FLIGHT CONTROLS

GENERAL

The Citation Sovereign flight control system is predominately manual with hydraulically operated speed brakes and roll spoilers. Flaps are electrically operated. Lateral control of the airplane is provided by the aileron and roll spoiler systems. The roll spoilers operate in conjunction with the ailerons to augment their roll control function. To optimize roll control for all flight conditions an electrical actuator automatically varies the aileron gearing. Pitch control is provided by the use of an electrically actuated movable horizontal stabilizer and mechanically actuated elevator. A single manually operated rudder provides yaw control. There is a pneumatically driven Rudder Bias system that will assist during single-engine operation. All manual flight control systems are interconnected to allow for pilot and copilot control inputs to be used in all flight conditions.

Trim is available for all three axes via trim tabs on the ailerons and rudder and a moveable horizontal stabilizer. The TRIM portion of the EICAS system displays the trim position of each individual trim surface relative to center.

The flaps are electrically powered and are composed of six segments (three per wing) that move simultaneously. They are driven from a DC power drive unit (PDU) through flexible shaft drive. Detents for the flap handle are provided for the 0°, 7°, 15°, and 35° positions and a stop detent is installed at the 35° position.

AILERONS AND ROLL SPOILERS

The airplane is controlled on the lateral axis by a manually operated aileron system augmented by hydraulically actuated roll spoilers. The ailerons and roll control spoilers each have an independent control system. The aileron control system is commanded by the pilot's control wheel, while the roll spoilers are commanded by the copilot's control wheel. While the systems are independent of each other they are interconnected at the roll disconnect system allowing the pilot or copilot to operate both systems.

Primary aileron control is available from a conventional control column and control wheel for the pilot. The aileron control system uses mechanical flight control components (i.e. cables, pulleys and associated linkages). A cable running from the pilot's control wheel to each aileron wing sector provides the output to the aileron control surfaces.

Maximum control wheel travel is ±70°, maximum aileron travel is full down deflection of 13.5° or the full up deflection of 18.6°. An automatic ratio changer is incorporated into the system to vary the aileron surface travel for a given control wheel input. Based on altitude and airspeed information an electrical actuator varies the gearing ratio between the control wheel and the ailerons automatically. At lower airspeeds (i.e. approach and landing) full control wheel travel results in full aileron deflection. At higher airspeeds and altitudes the gear ratio is reduced to increase the pilots mechanical advantage and improve roll response.

The roll spoilers are commanded by the copilot's control wheel and column through a mixer box. A cable system driven by the copilot's control wheel provides command inputs to the hydraulic roll spoiler actuator resulting in deflection to the appropriate roll spoiler panels. Roll spoiler panels 1, 5 and 6, 10 are non-modulated speed brakes. Panels 2,3,4 and 7,8,9 are modulated to perform the task of assisting the aileron system for lateral control functions. The amount of roll spoiler deflection is directly proportional to the movement of the pilot or copilot's control wheel.

ROLL SPOILER MIXER BOX

The roll spoiler mixer box uses a system of mechanical linkages and cams to transmit pilot roll input to the control arm of the roll spoiler hydraulic actuator. The cam profiles provide outputs to the spoiler actuators based on a summation of roll and speed brake control inputs.

AILERON TRIM

The aileron trim switch located on the center pedestal electrically controls the aileron system trim. Upon activation of the switch, a trim actuator moves the LH aileron trim tab in the selected direction. Both LH and RH trim tabs are servo tabs and will deflect proportionally to aileron deflection, while the LH tab can undergo a further pilot commanded travel of ±10°. The aileron trim actuator is an irreversible DC powered unit with an RVDT for position sensing.

RUDDER SYSTEM

GENERAL

Conventional rudder pedals provide primary yaw control for each pilot. These pedals are mechanically interconnected so the pilot and copilot pedals operate in unison. The primary rudder surface is mass balanced.

The rudder system is geared to allow ±30.5° trailing edge left and right travel in a no-load condition at a maximum pedal deflection of ±21°. Rudder stops are installed to limit rudder travel to the maximum trailing edge value. The rudder stops are conventional adjustable bolts. These stops act on the rudder torque tube.

RUDDER BIAS

The rudder system is equipped with a pneumatically powered rudder bias system. Rudder bias main purpose is to compensate for an asymmetric thrust condition. The rudder bias system is driven by a dual pneumatic actuator, which is equally supplied by dedicated bias bleed air from both the left and right engines. With both engines producing the same thrust, the bias system has no effect on rudder control. During asymmetric thrust conditions, the rudder bias system applies a "bias" of the rudder toward the side of the higher thrust engine. An electrical actuator in the system, automatically varies the rudder bias authority based on airspeed and altitude. The electrical actuator will automatically retract (minimizing bias authority) during thrust reverser deployment.

A rudder bias override control switch-annunciator is located to the left of the throttle quadrant on the center pedestal. In the NORM position, the rudder bias system functions to assist the pilot in an asymmetric thrust condition. Pressing the switch-annunciator to the O'RIDE position retracts the electric variable bias actuator providing minimum rudder bias output.

RUDDER SYSTEM

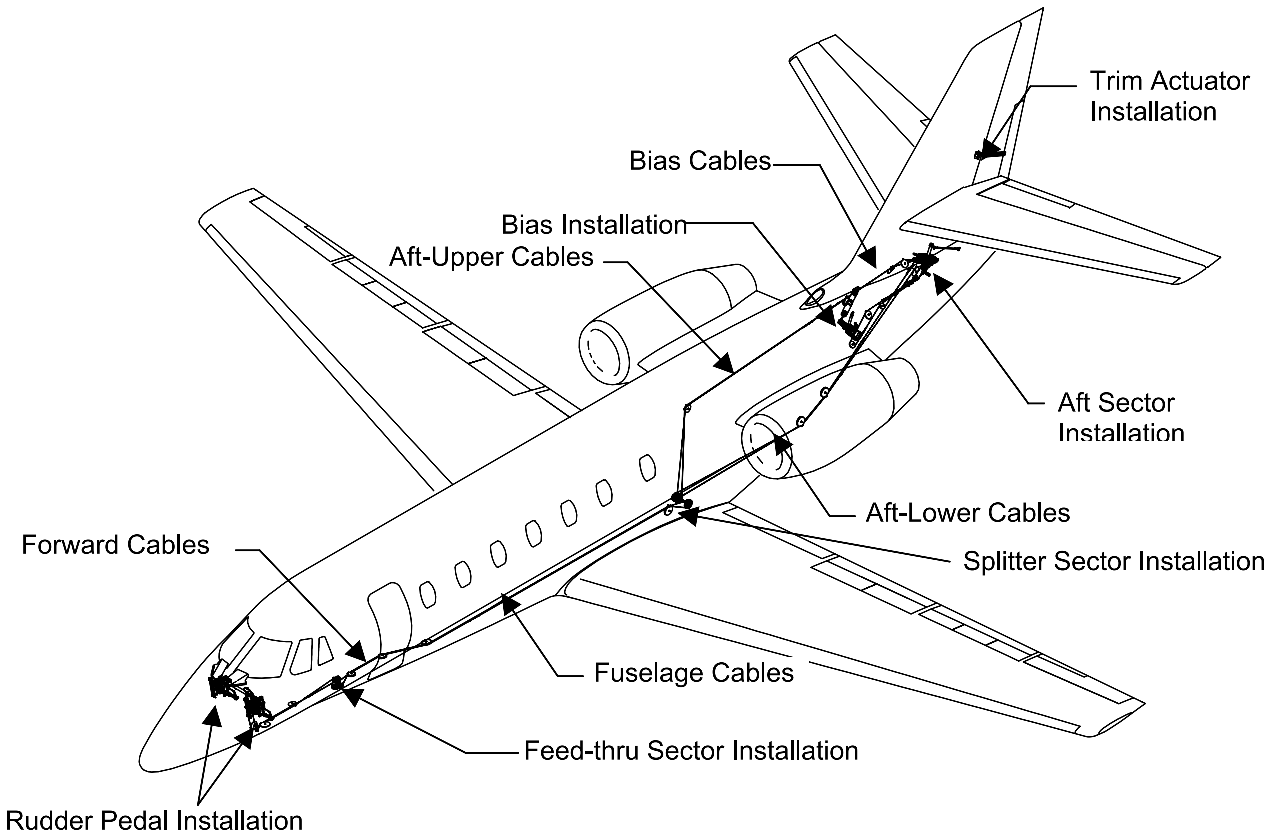

Figure 2-15

RUDDER BIAS SYSTEM

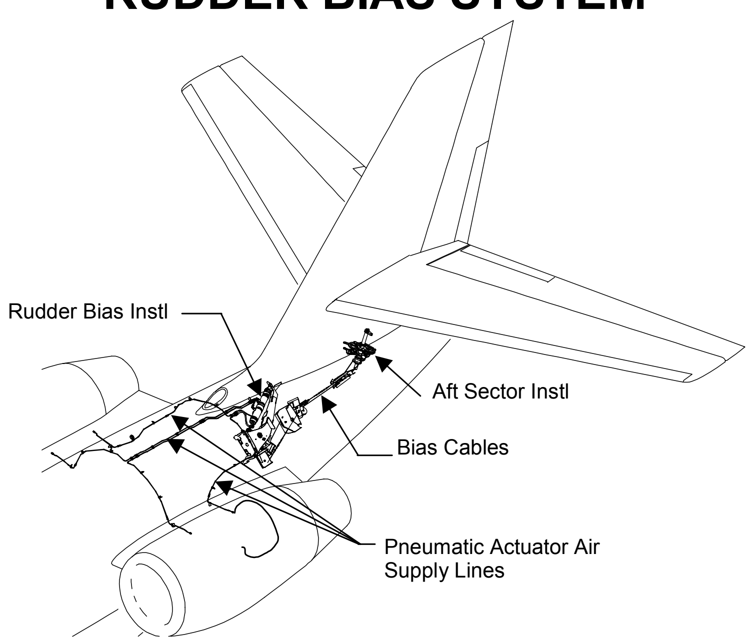

Figure 2-16

Icing protection is provided to the pneumatic cylinder by the use of service air that is used to heat the external shroud.

The Pneumatic Bias Actuator is equipped with a break away clevis which acts as a "load fuse." Should the Pneumatic Actuator jam when the variable bias actuator is retracted, the clevis will become loose on the shaft to prevent large loads from being transmitted into the support structure from rudder pedal inputs.

RUDDER TRIM

The rudder trim system is comprised the electric trim knob in the cockpit, an electrical trim actuator, mechanical linkages, and the rudder trim tab.

The RUDDER TRIM knob located on the aft end of the center pedestal actuates the rudder trim electrically. To activate the trim knob it must be depressed before trim adjustment can be made. Rotation of the trim knob left or right moves the rudder in the respective direction. Pilot commanded rudder trim tab travel is ±3°. The rudder trim tab is a servo tab where the tab will deflect proportionally to rudder travel in a manner, which reduces pilot effort.

The rudder trim actuator is an irreversible DC brushed motor. It is similar to the aileron trim actuator using a RVDT for position sensing.

NOSEWHEEL STEERING SYSTEM

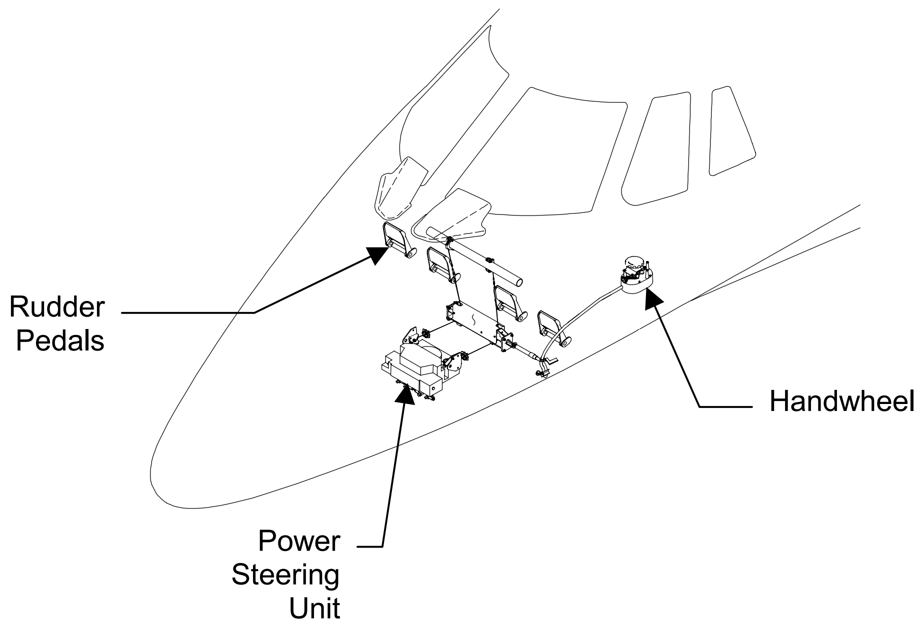

Figure 2-17

NOSEWHEEL STEERING

GENERAL

Nose wheel steering is provided through the use of the pilot or copilot pedals or a pilot's console mounted hand wheel operated gearbox. Steering inputs from the rudder pedals will allow a nosewheel travel of approximately 7° either side of center. Use of the hand wheel operated gearbox will allow a nosewheel deflection of approximately 81° either side of center. Hand wheel centering is accomplished by a spring-loaded cam mechanism attached directly to the gearbox. The hand wheel and rudder pedals may be used in conjunction to provide approximately 88° left or right deflection of the nosewheel when necessary for ground operations.

ELEVATOR SYSTEM

GENERAL

The elevator control system is of the conventional manually operated. The primary elevator surfaces are mass balanced. Main control of the system is provided through a series of cable and pulleys connected to the pilot and copilot's control column. Control column travel through the full range delivers a maximum no load elevator deflection of 10° down and 16.5° up.

Two sets of elevator control cables are installed in the airplane. The left cable system links the pilot's control column to the left elevator surface. The right cable system links the copilot's column to the right elevator surface. The cable system is interconnected by a torque tube and pitch disconnect mechanism that links both cockpit control columns enabling synchronized pilot or copilot control inputs to manipulate elevator position. Activating the disconnect allows the pilot and copilot to independently control their respective elevator surface.

HORIZONTAL STABILIZER TRIM

The horizontal stabilizer trim system is comprised of a primary stab trim actuator, actuator control unit, pilot and copilot control wheel switches, a secondary stabilizer trim actuator, and a secondary trim ON/OFF switch-annunciator.

The primary stab trim tab actuator is installed at the base of the vertical fin, forward of the horizontal stabilizer. The horizontal stabilizer pivots at the aft attach-point of the vertical stabilizer. Horizontal stabilizer travel is limited for primary trim by electrical stops at 1.2° Trailing Edge Down (TED), 6.9° degrees Trailing Edge Up (TEU), and for secondary trim by mechanical stops at 1.7° TED to 7.4° TEU.

A takeoff range switch is incorporated into the primary trim actuator. This switch is connected to the NO TAKEOFF warning system to make sure that the horizontal stabilizer is in takeoff position prior to takeoff.

The secondary trim actuator receives power from the airplane's DC emergency bus. A guarded cover switch-annunciator located on the center pedestal allows for the secondary trim system to be activated. Pitch Trim changes for the secondary trim system are available using the NOSE UP/NOSE DOWN trim control next to the secondary trim power switch. With the secondary pitch trim in the ON position, the primary trim actuator becomes disengaged. Activation of the secondary trim also removes power from the control wheel pitch adjustment switches and disengages the autopilot.

An audio clacker signal will be heard over the cockpit headsets and speakers under three sets of circumstances. The clacker will be heard when trim is commanded by the autopilot for more than 1 second. The clacker will also be heard when secondary trim is commanded for more than 1 second. Finally, the clacker will be heard when uncommanded trim actuator movement is detected for more than 1 second. The clacker will not be heard when trim is commanded from the split control wheel switches.

ELEVATOR TRIM

Two elevator tabs are installed on the airplane. One is located on the trailing edge of each elevator. The primary purpose of the elevator trim tabs is to maintain a faired position of the elevator relative to the horizontal stabilizer as the angle of incidence changes. Tab travel is varied automatically through a series of mechanical linkages tied to the movement of the horizontal stabilizer. Each elevator trim tab has a travel range of 2.0 to 11.9° trailing edge down relative to the elevator surface.

MACH TRIM

Mach trim is incorporated into the elevator tab system for improved pitch performance at higher mach numbers. At low mach, the system remains dormant. At higher mach numbers, the mach trim system is activated based on a preprogrammed schedule with no action required by the flight crew.

PITCH/ROLL DISCONNECT

A T-handle located at the aft end of the center pedestal allows the flight crew to disconnect or reconnect the ailerons and roll spoilers as well as the elevators. Pulling the T-Handle to disconnect will cause the ailerons and roll spoilers to be operated independently. The elevator systems will also be disconnected at the same time. The aileron system will be controlled by the pilot and roll spoilers by the copilot. Pitch inputs by the pilot or copilot will operate independent of each other after disconnecting the pitch/roll system. The pilot will control the left elevator and copilot control inputs are to the right.

Rotating the T-handle 90° clockwise from the NORM position to ROLL RECONNECT may reconnect the roll spoilers and ailerons. Moving the handle 90° counterclockwise from NORM to PITCH RECONNECT will cause the elevator to re-engage.

The pitch-disconnect system isolates the pilot and copilot side elevator control systems. Once the pitch/roll disconnect T-Handle is pulled, the pitch disconnect mechanism linking the pilot and copilot side control column torque tubes disengages the link of the system. In order to re-synchronize the left and right elevator control cable systems, the T-Handle is rotated 90° counterclockwise to the PITCH RECONNECT position.

To reset the system, rotate the T-handle 180° in either direction to the PITCH/ROLL RECONNECT PUSH-RESET position. Push the handle down and rotate to NORM to rearm the pitch/roll disconnect mechanisms.

GUST LOCK

GENERAL

The gust lock system is designed to protect the elevator, aileron, and rudder flight control systems from damage due to wind gusts. The gust lock system consists of three lock mechanisms, one for each primary flight control system. It is a "load" and "lock" system actuated by a T-handle located on the pilot's side of the pedestal, below the tilt panel and immediately outboard of the control column. The three lock mechanisms are simultaneously loaded when the T-handle is pulled, causing the mechanisms to move towards the locked position. A cyan "GUST LOCK ON" message is displayed on EICAS when any one mechanism moves away from the stowed position, but is not necessarily locked. Each mechanism then independently locks its corresponding flight control system when the flight control system is moved to the center position. When the T-handle is pulled and the throttles are in the takeoff position, the red "MASTER WARNING" indicator and the red "NO TAKEOFF" messages are triggered. Activation of the gust lock will restrict the RH throttle lever to approximately 20% thrust.

Gust lock control consists of a push/pull cable and T-handle. The gust lock control is unlocked when the T-handle is rotated 90° CCW to the vertical position and locked when the T-handle is rotated 90° CW to a horizontal position. The gust lock control must be unlocked prior to being pushed or pulled, and must then be locked after movement to the desired position. Loading the gust lock system requires pulling the T-handle all the way out (approximately 3") with a force of 25 lb or less. Less effort is required to disengage the gust lock system, as each mechanism is spring loaded to return to the disengaged position. All three lock mechanisms have a positive disengagement feature that disengages all three gust locks as the T-Handle is returned to the stowed position. If any of the mechanisms becomes jammed or cannot return to the stowed position, it will not be possible to fully stow the T-Handle.

The gust lock control is attached to the elevator shaft assembly which, when rotated, loads the elevator gust lock and moves a pushrod assembly that rotates the aileron shaft assembly. The aileron shaft assembly, when rotated, loads the aileron gust lock and pulls the rudder cable. The rudder cable, when pulled, loads the rudder gust lock.

FLAPS

GENERAL

The Citation Sovereign trailing edge flaps are mechanically controlled and driven by a single DC electric Power Drive Unit (PDU). Three flap panels are installed on the trailing edge of each wing. A common torque shaft system from the PDU drives pairs of mechanical actuators on all flap panels. A flap handle located on the center pedestal is used to select flap position. Monitoring of the flap system is performed electronically and is displayed on the Engine Indicating and Crew Alerting System (EICAS) display.

The flap control lever, located on the center pedestal, provides a mean of extension and retraction of the flaps. Four flap detent positions are marked on the left and right, side of the control handle. The positions are 0°, 7°, 15°, and 35°. Flap position is electronically monitored and displayed in the FLAP portion of the EICAS display.

In the event of a flap system malfunction, the system will automatically be, shut down. Warning messages relating to the cause of the flap system malfunction are displayed as appropriate on the EICAS display.

A FLAP RESET switch-annunciator is located immediately forward of the flap control handle; if a condition which has immobilized the flaps has cleared, the flaps will regain operation upon the pressing of the FLAP RESET button. If the condition has not cleared the flaps will remain inoperative. If the controller senses a fault and disables the flaps, an amber FLAPS FAIL message will appear in the EICAS display.

SPEED BRAKES

GENERAL

There are two dedicated speed brake panels installed on each wing of the Citation Sovereign. Three additional panels on each wing serve a dual role of roll spoilers and speed brakes. The speed brake system is hydraulically actuated through the use of Power Control Units (PCUS) combined with a mixer assembly that combines inputs from the copilot's wheel and the speed brake lever located on the center pedestal in the cockpit.

The speed brake control lever moves aft to deploy the speed brakes and forward to stow. Total travel of the lever is 57° for deployment of all ten speed brake panels. The first 47° of lever input modulate the roll spoiler/speed brake panels (2, 3, 4, 7, 8, and 9). A detent is reached at the 47° position of the lever, moving past this detent will allow deployment of the dedicated speed brake panels (1, 5, 6 and 10) to the full non-modulated deflection angle. Panels 1, 2, 3, 8, 9 and 10 are fully extended at 35°. Full travel for panels 5, 6, 7, and 8 is 24°.

FLIGHT CONTROLS

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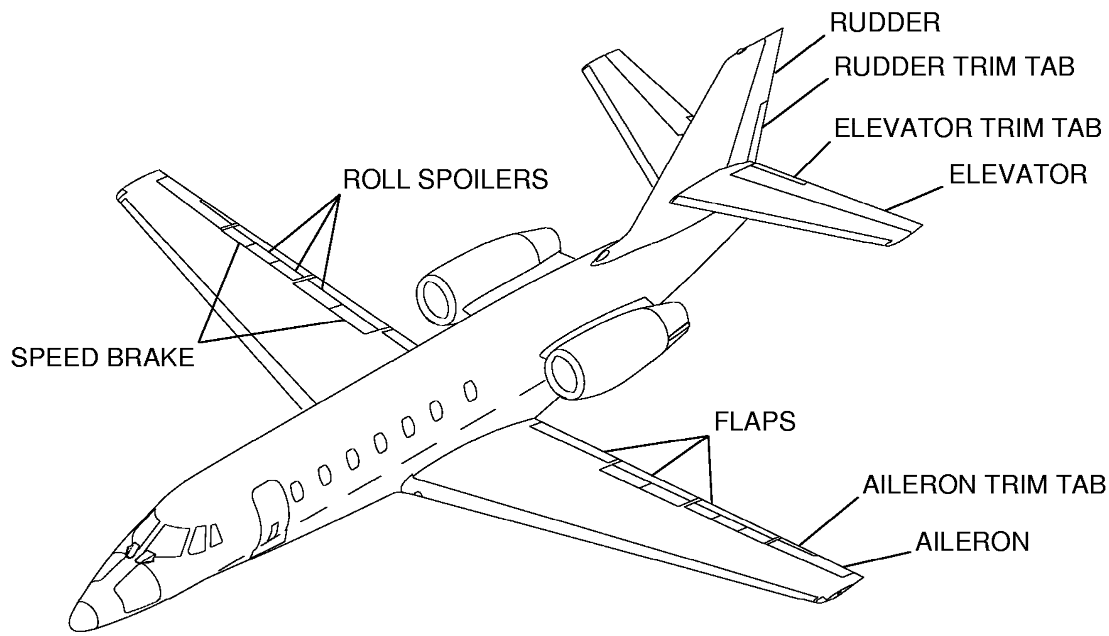

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Figure 2-18

Deployment of the speed brakes is displayed on the EICAS display as white bars corresponding to the ten separate speed brake panels. When on the ground the modulated panel will show deployment for speed brake system response as well as roll system response. In the air, with no weight on wheels, the modulated panels indicated will only show speed brake system response. The non-modulated panels will show an EICAS indication if the airplane is on the ground or in the air.

STALL WARNING - STICK SHAKER

GENERAL

Stall warning is provided by two independent Digital Stall Warning Computers, column shakers installed on the pilot and copilot's control columns, information displayed on the PFDS, and angle-of-attack (AOA) indexer lights. The main electrical busses provide electrical power for the AOA and Stall Warning Computers.

An AOA segment, located on the left hand side of the PFDS, displays the angle-of-attack of the airplane by percentage ranging from 0 to 1. A Low Speed Awareness (LSA) indication will also be displayed on the inside of the airspeed indicator tape in red. LSA corresponds directly to angle-of-attack.

AOA vanes are installed on the left and right side of the forward section of the fuselage. Each vane is heated by an electrically operated heating element for icing protection.

Stick shakers, installed on each cockpit control column, receive stall warning information from the warning computers. Each computer drives the respective stick shaker mechanism to warn the flight crew of a possible stall condition. An electric motor with rotating weights induces a vibration feel to the control column, alerting the pilots to the approaching stall.

An amber AOA HEAT FAIL L-R CAS message indicates failure of the applicable AOA probe heater. In icing conditions, information from that system should be disregarded or used with caution. An amber AOA/STALL WARN FAIL L-R CAS message indicates a system failure.

The rotary TEST switch on the center pedestal provides a means of checking the stick shakers and other features of the angle-of-attack system on preflight. To test the system select AOA on the test switch and observe following:

1. The three lights in the AOA indexer shall illuminate one at a time during the test sequence.
2. The AOA indicator shall slew to the zero end of the scale.
3. The AOA indicator OFF flag shall be displayed as well as the following CAS messages:
    a. AOA/STALL WARN FAIL L-R
4. After a few seconds, the AOA OFF flag shall be removed and the indicator will move up scale.
5. As the pointer approaches 0.60, the INDEXER lights (if installed), shall cycle from yellow to green to red.
6. At approximately 0.80 normalized AOA, the stick shaker should actuate. Vibration from left stick shaker should be felt through both control columns.
7. System should return to normal operation after a few seconds.