SECTION III

AVIONICS

SECTION III AVIONICS

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Chapter 1

INSTRUMENT PANELS

Introduction

To correctly determine how well an aircraft system is operating, you must be able to interpret instrument presentations. If a malfunction is indicated, you must determine if the information presented is an actual condition of the aircraft or incorrect instrument information.

An erroneous instrument reading could cause you to make a faulty decision on the best way to cope with a problem but, on occasions, true instrument readings could be contrary to your natural instincts. You are the determining factor. Abnormal instrument indications may be caused by a defective instrument or aircraft system. You can frequently recognize an abnormal instrument indication by "cross reference" with other instruments. Whenever possible, you should not base a conclusion on any one instrument. Instead, you should crosscheck other related instruments before any decision about system conditions is made. Each instrument displays only part of a much larger picture.

Instrument Panels

The flight deck has three major panels -- the main instrument panel, the engineer's instrument panel and the navigator's instrument panel.

In addition, some instrument testing and adjusting controls are located on the pilot's and copilot's side consoles.

The main instrument panel is divided into three sections. The left and right sides each contain a full set of flight instruments. The center section, containing engine and position instruments, is used by both pilots.

The engineer's instrument panel has an upper and lower section. The upper section has 12 subpanels for instruments and controls. The lower section has an engine instrument subpanel and a fuel system subpanel.

The navigator's instrument panel has several subpanels. Most of these subpanels are devoted to various navigational aids. However, three instruments covered in this section are mounted here. These are an altimeter, the true airspeed indicator and a clock.

Panel and Instrument Lighting

All instrument and panel lighting is controlled by toggle switches and rotary switches mounted on the overhead panels. The instruments are integrally lighted, using AC power provided by lighting stepdown transformers located behind each rotary switch. The lighting transformers are powered by the Essential AC Buses.

Instrument Types

All instruments are front mounted. Either a clamp-type or a bezel-type mounting is used.

The C-141 has two types of instruments. Most are the conventional dial-type indicators. However, the C-141 also has vertical scale tape indicators (VSI). On these indicators, a tape, moving vertically past an index, shows the desired indication.

Instrument Markings

On some instruments, limit and range markings are provided. The conventional colors are used -- red for limits, green for normal operation and yellow for caution. On instruments with range markings, a white index marker, partly on the cover glass and partly on the instrument case, indicates if the cover glass has slipped.

Chapter 2

INSTRUMENT SYSTEMS

115-Volt AC Instruments

APU EGT Indicator

The APU exhaust gas temperature (EGT) indicator is located on the engineer's panel. The thermocouple-type sensor is located in the APU exhaust. Power for this indicator comes from Essential AC Bus No. 2. A power-off flag in the center of the instrument is visible anytime power is not present at the indicator.

Oxygen Quantity Indicators

A 25-liter liquid oxygen quantity indicator is located on the copilot's side console. This is a capacitance-type indicator in which the indication is not affected by changes in density due to temperature variations. This indicator receives 115-volt AC power from the Essential AC Bus No. 2.

Two 75-liter liquid oxygen quantity indicators form part of the troop oxygen system. They are located on the troop oxygen panel at station 688. These two indicators are also of the capacitance type. Power for these indicators is obtained from two separate buses. Indicator No. 1 is supplied by Essential AC Bus No. 1, and Indicator No. 2 is supplied by Essential AC Bus No. 2.

All three liquid oxygen quantity indicators have a push-to-test button located alongside the indicator. Depressing the test button causes the indicator to move counterclockwise, as an operational check. If the indicator is driven far enough towards empty (2.5 liters crew system - 7.5 liters troop system), the LOW OXYGEN QUANTITY WARNING LIGHT will illuminate.

Fuel Quantity System

The fuel quantity system provides both individual tank readings and total fuel quantity. There are ten tank indicators and one total fuel indicator. All indicators are located on the engineer's fuel management panel.

The fuel quantity indicators use capacitor-type tank units and therefore measure fuel quantity in pounds rather than in gallons.

Each tank system contains the tank units, a density compensator and an indicator.

The system is essentially a bridge circuit which is unbalanced by the addition or subtraction of fuel in the tanks. Signals created by this unbalance are amplified within the indicator to power the indicator motor. As the indicating pointer reaches the proper fuel quantity reading, the bridge is again balanced and motion stops until the next fuel level change.

The total fuel quantity indicator is connected to a potentiometer in each of the tank indicators. Whenever the tank indicator motor responds to a change in fuel level, it also changes the potentiometer value, which in turn results in a change in total fuel quantity.

NOTE

> If a tank indicator should become inoperative, the value of that potentiometer is still in the totalizer circuit and is affecting the total fuel quantity reading.

Power requirements for the fuel quantity indicating systems are 115-volt, 400-hertz, single-phase AC power. The main tank indicators are supplied AC power from the Essential AC Bus No. 1 through four circuit breakers on the engineer's circuit breaker panel. The totalizer, auxiliary tanks No. 1 and No. 4, and LH extended range indicators are supplied by the Main AC Bus No. 1 through circuit breakers on the engineer's circuit breaker panel.

The No. 2 and No. 3 auxiliary tanks and the RH extended range indicators are supplied by Main AC Bus No. 4 through three circuit breakers on the engineer's circuit breaker panel.

A push-to-test button is located next to each fuel quantity indicator, except the total fuel indicator. Depressing the switch causes the indicator to drive towards zero. Each fuel quantity push-to-test button will also cause the fuel totalizer to rotate towards zero. When the switch is released, the fuel quantity indicator and the fuel totalizer will return to the proper fuel quantity reading. This is an operational check and not an accuracy check.

26-Volt AC Instruments

Fuel Pressure Indicator

There is one fuel pressure indicator on the engineer's fuel management panel. The transmitter for this indicator is located in the center wing dry bay area. The fuel pressure indicator can be used to ground check the output of individual fuel booster pumps by properly positioning the fuel crossfeed and separation valves. The fuel pressure indicator is also used to monitor fuel pressure during ground and air refueling.

Power for this indicator is supplied by 26-volt AC Bus No. 2.

Oil Pressure Indicators

The four oil pressure indicators are located on the engineer's engine instrument panel. The pressure transmitters are mounted on the main accessory section of each engine. Power for No. 1 and No. 4 indicators is supplied by 26-volt AC Bus No. 1, and for No. 2 and No. 3 indicators by 26-volt AC Bus No. 2.

Brake Pressure Indicators

Two brake pressure indicators are located on the pilots' center instrument panel. One indicator shows normal brake system pressure, and the other shows emergency brake system pressure. Operation of these indicators is dependent on the position of the brake selector switch. The proper indicator will show the pressure of the system selected.

The transmitters for these two indicators are located under the floor near the control columns.

The normal brake pressure indicator is powered by 26-volt AC Bus No. 1, and the emergency brake pressure indicator by 26-volt AC Bus No. 2.

Hydraulic Pressure Indicators

There are three hydraulic pressure indicators on the engineer's panel. They show Hydraulic System No. 1, No. 2 and No. 3 pressures. The transmitters for these indicators are located in their respective hydraulic system service centers.

The No. 1 and No. 3 system indicators are powered by 26-volt AC Bus No. 1, while the No. 2 system indicator is powered by 26-volt AC Bus No. 2.

Spoiler Position Indicator

The spoiler position indicator is a dual-pointer indicator located on the pilots' center instrument panel. The "L" and "R" pointers show the CLOSED and GRD positions of their respective wing spoilers. The transmitters are in the wings and are driven by the inboard spoiler drive tubes.

In the center of the spoiler position indicator is a window that shows either an UNLKD or LOCKED indication. This portion of the indicator is operated by the spoiler actuator limit switches.

Power for the spoiler position indicator pointers comes from the 26-volt AC Bus No. 1. Power for the position window comes from the Main 28-volt DC Bus No. 1.

Manifold Bleed Air Pressure Indicator

The manifold bleed pressure indicator is located on the engineer's environmental panel. The transmitter for this indicator is located in the wing manifold at a tee connection, and it receives pressure from both sides of the isolation valve.

The power for this indicator comes from the 26-volt AC Bus No. 2.

Regulated Bleed Air Pressure Indicator

The regulated bleed air pressure indicator is a dual pointer indicator. The "L" and "R" pointers show the pressure within their respective air-conditioning systems. The indicator is located on the engineer's panel. The pressure probes are located in the primary heat exchanger outlet air ducting.

Power for this indicator is supplied by 26-volt AC Bus No. 1 ("L") and 26-volt AC Bus No. 2 ("R").

28-Volt DC Instruments

Engine Oil Temperature Indicators

The four engine oil temperature indicators are located on the engineer's engine instrument panel. The temperature sensors are located downstream of the oil filter in the pressure passage. They sense the temperature of the oil going to the engine bearings.

The DC power for No. 1 and No. 4 indicators is supplied by Main DC Bus No. 1, while No. 2 and No. 3 indicators are powered by Main DC Bus No. 2.

CSD Oil Temperature Indicators

The four constant speed drive (CSD) oil temperature indicators are located on the engineer's panel. The temperature sensors measure the temperature of the oil as it leaves the CSD.

Power for No. 1 and No. 4 indicators is supplied by Main DC Bus No. 1, and for No. 2 and 3 indicators by Main DC Bus No. 2.

The engine oil temperature indicators and the CSD oil temperature indicators share the same circuit breakers on the engineer's circuit breaker panel.

Fuel Temperature Indicator

The fuel temperature indicator is located on the engineer's fuel management panel. Near the indicator is a switch marked: OUTBD and INBD. The temperature sensors are located in No. 1 engine feed line and in No. 2 engine feed line. If OUTBD is selected, the indicator shows the temperature of the fuel going to No. 1 engine, while the INBD position shows No. 2 engine fuel temperature.

Power for the fuel temperature indicator is furnished by Main DC Bus No. 1 through No. 1 engine CSD and engine oil temperature circuit breaker.

Primary Heat Exchanger Temperature Indicators

The two primary heat exchanger temperature indicators are located on the engineer's panel. They show the temperature of the left and right wing bleed air after it has passed through the heat exchangers. The temperature bulbs are located in the systems downstream from the heat exchangers.

The left-hand indicator is supplied power by the Isolated DC Bus, and the right hand indicator by the Main DC Bus No. 2. In the event of power failure on either bus, the applicable emergency pressurization switch on the emergency power circuit breaker panel can be placed to the EMERG position. Now power will be supplied to the indicators by the Emergency DC Bus.

Cargo Compartment Temperature Indicator

The cargo compartment temperature indicator is mounted on the engineer's panel. The temperature sensor is located in the cargo compartment temperature control unit installed in the aft upper deck (hayloft), wherein a fan draws cargo compartment air over it.

Power for this indicator comes from Main DC Bus No. 2.

Flap Position Indicator

The flap position indicator is located on the pilots' center instrument panel. The transmitter is driven by the flap drive gearbox. Calibration is from UP (zero percent) to DOWN (100 percent) with an increment at each 10 percent.

The wing flap indicator is supplied 28-volts of DC power by the Main DC Bus No. 1.

Landing Gear Position Indicators

The three type C-1 landing gear position indicators are located on the pilots' center instrument panel above the landing gear lever. A miniature wheel and tire indicate that the gear is DOWN; an UP marker indicates that the gear is UP; while a black and yellow striped flag indicates that the gear is not up or locked. Limit switches actuated by the gear movement operate the indicators.

Power is furnished by the Isolated DC Bus.

Bogie Position Indicators

Two type C-1 bogie position indicators are on the pilots' center instrument panel. The miniature wheel and tire indicate that the associated bogie is in position for landing. At all other times, a black and yellow striped flag is visible to warn the pilot. Limit switches, actuated by bogie beam movement, operate these indicators.

The Isolated DC Bus powers the bogie position indicators.

Trim Position Indicators

The horizontal stabilizer trim position indicator is mounted on the pilots' center instrument panel. It is calibrated in degrees of stabilizer travel for aircraft nose-up and nose-down. The transmitter is mounted in the empennage. Power is supplied by the Main DC Bus No. 1.

The aileron trim position indicator on the pilots' center instrument panel is calibrated in degrees for lower left wing and lower right wing. The transmitter is part of the aileron trim actuator. Power is furnished by Main DC Bus No. 1.

The rudder trim position indicator also is on the pilots' center instrument panel. It is calibrated in degrees for nose left and nose right. The transmitter is part of the rudder trim actuator. Power comes from the Main DC Bus No. 1.

Miscellaneous Instruments

Clocks

Four 8-day clocks are provided and are located on the pilot's, copilot's, engineer's and navigator's panels. These are spring-operated clocks. They are wound by using the knob in the lower left corner. A sweep second hand and minute totalizer are provided; both are controlled by successive depressions of the START-STOP-RESET knob in the upper right corner.

Accelerometer

The self-contained accelerometer is located on the pilot's panel. The dial is calibrated from -2 to +4 Gs. One pointer indicates continuously the vertical "G" forces on the aircraft. The other hands indicate the maximum plus and minus vertical "G" forces exerted until they are reset by the knob in the lower left corner.

Cabin Altitude and Differential Pressure Indicators

There are two of these indicators, one on the copilot's panel and one on the engineer's panel. One pointer indicates cabin pressure altitude in feet, and the other pointer indicates the pressure difference between aircraft and cabin altitudes in psi.

Cabin Rate of Climb Indicator

The cabin rate of climb indicator is located on the engineer's panel. It indicates the rate of pressure change in feet per minute as cabin altitude is moved up or down.

Direct Pressure Indicator

Each Hydraulic Service Center has a direct pressure reading indicator for system pressure. In addition, Hydraulic Service Center No. 3 has a direct pressure indicator for each of the two 400-cubic-inch accumulators and the APU surge accumulator. These indicators are mechanical instruments and require no electrical power.

Chapter 3

VERTICAL SCALE ENGINE INSTRUMENTS (VSEI)

(Figures 3-3 thru 3-5)

Introduction

There are five vertical scale engine instrument (VSEI) systems on the center instrument panel for the pilots and five on the engineer's panel. In turn, each of the five indicator systems gives indications for each of the four engines.

One engine instrument converter unit serves the following systems: EGT, N_1 and N_2 and fuel flow. This engine instrument converter unit is located in the right underdeck equipment rack. The converter contains amplifier modules, which take transmitter signals, amplify them, and then send the signals on to operate indicator servo motors. These modules are interchangeable in the same system. For example, the No. 1 EGT could be interchanged to operate the No. 2 EGT. A set of fuses on the converter protects each individual instrument presentation feeding through the converter unit. Power for the converter unit is supplied by the Essential Bus No. 2 through four circuit breakers (one per engine) labeled: ENG, EGT, RPM and FUEL FLOW.

The EPR system has four converter transmitters, one for each engine EPR indication. They are located in the base of each engine pylon. Power is provided by the Isolated AC Bus through four individual circuit breakers.

Exhaust Gas Temperature (EGT) Indicators

The four-channel EGT indicators on the pilots' center panel and the engineer's panel show the average temperature in the engine turbine case. Six thermocouples, electrically connected in parallel, are installed in each engine exhaust section. Signals developed by the thermocouples are fed through the converter unit to the indicator servo motor which positions the vertical scale. The scale is calibrated from 0°C to 700°C.

Power (115-volt, 400-hertz, single-phase AC) for the EGT indicators is supplied by Essential AC Bus No. 2. There are four circuit breakers on the engineer's circuit breaker panel. NOTICE that the EGT, tachometer, and fuel flow indicators for an engine share the same circuit breaker.

Tachometer Indicators

There are two four-channel tachometers on the pilots' center panel and two on the engineer's panel. This is because they have both an N_1 and N_2 compressor speed indicating system.

The N_1 indicating system shows the speed of the low-speed compressor. The N_2 indicator shows the speed of the high-speed compressor.

There are two tachometer generators per engine, one for each system. The N_1

tachometer generator is driven by the front accessory drive and routes the wires through the inlet guide vanes. The N_2 tachometer generator is driven by the main accessory drive.

As the engine speed changes, a variable signal is developed by the tachometer generator, fed to the converter unit, then to the indicator to move the vertical tape. The tape is calibrated from 0 to 110 percent.

Again, 115-volt, 400-hertz, single-phase AC power is supplied by Essential AC Bus No. 2. These indicators use the same four circuit breakers as the EGT and fuel flow indicators.

Fuel Flow Indicators

Two four-channel indicators are installed, one at the pilot's and one at the engineer's position. A transmitter is mounted on the forward right side of the low compressor case.

Fuel flowing through the transmitter creates a signal, which, after being amplified by the converter unit, positions the vertical tape. The tape is calibrated from 400 to 16,000 pph.

Power supply and circuit breaker protection is the same as for the two preceding instruments.

Engine Pressure Ratio Indicators

Two four-channel indicators are used in the engine pressure ratio indicating (EPR) system. These indicators show the ratio of engine turbine exhaust total pressure to compressor inlet total pressure. The EPR reading is used as a measure of engine thrust.

The converter-transmitter for each indicator is mounted in the engine pylon. An inlet pressure probe (PT_0) is mounted on the inboard side of each pylon. Six exhaust pressure probes (PT_7) are mounted in the exhaust of each engine. Changes in pressure are sensed by these devices and transmitted to the indicator. The EPR indicator moving tape is marked from 1.0 to 2.3.

The 115-volt, 400-hertz, single-phase AC power, required by the EPR indicators, is supplied by the Isolated AC Bus. There is an EPR circuit breaker for each engine.

VSEI Power OFF Warning

Each of the moving tapes in the VSEI indicators has a section of the tape, colored red, which will come into view at the top of the indicator column should there be a power loss to that indicating system. The word OFF is printed on the red section of tape. With normal power to the indicator, this red portion of the tape will be out of view.

Engine Vibration Indicators

Four vertical-scale type indicators, located on the flight engineer's lower panel, supply indications of engine vibration and aid in isolation of engine vibration. A vibration filter selector switch and an indicator pickup selector switch control monitoring on all four indicating systems. There are two vibration sensors on each engine: One on the forward end located on the left side of the compressor section, and one on the bottom of the turbine section. The sensors are monitored one at a time, depending on switch position. The vibration indicator registers the average vibration displacement on a zero-to-five MIL scale.

The vibration sensing system is capable of sensing both high and low frequency vibrations. The HI-LOW vibration filter selector switch is spring-loaded to the low position; in this position the amplitude of the total frequency range (HI & LOW) of vibration is presented on the indicator. Positioning the filter switch to HI allows only the high frequency vibration to be presented on the indicator. The pickup selector switch is a two-position switch. It remains in the desired position for either the FWD or AFT sensor. The PUSH-TO-TEST Switch is used to check continuity of the wiring and vibration sensors. When the switch is actuated, the indicator pointer should move to approximately 3.5 mil.

The vibration indicators use 115-volt, 400-hertz, single-phase AC power from the Isolated AC Bus. The pickup selector circuits require 28-volt DC power from the Isolated DC Bus to operate the pickup selector relay.

Chapter 4

INTERPHONE AND PUBLIC ADDRESS SYSTEM

(Figures 3-6, 3-7)

AN/AIC-18 Interphone System

The AN/AIC-18 Interphone System provides voice communication between the flight stations, cargo compartment and ground crew personnel. It also provides switching and mixing facilities for transmitting and receiving over the communications systems, and receiving the navigation systems.

The DC voltage for the interphone system consists of three independently operated circuits. Operational power for each circuit is provided through 28-volt DC circuit breakers.

Interphone control and monitor panels are installed at the pilot's, copilot's, navigator's, and flight observer's crew positions. The flight engineer's position has only the interphone control panel installed. The Center Avionic Equipment Rack contains both panels.

The control panel consists of a mike selector, 8 push-pull audio switches, a HOT MIC TALK switch, a master volume control and a CALL button. The monitor panel provides 8 additional audio switches to increase the receiving capability of the control panel.

The mike selector has seven positions and allows the operator to transmit and receive the following facilities:

I - Interphone and public address transmission

U1 - UHF Command Radio No. 1

U2 - UHF Command Radio No. 2

H1 - HF Radio No. 1

H2 - HF Radio No. 2

V1 - VHF Command Radio No. 1

V2 - VHF Command Radio No. 2

NOTE: Only the interphone and public address positions are wired for operation at the forward crew door and the two jumpmaster/loadmaster control panels. Only the interphone position is wired for operation at the flight engineer's panel, though all interphone control panels are interchangeable.

There is a MIC-OFF-INTERPHONE switch on the control yoke. When the switch is held in the up position (INTERPHONE), transmissions will be made only on interphone, regardless of the position of the MIC selector on the interphone control panel. In the down (MIC) position, the user is able to transmit over whatever facility he has selected on the MIC selector on the interphone control panel. In the spring-loaded middle position, the MIC is inoperative.

There is also an interphone button on the nose steering wheel that enables the pilot to transmit over interphone while steering the aircraft.

The 16 push-pull, rotary, audio switches enable the user to monitor any combination of audio facilities and individual volume control. The push-pull feature connects and disconnects the individual audio facilities, while the rotary feature affords the individual volume control.

The master volume control determines the output of an audio amplifier within the control panel, through which all signals are controlled simultaneously. This does not mean that all signals will be heard equally well, as most signals arriving at the amplifier have passed through two other volume controls. Therefore, the strength of each signal will be different. The audio amplifier will amplify each signal the same ratio, but it will not amplify a weak signal to the level of a strong signal.

The CALL feature is used to alert all crew members and to assure that they receive the information broadcast on the interphone circuit. The system is activated by depressing the CALL button on the control panel. Energizing the CALL button does not interrupt the other signals being received, but the CALL signal will be somewhat louder than the other signals.

The HOT MIC buttons on the control panel permit direct transmission to all interphone stations on the flight deck, avionics area, and forward crew door, without pressing the microphone switch at the interphone station. The push-pull buttons are labeled TALK and LISTEN and must be pulled out for hot microphone mode of operation.

An auxiliary control panel is installed in the vertical stabilizer. Operation from this position is limited to interphone.

There are external receptacles installed on the left forward fuselage, left wheel well, and right wheel well, to permit interphone communications between flight and ground personnel. The receptacle on the left forward fuselage is wired into the center avionics equipment rack control panel and can be used for radio communications from outside the aircraft on most aircraft. The left and right wheel well receptacles are wired into the left and right jump door interphone control panels respectively and can be used for interphone or PA system.

Public Address System AN/AIC-13

The public address system allows the crew to broadcast interphone or stations received on the ADF receivers throughout the cargo compartment. The system consists of a main control panel, three auxiliary control panels, a public address switch, three amplifiers and six speakers.

The required 28-volt DC power for operation is provided from the DC Avionics Bus No. 2, through a circuit breaker on the avionics circuit breaker panel.

Main Control Panel

The main control panel is located at the flight engineer's station. This panel has complete operational control of the system. The panel consists of a power

ON/OFF switch, a speaker selector switch, a volume control, and five toggle switches. The power ON/OFF switch turns the system on or off. The speaker selector switch selects the desired speaker or speakers to be used. In the JUMP position, the speaker forward of each troop door is in operation; in the ALL position, all six speakers are operational; in the FWD position, only the most forward speaker (just above the left forward side escape hatch) is operational; and in the AFT position, only the speaker aft of the pressure door is operational. The master volume control controls the volume of the entire system. Only the two toggle switches marked ADF 1 and ADF 2 are wired for operation. By placing them in the UP position, ADF reception is broadcast over the PA system. ADF reception is blocked out anytime the PA position is energized at the Jumpmaster/Loadmaster panels.

LH and RH Jumpmaster/Loadmaster Panels

These panels are located at the forward crew door and left and right jump doors. In addition to providing interphone reception and transmission capability, this panel affords the user the ability to transmit over the PA system, provided it is turned on at the main control panel. It also provides remote control of the volume level through the use of INCREASE and DECREASE buttons. These pushbuttons are electrically interlocked to prevent simultaneous increase or decrease operation.

Pilot's Side Console

The public address switch is located at the pilot's side console. With the switch in the INPH & PA position, any signal broadcast over the interphone system will be amplified and transmitted over the PA system, provided the PA system is turned on. In the INPH position, interphone communications are restricted to the normal interphone circuits. In the INPH and PA position, the ADF receivers will not be heard through the PA system.

Speaker Locations

Speakers No. 1 and 3 are located above the left forward side escape hatch, and speaker No. 2 is located just above the No. 2 hydraulic service center. Speakers No. 4 and 5 are just forward of the troop doors, and speaker No. 6 is aft of the pressure door.

Amplifiers

The three amplifiers are on the center avionics rack. The No. 1 amplifier supplies power for the No. 1 and No. 3 speakers, and the No. 2 amplifier supplies power for No. 2 and No. 4 speakers. The No. 3 amplifier powers the No. 5 and No. 6 speakers.

Chapter 5

EMERGENCY LOCATION/RECORDING DEVICES

(Figure 3-8)

Introduction

To quickly locate the aircraft upon an accident or crash landing, two systems are installed on the C-141 to aid in finding the aircraft and/or its recording devices. One is an underwater acoustic beacon (UAB); the other is an emergency locator transmitter (ELT). Once the aircraft has been located, two other devices will aid in the accident investigation board's findings of the causes of the accident. One such device is a cockpit voice recorder (CVR); the other is a digital flight data recorder (DFDR). All four of these systems are designed to be fire and crash resistant.

Underwater Acoustic Beacon (UAB)

The beacon is mounted on the digital flight data recorder, and is contained in a cylindrical watertight aluminum case capable of withstanding high-G impact shock and deep water immersion. The beacon consists of a self-contained battery, an electronic module and a transducer. The shock-mounted battery occupies approximately two-thirds of the case and is separated from the electronic module by a bulkhead that is integral with the case. The opposite end contains a teflon-insulated water switch.

The beacon radiates a pulsed acoustic signal into the surrounding water upon activation by its water-sensitive switch. Search operations for beacon-equipped aircraft that have crashed in water are conducted by utilizing a receiver equipped with a directional hydrophone (SONAR).

Operating Frequency	37.5± KHZ
Operating Depth	Surface to 20,000 feet
Pulse Length	Not less than 9 milliseconds
Pulse Repetition Rate	1 Pulse per second (1 Pulse per .5 second for Mod 1)
Operating Life	30 Days (12± 2 Days for Mod 1)
Battery Life in Beacon	1 Year

Emergency Locator Transmitter (ELT) System (AN/ARC-31)

General Description

The Emergency Locator Transmitter (ELT) system is installed on all C-141 aircraft. This system aids search, rescue and recovery operations by automatically signalling the geographical location of the downed aircraft. In the event of a crash landing, a radio beacon begins immediate transmission on the 121.5 and 243.0 megahertz distress frequencies.

ELT Components

The ELT system contains the following components: an ELT, a diplexer, an antenna, a remote control unit and the ELT mount.

Emergency Locator Transmitter

The ELT is a self-contained unit including a solid state transmitter, an inertial impact switch that will turn on the transmitter above a given impact force along the longitudinal axis of the aircraft, and a battery pack that consists of three D-size cells. The shelf life of the battery pack exceeds 24 months.

Diplexer

The purpose of the diplexer is to split the signal put out by the transmitter into a UHF and VHF signal and to send this signal to the antenna.

Antenna

The antenna is mounted on the top of the empennage bullet just aft of the HF antenna. It is both a UHF and a VHF antenna combined into one unit.

ELT Mount

The ELT mount, directly beneath the antenna, secures the ELT and diplexer to the underside of the empennage bullet and provides the ELT diplexer and antenna the ability to withstand up to 100 g's of impact force, even if the mount should separate from the rest of the aircraft.

Remote Control Unit (RCU)

The RCU, on the upper aft portion of the navigator's instrument panel, provides a means of checking the operational status of the ELT system, and a means of deactivating the unit due to inadvertent operation. This unit, the RCU, is not needed for the ELT to transmit an emergency signal if activated by a crash impact.

COCKPIT VOICE RECORDER (CVR) SYSTEM

(Figures 3-9 to 3-11)

The purpose of the CVR is to record continuously the last 30 minutes of flight station interphone and radio communications, as well as general flight station noise (talking and aircraft sounds). It accomplishes this by monitoring communications from the pilots, copilots, and flight engineers interphone stations. Recording anything that those crew members have selected to monitor and transmit on. The system picks up flight station noises by the use of an area microphone located in the front of the CVR control unit.

CVR Control Unit

The CVR control unit is in a fiberglass housing attached to the aft end of the pilot's overhead control panel. The control unit has, located on the front control panel, an area microphone, a test meter, a test button or switch, and a receptacle for a headset jack. Operation of the unit is automatic whenever the power select switch is positioned to "AUX" or "OFF" while the APU or engine-driven generators are on and powering the aircraft.

Cockpit Voice Recorder

The CVR is in the hayloft area, just aft of the right liferaft silo canister, and just forward of the No. 3 smoke detector. There are two day glow international orange-painted recorder boxes mounted on a tray there; the one on the right is the CVR. The unit is crash/fire/and salt water resistant and is used to record/store/erase any data sent to it from the control unit located on the flight station. Its operation is the same as the control unit above. The recording section consists of a four-channel recorder, a continuous loop magnetic tape that will record for a period of 30 minutes, and an eraser that will continuously erase the oldest data.

CVR Test Switch

The CVR test switch is in the right hand underdeck area, on the center avionics rack. This test switch, just aft and above the DFDR acquisition unit, is used to test the CVR system. By taking the switch to the test position, you bypass normal CVR operation and allow the CVR to operate while the power select switch, on the flight engineer's panel, is in the external power position. This position is used only to test the operation of the system during ground maintenance.

DIGITAL FLIGHT DATA RECORDER SYSTEM (DFDR)

(Figures 3-12 thru 3-14)

The DFDR system is used to record the last 25 hours of selected aircraft systems operation. The monitoring of these systems allows for the reconstruction of aircraft flight and ground operations, for accident/incident investigations. The system has five major components used in conjunction with each other to record/identify the various areas monitored. The components are:

1. Digitial Flight Data Recorder (DFDR) Unit
2. Flight Data Acquisition Unit (FDAU)
3. Triaxial Accelerometer
4. Underwater Acoustic Beacon
5. Rudder Position Synchro

Digitial Flight Data Recorder Unit

The DFDR unit is in the hayloft area, just aft of the right liferaft silo canister, and just forward of the No. 3 smoke detector. There are two day glow international orange-painted recorder boxes mounted on a tray there.

The one on the left is the DFDR. Mounted on the front of the case of the DFDR there is an underwater acoustic beacon. The DFDR unit's purpose is to record data sent to it from the acquisition unit. The DFDR is located toward the tail of the aircraft to help it withstand a crash impact. The unit is shock/salt water/fire resistant and contains an airborne magnetic tape recorder, with a six-channel tape and an eraser. The tape is a 1/4-inch mylar-based endless loop tape and has a storage capacity of 25 hours. Once part of the tape data becomes 25 hours old, it is erased and new data is recorded.

Flight Data Acquisition Unit (FDAU)

The Acquisition Unit, on the right side of the center avionics rack, is used to acquire and transmit data from 18 different aircraft monitored systems. The unit converts aircraft data into a 65-word-per-second computer format, and then transmits this data to the DFDR unit for recording. The following parameters are sensed by the FDAU.

1. Altitude
2. Airspeed
3. Heading
4. Vertical Acceleration
5. Lateral Acceleration
6. Longitudinal Acceleration
7. Pitch Attitude
8. Roll Attitude
9. Rudder Position
10. Pitch Trim Position
11. Engine Pressure Ratio
12. Flap Position
13. Spoiler Position
14. Thrust Reversers Extended
15. Thrust Reversers Locked
16. Landing Gear Position
17. Radio Transmission Keying
18. Aircraft Identification

The FDAU also contains the fault detection circuits to advise the crew or maintenance of faults. Once a fault is sensed through a built-in test circuit, the "FLT REC INOP" light on the annunciator panel will illuminate.

Triaxial Accelerometer

The triaxial accelerometer, on the aft center wing beam just forward of the flap pack in the cargo compartment, picks up movement around the three axes of the aircraft and sends this information to the FDAU.

Rudder Position Synchro

The rudder position synchro, mounted in the aft portion of the empennage and attached to the rudder torque tube, picks up movement of the rudder and transmits this information to the FDAU.

Operation of the DFDR System

The DFDR system's operation is controlled by the position of the power select switch. When the power select switch is placed to the "OFF" position, and the engine generators are operating, the DFDR system will start recording data as acquired. When the power select switch is positioned to "AUX" or "EXT," the DFDR is shut down. The system is designed, with the exception of the triaxial accelerometer and the rudder position syncro, to record data only. In this way, it does not affect the operational status of the aircraft. All system signals are picked up as generated through normal operation of the aircraft systems output to the flight deck instrumentation.

FDR Test Switch

The flight data recorder test switch, adjacent to the right side of the center avionics rack, allows for testing of the system on the ground and for operating tape copy. The switch will bypass the normal interlocks in the "EXT" and "AUX" positions of the power select switch when used for tape copy operations. A ground test/recorder unit plugged into a cannon plug in the center of the left aft chopping area is used to test or extract recorded data from the system. When the test switch is placed in test, all 25 hours of recorded tape information will be copied in 16 minutes, without erasing the original data.

Chapter 6

PITOT-STATIC SYSTEMS

(Figures 3-15 thru 3-20)

Introduction

Two independent pitot-static systems provide pressures to operate the central air data computers (CADC), two altimeters, and an airspeed indicator. These pressures are picked up by four pitot-static tubes, two on each side of the fuselage, mounted just aft and slightly above the crew entrance door.

Pitot-Static Tubes

A pitot opening and a static opening are in the head of each pitot-static tube. The lower tubes supply both pitot and static pressures. The upper tubes supply static pressure only as the pitot ports are plugged.

The static system is a balanced static system. The static vents of the upper tube on one side of the aircraft are connected to the static vents of the lower tube on the opposite side of the aircraft. This arrangement provides stable static pressure regardless of crosswinds, slipping or skidding of the aircraft.

Pilot's Pitot-Static System

The lower left and upper right pitot-static tubes form the pilot's (or No. 1) pitot-static system. This system furnishes pitot and static pressures to the No. 1 CADC only.

Copilot's Pitot-Static System

The upper left and lower right pitot-static tubes form the copilot's (or No. 2) pitot-static system. The copilot's system furnishes pitot or static pressures directly to the No. 2 CADC, navigator's and engineer's altimeters, and the pilot's standby airspeed indication.

Pitot-Static Systems Drains

Two drain boxes, one on each side of the fuselage, and internally mounted just below the pitot-static tubes, permit access to capped drain lines which can be used for moisture drainage from the tubes. Two capped drain lines, located under each CADC, permit drainage of the rest of the systems.

One manual shutoff valve is installed in the copilot's (No. 2) system. It is used to isolate a portion of the system in the event of a leak. The valve is in the right-hand underdeck area, just forward of the crew's latrine, and can be reached through an access door in the floor near the engineer's station. The shutoff valve is normally open. When the valve is closed, the engineer's and navigator's altimeters are isolated from the system. Pitot and static pressure will still be available to the No. 2 CADC and the pilot's standby airspeed indicator.

Pitot-Static System Anti-Icing

Anti-icing of the pitot-static tubes and masts is provided by 115-volt, 400-hertz, single-phase AC powered heating elements. Each pitot-static tube contains two anti-ice heating elements. One is in the tube head, to prevent ice from covering the pitot and static vents. The other heating element is in the tube mast and prevents ice buildup on the mast.

All the heating elements are powered by the Essential AC Busses except the pilot's pitot-static head heater elements. The pilot's pitot-static head heater elements are powered by the Emergency AC Bus and will keep the No. 1 system pitot-static ports open when operating with only emergency power available.

The anti-icing systems are controlled by two OFF-ON switches on the pilot's overhead panel.

A HTR FAULTED light above each switch will illuminate if a power loss should occur in either head heating element and the switch is ON. In addition, a PITOT HEAT light on the annunciator panel and both master CAUTION lights will illuminate.

Standby Instruments

The instruments operating on direct pitot and/or static pressures (not connected through a CADC) are the pilot's standby airspeed indicator and the navigator's and engineer's altimeters.

The pilot's standby airspeed indicator is mounted on the pilot's instrument panel. The pilot's instrument is of the conventional design, but only one-half the size of a conventional airspeed indicator. It is used only if the vertical scale flight instruments have malfunctioned.

The navigator's and engineer's altimeters are the conventional type and can be adjusted by using the knob and barometric scale provided on each altimeter. The pilot also has a standby compass mounted forward of the pilot's overhead panel.

Total Temperature Indicators

There are two total temperature indicators. One is located on the pilots' center instrument panel and one is on the engineer's instrument panel. They indicate total temperature in degrees centigrade. Total temperature is defined as ambient air (outside) temperature plus ram effect.

The total temperature indicators use 115-volt AC power from the Main AC Bus No. 1. Both indicators are protected by a single circuit breaker labeled TOTAL TEMP IND. A power-off mechanism displays the word OFF through a window in the face of the indicator, should electrical power be lost.

A total temperature probe is on each side of the fuselage, below the pilot's windows. Each temperature probe contains two temperature sensing elements. Probe No. 1 is on the left side and has one sensor connected to the engineer's total temperature indicator and the other connected to CADC No. 1. Probe No. 2 is on the right side of the fuselage and has one sensor connected to the total temperature indicator on the center instrument panel. The second sensor is connected to CADC No. 2.

Each temperature probe has a 115-volt AC de-icing element which is energized by a switch on the pilots' overhead panel, labeled TEMP PROBE DE-ICE.

The No. 1 probe de-icing element is powered by Essential AC Bus No. 1. No. 2 probe de-icing element is powered by Essential AC Bus No. 2.

NOTE: There is no warning for a de-icing system failure.

Chapter 7

CENTRAL AIR DATA COMPUTERS

(Figures 3-21 thru 3-23)

Introduction

Two complete and independently operated Central Air Data Computers (CADCs) are installed in the C-141 to supply primary flight information to the pilot's and copilot's Vertical Scale Flight Instruments (VSFI). They also provide control and warning information to the Automatic Flight Control System (AFCS), air conditioning systems, elevator artificial feel system, rudder pressure reducers, overspeed warning system, stall warning system, rudder pressure warning, and navigator's true airspeed indicator.

Pitot-static pressures, from the aircraft pitot-static systems, and temperature information, from the total temperature probes, are supplied to each CADC. The CADCs transform these primary input variables into electrical servo signals representing true airspeed, calibrated airspeed, Mach number, pressure altitude, and vertical velocity.

CADC No. 1 receives pitot-static pressures from the pilot's system and temperature information from a temperature sensor mounted in the left total temperature probe. The electrical signal outputs are amplified and displayed on the pilot's primary flight instruments and the navigator's true airspeed indicator (if selected). Electrical power is supplied by the 115-volt, 400-hertz Emergency AC Bus. A circuit breaker, labeled CADC No. 1, provides power circuit protection. There are no control circuits, and the system is in operation whenever the Emergency AC Bus is powered.

CADC No. 2 has the same power requirement, but power is supplied from the Navigation AC Bus No. 2. Whenever power is applied to Navigation AC Bus No. 2, the system is in operation. Circuit protection is supplied by a circuit breaker labeled CADC No. 2. Pitot-static inputs for CADC No. 2 are provided by the copilot's system. A temperature sensor in the right total temperature probe provides temperature information. Electrical signal output is displayed on the copilot's primary flight instruments and the navigator's true airspeed indicator (if selected).

Both CADCs are located on the underdeck equipment racks. CADC No. 1 is on the left side, on the forward equipment rack, and CADC No. 2 is on the right side of the center avionics rack.

Testing

On the front panel of each CADC is a push-to-test switch and four malfunction warning lights. The warning lights are labeled Hp (pressure altitude), V_t (true airspeed), V_c (calibrated airspeed) and M (Mach). Monitor circuits within the CADCs will sense malfunctions and illuminate one or more warning lights, depending on which circuit has malfunctioned.

Actuating the push-to-test button will drive the CADC through a self-test cycle. The moment the test cycle starts, all four warning lights should illuminate.

The lights will not go out until the preset values are reached. The time limit for self-testing is 150 seconds, at which time all four warning lights should be OUT. Should a light remain ON, a malfunction is indicated in that circuit. Readout of the preset values will be displayed on the corresponding primary flight instruments. In addition, a CADC INOP light will be displayed on the annunciator panel, and both master CAUTION lights will illuminate.

In addition to the test switches on the CADC, there is a test switch on each side console. These switches have three positions: SELF TEST, NORMAL and MON TEST.

The test switch for No. 1 CADC is on the pilot's side console, and the test switch for No. 2 CADC is on the copilot's side console.

The SELF TEST position is used for normal testing of the system, and this position corresponds to the push-to-test switch on the CADC front panel. When the switch is held to the SELF TEST position, the CADC reacts and drives the vertical scale flight instrument to preset test values within 150 seconds. These values are:

Altitude	50,000' ± 110'
KCAS	225 knots ± 3.5
Mach	.92 ± 0.015
Navigator's TAS	527 knots ± 5

During the test cycle, the vertical velocity indicator will indicate a 20,000 foot per minute rate of climb.

As the Mach indicator passes .3 Mach, the EJECTOR ON light on the flight engineer's environmental panel will go OUT. An audible aircraft overspeed warning will be heard in the headsets as Mach 0.825 is reached.

Throughout the entire test cycle, flags will appear on the Mach, on the calibrated airspeed, and on the altitude indicators. The CADC INOP lights on the annunciator panel will come ON, and both master CAUTION lights will illuminate.

The test cycle is complete when the vertical velocity indicator returns to zero. The switch is then released, and the CADC system will return to normal mode of operation.

Holding the switch in MON TEST (Monitor Test) will test the CADC monitor circuits. This is done to assure the monitor circuit will operate at minimum prescribed values. The monitor circuits normally sense mechanical or electrical malfunctions within the CADC and trigger warning signals which warn the pilots of the malfunctions. During monitor test, flags should appear in the Mach and calibrated airspeed instruments. The CADC INOP light will illuminate on the annunciator panel, and the three warning lights on the front panel of the respective CADC will come ON (Hp [pressure altitude] light will not illuminate). In addition, the master CAUTION lights will illuminate.

Releasing the switch will return the CADC to normal mode of operation. All CADC test switches are interlocked through the touchdown relays to prevent testing the CADCs in flight.

Chapter 8

VERTICAL SCALE FLIGHT INSTRUMENTS (VSFI)

(Figures 3-24 thru 3-26)

Introduction

The pilot's and copilot's instrument panels are equipped with independently operated electro-mechanical vertical scale flight instruments. Each panel has two group-type indicators. One indicator will supply airspeed and Mach information, and the other will supply vertical velocity and altitude. Each indicator is integrally lighted and divided into three display columns. Each column contains a vertically moving tape moving past a stationary horizontal reference line across the center of the indicator.

Each indicator is driven by a separate amplifier mounted on the top shelf of the center equipment rack and receives electrical signals from the central air data computers. The pilot's indicators receive signals from CADC No. 1, and the copilot's indicators receive signals from CADC No. 2.

The indicators are powered by 115-volt, 400-hertz single-phase AC power. The pilot's VSFIs are powered by the Emergency AC Bus and the copilot's by the Navigation AC Bus No. 2. An electrical or mechanical failure of one indicator will not affect any other VSFI.

Mach-Airspeed

Two Mach-Airspeed Indicators provide the pilot and copilot with indications of Mach number and calibrated airspeed. The Mach number is on the center tape, and calibrated airspeed is on the right tape.

The Mach number is indicated on a vertically moving scale in the middle column. The Mach tape also moves past a fixed index line and is graduated from 0.2 Mach through 1.0 Mach in hundredths of Mach increments. The tape is numbered at each one-tenth Mach increment. A hashed maximum allowable MACH marker is painted on the tape at 0.825 Mach and indicates the maximum allowable Mach number speed when the symbol is aligned with the fixed index line.

A command Mach marker is used in conjunction with the Mach tape and may be set to any Mach number between 0.2 and 1.0 Mach. The command marker position is set by moving the command Mach slewing switch on the base of the indicator, up or down. The set position of the command Mach marker is indicated in the command Mach readout window, located above the slewing switch.

Releasing the slewing switch will lock the command marker at the set Mach value. The command marker remains at the top or the bottom of the Mach column until the set Mach value comes into view. Then the command marker will follow the tape at the set Mach value. When the command marker is aligned with the fixed index line, the aircraft is flying at the command Mach value.

A warning flag, that reads MACH, will come into view just below the fixed index line to indicate unreliable Mach information is being displayed or power has failed.

Calibrated airspeed is indicated in the right column on a vertically moving tape, moving past a fixed index line. The tape scale is graduated from 50 through 600 knots in 10-knot increments. Between the 100 through 200 knot scale, each 20-knot increment is numbered. Above 200 knots, each 50-knot increment is numbered.

A command airspeed marker is used in conjunction with the tape, the command airspeed readout window, and the command airspeed slewing switch. Pre-selection and operation of the command airspeed marker is identical to that of the command Mach marker, with one exception. The command airspeed slewing switch has a side detent position, in which the command airspeed marker will align itself with the fixed index line and the calibrated airspeed (as read at the fixed index line) will appear in the command airspeed readout window. Any change in airspeed will be indicated immediately in the command readout window.

A warning flag, reading IAS, will come into view if the displayed airspeed is unreliable or power is lost.

Vertical Velocity - Altitude Indicators

The pilot and copilot indicators have three columns with a vertically moving tape. The left column contains the vertical velocity indicator, the center column has the vernier (sensitive) altitude scale, and the right column has the gross altitude scale. Altitude and vertical velocity for the pilot's indicator comes from CADC No. 1. The copilot's information comes from CADC No. 2.

Vertical velocity information is displayed in the left column. Its operation is somewhat different than the other flight instruments, in that a moving pointer will move past a fixed scale on the face of the indicator, when the vertical velocity is 1,500 feet-per-minute or less. The fixed scale is graduated from 0 through 1,500 feet-per-minute climb or descent, in 100 feet-per-minute graduations. Each 500 fpm increment is numbered.

When the vertical velocity increases above 1,500 feet-per-minute, the pointer will stop at the top (or bottom) of the column, and a vertical tape will move past it. The vertical tape is graduated in 1,000 fpm increments from 2,000 up to 6,000 fpm, in 2,000 fpm increments up to 10,000 fpm, and in 5,000 fpm increments up to 20,000 feet-per-minute. The vertical speed of the aircraft is indicated at all times by the pointer. The altitude information is indicated in the middle and right columns. The middle column has a vertically moving tape indicating vernier (sensitive) altitude and the right column has a vertical moving tape indicating gross altitude. Both tapes move past a fixed index line and must be used together to indicate aircraft altitude.

The vernier tape (middle column) is graduated from zero through 1,000 feet, in 50-foot increments. Each 100-foot increment is numbered. The vernier tape will make one complete revolution for each 1,000 feet indicated on the gross altitude tape. The gross tape is graduated from minus 1,000 feet to plus 60,000 feet in 500-foot increments, with each 1,000-foot increment being numbered.

A command altitude marker is used in conjunction with the vertical moving tapes, command altitude slewing switch, and command altitude readout window. Operation of the command marker is identical to that of the airspeed indicator. The slewing switch has a side detent position. Command altitude is indicated in the readout window when the slewing switch is in the CENTER position.

The command altitude marker will remain at the bottom or top of the column until the preset command altitude appears in the gross altitude column. The command marker will then position itself to the preselected command altitude value on the vertically moving gross altitude tape. Command altitude is reached when the command marker is aligned with the fixed index line.

Placing the altitude slewing switch in the side detent position allows digital readout of aircraft altitude in the command readout window. When self-testing the CADCs or the VSFIs, the slewing switch must be in the side detent position.

On the lower left portion of the instrument case is a BARO set knob, which is used to make barometric pressure (altimeter settings) adjustments to the altitude indicators. Digital readout of the barometric setting appears in a window just above the BARO set knob.

Power failure or unreliable altitude information is indicated when a flag (reading ALT) appears in the gross altitude column above the fixed index line.

VSFI Testing

Two VSFI switching panels are installed for the purpose of testing the vertical scale flight instruments. The pilot's VSFI switching panel is on the pilot's side console and will test the pilot's Mach, IAS and altitude indicators. The copilot's VSFI switching panel is on the copilot's side console and is used to test the copilot's Mach, IAS and altitude indicators.

Each switching panel has two test switches. One switch is labeled "IAS" and will test the Mach and IAS indicator. The other is labeled "ALT" and will test the altimeter. Each switch has three positions (NORM, SELF TEST and MON TEST), and is spring-loaded to the NORM position. Holding either (or both) switches to the SELF-TEST position will drive the indicators to the same preset values. The warning flags will appear in the indicators being tested. To attain the correct altimeter preset test value, 29.92" Hg must be set into the barometric pressure readout window. Releasing the switches will allow the indicators to return to normal operation.

The MON TEST position of either switch is used to assure the pilots that the indicator monitor circuits will operate at a minimum prescribed value. Holding the switches in the MON TEST position will trip the monitor circuits within the indicators and display a warning flag in the indicator being tested. Releasing the switches will reset the monitor circuits and pull the warning flags out of view.

The VSFI test switches will operate in flight or on the ground and are primarily used to check the accuracy of the indicators.

Chapter 9

AUTOPILOT

Introduction

The autopilot provides the means by which the aircraft is controlled automatically in flight. The autopilot can guide the aircraft to the runway for landings. The action of the autopilot is smooth since autopilot signals for corrective action are directly proportional to the amount of displacement. Coordinated turns are made in all modes of operation.

Any desired pressure altitude or Mach can be maintained. Any desired navigation course can also be maintained using VOR No. 2, ILS No. 2, TACAN No. 2 or INS signals.

Power Requirements

The autopilot uses 115-volt, 400-hertz, single-phase AC power from AC Nav Bus No. 1, and 28-volt DC power from the Main DC Avionics Bus No. 1.

The autopilot warning circuits are connected to the Isolated DC Bus.

Autopilot Controls and Indicators

The Automatic Flight Control System (AFCS) panel is located on the pilots' center console. With this panel, the pilot can control the desired mode engagement. In addition, he can change altitude or make coordinated turns without disengaging the autopilot.

All of the switches on the AFCS control panel with the exception of the MACH INC-MACH DEC are held to their engaged position by a holding solenoid.

An interlock system prevents engagement or releases the switch if an improper engagement setup is selected by the pilot.

An AFCS trim indicator panel is located on the pilots' center instrument panel. This panel has four indicators and four indicator lights.

AFCS Control Panel Procedures

The AUTOPILOT switch on this panel is used to engage the autopilot. It can also be used to disengage the autopilot. If the autopilot disconnect switch on either control wheel is depressed, the autopilot will disengage and the AUTOPILOT switch will return to the OFF position. The yaw damper system does not disengage when the autopilot is disengaged.

The NAV SEL/LAT OFF switch allows the navigation aid selected on the copilot's Navigation Selector Panel (NSP) to furnish signals to the autopilot. Moving the switch to NAV SEL engages the navigation aid. Move the switch to neutral if the navigation aid is to be disengaged. The switch will be returned to neutral automatically if the TURN controller knob is moved out of detent or the autopilot is disengaged.

Aileron control only is disengaged by moving the NAV SEL/LAT OFF switch to LAT OFF. Aileron control by the autopilot is restored when the switch is returned to neutral.

The ALT HLD/PITCH OFF switch allows the pilot to disengage pitch control only by placing it to PITCH OFF. The ALT HLD position allows the autopilot to keep the aircraft at the pressure altitude existing at the time the switch is engaged to this position. Altitude hold can be disengaged by placing this switch to neutral.

Next to the ALT HLD/PITCH OFF switch is the PITCH controller. The PITCH controller can be rotated toward UP or DOWN to produce a pitch change. Maximum pitch angle is 30 degrees. The PITCH controller is inoperative under the following conditions: ALT HOLD engaged, PITCH OFF engaged, G/S (glideslope) engaged, or MACH HLD EL (Mach hold-elevator) engaged.

The TURN controller, located to the right of the PITCH controller, is used to make coordinated turns with the autopilot. Turn it "L" or "R" as desired. Maximum bank angle is 38 degrees. Whenever it is used, the compass heading or navigation aid being used is automatically disengaged. When the TURN controller is returned to neutral (detent) position, the compass is automatically reengaged as the aircraft rolls to a wings-level attitude. However, the navigation aid will be reengaged only if the NAV SEL/LAT OFF switch is placed to NAV SEL.

The G/S/VER NAV (glideslope/vertical navigation) switch is used, in the G/S position, with the navigational aid function of the autopilot in the ILS mode. An ILS frequency must be selected on VHF/NAV No. 2, and VOR/ILS must be selected on the copilot's NSP before the G/S switch will engage. The G/S arm light illuminates when the G/S switch is set to ON. The glideslope function engages automatically as the aircraft intercepts the glideslope. Altitude hold, if on, will automatically disengage, the G/S ARM light goes out, and the aircraft tracks the G/S signal. The VER NAV position of the switch allows the FSAS (Fuel Savings Advisory System) to utilize the old VER NAV channel of the autopilot, which had been disconnected, to give the autopilot climb, cruise and descent information. This is so that the aircraft can maintain those modes of operation inserted in the FSAS.

AFSC Trim Indicator Panel Procedures

The four trim indicators, two for the rudder and one each for the aileron and elevator, allow visual monitoring of the AFCS. Anytime a correction is applied to an autopilot control axis, the indicator bar deflects from the index mark.

The A/P OFF light comes ON to indicate that the autopilot has been disengaged by one of the system safety interlocks. The light may be turned OUT by depressing either of the control wheel A/P disconnect switches or reengaging the autopilot.

Control Wheel Steering

Control wheel steering (CWS) is incorporated into the pilot's control wheel only. It allows aircraft attitude control without disengaging the autopilot.

Control Wheel Steering Mode is selected by placing the CWS selector switch, on the AFCS control panel, forward to the CWS position.

A wheel pressure of more than 2.5 pounds will activate the system. The bank or pitch angle will change up to the limits of the engaged operating mode.

If CWS force is released, and bank angle is greater than three degrees, the aircraft will stay at the existing bank angle.

If CWS force is released, but bank angle is less than three degrees, the aircraft will roll level and hold the existing heading.

In pitch, if the CWS force is released, the aircraft will remain at the pitch angle existing at the time of release; however, a CWS force must be sustained to hold a deviation from the normal glideslope path during coupled ILS approaches.

CWS action is present if: AUTOPILOT switch is ON, TURN controller is in detent, and CWS switch is forward.

Roll CWS is inactivated when the NAV SEL/LAT OFF switch is in LAT OFF. NAV SEL has priority over roll CWS except when flying a coupled ILS approach. Pitch CWS is inactivated when ALT HLD or PITCH OFF is selected. Altitude hold has priority over pitch CWS. Pitch CWS has priority over MACH hold.

Autopilot Mode Selection

The various autopilot modes have definite priorities and compatibilities. The term compatibility is used to indicate that two different modes can be selected at the same time. For example, altitude hold and roll CWS can be selected at the same time.

Some modes have priority over other modes. If a higher priority mode is selected, the lower priority mode will drop out of the control circuit. The interlock system prevents the pilot from selecting a low priority mode when a higher priority mode has already been selected.

AUTOPILOT OPERATIONAL MODES

Introduction

The autopilot modes are selected by switches on the AFCS control panel. Selection of navigation aids used with the autopilot are made on the copilot's navigation selector panel.

Heading Hold Mode

This is a primary mode. In this mode, the autopilot uses the existing aircraft heading at time of switch engagement as the heading reference. This mode is selected when the AUTOPILOT switch (AFCS panel) is turned ON. It is not in operation when CWS or the TURN controller is being used.

Altitude Hold Mode

This mode holds the aircraft at the reference altitude existing at time of engagement. Setting the ALT HLD/PITCH OFF switch to ALT HLD engages this mode. The heading hold mode is retained. Altitude hold mode drops out if MACH HLD EL is selected, or if G/S ARM is selected and the aircraft intercepts the ILS glideslope. No pitch CWS is available when this mode is used.

Mach Hold Mode

This mode maintains the aircraft at the Mach existing at the time of engagement. The Heading Hold Mode is retained. Other heading modes may be selected. Use MACH HLD EL switch (AFCS panel). The Mach reference can be increased/decreased up to .05 Mach by use of MACH INC/MACH DEC switch (AFCS panel). This mode drops out if ALT HLD switch is used or if G/S ARM is selected and the aircraft intercepts the glideslope. CWS pitch control has priority over this mode.

Pitch Off Mode

Selecting PITCH OFF with the ALT HOLD/PITCH OFF switch turns off pitch control only. The rest of the autopilot is still engaged. The pilot and/or copilot has to fly the pitch axis manually in this condition.

Lateral Off Mode

Selecting LAT OFF with the NAV SEL/LAT OFF switch turns off lateral steering only. The rest of the autopilot is still engaged. The pilot and/or copilot flys the lateral axis in this mode.

Navigation Select Operation

Basic autopilot functions may be supplemented by signals from the navigation systems installed on the C-141. Selection of the desired aid is made on the copilot's navigation selector panel.

Heading Select Mode

In this mode, the autopilot maintains the aircraft on the heading set with the heading marker on the copilot's HSI. Set the HDG SEL/NAV button on the copilot's navigation selector panel to HDG, and the NAV SEL/LAT OFF switch (AFCS panel) to NAV SEL.

VOR/ILS Mode

This mode is placed in service by depressing the VOR/ILS button on the copilot's navigation selector panel, and placing the HDG SEL/NAV button to NAV. After these two have been set, set the NAV SEL/LAT OFF switch (AFCS panel) to NAV SEL. When intercept heading is established, the autopilot will remain responsive to the heading marker until the VOR course deviation is one dot deflection; then the navigation aid takes over control.

To use ILS, tune a localizer frequency on the VHF NAV No. 2 and proceed as above. The autopilot will begin to track the localizer when course deviation is slightly greater than two dots. After localizer intercept, position the G/S switch to G/S ARM. The autopilot will maintain level flight until intercepting the G/S; then it will begin to track the G/S signal.

TACAN Mode

For this mode, use the TACAN button on the copilot's navigation selector panel. Next, set the HDG SEL/NAV button to NAV. On the AFCS panel, the NAV SEL/LAT OFF switch is set to NAV SEL.

Turn the aircraft with the copilot's HSI heading marker to intercept the desired TACAN radial. The autopilot will receive TACAN course signals when the TACAN beam coupler is energized at one dot course deviation.

INS Select Mode

To use INS, select INS 1 or INS 2 on the copilot's navigation selector panel. On the AFCS panel, place the NAV SEL/LAT OFF switch to NAV SEL. The autopilot will then intercept and track the active INS course if the copilot's HDG SEL/NAV button is in NAV.

Fuel Savings Advisory System Mode (FSAS)

To use the FSAS, select the mode of operation in the FSAS (refer to the 1C-141B-1 for modes to select from and their use) press the lighted engage button on the FSAS CDU, and then take the G/S/VER NAV switch to the VER NAV position. The aircraft will now maintain or fly the mode selected in the FSAS.

Chapter 10

YAW DAMPER SYSTEM

(Figure 3-27)

Introduction

The Automatic Flight Control System (AFCS) includes a yaw damper system. This yaw damper system operates independently of the autopilot portion of the AFCS and is electrically isolated from the autopilot.

This yaw damper system controls the rudder in all modes of automatic operation.

Theory of Operation

Yaw damper engagement means that the rudder control surface is being positioned by the rudder servos of the AFCS. This engagement is accomplished by placing the yaw damper switch on the yaw damper panel to ON. The rudder servos have automatically assumed the position of the aircraft prior to this, resulting in a smooth takeover by the yaw damper.

If the autopilot is not operating, yaw damper rate gyros No. 1, No. 2, and No. 3 sense any deviation of the aircraft in the yaw axis. Immediately, a signal is generated proportional to the yaw, sent to the yaw damper computer, amplified, and then sent to the servos to dampen out the deviation.

If the autopilot is operating, a dynamic vertical sensor (pendulum) sends signals from roll crossfeed channel of the aileron computer into the rudder channel that is used for turn coordination. That is, if the turn is not coordinated, this device senses this fact and generates a signal. The magnitude of the signal is proportional to the amount of slipping or skidding. The vertical sensor's signal is then summed in with the yaw rate gyro signals to obtain the proper rudder position for a coordinated turn. The YAW damper is also used by the AWLS system for course corrections during automatic landings.

Operation

The yaw damper system uses 115-volt, 400-hertz, single-phase AC power from the Emergency AC Bus through three yaw damper circuit breakers on the emergency circuit breaker panel. In addition, the yaw damper uses 28-volt DC power from the Emergency DC Bus. There are two DC yaw damper circuit breakers on the emergency circuit breaker panel.

The yaw damper control panel is located on the pilot's center console. To turn on the yaw damper, place the OFF-ON switch to ON.

Warning System

If one of the yaw rate gyros or servos malfunctions, the monitor system will turn on the YAW DAMPER FAULT light located on the pilot's annunciator panel.

If more than one of the yaw rate gyros and/or servos malfunction, it is detected by the monitor circuit which turns on the YAW DAMPER INOPERATIVE lights located on the pilot's and copilot's instrument panels and automatically disengages the YAW DAMPER.

Testing

The yaw damper test circuit is wired through the touchdown relays, so that it is rendered inoperative whenever the aircraft is in flight.

Testing is accomplished from the yaw damper control panel located on the pilot's center console. Procedures for testing are located in TO 1C-141B-1.

Chapter 11

FLIGHT DIRECTOR SYSTEM

The C-141 aircraft is equipped with dual flight director systems (FDS). They are designated the pilot's and copilot's, and are completely independent. Each has its own compass and navigational aids inputs. The pilot's FDS receives its signals from the No. 1 navigational aids. The copilot's FDS receives its signals from the No. 2 navigational aids.

Attitude is by a vertical gyro signal from INS or AHRS (INS No. 1 or AHRS for the pilot's, and INS No. 2 or AHRS for the copilot's). Attitude information for both Flight Director Computers is supplied by the Test Programmer and Logic Computer. This signal is the median signal of the three attitude sources in the C-141.

Each FDS consists of six major components: Attitude Director Indicator (ADI), Horizontal Situation Indicator (HSI), Flight Director Computer, Roll and Pitch (Attitude) from INS or AHRS, Rate Transmitter, and the Navigation Selector Panel.

In addition to the above, there is the RATE OFF warning flag, Switching Rate Gyro, AWLS/Flight Director Test Panel, AWLS Fault Identification Panel, and AWLS Progress Display Panel. The Flight Director Computers, INS and AHRS Gyros, and the Rate Transmitter are located in the underdeck equipment racks.

The FDS is simply a combining of indicators, i.e., a bringing together or integration of many indicators into two. These two instruments are used to portray the aircraft's horizontal and vertical attitudes. The navigation selector panel is used to select different navigational systems for presentation on the flight director instruments.

ADI

Basically, the Attitude Director Indicator is a roll and pitch indicator. It employs a conventional artificial horizon to indicate the aircraft's attitude, relative to the earth. The INS or AHRS gyros and attitude sphere are used to provide this information.

The attitude warning flag comes into view when power is lost to either an attitude gyro or ADI.

A pitch trim knob is on the lower right-hand corner of the ADI to adjust the position of the attitude sphere in the pitch axis.

Turn and Bank Indicator

At the bottom of the ADI is a turn and slip indicator and a 4 minute rate-of-turn needle. These indicators provide regular needle and ball operation.

On the left side of the ADI is the vertical deviation indicator. The aircraft's position is represented by the center line, which is an extension of the miniature airplane. Vertical deviation is presented as a variable marker. The VDI has a warning flag which will appear if the glideslope signal is lost or unreliable. However, until the VDI is used, both the VDI and warning flag will be biased out of view.

An altitude indicator (rising runway) moves up into view at 180 feet Radar altitude during approaches. At touchdown, the altitude indicator should be touching the wheels of the miniature aircraft.

The ADI has two other items of importance: the bank and pitch steering bars. These are used to reflect computed bank and pitch steering commands and will be discussed in a later paragraph. Therefore, the ADI is primarily an artificial horizon with a turn and slip indicator and the VDI.

HSI

The Horizontal Situation Indicator is primarily a master repeater for the compass system. Also incorporated into the HSI is a range (DME) indicator and radio magnetic bearing pointer (RMI needle).

Aircraft heading (magnetic, true or grid) is read under the upper lubber line. The aircraft symbol is affixed to the face of the indicator, with the upper and lower lubber lines representing an extension of the aircraft's nose and tail. The compass card rotates around the aircraft symbol and lubber lines as aircraft heading changes.

The heading set knob is used to manually select different heading references as indicated by the position of the heading marker, relative to the compass card. Once set, the heading marker will rotate with the compass card as aircraft heading changes.

The bearing pointer is basically an RMI (Radio Magnetic Indicator) needle. However, in two cases it does not indicate the magnetic bearing to the station. This will be covered with the different modes of operation.

The center portion of the HSI (except for the aircraft symbol) is the course deviation section.

The course deviation indicator (CDI) reflects the actual deviation from the desired course (track or radial), as selected in the course window and indicated by the head of the course arrow. The TO-FROM indicator solves ambiguity, i.e., if the course selected is toward or away from the station. When the diamond points toward the course arrow head, the course is TO; when it points toward the course arrow tail, it is FROM the station. The desired course is selected by rotating the course set knob. Any one of 360 different courses can be selected. The complete CDI section will rotate when selecting a new course or changing the aircraft heading. The course warning flag will be out of view when signals are reliable.

The range indicator (DME) provides a miniaturized, digital readout of the distance to the selected station, if the mode selected affords the capability. The range indicator's maximum reading is 999 nautical miles.

Navigation Selector Panels

Two NAVIGATION SELECTOR panels (NSP), located on the pilot's and copilot's glare shields, contain controls for selecting the navigation modes for the flight director systems. The pushbuttons on the pilot's NAVIGATION SELECTOR panel control inputs to No. 1 flight director system, and the pushbuttons on the copilot's NAVIGATION SELECTOR panel control inputs to No. 2 flight director system and autopilot. Mechanical interlocks are incorporated in the top row of pushbuttons to prevent depressing any two pushbuttons together. The four pushbuttons on the bottom row of each unit (GRID HDG is light only) may be selected independent of the top row selection.

INS-1 and INS-2 pushbutton - applies INS navigation information and true heading from the selected INS to the associated HSI.

INS-1 and INS-2 ALERT (bottom half of INS No. 1 and INS No. 2 pushbuttons) illuminate when the ALERT light on the CDU of the associated INS illuminates.

TACAN pushbutton - applies TACAN signals to the associated HSI.

VOR/ILS pushbutton - applies VOR/ILS signals to the associated HSI.

NAV OFF pushbutton - removes all nav receiver and computer inputs to FDS and HSI.

GRID HEADING - consists of two independent lights, HSI and BDHI. They illuminate when GRID heading from INS is displayed on the associated HSI or BDHI as follows:

Pilot's NSP - GRID HD (HSI light) will illuminate when INS-1 is in Grid and pilot has selected MAG HDG (INS) and NSP is not in INS-1 or INS-2.

Copilot's NSP - GRID HDG (HSI light) will illuminate when INS-2 is in Grid and copilot has selected MAG HDG (INS) and NSP is not in INS-1 or INS-2.

Copilot's NSP GRID HDG (BDHI light) will illuminate when INS-1 is in Grid and pilot has selected MAG HDG (INS).

NOTE - These lights are not affected by AHRS.

ATT SEL pushbutton - allows selection of INS or AHRS as the source of attitude signals for the associated ADI. When INS is selected, the pilot's source is INS No. 1 and the copilot's source is INS No. 2. The pilot's ATT SEL switch receives power from the emergency DC bus.

MAG HDG pushbutton - If TACAN, VOR/ILS, or NAV/OFF is selected, MAG HDG pushbutton allows selection of either the INS or AHRS as the source of magnetic heading for the associated HSI.

ADI REP pushbutton - if selected on the pilot's NAVIGATION SELECTOR panel, pilot's FDS repeats indication of copilot's FDS; if selected on copilot's NAVIGATION SELECTOR panel, copilot's FDS repeats indication of pilot's FDS.

NAV/HDG pushbutton - allows selection of heading mode or navigation mode.

The ADI REP Function

The ADI REP function is a feature whereby one pilot can display the same steering and glideslope information on his ADI that is displayed on the other pilot's ADI. Whenever the ADI REP button is depressed, the associated ADI will repeat the same bank and pitch steering commands, flags, and glideslope deviation information that are being presented on the other FDS ADI.

It must be emphasized that attitude, attitude warning, and turn and bank information will not be repeated. The ADI REP function can be used when an ILS/glideslope receiver or flight director computer fails.

HDG SEL/NAV Button

Anytime the HDG SEL/NAV button is positioned to HDG SEL, the FDS will disregard the NAV signals and use the heading marker's setting to compute the bank angle necessary for the new heading. That is, in the HDG SEL mode of operation, the bank steering bar will present steering commands to turn to whatever heading is selected under the heading marker.

A practical use of this feature can be seen during a missed approach situation. With the FDS in the ILS APPROACH mode, the missed approach heading could be set under the heading marker. In the event of a go-around, moving the HDG SEL/NAV button to HDG SEL would provide bank steering information to the missed approach heading.

The HDG SEL position will have no effect on the presentation of the HSI. It merely changes the reference used by the FDS Computer to compute bank steering signals from the desired course to a desired heading. Maximum bank, as commanded by the FDS in HDG SELECT, is 30°.

When the HDG SEL/NAV button is in NAV, the FDS is now armed for capture of the selected NAV/AID. When the aircraft is within the necessary capture zone, the FDS will command intercept and tracking of the desired course or track.

Maximum bank, as commanded by the FDS in NAV, is 30° in all modes except ILS and INS. At glideslope capture, the bank command is further reduced to 7.5°. For INS, the bank angles below 10,000 feet - 23° MAX; 10,000 feet to FL 250 - 15-20°; above FL 250 - 15° MAX.

NAV-OFF/NAV Mode

HSI

The only usable information displayed on the HSI while operating in this mode will be aircraft heading. Aircraft heading will be displayed under the upper lubber line and also as a digital "readout" in the course window. (This is the result of the course arrow head being slaved to the upper lubber line.) The bearing pointer will be slaved to the lower lubber line, the CDI will be centered and the warning flag in view, the distance indicator will be masked, and the heading marker will remain at whatever position it was set, i.e., it will rotate with the compass card until it is manually changed.

ADI

The ADI will be in the basic navigation mode and afford only basic flight indications. The attitude sphere (artificial horizon) and turn and slip (needle and ball) are the only active indications. The VDI, GSI warning flag, bank and pitch steering bars, and course warning flag will all be biased out of view.

Inasmuch as the VDI and bank steering bar are inactive and pulled out of view in this mode, their associated warning flags are also pulled out of view.

VOR-NAV Mode

HSI

With the VOR/ILS-1 button depressed and a VOR station tuned on the VHF-NAV receiver, VOR signals will be available for use by the HSI. By selecting the desired course (either inbound or outbound) in the course window, conventional navigation information will be available to the pilot. Aircraft heading will be presented under the upper lubber line. Course deviation (left or right) will be reflected by the CDI. In this mode, each dot of deviation represents 5° off course. TO-FROM information will be determined and presented by the position of the TO-FROM indicator. Signal reliability is ascertained from the course deviation and warning flag. Magnetic bearing to the station is read directly from the compass card, under the bearing pointer's head. The reciprocal is read at the bearing pointer's tail. The range indicator is inactive and it will be masked.

ADI

In addition to basic flight indications, computed bank information to the selected course (radial) will be reflected on the bank steering bar. The bank steering bar is directional (fly-to-indication) during normal intercepts and will be centered if one of two conditions exists: the aircraft is on course, or the aircraft has been banked sufficiently for a proper intercept angle to the desired course. Bank angle signals, along with course error and course deviation signals, are sent to the FDS, where the necessary bank angle is resolved and the necessary amount of bank angle is deflected on the bank steering bar. The maximum bank, as commanded from the computer for all NAV modes, will be 30°.

Bank steering bar reliability can be ascertained by the course warning flag. When the flag is in view, it indicates either lost or unreliable course or steering information. Failure of the FDS Computer will affect only the bank steering bar, as it does not have inputs to the HSI. In the above condition, the bank steering bar will also be retracted from view as an added precaution.

ILS-NAV Mode

HSI

When a localizer frequency is selected on the VHF-NAV control, the VOR/ILS-1 button depressed, and sufficient localizer signals received, the system is in the ILS-NAV mode. Set the localizer course in the course window. The CDI will display the relative position of the center line. Each dot of deviation in the ILS mode represents a deviation off the center line. The TO-FROM indicator will be out of view, since solving for ambiguity is not a function of the localizer system. The bearing pointer will freeze at its last position. The range indicator will be masked.

As long as the front course approach heading is selected in the course window, the CDI will ALWAYS be directional.

ADI

All of the features discussed with VOR-NAV operations will be active on the ADI (attitude sphere, turn and bank, and bank steering bar). The bank steering bar will command intercept and tracking of the localizer when the aircraft is within slightly less than 2 dots of CDI. As the aircraft intercepts the glideslope, the VDI will appear from the top of the scale and move downward as the aircraft approaches the center of the glideslope. Whenever the aircraft is within 1/4 of a dot from the glideslope centerline, the pitch steering bar will be deflected into view close to the center of the ADI. The FDS Computer is now in the ILS-APPROACH mode, and closer lateral guidance is offered. In this mode, the maximum bank angle that the bank steering bar will command is 15°. As long as the aircraft stays within two dots of deviation, either CDI or GSI, the FDS will remain in the ILS-APPROACH mode.

NOTE: Steering information cannot be used for flying inbound on the back course or outbound on the front course of a localizer.

TAC-NAV Mode

TACAN

With TACAN pushbutton depressed and the TACAN receiver tuned, the HSI will display bearing, course deviation, distance, and to-from information relative to the TACAN station. Each dot displacement on CDI is 5°. The course capture zone is a function of distance. The intercept threshold decreases with distance from the ground station. The intercept starts at approximately 2 dots at 50 miles and is 2/3 dot at 150 miles.

INS-NAV Mode

HSI

Selection of INS-1 or INS-2 will display somewhat different indications than VOR or TACAN on the HSI. The CDI will reflect miles off desired track instead of degrees. Each dot of deviation is 1.5 NM. The desired course will be automatically displayed. The bearing pointer will indicate true track.

The range indicator will be active, and the maximum readout is 999 NM. But, when operating in the INS mode, the actual readout on the distance indicator will be the DME for TACAN No. 1 on the pilot's HSI and the DME for TACAN No. 2 on the copilot's HSI.

ADI

The FDS will command a maximum intercept angle of 45°, regardless of cross-track distance. The ADI indications will remain the same as VOR-NAV, with the understanding that bank steering information will be computed to the desired INS course, as displayed in the course window. The VDI, glideslope warning flag, and pitch steering bar will all be biased out of view.

FDS Malfunctions

Malfunctions affecting the valid operation of the pilot's and copilot's attitude source, the ADI attitude spheres, and the command steering bars are indicated by the illumination of caution lights on the progress display panels and by one or more lights on the fault identification panel.

Attitude Gyro Malfunction

As previously mentioned, furnishes pitch and roll displacement signals directly to the ADIs.

These displacement signals are also furnished to the TPLC, where these signals are compared. The median signal is then selected and sent to both FDS computers and the AFCS.

The TPLC also monitors the ADI sphere relationship to the signal.

> NOTE: For this discussion, the pilot's system will be used. Refer to Fault and Caution panels for indications.

1. Problem

   Pilot's gyro exceeds 5° of deviation, either pitch or roll.

   Display

   The AUTO CAUTION, INS/ATT, and TPLC fault lights will illuminate.

2. Problem

Pilot's ADI sphere fails to drive within 5°, pitch or roll, of the gyro displacement signals.

Display

The AUTO CAUTION and INS/ATT fault lights will illuminate.

If the above deviations were caused by a low voltage condition affecting the GYRO or ADI Amplifier, the ATTITUDE WARNING flag would also be displayed.

FDS Malfunctions

Any malfunction of the Flight Director Computer should be indicated by the illumination of the appropriate FLT DIR fault light and the MAN caution light. If the malfunction affects the bank channel, the ADI Course Warning Flag will also be in view.

FDS Self-Test

The FDSs have the capability of self-test.

The test is simple and short, taking a maximum of 6.5 seconds to complete.

Chapter 12

STALL PREVENTION SYSTEMS

There are two independent stall prevention systems. Each system is independent of the other and is composed of a control panel, an electrically heated angle of attack sensing vane, a computer channel, an overhead panel switch, and a control column shaker. The systems are identified as No. 1 and No. 2. No. 1 system acts on the pilot's control column, and No. 2 acts on the copilot's control column. Separate electrical power sources and emergency shutoff switches are provided for each system. During an approaching stall condition, the control shakers are energized. This imparts vibration to the control columns, sufficiently violent as to be immediately identifiable by the pilots. Power is removed from the pitch trim noseup mode. Placing one switch "OFF" does not affect operation of the other system.

The stall computer receives input signals from the angle-of-attack vanes, the CADC's, and flap and gear position. The stall computer has two alarm schedules predicated on the position of the flaps and gear. In the dirty configuration (gear down and flaps extended beyond 60% plus or minus 8%), shaker onset is initiated as a function of angle of attack. In the clean configuration (gear up or flaps NOT extended beyond 60% plus or minus 8%), mach number is also introduced into the stall warning schedule.

The shakers continue to operate until the aircraft has responded and the angle of attack has been reduced below the level at which the shakers were actuated. When this point is reached, the copilot's shaker function is immediately removed. Approximately three seconds later, in the clean configuration only, the pilot's shaker function is removed. In the event recovery is not initiated at the shaker warning, an audible warning of a stall condition occurs at a computed point that is a function primarily of angle of attack and MACH number.

Stall Prevention System Panel

A STALL PREVENTION SYS panel is provided on the pilots' overhead panel. The panel contains two toggle switches, one for the No. 1 system (PILOT) and one for the No. 2 system (COPILOT). Each switch has two positions: (1) Norm, (2) Off. In the "NORM" position, the applicable system is armed to provide stall warning by control column shaker operation. When either the PILOT or the COPILOT switch on the overhead control panel is placed at "OFF," the corresponding system is deactivated and does not provide stall warning. The "OFF" position is used as a system emergency shutoff, or is used if a system malfunction is suspected. Placing one switch "OFF" does not affect operation of the other system.

Stall Prevention Panels

Two STALL PREVENTION panels are provided, one on the pilot's side console and the other on the copilot's side console. Each panel contains a three-position ("TEST," "NORM," "MACH TEST") toggle switch and a stall warning light. The panel on the pilot's side console provides test and visual stall warning capabilities for the No. 1 system, while the panel on the copilot's side console provides the same capabilities for the No. 2 system. The switch on each STALL PREVENTION panel, when placed in the "TEST" position with the aircraft on the ground, actuates touchdown relays to place the corresponding system in the airborne condition. Shaker operation may then be checked by suitable positioning of the angle-of-attack vane of the system. When the switch is placed in the "MACH TEST" position, the system is similarly placed in the airborne condition, and may be tested for shaker operation at the correct Mach numbers by operating the test switches on the corresponding CADC test panel.

NOTE: Do not place the stall prevention test switch to the "MACH TEST" position if the aircraft is flying at M 0.365 or greater.

The STALL warning light illuminates when the corresponding system has sensed an approaching stall condition and the related computer has transmitted the stall signal to the shaker circuits of that system. The light remains illuminated until the stall condition has been corrected and shaker operation has ceased. The light is included in the instrument dimming circuit, and is dimmed when the instrument lights are dimmed.

System Warning Lights

Individual warning lights are provided on the annunciator panel for each of the stall prevention systems. These consist of a STALL PREV NO. 1 light and a STALL PREV NO. 2 light. The STALL PREV NO. 1 or STALL PREV NO. 2 light is illuminated if a failure occurs in the computer circuits or the vane heating circuit of the corresponding system, or if the corresponding switch on the overhead control panel is off. Placing the applicable switch on the overhead control panel in the "OFF" position deactivates that system and illuminates the annunciator light.

Stall Prevention Computer

The stall prevention computer, on the left side of the center avionics rack, contains two completely independent channels. The No. 1 channel receives inputs from the left angle-of-attack vane transducer, the left outboard flap position switch, nose gear position switch, the No. 1 CADC, and from a yaw rate gyro. The No. 2 channel similarly receives inputs from the right angle-of-attack vane transducer, the right outboard flap position switch, nose gear position switch, the No. 2 CADC, and from a yaw rate gyro.

The computer outputs operate the stall prevention systems and also provide visual and audible warning if the spoiler lever is armed during an approach to a stall condition and the aircraft is airborne.

Angle of Attack Vanes

An angle-of-attack vane is mounted on each side of the forward fuselage. The left vane provides angle of attack signals from the No. 1 stall prevention system, and the right vane performs the same function for the No. 2 stall prevention system. The vanes are electrically heated for protection from icing. The vanes are positioned by airflow over them during flight. Vane movement rotates a shaft in each vane assembly to which a transducer is coupled. The transducer transmits continuous local angle-of-attack signals to the related channel of the stall prevention computer so long as the applicable system is energized.

Stall Prevention System Operation

When either computer channel transmits a stall signal to the corresponding system, the shaker motor of that system is energized. When the stall condition has been corrected, the stall prevention computer removes the shaker signal and the shaker motor of that system is deenergized.

Audible Stall Warning

On aircraft modified by T.O. 503, an audible stall warning is provided to indicate that immediate action should be taken to restore the airplane to a safe angle of attack. The audible warning is provided through the intercom system and the overhead warning horn and loudspeaker. The audible stall warning does not occur unless the shaker has already energized. The audible warning can be energized by either or both of the stall prevention computer channels whenever one of the following conditions is reached:

1. After 5.0 seconds of shaker operation (gear or flaps or both are up) or at a computed point which is a function of MACH number and angle of attack.

2. With the flaps extended beyond 25 degrees and the gear down, stall event audible warning is a function of angle of attack.

In both the above modes, the computer activation points are adjusted by angle of attack rate and yaw rate, which serve to anticipate a stall condition. A higher rate of change in angle of attack results in activation of the stall event warning at a lower angle of attack, as does an increase in yaw rate.

An audible stall warning is also provided if the spoiler lever is armed during an approach to a stall condition.

Any time the audible stall warning activates, the stall warning lights, located on the pilot's and copilot's side console panels, also illuminate.

Chapter 13

FUEL SAVINGS ADVISORY SYSTEM (FSAS)

(Figures 3-28 thru 3-33)

With the addition of TCTO 602 to the C-141 fleet, fuel savings of approximately 2.5 percent can be achieved in most C-141 operations. Over a period of time, this will reduce the overall fuel usage. The FSAS system, in performing its function, also will aid the aircrew in all phases of flight.

FSAS - INS Navigation Aiding

The FSAS system will make inertial navigation less of a work load requirement on the aircrew by increasing the number of waypoints that can be programmed before and after takeoff. The FSAS has a total memory capability of 40 waypoints. As the INS system uses a waypoint, the FSAS system will program into the INS the next waypoint automatically. The programming of waypoints is also reduced by allowing the aircrew to select waypoints from one of a possible 200 TACAN stations that can be stored in the memory of the FSAS computer. The TACAN stations in the FSAS computer are of two types. The first group of 160 TACAN stations are preprogrammed and cannot be changed. The second group of 40 TACAN stations can be entered and changed at the operator's convenience. Each TACAN station can be called up for display by simply entering the three- or four-letter ICAO identifier for that station.

FSAS TOLD Card Information

The FSAS will complete a takeoff and landing data (TOLD) card for the aircrew when the conditions affecting the takeoff and type of takeoff desired are selected and inserted into the system. The TOLD card can be based on a reduced EPR or TRT takeoff with or without bleed air penalties. When performance parameters are exceeded, the system will warn the aircrew to that effect. For example, "Gross weight is limited by critical field length." If the takeoff is still required with the conditions present, the system will compute a new takeoff (STRT) based on setting TRT prior to break release. If parameters for takeoff are exceeded, the system will warn the crew and display the gross weight required to meet all parameters. The FSAS will also compute emergency return and landing data based on real-life conditions of flap settings, anti-skid on or off, and others.

Flight Planning

The FSAS system will help the pilots compute the exact fuel required to fly either an entire mission or a leg segment. The winds and temperatures between waypoints, desired flight altitude, climb, cruise and descent profiles to be flown can be entered to determine the fuel requirements for an entire flight.

The system will also display fuel, time and range remaining based upon present conditions or climbing to optimum altitudes.

Flight Profiles

Once the aircraft is 1,500' above ground level, the full capabilities of the FSAS can be taken advantage of. If the autopilot and auto throttle systems are turned on and engaged with the FSAS, the aircraft can automatically climb in one of four modes up to a selected altitude. The modes will allow for climb profiles based on 100 percent maximum specific range (economy). In addition to maximum rate of climb, maximum angle of climb, and a manual climb mode, the operator selects the airspeed or mach desired and the aircraft will maintain it. Once the aircraft is within 1,000' of the selected level-off altitude, the aircrew selects one of three cruise modes available, engages the mode selected, and the aircraft will then level off and smoothly transition into that cruise mode. The types of cruises available are: an economy cruise (100 percent maximum specific range), a long range cruise (99 percent maximum specific range), and a manual cruise mode. When it comes time to descend, one of four descend modes is selected and engaged. The auto throttles are disconnected and the throttles retarded manually, then the aircraft can fly a descend in either an economy descend, enroute descend, manual descend or rapid descend.

FSAS - INS Radar Displays

The FSAS system will allow for the display of information on the AN/APS-133 radar scope. Some of the information that can be displayed, if the operator desires, is: reference ground speed, actual ground speed, optimum altitude, airspeed or mach, INS true heading, engine pressure ratio, and variable waypoint latitude and longitude. As an aid to the aircrew, the FSAS will also display on the radar scope a scale display of the aircraft's position with course lines and waypoint/TACAN symbols -- this is to show the aircraft's present position and the present position of relative points of interest. The aircrew can then evaluate graphically the aircraft's course and position and insert changes as needed. A trend vector can be displayed that will show, based on present conditions, where the aircraft will be in 30-, 60-, and 90-second intervals. An air refueling track can be displayed as well as the airdrop initial point, point of impact, and trailing edge positons. If desired, a runway marker for the destination runway can be displayed to aid in the location of the runway.

Malfunction Analysis and Data

The FSAS will determine faults in related aircraft systems and advise the aircrew of the problem areas. Example: "A/P Pitch and Roll" (Autopilot pitch and roll malfunction), "ATS-INOP" (the auto throttle system is inop), etc. The display of engine-out performance data can be evaluated at any time. If an actual engine loss is experienced, the FSAS will display actual altitude, airspeed, and EPR for three-engine operation. If the loss of another engine is experienced, the system will now give the aircrew 2 ENG VMCA.

FSAS Components

The four major components of the FSAS are the FSAS computer, the FSAS display interface unit, the display interface control unit, and the FSAS/INS control display unit.

FSAS Computer

The FSAS computer, on the upper right side of the forward equipment rack in the left hand underdeck area, handles all calculations and displays of the FSAS.

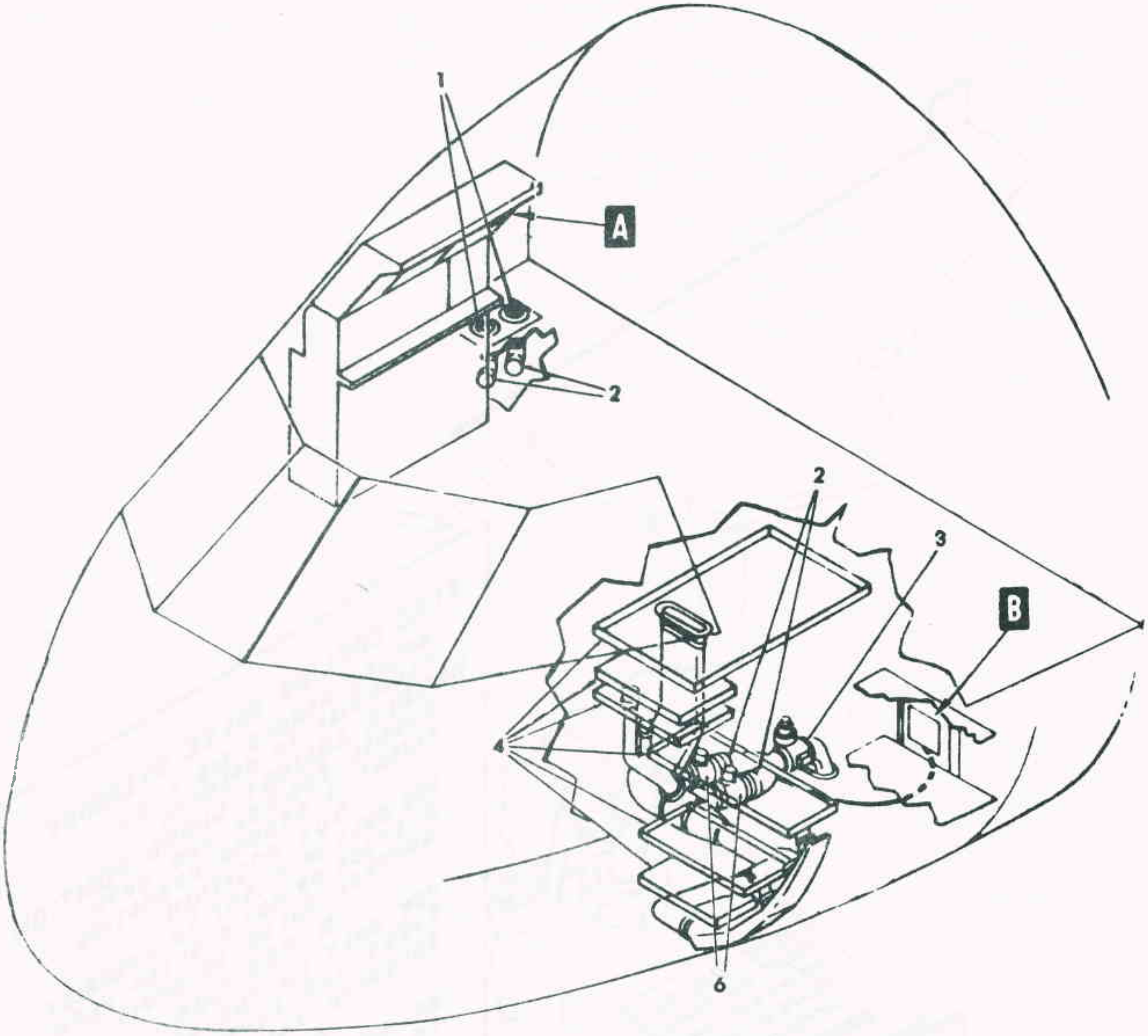
FSAS Display Interface Unit

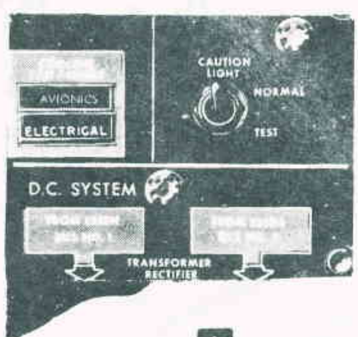
The FSAS display interface unit (DIU), on the left side of the center avionics rack in the left hand underdeck area, converts and then displays information from the FSAS computer to the AN/APS-133 radar scope. The system is designed so that even with the radar off, the DIU will display information on the scope to be viewed by the aircrew.

FSAS Display Interface Control Unit

The display interface control unit (DICU), in the center row of the pilot's center console just aft of the autopilot control panel, controls the FSAS system's operation, on or off, and allows the selection of the different displays and information to be presented on the AN/APS-133 radar scope. The DICU controls the selection of an altitude for an altitude alerting feature. This feature will warn the aircrew both visually and orally if the aircraft reaches or departs a selected altitude. The DICU also contains a altitude alert inhibit switch, which will momentarily inhibit the oral altitude alerting feature. A slewing switch, which is used to move a variable marker on the radar screen for position selections, is also on the DICU. The DICU also has three lights advising the crew of the flight profile being flown, climb, cruise, and descent, and a battery light. The battery light advises that the INS selected is operating off of INS battery power.

Just aft of the DICU is the FSAS/INS control display unit (CDU), which controls and displays all information provided by the FSAS to the aircrew. This CDU is also the pilot's only direct link with the INS system, as it replaces the INS CDU No. 1. The CDU contains INS mode/function keys, FSAS mode keys and miscellaneous alfa numeric keys for entering data into the FSAS/INS systems. It also contains a small cathode ray tube (CRT) display screen, a brightness control, and a green engage key. The engage key is used to couple the FSAS with the autopilot system.



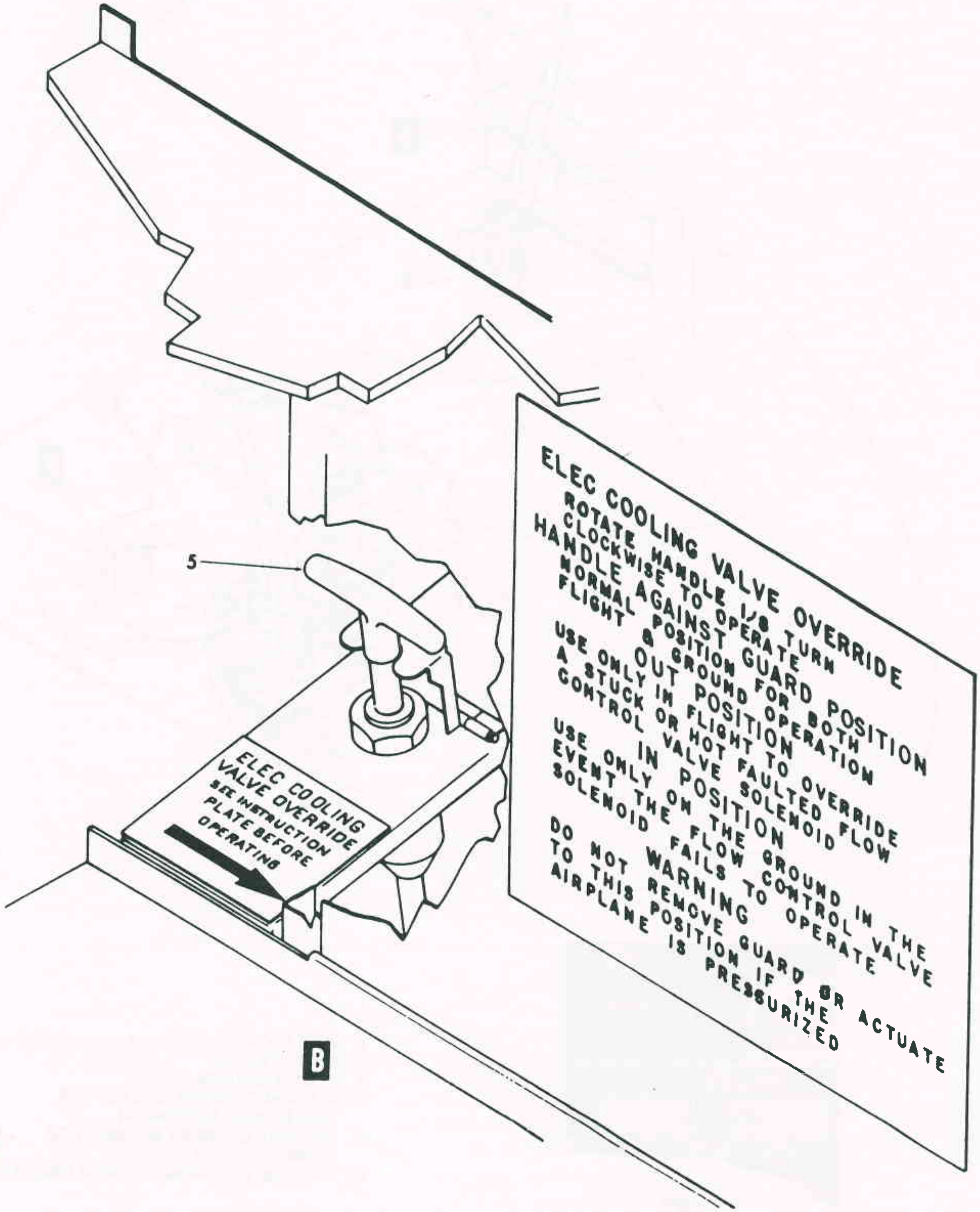


A

1. ELECTRICAL EQUIPMENT COOLING FANS
2. CHECK VALVES
3. FLOW CONTROL VALVE
4. PLENUM CHAMBER SHELVES FOR ELECTRONIC EQUIPMENT
5. MANUAL OVERRIDE HANDLE FOR FLOW CONTROL VALVE
6. AVIONICS EQUIPMENT COOLING FANS

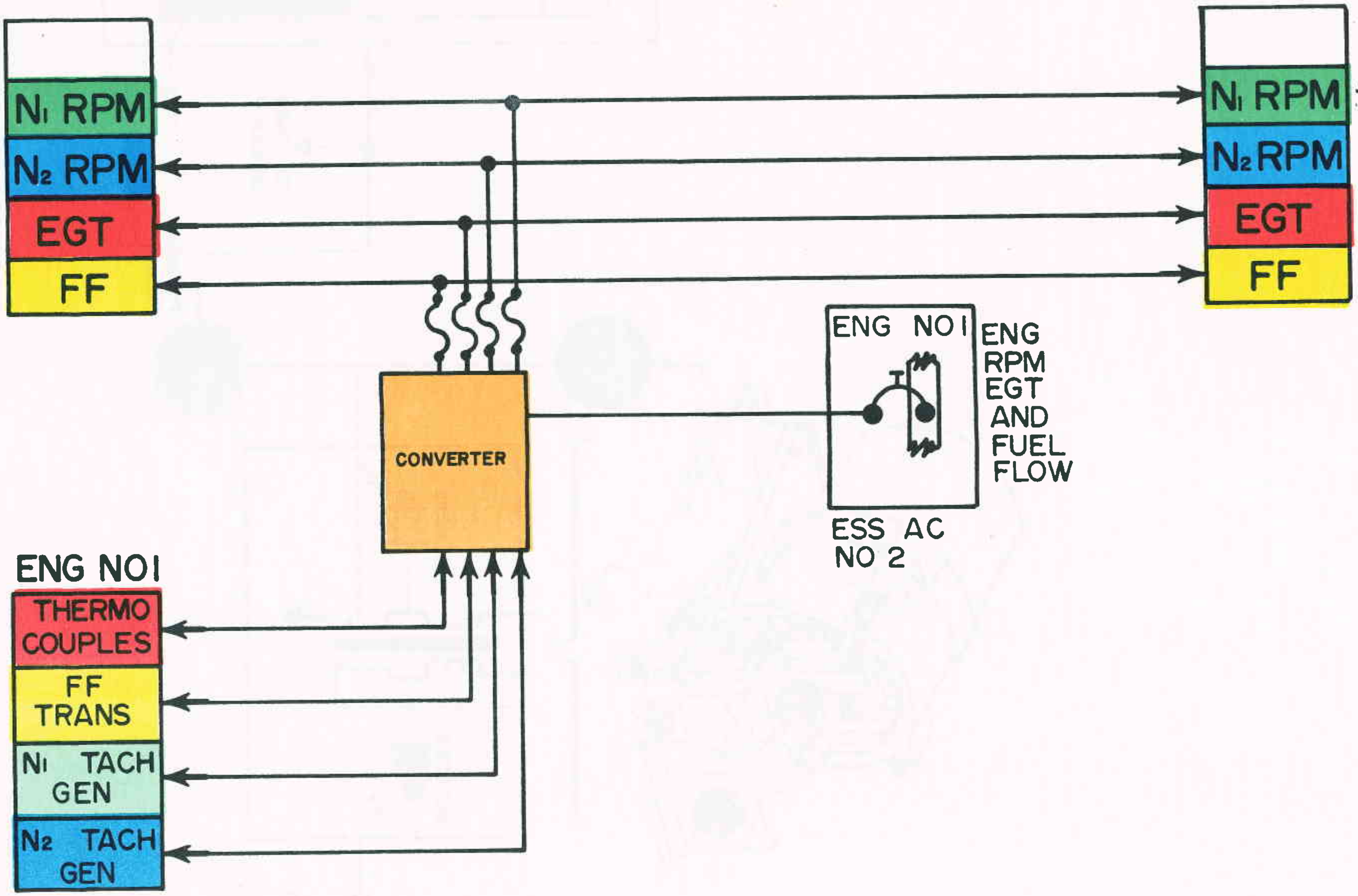
AVIONICS & ELECTRICAL COMPARTMENTS
COOLING FAN LOCATIONS (sheet 1 of 2)

Figure 3-1



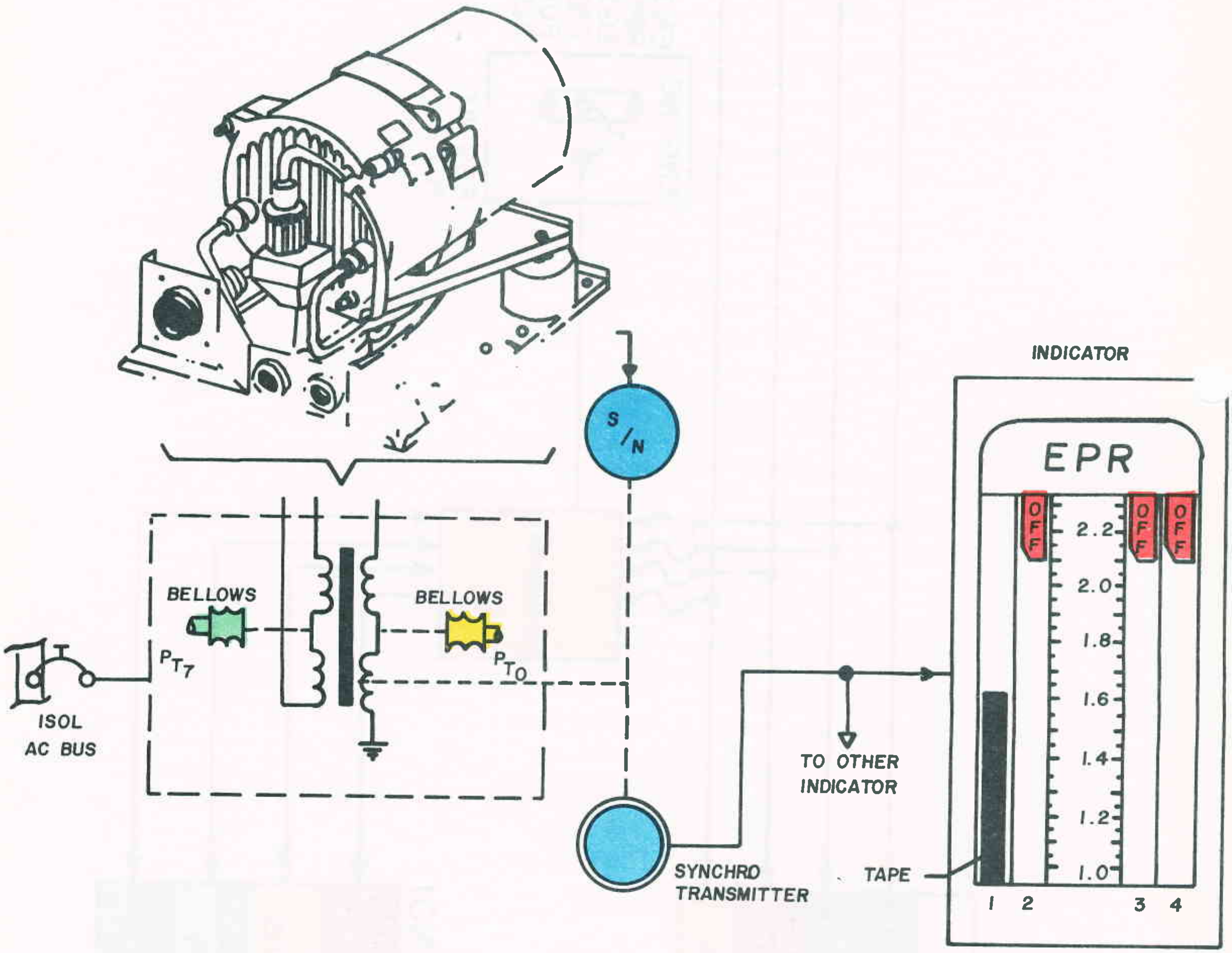
Avionics and Electrical Compartments Cooling Fan Locations (Sheet 2 of 2)

Figure 3-2



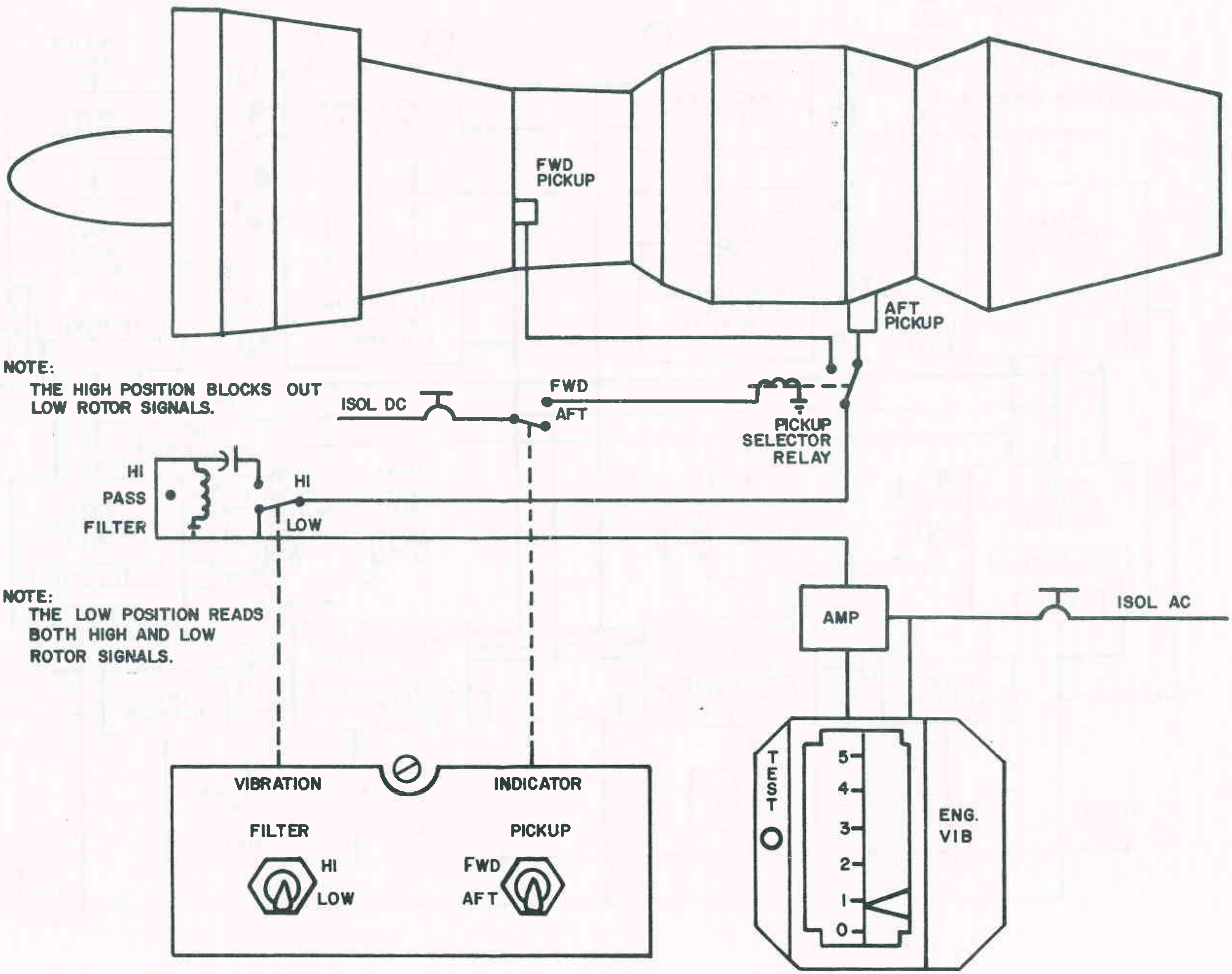
N1 RPM
N2 RPM
EGT
FF
N1 RPM
N2 RPM
EGT
FF
CONVERTER
ENG NO1
ENG RPM EGT AND FUEL FLOW
ESS AC NO 2
ENG NO1
THERMO COUPLES
FF TRANS
N1 TACH GEN
N2 TACH GEN

Figure 3-3



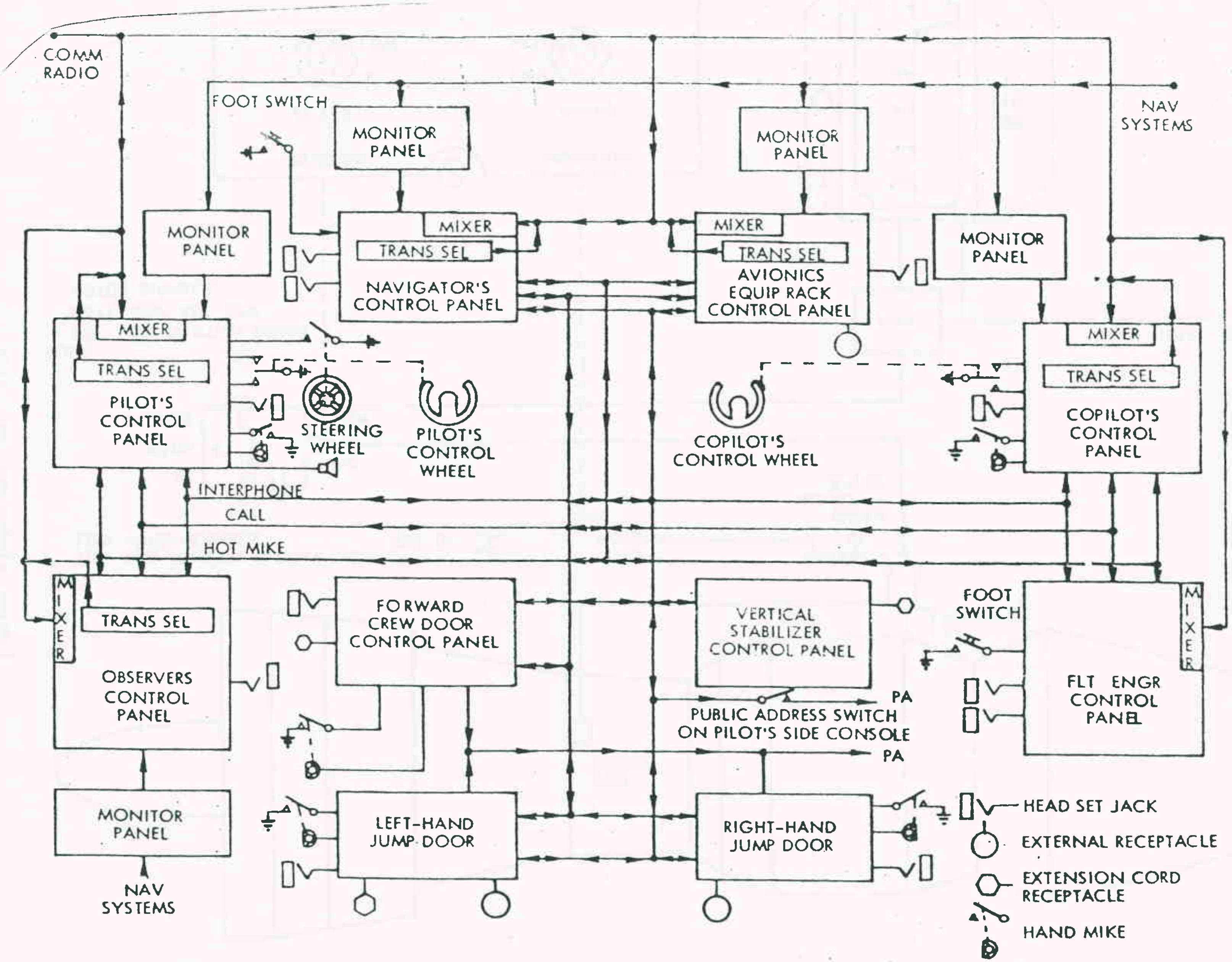
INDICATOR
EPR
OFF
OFF
OFF
2.2
2.0
1.8
1.6
1.4
1.2
1.0
1
2
3
4
TAPE
S/N
BELLOWS
BELLOWS
P_{T7}
P_{TO}
ISOL
AC BUS
TO OTHER
INDICATOR
SYNCHRO
TRANSMITTER

Figure 3-4



ENGINE VIBRATION INDICATOR

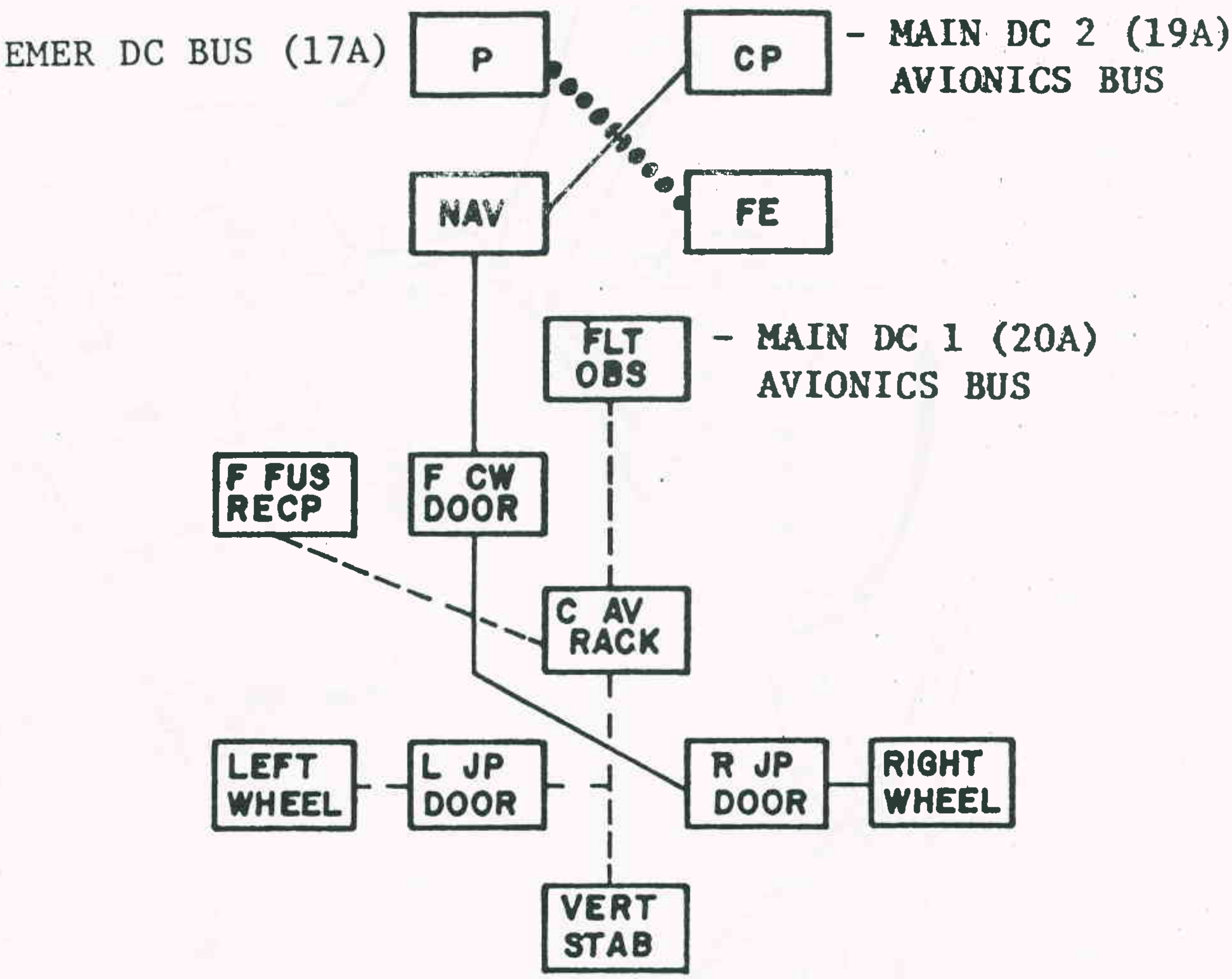
Figure 3-5



Intercommunication System Schematic Diagram

Figure 3-6

C-141 Interphone System



UNDERWATER ACOUSTIC BEACON SYSTEM

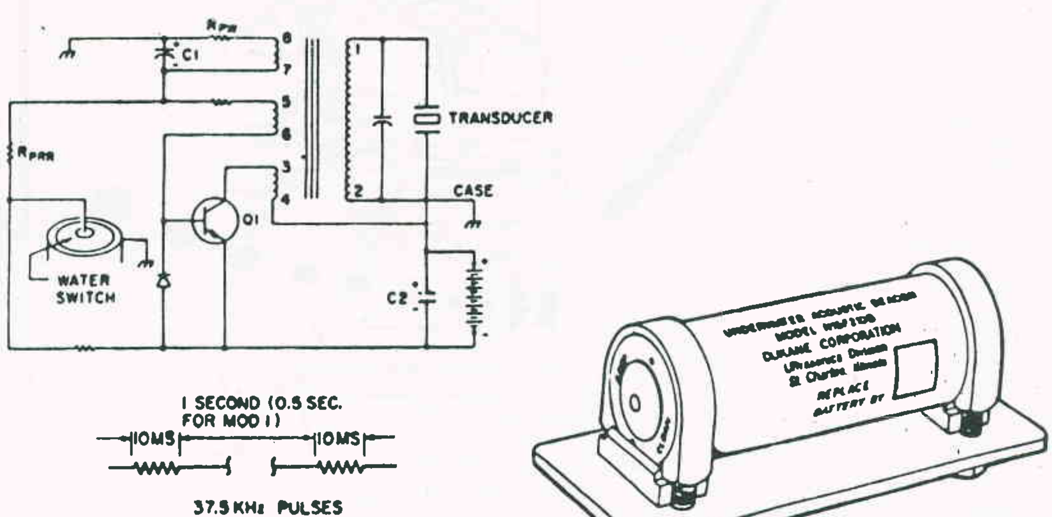
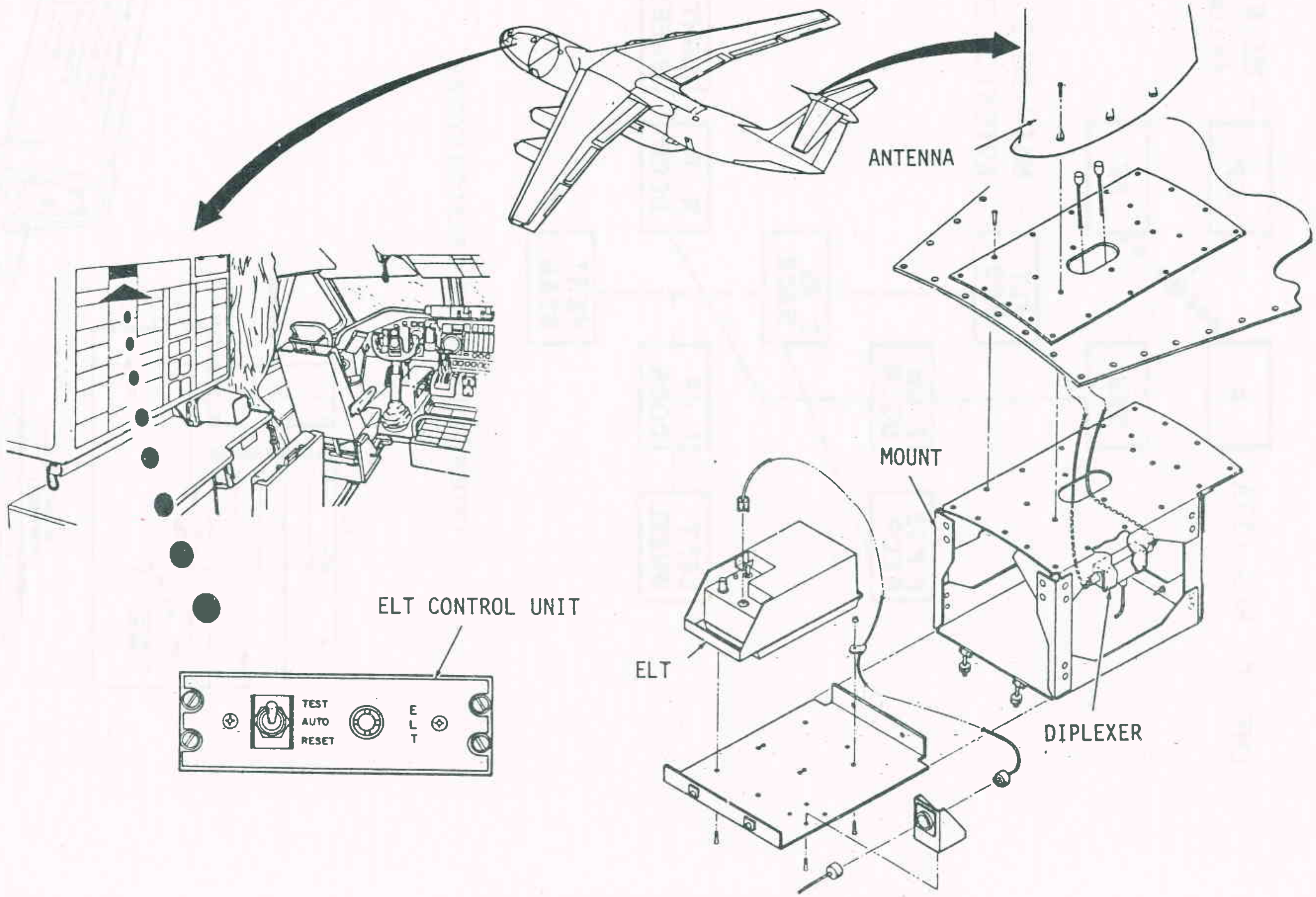
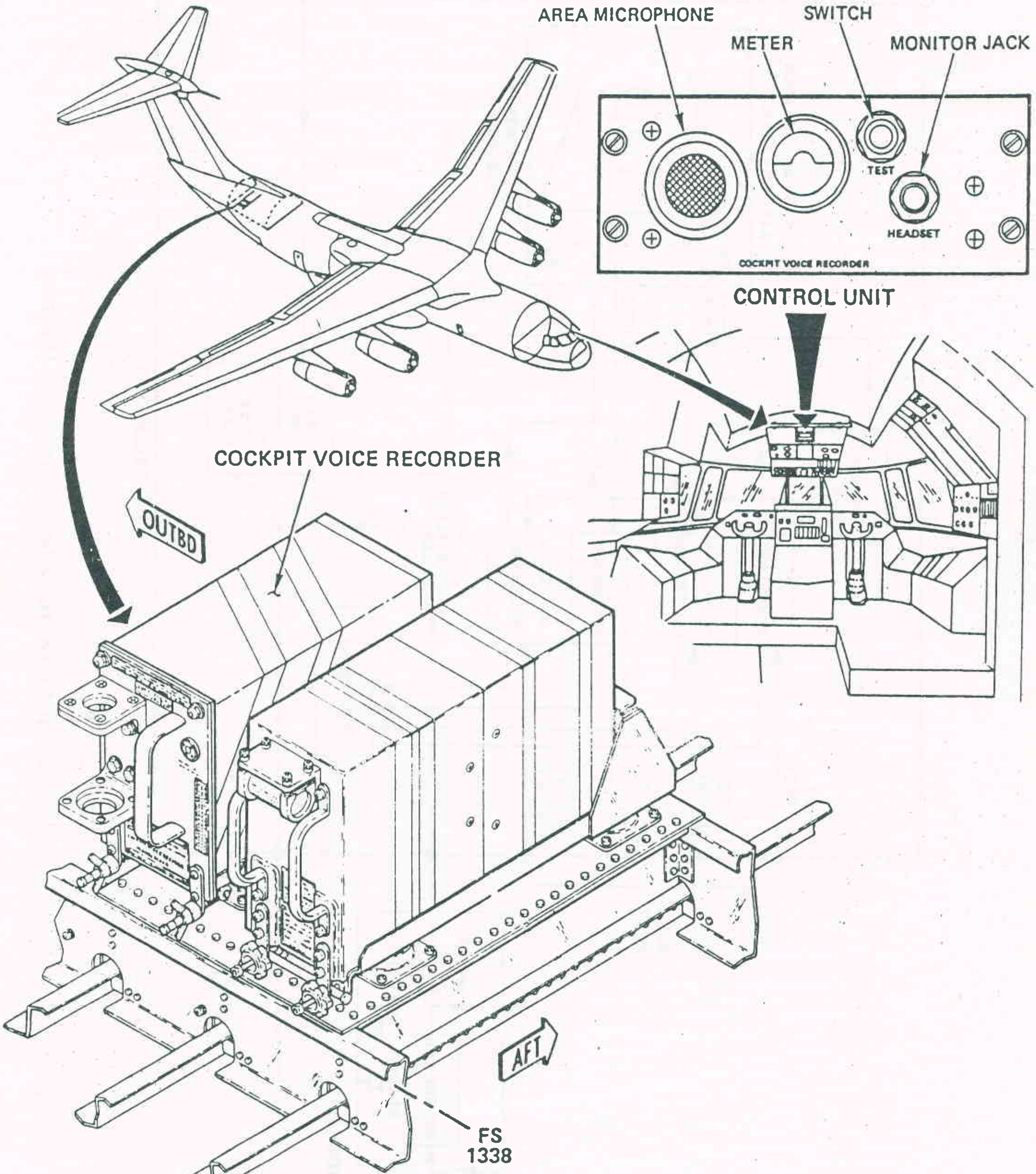


Figure 3-7



Emergency Locator Transmitter System

Figure 3-8



Cockpit Voice Recorder System Component Locations

Figure 3-9

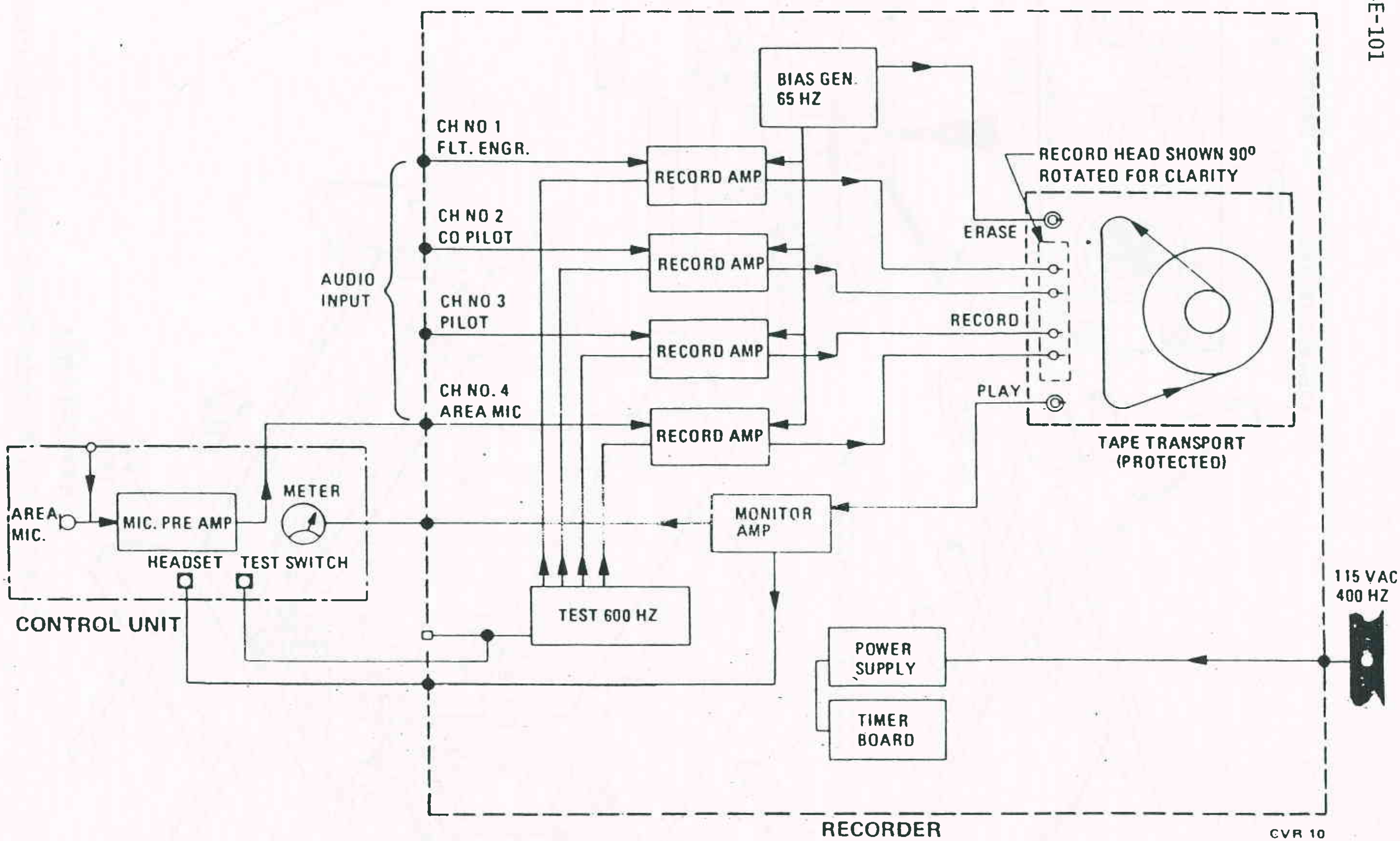
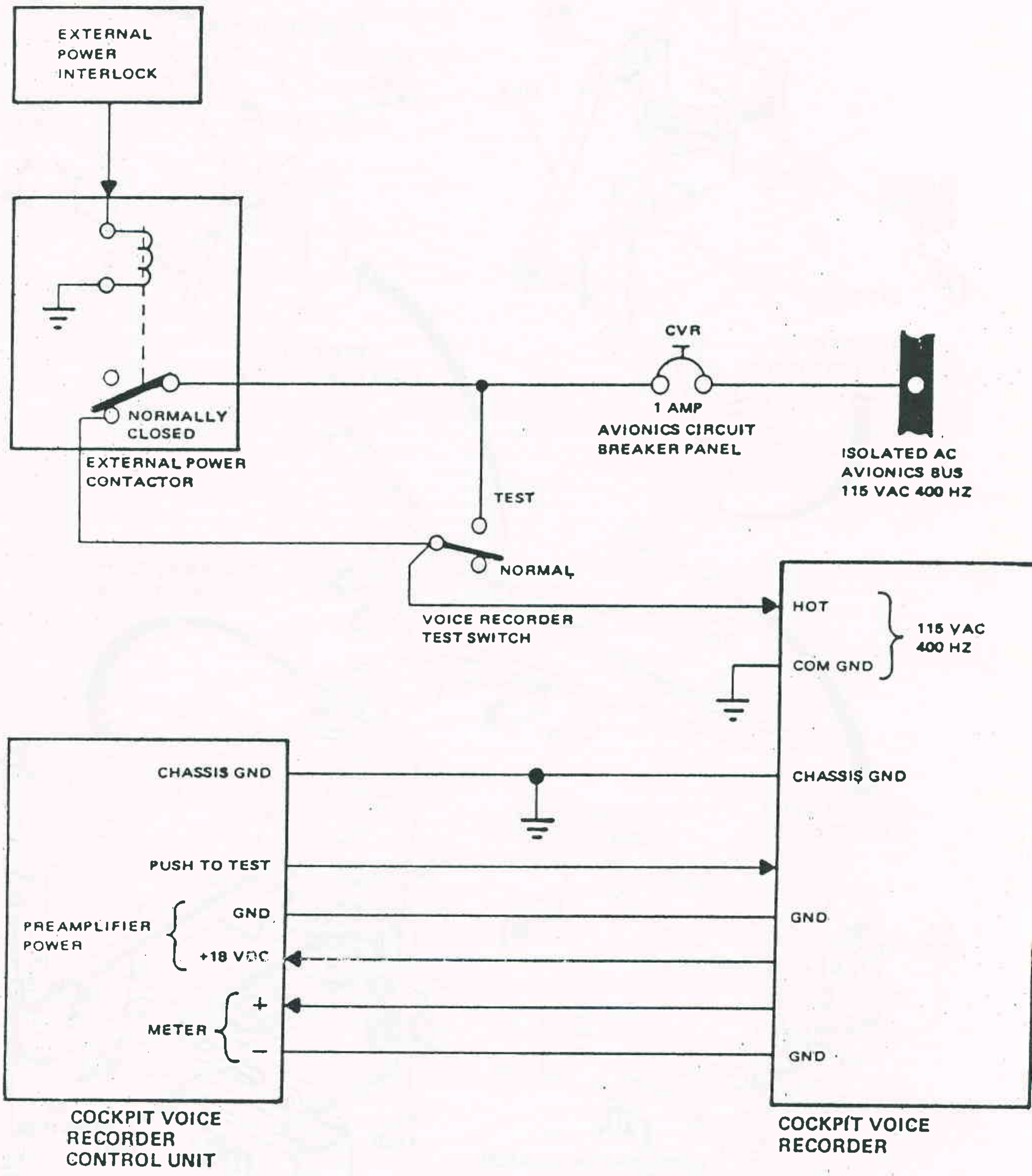


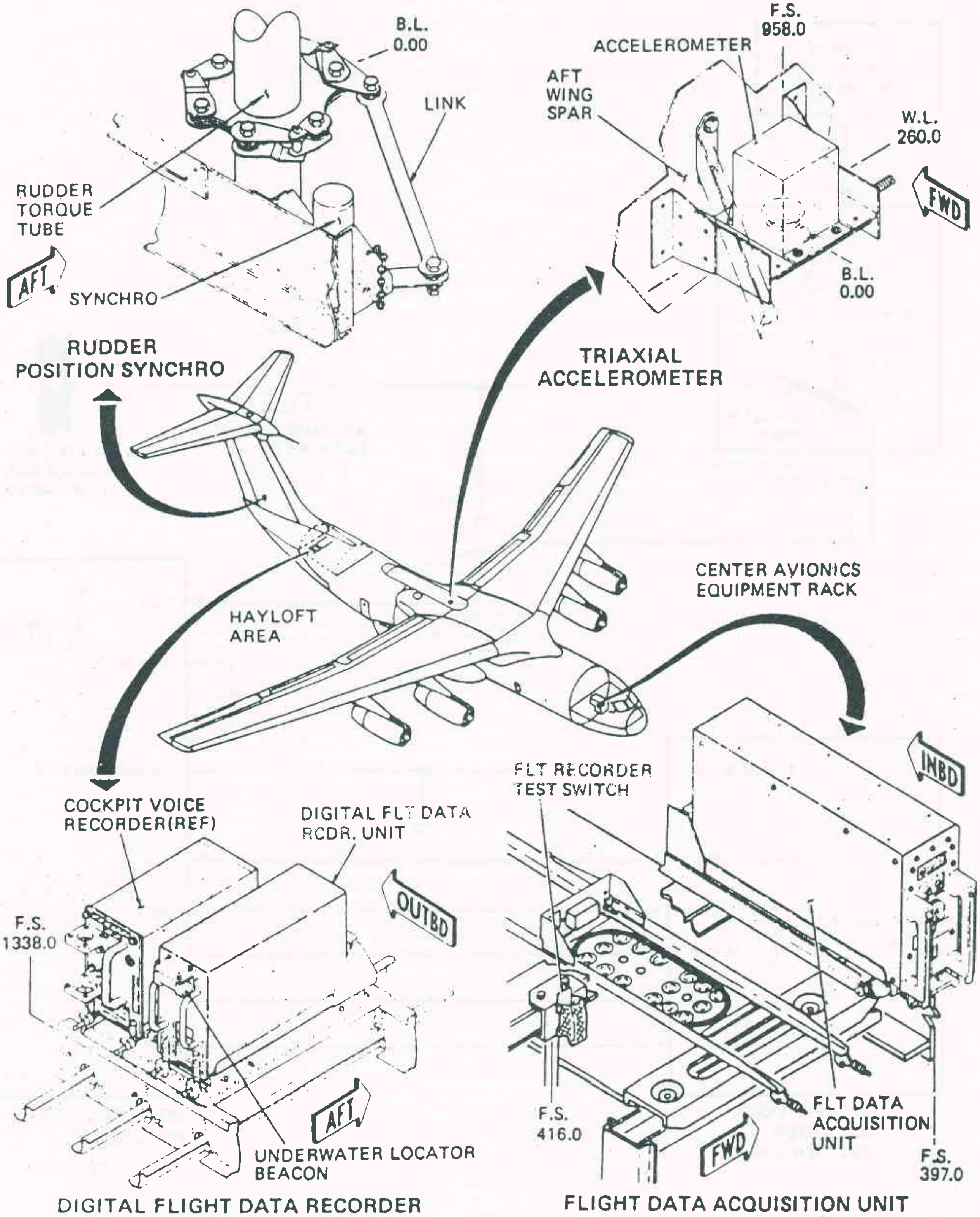
Figure 3-10

Cockpit Voice Recorder System Block Diagram



Cockpit Voice Recorder System Power Sources and Switching Diagram

Figure 3-11



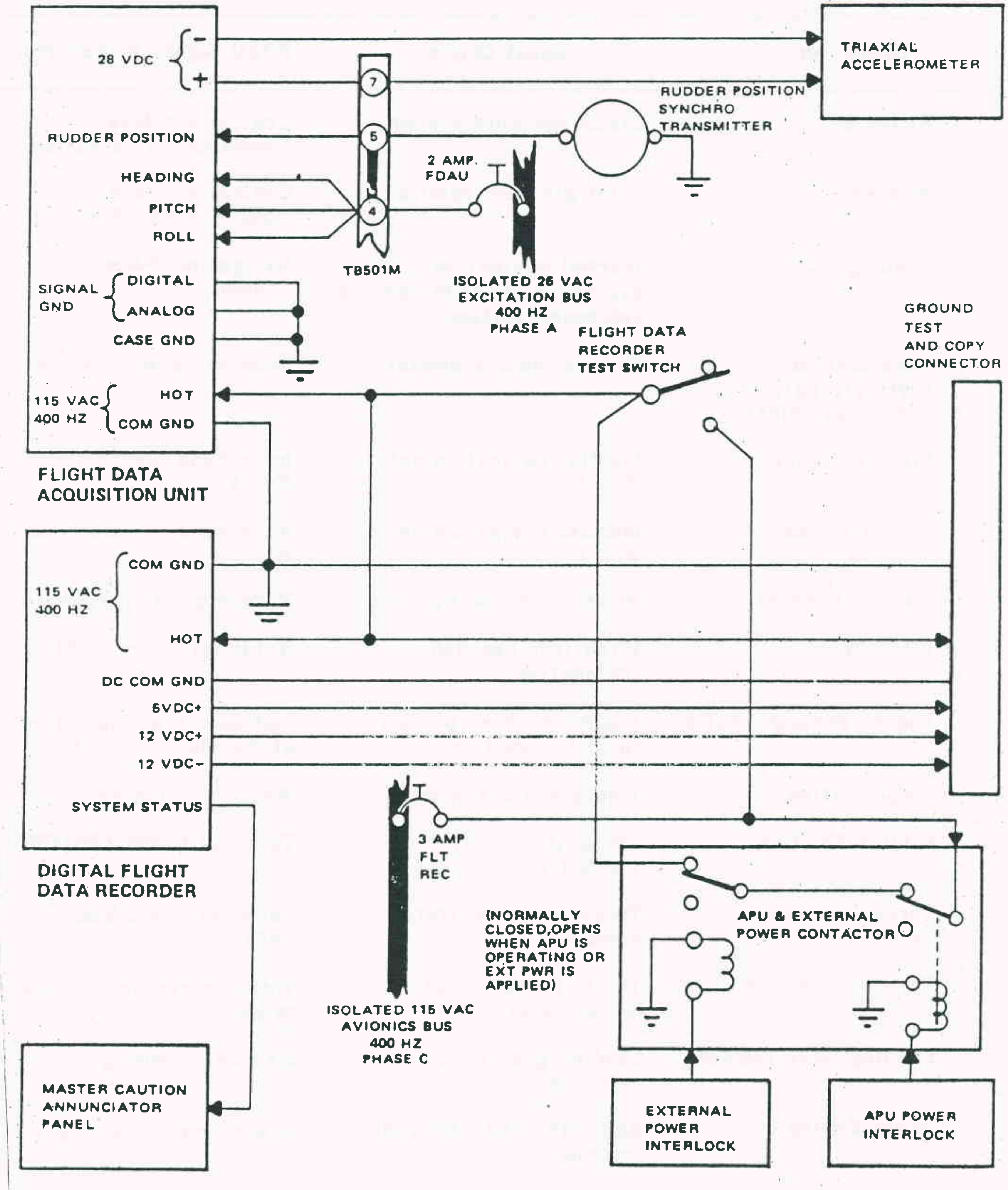
Digital Flight Data Recorder System Component Locations

Figure 3-12

Flight Data Recording Parameters

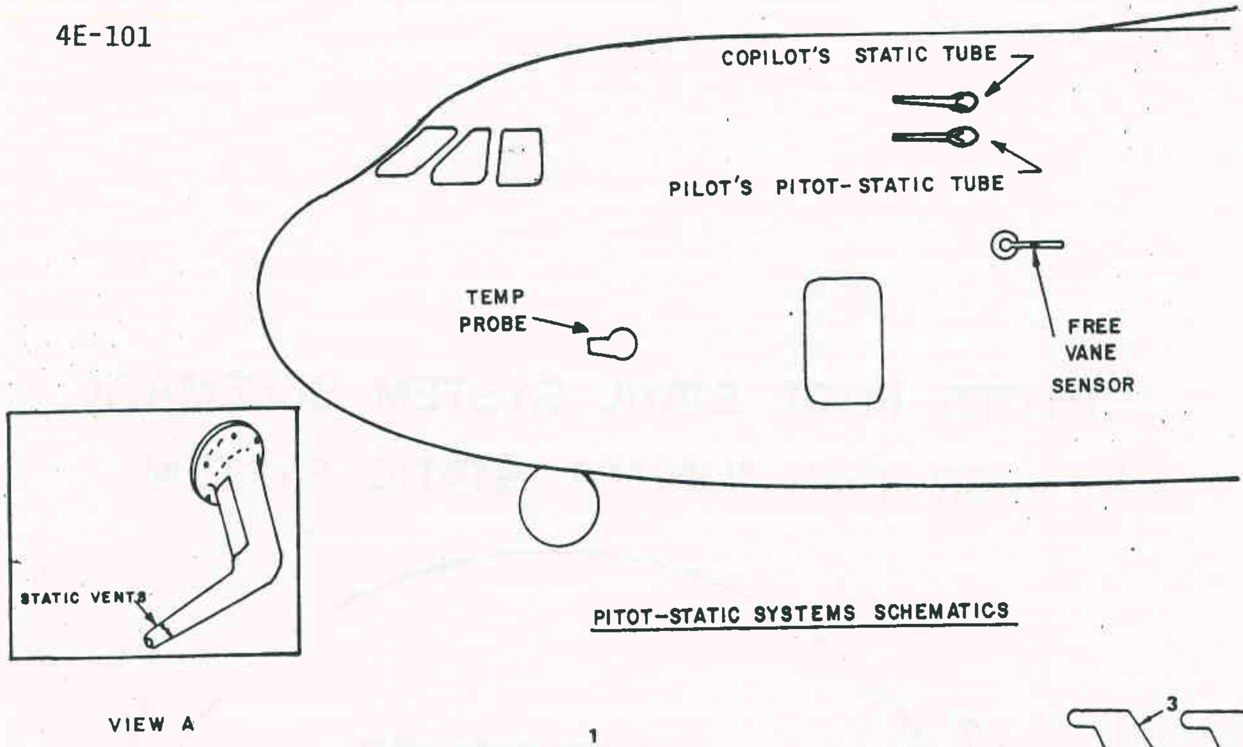
Parameter	Signal Origin	FDAU Signal Pickup Point
Altitude	Static pressure system	Central air data Computer No. 2 wiring
Airspeed	Pitot pressure system	Central air data Computer No. 2 wiring
Heading	Inertial navigation system or attitude heading reference system	Navigation J-box (TB8BP)
Acceleration - Vertical, Lateral, and Longitudinal	Triaxial accelerometer	Triaxial accelerometer
Pitch Attitude	Inertial navigation unit No. 1	Interphone junction box No. 1
Roll Attitude	Inertial navigation unit No. 1	Interphone junction box No. 1
Rudder Position	Rudder position synchro	Rudder position synchro
Pitch Trim	Pitch trim position transmitter	Autopilot J-box (A21)
Engine Pressure Ratio	Engine mounted pressure ratio transmitter	Instrument wiring splice at FS 400
Flap Position	Flap position synchro	AWLS junction box
Spoiler Position	LH spoiler position transmitter	Terminal board (TB24BB)
Thrust Reversers - Extended	Thrust reverser status circuit	Terminal board assembly (A69)
Thrust Reversers - Locked	Thrust reverser not locked relay	Thrust reverser not locked relay
Landing Gear Positon	Landing gear limit switches	Instrument wiring
Radio Keying	HF, VHF, and UHF radio systems	Interphone J-box (A22)
Aircraft Identification	Discrete input logic state	Terminal board jumpers

Figure 3-13

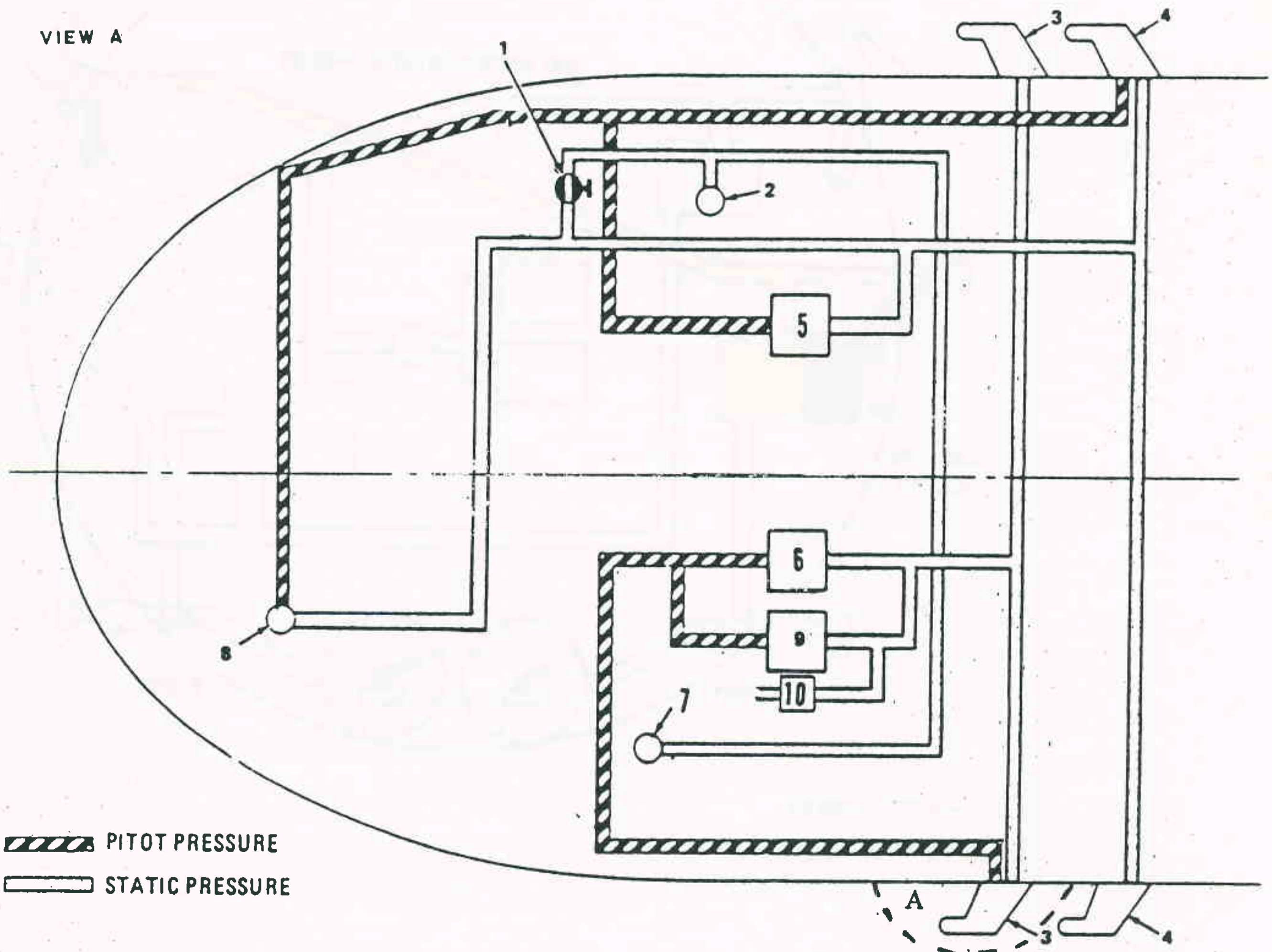


Digital Flight Data Recorder System Power Sources and Switching Diagram

Figure 3-14



PITOT-STATIC SYSTEMS SCHEMATICS



1. MANUAL SHUTOFF VALVES
2. FLIGHT ENGINEER'S ALTIMETER
3. PILOT'S PITOT-STATIC TUBES (UPPER RIGHT AND LOWER LEFT SIDES)
4. COPILOT'S PITOT-STATIC TUBES (UPPER LEFT AND LOWER RIGHT SIDES)
5. NO. 2 CENTRAL AIR DATA COMPUTER
6. NO. 1 CENTRAL AIR DATA COMPUTER
7. NAVIGATOR's ALTIMETER
8. STANDBY AIRSPEED INDICATOR
9. ALTITUDE/AIRSPEED TRANSDUCER
10. CABIN DIFFERENTIAL PRESSURE TRANSDUCER

Figure 3-15

PILOTS PITOT STATIC SYSTEM SCHEMATIC OR THE #1 PITOT STATIC SYSTEM

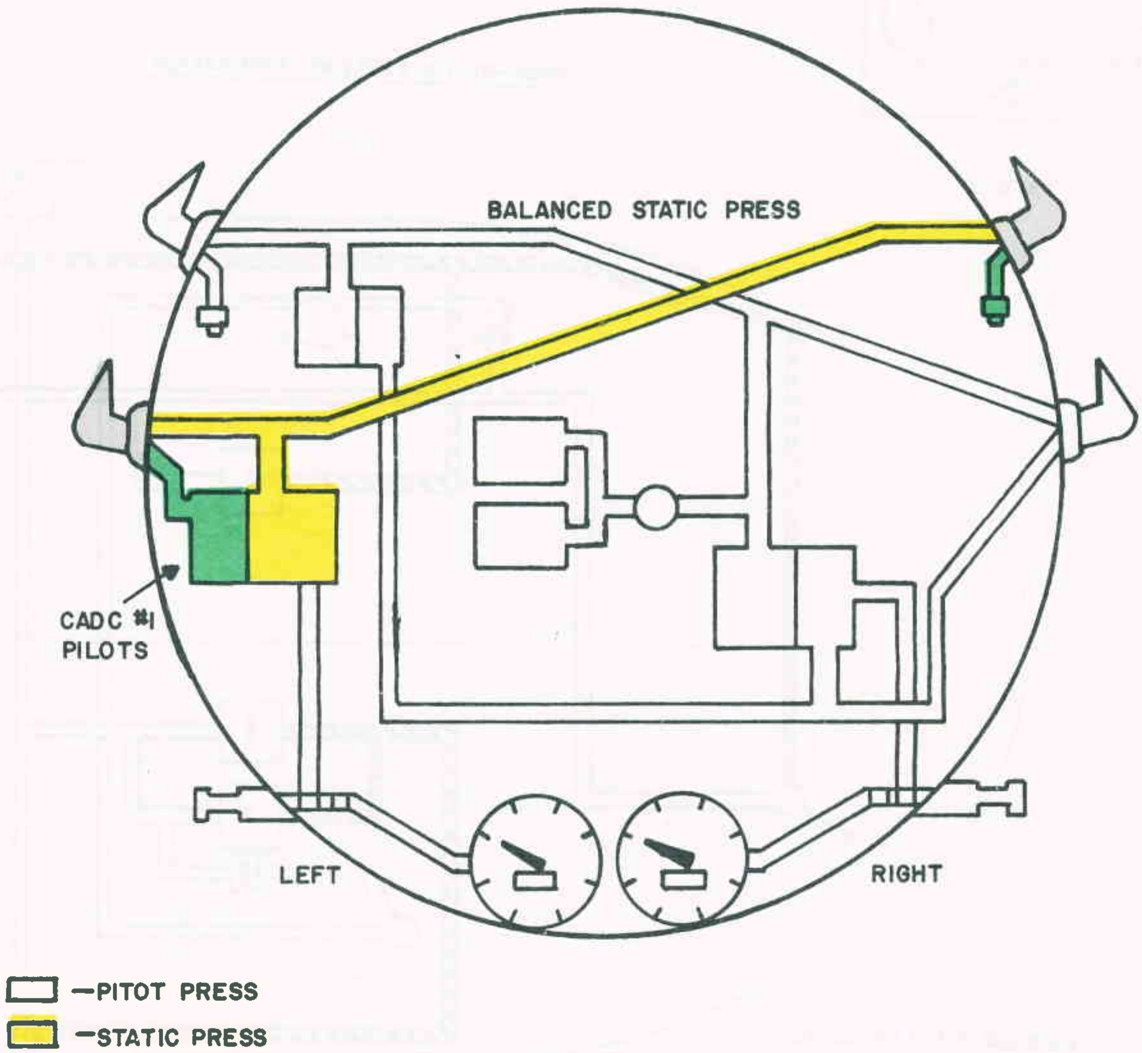


Figure 3-16

COPILOTS PITOT STATIC SYSTEM SCHEMATIC OR THE #2 PITOT STATIC SYSTEM

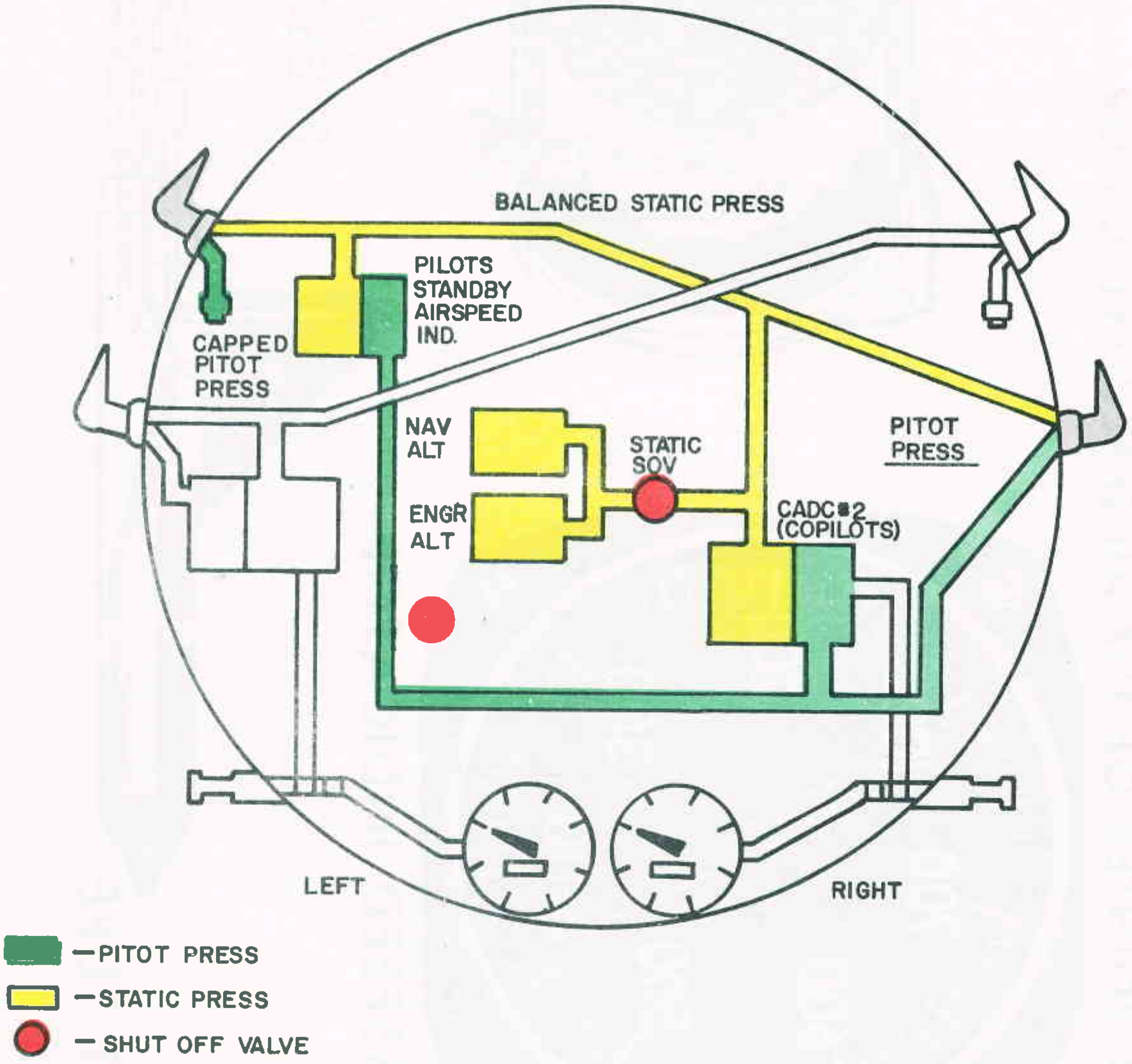
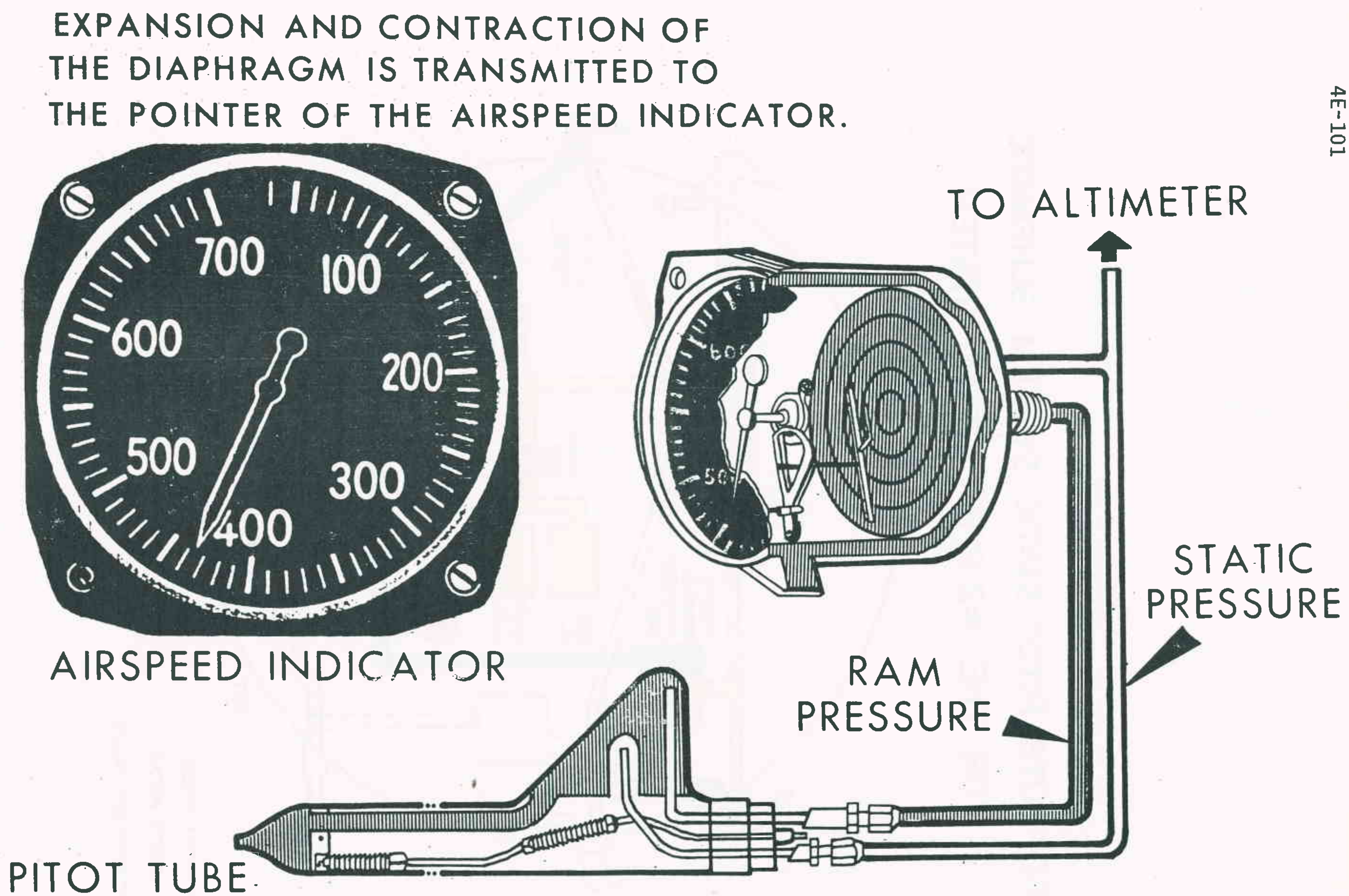
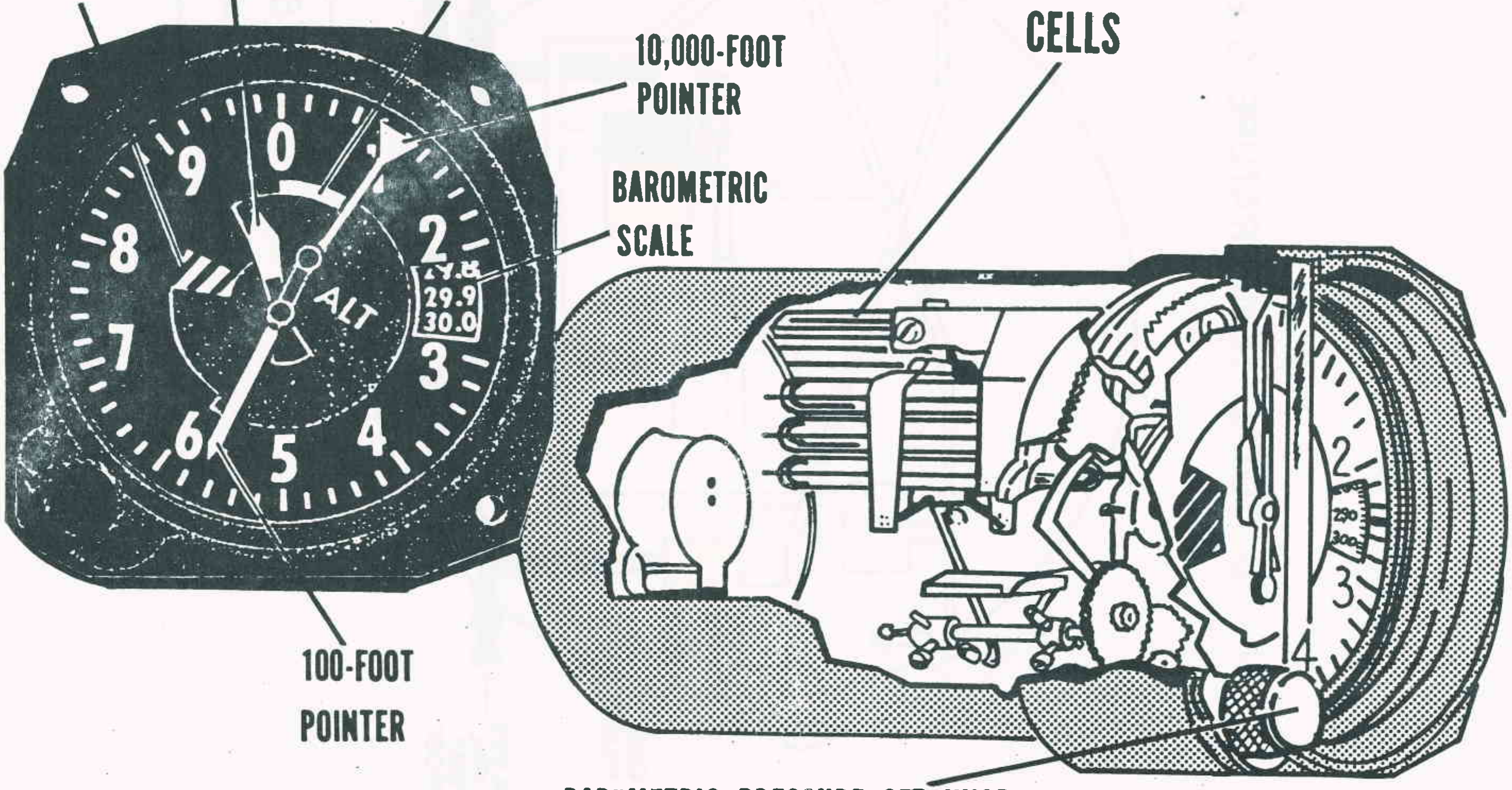


Figure 3-17



OPERATING PRINCIPLE OF THE AIRSPEED INDICATOR

Figure 3-18



THREE-POINTER ALTIMETER

Figure 3-19

4E-101

C-141 B TOTAL TEMPERATURE SYSTEM

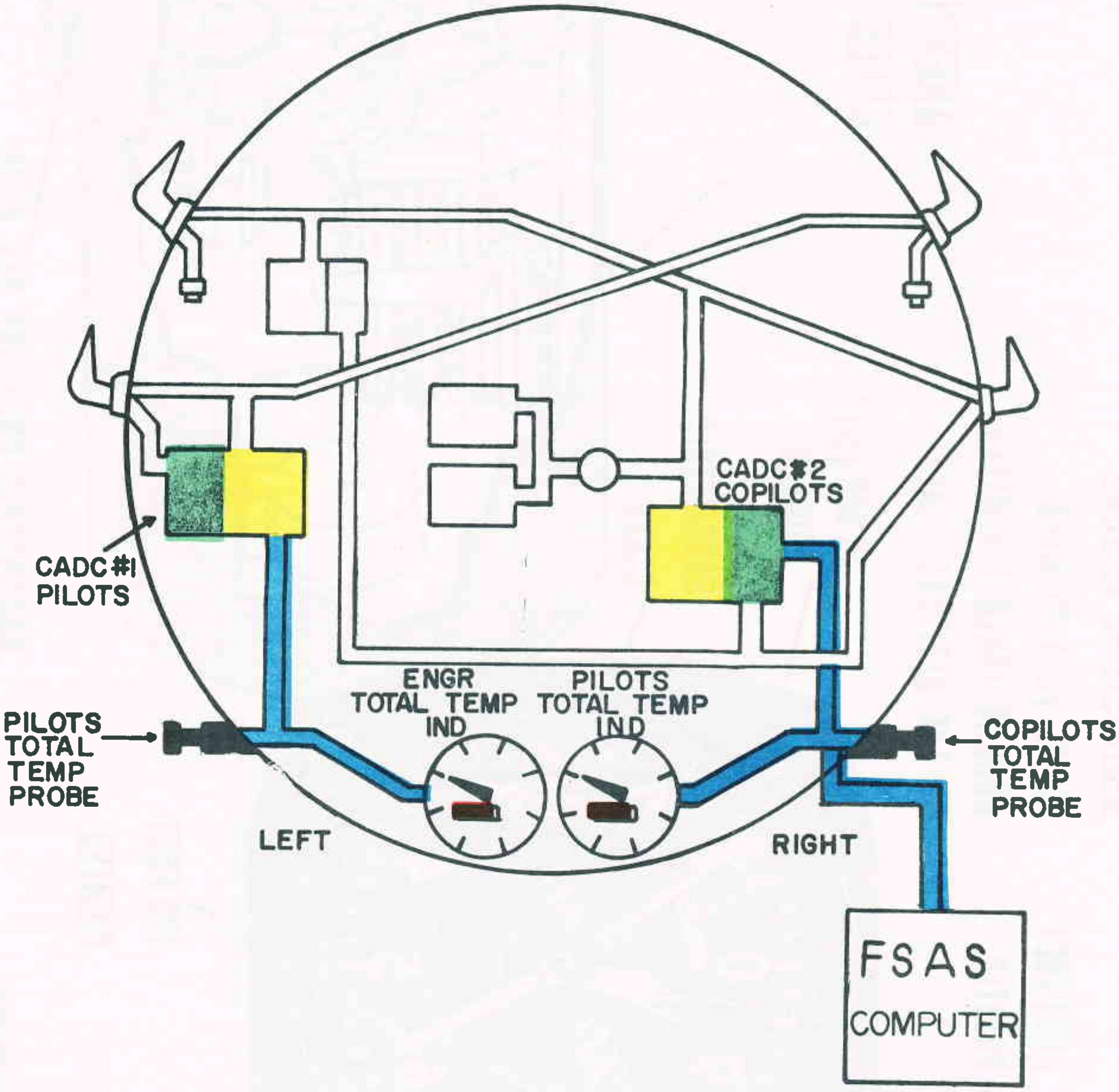


Figure 3-20

C-141B #1 CADC-VSFI GROUP

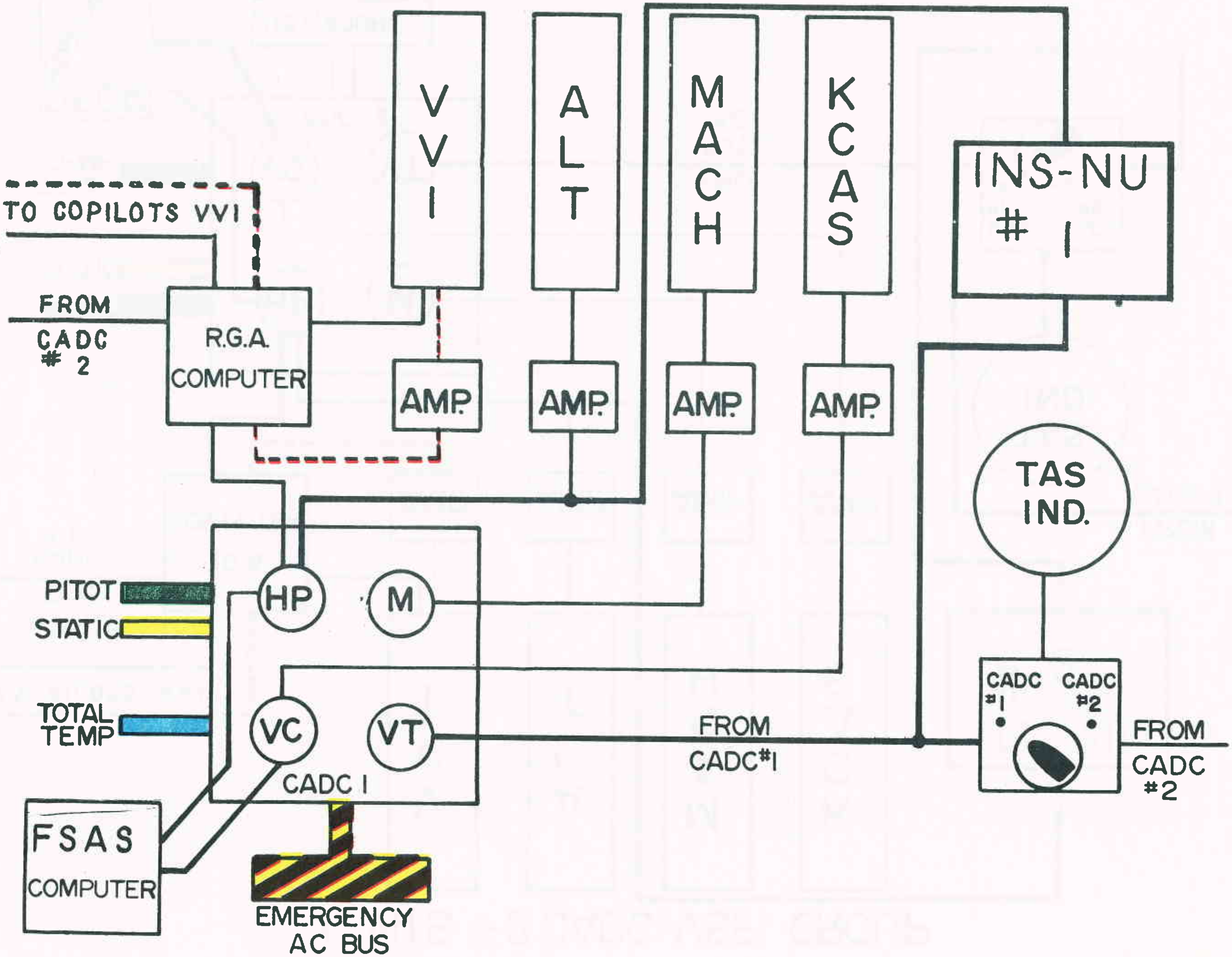


Figure 3-21

C-141B #2 CADC-VSFI GROUP

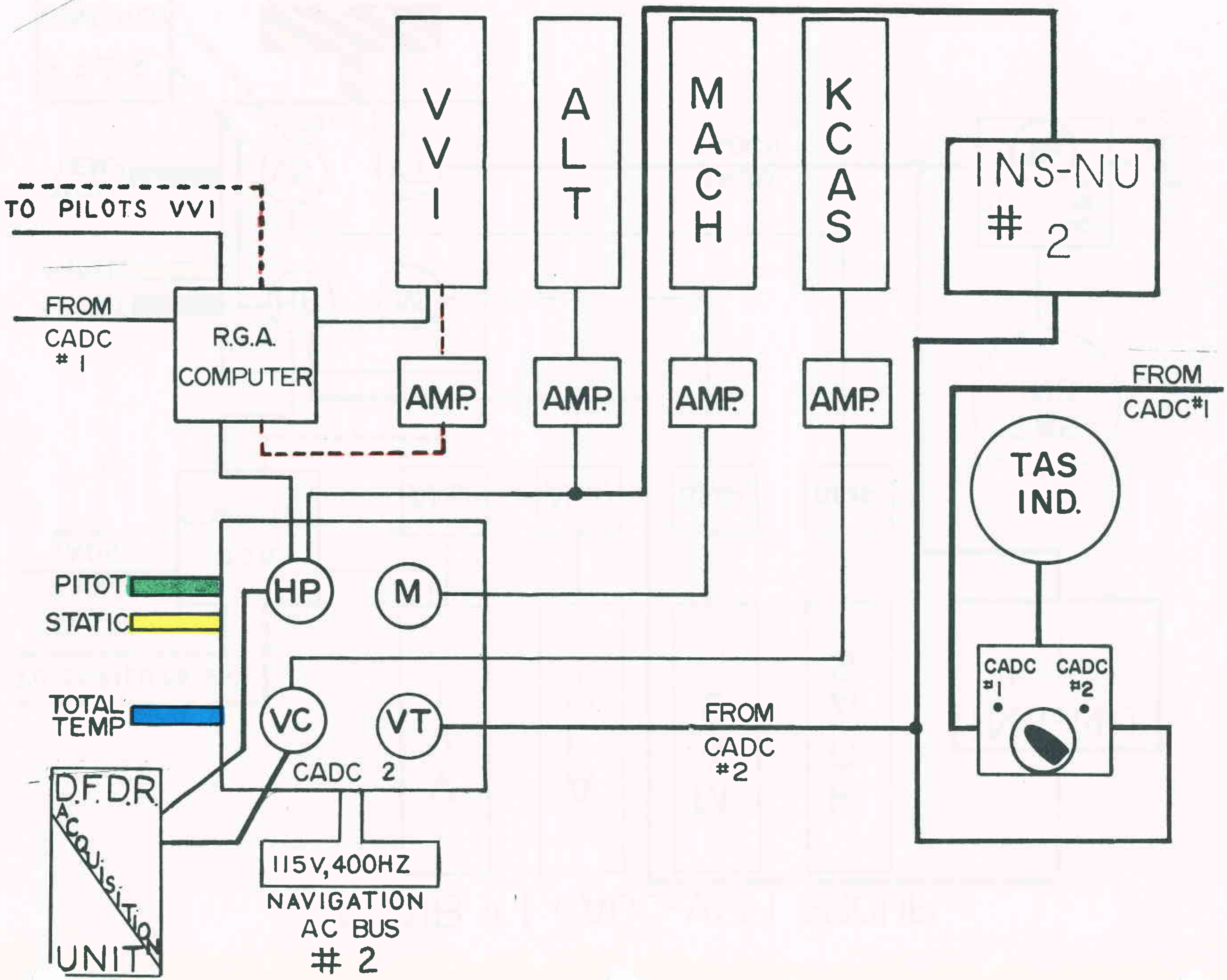
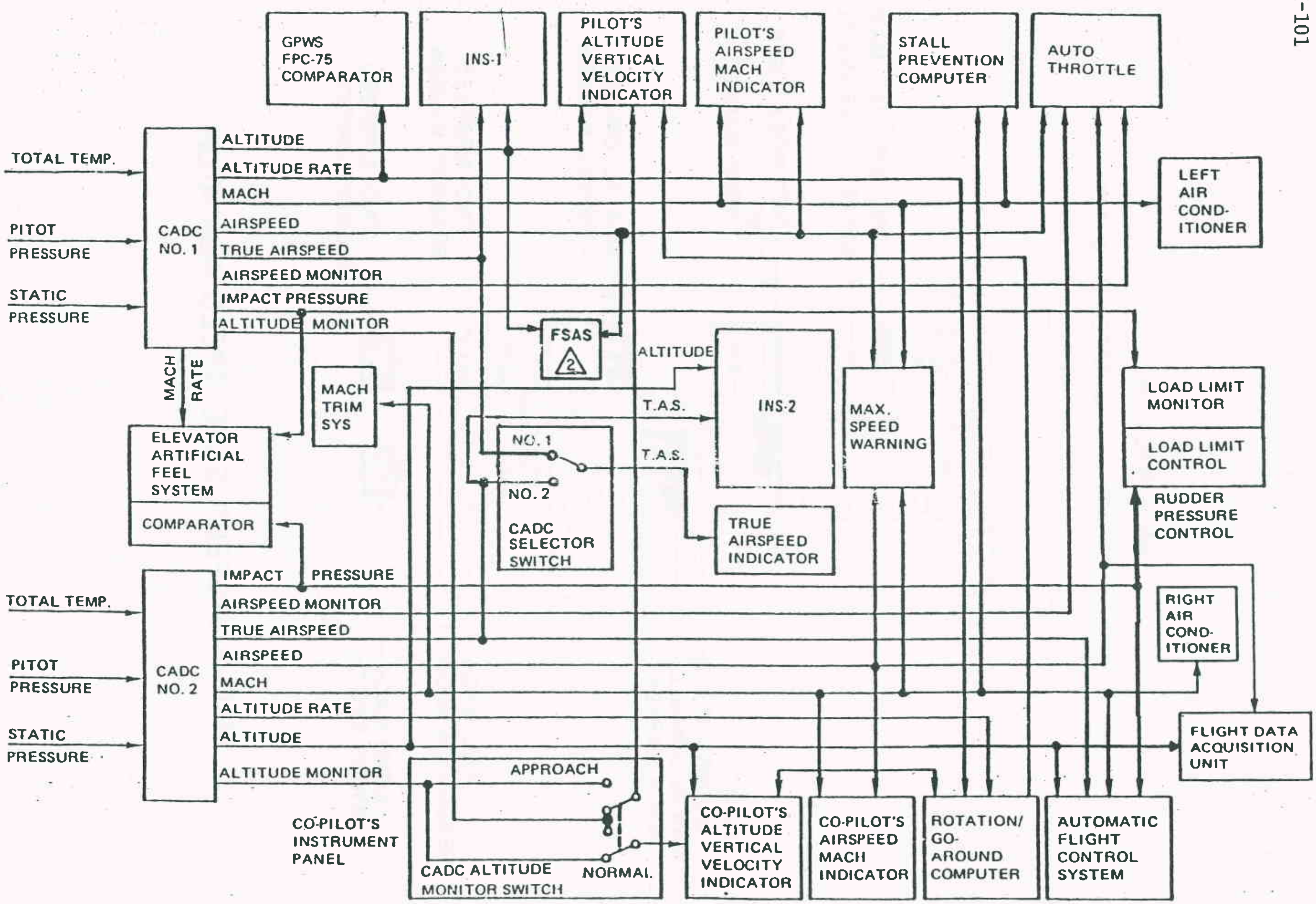
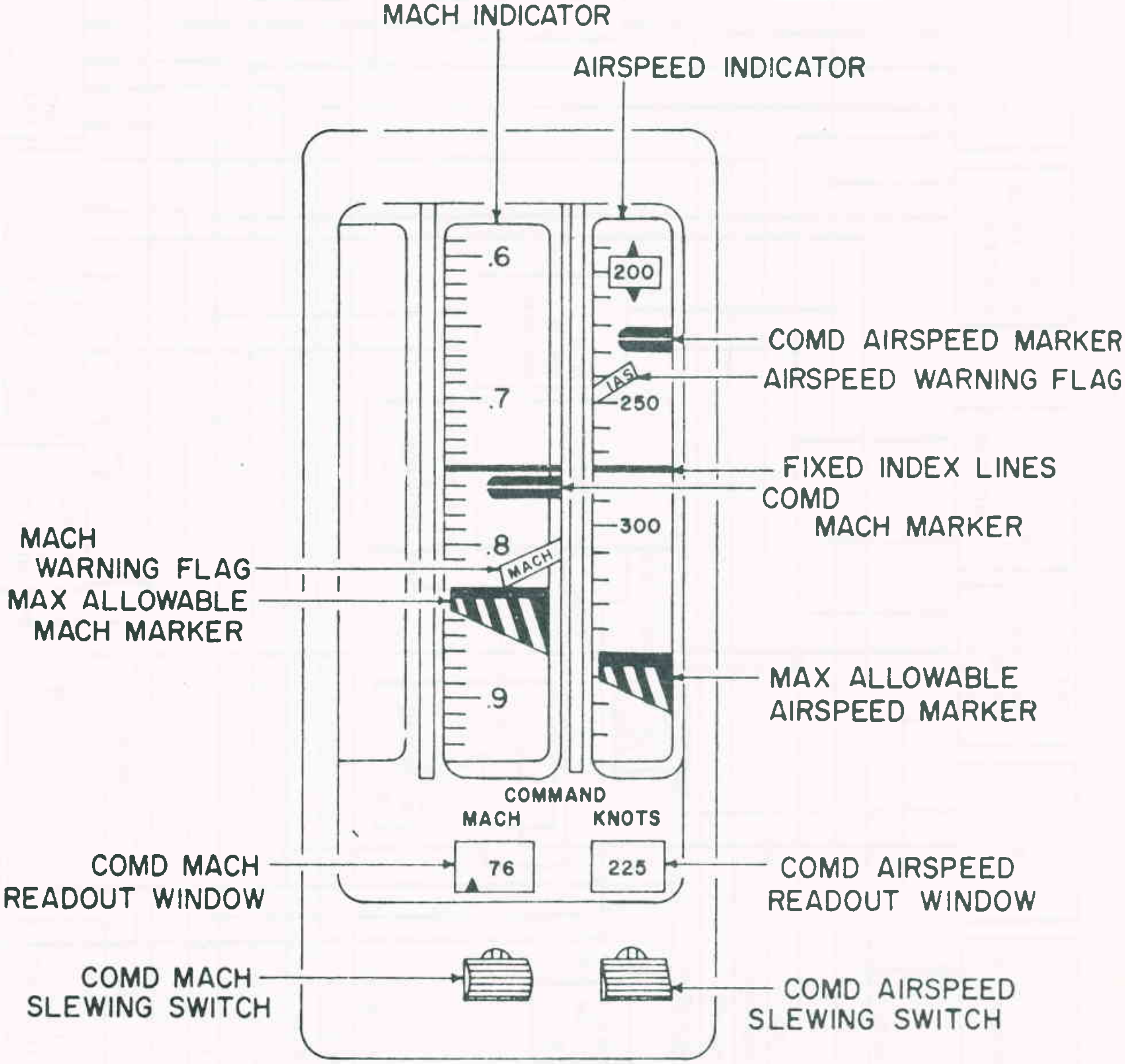


Figure 3-22



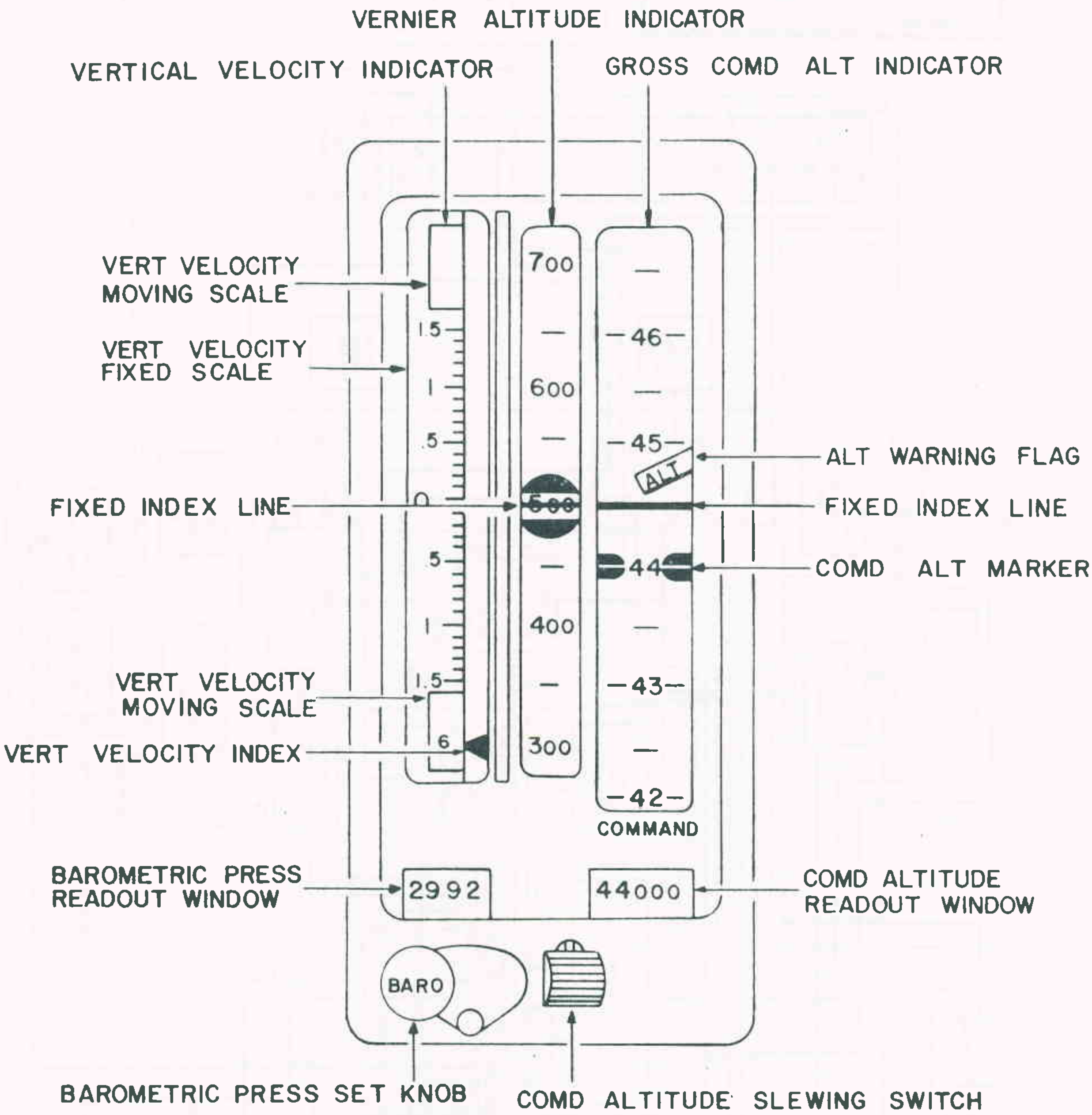
Central Air Data Computer System Block Diagram

Figure 3-23



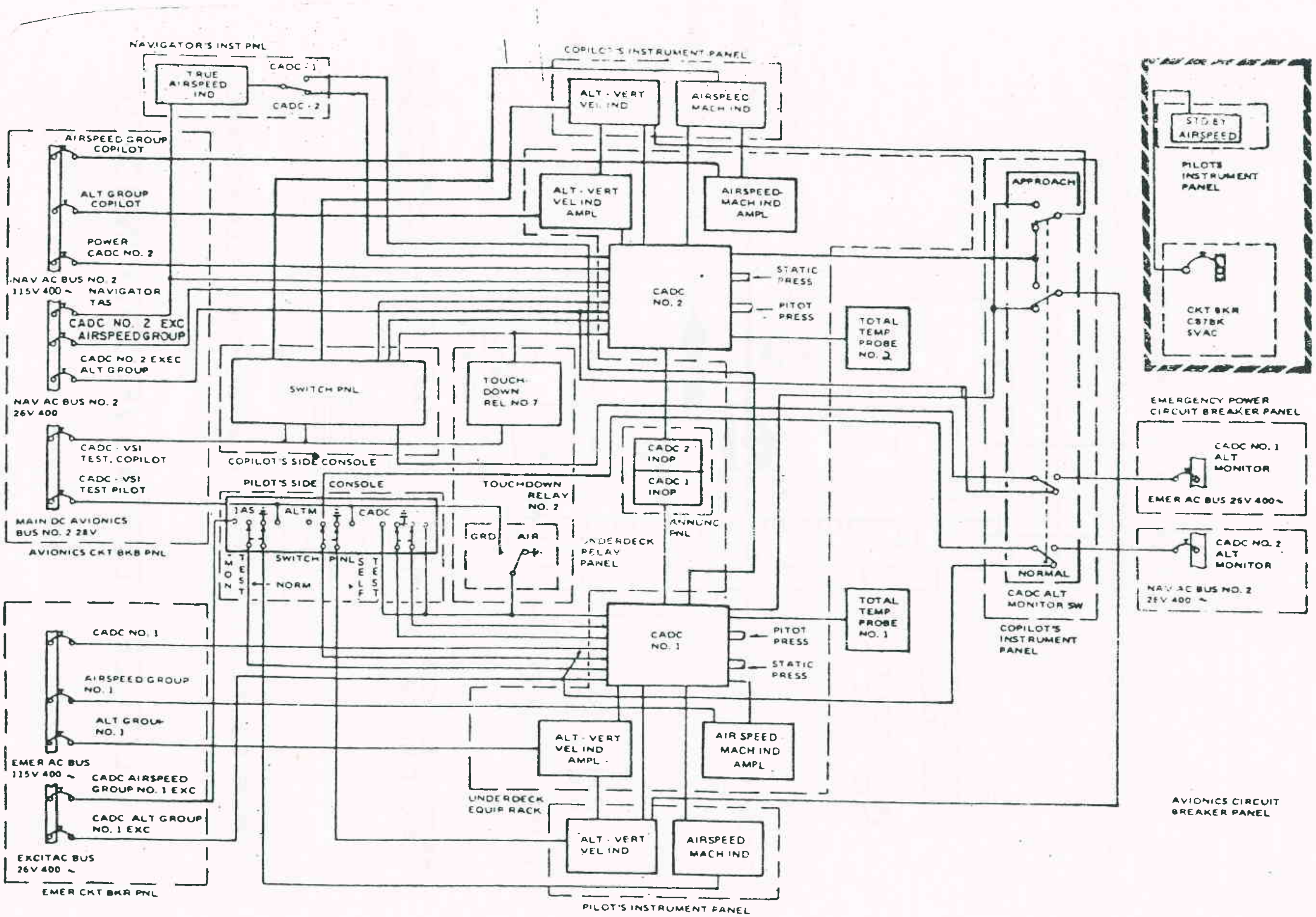
MACH-AIRSPEED, SAFE SPEED INDICATOR

Figure 3-24



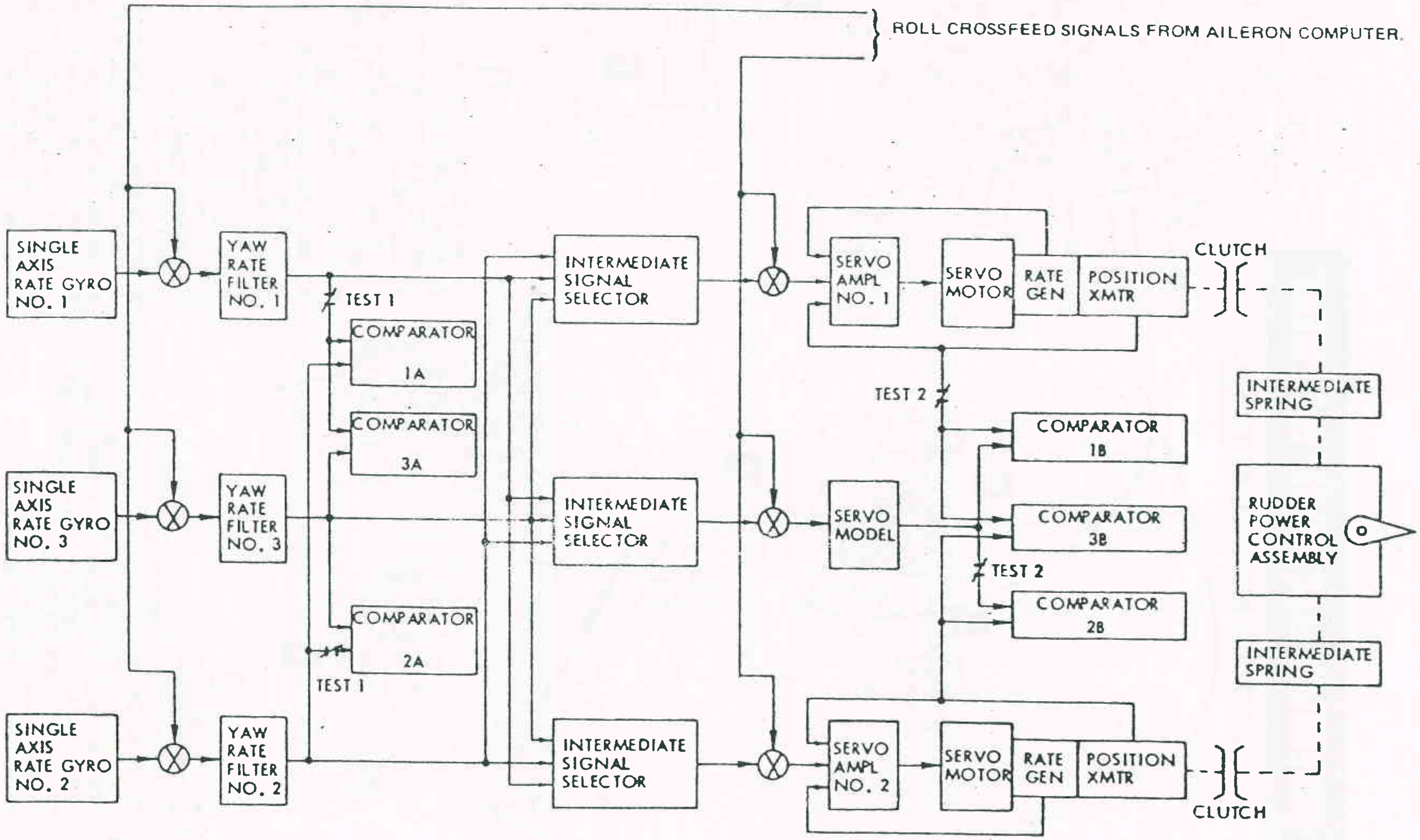
ALTITUDE-VERTICAL VELOCITY INDICATOR

Figure 3-25



Flight Instrument Indicating System Schematic

Figure 3-26

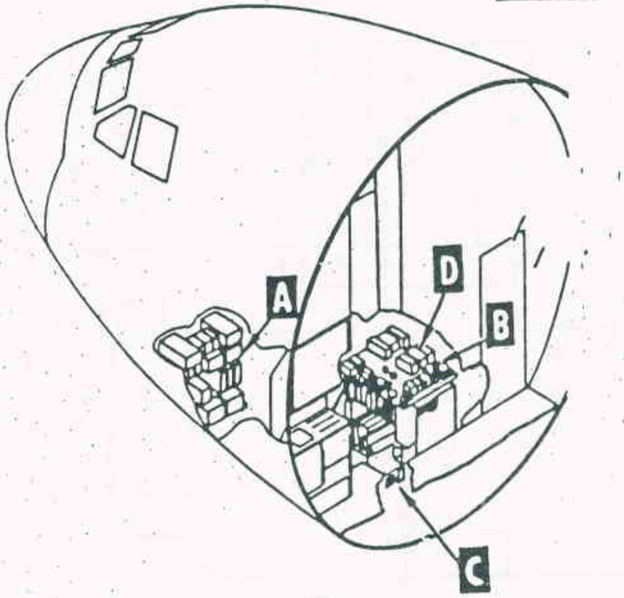


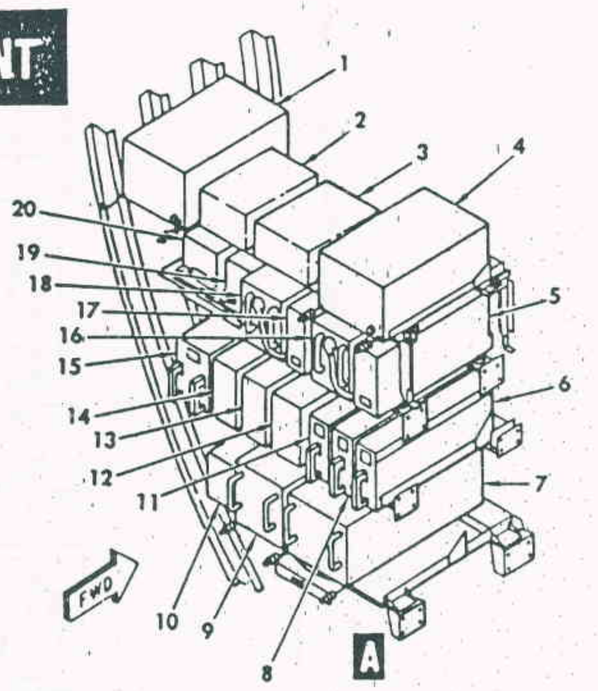
Yaw Damper Block Diagram

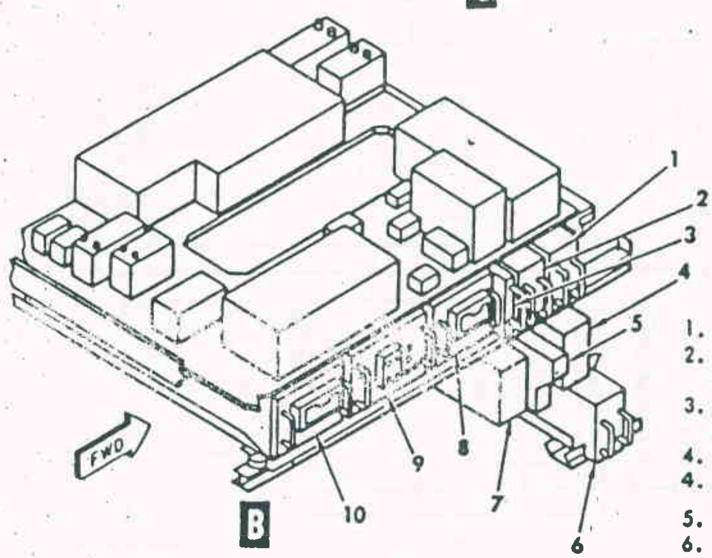
Figure 3-27

UNDERDECK ELECTRONIC EQUIPMENT

AIRCRAFT MODIFIED BY TO 514.







C

1. ELECTRIC COOLING VALVE OVERRIDE HANDLE
2. INSTRUCTION PLATE

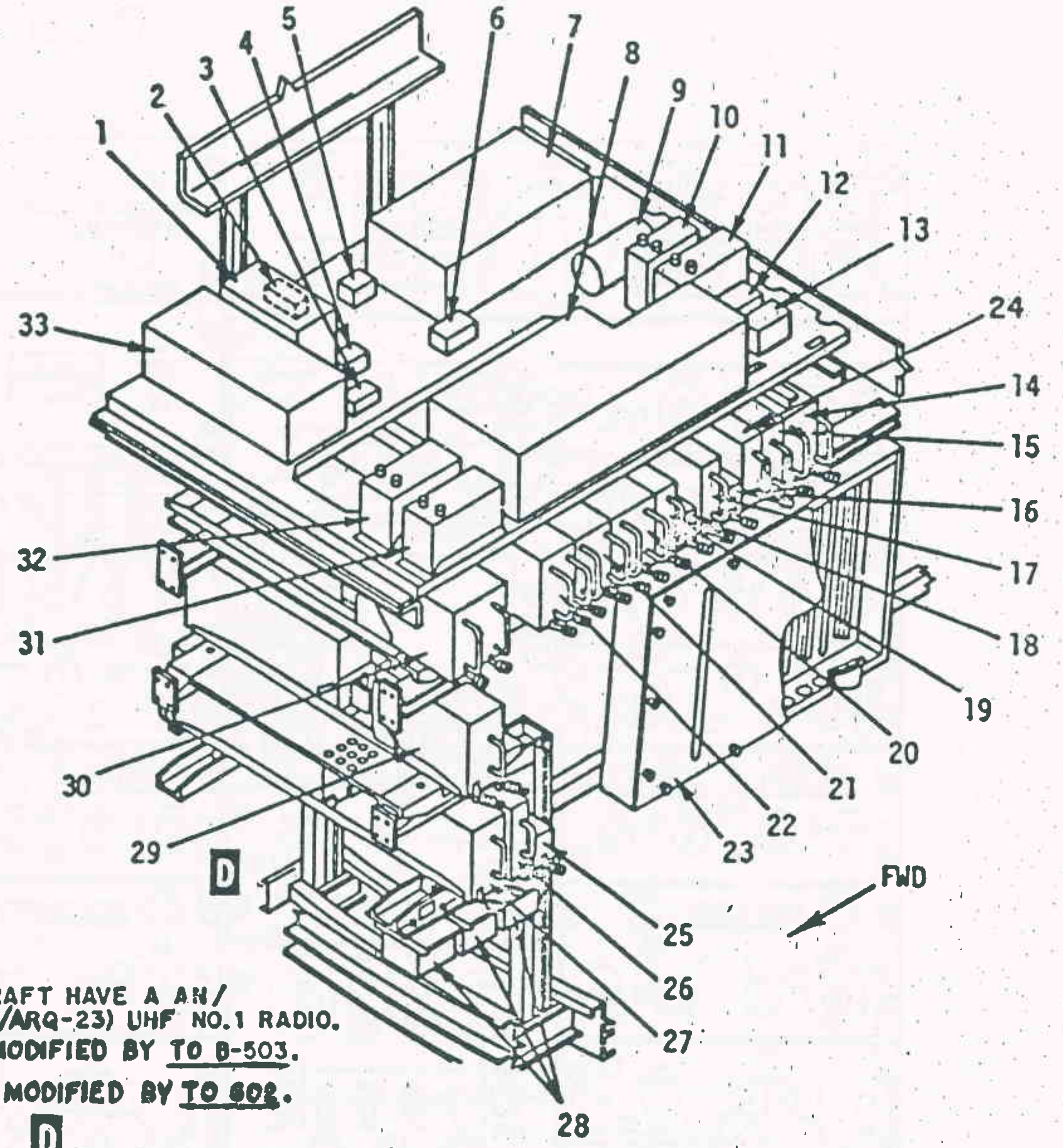
A

1. UHF NO. 1 TRANSCEIVER (TYPE RT-1168/ARC-164(V)) ⚠1
2. ARQ-23 AMPLIFIER ⚠1
   HF1 635V-1 FILTER (JACC/CP) ⚠2
3. ARQ-23 RCVR-XMTR ⚠1
   HF2 635V-1 FILTER (JACC/CP) ⚠2
4. UHF NO. 2 TRANSCEIVER (TYPE RT-1168/ARC-164(V))
4. FSAS COMPUTER ⚠3
5. TACAN NO. 2 D/A CONVERTER (TYPE MX-9577A)
6. AILERON COMPUTER
7. NAVIGATION UNIT NO. 2
8. YAW DAMPER COMPUTER
9. NAVIGATION UNIT NO. 2
10. INS JUNCTION BOX NO. 2
11. ELEVATOR COMPUTER
12. AFCS COMPUTER
13. TEST PROGRAM AND LOGIC COMPUTER (TPLC)
14. FLIGHT DIRECTION COMPUTER
15. CENTRAL AIR DATA COMPUTER NO. 2 (TYPE CPU-43A)
16. TACAN NO. 2 RCVR-XMTR (TYPE RT-1159A/ARN-118)
17. TACAN NO. 1 A/D CONVERTER (TYPE MX-9577A)
18. TACAN NO. 1 RCVR-XMTR (TYPE RT-1159A/ARN-118)
19. TACAN NO. 2 COUPLER
20. TACAN NO. 1 COUPLER

B

1. BATTERY UNIT NO. 3
2. BATTERY UNIT NO. 2
3. FLIGHT DIRECTOR COMPUTER NO. 2
4. FLIGHT RECORDER (TYPE FA-542)
5. FLIGHT PROFILE COMPARATOR (TYPE FPC-75)
6. CENTRAL AIR DATA COMPUTER NO. 2 (TYPE CPU-43K)
7. SIGNAL DATA CONVERTER (TYPE CV-3524/APN-169)
8. HF NO. 1 TRANSCEIVER (TYPE 618T-2)
9. IFF RCVR-XMTR (TYPE RT-731/APX-64)
10. HF NO. 2 TRANSCEIVER (TYPE 618T-2)

Figure 3-28



NOTE

1. SOME AIRCRAFT HAVE A AN/ARC-90(AN/ARQ-23) UHF NO.1 RADIO.
2. AIRCRAFT MODIFIED BY TO 8-503.
3. AIRCRAFT MODIFIED BY TO 602.

D

1. INDICATOR-COUPLER (type CU-2140/APN-169B)
2. IFF TEST SET (type TS-1848A)
3. SINGLE AXIS RATE SENSOR GYRO NO. 2
4. AUDIO AMPLIFIER (type AM-6308/APN-169)
5. ARTIFICIAL FEEL COMPARATOR
6. IFF ANTENNA SWITCHING UNIT (type SA-1474/A)
7. DISPLACEMENT GYROSCOPE ASSEMBLY
8. NAVIGATION SELECTOR JUNCTION BOX
9. RATE GYRO TRANSMITTER NO. 2 (type TRU-2A/A)
10. ALTITUDE-VERTICAL SPEED AMPLIFIER NO. 2
11. INDICATED AIRSPEED AMPLIFIER NO. 2
12. SINGLE AXIS RATE SENSOR GYRO NO. 1
13. SINGLE AXIS RATE SENSOR GYRO NO. 3
14. VOR/LOC RECEIVER NO. 1 (type 806-A)
15. FLARE COMPUTER
16. GLIDE SLOPE RECEIVER NO. 1 (type 800-B)
17. AMPLIFIER POWER SUPPLY NO. 1
18. MARKER BEACON RECEIVER (type 51Z-3)
19. AMPLIFIER POWER SUPPLY NO. 2
20. STALL PREVENTION COMPUTER
21. VHF NO. 1 TRANSCEIVER (type 618M-1 or WILCOX 807A)
22. AMPLIFIER COMPUTER
23. AUTOPILOT JUNCTION BOX
24. ROTATION/GO-AROUND COMPUTER
25. GLIDE SLOPE RECEIVER NO. 2 (type 800-B)
26. ADF RECEIVER NO. 2 (type DFA-73A)
27. VOR/LOC RECEIVER NO. 2 (type 806-A)
28. AUDIO FREQUENCY AMPLIFIERS (type AM-944/AIC-13)
29. VHF NO. 2 TRANSCEIVER (type 618M-1 or WILCOX 807A)
30. ADF RECEIVER NO. 1 (type DFA-73A)
30. (3) DISPLAY INTERFACE UNIT (DIU)
31. INDICATED AIR SPEED AMPLIFIER NO. 1 (type ASK-10/A24G-10)
32. ALTITUDE-VERTICAL SPEED AMPLIFIER NO. 1 (type ASK-11/A24G-11)
33. WEATHER RADAR RCVR-XMTR (type RTA-1EB/APS-133)

Figure 3-29

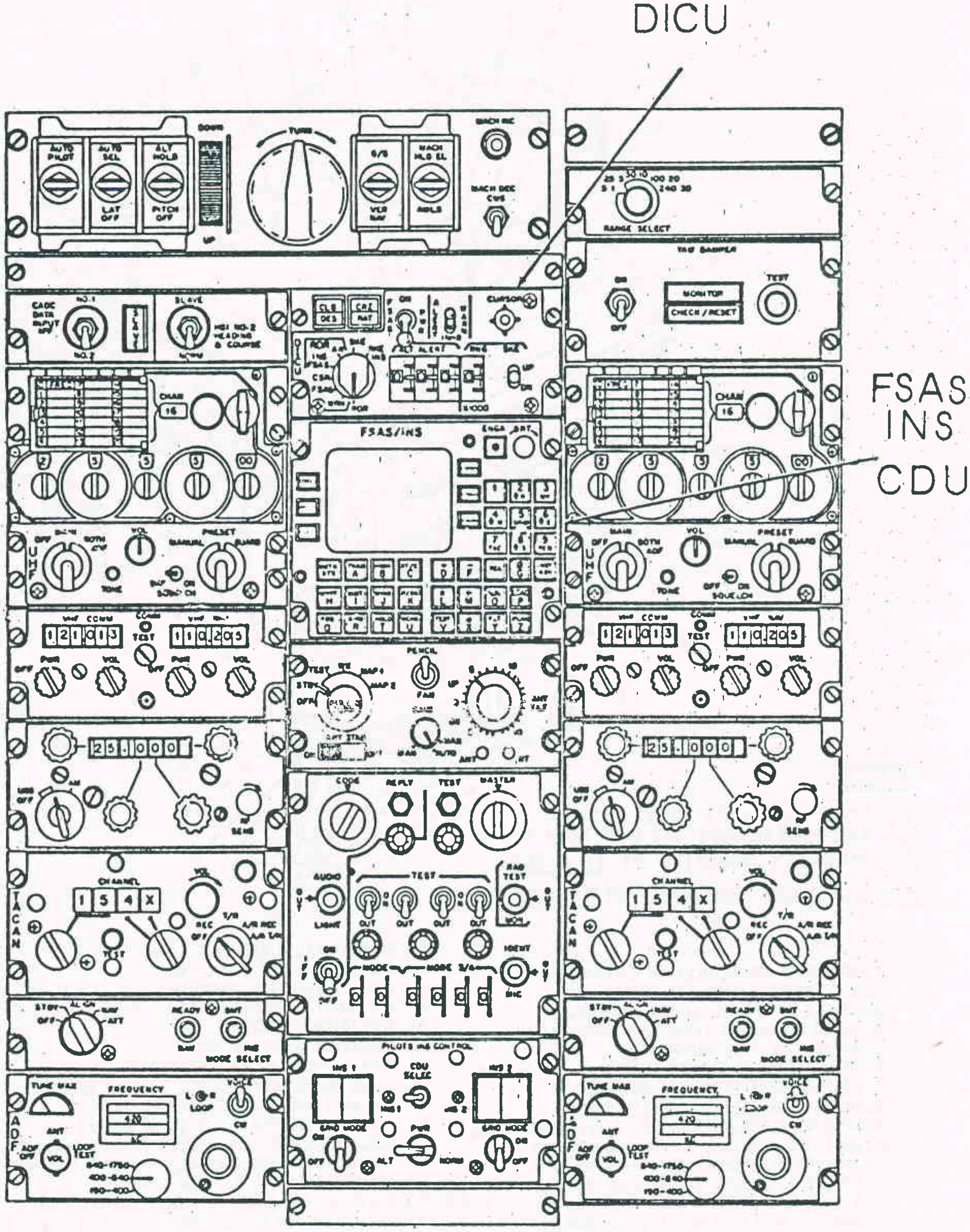
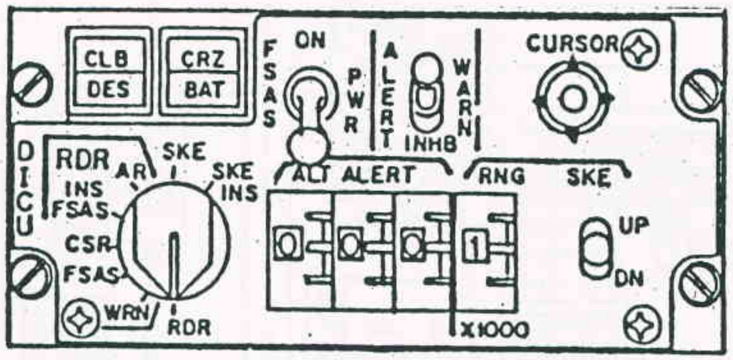


Figure 3-30



FUEL SAVINGS ADVISORY SYSTEM CONTROL DISPLAY UNIT (FSAS CDU) ARRANGEMENT

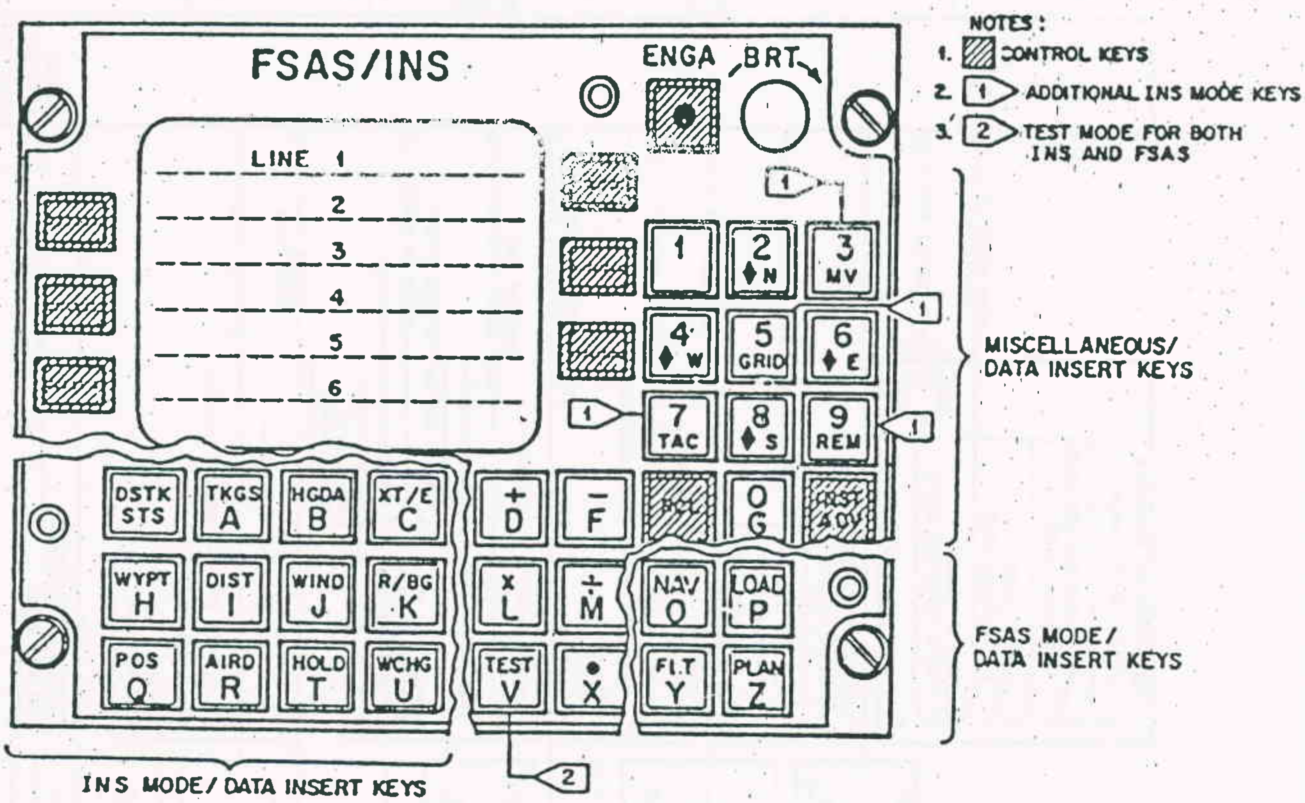


Figure 3-31

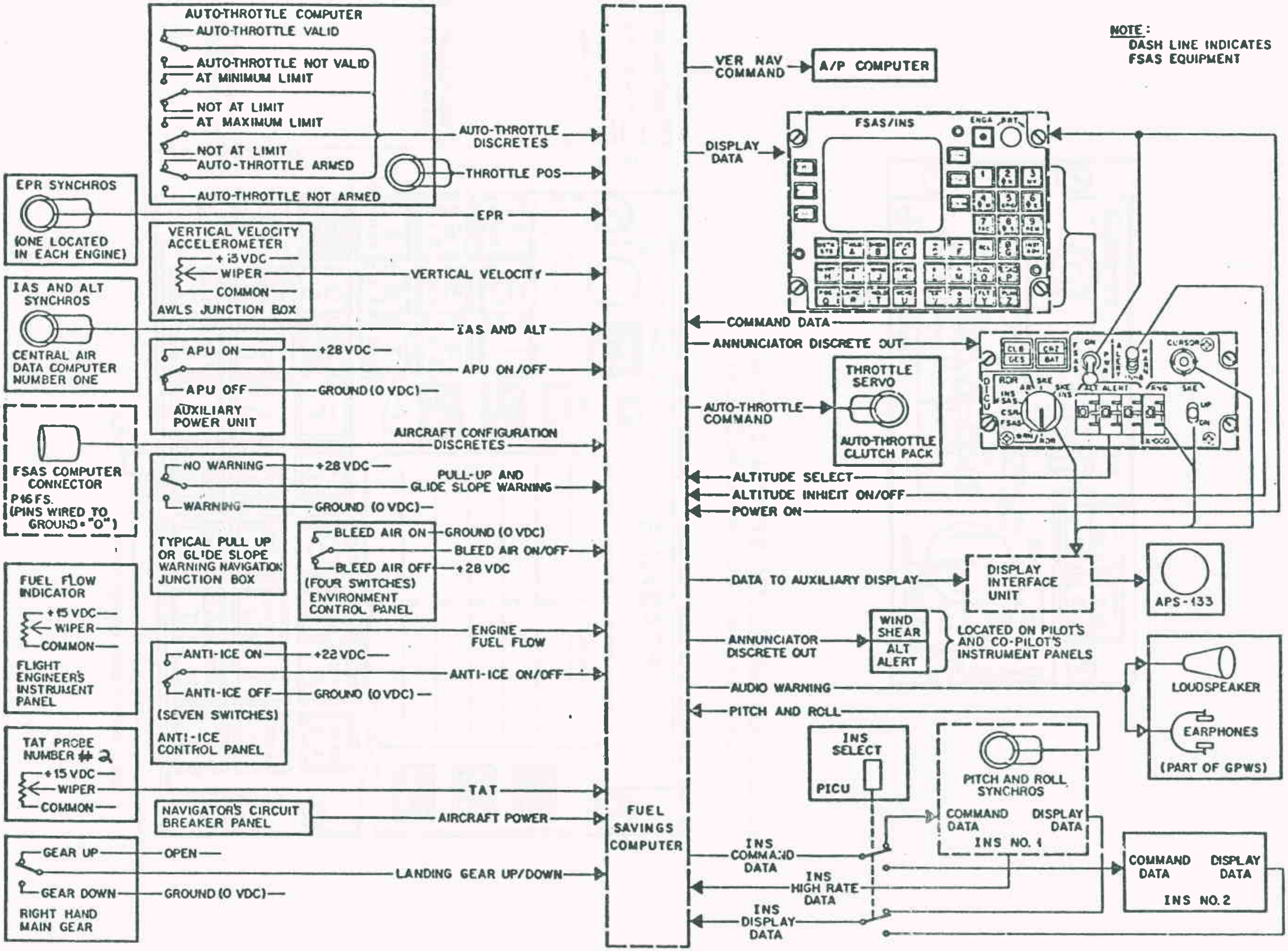
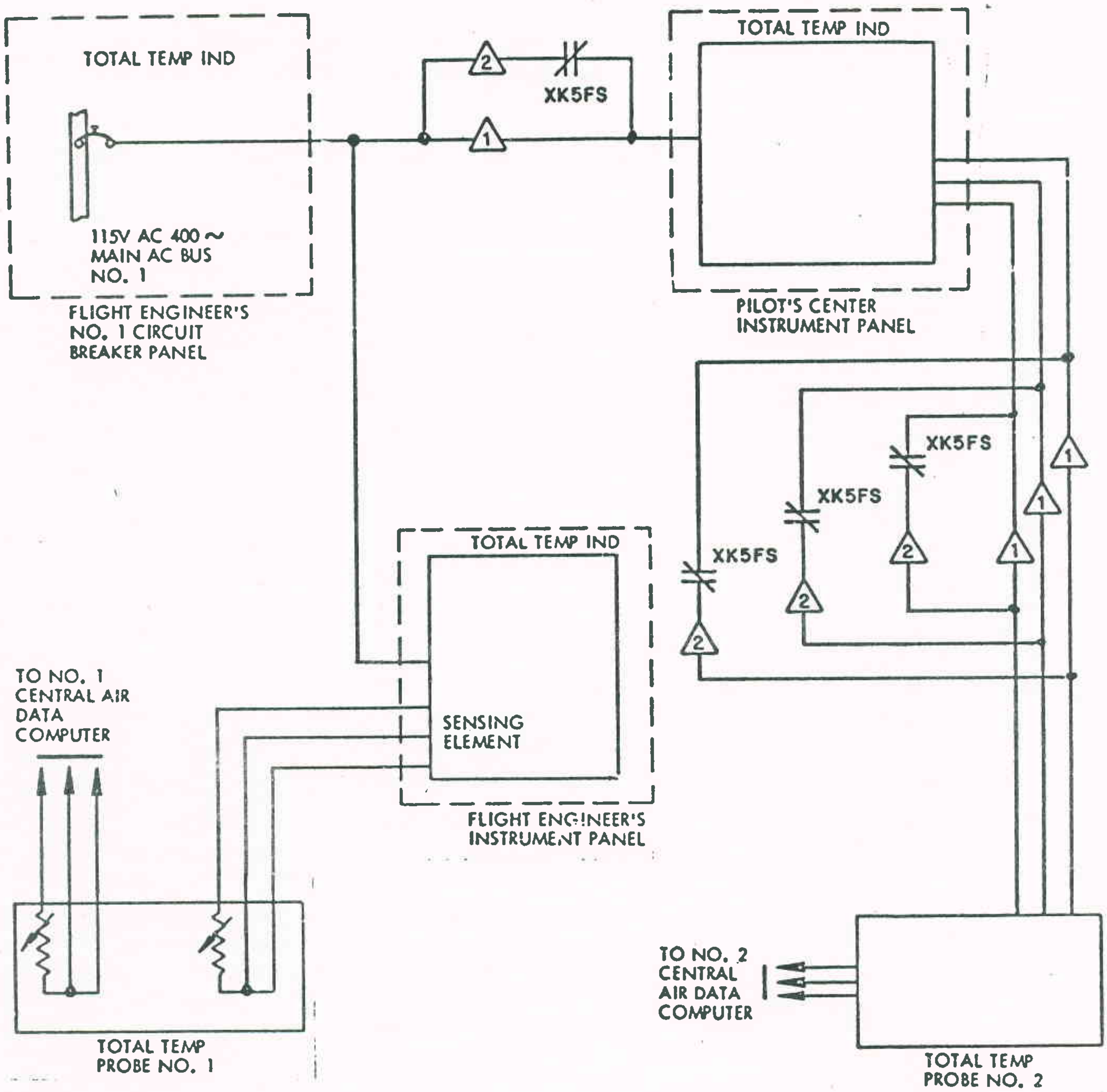

NOTE:
DASH LINE INDICATES FSAS EQUIPMENT
AUTO-THROTTLE COMPUTER
AUTO-THROTTLE VALID
AUTO-THROTTLE NOT VALID
AT MINIMUM LIMIT
NOT AT LIMIT
AT MAXIMUM LIMIT
NOT AT LIMIT
AUTO-THROTTLE ARMED
AUTO-THROTTLE NOT ARMED
AUTO-THROTTLE DISCRETES
THROTTLE POS
EPR SYNCHROS
(ONE LOCATED IN EACH ENGINE)
EPR
VERTICAL VELOCITY ACCELEROMETER
+15 VDC
WIPER
COMMON
AWLS JUNCTION BOX
VERTICAL VELOCITY
IAS AND ALT SYNCHROS
CENTRAL AIR DATA COMPUTER NUMBER ONE
IAS AND ALT
APU ON
+28 VDC
APU ON/OFF
APU OFF
GROUND (0 VDC)
AUXILIARY POWER UNIT
FSAS COMPUTER CONNECTOR
P16 FS.
(PINS WIRED TO GROUND = "0")
AIRCRAFT CONFIGURATION DISCRETES
NO WARNING
+28 VDC
PULL-UP AND GLIDE SLOPE WARNING
WARNING
GROUND (0 VDC)
TYPICAL PULL UP OR GLIDE SLOPE WARNING NAVIGATION JUNCTION BOX
BLEED AIR ON
GROUND (0 VDC)
BLEED AIR ON/OFF
BLEED AIR OFF
+28 VDC
(FOUR SWITCHES)
ENVIRONMENT CONTROL PANEL
FUEL FLOW INDICATOR
+15 VDC
WIPER
COMMON
FLIGHT ENGINEER'S INSTRUMENT PANEL
ENGINE FUEL FLOW
ANTI-ICE ON
+22 VDC
ANTI-ICE ON/OFF
ANTI-ICE OFF
GROUND (0 VDC)
(SEVEN SWITCHES)
ANTI-ICE CONTROL PANEL
TAT PROBE NUMBER #2
+15 VDC
WIPER
COMMON
TAT
NAVIGATOR'S CIRCUIT BREAKER PANEL
AIRCRAFT POWER
GEAR UP
OPEN
LANDING GEAR UP/DOWN
GEAR DOWN
GROUND (0 VDC)
RIGHT HAND MAIN GEAR
FUEL SAVINGS COMPUTER
VER NAV COMMAND
A/P COMPUTER
DISPLAY DATA
FSAS/INS
COMMAND DATA
ANNUNCIATOR DISCRETE OUT
THROTTLE SERVO
AUTO-THROTTLE COMMAND
AUTO-THROTTLE CLUTCH PACK
ALTITUDE SELECT
ALTITUDE INHIBIT ON/OFF
POWER ON
DATA TO AUXILIARY DISPLAY
DISPLAY INTERFACE UNIT
APS-133
ANNUNCIATOR DISCRETE OUT
WIND SHEAR
ALT ALERT
LOCATED ON PILOT'S AND CO-PILOT'S INSTRUMENT PANELS
AUDIO WARNING
LOUDSPEAKER
EARPHONES
(PART OF GPWS)
PITCH AND ROLL
INS SELECT
PICU
PITCH AND ROLL SYNCHROS
COMMAND DATA
DISPLAY DATA
INS NO. 1
INS COMMAND DATA
INS HIGH RATE DATA
INS DISPLAY DATA
COMMAND DATA
DISPLAY DATA
INS NO. 2

Figure 3-32



NOTES:

(1) AIRCRAFT NOT MODIFIED PER T.O. 1C-141-602

(2) AIRCRAFT MODIFIED PER T.O. 1C-141-602

Figure 3-33