transfer switch. After discharging the first bottle and selecting the other, pressing the bottle discharge button again discharges the remaining bottle and the associated bottle discharge light comes on when bottle pressure is diminished. Ordinarily, the bottle transfer switch is positioned to the left bottle.

Three discharge discs are incorporated in the engine fire extinguishing system. Each bottle has a red disc, which ruptures in the event of bottle discharge due to thermal relief. There is a single yellow disc, which is ruptured by intentional discharge of either extinguisher bottle. These discs are located on the right aft fuselage under the #3 engine strut.

Electrical power for discharge of the engine fire extinguisher bottles comes from the battery bus. Engine fire detection power source is ESSENTIAL AC. Detector circuit ground fault lights monitor the integrity of the engine fire detection system on some aircraft. They are installed on the P-6 circuit breaker panel next to the fuel dump panel. If an electrical ground fault in any engine system occurs. Its light comes on. This provides warning of a potential malfunction in the engine fire detection system.

The APU fire detection system is identical to the systems used on the engines. The fire detection sensor is looped around the APU within its shroud. The electrical power required for activation of the APU fire detection circuit comes from the battery bus.

APU Fire Protection

The APU fire detection system provides cockpit and external visual and aural warnings. The external warnings consist of a horn located in the nose wheel well, which sounds intermittently, and a flashing red light below the APU control panel in the left wheel well. The cockpit warnings are illumination of the master FIRE WARN lights on the glareshield, if installed, the fire warning bell rings, and a steady red light in the fire shutoff handle on the engineer's auxiliary panel illuminates.

The fire warning bell can be silenced by pressing the bell cut-out button on the pilot's overhead panel; pressing either master FIRE WARN light on the glareshield; by selecting RESET on the APU test reset switch on the auxiliary panel; or by pressing the horn cut-out button on the exterior APU control panel in the left wheel well. Pressing any of these cutouts silences both the horn and the bell and the red light in the left wheel well changes from flashing to steady.

The APU fire extinguishing system includes a fire extinguisher battle charged with Freon, and the associated discharge line and circuits. The fire extinguisher-bottle is mounted behind the APU control panel in the left wheel well. When this bottle is properly charged. The pressure gauge should indicate 350 PSI at a bottle temperature of 70 degrees Fahrenheit.

The APU fire extinguishing circuit can be activated by pulling the fire shutoff



System Descriptions

handle and pressing the bottle discharge button on the APU cockpit control panel or by use of the fire shutoff handle and bottle discharge button on the exterior control panel. By pulling either APU fire shutoff handle, the following things occur: Arms the discharge button, Closes the fuel valve at the tank and the fuel solenoid valve on the fuel control unit, Closes the air load control valve, After a 5-9 second delay, trips the APU generator field relay, After either APU fire shutoff handle has been pulled, pressing the associated APU bottle discharge button releases the extinguishing agent into the discharge line. There is no visual indication that the extinguisher bottle has been discharged, as is the case with the engine system. The battery bus provides the electrical power required for the extinguisher circuit. In addition, the aircraft battery switch must be on in order for the APU to run. Without the battery switch in the on position, there is no fire protection or detection for the APU. It will shut down automatically if the switch is inadvertently turned off. A red thermal discharge disc is installed on the fuselage Just forward of the left wheel well. The disc ruptures if a thermal discharge of the fire extinguisher bottle occurs.

To conduct a fire test, the Fire Test switch is held in the TEST position. The fire bell should ring within 60 seconds. When the detector is heated to the alarm levels the red light in the APU fire shutoff handle comes on. the master FIRE WARN lights on the glareshield illuminate. The fire bell in the cockpit will ring and the horn in the nose wheel well will sound intermittently. And the red light on the APU panel in the left wheel well begins to flash.

There is no need to fire test an operating APU, however, the automatic fire shutdown circuit can be defeated so that this can be done. The auto fire shutdown switch controls this feature, which is labelled ARMED/OFF. With the switch in armed, the APU will shut down when the fire detection circuit is energized. When the switch is in the OFF position. A fire test may be made while the APU is operating.

Since the fire test system energizes the fire detection systems it must be reset after each fire test. This is done by momentarily holding the fire test switch in the reset position after the fire warning light has gone out. Remember, reset must be selected twice. Once to silence the bell and once to reset the auto fire shutdown circuitry.

Wheel Well Detection

The overheat detection system for the wheel wells provides detection only. There is no extinguishing equipment installed in the wheel wells. Fire detection elements are installed in the top of each of the three wheel wells. When the detector is activated by an overheat condition, both visual and aural warnings are given. The master FIRE WARN lights on the glare shield and the wheel well fire warning light on the pilot's overhead panel illuminate. and the fire warning bell rings. The bell can be silenced by use of any of the bell cut outs used for engine or APU fire warnings. The wheel well light remains on until the overheat condition no longer exists. The electrical power source for the wheel well fire detection circuit is essential AC.



System Descriptions

To test the fire detection system.

A fire test switch is located on the pilot's overhead panel or Glareshield (locations vary). The switch is spring loaded to the centre. The sensitivity of the circuits is such that it should take no more than 60 seconds to activate all three of the fire warnings. Holding the switch in the up position tests the detection circuits for the firewalls. Holding the switch in the down position tests the engine sensors and the wheel well detection loops. The wheel well warning is activated immediately upon initiation of the test. There is a delay for the engine and firewall warnings.

Fire extinguishing devices are located in the lavatories. Heat sensitive plugs automatically release the extinguishing agent to flood the water heater or towel dispenser if a fire should occur. A temperature sensitive indicator records temperature in the critical areas. The appropriate circle will blacken to indicate temperature reached. There is no visual or aural warning if this system in the toilet is activated.

FLIGHT CONTROLS

The 727 wings have 28 Deg sweepback and many devices that affect its aerodynamics. All flight control surfaces are normally powered hydraulically except for the stabilizer, which is trimmed electrically. Switches on the overhead panel control system "A" and system "B" hydraulic pressure to the ailerons, elevators, rudders, and flight spoilers. Normally they are guarded ON. Moving these switches OFF shuts off pressure to the associated control units. The system "A" rudder switch is ganged with the ON-OFF switch for the standby hydraulic system pump motor. Should hydraulic pressure to the primary flight controls drop below an acceptable level the appropriate amber light on the annunciator on the First Officer's forward instrument panel will come on.

ROLL CONTROL

There are two sets of ailerons on the 727; they are designated inboard and outboard. A hydraulic power unit located in the left main wheel well operates the ailerons. This unit is supplied hydraulic pressure from both systems "A" and "B". Either system will provide sufficient power to displace the ailerons through their full travel. When the control wheel is turned, the hydraulic power unit operates the ailerons through a cable system. When the trailing edge flaps are up, the outboard ailerons are held in a faired position by a lockout mechanism. When the flaps are extended, the lockout mechanism allows outboard aileron movement. The amount of outboard aileron movement, in relation to inboard aileron movement, is dependant on the degree of outboard trailing edge flap extension. Full outboard aileron response is available before the flaps reach normal landing configuration.

Aerodynamic balance tabs and balance panels assist in the operation of the



System Descriptions

ailerons. In powered operation, the tabs on all ailerons move opposite to the direction of aileron movement to assist in positioning the aileron aerodynamically. If all hydraulic power to the aileron is lost the system will automatically shift to manually operate aerodynamic tab control. That is the inboard aileron tabs operate as control tabs.

Now when the control wheel is turned, the aileron cable moves the inboard aileron tabs and the inboard ailerons are repositioned through aerodynamic action. Movement of the inboard ailerons is transmitted to the outboard ailerons through the normal cable system. If the flaps are retracted, the outboard ailerons are locked out as in powered operation. During manual operation the outboard aileron tabs do not shift to a control function, but move in response to the movement of the aileron.

Aerodynamic balance panels are located in the wing structure attached to the leading edge of the aileron and the aft wing spar. Differential air pressure acting on these panels assists in aileron operation. This is particularly significant during manual flight control operation.

Artificial feel for aileron movement is provided by a spring-loaded roller and cam mechanism. This mechanism also centres the control wheel when control pressure is released. The aileron trim wheel is located on the aft end of the control stand. Movement of the trim wheel repositions the roller and cam mechanism, changing the control wheel forces. When trim is changed, the control wheel will have a new neutral feel position. The ailerons will not move in response to a trim input if there is no hydraulic pressure available, however, the control wheel will reposition. When the ailerons are operated in flight without hydraulic pressure. Aerodynamic forces provide feel, and the trim wheel is ineffective.

SPOILERS

There are seven spoiler panels on each wing. The two inboard panels are ground spoilers and can be extended only on the ground through the action of the speed brake lever. They are operated by hydraulic system "A" pressure. With the left main gear strut compressed, this linkage opens a hydraulic valve allowing pressure to go to the ground spoiler actuators when the speed brake lever is moved aft. Since these panels are used only on the ground to spoil lift after landing, they go to the full up position when actuated. There are no intermediate positions for the ground spoilers.

The remaining five panels on each wing are flight spoilers. The three inboard flight spoilers are powered by hydraulic system "B". The two outboard flight spoilers are powered by hydraulic system "A". Individual actuators operate the spoiler panels.

The flight spoilers, which operate in conjunction with the ailerons, provide the major portion of roll control by spoiling lift on the low wing. After a small movement of the control wheel, spoiler action is programmed in proportion to the further movement of the control wheel. A full roll control input would cause the flight spoilers of the low wing to rise to a maximum of approximately 25 degrees. The spoiler panels of the high wing would remain down.



System Descriptions

The flight spoiler panels are also used as in-flight speed brakes when extended by the speed brake lever. They extend increasingly as the speed brake lever is moved aft. Maximum extension of the flight spoilers when used as speed brakes is 45 degrees; however, the actuators will allow spoiler blow down at high speed. While in flight a warning horn will sound intermittently if speed brakes are extended with the wing flaps also extended.

When the speed brakes are extended, spoiler input is still received from control wheel movement; however, this input is influenced by the amount of speed brake extension. Although there is no operational restriction, care should be exercised in operating ailerons with partial speed brakes due to spoiler mixer inputs. Extreme roll rates can be experienced in this configuration. Two switches on the upper right of the flight control hydraulic power panel are OFF-ON switches for hydraulic power to the flight spoilers.

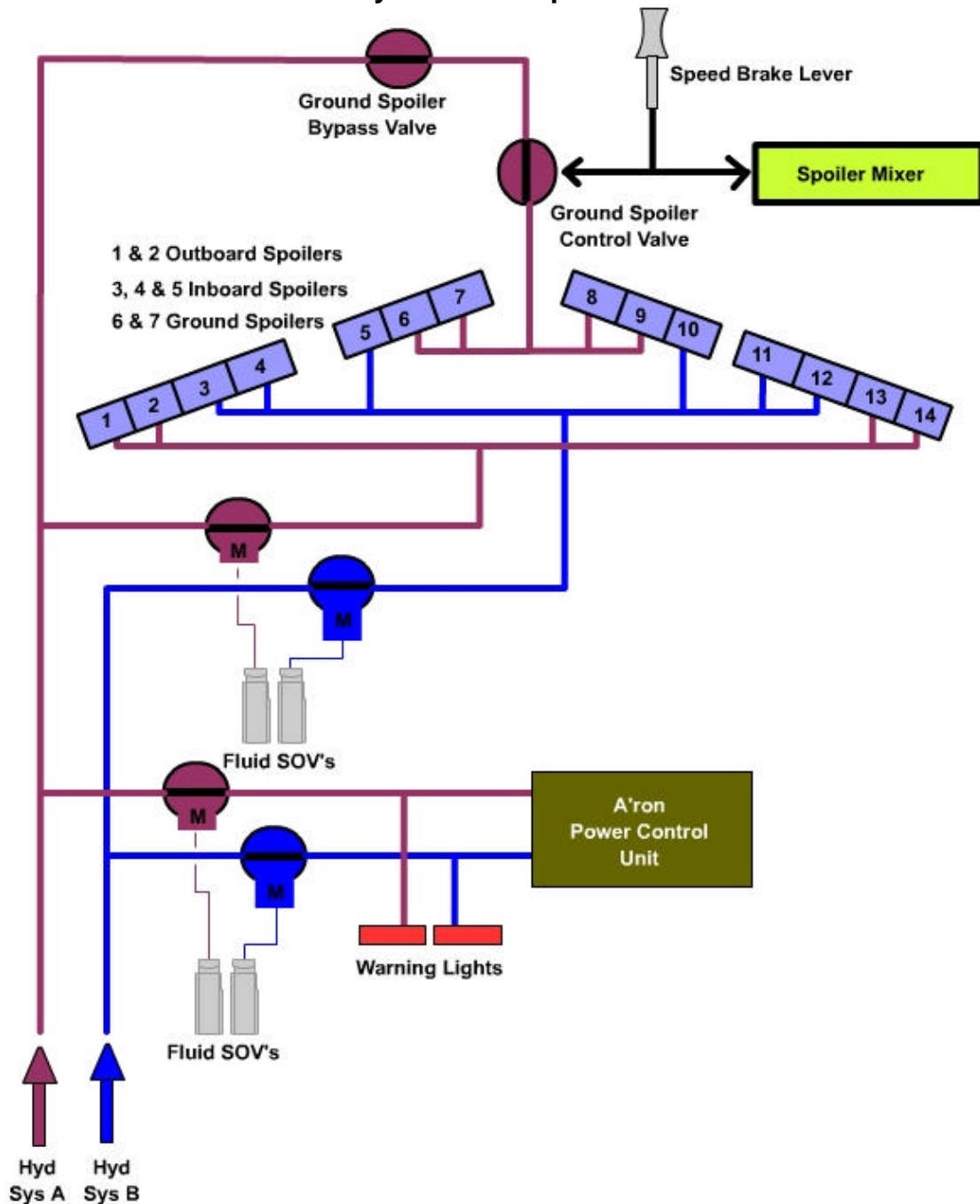
AUTO SPOILERS 200 series variants

An auto spoiler system is installed and extends the flight and ground spoiler's automatically after landing. To arm the auto spoiler system, the speed brake is moved to the arm position. The spoiler will extend when armed with the main wheel rotation above 60 kts, and strut compression has taken place. The speed brake armed light will illuminate to show that the auto speed brake circuitry is complete. If armed and a fault exists in the auto spoilers, the DO NOT ARM light will illuminate and the speed brake lever must be moved out of the armed position. The speed brakes may then be armed manually. The DO NOT ARM light will illuminate after landing with the speed below 60 kts, until the speed brake lever is restowed. When either No. 1 or No. 2 reverser is actuated during an aborted take off above 60 kts, the spoilers will automatically deploy. Auto spoilers will also deploy automatically if a landing is made without speed brake in arm when either No.1 or No.2 reverse throttle is actuated, with speed above 60 kts. If a go-around is initiated after landing, moving no.1 or no. 3 thrust lever forward automatically moves the speed brake handle forward to the down position.

To test the system, place the speed brake lever in arm and note the ARMED light illuminated, press each of the test buttons. When pressed the DO NOT ARM light will be illuminated.



System Descriptions



PITCH CONTROL

Pitch control of the 727 is accomplished through two independent elevators and a stabilizer. There are balance panels to assist in elevator movement. The elevators are powered jointly by systems "A" and "B" and will operate normally with either system alone. With loss of all hydraulic power, the elevator tabs act as control tabs, the same as the inboard aileron tabs. Full elevator movement by means of the control tabs is only about 50% of the movement available in powered operation. Actual position of the elevator with respect to the stabilizer can be observed by reference to the indices on the



System Descriptions

left and right sides of the rudder and elevator position indicator. Under normal flight conditions the elevator pointers should be centred.

To provide artificial feel, an elevator feel computer is installed. With inputs of system "A" and "B" hydraulic pressure, pitot-static pressure and stabilizer position the feel computer furnishes proportional feel to the elevator control system. The elevator feel computer does not supply boost to the elevator. When substantial differences in the computer pressure outputs occur, the elevator feel differential pressure light on the SID's panel will come on. It indicates a possible erroneous control feel. When the light is on the pilot should avoid abrupt elevator inputs. Spring tension and aerodynamic forces give feel when the elevators are operated with no hydraulic power.

Pitch trim is accomplished by repositioning the stabilizer. The stabilizer can be controlled by either of two electric motors or a manual system. Both of the electric motors and the manual system operate the same jackscrew, which drives the stabilizer. The high-speed electric trim motor is controlled by these main electric trim switches on the control wheels. Each main electric control consists of two thumb switches; one is for motor power, the other clutch power. Both have to be activated to move the stabilizer. This is a safety feature to prevent one faulty switch from causing a runaway stabilizer. The other trim motor is slow speed, and is controlled by the cruise trim switch on the control stand. The autopilot uses the slow speed motor for pitch trim. Manual trimming of the stabilizer is accomplished by using the cranks stored in the trim wheels located on either side of the control stand. The manual trim will override either electrical trim motor.

When the stabilizer is being trimmed electrically by either the main electric trim switch or cruise trim switches the stabilizer trim light on the control stand will come on. This indicates that one of the trim motor circuits is energized. When the autopilot is engaged, however, the cruise trim motor runs continuously. To avoid continuous illumination of the lights it is deactivated during autopilot operation. The two lever switches to the right of the light are cutout switches to remove electrical power from the motors.

Trim indices, located on both sides of the control stand, show the position of the stabilizer in units. The green band denotes limits of stabilizer trim in % MAC. If the stabilizer trim is not in the green band for takeoff, an intermittent horn will sound when the throttles are advanced toward takeoff setting. To prevent running the stabilizer to the stops there are electrical trim limits, both nose up and nose down. When the main trim reaches one half-degree nose up during nose down travel, control automatically switches to the cruise trim motor. Once the electrical limits are reached, the stabilizer can be trimmed slightly further manually. As the stabilizer is trimmed into the range near the nose up limit, the elevator neutral will gradually move a few degrees up from faired. This provides more effective stabilizer trim and more nose down elevator capability when the airplane is trimmed nose up. Operation of this mechanical linkage can be noted in the cockpit by aft movement of the control column when the stabilizer is trimmed into the range where the system is effective. To stop a runaway stabilizer a stabilizer brake is installed. A control force opposite to the direction of the runaway will engage the brake. Once



System Descriptions

engaged, the stabilizer brake should automatically disengage when opposing pressure is relieved. If it does not, pulling the stabilizer brake release knob will mechanically release the brake.

YAW CONTROL

Yaw control is accomplished through dual rudders and yaw dampers. The rudders are hydraulically powered, the upper rudder by system "B" pressure, the lower rudder by system "A" pressure. As a backup, the lower rudder can also be operated by the standby hydraulic system, which powers a separate rudder actuator. If all hydraulic power is lost there is no rudder control. Full system "A" pressure is provided to the lower rudder when the flaps are extended. When the flaps are up pressure to the lower rudder is decreased through action of the rudder load limiter. Illumination of the rudder load limiter light on the FE's panel indicates that the pressure to the lower rudder is not proper for the inboard flap position. The upper rudder always operates at reduced system "B" pressure. Therefore, the pressure changeover monitored by this light is applicable only to the lower rudder. Both rudders have anti-balance tabs, which move in the same direction as the rudder. Artificial feel for rudder inputs is provided through a spring-loaded roller and cam assembly.

The rudders are trimmed by positioning the rudder trim control on the control stand. The rudder pedals will reposition during trim input. As with the ailerons, rudder trim is available only when hydraulic power is available.

YAW DAMPERS

The tendency to "Dutch Roll", at high altitude and high airspeed, is counteracted in the 727 by a yaw damper system for each rudder. The lower rudder yaw damper does not function when that rudder is powered by the standby system. The yaw dampers receive electrical signals from the rate gyros. As the nose moves left or right, the rate gyro senses a yaw. This signal is sent to the yaw dampers, which direct the rudders opposite to the direction of the yaw. Rudder movement caused by the yaw dampers is not transmitted to the rudder pedals and does not interfere with pilot input to the rudders. A yaw damper test switch on the Captain's forward instrument panel allows testing of the yaw dampers before taxiing. The yaw dampers are designed to be used continuously. Therefore, the guarded yaw damper engage switches on the centre forward instrument panel are normally ON at all times. On the Captain's forward panel is a rudder-elevator indicator. The yellow YID flag will appear if the respective yaw damper system loses electrical power or is turned off. If one or both yaw dampers fail, airspeed and altitude restrictions are imposed. These restrictions are listed in the Limitation section of the Flight Manual.

[Find out more about yaw dampers by clicking here](#)



System Descriptions

To prevent airflow separation over the rudders at low speeds, vortex generators are installed at the vertical stabilizer. A maximum of two may be missing (not adjacent).

HIGH LIFT DEVICES

Combining both leading and trailing edge devices provides high lift capability in the 727. The trailing edge flaps are triple slotted Fowler flaps and are divided into two sets, inboards and outboards. Each set is bussed separately by torque tubes.

The flap handle on the control stand normally controls flaps. Flap positions are: UP 2, 5, 15, 20, 25, 30, and 40 with gates at the 2 and 25 positions. Though operational reasons sometimes prevents these all being used (certain stage three requirements). The position 2 gate prevents inadvertent retraction of the flaps prior to reaching air speed requirements and also acts as a reminder to check the leading edge annunciator before selecting another position. The position 25 gate establishes the lower end of the takeoff range. This gate also facilitates locating the go-around position without visual reference to the flap handle. Both gates require that the lever be lowered into the detent and passed under the gate before it can be moved further.

The flaps are normally operated by hydraulic motors each turning a torque tube for a set of flaps. One of these motors is located in each main wheel well. An alternate method of operating the flaps is by electric motors. One located in each wheel well. The same torque tubes and jack screw assemblies are used by either the hydraulic or electric motors.

Switches on the overhead panel control the inboard and outboard flap electric motors. With the Alternate Flaps Master switch ON, the inboard and outboard switches are armed. The switches must be held down for flap extension but will stay up for flap retraction. The Alternate Flap Master switch should not be turned ON unless airspeed is 245 knots or below. The inboard and outboard switches should be positioned off for a few seconds before reversing the direction of flap movement. Indicators on the centre instrument panel show actual flap position. Each flap indicator has dual pointers, one for each associated left and right flaps.

To prevent a roll control problem due to a differential in trailing edge flap extension, there are asymmetrical protection systems for the inboard pair and outboard pair of flaps. The indicators have an electric asymmetrical protection system based on separation of the left and right pointers on the flap indicators by about one quarter inch. This protection system stops flap travel by stopping flow of fluid to the hydraulic motor. There is no asymmetrical protection when using the alternate flap system other than monitoring the flap position indicators.

Leading edge devices, consisting of four slats and three flaps on each wing, are automatically sequenced to operate with the trailing edge flaps. The movement of the outboard trailing edge flaps positions the control valve for the leading edge device actuators. At the position 2, the two middle slats on each wing will extend. When the trailing edge flaps are extended to position 5, all of the leading edge devices extend. Once extended the actuators are



System Descriptions

mechanically locked in the extended position and must be unlocked hydraulically to be retracted. The three inboard devices, or flaps, are held in the retracted position with the system "A" pressure. The remaining devices, the slats, are mechanically locked in the retracted position and must be unlocked hydraulically to be extended. Each leading edge device has its own actuator. The leading edge annunciator on the forward instrument panel will be amber while the leading edge devices are in transit or not properly positioned for trailing edge flap configuration. When the correct leading edge devices are extended, in agreement with trailing edge flaps, the green light comes on. Both lights will be off when the leading edge devices are up. The leading edge device annunciator on the FE's auxiliary panel has a 3 position selector, spring loaded to OFF. Holding the selector to the right will test all of the lights. To the left, the actual position of each individual leading edge device is indicated, that is, with no light the device is retracted; amber, the device is partially extended; or green, the device is fully extended. This annunciator is used when the one on the forward instrument panel remains amber after flap operation, indicating improper leading edge device positioning.

An alternate means of extending the leading edge devices is provided. With the Alternate Flaps Master switch ON, momentarily moving either of the alternate flaps inboard or outboard switches to DOWN will extend all leading edge devices.

The standby hydraulic system powers a hydraulic motor-pump assembly, which pressurizes the leading edge device actuators to the down position with fluid from a reserve section of the system "B" reservoir. The devices cannot be retracted by the alternate system.

Under certain conditions a warning horn will sound if high lift devices, speed brakes, or the stabilizer are improperly positioned. On the ground, the warning horn will sound intermittently when the no. 1 or 3 throttle is advanced to takeoff thrust if the stabilizer trim is not in the green band, the flaps are not in the takeoff range, the speed brake lever is not in the forward detent, or the leading edge devices are not extended.

The trailing edge flaps control various other components on the airplane.

Outboard Trailing Edge Flaps	Inboard Trailing Edge Flaps
Leading edge devices	Stall warning systems
Outboard ailerons	Air conditioning pack fan
Takeoff warning horn	Auto pack trip system.
Landing gear-flaps warning horn	Lower rudder load limiter
Ground proximity warning system.	Speed brake - flaps warning

Most commercial swept-wing airplanes have a device to warn of an approaching stall. As an advanced warning if an approach to stall occurs, the computer activates a stick shaker at 1.15 times stall speed. The stall-warning portion of the overhead panel contains the fail light and test/heater off switch for the Stall Warning unit in the 727. There is one unit installed on the airplane, which consists of a computer, an angle of attack vane and a stick



System Descriptions

shaker. The angle of attack vane, located on the right side of the airplane, sends angle of attack information to the stall-warning computer. The computer also receives information of actual inboard trailing edge flap position. The computer is powered and the vane is heated any time an airplane generator is powering the electrical bus, provided external power is disconnected. When the vane heater and computer are both powered, the power off lights will be off. To avoid stall-warning signals while taxiing, the computer, although powered, is not activated until the airplane is airborne.

FUEL SYSTEM

General

There are various options of auxiliary fuel tank installations, which are located in the fuselage, they will not be discussed here.

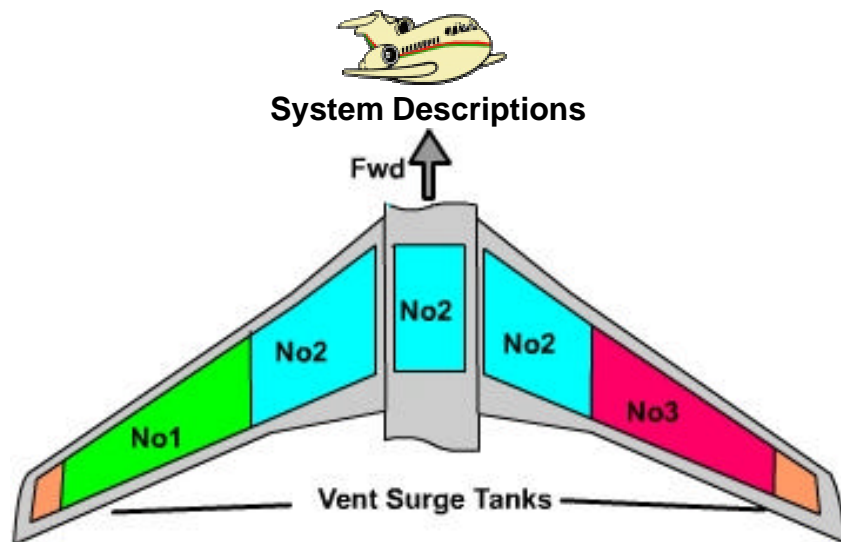
Fuel is stored within vented areas of the wing and wing centre section. These fuel storage areas are divided into three main tanks, one for each engine. The tanks are located in the interspar area of each wing. They are identified as Tank1, Tank2 and Tank3. Normally each tank supplies its respective engine, which is referred to as "tank to engine". Tank No1 is contained entirely in the left wing, Tank No3 entirely in the right wing. Tank No2 consists of an integral portion in each wing and removable cells in the wing centre section.

Low points on all the integral tanks are fitted with fuel sump drain valves to permit draining of accumulated water from the tanks and for draining fuel when the tank is defueled.

Tanks can be fuelled from the pressure refuelling station at the right wing, via this system any or all of the tanks can be filled rapidly. Tanks may also be defueled through this pressure system. Overwing filler points are provided for tanks 1 and 3. Fuel may be transferred from tanks 1 and 3 into tank 2.

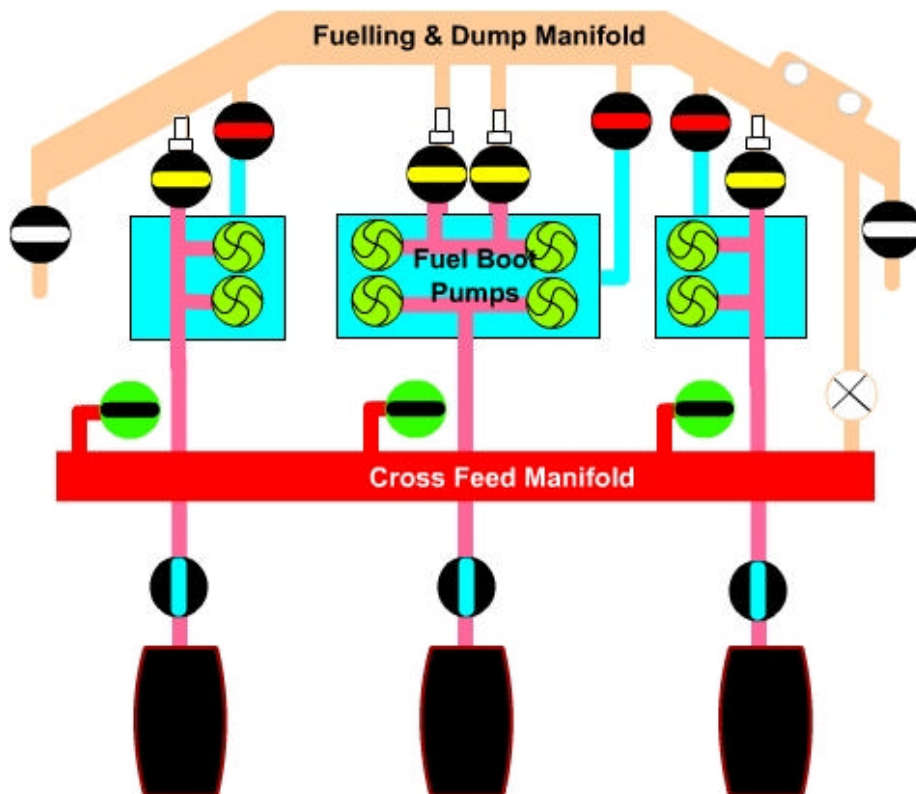
Transfer of fuel from one tank to another is possible on the ground only, through a manually operated defuel valve located at the lower right hand wing root. When overwing refuelling is taking place it's normal for some operators to shut down the APU.

An electronic capacitance type indicating system provides fuel quantity indications on the flight engineers panel and at the refuel station. Vent surge tanks are provided to accommodate fuel surges, any fuel in the vent surge tanks drains directly into the adjacent No1 or No3 tanks. Tanks are vented through the surge tanks to a single opening at each wing tip. The vent system also provides ram air pressure within the tanks. The centre tank cavity is vented and drained overboard through a separate system of vent and drain lines.



Fuel Boost Pumps

Eight AC motor driven fuel boost pumps are located in the fuel tanks, two per tank in one and three and four boost pumps in tank 2. AC power to the boost pumps is distributed so that failure of any single AC bus will not cause complete loss of all boost pumps in any one tank. A fuel boost pump bypass valve allows engine driven fuel pumps to draw fuel from the main tanks if all boost pumps in a tank are inoperative.



Valves with a red band are fuelling valves, yellow band are dump valves, black band cross feed valves, white band nozzle valves, beige cross manual defuel valve, white circles refuel station, Black lumps engines.



System Descriptions

Fuel Heaters

A fuel heating system is provided to prevent the formation of ice crystals in the fuel being delivered to each engine. Fuel flowing to the engine is passed through an air to fuel heat exchanger just prior to entering the fuel filter. Heating is accomplished by ducting 13th stage engine compressor air through the heat exchanger. After passing through the heat exchange, the air is dumped into the CSD oil cooler exhaust air duct. Control for the system is via switches on the flight engineers panel. Valve operation is confirmed by Blue in transit lights. When operating you will see a rise in oil temperature to the engine receiving fuel heat. Icing lights are provided and are also located on the same panel as the control switches. The icing light will come on if a differential pressure exists across the main fuel filter. The difference in pressure is assumed to be caused by ice crystals partially blocking the filter. If the icing light does not go out with the application of fuel heat, the filter may be clogged.

Dripsticks

A calibrated dripstick is installed in each tank. These can be used to determine the fuel level in each tank in the normal taxi attitude. When the locking sealing cap and the hollow dripstick assembly is withdrawn from the wing lower surface, fuel enters the open top of the stick and flows out through a drip hole near the base (indicated by an arrow on the surface). The dripstick should be pushed up slowly until fuel stops and then inched down again until you obtain a steady drip/flow of fuel. Reading the calibrations on the stick from the top down, the calibrated reading plane is the inner surface of the recess. Calibrations maybe in inches, pounds or Kgs.

They are located on the tank centre lines. No1 and No3 are near the inboard end of the respective tank. No2 is near the inboard end of the integral tank on the right side. Additional dripstick may be installed. The normal minimum stickable fuel value is 938 lbs.

HYDRAULICS

"A" System

System consists of a single reservoir. Hydraulic power is supplied to the A system from two engine driven pumps fitted to engines No1 and No2. The pump switches control solenoid operated blocking valves, with the switches on, normal pump output is supplied to the system. A small portion of the fluid that enters the pump is circulated through the pump case for cooling and lubrication. As long as the engines are operating regardless of the blocking valve position this small amount of fluid leaves the pump and is cooled by a heat exchanger in the No3 fuel tank heat exchanger before being returned to the system.

Fluid shutoff valves controlled by their respective switches control the fluid in

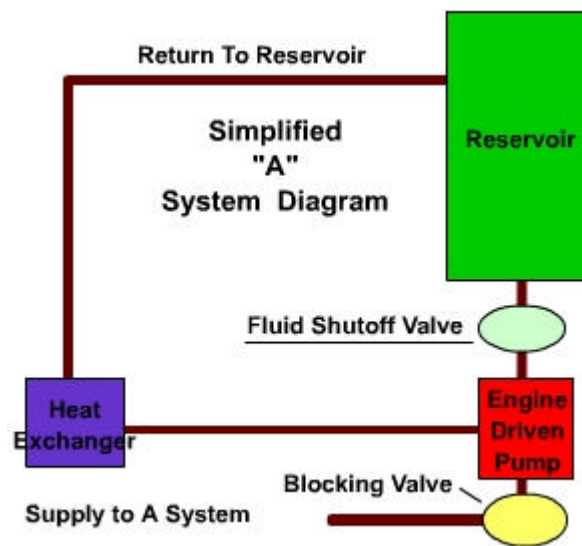


System Descriptions

the engine pump supply lines, when closed fluid flow to the pump is isolated. (Includes lubricating and cooling fluid). Pulling the fire switch on engine 1 or 2 will close these shutoff valves; the fire switch also deactivates the low-pressure warning light.

System A and therefore system B and stand by reservoirs are pressurized by engine bleed air from engines 1 & 2 this is applied to the reservoir to ensure a positive supply of fluid to the pumps.

A balance line connects the system A reservoir to the system B reservoir at a level of 2.5 gall.



A system supplies the heavy load items landing gear, trailing edge flaps.

Fluid Quantity.....4.4 Gall (gear down)

.....3.8 Gall (gear up)

™it's normal to see a reduction in the fluid quantity indication in flight, due to the cold soak characteristics of the system

System Pressure.....2,800 - 3100 PSI (red line @ 3,500)

Balance line to "B" system.....2.5 Gall

Operating time with the "A" fluid SOV closed 5 Min's

™ A system pumps max demand flow rate is 22 gal min 100% N2

Systems Operated:

Nose Wheel Steering

Landing Gear

Tail Skid

Trailing Edge Flaps

Ground Spoilers

Leading Edge Devices

Lower Rudder

Ailerons



System Descriptions

Elevators
Outboard Spoilers

"B" System

"B" system again contains a single reservoir supplying two electric AC pumps. It's reservoir is more complicated than system "A" in that it has an auxiliary reservoir separated by a baffle which is open at the top ("little b"), this is reserved for alternate extension of the leading edge slats and flaps using stand by hydraulic power (more about that later).

Each pump is controlled by its individual selection switch; with the switch ON normal hydraulic output pressure is provided to the system. As in the "A" system a portion of the fluid is used for cooling and lubrication of the pump but is returned via a heat exchanger in the no1 fuel tank. If the pressure falls below the preset level a warning light will illuminate, sensors monitor the fluid temperature and if activated will bring on an overheat light. A relay in the overheat sensing circuit will cause the overheat light to go out when the affected pump is turned off. A simplified view of the system is the same as the above diagram but with no blocking valve or shut off valve (items 2 & 4).

Fluid Quantity.....Full if system "A" quantity over 2.5 gall

System Pressure.....2,800 - 3100 psi (red line @ 3,500)

Baffle level (little B).....1.1 gall

Do not operate a system "B" pump with less than 1,300 lbs in fuel tank No1
Any one hydraulic system "B" pump should not be operated more than 5 times in 5 min's. After this period the pump must be left on for 5 min's or off for 30 min's.

Systems Operated:

Brakes
Inboard Spoilers
Upper Rudder
Aft Airstairs
Ailerons
Elevators

Stand-By System

The standby system reservoir supplies fluid to an AC electric motor driven pump. Positioning the standby rudder switch or the alternate flap master to on will provide power to the standby motor, which in turn supplies fluid pressure to the standby power module. System pressure is indicated by a light. There is an overheat sensing to monitor fluid temperature which will illuminate the overheat light for the standby system.

Fluid Quantity.....0.3 gall

System pressure..... 2,200 psi @ 1.14 gall per min for leading edge devices (there is no pressure gauge for the stand-by system only a green light.

..... 2,975 to 3,075 @ 3 gll per min for lower rudder

Systems Operated:



System Descriptions

Lower Rudder Stand-By Actuator
Leading Edge Devices (Extension Only)

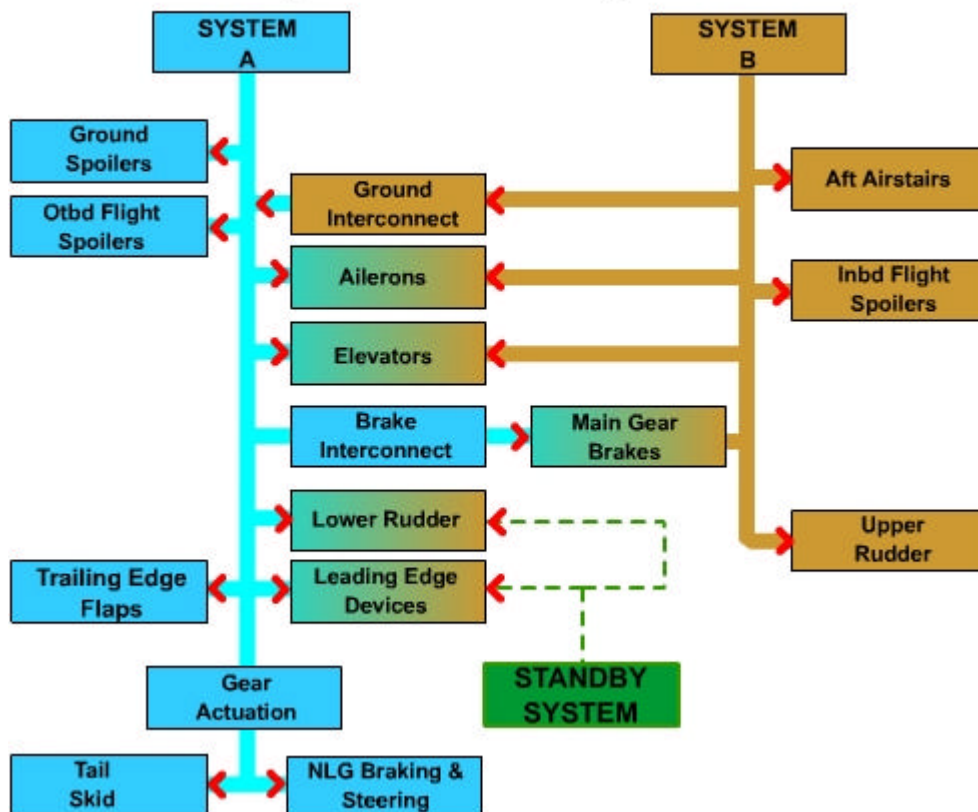
Standby Rudder

Positioning the standby rudder switch to on, the system "A" rudder power switch is selected off. With "A" system off hydraulic fluid is isolated from the lower rudder power unit, standby system pressure repositions a selector valve to allow the standby system to power the lower rudder standby power control unit.

Alternate flap

Positioning the alternate flaps master switch on arms the alternate flap switches (2). Moving either switch to down opens the leading edge flaps and slats sov. The standby system pressure then drives a hydraulic motor portion of a pump assembly. The pump portion of this assembly uses fluid from "little b" to extend the leading edge devices.

Hydraulic Power Systems





System Descriptions

GPWS

The Ground Proximity Warning System alerts the flight crew when one of the following thresholds are exceeded between 50 and 2450 feet radio altitude.

- Mode 1 - Excessive Decent Rate.
- Mode 2 - Excessive Terrain Closure Rate.
- Mode 3 - Altitude Loss After Take - Off or Go Around.
- Mode 4 - Unsafe Terrain Clearance During High Speed Flight or While Not in the Landing Configuration.
- Mode 5 - Below Glideslope Deviation Alert.

Inputs to the ground prox computer are radio altitude from the No1 radio altimeter, barometric altitude rate and mach from an air data computer; glide slope deviation signals from capt side, landing gear and flap positions.

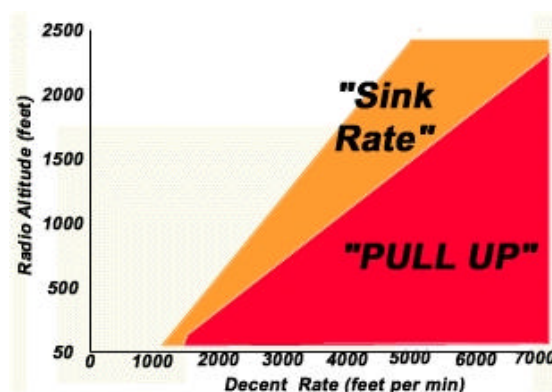
The loss of one of these inputs will deactivate only the affected mode or modes.

Aural alerts and warnings for modes 1 through 4 are accompanied by red PULL UP lights

Modes Explained (Note that these diagrams are simplified)

Mode 1 - Excessive Decent Rate

Has two boundaries and is independent of airplane configuration. Penetration of the first boundary generates the repeated aural alert of SINK RATE. Penetrating the second boundary causes the repeated aural warning of WOOP WOOP PULL UP, until the rte of decent has been corrected.



Mode 1 Envelope

Mode 2 - Excessive Terrain Closure Rate

Monitors Mach number, radio altitude and radio rate of change, barometric altitude and airplane configuration. Has two boundaries, the first causes an



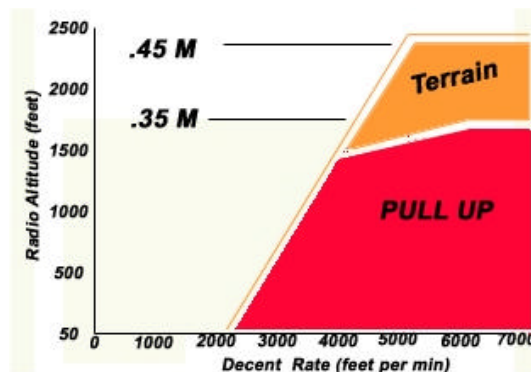
System Descriptions

aural alert of TERRAIN repeated twice, followed by the repeated aural warning of WOOP WOOP PULL UP. After leaving the PULL UP area the repeating TERRAIN message will be heard while in that portion of the envelope.

If both boundaries are penetrated while in the landing configuration, only the repeating TERRAIN aural alert will occur.

If the landing gear and wing flaps are not in the landing position when leaving the PULL UP area, 300 ft of barometric altitude must be gained before the aural TERRAIN alert is silenced.

As the Mach number increases from .35 to .45 with the gear up, the highest altitude at which a mode two alert warning will occur is increased to 2450 ft. This higher portion is inhibited with the flap inhibit switch in the FLAP INHIBIT position.

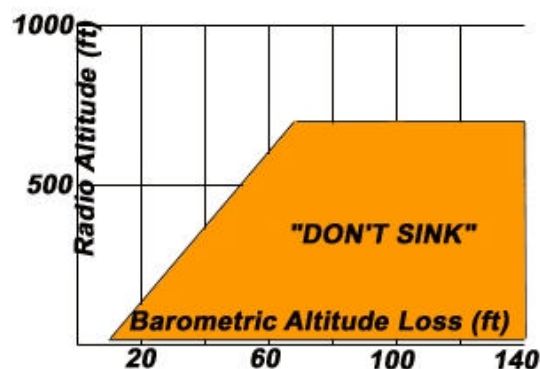


Mode 2 Envelope

Mode - 3 Altitude Loss After Take off Or Go Around

Provides an alert if a decent is made during initial climb or go around. The aural alert is a voice message "DON'T SINK", repeated until the condition is corrected. It is effective between 50 and 700 feet radio altitude and generates the alert when accumulated barometric loss equals approx. 10% of the existing radio altitude.

Mode 3 does not arm during decent until below 200 ft radio altitude.



Mode 3 Envelope

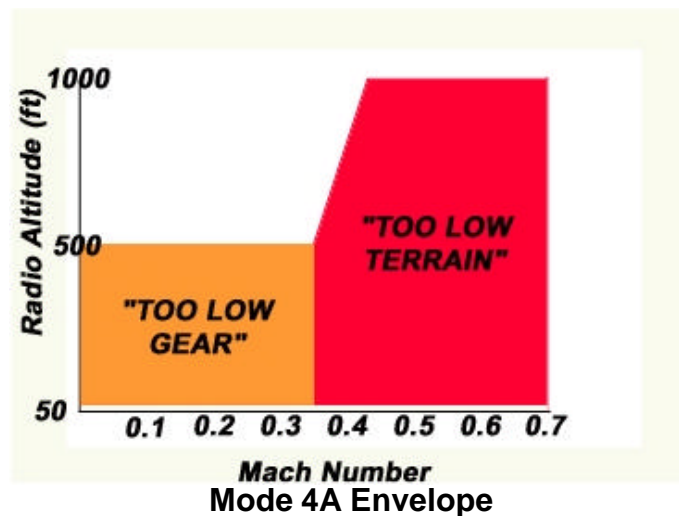
Mode - 4 Two modes 4A and 4B



System Descriptions

4A - Unsafe Terrain Clearance With Landing Gear Not Down

This is the terrain clearance mode with the gear retracted, is armed after take off upon climbing through 700 ft radio altitude. When this envelope is penetrated at less than 0.35 mach the aural alert TOO LOW, GEAR is sounded. When the envelope is penetrated at more than 0.35 mach, the aural alert TOO LOW TERRAIN is sounded and the upper boundary of the envelope is increased to 1000 ft rad alt. The message is repeated until the flight condition has been corrected.



MACH -IAS Conversion

MACH	SL	5000'	8000'	10000'
0.34	232	211	200	193
0.45	298	272	258	249

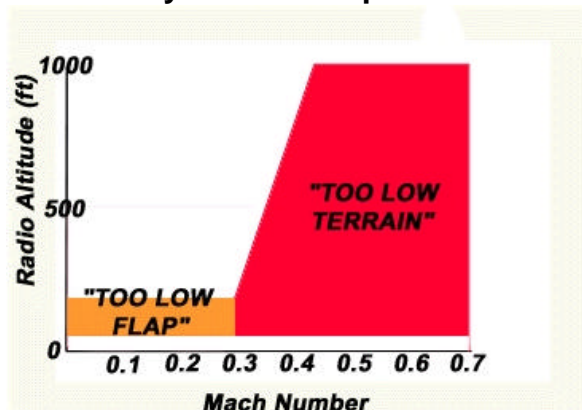
4B - Unsafe Terrain Clearance With The Flaps Not In Landing Position

This mode provides an alert when the gear is down and the flaps are not in landing position. If the envelope is penetrated at less than 0.28 mach with the flaps not in the landing position, the aural alert TOO LOW FLAPS is sounded. When the envelope is penetrated at more than 0.28 mach, the aural alert of TOO LOW TERRAIN is sounded and the upper boundary is increased to 1000 ft rad alt. The voice messages continue to occur until the flight condition has been corrected.

The TOO LOW GEAR alert takes priority over TOO LOW FLAPS. The TOO LOW FLAPS and associated TOO LOW TERRAIN alert are inhibited with the flap inhibit switch when moved to the FLAP INHIBIT position



System Descriptions



Mode 4B Envelope

MACH -IAS Conversion

MACH	SL	5000'	8000'	10000'
0.34	185	169	160	154
0.45	298	272	258	249

Mode 5 - Below Glideslope Deviation Alert

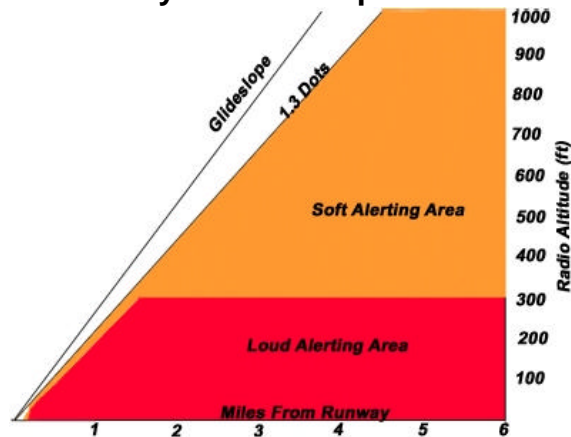
This mode alerts you of a descent of more than 1.3 dots below an ILS glideslope. The envelope has two areas of alerting, soft and loud. In both areas is a repeated aural warning of GLIDESLOPE and illumination of the pilots BELOW G/S lights. The voice message amplitude is increased when entering the loud area. In both areas, the aural warning repetition rate is increased as glideslope deviation increases and radio altitude decreases.

The mode is armed when a valid signal is being received by the capt's glideslope receiver and the radio altitude is 1000 feet or less. This mode may be cancelled or inhibited by pressing either pilot's BELOW G/S light while below 1000 ft radio altitude. The mode will re arm when climbing above 1000 ft rad alt.

Mode 1 through 4 aural alerts and warnings have priority over mode 5 aural alerts, though both PULL UP and BELOW G/S lights could be illuminated at the same time.



System Descriptions



Mode 5 Envelope

ICE & RAIN PROTECTION

Ice protection is the prevention and removal of ice accumulation (anti-icing and de icing respectively) The pneumatic and electrical systems supply the required heat.

Engine bleed hot air for:

Wing anti-icing

Engine nose cowls and inlets and centre engine inlet duct

The upper VHF antenna

Fuel filter de-icing (more under power plant)

Electrical power provides heat for:

Pitot tubes

Static ports

Temperature probe

Cockpit windows

Stall warning heater

Lavatory and galley drains

WING ANTI-ICING

For wing anti-icing, bleed air from engines 1 and 3 flows through ducts in the leading edges of the wings and is discharged overboard.

Air is passed through the leading edge slats, leading edge flaps (2 thru 5), fixed inboard wings above the leading edge flaps and the upper VHF antenna this is a mix of high and low pressure air. There is a automatic trip off system installed as a safety measure against a ruptured wing anti-ice duct in the pressurized area of the fuselage which would result in a rapid pressurization of the cabin (increase). If this occurs the wing anti-ice valves close automatically and the WING ANTI-ICE AUTO TRIP OFF light illuminates.



System Descriptions

(Located on the flight engineers panel) There is a test switch installed to allow a check of the system before flight. Overheat protection is provided by temperature sensors located in the wing anti-ice ducting. When it gets too hot the DUCT OVERHEAT light illuminates. It's value is different depending whether you are in flight or on the ground. If you are on the ground is setting is less, and the anti-ice valves will close. In flight it has a higher value and the valves do not close. You can verify and locate the source of the overheat by looking at the DUCT TEMP indicator and using the anti-ice temp selector at the co pilots overhead panel. The wing anti-ice shutoff valves are overridden closed by pulling the fire handle. Valves are electrically operated.

ENGINE ANTHICING

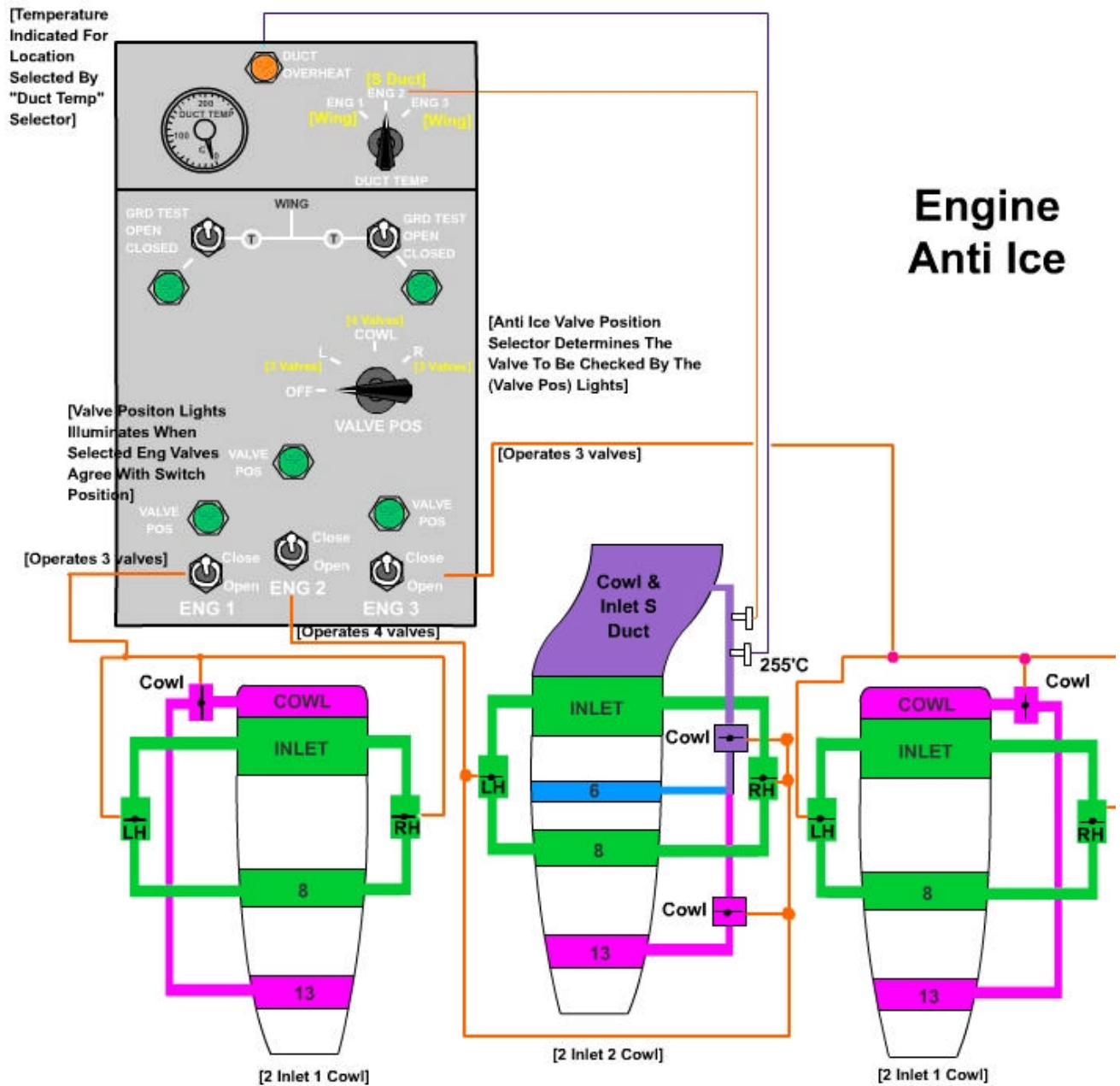
Each engine supplies it's own anti-icing air and is separate and independent of the other two engine anti-ice systems. The engine nose cone, EPR port and inlet guide vanes are anti-iced by engine low-pressure bleed air. The air is ducted through the left and right engine anti-ice valves, to the inlet guide vanes and nose cone, and vented into the engine intake. When operating engine ant-ice you will see a drop in engine EPR.

POD ENGINES. Nose cowls and CSD oil cooler scoops are anti-iced by engine high pressure bleed air ducted through the nose cowl anti-ice valve and mixed with ambient air. The mixed air is directed against the cowl and CSD scoop leading edges, exhausted overboard the a opening in the bottom of the engine cowl.

CENTER ENGINE. The inlet leading edge, part of the surface of the inlet duct, and the CSD oil cooler inlet are anti-iced by a combination of high and low-pressure bleed air. Air is vented into the rear fuselage section and exhausted overboard through an opening in the left hand side. Bleed air is also used to anti-ice the vortex generators installed on the bottom of the inlet duct, this is vented into the engine intake.



System Descriptions



Engine Anti Ice

COMPONENTS OF INTEREST

Thermostatic Modulating Valves.

Control the volume of flow in relation to the temperature of the air, Higher the temperature the lower the flow.

Left Hand and Right Hand Engine Anti-Ice Valves.

Left and right anti-ice valves on each engine control the bleed airflow through the guide vanes to the nose cone, which also contains the EPR port. Valves are electrically operated.

Nose Cowl Anti-Ice Valves.

Control high-pressure bleed airflow to the pod engine nose cowls and centre engine inlet duct. On the No2 engine, low pressure bleed air is also used for inlet duct anti-icing. A mixed air shutoff valve controls the flow of both high

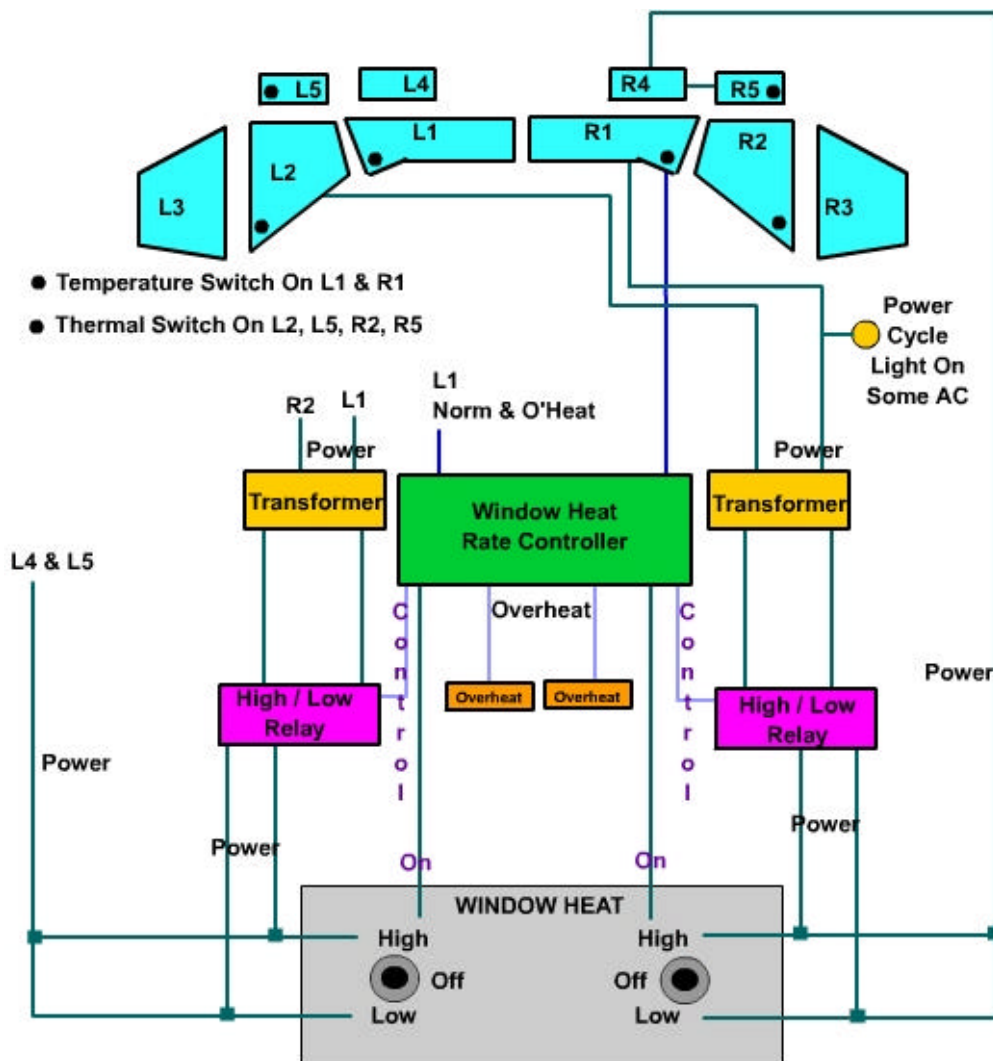


System Descriptions

and low pressure bleed air to the inlet duct. The No 2 engine mixed air shutoff valve is overridden closed when the engine fire handle is pulled, it is electrically operated.

Window Heat

Cockpit windows, except No3 are electrically heated to provide anti-icing, defogging and impact resistance. Power to a electrically conductive coating within the laminated window is controlled by switches on the overhead panel above the co pilot Window heat is regulated by, The window heat controllers, Temperature sensors and thermal switches.



Window Overheat Protection

An overheating window will not cause damage during flight. Due to the cooling airflow however when this is lost after landing damage could quickly occur if it was not for the protection system. At first overheating appears as small bubbles (like water droplets) this does not affect the structural integrity, but may cause problems with visibility. If the overheat condition persisted the window would splinter and crack.



System Descriptions

™ Don't assume the system overheat protection is working as I've had a window fail.

Windshield Anti-Icing. Anti-icing is provided for the No1 and No2 windows, control is through two switches on the control panel. Marked High, Low, Off these select high or low voltage to control the rate of heating. If an overheat condition is detected a window overheat light will illuminate of the flight engineers door panel. It locks out the control and removes power from the respective No1 and opposite No2. It may be reset by cycling the switch to off.

Window Defogging No 4 and No5

These windows receive heat for defogging only and are powered when the switch is in the high or low position; they have no controller or overheat protection. Control of the heating is by a thermal switch on the No5 Window. Note the No3 window is not heated.

Pitot - Static Heat

The left and right pitot probes, auxiliary pitot probe, left and right static ports, total air temperature probe (TAT) and the left and right elevator feel pitot probes are heated to prevent ice formation which would affect sensing accuracy. The heat is controlled by two switches on the co pilot's overhead panel. It supplies 115V AC, When the left switch is placed on heating is applied to:

Captains pitot probe

Left static ports

TAT probe

Left elevator feel pitot probe

The right switch controls in the same manner as the left but operates

Co pilot pitot probe

Right static ports

Auxiliary pitot probe

Right elevator feel pitot probe

Static port heating has been deactivated on some 100's but is required on the 200.

Stall Warning Sensor Heat

The attitude sensor of the stall warning system is anti-iced by an integral electric heater. it is available when the airplane is on engine generators. When external or APU power is on the sync bus the heater is off except when checked by the test switch.

Rain Protection

Consists of electrically operated windscreen wipers operated by a switch on the overhead panel, these are install on both the No1 windows. The switch has the following positions Park, Off, Low, 1/2, 3/4 and High. There is a rain repellent system also for the No1 windows, which is controlled by two switches, two time control valves and a fluid container. When a switch is pressed it opens a valve letting a fixed amount onto the windscreen on



System Descriptions

application takes approx 1/2 second. The system is little used and may be removed; there is an SB on this.

LANDING GEAR & BRAKES

LANDING GEAR

The landing gear consists of two dual wheel main gear and one dual nose gear, each main gear is equipped with Disk brakes, anti skid protection and thermal tire deflators (fusible plugs).

The landing gear is positioned hydraulically as selected by the landing gear lever in the cockpit on the centre instrument panel.

Door and gear sequencing is automatic. Except for the nose gear, which is mechanically opened and closed by the movement of the gear. There is a door release handle in each main gear well for ground access.

Hydraulic system "A" provides power for the landing gear system and nose wheel brakes if installed. "B" system provides power for the main wheel brakes. System "A" can also be used as an alternate power source by selecting open the brake interconnect switch on the flight engineers panel.

Extension and Retraction

Gear Doors. Each gear is sequenced automatically with its gear door, opening of the door is controlled by the gear lever. The main gear cannot extend or retract unless the gear door is open and cannot close unless the gear is locked in the up or down position all due to sequence valves being installed.

The nose gear is controlled mechanically by linkages to the gear. The forward doors are closed in both the gear up and down positions but the aft doors remain open when the gear is down.

Gear Air-Ground Logic. Air ground sensing for various systems is provided by safety switches on the left main gear and nose gear. These are actuated by the extension (air logic) or compression (ground logic) of the left main gear and nose gear. [Click here to see some of the system inputs for logic](#)

Nose Gear Steering

The nose wheels can be turned by a steering handle to the left of the captain (some aircraft have two handles one on each side), or by either set of rudder pedals if the nose gear strut is compressed. Internal cams in the strut automatically centre the nose wheels if the strut is extended. Power for the nose wheel steering is supplied by hydraulic system "A" through the landing gear down line. Steering wheel movement of 95 deg at the handle will produce 78 deg of nose wheel turning. Full rudder deflection will give you about 8 deg of nose wheel turn. If steering hydraulic pressure is lost and the steering control valve is in neutral, restrictions in the hydraulic circuit prevent nose gear castering. Movement of the handle or rudder pedals will displace the control valve and allow castering.



System Descriptions

Manual Gear Extension

In the event of a "A" system hydraulic failure the landing gear can be extended and locked mechanically (manual). Three hand crank drums located in the cockpit floor can be operated. The hand crank lever is stowed on the aft left bulkhead of the cockpit. Instructions for operating are placarded on the back of the access panel. Operation of the system for the main gear unlocks the door first and then unlocks the gear which free falls. The gear doors will then remain open. For the nose gear it unlocks and free falls as the doors are mechanically link to the gear. Reversing the hand crank rotation after the gear has extended locks the gear in place.

Mechanical Lock Indicators

Viewing ports are provided to visually check that the landing gear is down and locked. These are located at the following locations (approx), Nose gear four feet aft of the cockpit door in the centre, main gear five feet aft of the over wing exits to the left and right of the centreline. The visual lock indicator is operated by the down lock linkage and is located near the top end of the side strut. For the nose gear a stripe on the actuating arms aligns with the lock housing. There are lights installed to allow viewing at night and it is controlled by the wheel well light switch on the pilots overhead panel.

Tailskid

If there is cases of over rotation on take off it's this that will first contact the runway. It is equipped with an energy absorber, which consists of a cylinder with a crushable honeycomb core in the upper half. The core is replaceable. An indicator clip is riveted to the strut and attached to a wire. When the clip is sheared off by compression of the tailskid, it will be retained by the wire and a red area beneath the clip will be exposed to indicate that the core has been crushed. Operation is by the electrical system and extends when the landing gear lever is in the down position and the outboard flaps have been lowered 15 deg or more. It retracts when the gear lever is placed in the up position. It has it's own warning light on the flight engineers door annunciator panel and comes on when there is a disagreement with the landing gear lever in the up or down positions.

BRAKES

Normal Operation. Self-adjusting, multidisc hydraulic brakes with incorporated brake wear indicators are installed at each main gear wheel. They are operated by the pilots brake pedals or the pneumatic brake handle (main gear only). An anti skid system is installed to maximize normal braking capability and prevent locked wheels. Pressure sources are available from hydraulic system "B", "A", brake accumulator and pneumatic pressure source, nose brake power is from system "A" (if installed). As previously mentioned system "A" may power the brakes via the brake interconnect valve. Check valves retain pressure in the brake system if hydraulic pressure is lost. A fully charged brake accumulator stores enough fluid under pressure for several brake applications.

There are brake pressure gauges in the cockpit and left hand wheel well this

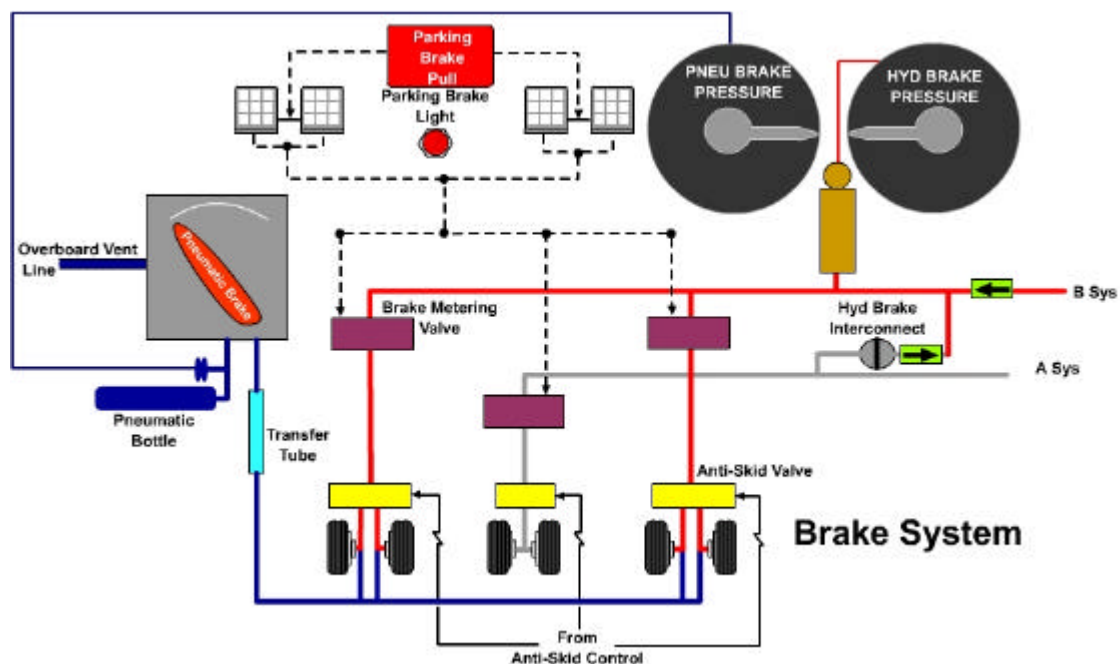


System Descriptions

will show hydraulic pressure and accumulator air pressure. Normal operating pressure is 3000 psi. Pressure surges trapped by the check valve may cause indications to rise to 3500 psi. When all the fluid pressure is depleted from the accumulator the indicator will read precharge pressure, about 1000 psi. Pilot control of the braking is through the brake pedals to the brake metering valves, one for each main gear. Stepping on the brake pedals actuates the respective metering valve. As the metering valve moves a proportional amount of hydraulic pressure is directed to the anti skid valves and lock out deboosters, then to the wheel brakes. The deboosters reduce hydraulic pressure and isolate the fluid downstream. If a leak occurs between the deboosters and the brake, only the isolated fluid is lost, and you won't have a system "B" hydraulic loss. A servicing handle on the deboosters replenishes the isolated fluid (part of pre flight).

Pneumatic Braking

The pneumatic braking system is an alternate system and is a way of providing pressure to main brakes in the event of hydraulic system failure. There is no anti skid or differential braking available from the pneumatic source. A pneumatic brake control valve operated by a handle on the captain's instrument panel opens and modulates air bottle pressure to a transfer tube. Pressurized hydraulic fluid from this tube is routed to a shuttle valve on each main wheel brake. The shuttle valve moves to block the hydraulic pressure port of the main brake line and permits fluid from this tube to apply the brakes. Pneumatic braking is only used when hydraulic pressure is lost. TMMost guy's I've know that have used it have blown a tire or two, (a wheel and tire only costs you about \$ 1200 service exchange!).





System Descriptions

Gear Up Braking

Light braking is automatically applied to the main wheels from the gear up hydraulic line during retraction. The nose gear tires rub against brake shoes in the forward wheel well (spin brakes) on retraction.

Parking Brake

This provides a means of locking the brake pedal linkage in a brakes applied condition. They maybe set (correct term for brakes on) by depressing either set of brake pedals and pulling up on the parking brake lever, then release pedals. Brakes will be applied if pressure is available the accumulator in the brake system is the backup source and will hold for many hours. The parking brake lever also closes an electric solenoid valve in the anti skid return line on the main gear anti skid valves thus preventing leakage through the anti skid valve as depleting accumulator pressure. There is light adjacent to the lever to tell you when the brake lever is up (set position). A second light is installed at the external power receptacle panel. Releasing the brakes is by pressing on the pedals.

Anti Skid

A skid is prevented by controlling the deceleration rate of each wheel. Locked wheels due to hydroplaning are prevented by comparing the speed of each wheel to the speed of the other wheels. This is achieved by releasing some or all of the brake pressure applied by the pilot through the modulating anti skid valves. When a brake is released by the anti skid system the corresponding indicator in the cockpit will display REL. touchdown protection prevents landing with the brakes on and keeps all brakes released until landing gear logic is satisfied that the aircraft is firmly on the ground. In flight when a REL is displayed with the gear down, touchdown protection is operative. After landing wheel spin up can override faulty touchdown logic. Do not test the anti skid while applying brakes, brake release may occur.

TMA more in depth look at anti skid will come when I get more time.

OXYGEN

There are two fixed oxygen systems, one for the cockpit and one for the cabin. In addition there may be up to five portable oxygen bottles. One in the cockpit and four located throughout cabin. Oxygen for both systems, cockpit and cabin is provided by high-pressure cylinders. Three cylinders are located along the right wall of the electronic equipment compartment. The forward cylinder provides oxygen to the cockpit. The other two cylinders provide oxygen to the cabin system.

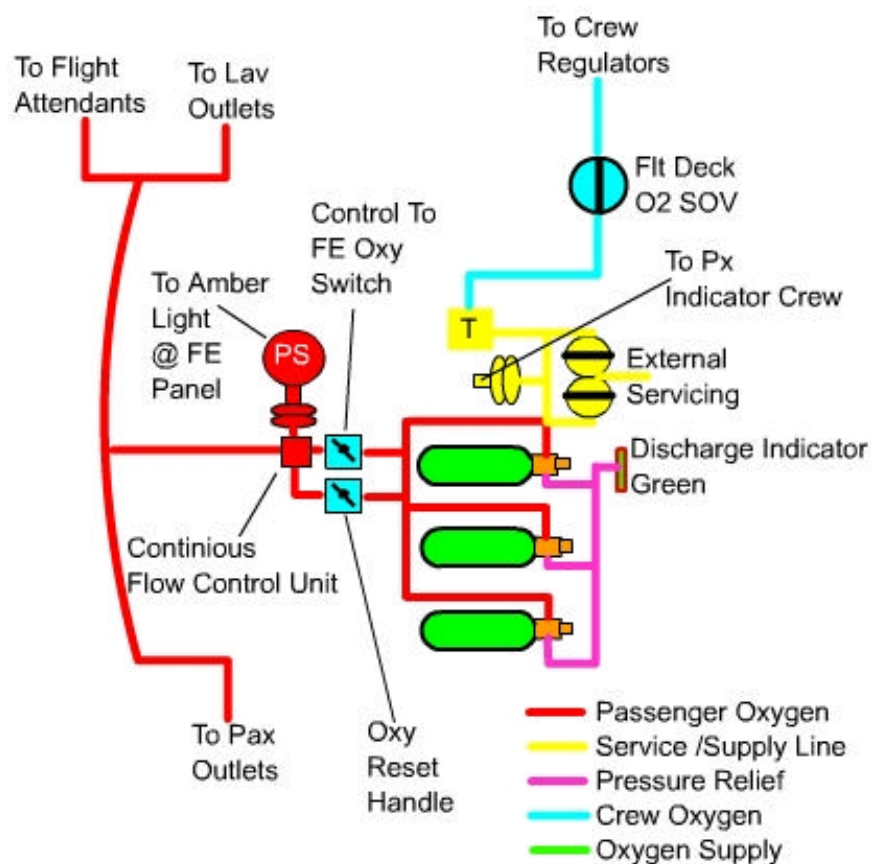
Each cylinder has its own shutoff valves, pressure gauge and overpressure relief. In the event any cylinder overpressure, a valve relieves all of that bottle's oxygen. It will be dumped overboard through the overpressure discharge line, common to all three cylinders. The overboard discharge port



System Descriptions

is located on the right side of the fuselage just below the E & E cooling exhaust and is sealed with a green disc. The overpressure discharge lines from all three cylinders vent through this one port. a missing green disc indicates a possible thermal overpressure discharge from one or more cylinders. If the disc is missing or broken, relief may have taken place and further investigation is required.

The high pressure of the oxygen from the crew cylinder is lowered by a reducer/regulator. Lower pressure then flows to the cockpit shut-off valve. This shut-off is located on the sidewall just aft of the F/O seat. Oxygen from the reducer/regulator flows to each individual regulator and mask in the cockpit.



Minimum crew oxygen pressure is normally around 1200 PSI for dispatch and is read on the gauge located top centre of F/E panel. This gauge is an electrical repeater of the pressure gauge on the bottle and operates when the aircraft is powered.

All cockpit oxygen regulators are diluter demand type. At the top right of the regulator is the flow indicator, which blinks when oxygen is flowing to the mask. The supply lever is a two position off /on selector on the lower right. The normal position of the lever is at all times ON. To the left of the supply lever is the oxygen lever. This lever should always be in the 100% position in preparation for immediate use in the event of smoke, contaminated air in the cockpit or a depressurisation at altitude. The last lever to the left is the



System Descriptions

emergency lever. UP is the ON position. This lever provides 100% oxygen to the mask at a slight overpressure, regardless of the position of the oxygen lever. The OFF position of the emergency lever allows the oxygen lever to regulate the oxygen flow, 100% or diluter demand as selected. The masks in the cockpit are of the "quick-donning", or "sweep-on" type. The oxygen / boom selector for the flight engineer is located on his jack box. The boom position is deactivated. His oxygen mask mike is activated by the associated push to talk button, labelled "PTT". For the pilots, the oxygen/boom selector is on their respective jack boxes. They also have a PTT (push to talk) button. When selected to oxygen and mike selection is in PA, the associated oxygen mike is "HOT". The BOOM position is deactivated on some airplanes. and the oxygen position should always be selected. Then the push to talk button or the rocker switch on the rear of each control wheel will activate the oxygen mike to what ever is selected to transmit.

Cockpit indication of passenger oxygen system pressure is read on the electrical repeater gauge next to the crew oxygen pressure gauge on the top centre of the SIC panel. Minimum pressure for dispatch is around 1450 psi for the passenger oxygen system. Oxygen from the two passenger cylinders flows from a common manifold into two paths. The pressure is reduced by two parallel pressure reducing regulators and the flow is controlled by two flow valves. Downstream of the two valves the two paths rejoin. When either flow valve is opened, oxygen pressurizes the passenger oxygen manifold, causing a pressure switch to illuminate an amber light. When cabin oxygen is required, one or both of these valves opens. The valve can be opened on of three ways, Pneumatically, electrically or manually. Both valves are designed to open pneumatically when cabin altitude exceeds 14,000 feet. One opens electrically by use of the oxygen switch on the FE panel, and the other by using the manual "T" handle located on the FE aux panel below the APU control panel or under floor access panel on the 100 series. This "T" handle can be used to close and reset both valves when oxygen is no longer needed, regardless of how the valves are activated. When the cabin oxygen manifold is pressurized. The surge of pressure trips the latches on the oxygen mask doors and the doors open. Four masks are then dropped at each passenger service unit and two at each flight attendant's station and in the lavatories. Oxygen is now available to each mask valve in the cabin. In order to establish Flow, the user must pull down the mask. This action extracts the valve activating pin to allow flow of oxygen. The mask delivers oxygen diluted with cabin air to the user depending on cabin altitude. If the automatic system does not open them, pushing a sharp object into a slot can open the passenger doors. Flight attendant doors can be opened by moving a latch to the side, if the automatic system failed. Lavatory doors are similar to the flight attendants doors. The three ways of terminating oxygen flow are:

1. The valve can be closed by reinserting the pin into the shut-off valve assembly.
2. By pushing the manual toggles on the ends of the shut-off valves to there up position.



System Descriptions

3. By pushing the "T" handle to reset, all the way down for 5 seconds to dump the oxygen manifold pressure, and then back to the "OFF" position. This will close both valves.

There is insufficient oxygen flow below 10,000' cabin altitude for passenger use. At a cabin altitude of 10,000' an intermittent horn will sound and can be silenced by a cutout button on the pressurization panel.

Portable Oxygen Bottles

There are approximately five portable oxygen bottles aboard the airplane. Four bottles in the cabin are for passenger use, and one with a full face mask is carried in the cockpit. Each bottle has a yellow shut-off knob and pressure gauge.

PITOT STATIC SYSTEM

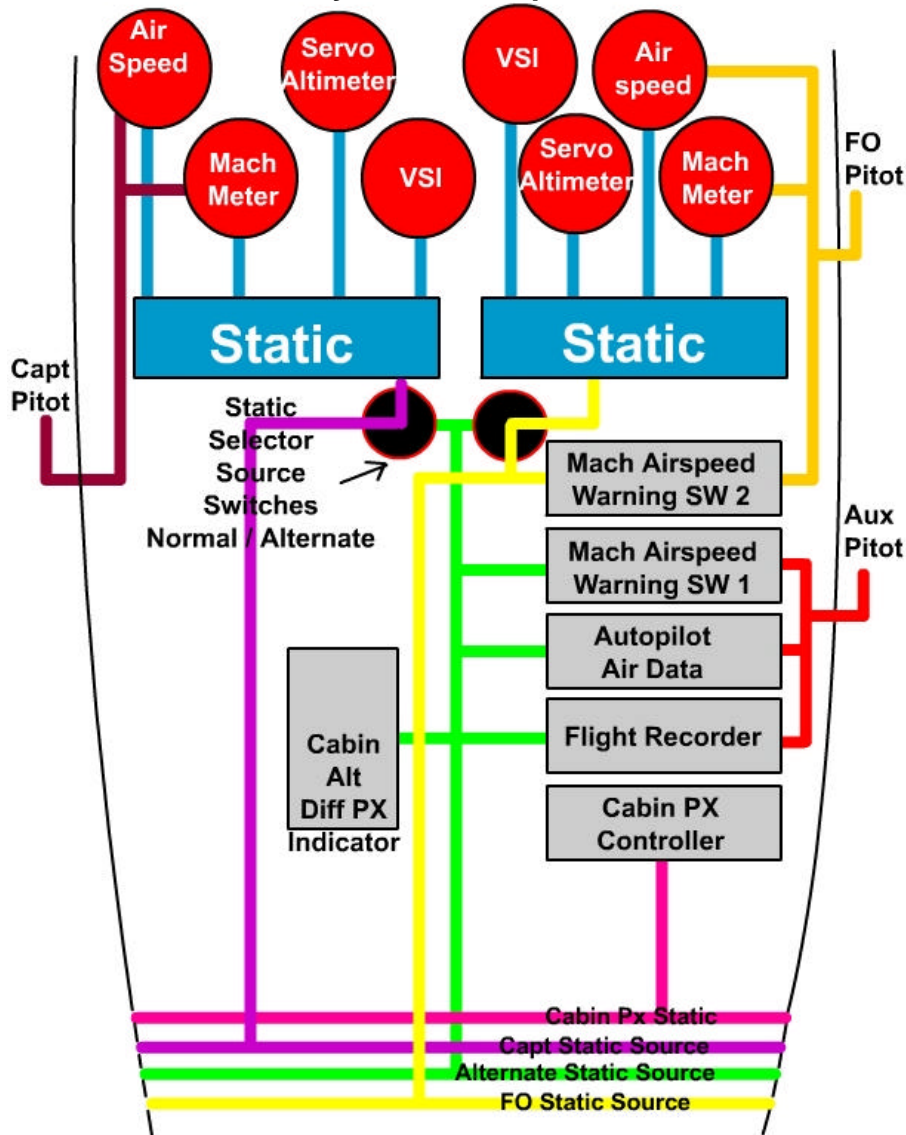
The pitot static system provides total and static inputs for the pressure sensing instruments and systems, which have functions that vary with altitude and airspeed.

A sample system as show below would have three independent systems referred to as Capt, FO's and Aux. A forth port is connected to supply pressure inputs for the cabin pressure controller. All of the pitot probes are equipped with heaters for anti-icing protection. Along with the ability for the captain or fist officer to select an alternate pressure source.

A simplified sample system is shown on the next page.



System Descriptions



PNEUMATICS

In general, the pneumatics system provides compressed air at a constant flow rate to each of the air conditioning units, where its temperature is controlled and it is ducted into the cabin. Boeing refers to the air conditioning units as packs. The pressurization control system restricts the escape of this air from the cabin to maintain proper pressurization in the cabin. The pneumatic system also provides compressed air for engine starting. The engine compressors, auxiliary power unit, or a ground unit can be used to supply the pneumatic system. The pneumatic manifold is normally supplied from engines one and three, with backup from engine two. Engines 1 and 3 are usually referred to as the "pod" engines. The APU or an external air source may also be used. AC powered valves in the pneumatic manifold

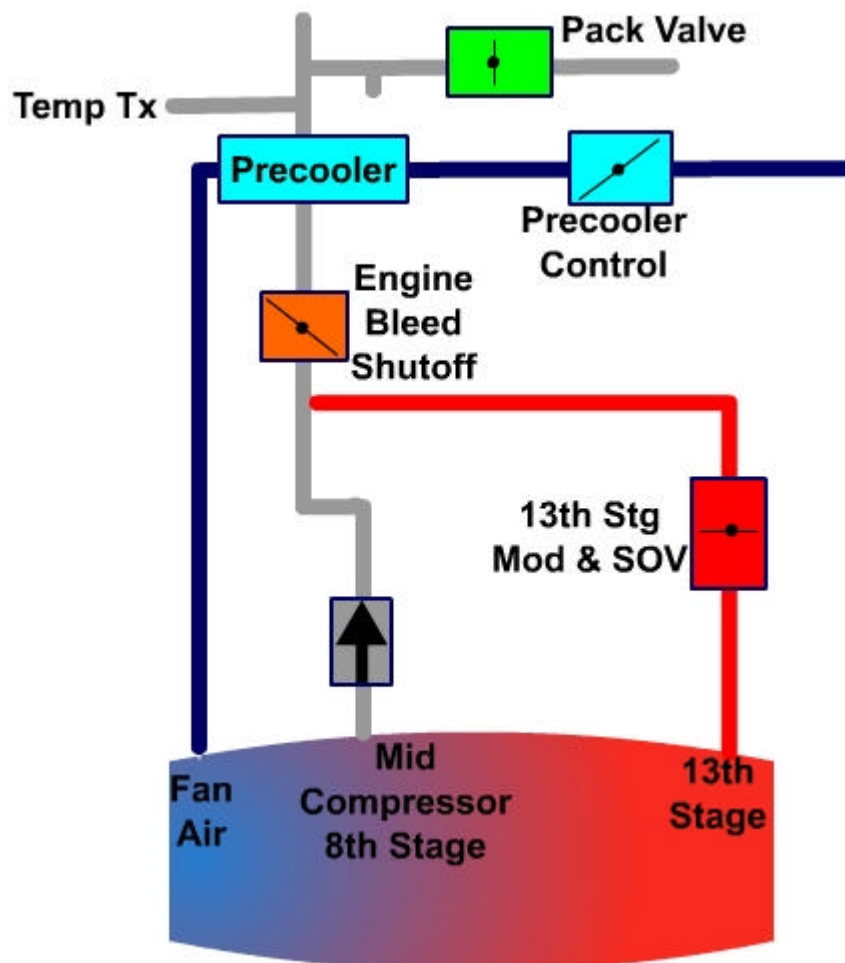


System Descriptions

control the delivery of engine bleed air into the manifold. The engine 2 bleed valves, when closed, isolate the two sides of the pneumatic manifold so that each air conditioning pack is supplied by a separate air source.

The bleed air switches flight engineers panel normally control the engine bleed valves. The engine 2 bleed switches also control the APU's bleed air valve. The fire switches will close the bleed valves when those switches are pulled. The number 2-fire switch closes both engine 2 bleed valves. In order for the engine bleed valves to respond to the positions of the bleed switches and fire switches, the AC buses must be powered.

There are two bleed valves an each pod engine. The 8th stage bleed normally provides most of the air to the manifold except at low engine power, at which time the 13th stage valve opens automatically to augment the flow. In normal operation, pneumatic flow is arranged so that engine one provides air for the left air conditioning pack, and engine three provides the right pack. As the compressed air passes from the pneumatic manifold to an air conditioning pack, it is fed through a flow sensing venturi. If the flow rate is sensed to be too low, the venturi signals the 13th stage bleed valve to open and increase flow.





System Descriptions

At high engine powers there is more than sufficient airflow from an 8th stage bleed for the associated air conditioning pack. As the compressed air leaves the pneumatic manifold it passes through a modulating and shut off valve on its way to the flow sensing venturi. The modulating and shut off valve is signalled from the flow sensing venturi to control the flow at high power settings.

As the 13th stage valve on a pod engine opens or closes, the temperature of the air from that engine varies. The temperature of this bleed air must be controlled for air conditioning pack operation. This is accomplished automatically on engines one and three by a pre-cooler on the bleed lines from each engine. The pre-cooler uses fan stage air to cool the bleed air. A temperature sensitive valve controls the rate of flow of fan stage air through the cooler. There is no flight deck control for the pre-cooler.

To protect against excessively high temperature in the pneumatic duct from a pod engine, an automatic trip off feature is installed. When the temperature of the bleed air is too high, the bleed air valve closes and a trip off light next to the affected bleed switch illuminates.

After a trip has occurred and the temperature of the bleed air has dropped sufficiently, pressing the reset button on the air conditioning panel will return the bleed valve to normal operation. If the condition that caused the trip to occur still exists, however, the bleed valve trip will reoccur.

Engine No. 2 supplies air only from the 8th stage. Since supplemental air is not supplied from the 13th stage on this engine, a pre-cooler is not fitted. To warn of excessively high temperature in the engine No 2 bleed system, a high temperature light is provided. No automatic trip off is associated with illumination of this light.

Opening the engine No. 2 left bleed switch will open the left engine 2 bleed valve to supply air to the left air conditioning pack. The engine 1 bleed switch should be closed in this case so that only one engine is supplying bleed air to that pack.

Pressure in the bleed air distribution system can be read on the duct pressure gauge at the flight engineers panel. There are duct pressure transmitters installed on both sides of the ducting. If both engine 2/APU bleed switches were open, the left and right pressures would be equal indicating common pressure.

There is provision for using an external air cart for pneumatic supply. The external air cart is connected to the bleed air distribution system between the right No. 2 and No. 3 bleed valves. On the exterior of the airplane this connection is on the aft right side of the fuselage.

The APU can be used on the ground to deliver compressed air to the air conditioning packs or the pneumatic manifold. The APU bleed air valve, which bleeds compressed air from the APU, will open when either (or both) engine 2/APU bleed switch is in the open position.

On the 200 series aircraft there is a flow multiplier. The purpose of the flow multiplier is to augment the bleed air output of the APU so that there will be



System Descriptions

sufficient quantity of compressed air to operate both air conditioning packs from the APU for ground operation.

With both air conditioning packs operating the augmented APU bleed air is ducted directly to both packs. If the air conditioning packs are not operating, APU bleed air travels back through the ducting, which contains the flow sensing venturi and modulating and shutoff valves to the pneumatic manifold. This allows the APU bleed air to be used for engine starting.

With the APU operating, when one air conditioning pack is turned on, both modulating and shutoff valves close to isolate the air conditioning packs from the pneumatic manifold. Therefore, with at least one engine 2/APU bleed switch open and one air conditioning pack on, there is no flow of air through either modulating and shutoff valve. The duct pressure gauges are installed in the pneumatic manifold. If the APU is the only source of bleed air to the pneumatic system, and at least one air conditioning pack is turned on, no compressed air will reach the gauges, and they will read zero pressure. (200 Series, 100 Series will show a reduced pressure indication).

If the air conditioning packs are not operating, the APU provides compressed air to the pneumatic duct through the modulating and shutoff valves. This is the normal configuration for engine starting.

Heat from a broken pneumatic or anti-ice bleed air duct could cause damage to the airplane structure. Three detection systems are installed in the areas of these ducts to give warning of duct failures.

A detection system is installed in each pod engine strut area inboard of its engine firewall. A third system combines several sensors to detect overheat in what is referred to as lower aft body. These lower aft body sensors are located on either side of the aft airstairs, above the ceiling of the aft cargo compartment, and in the fuselage keel beam. An overheat sensed by any of the three detection systems is reflected in the flight deck by illumination of the appropriate amber warning light on the flight engineers panel. The adjacent test button is used to test simultaneously the light bulbs and the continuity of the overheat detection sensors for the struts and lower aft body.

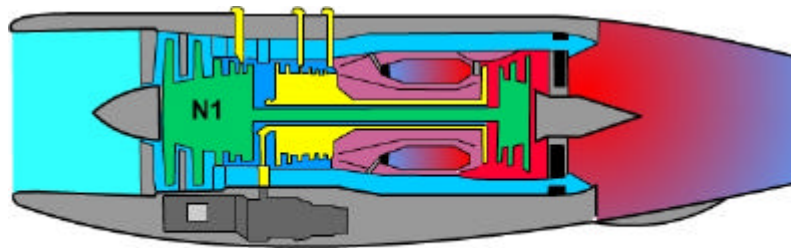
POWERPLANT

Three aft mounted Pratt and Whitney JT8D series dual compressor turbofan engines power the Boeing 727. Each engine produces Between 14,000 and 16,900 pounds of thrust at sea level depending on model installed (new fan mod excluded). The design of the engines includes an integral fan bypass which routes fan air the entire length of the engine. Cooling the engine jacket. The fan air mixes with the turbine exhaust just forward of the reverser section where it increases thrust and reduces engine noise. The engine has two axial flow compressors. The low-pressure compressor includes the fan stages and is driven by the low-pressure turbine stages. The high-pressure compressor is driven by the high-pressure turbine stage. The dual compressor feature allows for more closely matching compressor blade speed to the increasing



System Descriptions

pressure and temperature of the air as it passes through the compressor stages. This results in increased compressor efficiency, a higher compressor ratio and increased thrust. Metering fuel to the combustion chambers controls engine thrust. The fuel control unit accomplishes this function by sensing, in addition to inlet temperature and pressure and combustion chamber pressure. The RPM of the high-pressure turbine-compressor rotor. The fuel control does not sense low-pressure rotor RPM, its speed depends on airflow through the high-pressure rotor.



N1 & N2

The accessory section of the engine is driven by the high-pressure rotor. The engineering symbol for the RPM of the low-pressure rotor is N1, and for the high pressure rotor is N2. Common usage has made the terms N1 and N2 refer to the rotors themselves. N1 is measured off the low-pressure rotor directly, and N2 is measured at the accessory section. If the shaft to the accessory section from the N2 rotor fails, N2 will read zero even though the rotor may still be turning. The tachometers in the cockpit for N1 and N2 are driven by self-powered tachogenerators.

Exhaust Gas Temperature

Exhaust gas temperature is measured at the low-pressure turbine outlet. Standby AC power is required to display EGT in the cockpit.

Engine Pressure Ratio

Engine pressure ratio or EPR is the relationship between engine inlet pressure. PT2, and turbine exhaust pressure. PT7. It is a measure of the thrust being produced by the engine. EPR is displayed on the face of each EPR gauge with a needle and a digital counter. Any blockage of the inlet pressure probe, which is located in the engine nose dome, will result in erroneous EPR indications, which can cause serious errors in power settings. N1 RPM is also an excellent measure of power and can be used as a cross check in icing conditions when the EPR probe may be blocked.

Electrical power is required to operate the EPR gauges. The EPR gauges receive inputs from the PDCS when the set knob on the instrument is pushed in. These inputs drive the internal reference marks or "bug", and a digital counter on the face of the instrument.

Fuel Flow

Fuel flow is measured between the fuel control and the burner nozzles. Both AC and DC power are needed for the fuel flow gauges.

The engine accessory section is driven by the high-pressure, or N2, rotor. It

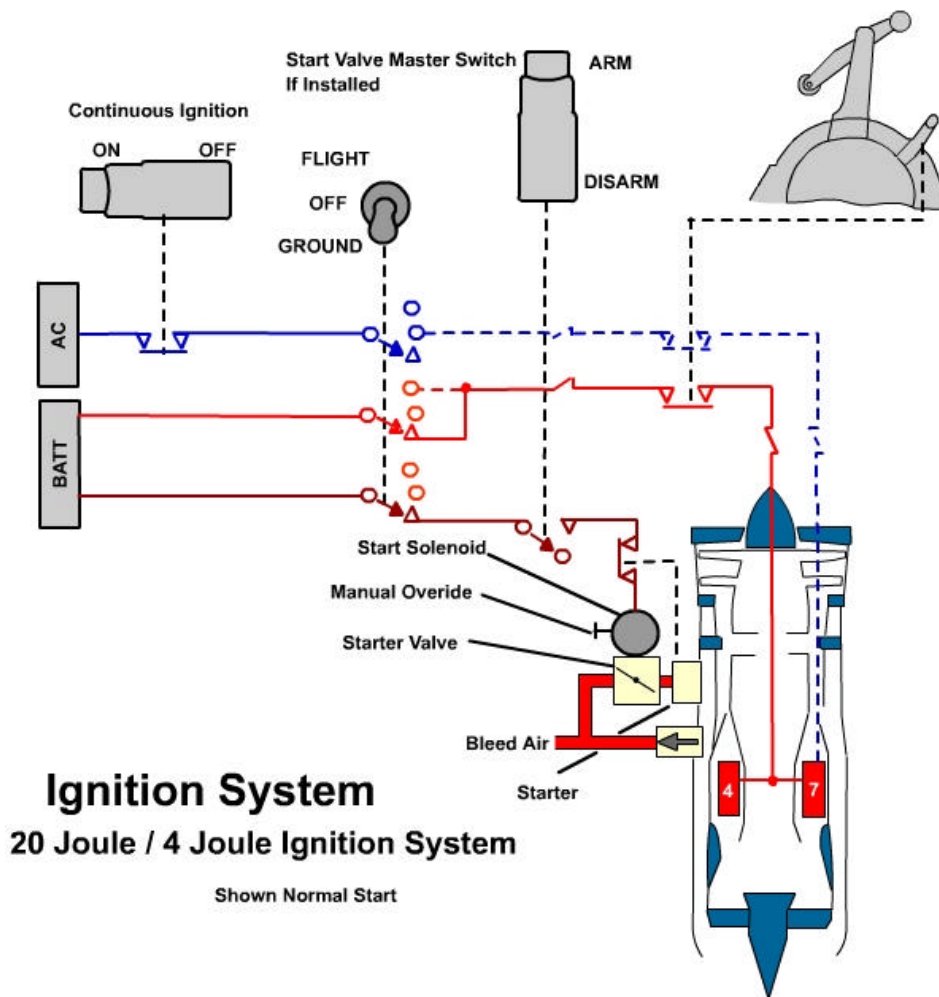


System Descriptions

has pads for the engine's own fuel and oil systems, an AC generator, a hydraulic pump, and a starter. Only engines one and two have hydraulic pumps installed. Bleed air is extracted from the sixth, eighth and thirteenth compressor stages for the pneumatic and anti-ice systems. When any bleed air valve is opened, that engine's EPR will vary slightly.

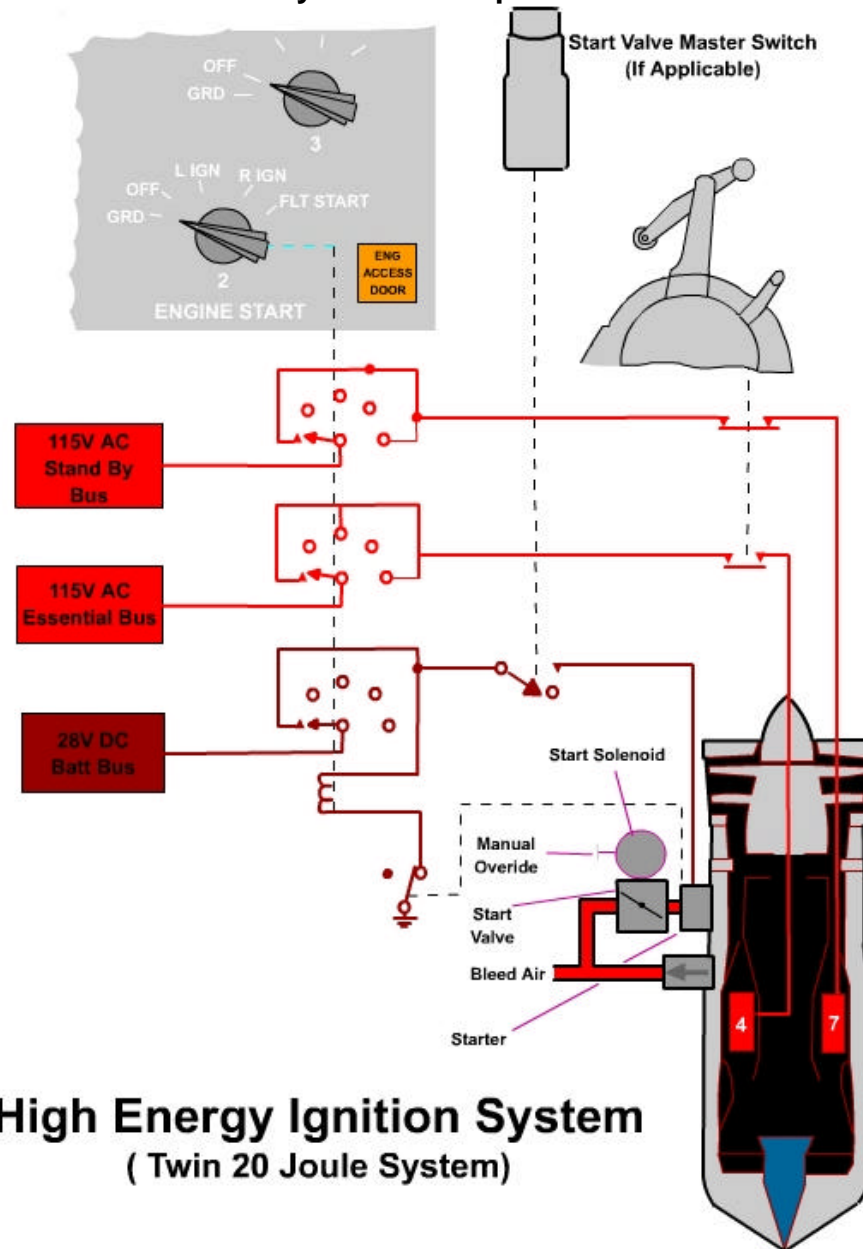
Starter

A pneumatically driven starter is attached to the accessory section. An electrically operated starter valve allows compressed air from the pneumatic system to drive the starter. The GROUND position of the engine start switch controls the starter valve.





System Descriptions



High Energy Ignition System (Twin 20 Joule System)

Reversers

There is a thrust reverser unit on the aft end of each engine that deflects the exhaust gases forward to shorten the landing roll. The reversers may be of a cascade type, which consist of internal clamshell doors and external cascade vanes. Alternatively there is bucket door system with deflector doors only. The clamshells are pneumatically operated by bleed air from their respective engines. But not by compressed air delivered from the APU or an external air source. The reverser mechanisms are operated by levers on the throttles. With the throttles out of the idle position an interlock prevents operation of the reverse levers. If a malfunction occurs which causes an engine to go into reverse thrust with its throttle in forward thrust, that throttle will move forcibly to the idle position. When an engine is in reverse thrust, the interlock prevents



System Descriptions

forward movement of the throttles. As with the throttles, if the engine goes into forward thrust with the reverse lever in reverse thrust the reverse lever will move forcibly into the forward thrust position. REVERSER NOT STOWED lights, located above the engine instruments, indicate that the clamshells of the reverser are not fully stowed in the forward thrust position. The reverse interlock on each engine will prevent motion of the throttle into the forward thrust range if the engine is actually in reverse thrust and the reverse interlock will also prevent the reverse lever from applying reverse thrust until the clamshells are in the full reverse position.

Oil System

Each engine has an independent oil storage and distribution system, which provides cooling, and lubrication of gears and bearings. The tank has a useable capacity of four gallons. An engine driven pump pressurizes the oil from the oil tank. The pressurized oil passes through a filter, is cooled and is piped to the bearings and accessory section gears. The oil is then returned to the tank by scavenge pumps. After it leaves the cooler the pressure and temperature of the oil are measured and the values are transmitted to gauges on the flight engineer's panel. In addition a separate pressure switch will turn on a light on the pilots' centre instrument panel if oil pressure is too low. If the oil filter is unable to process the output of the oil pump because the filter is clogged a bypass will open to allow oil to reach the engine. If this occurs, the difference in oil pressure across the filter will cause the same low oil pressure light on the centre instrument panel to illuminate. The label on the light signifies this dual purpose.

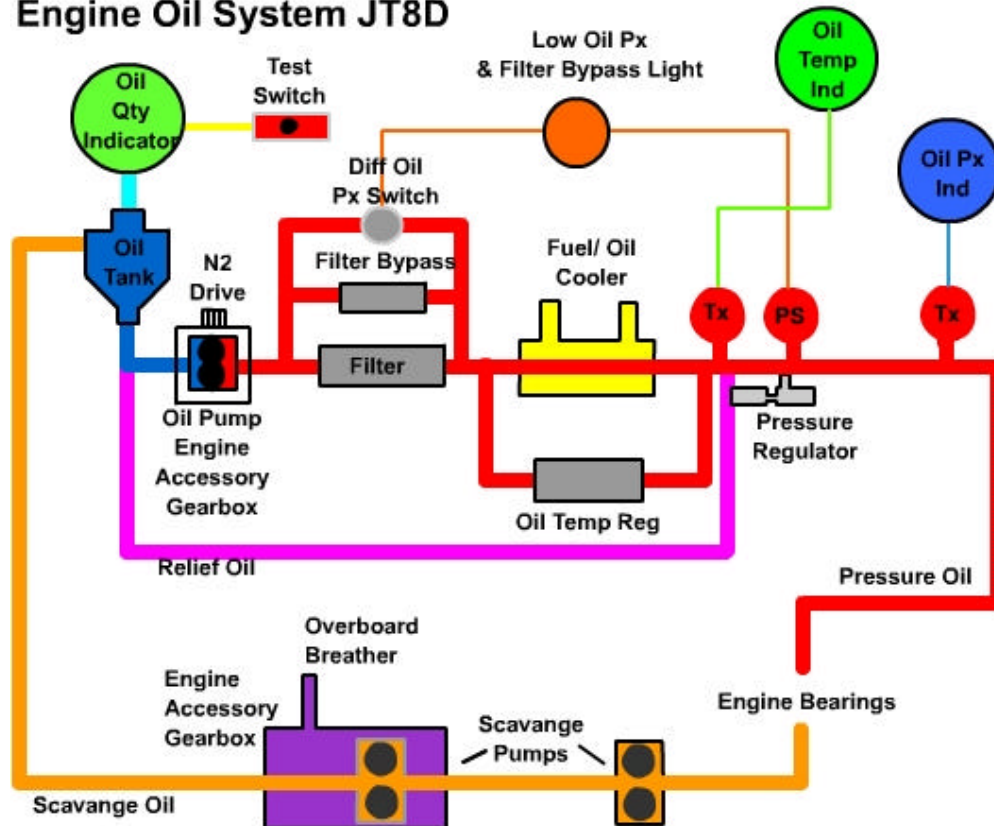
LOW OIL PRESSURE OR FILTER BYPASS

If the low oil pressure or filter bypass light comes on, the cause can be determined by reference to the corresponding oil pressure gauge. If the oil pressure reading is normal the light indicates that the oil filter is being bypassed. The oil is cooled in a heat exchanger through which fuel from the engine fuel system is circulated. As fuel temperature and flow rate vary, oil cooling will vary, and oil temperature will change. Gauges on the flight engineers panel monitor oil quantity, temperature, and pressure. A test button in the lower right corner of this panel is used to test the operation of the oil quantity gauges. Standard instrument markings are used on the temperature and pressure gauges: green indicating the normal range, yellow indicating caution and red showing the operating limit.



System Descriptions

Engine Oil System JT8D



Oil temperature and pressure limits are indicated by coloured arcs on the gauges. The temperature and pressure limits for continuous operation are shown by green arcs. There is a 15-minute time limit an engine operation with the oil temperature indicating in the caution ranges shown by a yellow arc. Operation with the oil pressure in the caution range, a yellow arc, is allowed for a short period at reduced thrust. Red radials show the maximum oil temperature and maximum and minimum oil pressures. An amber light on the forward instrument panel will illuminate if the oil pressure in the associated engine is too low. The minimum oil quantity for dispatch is one gallon (US) and a quart (US) for each hour of planned flight.

Fuel

Fuel is normally transferred under boost pump pressure to the engine from the fuel tanks. At the engine the fuel is pressurized by a low-pressure engine driven pump, passes through a heater and is filtered. The high-pressure engine driven pump increases the pressure of the fuel before it reaches the fuel control. The fuel control modulates the flow of fuel to the engine to maintain its power at the level selected by the position of the throttles in the cockpit. Between the fuel control and the burner nozzle in each engine is a fuel flow transmitter. It measures the rate at which fuel is delivered to the burner nozzles in that engine and sends this information to the fuel flow gauge on the centre instrument panel. To eliminate the need for a drag producing oil cooler in the slipstream, the fuel is used as the cooling medium to cool the engine oil. From the cooler the fuel is directed to the burner nozzles by the



System Descriptions

pressurizing and dump valve. At low power settings, only the primary fuel manifold is pressurized. As power is increased the secondary manifold is also used. At low temperatures the fuel filter is susceptible to clogging by ice particles in the fuel. If this occurs a bypass around the filter opens allowing fuel to reach the engine. The pressure drop across the filter is sensed, causing an amber icing light on the flight engineer's panel to illuminate upstream of the fuel filter. A fuel heater is installed. When the fuel heat switch on the flight engineer's panel is moved to ON, a valve opens allowing high stage engine bleed air to pass through the heater heating the fuel rapidly. The heated fuel flows to the fuel filter melting the ice in the filter. The heated fuel continues on through the fuel control and on to the oil cooler. Since the fuel is now warmer it cannot cool the engine oil as efficiently, and oil temperature on that engine rises.

A blue IN TRANSIT light on the flight engineer's panel will illuminate as long as the hot air valve does not match the position of the fuel heat switch. When the hot air valve opens, a slight drop in EPR can be seen. A rise in engine oil temperature verifies that the fuel is being heated. The low-pressure engine driven pump normally provides for suction feed if necessary from a fuel tank to force fuel through the fuel heater and filter to the high-pressure engine driven pump. If the low-pressure pump fails, a bypass allows fuel to flow directly to the high-pressure pump without passing through the heater or filter. In this situation the fuel cannot be heated and use of the fuel heat switch will not cause a rise in engine oil temperature. If fuel temperature fails to rise while the fuel heat is in use, the low-pressure engine driven pump may have failed.

Fuel Temp

Fuel temperature in the number one fuel tank is displayed on the flight engineer's panel to determine if the fuel is approaching a temperature limit or if any fuel filter blockage could be caused by ice. Number one tank was chosen since its fuel is the coldest.

Ignition & Starting

Ignition and start valve operation are controlled by the ignition switches and the start levers. The ignition switches are located on the overhead panel, and the start levers are on the centre console throttle quadrant. Each engine's ignition system provides two levels of ignition energy. High-energy ignition, used for engine starting. It receives its power from the DC circuits. Low energy ignition, which can be used continuously without decreasing the life of the engine igniters, receives its power from the AC circuits. Each ignition switch arms its ignition circuits, however, the ignition will not be activated unless the associated start lever on the throttle quadrant is moved to start or idle. With the ignition switch in either flight or ground, and the start lever in start high-energy ignition is activated. With the ignition switch in either flight or ground and the start lever in idle, low energy, continuous ignition is activated. The ground position of the start switch has the additional function of



System Descriptions

opening the associated engine start valve to initiate engine rotation for start. Once the engine reaches the proper RPM, the start lever is moved to start, which causes the high-energy ignition to activate as well as the fuel to be introduced to the engine. With the start lever in its normal in-flight position

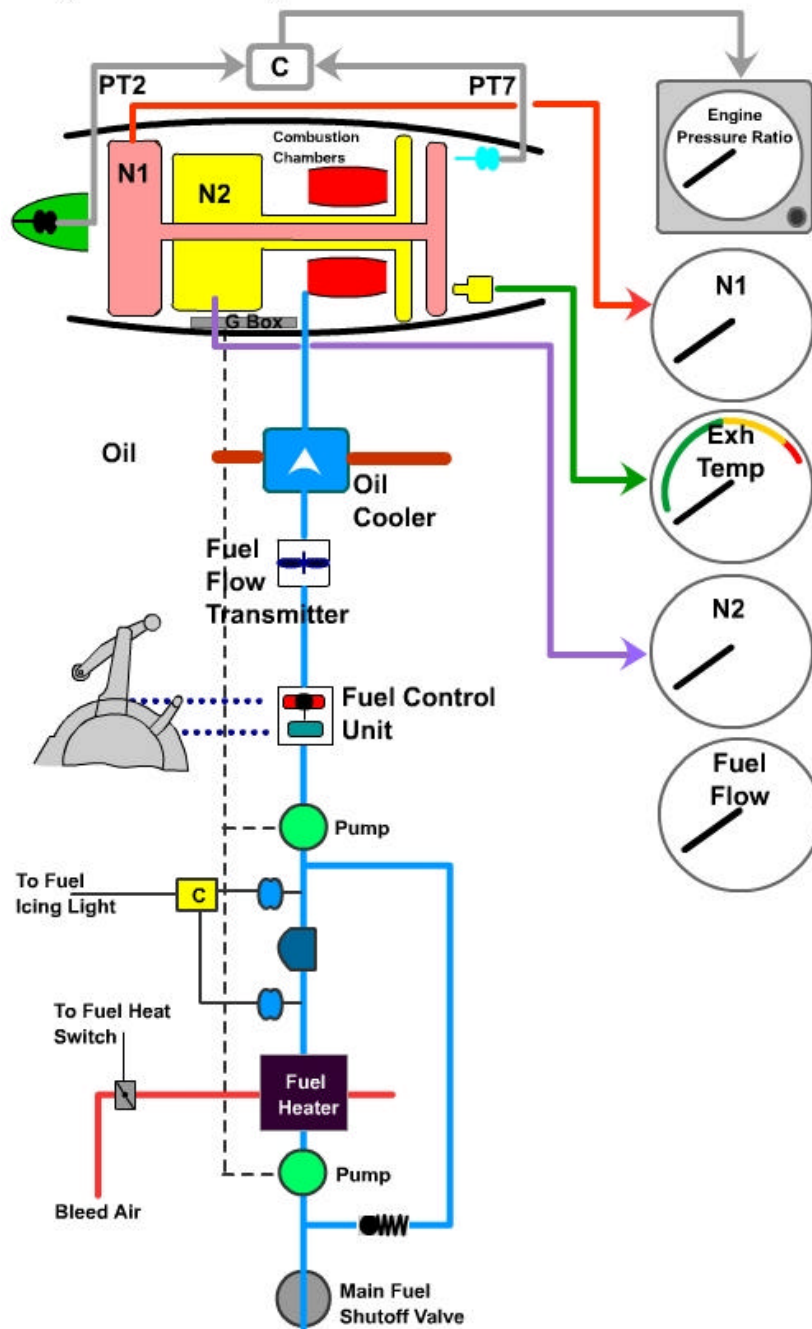
idle, moving the start switch to flight provides low energy ignition. The ground position of the start switch would also activate the low energy ignition but this would open the start valve, subjecting the starter to potential damage. Low energy ignition should be used during takeoff, landing, icing conditions, turbulence and when using fuel heat.

Fuel to the engine as well as ignition is controlled by the start lever. With the start lever in start or idle the fuel valve in the fuel control is open providing fuel to the engine. In addition, the fuel shutoff valve at the fuel tank is open. With the start lever in cut-off, the fuel is shut off at the fuel control and at the wing tank shutoff valve.



System Descriptions

Engine Fuel System 100 Series Aircraft



S Duct

The "S" duct which supplies air to number two engines has an access door for the number two engine inlet. It is located directly in front of the number two-engine inlet in the duct. A microswitch senses if the access panel is secured. If not, an amber light labelled ENGINE ACCESS DOOR next to the engine start switches will illuminate.



System Descriptions **PRESSURISATION**

There are two types of pressurisation installed on the B727, Pneumatic controlled and electronic controlled.

The electronic pressurisation system

Four independent modes of operation: Automatic, Standby, Manual AC, and Manual DC. The Standby mode is a semiautomatic system that acts as a backup for the automatic mode of operation. The two manual modes, acting as backups for the entire system. The chart below the pressurisation panel is used to set or check the cabin altitude in relation to airplane altitude in the Standby mode.

The gate-type electrically controlled outflow valve is located on the right rear side of the airplane near the tailskid. Although there are other acceptable pressure bleeds, most of the air exits through this valve. Two safety pressure relief valves located just forward of the tailskid limit the cabin to a maximum of 9.6-psi differential pressure. A negative relief valve prevents negative cabin pressure from exceeding 1 psi.

Automatic warning of cabin altitude exceeding 10,000 feet is provided by a horn. Depressing the horn cutout on the pressurisation panel raises cabin PA volume as well as silencing the warning horn.

Correct operation of the pressurisation control system requires information for the system from the air data computer. The Captain's or First Officer's altimeter setting, (Captain's altimeter setting is used in Auto mode and First Officer's setting in Standby), ambient air pressure from an external part, and the landing gear ground safety sensor. On some airplanes the barometric pressure must be set on a separate counter on the flight engineer's panel.

AUTOMATIC CONTROL

When the proper information is set, the automatic mode will smoothly control the cabin from before takeoff to after landing with little or no further input from the crew. The left side of the pressurisation section of the flight engineer's panel is used for manual inputs to the Auto mode. Planned cruise altitude is entered in the Flight Altitude window and landing field altitude in the Land Altitude window. The Flight/Ground switch, when moved to Flight, signals the pressurisation system to begin pressurising the airplane by moving the outflow valve from the full open position. With the mode selector in Auto and

the airplane on the ground, it will cause cabin altitude to descend to 200 feet below the present field elevation, resulting in a differential pressure of less than .125 psi.

After takeoff, the cabin climbs in proportion to airplane climb. Cabin rate is limited to approximately 500 feet per minute during climb and 350 feet per minute during descent. The cabin climbs automatically holding at intermediate altitudes as the airplane holds until the airplane reaches cruise altitude. During high altitude cruise the cabin enters a barometric hold phase.



System Descriptions

The cabin will maintain a constant altitude at a nominal differential pressure of 8.5 psi. Slight variations of airplane flight altitude will cause no change in cabin altitude, but the differential pressure will vary. If the airplane climbs, so that the pressure differential reaches 8.65 psi, however, it will remain at that differential and the cabin will climb. Upon descent, the cabin will smoothly descend holding again as the airplane holds until the airplane reaches the landing altitude programmed. With the barometric pressure set for the destination, the cabin descends to 300 feet below the setting in the Land Altitude window and at touchdown the cabin climbs to 200 feet below the Land Altitude setting. Placing the Flight/Ground switch to Ground after landing depressurises the cabin slowly. In the auto mode during low altitude cruise, the full differential pressure is not needed. The cabin climbs or descends to a cabin altitude 300 feet below the landing altitude and stays there until touchdown.

If it becomes necessary to return to the departure airport before reaching cruise altitude, the system automatically sets the cabin for landing at the takeoff field. If the airplane has not reached the cruise altitude set in the Cruise Altitude window before descending, the OFF SCHEDULE DESCENT sequence will occur. If the landing is to be made at the departure airport, the system will set cabin altitude without further crew input.

If, however, the flight has reached its destination without climbing to the final cruise altitude, the OFF SCHEDULE DESCENT mode must be cancelled. The light may be extinguished by rotating the flight altitude digital readout to the airplane's present altitude. At this point the system will be returned to normal operation and the cabin pressure will be set for landing at the altitude in the Land Altitude window. The correct land altitude value must be set in the Land Altitude window prior to beginning this procedure if landing at other than the original destination.

STANDBY CONTROL

The Standby Mode. The green Standby light will illuminate as a result of automatic transfer from the auto mode, or crew selection of the Standby mode. Control of pressurisation in Standby mode is through the Cabin Altitude selection in the centre of the pressurisation panel. The desired cabin altitude is set by the flight engineer to control the pressurisation he uses the chart next to the panel to determine the proper cabin altitude for the airplane altitude being flown. The Standby rate knob controls the rate at which cabin altitude changes in the Standby mode. The rate knob has 50 to 2000 feet rate

of cabin climb or descent capability. Normally the rate knob is set on the Index Mark, which results in a rate of climb or descent of 300 feet per minute. To operate the pressurisation system in Standby, the Cabin Altitude is set to 200 feet below the takeoff airport altitude for takeoff, the barometric pressure is set in the appropriate instrument or counter, and the Flight/Ground switch is moved to Flight. After takeoff, the proper cabin altitude is found on the chart next to the pressurisation panel and is set in. As the airplane begins its



System Descriptions

descent for landing, the flight engineer sets the Cabin Altitude to 200 feet below the destination airport's altitude, and the barometric pressure is set. The airplane is depressurised after landing by moving the Flight/Ground switch to Ground.

The pressure profiles that might be experienced, on a flight are. In this example, the takeoff airport elevation is at sea level. Since the takeoff airport elevation is sea level, the controller is set to minus 200 feet prior to takeoff. This allows the airplane to make a smooth transition to pressurised flight. The cruise altitude will be 35,000 feet. After takeoff, the corresponding cabin altitude is set in the controller, 5,600 feet. Prior to descent, the landing elevation minus 200 feet is set in the controller. The airport elevation is 2,000 feet, so the controller is set at 1,900 feet.

Control will automatically shift from the Auto mode to the Standby mode for any of the following reasons: an excessive cabin rate of change, a power loss of more than 15 seconds to the auto portion of the pressurisation system, or if the cabin altitude exceeds 14,000 feet. If an auto mode failure occurs, the amber AUTO FAIL light will illuminate and the green STANDBY light will also illuminate because the pressurisation system has reverted to the Standby mode. If any of the modes fail to control the cabin pressure so that the cabin altitude rises to 14,500 feet, the outflow valve will be driven fully closed by DC power from the battery transfer bus.

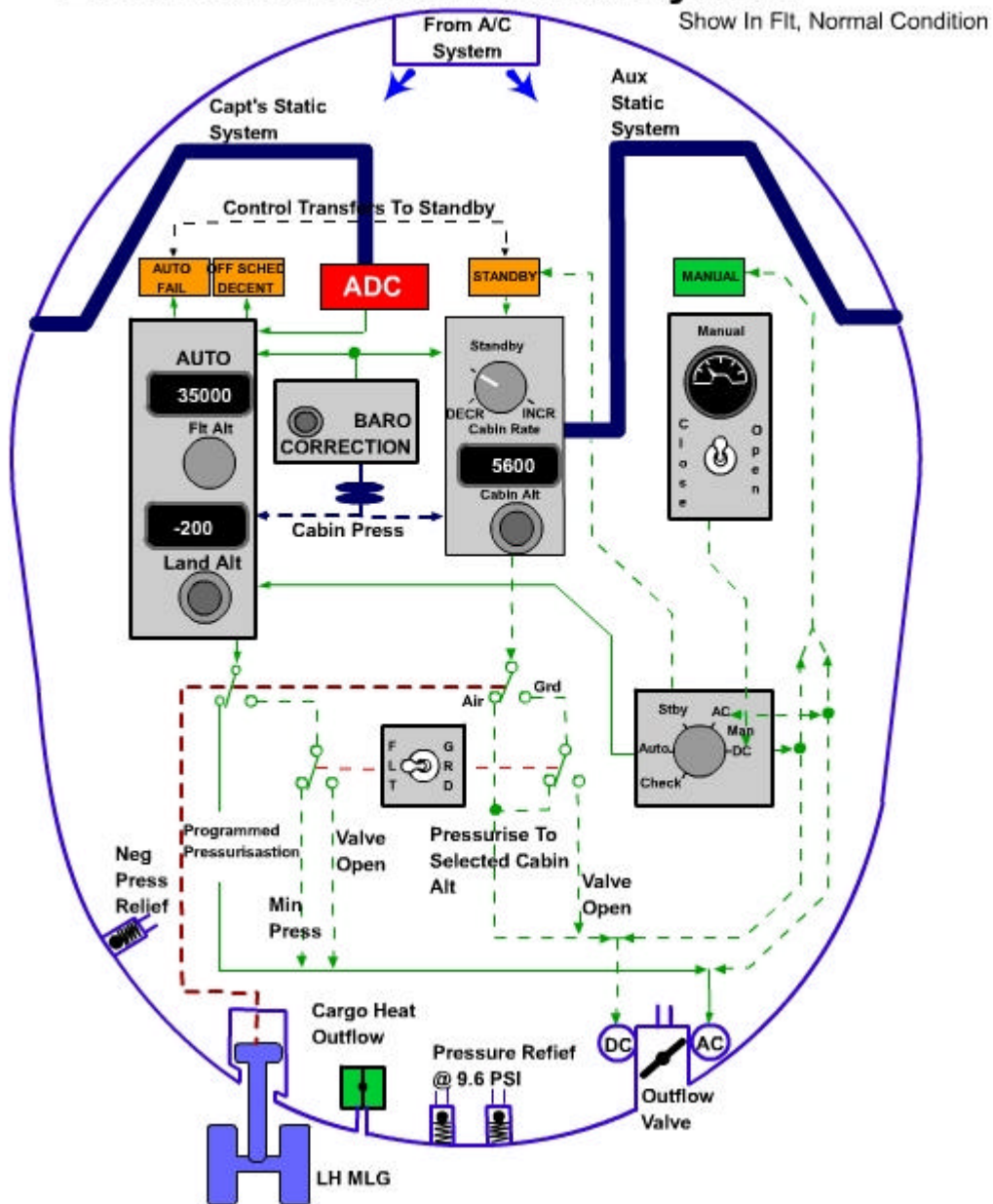
MANUAL CONTROL

The green Manual light will illuminate when either MANUAL AC or MANUAL DC is selected on the mode selector. For use in checking the outflow valve, a valve position indicator is provided. This indicator functions in all modes. The CLOSE/OFF/OPEN toggle switch below this indicator will move the outflow valve towards the position selected and thereby control the pressurisation system. Electrical power to the DC Manual mode should always be available from the battery transfer bus. Manual AC, powered by the Essential AC bus, operates much faster than Manual DC and should be used during a rapid depressurisation. Manual DC should be used during an AC power failure, as this will be the only operational mode at this time. I personally prefer to use DC when in manual as it gives a much smoother control of the pressurisation



System Descriptions

Electronic Pressurisation System

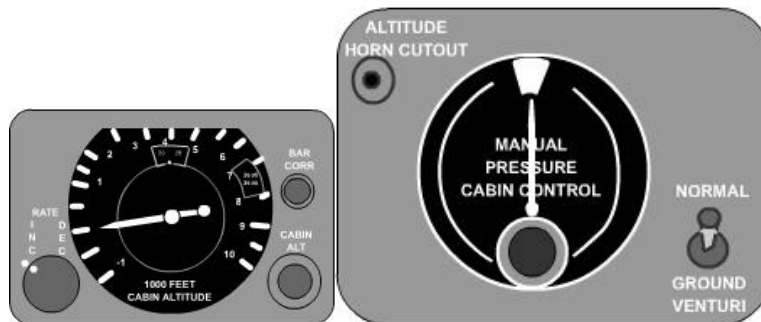


Pneumatic Control System

It is a little long in the tooth now, but it is still around and in use on a great number of aircraft, mostly 100's. Affectionately know as "steam driven". There are two control panels, again at the flight engineers panel. One for automatic control and one for manual mode. You set these by markings on the instrument and it is then entirely controlled by sensed pressures and venturi's. It's basic, but robust, though pressure bumps are quite a common feature of this system.



System Descriptions



Pneumatic Control Panels

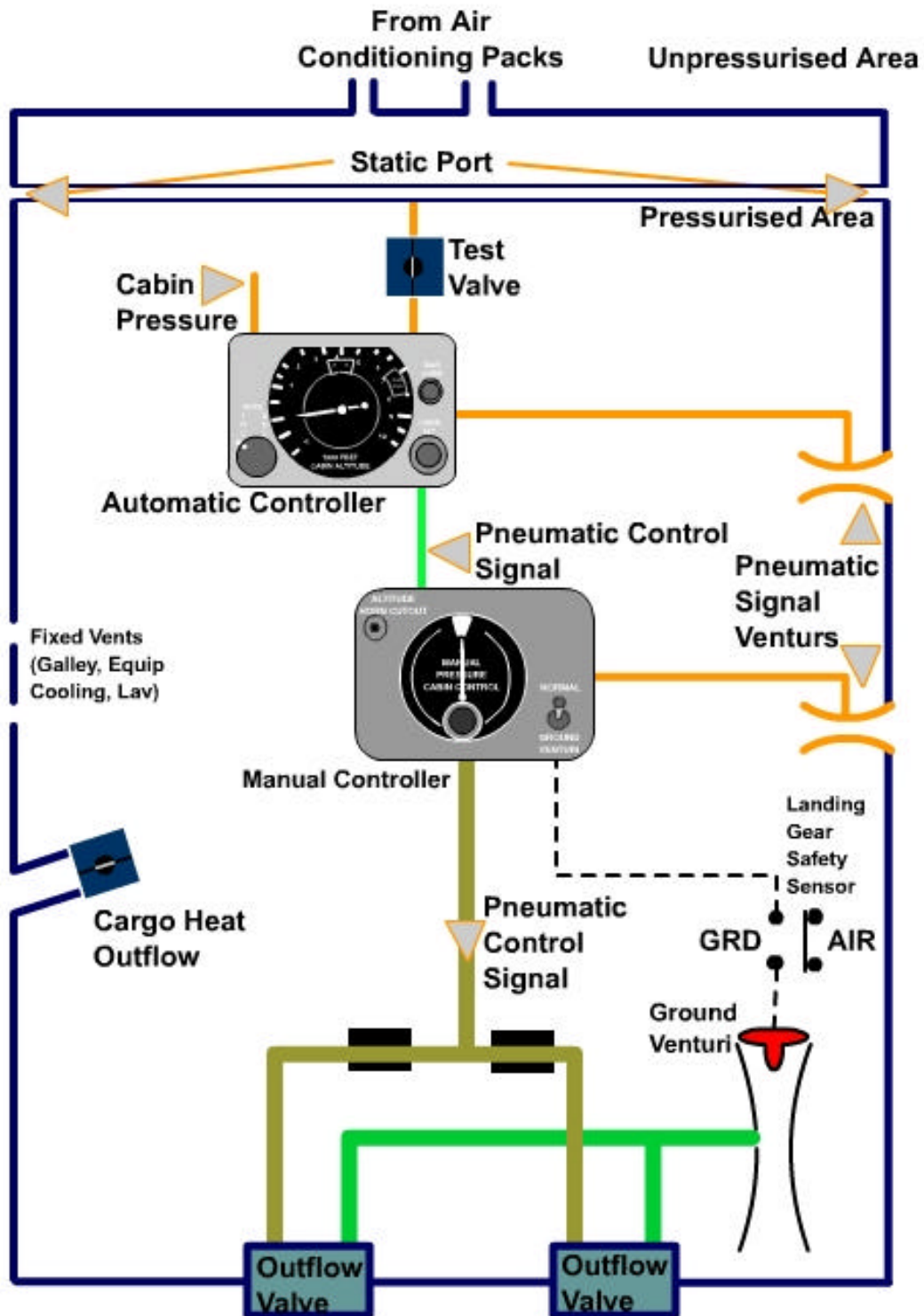
On the Auto controller you also have a rate control knob to increase or decrease the rate of change. Cabin altitude selector, with which you select the desired altitude from the instrument markings. Finally a barometric setting control knob with which you set the local pressure datum.

On the manual controller you have a knob with which you can control the outflow valves, either increase or decrease pressure by moving it clockwise or anti clockwise respectively.

Note that the valves are operated by sensed pressures and no physical link, unlike the manual mode on the electronic system, which is controlled by electric actuators. The manual controller will override the auto controller, also on the same panel is the ground venturi blower switch which is used only on the ground and performs the same function as the ground / flight switch of the electronic system. On the top left of is panel is the altitude horn cutout switch which will silence the altitude warning horn.



System Descriptions



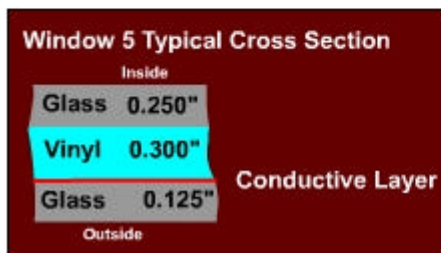
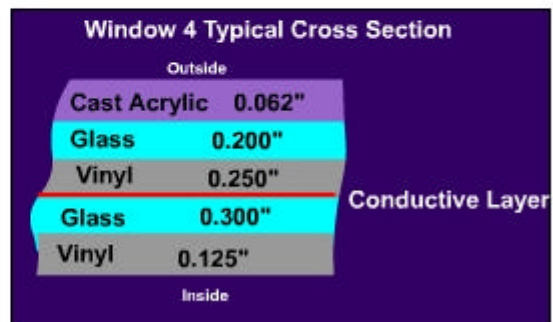
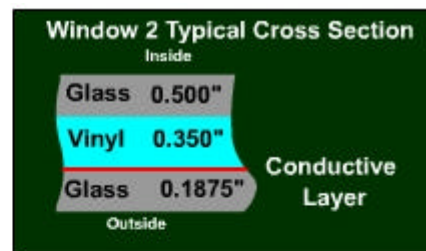
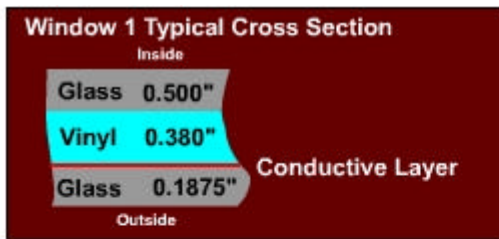
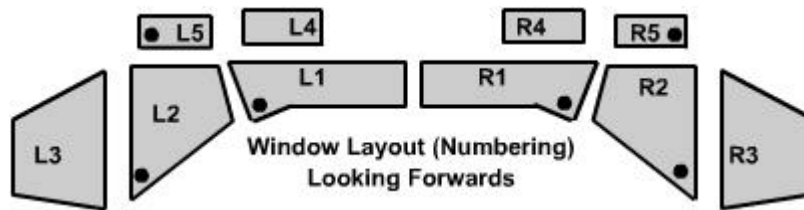
When Manual mode selected, Manual controller overrides the auto controller signals



System Descriptions WINDOWS

Windows are numbered 1 to 5 (identified as LH or RH)

Windows are heated to improve their impact resistance and to help prevent misting/fogging. They are turned on at least 10 min's before departure to allow them to warm up. There are two settings Low and High. Low is first used to prevent any thermal shock to the window and allow a gentler warming up, and then during the taxi out they are placed in the high position for flight for normal operations.





System Descriptions

Window Failure

Window failure does not compromise the integrity of the flight, it just changes your operating procedures and limitations. I've had windows fail at altitude the only thing you do is spill your coffee, as it goes bang.

PANE	WINDOW	ACTION
Outer Pane Operations	All Windows	Normal
Middle Panes psi (This is the only window having a middle pane)	Window 4	Max Cabin Diff 5
Inner Pane psi	Window 1, 2, 5	Max Cabin Diff 5
Inner Pane Operations	Window 3, 4	Normal
Both Panes psi	Windows 1, 2, 4, 5	Max Cabin Diff 2
Both Panes psi	Window 3	Max Cabin Diff 0

Max Airspeed with Window Heat Inoperative is 250 kts Below 10,000'

YAW DAMPERS

Yaw Damping is provided by two completely independent dampers, which can be operated singly or simultaneously. The prime function of the system is to minimise Dutch Roll by providing automatic rudder displacement proportional to and opposing the amount of yaw experienced.

One yaw damper controls the upper rudder the other the lower. Each yaw damper has an associated coupler, which operates a rate gyro, and senses yaw. The damper system then provides the necessary rudder movement to oppose and damp out the yaw. Rudder displacement resulting from yaw damper input is limited to 5 deg to prevent full rudder being applied in the event of a yaw damper malfunction.

The lower yaw damper is powered electrically from the Essential Radio Bus; upper yaw damper is powered from the No 2 Radio Bus.

Two yaw damper warning flags on the rudder and elevator position indicator are biased out of view when the respective yaw damper is engaged. Some aircraft have green lights instead of the warning flags; lights are green when the damper is engaged.

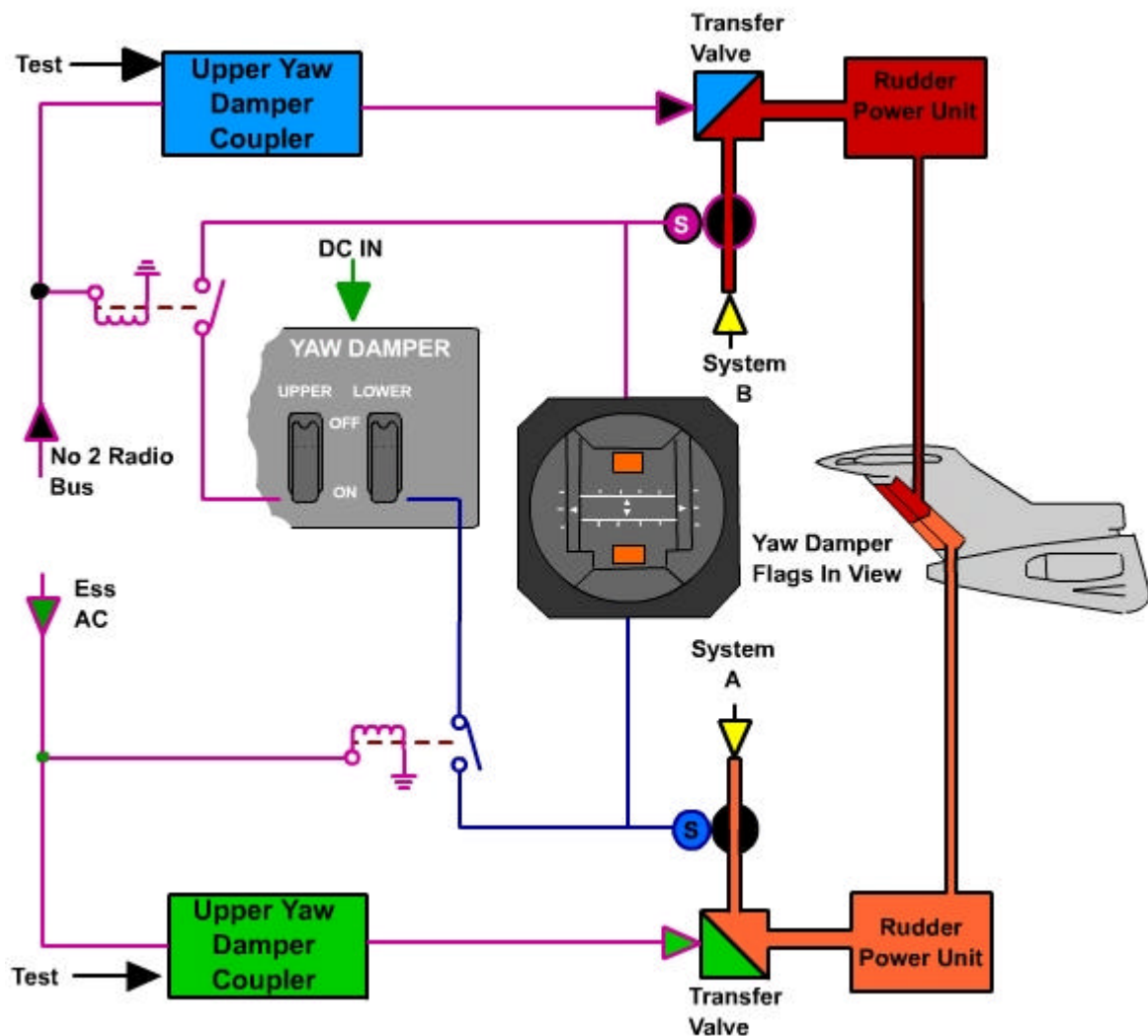


System Descriptions

Each yaw damper controls its associated rudder through a transfer valve on the rudder power unit. The upper damper uses system B, the lower damper system A.

The loss of either hydraulic system pressure will result in the loss of the associated yaw damper. If this occurs a loss damper disengaged warning will **NOT** occur.

The only common circuitry between the yaw dampers and the autopilot is an interlock that requires at least one yaw damper to be on in flight.



*Note * Not all aircraft have a test facility on the yaw damper system