**ON A/C ALL

31–00–00–001

INDICATING AND RECORDING SYSTEMS, GENERAL

Introduction

The indicating and recording system has different sub–systems to do the functions that follows:

- Show time
- Records aircraft parameter data
- Receive data from sensors and avionics systems and supply it to other systems
- Make warning tones
- Show caution and warning lights
- Show advisory lights
- Take–off warning calculation
- Show navigation, engine, and system parameters.

General Description

Refer to Figure 1.

The indicating and recording system has the sub–systems that follow:

- 31–21–00 Clocks
- 34–32–00 Flight Data Recorder System (FDR)
- 34–40–00 Central Computer
- 34–50–00 Central Warning System
- 34–61–00 Electronic Instruments System.

Clocks

The electronic clock has a quartz timebase that supplies a continuous display of Greenwich Mean Time (GMT) or Local Time (LOC). The electronic clock can also be set to show the Elapsed Time (ET), the date or set to the chronometer function (CHR).

Flight Data Recorder System (FDR)

The Solid State Flight Data Recorder (SSFDR) records aircraft parameter data and stores it in crash–protected memory for future retrieval purposes.

Central Computer

The central computer has a Flight Data Processing System (FDPS) to receive data from sensors and avionics systems and supply it to other systems. The Flight Data Processing System (FDPS) also supplies warning tones that alert the crew to specific events or system failures.

Central Warning System

The central warning system is divided into the two parts that follow:

- Caution and Warning Lights
- Take–off Warning.

The caution and warning light system is divided into the two parts that follow:

- Caution and warning lights
- Advisory lights.

Caution and Warning Lights: The caution and warning lights system shows system malfunctions and other conditions that require a corrective action.

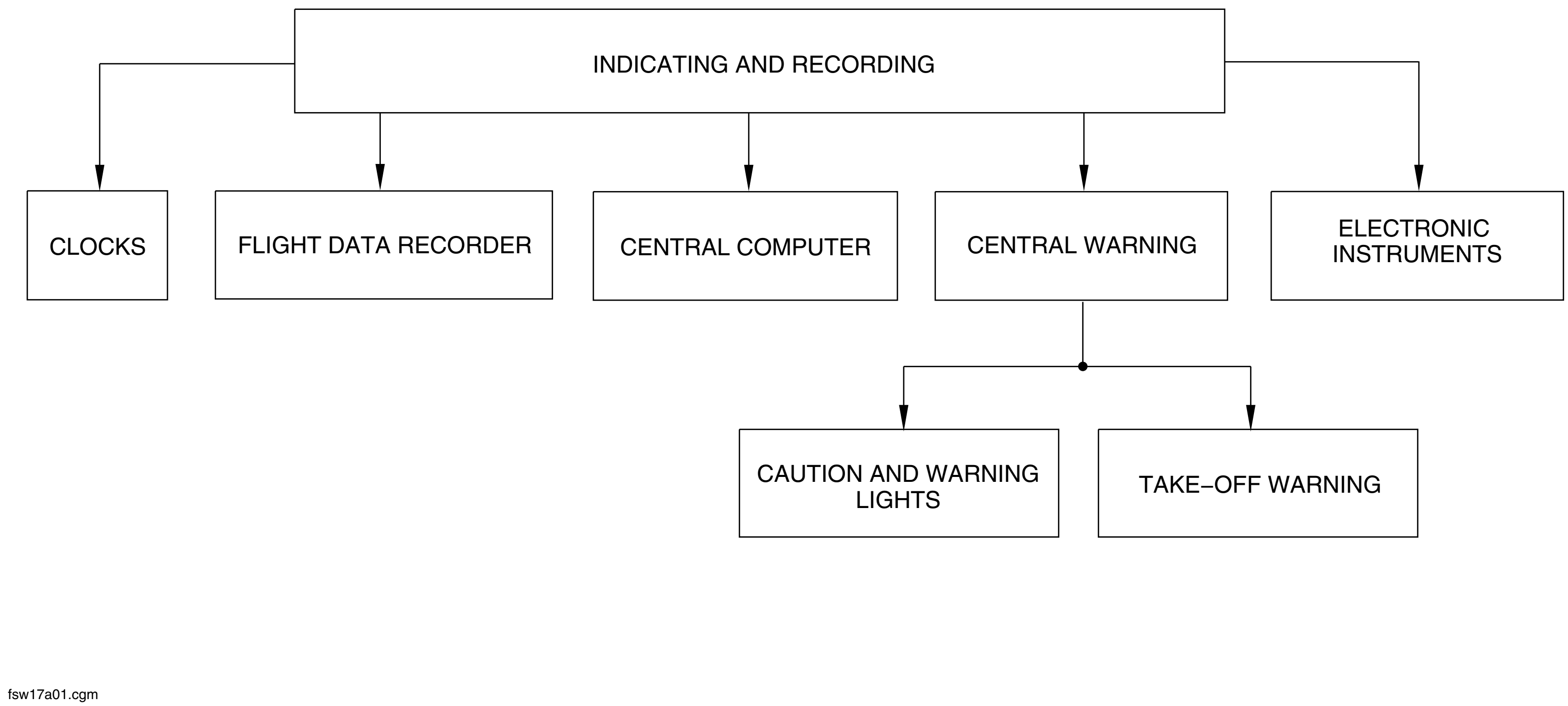
Advisory Lights: The advisory lights show malfunctions and other conditions that require a corrective action and safe and normal system operation.

Refer to 31–40–00, Central Computer.

The take–off warning system supplies an aural warning when a take–off is attempted with the aircraft not in the correct take–off configuration.

Electronic Instruments System

The Electronic Instrument System (EIS) shows navigation, engine, and system parameters. It interfaces with other systems to calculate, make, and show their images. The Electronic Instrument System (EIS) also monitors is calculations to prevent misleading information from being shown.



INDICATING AND RECORDING BLOCK DIAGRAM
Figure 1

**ON A/C ALL

31–15–00–001

ENGINE COCKPIT INTERFACE UNIT

General

Refer to Figure 1.

The engine cockpit interface unit (ECIU) transfers data between the cockpit and both full authority digital engine controllers (FADECs).

General Description

Refer to Figure 2.

The ECIU is located in the flight compartment, on the lower shelf of the left circuit breaker console. It contains a single main circuit card assembly (CCA) that consists of two redundant channels (A and B).

The ECIU receives 24 inputs per channel from the cockpit that are transmitted to the FADEC and 16 inputs per channel from the engines that are transmitted to the cockpit.

Detailed Description

Refer to Figure 3.

Inputs to the ECIU from the cockpit are hardware discretes and supply either a ground or an open indication. The ECIU monitors these discrete inputs and processes the information into ARINC data. The ECIU then transmits this data across four independent buses to each channel of both FADECs. The cockpit discrete inputs are shown in Table 1.

TABLE 1 DISCRETE INPUTS

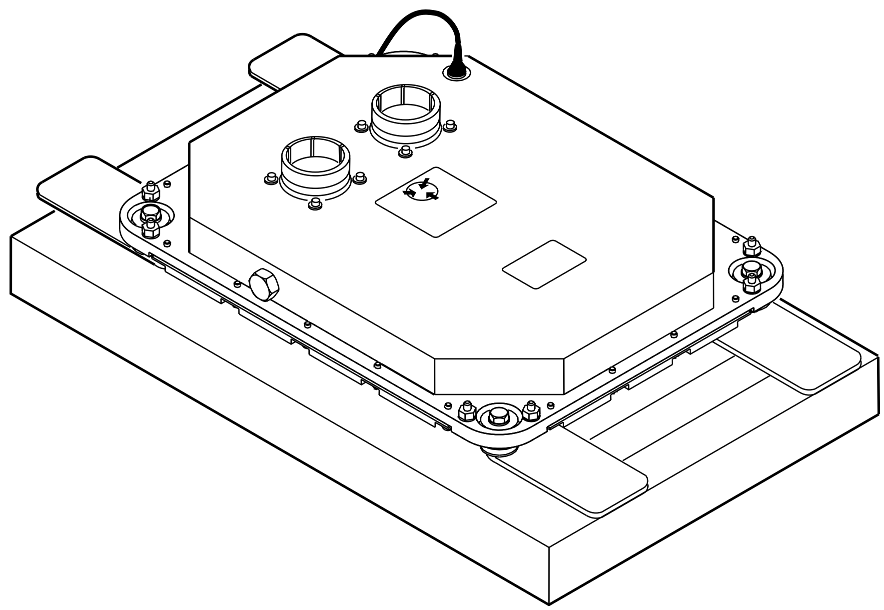
SOURCE	DISCRETE
Engine Control Panel	Maximum climb select* Maximum climb select* Maximum power select Power derate rest* Power derate decrement* 850 RPM approach select
Maintenance Panel	Maintenance test select * Rigging trim select*
Ignition Panel	Engine 1 manual ignition select Engine 1 ignition on/off Engine 2 manual ignition select Engine 2 ignition on/off
ECS Panel	ECS bleed 1 on/off ECS bleed 2 on/off ECS maximum select ECS minimum select
Spares	Eight spares available (two momentary)
* Momentary switches	

The ECIU can supply 16 discrete outputs per channel as either a ground or open indication. A 1.0–ampere discrete output drives the oil ejector solenoid while a 0.5–ampere output drives the other systems. The ECIU monitors the ARINC data and sets these discretes when the appropriate bits are set. The discrete outputs are shown in Table 2.

TABLE 2 DISCRETE OUTPUTS	
DESTINATION	DISCRETE
Lamps – O Red channel A and B	Engine 1 low oil pressure Engine 2 low oil pressure Engine 1 low fuel pressure Engine 2 low fuel pressure Engine 1 fuel filter bypass Engine 2 fuel filter bypass
Relay Drivers – simplex, engine 1 channel A only, engine 2 channel B only	Engine 1 oil cooler ejector Engine 2 oil cooler ejector Engine 1 oil cooler intermediate flap position Engine 2 oil cooler intermediate flap position Engine 1 oil cooler maximum flap position Engine 2 oil cooler maximum flap position
Spares	Four spares available

T1.8 information is also input to the ECIU from each engine FADEC channel (four per aircraft) across the ARINC 429 data buses. The ECIU processes the T1.8 signal and supplies this information on the four ARINC data buses.

The dual channel ECIU operates on 28 Vdc electrical power through 3–ampere circuit breakers. Channel A receives power from the left essential bus and channel B receives power from the right essential bus. The circuit breaker for channel A is located at position H5 on the left DC circuit breaker panel. The circuit breaker for channel B is located at position H5 on the right DC circuit breaker panel.



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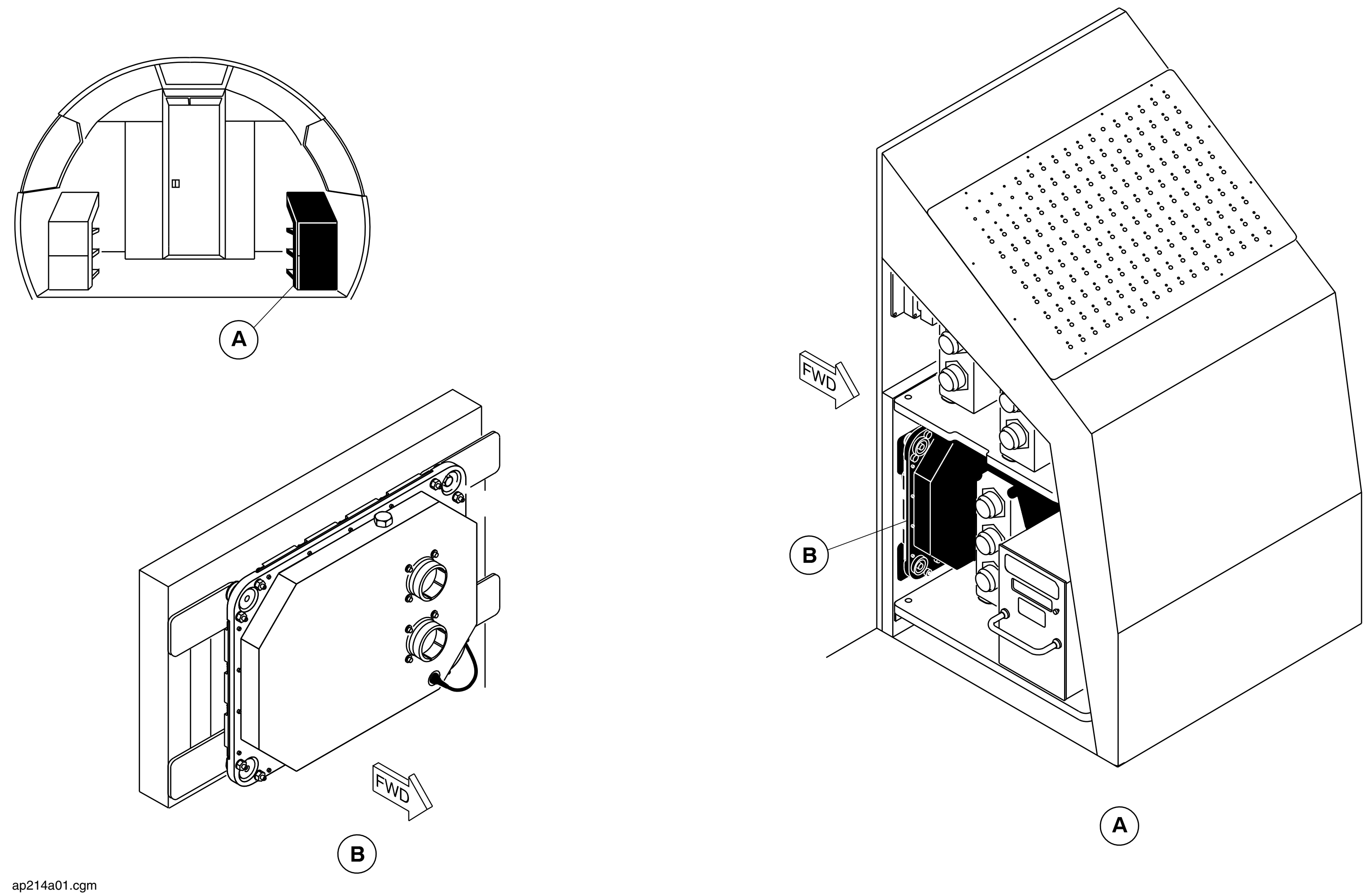
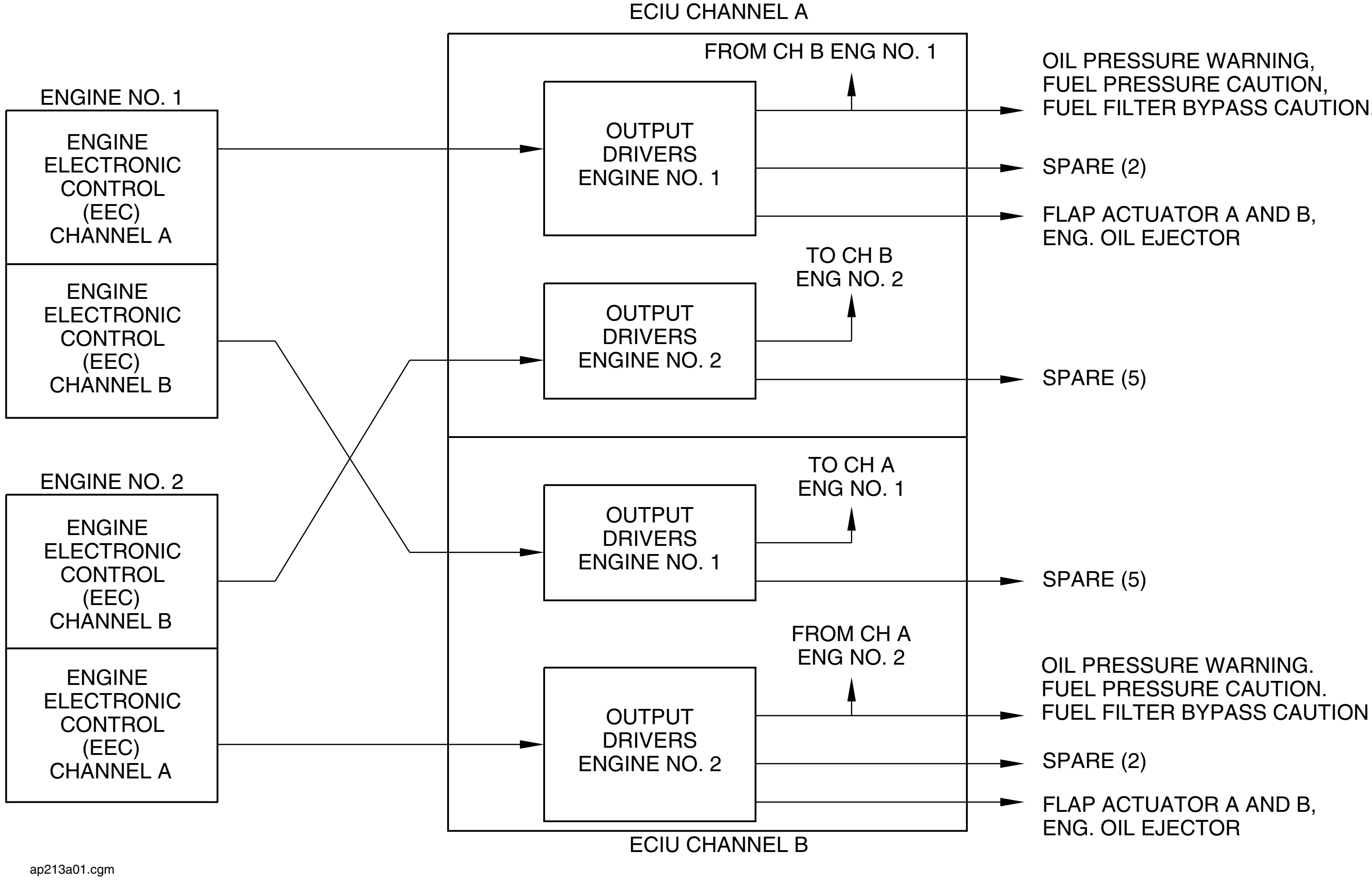
ENGINE COCKPIT INTERFACE UNIT
Figure 1



ENGINE COCKPIT INTERFACE UNIT (ECIU) LOCATION
Figure 2



ENGINE COCKPIT INTERFACE UNIT – SYSTEM BLOCK DIAGRAM
Figure 3

**ON A/C ALL

31–20–00–001

INDEPENDENT INSTRUMENTS

Introduction

On aircraft without ModSum 4–126403 and 4–126434 incorporated, the electronic clock has a quartz timebase that supplies a continuous display of Greenwich Mean Time (GMT) or Local Time (LOC). The electronic clock can also be set to show the Elapsed Time (ET), the date or set to the chronometer function (CHR).

On aircraft with ModSum 4–126403 or 4–126434 incorporated, the electronic clock has a quartz timebase that supplies a continuous display of Universal Time Coordinated (UTC) time, the current date, Local Time (LT) on the top six–digit Liquid Crystal Display (LCD). The clock can also be set to show the Elapsed Time (ET) or Chronometric Time (CHR) on the bottom four–digit LCD.

General Description

On aircraft without ModSum 4–126403 and 4–126343 incorporated (Refer to Figure 1.), the electronic clock has a quartz timebase which operates a special function Large Scale Integrated (LSI) circuit and supplies the functions that follow:

– Continuous digital display of GMT or LOC
– Digital display of ET
– Display of the chronometer (CHR) function. In this mode, the display of seconds is analog (sweep–hand) and that of minutes is digital.
– Display of date and year, when requested.

On aircraft with ModSum 4–126403 or 4–126434 incorporated (Refer to Figure 1.), the electronic clock has a quartz timebase which operates a special function LSI circuit and supplies the functions that follow:

– Continuous digital display of UTC or LOC
– Digital display of ET
– Display of the chronometer (CHR) function. In this mode, the displays of seconds and minutes are digital.
– Display of date and year, when requested.

There are two clocks located in the flight compartment one on each side of the glareshield. The pilots set the type of time–based information to be shown on the display. A four–position switch, located on the lower left corner of the clock face, controls the clock functions.

The clock system includes the component that follows:

– Clock, Electronic (31–21–00).

Detailed Description

On aircraft without ModSum 4–126403 and 4–126434 incorporated, Refer to Figure 2.

Each clock is independently operated from the switches located on the bezel, and has instrument glass which is coated with an anti–reflection treatment on both sides.

The function selector positions are labeled DATE, LOC, GMT and SET. The selector is pushed to exit the SET mode.

When the function selector is placed in the SET position, the Elapsed Time (ET) button is pushed to cycle through the modes that follow:

– GMT minutes (displayed immediately when SET is selected)
– GMT hours
– LOC minutes
– LOC hours
– Days
– Months
– Years (default on power–up is 90).

At each momentary activation of the ET switch, the applicable area of the display flashes and the data is then entered using the CHR button. The CHR button may either be pushed steadily to cause the target display to increment automatically at a rate of 1 unit per 0.5 seconds, or may be activated by the operator in discrete steps.

The Elapsed Time (ET) switch located in the lower left area of the clock face gives 3–state and 2–state sequences dependent on the Weight On Wheels (WOW) status of the aircraft.

On the ground:

– First activation: Display of Elapsed Time
– Second Activation: Elapsed Time is reset to zero
– Third activation: Display of Chronometer minutes

Airborne:

– First Activation: Display of Elapsed Time
– Second Activation: Display of Chronometer minutes.

The Chronometer function switch (CHR) located in the top right corner of the clock face supplies the three states, in order, that follow:

First activation: START	Temporary removal of the Elapsed Time hour display Return to zero Chronometer minute count start Chronometer sweep–hand start
Second activation: STOP	Maintains the display of the current indication
Third activation: RESET	Sweep–hand returns to zero Elapsed Time display returns

When primary electrical power is removed, the time base is maintained by the aircraft battery bus, all displays are blanked, and the sweep–hand, if active, stops. Current parameters continue to increment with the exception of the Chronometer and Elapsed Time functions. When primary power is restored, the upper LCD display shows the original function data and the lower display indicates 00 00. The Chronometer sweep–hand returns to zero and can be re–enabled if set to start from zero.

The ARINC 429 output bus is not active during standby power operation.

Greenwich Mean Time (GMT) is shown in the top 4–digit readout area of the clock face from 00:00 to 23:59 minutes when the function selector is in the GMT position. A single dot is displayed above the GMT legend.

Local time (from 00:00 to 23:59) is shown in the same location as GMT when the selector switch is in the LOC position. A single dot appears above the LOC legend, to give an alternative means of distinguishing local time from GMT, additional to switch position.

When the selector switch is set to DATE, the day and month are shown in the top 4–digit area of the clock face. The two left digits (01 to 12) identify the month and the two right digits identify the day (01 to 31) and the year (0 to 99). As the day and year occupy the same area, the display alternates each second between the two parameters. To aid interpretation while displaying the year, the left digits are blank. Leap years are programmed into clock operation.

Elapsed Time (ET) is indicated from 0 to 99:59 in the lower digital display area of the clock face and gives an indication of aircraft flight time. The mode is automatically enabled by a discrete Weight–Off–Wheels discrete input from the Proximity Switch Electronics Unit (PSEU) when the aircraft becomes airborne and can only be reset during a Weight–On–Wheels (WOW) condition. A colon separates the hours and minutes.

Minutes are indicated from 0 to 59 by the two right digits in the lower display area of the clock face with the left digits blanked. Seconds are shown against the round dial of the clock face by a sweep–hand activated by a stepper motor.

On aircraft with ModSum 4–126403 or 4–126434 incorporated, Refer to Figure 3.

Each clock is independently operated from the switches located on the bezel, and has instrument glass which is coated with an anti–reflection treatment on both sides.

The clock has the indications that follow:

- UTC
- CHR
- ET.

UTC: The UTC time is displayed from 0 to 23 hours, 59 minutes and 59 seconds on the six digit LCD (upper).

The display can be:

- Date, if this mode is selected (DT flag shown) (date format is mm/dd/yy)
- Local time, if this mode is selected (LT flag is shown).

The colon between the hours digits and the minutes digits is lit when the clock is on.

CHR: The chronometric time (minutes and seconds) is shown on the lower four–digit LCD from 0 to 99 hours, 59 seconds. The colon between the minutes and seconds is lit when the chronometer is on. The display is blanked when reset.

ET: Elapsed time is displayed from 0 to 99 hours, 59 minutes on the lower four–digit LCD. The elapsed time indication starts automatically when the aircraft is weight off wheels. The colon is lit when the ET indicator is on. Reset of the display is possible only when the aircraft is weight on wheels.

Mode: The MODE pushbutton switch allows selection of the various modes in turn:

- INT: In the INT mode the clock runs in internal mode and uses its own time base.

- LT: In LT mode the local time is shown and the clock uses its internal time base.
- DT: In this mode the date is shown and the clock uses its own internal time base.

At start−up the clock automatically starts in the INT mode.

The time can be set with the SET function which is made available by pressing the MODE pushbutton for a minimum of two seconds. In this function the UTC minute digits flash (the INT flag is on). Use the ET RST pushbutton to increase the minute indication; use the ET SEL pushbutton to decrease the minute indication. While in this function, each push of the MODE button will cycle through:

- The hour indication (the INT flag is on)
- The year indication (the DATE flag is on)
- The month indication (the DATE flag is on)
- The day indication (the DATE flag is on)
- The LT minute indication (the LT flag is on)
- The LT hour indication (the LT flag is on).

For each of these indications, use the ET RST pushbutton to increase the value and the ET SEL pushbutton to decrease the value.

CHR: This function operates with the CHR pushbutton switch which, with each push, cycles through start (colon is lit), stop (colon is blanked) and reset (display is blanked). If the display was occupied by the ET function before CHR is selected, the indicator flag toggles between ET and CHR.

ET SEL and ET RST: The elapsed time function is operated with these two pushbutton switches. If the display was occupied by the CHR function before ET SEL is selected, the indicator flag will toggle between CHR and ET. A push of the ET RST button blanks the ET display (only when the aircraft is weight on wheels).

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The left essential and battery power busses supply 28 Vdc electrical power through two 1 ampere circuit breakers to the CLOCK 1. The circuit breakers are located in positions E9 on the avionics circuit breaker panel and F3 on the right circuit breaker panel.

The right main and battery power busses supply 28 Vdc electrical power through two 1 ampere circuit breakers to the CLOCK 2. The circuit breakers are located in positions E6 on the avionics circuit breaker panel and G3 on the right circuit breaker panel.

Primary power is supplied from the left essential and right main 28 Vdc busses for CLOCK1 and CLOCK2 with a keep−alive voltage automatically supplied by the battery power bus when the primary electrical power is removed. The clocks are also supplied with variable 5 Vdc from the aircraft dimming bus to 4 display backlighting bulbs.

Each clock operates independently. The CLOCK1 is interfaced directly with the Cockpit Voice Recorder and both clocks are interfaced with the Flight Data Recorder (FDR) through the Flight Data Processing System (FDPS). The FDR normally records time from the CLOCK1 but will switch to CLOCK2 if the CLOCK1 malfunctions. Real time is recorded on both the Cockpit Voice Recorder (CVR) and Flight Data Recorder (FDR) to establish synchronization between the two recording systems.

The date is also supplied to the Flight Data Processing System (FDPS) located in the Integrated Flight Cabinet (IFC1) to show when the Type 1 systems malfunctioned for events in Central Diagnostics System (CDS) BITE reports.

Type 1 systems include:

- Electronic Indication System (EIS)
- Remote Control Audio Unit (RCAU)
- Audio and Radio Management System (ARMS)
- Engine Monitoring Unit (EMU).

The clocks send data to the CVR and FDPS through an ARINC 429 databus.

The clock installations interface with the aircraft systems that follow:

- IOP1, located in IFC 1, for input to the CDS and FDR
- PSEU for Weight–Off–Wheels discrete (to activate the Elapsed Time function)
- CVR for recording of real time and synchronization with the FDR.

Clock, Electronic (31–21–00)

Refer to Figure 4.

Two clocks are installed on the left and right sides of the glareshield panel assembly.

On aircraft without ModSum 4–126403 and 4–126434 incorporated, the clocks have dichroic LCDs that show white digits against a grey background.

On aircraft with ModSum 4–126403 or 4–126434 incorporated, the clocks have one six–digit and one–four digit LCDs that show white digits against a black background.

On aircraft without ModSum 4–126403 and 4–126434 incorporated, the five timebase functions that are available are as follows:

- GMT
- LOC
- ET
- Date
- Chronometer function (stopwatch) with a seconds sweep–hand.

On aircraft with ModSum 4–126403 or 4–126434 incorporated, the five timebase functions that are available are as follows:

- UTC
- LOC
- ET
- Date
- Chronometer function (stopwatch).

Dichroic liquid crystal decimal indicators (7–segment) display white digits against a black background. Digits, graduations and GMT, LOC, DATE, SET, ET, CHR markings are shown in light grey on a black background.

Four labels are attached to each clock that include the information that follows:

- An Identification label

- A Modification Status label
- A NO HIGH VOLTAGE TEST label
- A CAUTION label regarding the handling of electrostatic–sensitive devices.

The identification label contains the information that follows:

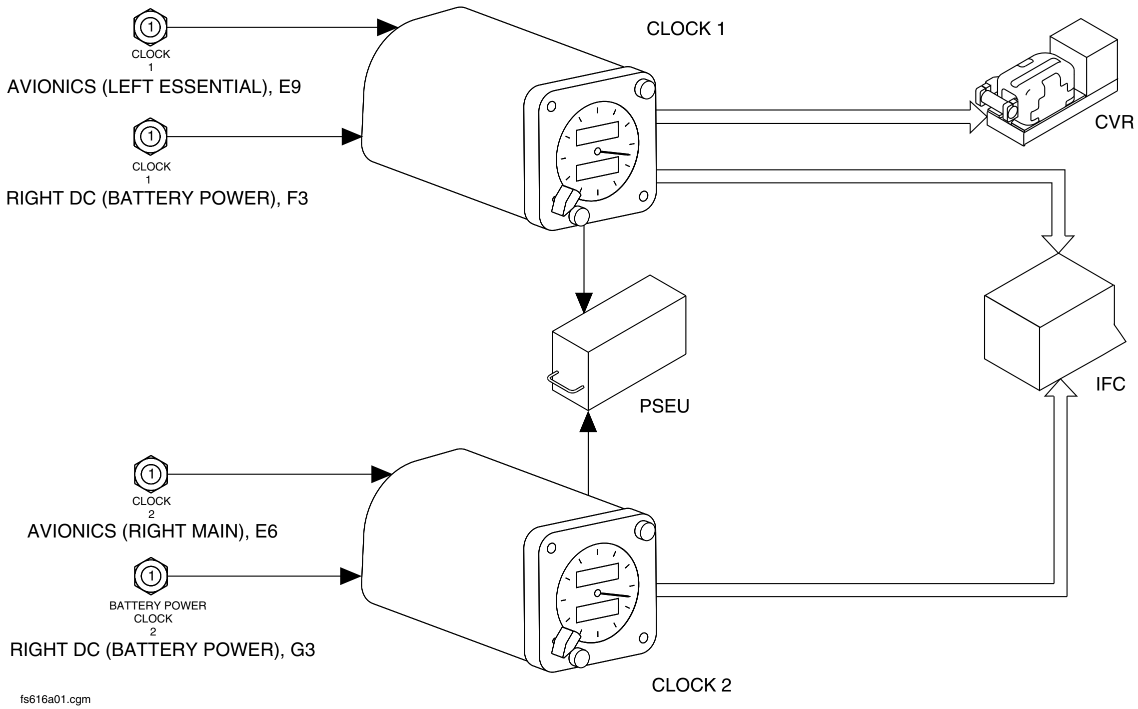
- Manufacturer's logo
- Manufacturer's code
- Name of equipment
- Equipment Part Number
- Equipment Serial Number
- Date of Manufacture
- Manufacturer's Inspection Stamp
- Administration Inspection Stamp
- Power Requirements
- Backlighting Requirements.

On aircraft without ModSum 4–126403 and 4–126434 incorporated, the electronic clock weighs 1.04±0.031 lb. (0.475±0.01 kg). It measures 2.4 in. (61 mm) high by 2.4 in. (61 mm) wide by 6.24 in. (156 mm) long (including front controls and the rear connector).

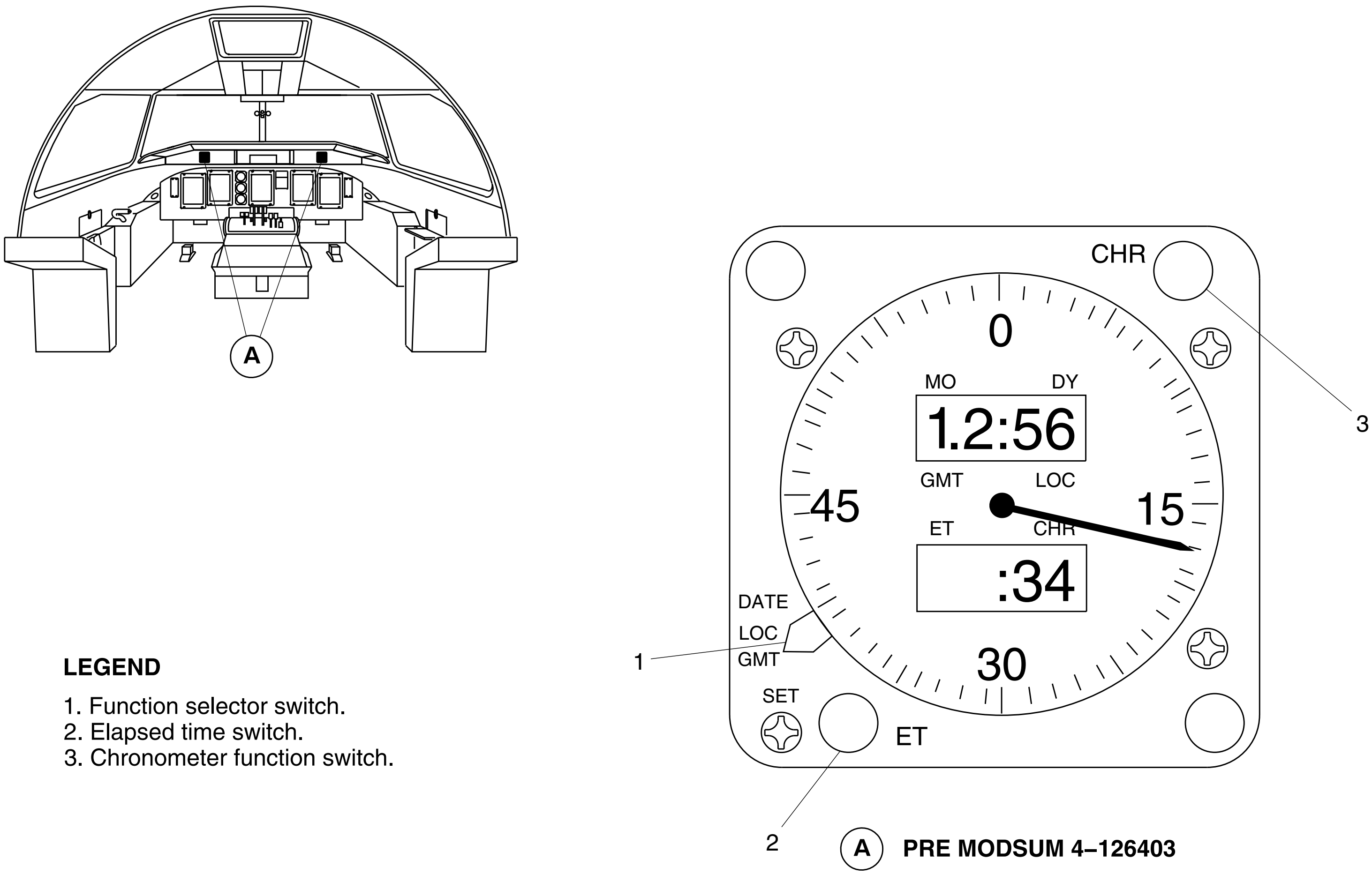
On aircraft with ModSum 4–126403 or 4–126434 incorporated, the electronic clock weighs 0.78±0.02 lb. (355±10.65 g). It measures 2.37 in. (60.32 mm) long by 2.37 in. (60.32 mm) wide.

On aircraft without ModSum 4–126403 and 4–126434 incorporated, the clocks are attached to the glareshield panel with one mounting screw.

On aircraft with ModSum 4–126403 or 4–126434 incorporated, the clocks are attached to the glareshield panel with two mounting screws.



CLOCK SYSTEM BLOCK DIAGRAM
Figure 1



LEGEND

1. Function selector switch.
2. Elapsed time switch.
3. Chronometer function switch.

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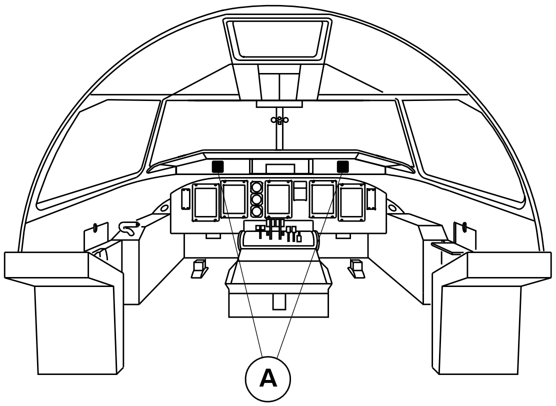
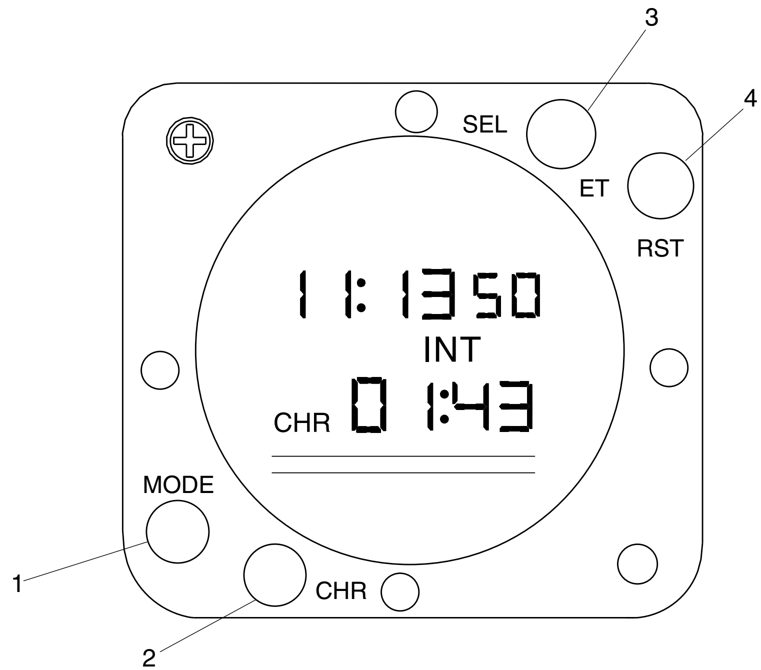
ELECTRONIC CLOCK
Figure 2





A POST MODSUM 4–126403

LEGEND

1. Mode pushbutton switch.
2. Chronometer function switch.
3. ET SEL pushbutton switch.
4. ET RST pushbutton switch.

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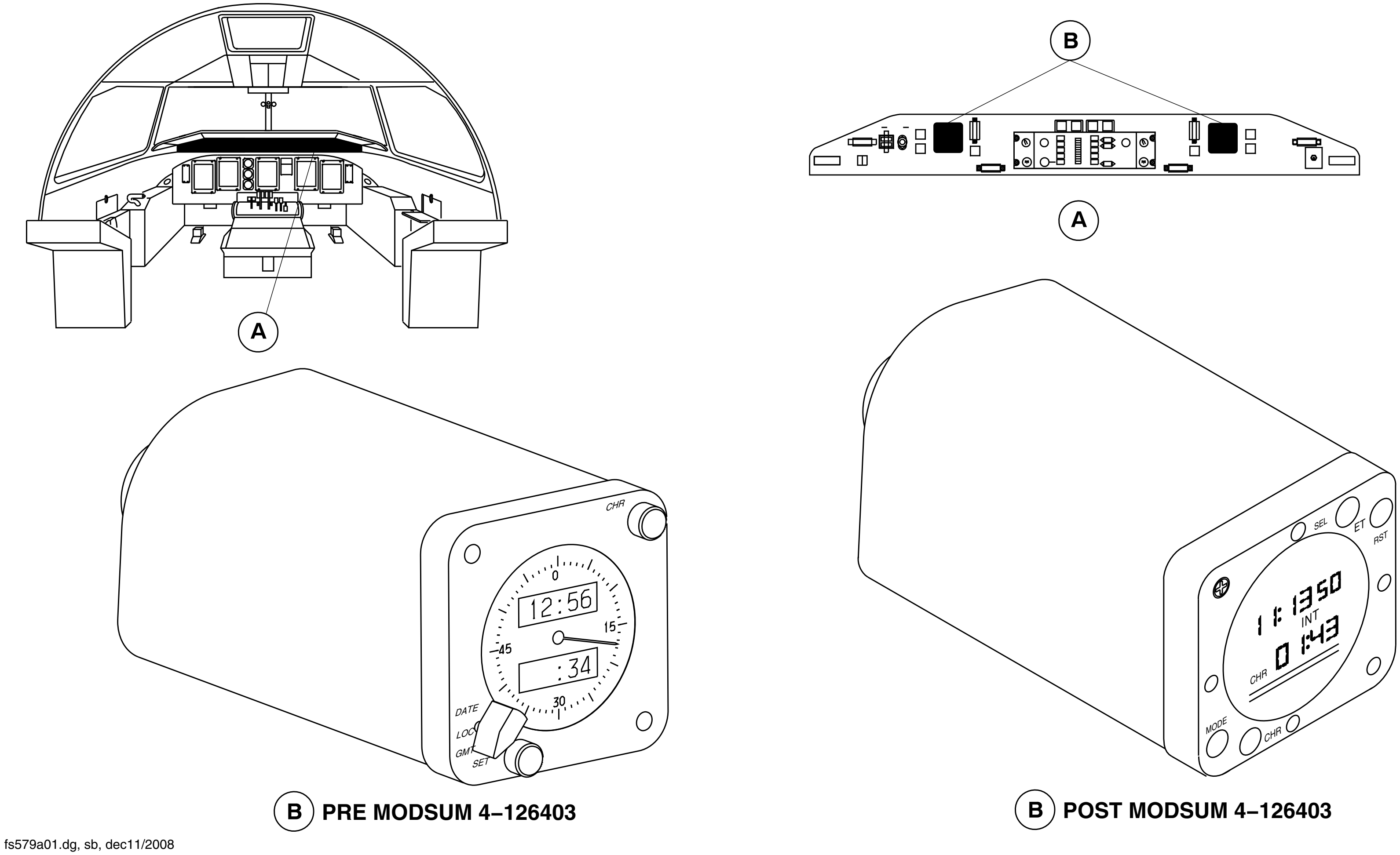
Electronic Clock
Figure 3



ELECTRONIC CLOCK LOCATOR
Figure 4

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**ON A/C ALL

31–30–00–001

RECORDERS

Introduction

The Solid State Flight Data Recorder (SSFDR) records aircraft parameter data and stores it in crash–protected memory for future retrieval purposes.

General Description

Refer to Figure 1.

Refer to Figure 2.

The Solid State Flight Data Recorder (SSFDR) is a crash survivable unit that can record up to 25 hours of aircraft parameters and clock data.

The Solid State Flight Data Recorder (SSFDR) receives from the Integrated Flight Cabinet (IFC 1), the aircraft parameters that follow:

– Flight path
– Speed
– Attitude
– Engine power
– Configuration
– Operation.

An inertia switch removes power to the SSFDR and Solid–State Cockpit Voice Recorder (SSCVR) when exposed to a high acceleration. Pushing a mechanical reset button on the switch assembly manually resets the switch following activation.

On aircraft with ModSum 4Q126473 incorporated, two inertia switches are there, one each, for both the SSFDR and SSCVR is installed, which removes power to the SSFDR and SSCVR when exposed to a high acceleration. Pushing a mechanical reset button on the switch assembly manually resets the switch following activation.

The Flight Data Recorder System (FDR) has the components that follow:

– Recorder, Flight Data (31–32–01)
– Tray, Mounting (31–32–06)
– Switch, Inertia (31–32–16)
– Switch, Test (31–32–21)
– Switch, Anticollision (31–32–26).

An Underwater Locator Beacon (ULB) (also known as Underwater Locating Device (ULD)) is attached to the Solid State Flight Data Recorder (SSFDR) handle. When immersed in either fresh water or saltwater, the ULB transmits an acoustic signal to help locate the SSFDR.

Detailed Description

The flight data recorder system has SSFDR and an acceleration inertia switch. The system interfaces with:

– Flight Data Processing System (FDPS)

- Test circuit
- Input power interlock arrangement
- Caution and warning panel.

The ULB is attached to the front of the SSFDR to easily access it for servicing and to read its battery expiratory date. It may also be used as a carrying handle.

The ULB is automatically activated when immersed in either fresh water or salt water at depths from 0.5 to 20,000 feet (0.15 to 6096 meters).

On aircraft without ModSum 4Q126626 incorporated, when the ULB is activated, it will transmits an acoustic signal of 37.5 ±1 kHz frequency for a duration of 30 days or more. The ULB is self–powered by a battery with a typical service life of six years from the date of manufacture. A label that shows the battery expiry date is attached to the ULB.

On aircraft with ModSum 4Q126626 incorporated, when the ULB is activated, it will transmits an acoustic signal of 37.5 ±1 kHz frequency for a duration of 90 days or more. The ULB is self–powered by a battery with a typical service life of seven years from the date of manufacture. A label that shows the battery expiry date is attached to the ULB.

The SSFDR system receives mandatory and non–mandatory parameters and discrete from aircraft systems at 128 words per second through an ARINC 717 databus. The unit records up to 25 hours of data in a crash–survivable memory module for subsequent retrieval and analysis. Ground–Based Equipment (GBE) and an RS422 interface are used to download the recorded data.

On aircraft with ModSum 4Q126473 OR SB84–31–82 incorporated, the SSFDR system receives mandatory and non–mandatory parameters and discrete from aircraft systems at 128 words per second or 512 words per second through an ARINC 717 databus. The unit records up to 533.1 hours of data at 128 words per second data transmission rate or 140.8 hours of data at 512 words per second data transmission rate in a crash–survivable memory module for subsequent retrieval and analysis. Ground–Based Equipment (GBE) and an ethernet interface are used to download the recorded data.

The GBE performs the functions that follow:

- Enables the real–time monitoring of aircraft parameters and discrete
- Shows the internal status of the SSFDR
- Downloads data from the Crash Survivable Memory Unit (CSMU)
- Does a functional test of the SSFDR.

The GBE interface connector (with protective cover) is installed in the front panel of the SSFDR and is used for download, test and maintenance functions without removing the LRU from the aircraft.

SSFDR system operations are controlled by a microprocessor which supplies the logical processing and data flow necessary to do all system functions.

The SSFDR operates in the modes that follow:

- Power off mode
- Initialization mode
- Record mode

- Continuous monitor mode
- Test mode
- Down load mode
- Power down mode.

Power Off: The SSFDR operates when power is removed for more than 200 milliseconds. No functions are available. Applying power exits the power off mode.

Initialization: The SSFDR operates in the initialization mode immediately upon receiving power. It initializes and performs 95 percent of the Built–In–Test (BIT) within 500 millisecond (On aircraft with ModSum 4Q126473 OR SB84–31–82 incorporated the initialization is completed within 250 millisecond). It determines the SSFDR status on power–up in addition to normal recorder operations. A history of the BIT status is maintained within the CSMU and can be downloaded with the GBE. During initialization, the system records the unit configuration that follows:

- Memory size
- Rate of transfer
- Address of next available memory location to be recorded.

If the BIT fails, the external BIT light on the front panel of the SSFDR comes on. The SSFDR exits the initialization mode of operation if it cannot record data due to a critical failure. It transmits a signal to the continuous monitor mode to stop it from recording, and the FLT DATA RECORDER annunciator on the Caution Warning Panel comes on. The Central Diagnostics System (CDS) receives and records an equivalent maintenance output through the Integrated Flight Cabinet (IFC 1).

Record Mode: Recording data begins immediately following the initialization and self–test mode. It operates in the record mode within 500 millisecond of applying power (On aircraft with ModSum 4Q126473 OR SB84–31–82 it is within 250 millisecond). The SSFDR stops recording if it does not receive data for more than five seconds, causing the FLT DATA RECORDER caution light to come on.

The SSFDR stays in the record mode until it is connected to the Ground Based Equipment (GBE). The type of GBE selects what mode the SSFDR will enter next.

The SSFDR transmits the received data back to the Input/Output Processor (IOP1). This makes sure that the SSFDR is properly receiving the input data. If the status discrete senses a failure, the "loopback" data is not present. The FLT DATA RECORDER caution light on the caution and warning panel will come on.

The conditions that follow exit the SSFDR from the record mode:

- Removing power
- Detection of the download mode
- Command from the IOP.

Monitor Mode: The monitor mode supplies aircraft installation diagnostics and troubleshooting.

The SSFDR continuously does the following:

- Background tests on system hardware
- Monitors data inputs.

In the monitor mode, the SSFDR records normal flight data in parallel with the data bus to the GBE. The GBE commands the SSFDR to operate in the monitor mode and to output a "data frame", once per second, that shows the status of the recorder, and the record data, 8 times per second.

Test Mode: The test mode supplies "Return to Service" testing of the SSFDR. The external Automated Test Unit (ATU) commands the SSFDR to operate in the test mode. The normal record mode stops when the unit is operating in the test mode. The ATU controls the exit from the test mode.

Down Load Mode: The SSFDR starts the down load mode when initiated by the external Down Load Unit (DLU). A DLU is a PC−based device that has the necessary software to download the contents of the CSMU and monitor the data parameters as they are received. The data can be downloaded through the front connector of the SSFDR. When the information is downloaded, the SSFDR record mode stops and the status discrete will show a failure.

A full 25 hours download of data recorded at 128 words per second can last approximately 10 minutes dependent upon PC speed.

On aircraft with ModSum 4Q126473 OR SB84−31−82 incorporated, a full 140.8 hours download of data recorded at 512 words per second or 533.1 hours of data recorded at 128 words per second can last approximately 45 minutes dependent upon PC speed.

Power Down Mode: The SSFDR power supply maintains internal voltages to continue normal operation during a power interruption lasting less than 200 milliseconds. For power interruptions greater than 200 milliseconds:

- The data in the RAM is lost
- The initialization mode starts.

In the power down mode, the SSFDR continues to receive data. It records data in the volatile Random Access Memory (RAM) rather than the non−volatile CSMU. If the power restores within 200 millisecond, the data in the RAM is transferred to the CSMU. The normal recording process then continues.

The impact switch, when triggered, removes aircraft power to the SSFDR system caused by a high impact. An acceleration greater than 5.5 g opens the impact switch. It makes sure that the recorded data before the impact is not overwritten. The RESET BUTTON on the side of the impact switch is pushed to re−arm the switch following activation.

The SSFDR is electrically powered by 28 VDC through an interlock system. The SSFDR starts to record when any of the conditions are met that follow:

- Red anti collision lights are switched ON
- White anti collision lights are switched ON
- Both engines are operating, engine−operating oil pressure is sensed from both engines
- Weight−Off−Wheels is sensed

– Test switch is set to GND TEST.

Refer to Figure 3.

The FLIGHT DATA RCDR toggle switch on the Flight Data Recorder Panel is set to the GND TEST position to test the SSFDR while the aircraft is on the ground.

Refer to Figure 4.

If the SSFDR BITE senses satisfactory system operation, the FLT DATA RECORDER indication in the caution and warning panel goes out.

Refer to Figure 5.

The RED–OFF–WHITE A/COL toggle switch on the Exterior Lights Panel is set to the WHITE or RED position to power the SSFDR.

The Left Main Bus supplies the Solid State Flight Data Recorder (SSFDR) with 28 VDC electrical power through a 1 Ampere circuit breaker. The circuit breaker is located on the 28 VDC avionics circuit breaker panel at position F3.

On aircraft with ModSum 4Q126473 OR SB84–31–82 incorporated the Right Essential Bus supplies the Solid State Flight Data Recorder (SSFDR) with 28 VDC electrical power through a 5 Ampere circuit breaker. The circuit breaker is located on the 28 VDC avionics circuit breaker panel at position E10. The circuit breaker E10 also provides 28 VDC power to the Signal Conditioning unit (SCU).

Two digital clocks supply the SSFDR and Solid State Cockpit Voice Recorder (SSCVR) with time information. The clocks synchronize the aircraft data with audio information recorded by the SSCVR. The SSCVR receives ARINC 429 time information directly from Digital Clock 1. Digital Clock 2 supplies time information to the SSFDR through the Input/Output Processor (IOP1). The IOP1 converts ARINC 429 data to ARINC 717.

The Flight Data Processing System (FDPS) includes the Flight Data Concentrator which is located in the IFC1 within the IOP1 module. The acquisition of discrete and analog signals is done by the Input Output Module 1 (IOM1) while IOP1 receives digital data. The flight data concentrator receives, samples, conditions and digitizes the received flight data and transmits it to the SSFDR through an ARINC 573/717 bus generated within IOP1. The output of the flight data concentrator is organized into frames which are repeated every four seconds. Each frame has four sub–frames of 1 second, that consist of 128 twelve–bit words each. Discrete data are packed into discrete words. The data transmission rate is therefore 128 words per second (On aircraft with ModSum 4Q126473 OR SB84–31–82 incorporated, the data transmission rate is 128 or 512 words per second).

The SSFDR system interfaces with the aircraft systems/busses that follow:

– Left Main 28 VDC bus through 1 A circuit breaker F3
– On aircraft with ModSum 4Q126473 OR SB84–31–82 incorporated, Right Essential 28 VDC bus through 5 A circuit breaker E10.
– Avionics circuit breaker panel
– IOM1 (located in IFC1) for input to the Centralized Diagnostic System (CDS) (maintenance flag)
– IOP1 (located in IFC1) for input ARINC 573/717 data and loop–back data
– Overhead anti collision lights (red and white) switch for power interlock

- Proximity Sensor Electronic Unit (PSEU) for WOW input (part of power interlock)
- Overhead test switch
- Caution panel (status flag)
- Inertia switch (removes power for accelerations in excess of 5.5 g).

The SSFDR operates in the modes that follow depending on its interface:

MODE	INTERFACE
Record	IOP1 using ARINC 717 data bus
Monitor and Test	Automated Test Unit using RS–422 data bus
Download	Download Unit using RS–422 data bus

NOTE

On the aircraft with V800/810 Avionics Suite installed, the flight data recorder parameters that follow are invalid:

- MFD1 Selection (WX/TERRAIN)
- MFD2 Selection (WX/TERRAIN)

The flight data recorder parameters that follow are controlled using ARINC 429 low speed busses (12.0 to 14.5 kilobits per second):

PARAMETER	SOURCE
Barometric Corrected Altitude	ADU1, ADU2
Computed Air Speed	ADU1, ADU2
Static Air Temperature	ADU1, ADU2
Pilot's Barometric Setting	ADU1
AFCS engagement	FGM1, FGM2
YD engagement	FGM1, FGM2
AFCS Mode Selection	FGM1, FGM2
AFCS Selected Speed	FGM1, FGM2
AFCS Selected Vertical Speed	FGM1, FGM2
Tactile Control Steering Status	FGM1, FGM2
HSI selection	IOP1, IOP2
Heading Selection	IOP1, IOP2
Altitude Selection	IOP1, IOP2
DUs Health Status	IOP1, IOP2
Course Selection	IOP1, IOP2

PARAMETER	SOURCE
DH Selection	IOP1, IOP2
Recorder Elapse Time	IOP1
Pitch Angle	AHRS1, AHRS2
Roll Angle	AHRS1, AHRS2
Magnetic Heading	AHRS1, AHRS2
Normal Body Acceleration	AHRS1, AHRS2
Lateral Body Acceleration	AHRS1, AHRS2
Longitudinal Body Acceleration	AHRS1, AHRS2
Time	Clock 1, Clock 2
Condition Lever Angle	FADEC1, FADEC2
Power Lever Angle	FADEC1, FADEC2
ECS Bleed Valve Position	FADEC1, FADEC2

PARAMETER	SOURCE
NH	FADEC1, FADEC2
NP	FADEC1, FADEC2
Torque Command (QPLA)	FADEC1, FADEC2
Torque Target	FADEC1, FADEC2
TQ	FADEC1, FADEC2
ITT	FADEC1, FADEC2
Thrust Reverse Status	FADEC1, FADEC2
Autofeather Event Marker	FADEC1, FADEC2
Autofeather Armed Indication	PEC
DUs on or off status	EIS
MFD Reversions	EIS
ADU Reversions	EIS
AHRS Reversions	EIS

PARAMETER	SOURCE
Mono Mode	EIS
DUs Image Selections	EIS
Glideslope Deviation	NAV1, NAV2
Localizer Deviation	NAV1, NAV2
VOR Bearing	NAV1, NAV2
APU Bleed Valve Position	APU FADEC
VHF Communications PTT	RCAU
HF Communications PTT	RCAU
Standby Battery Hot	CWP
No. 3 Standby Hydraulic Pump	CWP
Cabin Pressure	CWP
Touched Runway	CWP
AC Busses	CWP
TRUs	CWP
DC Generators	CWP
AC Generators	CWP
DC Busses	CWP
FADEC Malfunctions	CWP
Doors	CWP
Check Fire Detection	CWP
Smoke	CWP
Main Battery Hot	CWP
Auxiliary Battery Hot	CWP
Engine Oil Pressure	CWP
Engine Hydraulic Pumps	CWP

Recorder, Flight Data

Refer to Figure 6.

Refer to Figure 7.

The SSFDR has the components that follow:

- Circuit Card Assemblies (CCAs)
- Controller

- Memory
- Power supply regulator
- Power supply filter.

The CCAs are interfaced with each other or to external devices through a series of connectors.

The Controller CCA performs the main SSFDR control functions. The CCA controls the reception and transmission of all aircraft parameter and discrete data. The data inputs flow through the controller and are presented at the CSMU for storage. All circuit card assemblies are conformably coated for protection from the effects of moisture, humidity and vibration.

The memory functional block includes 18 flash Electrically Erasable Programmable Read Only Memory (EEPROM) memory chips, each capable of 8 megabits (1 mega byte) storage capacity and powered by 5 VDC. The unit can store the last 25 hours of recorded data. An additional EEPROM chip is installed for BITE messages generated during BIT operations. An Elapsed Time Indication (ETI), repair history, bad memory locations and pointers to the most recent ARINC 717 data are also stored.

The power supply uses the 28 VDC input from the rear connector through the controller CCA, then filters and regulates the power to create the necessary voltages for system operation. The power supply circuitry also monitors input and output power characteristics. Voltages used by the SSFDR are +5, +12, and −12 VDC. Additional functions of the power supply are the generation of system reset and power−down interrupt signals. The reset signal is triggered when the +5 VDC regulated signal drops to approximately 4.75 VDC. The Power−Down Interrupt occurs for a primary power input drop of approximately 13 VDC.

The SSFDR has an identification label that includes the information that follows:

- Unit part number
- Weight
- Serial number
- Manufacturing date
- TSO certified number
- DO−160C environmental certification categories.

A modification status label (hardware/software) is also attached.

The SSFDR chassis is painted a bright international orange and is marked with the black letters "FLIGHT RECORDER DO NOT OPEN" and "ENREGISTREUR DE VOL NE PAS OUVRIR".

Reflective tape is attached to the SSFDR external surface to further help recover it.

The SSFDR has two flex print assemblies; one between the controller CCA and the LRU rear connector and the other between the controller CCA and the CSMU. This latter assembly is constructed such that in the event of a crash, should the flex print be torn, corruption of the FLASH memory will not occur.

The housing is designed such that the crash−survivable enclosure is installed on top and towards the front of a thin−walled chassis. The recorder housing has been designed to be Electro−Magnetic Interference (EMI) tolerant.

The SSFDR front fasteners are electrically bonded to the chassis ground. The rear interface connector shell includes bonding surfaces which maintain good ground continuity between the recorder chassis and the recorder mounting frame.

On aircraft with ModSum 4Q126473 OR SB84–31–82 incorporated, the SSFDR is directly mounted on to the tray with help of four bolts, which provides bonding to the chassis ground.

The SSFDR is designed to operate in extreme environmental conditions with no forced air cooling. When installed, air can flow around the unit and supply cooling.

The SSFDR is installed in the rear fuselage of the aircraft between station points X942.57 and X958.23 at Z134.46 and to the left of aircraft centerline. The SSFDR weighs less than 16 pounds (7.26 kilograms).

On aircraft with ModSum 4Q126473 OR SB84–31–82 incorporated, The SSFDR weighs 8 pounds (3.63 kilograms).

The SSFDR is installed to the tray with corrosion resistant steel fasteners. The unit is located adjacent to the Solid–State Cockpit Voice Recorder (SSCVR).

Tray, Mounting

The SSFDR is installed in a ½ ATR short mounting tray that is electrically bonded to the airframe by a bonding strap.

On aircraft with ModSum 4Q126473 OR SB84–31–82 incorporated, the SSFDR tray is directly mounted on to the aircraft structure with help of four bolts, which provides bonding to the chassis ground.

The SSFDR mounting tray is located in the tail of the aircraft at stations X958.23 and Z134.43.

Switch, Inertia

Refer to Figure 8.

The inertia switch supplies the SSFDR system with the capability to remove power from the system if an acceleration of greater than 5.5 g is applied to the switch as a consequence of high impact landing. This makes sure that SSFDR recorded data prior to the impact is stored and cannot be overwritten.

The normally–closed inertia switch has an impact activated double–pole latching switch; one pole of which is connected in series with the 28 VDC aircraft supply to the SSFDR. (The second pole is connected to the SSCVR). The switch latches to an open state when activated.

On aircraft with ModSum 4Q126473 incorporated, two inertia switches are there, one each, for both the SSFDR and SSCVR is installed. The existing inertia switch provides 28 VDC power to the SSFDR and the new inertia switch, installed above the SSFDR inertia switch, provides the 28 VDC power to the SSCVR.

A manually operated RESET BUTTON, located on the side of the inertia switch is used to re–arm the switch.

The current rating of each pole of the switch is 5 A at 28 VDC.

NOTE

The maximum DC current of the SSFDR is less than 500 mA.

The inertia switch connector is installed on the side panel of the SSFDR.

The unit is installed at 45 degrees with the reset switch pointing towards ground and the front of the aircraft to sense upward and downward acceleration. It is attached to a bracket located below the cabin floor at station point X195.8, Y0, Z88.9.

Switch, Test

Refer to Figure 9.

When the FLIGHT DATA RCDR switch is set to the GND TEST position, the normal "on ground" power removal circuits are bypassed. The pre–flight GND TEST switch located in the flight compartment on the overhead panel, supplies power to the SSFDR for the duration that the switch is held. If the SSFDR BITE indicates satisfactory system operation, the caution panel FLT DATA RECORDER light goes out.

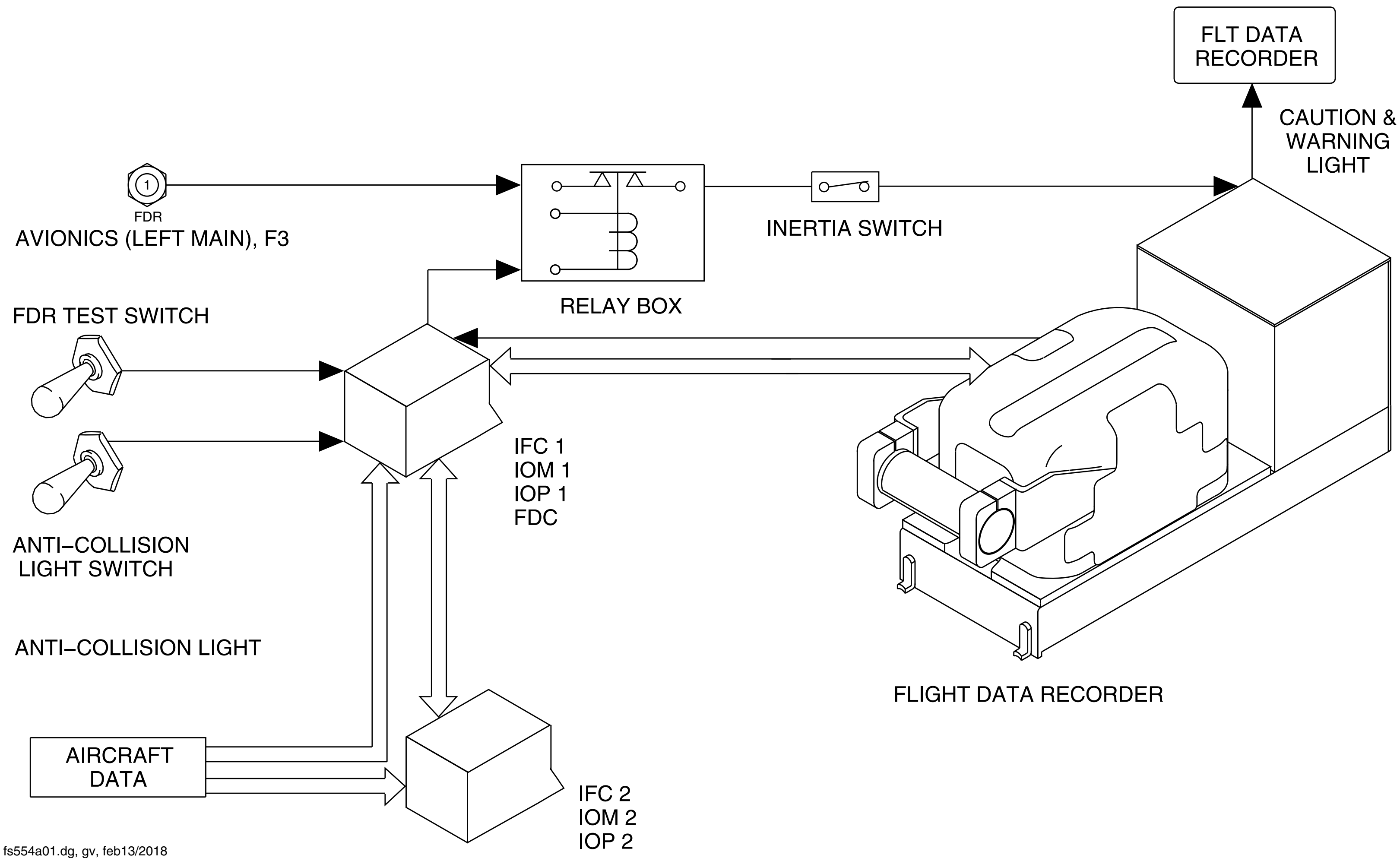
The NORM–GND TEST toggle switch is located in the flight compartment on the overhead FLIGHT DATA RCDR panel.

Switch, Anti–Collision

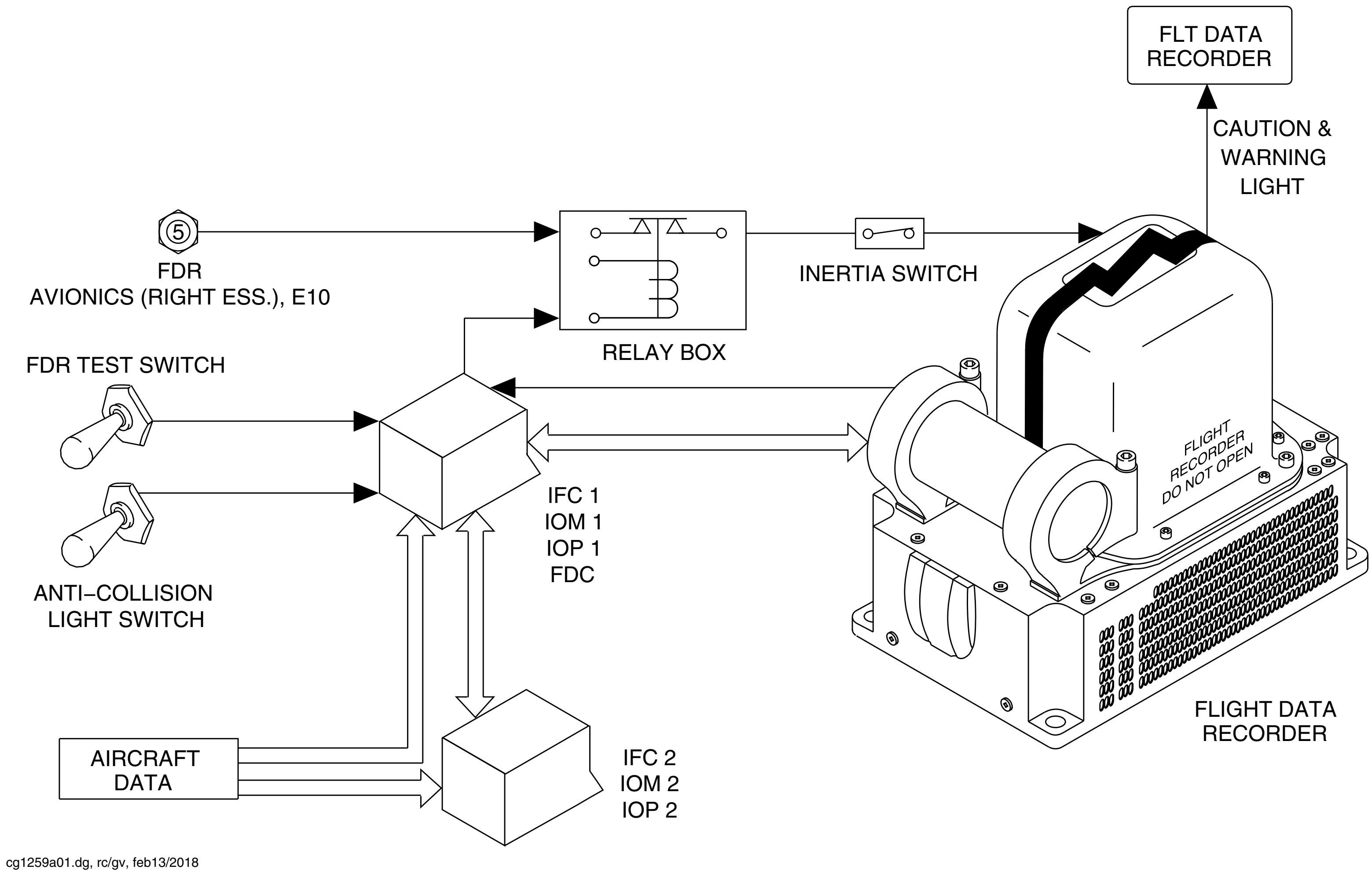
Refer to Figure 10.

When the anti collision lights switch is set to the RED or WHITE position, the normal "on ground" power removal circuits are bypassed. The anti collision lights switch is located in the flight compartment on the overhead panel. If the SSFDR BITE indicates satisfactory system operation, the caution panel FLT DATA RECORDER light goes out.

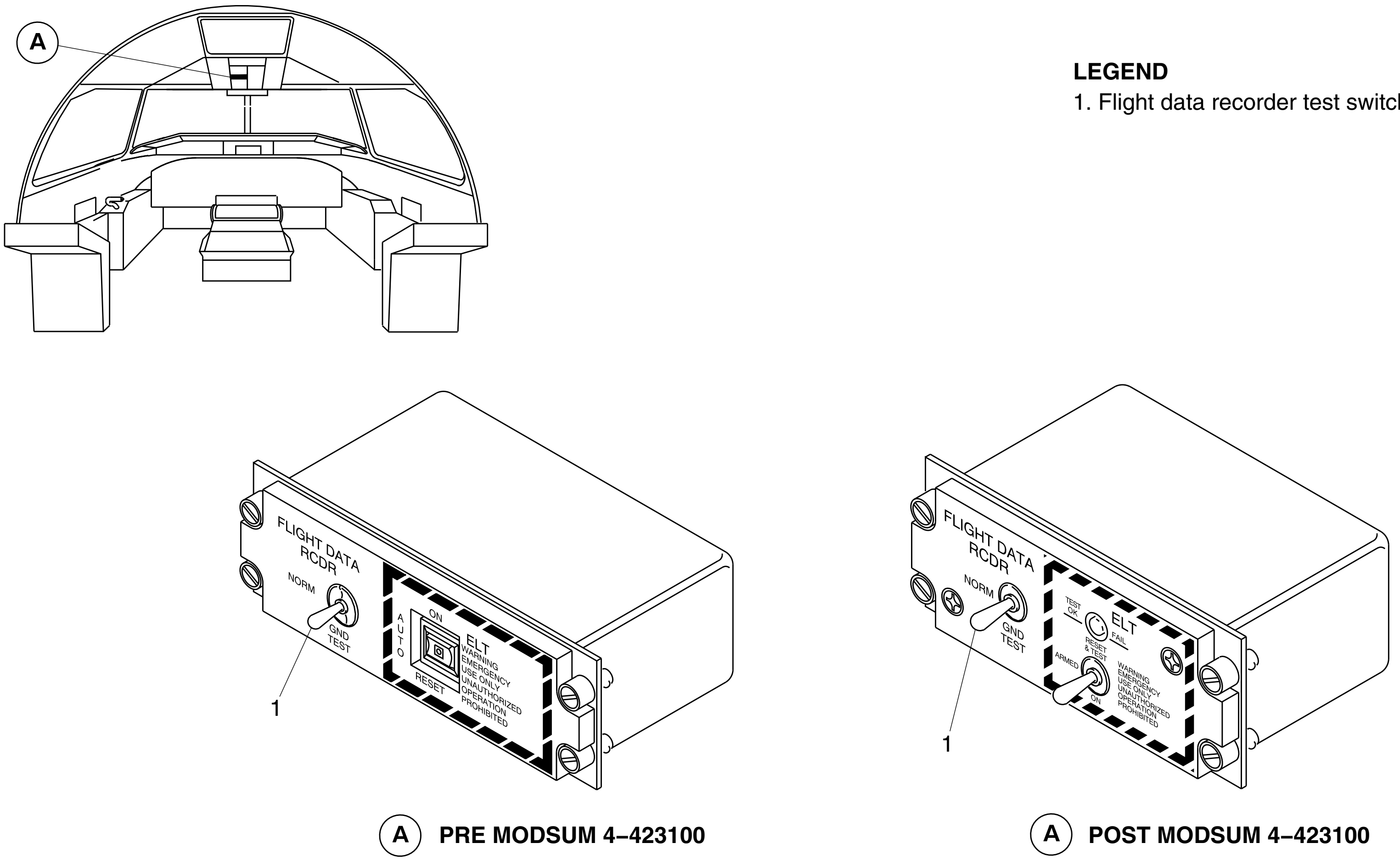
The RED–OFF–WHITE A/COL switch is located in the flight compartment on the overhead EXTERIOR LIGHTS panel.



FLIGHT DATA RECORDER BLOCK DIAGRAM
Figure 1



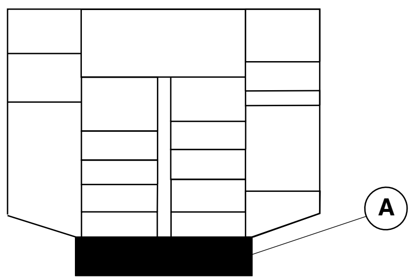
Flight Data Recorder (Universal) Block Diagram
Figure 2



LEGEND
1. Flight data recorder test switch.

fs537a01.dg, rc, oct04/2010

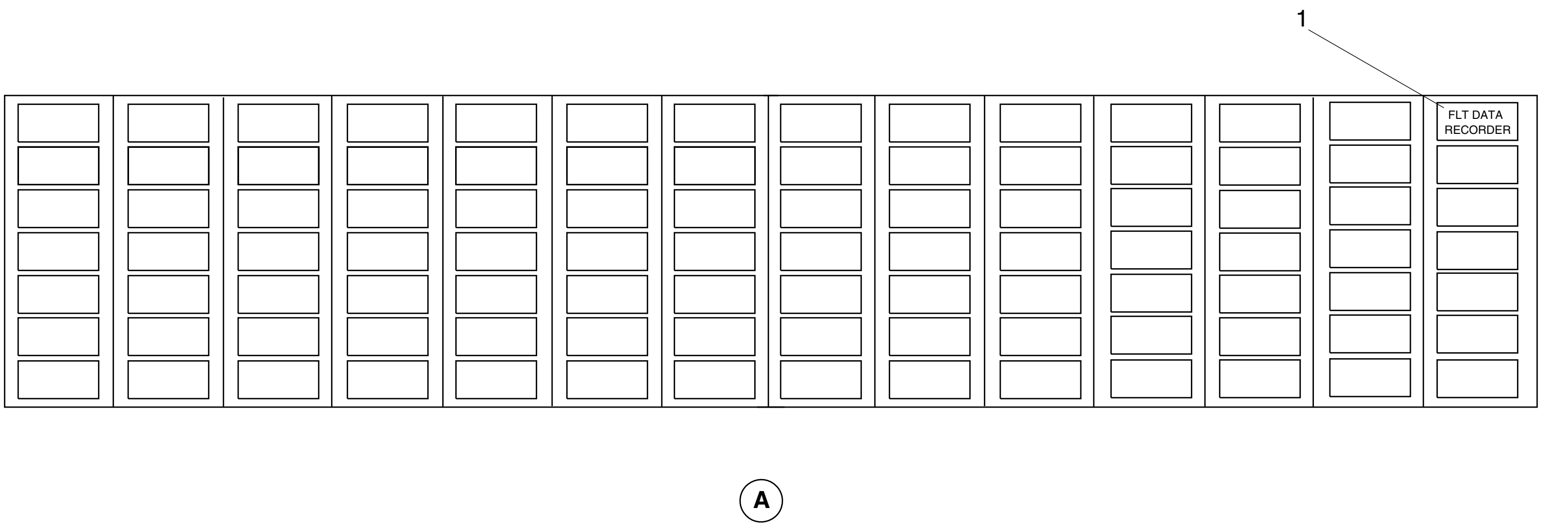
Flight Data Test Switch
Figure 3



OVERHEAD CONSOLE

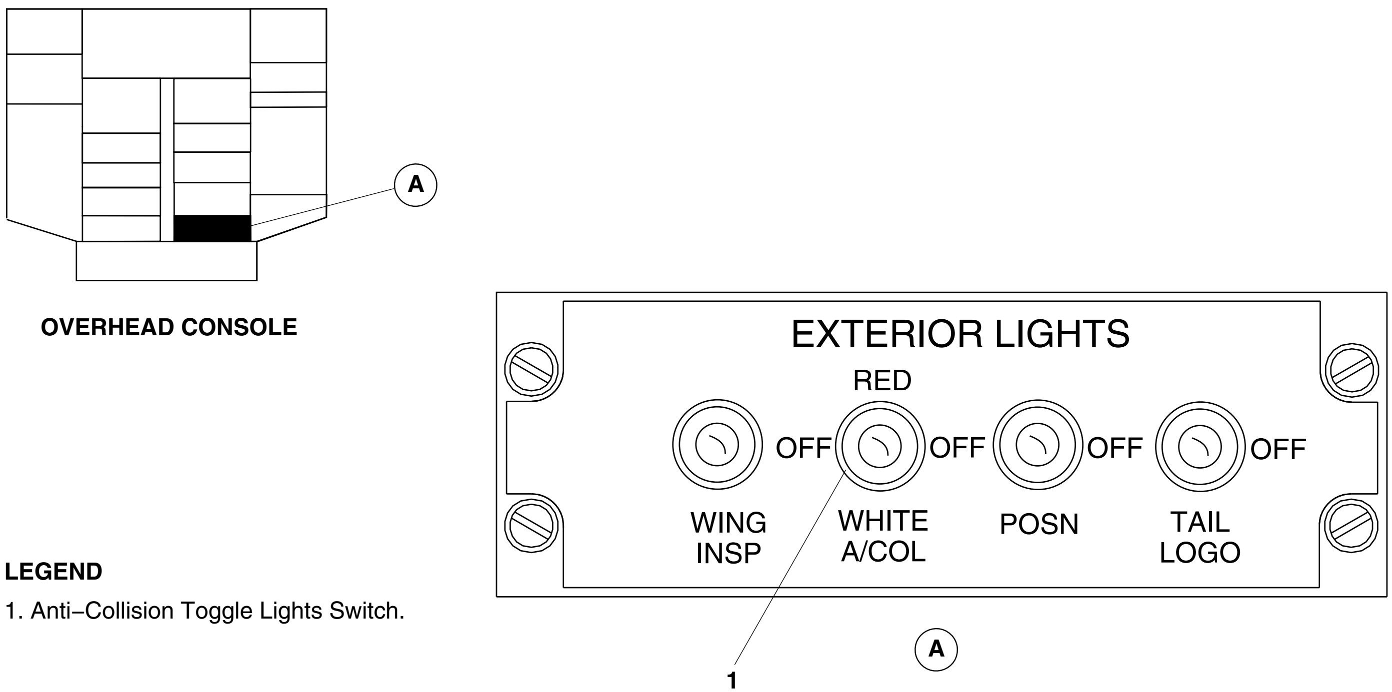
LEGEND

1. Flight Data Recorder (Amber).



fsa50a01.cgm

FDR CAUTION INDICATION
Figure 4

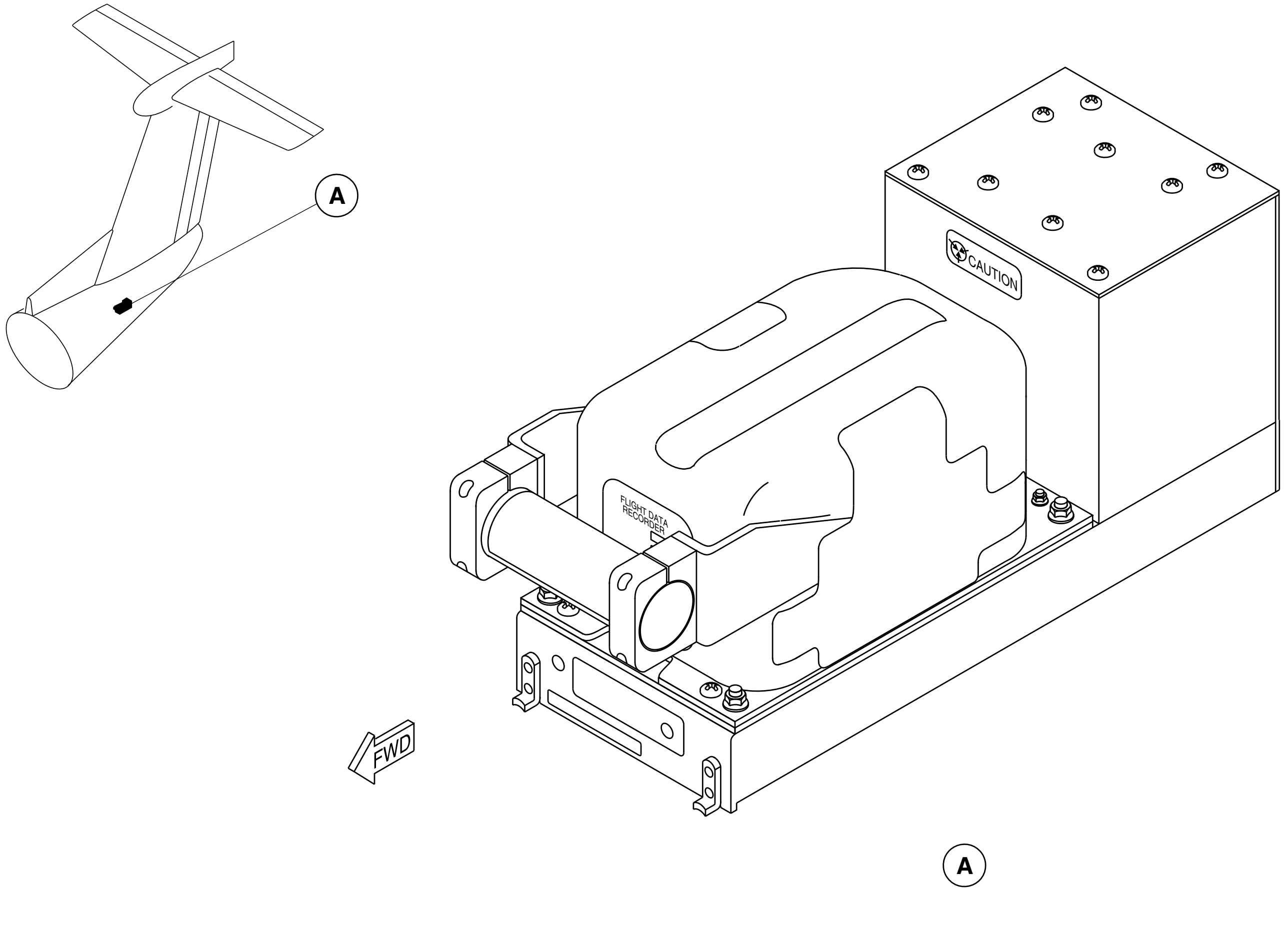


LEGEND

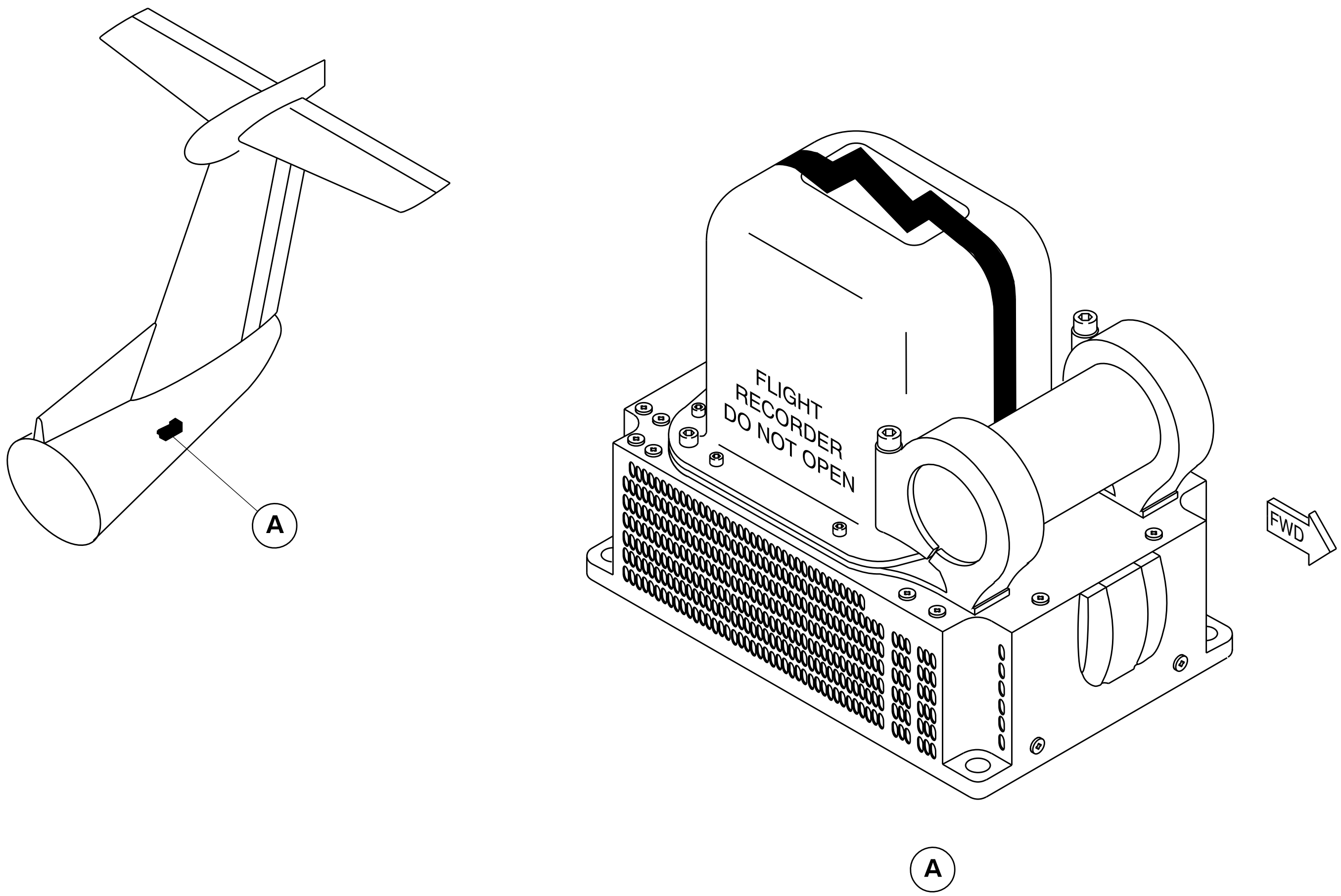
1. Anti–Collision Toggle Lights Switch.

fs084a01.cgm

ANTI–COLLISION LIGHT SWITCH
Figure 5



FLIGHT DATA RECORDER LOCATOR
Figure 6



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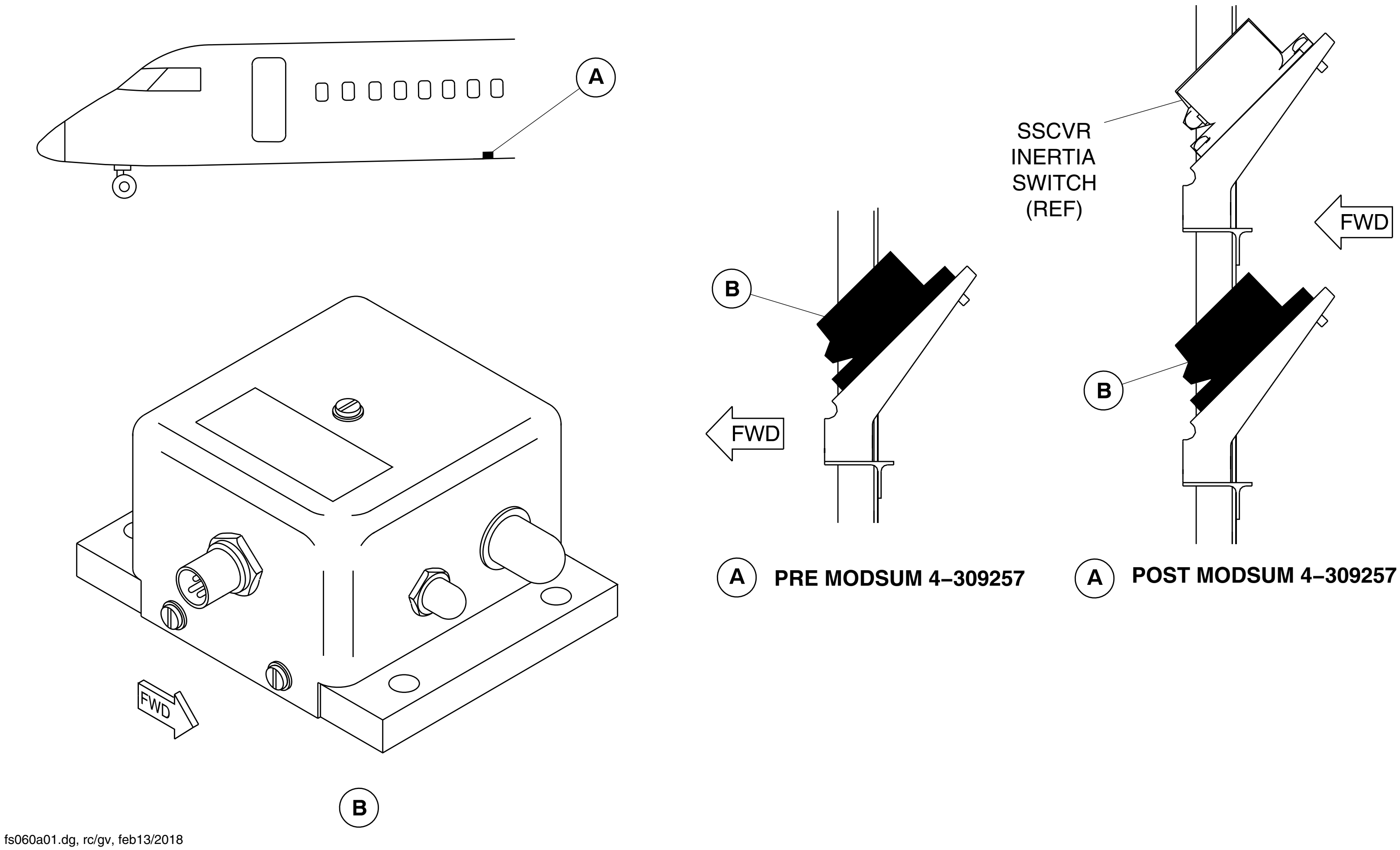
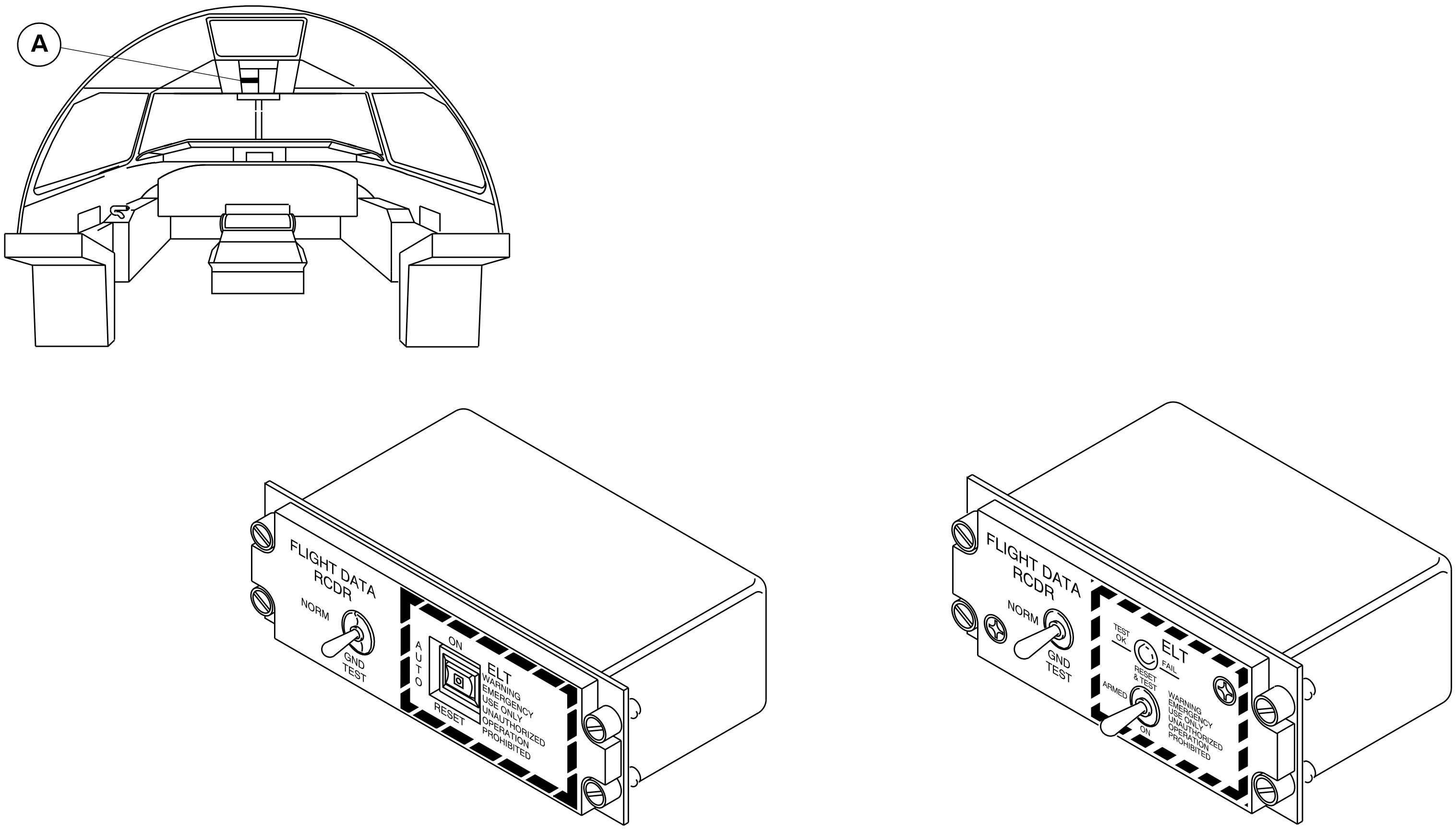
Flight Data Recorder (Universal) Locator
Figure 7



Flight Data Recorder Inertia Switch Locator
Figure 8

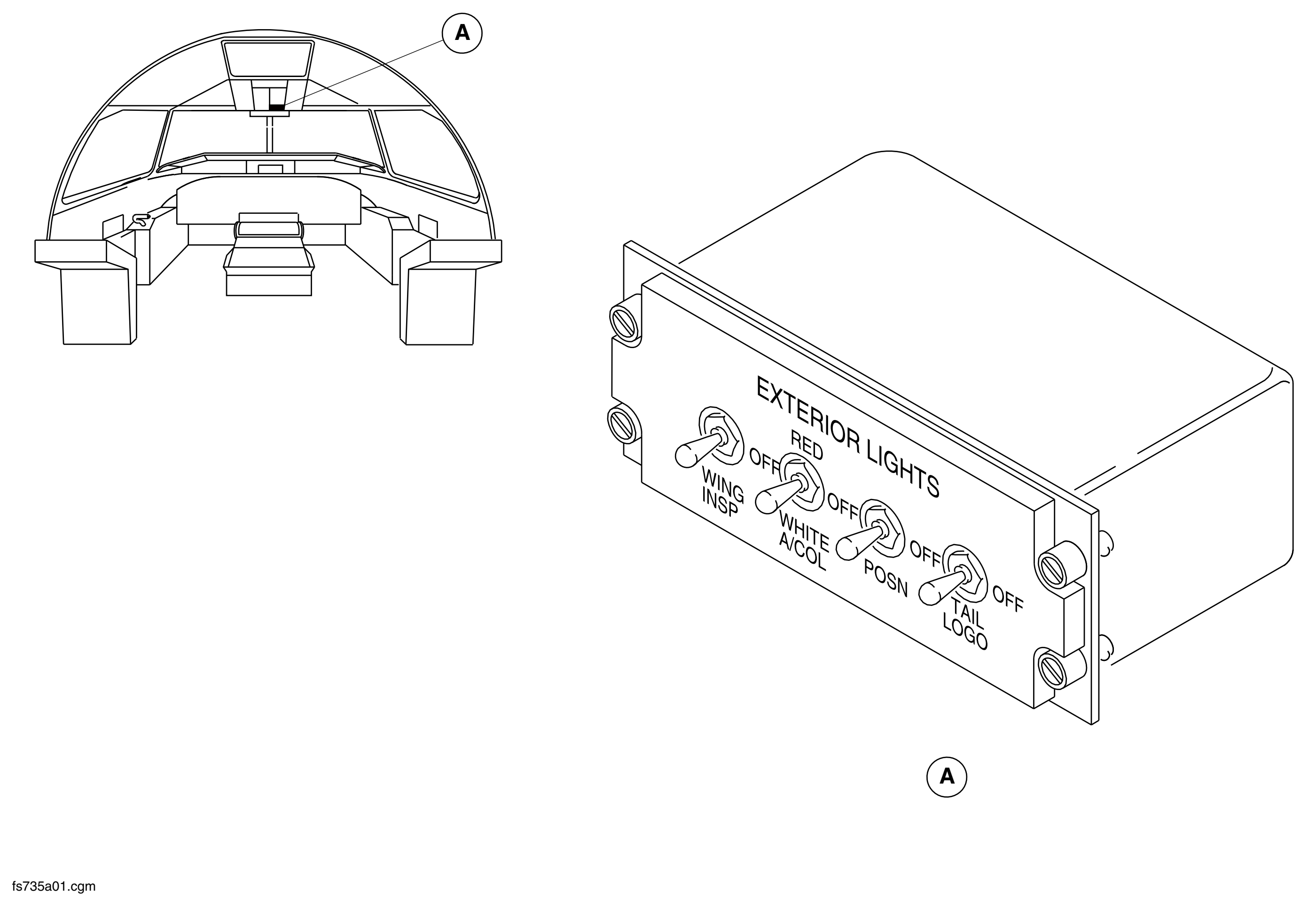


A PRE MODSUM 4–423100

A POST MODSUM 4–423100

fs536a01.dg, rc, oct04/2010

Flight Data Recorder Panel Locator
Figure 9



RECORDERS, EXTERIOR LIGHTS PANEL LOCATION
Figure 10

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31–30–00–002

SIGNAL CONDITIONING UNIT

Introduction

Refer to Figure 11.

The discrete, force, position and pressure sensors measure the control column inputs and also the affector response. The aircraft system sensors convert the control column inputs into electrical signals and transmit them to Flight Data Recorder Signal Conditioning Unit (FSCU). The FSCU supplies excitation, demodulation and filtering signal for the sensing devices. The FSCU also supplies ARINC 429 data for the Flight Data Recorder (FDR). The FSCU operates with two Computer Software Configuration Items (CSCI).

General Description

Refer to Figure 12.

The FSCU operates in the states that follow:

– On
– Off

The FSCU interfaces with the aircraft 28 VDC power supply. The power supply converts the input into the conditioned supply and reference voltages.

The FSCU interfaces with the 24 discrete inputs and 4 discrete outputs. The impedance of each discrete input to the ground is 300 KΩ.

The FSCU receives data from at least five ARINC 429 receivers at 100 kbps.

The FSCU supplies analog inputs to interface with the devices that follow:

– Potentiometers
– Linear Variable Differential Transformers (LVDTs)
– Pressure Sensors

The FSCU performs the tests to detect the critical faults that follow:

– Power supply fault
– A/D converter fault
– SRAM fault
– Program memory fault
– EEPROM checksum fault
– Watchdog fault

Detailed Description

Refer to Figure 12.

The FSCU operates in the modes that follow:

– Start mode
– Power–up self–test mode
– Halt mode
– System operation mode
– CSCI 1 Download mode

Start Mode: First when the power is applied, the FSCU initiates in this mode. Based on the pattern of the mode pins, the FSCU change over to the appropriate mode.

Power−Up Self−Test Mode: The FSCU enters in this mode on power−up. In this mode, the software performs Built In Test (BIT) for hardware functions critical to the microcontroller's operation, followed by BIT on the watchdog. If all the tests pass, the software change over to the system operational mode. If any of the power−up test fails, the controller enters to the halt mode.

Halt Mode: The controller enters halt mode when either software or hardware detects a critical hardware failure. In the halt mode, the controller acts as if power has been removed from it. This mode may be exited by removing power from the FSCU.

System Operation Mode: In system operational mode, the FSCU performs all monitoring and transition functions, performs BIT on the interfaces and on all the electrically connected LRUs and transmits data in the ARINC 429 data bus. The FSCU change over to halt mode if either the software or hardware detects a critical failure.

CSCI 1 Download Mode: The purpose of this mode is to download the operational software. This mode also requires that the FSCU be connected to an external data terminal designed for download. To start downloading, a correct pattern of the mode pins must be set and a correct download password must be supplied by the operator. The FSCU will accept the new program code when the password is confirmed. This mode may be exited by removing power from the FSCU.

ARINC 429 Data

The FSCU supplies ARINC 429 bus communication via one ARINC 429 transmitter and has one ARINC 429 receiver to monitor the transmitted data. The FSCU receives data with a minimum of five ARINC 429 receivers and supports high speed ARINC 429 at 100 kbps.

CPU

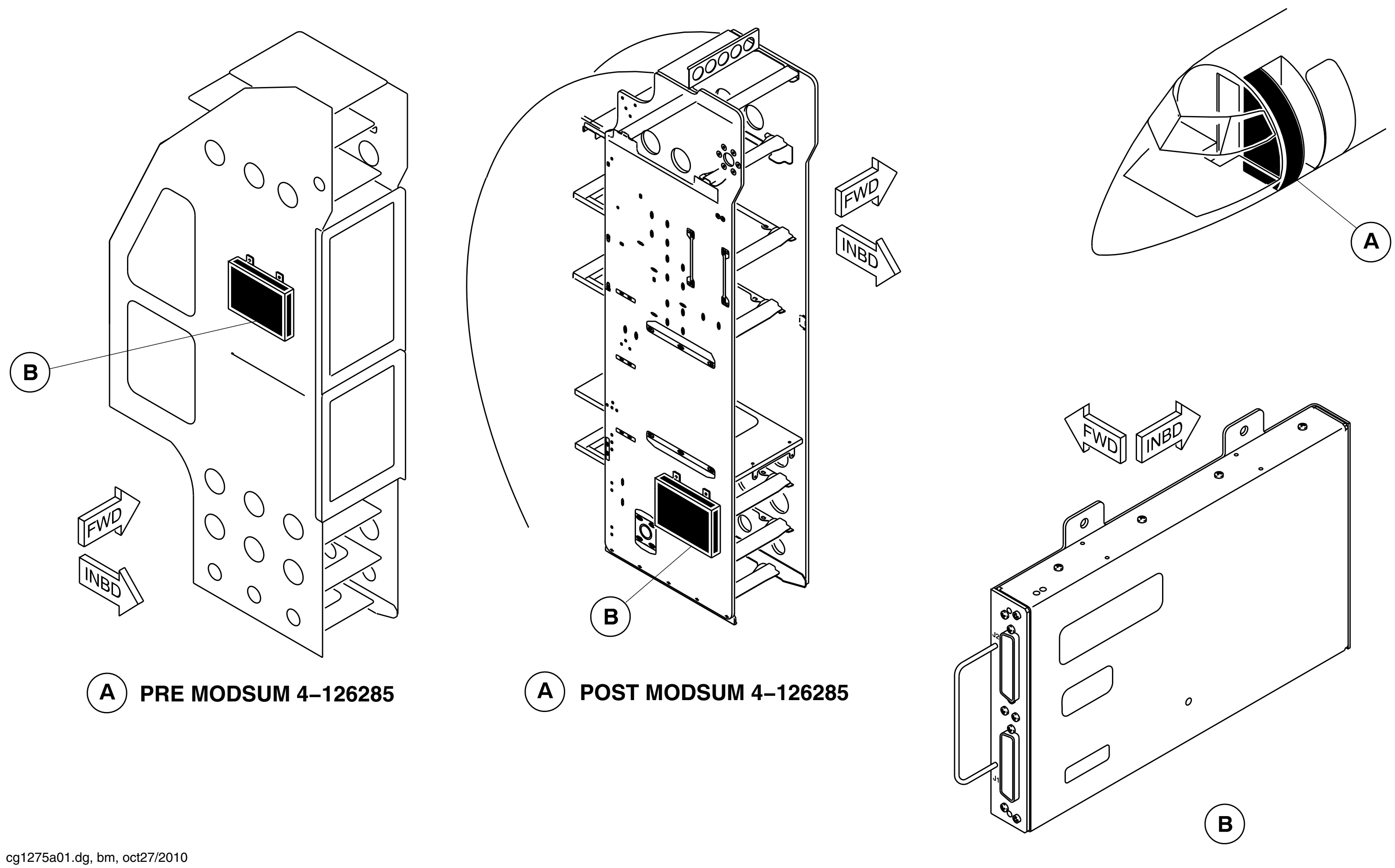
The controller consists of a 16 bit CPU with a minimum throughout of 8.4 MIPS (at 16.78 MHz) to implement the required control, BIT processing and allows 25% growth.

Memory

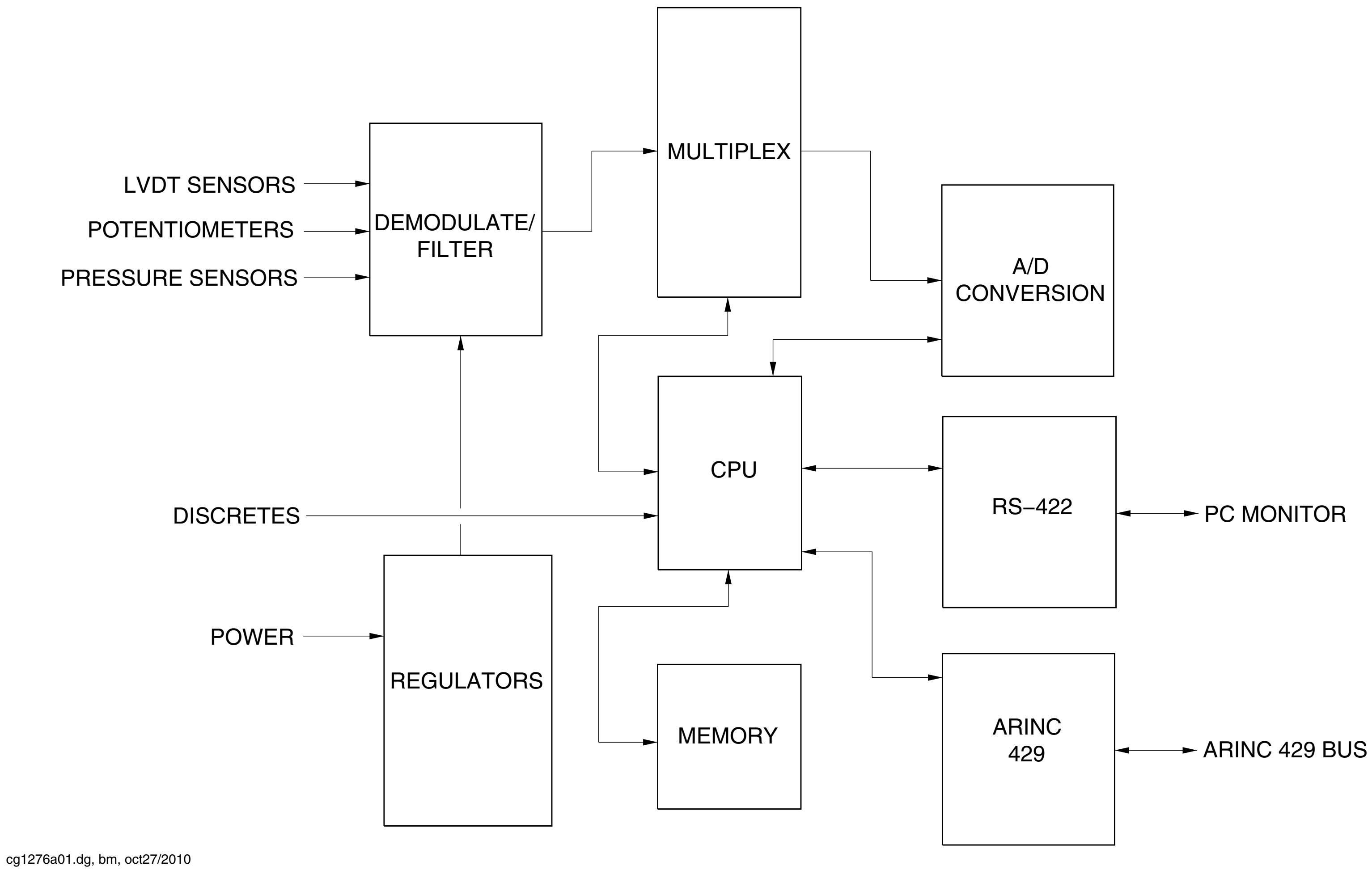
A minimum of 32 KB of non−volatile memory is supplied for storage of all detected faults by BIT, the downloader and allows 25% growth. A minimum of 256 KB Random Access Memory (RAM) is supplied to hold all control/BIT data and allows 25% growth. A minimum of 512 KB on−board flash memory is supplied for the storage of the control and BIT software and allows 25% growth.

RS−422 Bus

The FSCU supplies a bi−directional RS−422 interface which is used in ground operation to allow the downloader to receive the data used to program the flash memory.



Signal Conditioning Unit
Figure 11



Signal Conditioning Unit Block Diagram
Figure 12

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31–34–00–001

EXTENDED–STORAGE QUICK ACCESS RECORDER SYSTEM (EQAR)

Introduction

The extended–storage quick access recorder (EQAR) is an on–board optical recorder which is based on rewriteable disk technology. It is used as a digital aircraft condition monitoring system (ACMS) recorder (DAR) and records on–board flight and message data. This data is supplied by the aircraft flight data processing system (FDPS).

General Description

Refer to Figure 1.

On PRE MODSUM 4Q900161 aircraft, the EQAR receives aircraft data from the solid state flight data recorder (SSFDR) through input/output processor (IOP) #2 and run control input from input/output module (IOM) #2. It outputs EQAR status, disk low, and disk full data to IOM #2.

Refer to Figure 2.

On POST MODSUM 4Q900161 aircraft, the EQAR receives aircraft data from the SSFDR and run control input from IOM #1. It outputs EQAR status, disk low and disk full data to IOM #1.

Detailed Description

The EQAR interfaces with:

– Solid state flight data recorder (SSFDR) system

– Flight data processing system (FDPS)

The system is housed in a 1/2 ATR enclosure with an ARINC 404A rear connector and is mounted in the aircraft wardrobe shelf area. Data is input through the FDPS from the SSFDR. It is transmitted in a harvard biphase format that conforms to ARINC 717–7. The unit receives data from the aircraft SSFDR and duplicates its readings.

The recording medium is a standard commercial 3.5–inch polycarbonate or glass removable rewriteable optical disc. The disks supply a large storage capacity of 128 Mb, 230 Mb or 540 Mb per disk. On a 128 Mb disk, the recording capacity is 17 to 276 hours. For a 230 Mb disk, the recording capacity is 30 to 495 hours and on the 540 Mb disk, the capacity is 71 to 1139 hours.

The storage capacity is dependent on the ARINC 717 data rate and disk capacity. The EQAR receives and records the aircraft data at a programmable recording speed of 2 x 128 x 16–bit words per second through the ARINC 717 databus. The capacity is user–configureable through the use of an upload disk and reconfiguration can be done either directly on the aircraft or with the use of EQAR ground diagnostic equipment (GDE).

Operation

The EQAR optical disk recorder is based on a CPU board that ensures management and monitoring of the equipment. Communication between the aircraft systems and the CPU board is accomplished through the back panel connector board and the I/O interface board.

Data to be recorded is transmitted to the disk drive through the SCSI controller board. Data transfer between the I/O interface board, CPU board and SCSI controller board is done over a PC/AT internal bus.

Three LEDs located on the front panel board show the status of the unit. This status is also transmitted over a databus by I/O discrete signals to the IOM. The power supply unit delivers the internal voltages required for the equipment to function.

The battery board allows for the backup of the calendar clock and data buffer memory in case of power loss.

The Left Main Bus supplies the EQAR with 28 V dc electrical power through a 5 ampere circuit breaker. On PRE MOD 4Q900161 aircraft, the circuit breaker is located on the Left DC Circuit Breaker panel (Left Main Bus) at position E4 and on post modsum 4Q900161 aircraft, it is located on the Avionics Circuit Breaker panel (Left Main Bus) at position L1.

Extended–storage Quick Access Recorder

Refer to Figure 3.

The EQAR comprises the components that follow:

- Back panel connector board
- I/O Interface board
- CPU board
- Drive assembly
- Power supply
- Battery board
- Front panel board

The back panel connector board connects the EQAR with the aircraft systems through connector P1. It transmits operational inputs to the I/O interface board, test signals to the CPU board and the 28 V dc voltage to the power supply unit. It also transmits discrete outputs (status) from the I/O interface board to the aircraft systems. It contains EMI filters and diodes to protect the internal circuits against interference overvoltages.

The I/O interface board adapts, decodes and transmits the different signals. It consists mainly of a network programmed architecture. It transmits discrete outputs, such as status, disk low, or disk full, to the aircraft systems. It also controls the front panel LEDs with the test signals supplied by the CPU and interconnects the CPU board to the SCSI board. The I/O board also transmits to the CPU board the data that follows:

- Operational inputs (flight parameters, monitoring and control signals) from the back panel connector board
- Power supply voltages (5 V and 12 V)
- Status of the power supply voltages
- Status of the battery (from the battery board)
- Status of the disk drive sensors (accelerometer and temperature probes)

The CPU board is a PC type of CPU board which contains three 4 Mb SRAM components and one 256k flash EPROM memory component. The board connects to the I/O interface board through the PC/AT bus, the back panel connector board (for test signals through the serial port COM1, and for a serial RS422 serial link) and the battery board (battery connector).

The memories allow for storage of the operational software (flash EPROM) and the parameters being recorded (SRAM). An upload function allows for the direct loading of the operational software from an optical disk.

The internal calendar clock of the CPU board allows for creation of the volume name of the disk and the file name during the recording phase. The clock can be monitored and set through the RS232 link at port COM1.

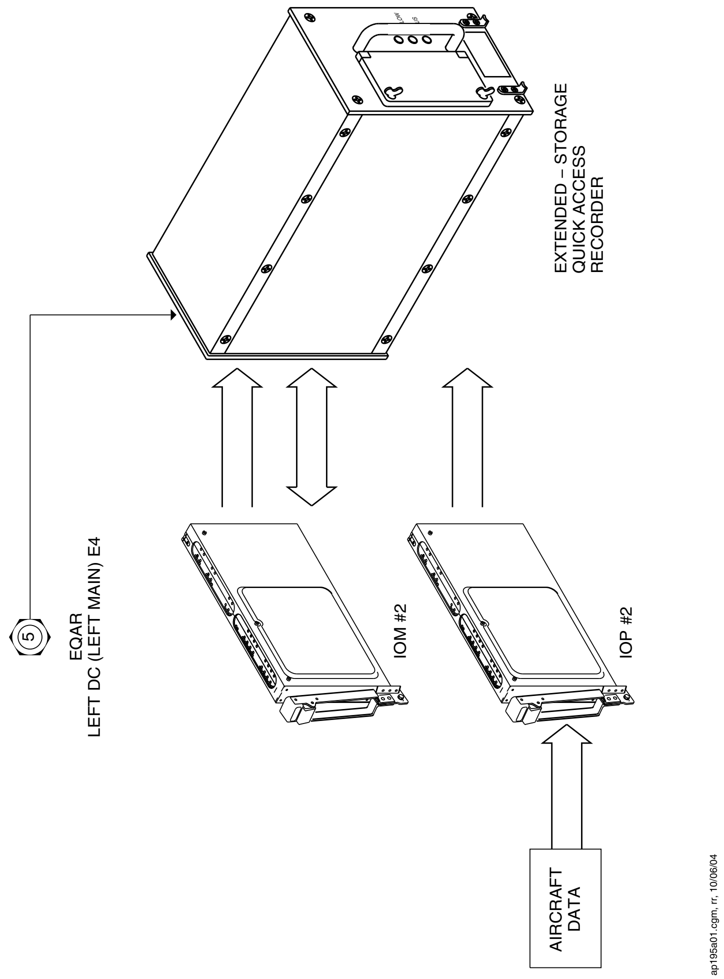
The SCSI controller board is used to control access to the optical disk.

The drive assembly includes the optical disk drive mounted on a support assembly fitted with shock absorbers, an accelerometer and temperature probes. It allows for the recording of the flight parameters on a 3 1/2 in. formatted optical disk, with a storage capacity of 128 Mb, 230 Mb or 540 Mb. The storage capacity depends on the ARINC 717 data rate and the disk capacity.

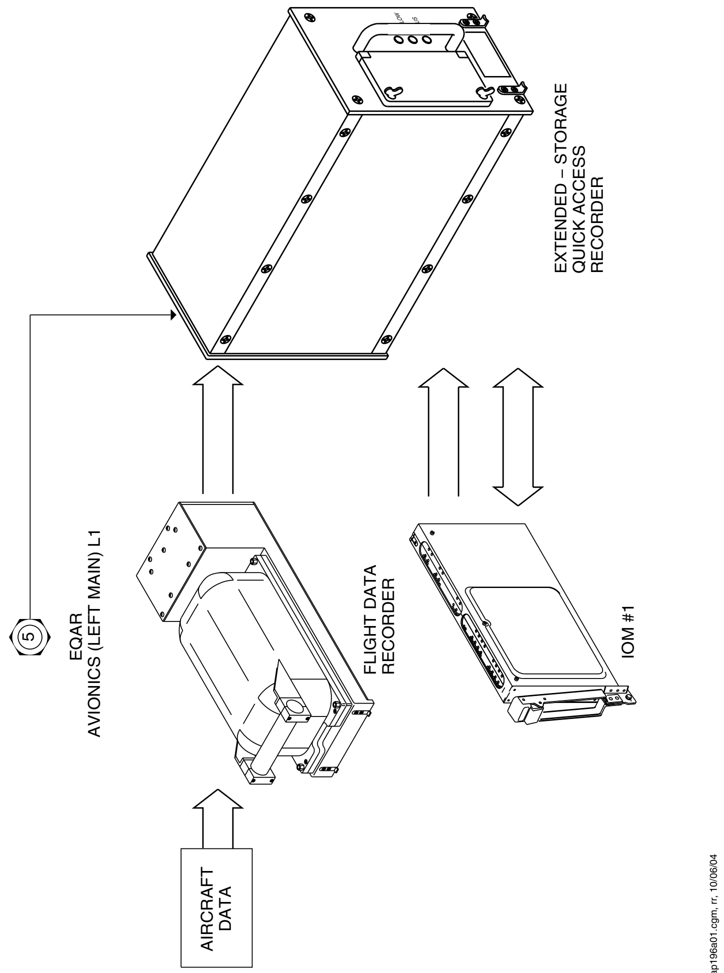
With the aircraft 28 V dc voltage, the power supply unit delivers 5 V and 12 V for the I/O interface board and for the disk drive. These voltages are sent through the I/O interface board to the CPU and SCSI boards.

The battery board contains a 3.6 V battery that allows for backup of the CPU board R/W memory. The output voltage is also sent, through the front panel board, to the I/O interface board. The I/O interface board monitors its level and transmits the results to the CPU board. In case of power cut, the battery board allows for backup of the calendar clock and any flight parameters that have been received but not yet recorded. These parameters are recorded as soon as the system is reenergized.

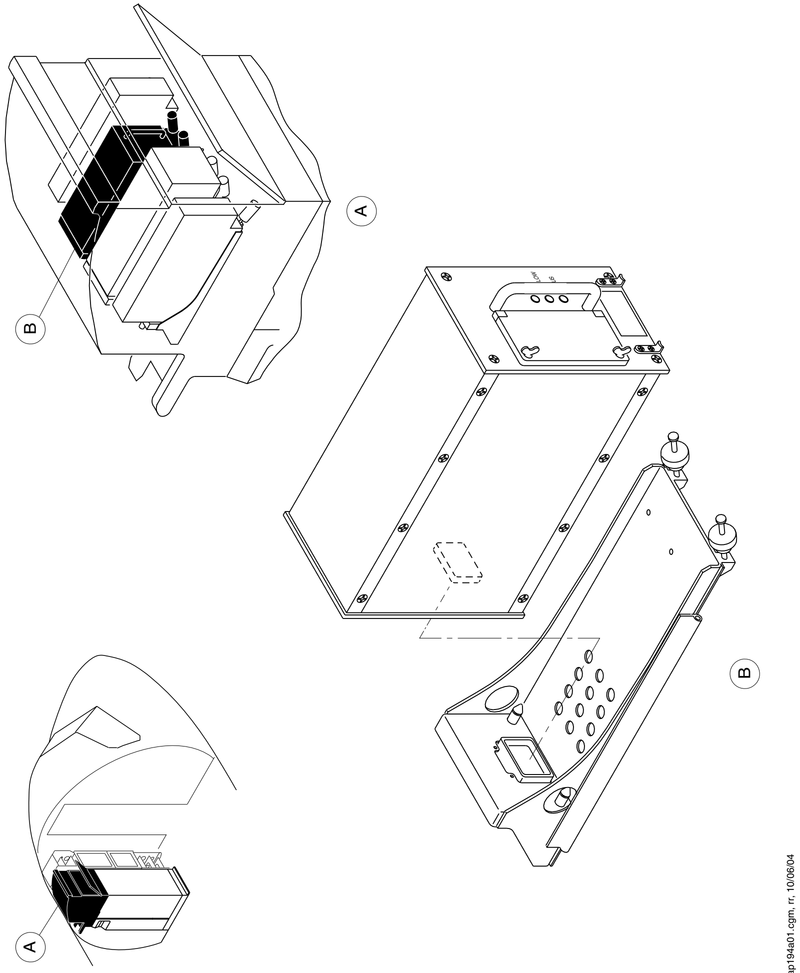
The front panel board contains the three LEDs that are controlled by the CPU board or the SCSI board, through the I/O interface board. The red LED DS1 comes on when the CPU board detects a fault while the power–on test is performed. The red LED DS2 comes on to indicate the optical disk is 70% full and the green LED DS3 comes on to indicate that the unit is powered on.



Extended – Storage Quick Access Recorder (Pre Modsum 4Q900161)
Figure 1



Extended – Storage Quick Access Recorder (Post Modsum 4Q900161)
Figure 2



Extended Quick Access Recorder (EQAR)
Figure 3

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31–36–00–001

MICRO QUICK ACCESS RECORDER SYSTEM (MQAR)

Introduction

The Micro Quick Access Recorder (MQAR) is an on–board flash card recorder which is based on compact flash card technology. It is used as a digital aircraft condition monitoring system (ACMS) recorder (DAR) and records on–board flight and message data. This data is supplied by the aircraft solid state flight data recorder (SSFDR). The MQAR is a “Wrap” type, in which the memory is overwritten, when the memory is full.

General Description

Refer to Figure 1.

The MQAR receives aircraft data from the SSFDR and records it on a compact flash card.

The MQAR data can be downloaded with the built–in mini Universal Serial Bus (USB) port interface to a portable computer. Alternately, the filled compact flash card can be removed from the MQAR and accessed through a standard PCMCIA interface reader on a portable computer.

Detailed Description

The MQAR interfaces with:

– Solid state flight data recorder (SSFDR) system

– Input/Output Processor (IOP) #1.

The system is mounted in the aircraft wardrobe shelf area. The system interfaces with the aircraft through the standard Boeing 15–Pin 28 VDC connector at the rear of the MQAR. Data is input from the SSFDR. It is transmitted in a Harvard biphase format that conforms to ARINC 717–10. The unit receives data from the aircraft SSFDR and duplicates its readings.

The recording medium is a standard commercial compact flash card. The cards supply a large storage capacity of 128 MB, 256 MB, 512 MB or 1 GB per card. On a 128 MB card, the recording capacity is 20 to 160 hours. On a 256 MB card, the recording capacity is 40 to 320 hours. For a 512 MB card, the recording capacity is 80 to 640 hours and on the 1 GB card, the capacity is 160 to 1250 hours.

The storage capacity is dependent on the ARINC 717 data rate and card capacity. The MQAR receives and records the aircraft data through the ARINC 717 databus.

The algorithm for the approximate record time in hours is:

– Record Time ≈ 160 * (Compact Flash Card Capacity in Mega–Bytes (MB) / Data Input Rate in Words–Per–Second (wps))

Operation

The MQAR operates on 28 VDC and draws less than 2 watts of power.

The MQAR contains an optically coupled data input circuit, which provides isolation and eliminates any interference to aircraft systems. This input interface also maintains data symmetry throughput to the MQAR and records ARINC 717 data at the rate of 64 wps, 128 wps, 256 wps or 512 wps.

Two LEDs located on the MQAR front panel show the status of the unit. The two LEDs illuminate momentarily upon power–up of the MQAR, to indicate the normal operation.

The battery board allows for the backup of the calendar clock and data buffer memory in case of power loss.

The Left Main Bus supplies the MQAR with 28 VDC electrical power through a 5 ampere circuit breaker. The circuit breaker is located on the Avionics Circuit Breaker panel (Left Main Bus) at position L1.

The mini USB port of the MQAR supplies 5 VDC operating power to the MQAR from a host computer when the aircraft power is not available.

Micro Quick Access Recorder

Refer to Figure 2.

The MQAR comprises the components that follow:

- Aircraft interface connector
- Optically coupled data input circuit
- Internal circuits
- Compact flash card interface
- Mini USB port
- Front panel LEDs.

The aircraft interface connector at the rear of the MQAR connects the MQAR with the aircraft systems through connector P1–A1. It transmits operational inputs and the 28 VDC power to the internal circuits of the MQAR. The operational inputs include the flight parameters and data from the SSFDR.

The optically coupled data input circuit provides isolation and eliminates any interference to the aircraft systems. This input interface also maintains data symmetry throughput to the MQAR and records ARINC 717 data at the rate of 64 wps, 128 wps, 256 wps or 512 wps.

The internal circuits of the MQAR provide the operational and control logics for the proper functionality of the MQAR. The memories on the internal circuits allow for storage of the operational software (flash EPROM). The calendar clock on the internal circuits allows for creation of the volume name of the compact flash card and the file name during the recording phase. The clock can be monitored and set through the MQAR utility software. In case of power failure, the internal circuits of the MQAR allow for backup of the calendar clock and any flight parameters that have been received but not yet recorded. These parameters are recorded as soon as the system is re–energized.

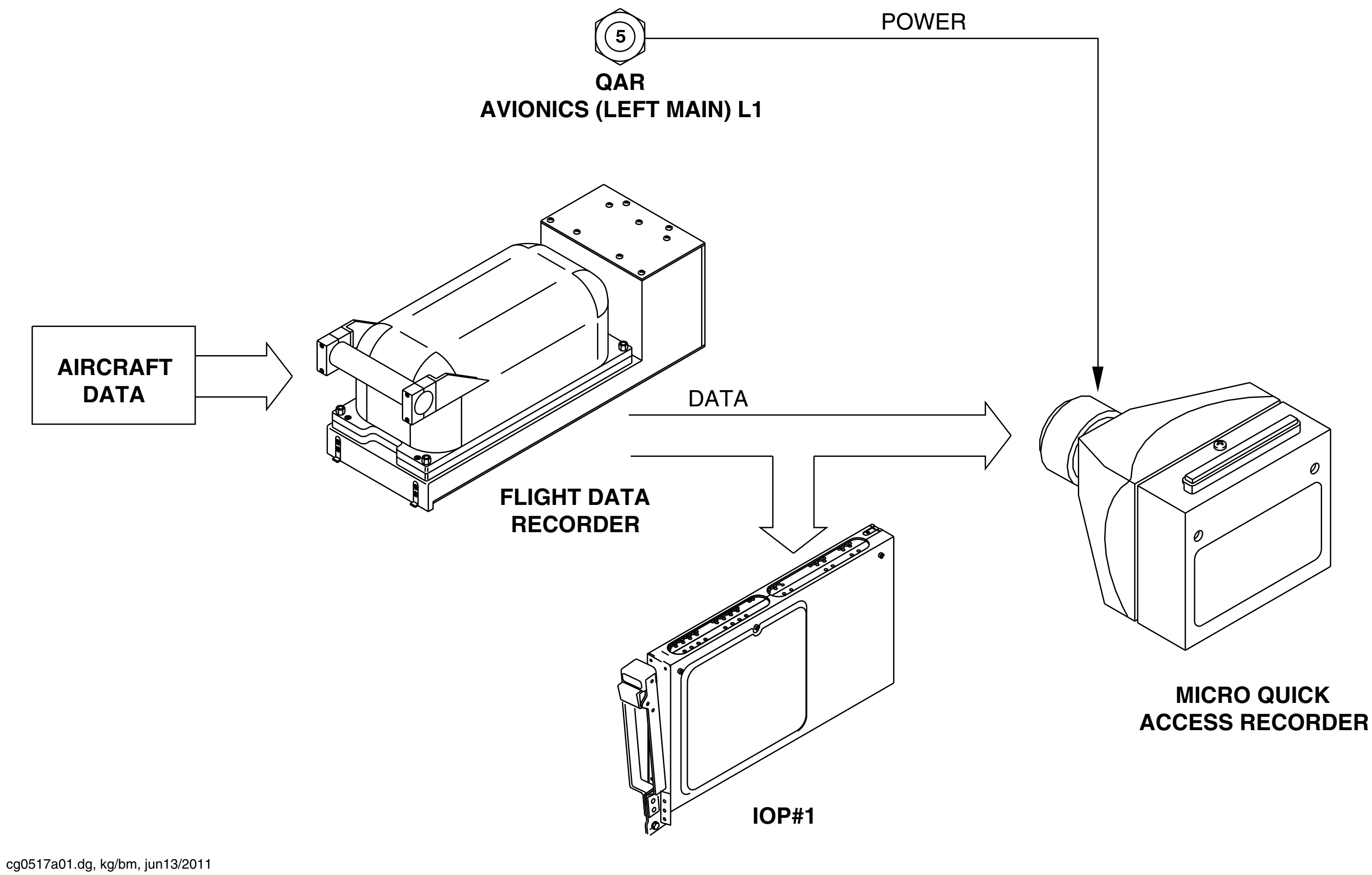
The compact flash card interface is a 50–pin connector that accepts Type 1 (thin) compact flash cards. The compact flash card acts as the media for recording of the flight parameters. The storage capacity of the compact flash card may be 128 MB, 256 MB, 512 MB or 1 GB. The storage capacity of the MQAR depends on the ARINC 717 data rate and the compact flash card capacity.

When connected to a host computer through an USB interface cable, the mini USB port on the MQAR acts as an easily accessible interface to:

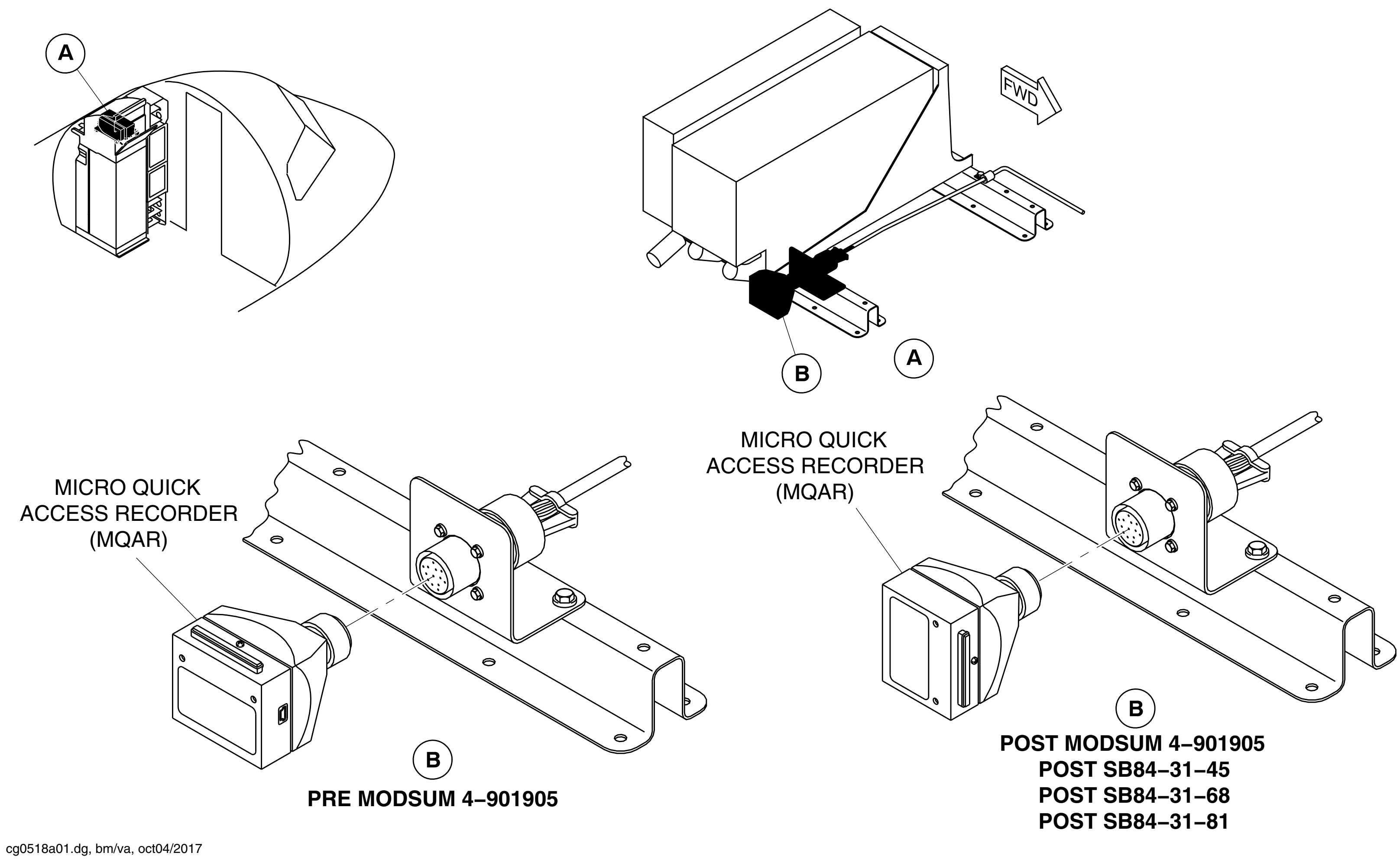
- Download the recorded data from the MQAR
- Monitor and set the MQAR internal clock's time and date
- Erase the flight data recorded in the MQAR.

The mini USB port on the MQAR also supplies 5 VDC power to the MQAR from the host computer, if the aircraft power is not available.

The front panel of the MQAR contains the two LEDs that are controlled by the internal circuits of the MQAR. Upon power up of the MQAR, the two LEDs illuminate two times, and go off to indicate normal operation. The amber FAULT LED comes on when the MQAR stops recording for any reason, or when the compact flash card is not installed in the MQAR. The illumination of the amber MEMORY FULL LED in the "No–Wrap" type MQAR indicates that the compact flash card is full. The amber MEMORY FULL LED does not illuminate in the "Wrap" type MQAR, in which the memory is automatically overwritten when it is full.



Micro Quick Access Recorder (MQAR)
Figure 1



Micro Quick Access Recorder (MQAR)
Figure 2

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31–40–00–001

CENTRAL COMPUTERS

Introduction

The central computer has a Flight Data Processing System (FDPS) to receive data from sensors and avionics systems and supply it to other systems. The Flight Data Processing System (FDPS) also causes different warning tones to sound if an important system has malfunctioned or if the aircraft is in a dangerous condition.

General Description

Refer to Figures 1 and 2.

The two Flight Data Processing System (FDPS1, FDPS2) do the functions that follow:

- Receives and calculates aircraft and avionics parameters for the Flight Data Recorder (FDR)
- Receives data from different avionics and aircraft systems and routes their parameters to a other systems through a single output
- Receives analogue, discrete, and digital data from other systems and changes it to ARINC 429 format
- Mismatch message calculations
- Monitors the Electronic Instrument System (EIS)
- Message generation
- Aircraft Configuration Management (ACM)
- Supplies Warning Tones (WTG 1, WTG 2) and manages its priority
- Maintenance tests
- Software teleloading.

Each Flight Data Processing System (FDPS 1, FDPS 2) has the modules that follow:

- Integrated Flight Cabinet (31–41–00)
- Input/Output Processor (31–41–06)
- Input/Output Module (31–41–11)
- Prime Power Supply Module (31–41–16)
- Aircraft Configuration Module (31–41–21).

The modules are located in two Integrated Flight Cabinets (IFC 1, IFC 2) installed in the Avionics rack.

Detailed Description

The Flight Data Processing System (FDPS 1, FDPS 2) functions in the modes that follow:

- Initialization (INIT)
- Power–On Self Test (POST)
- Operational (OPER)
- Maintenance
- Teleloading.

Initialization (INIT) State: The FDPS functions in the initialization mode after an electrical power interruption. It initializes the hardware and software when it is on the ground.

If the FDPS does not receive a maintenance or teleloading request, it will then function in the Power−On Self Test (POST) mode.

Power−On Self Test (POST): The FDPS does a (POST) to make sure the hardware operates correctly before starting the operational mode.

The FDPS does a POST when the aircraft is on the ground and there is a long electrical power interruption that continues for more than 200 milliseconds.

The FDPS tests the interfaces that follow:

- Input/Output Processor (IOP 1, IOP 2)
- Input/Output Module (IOM 1, IOM 2)
- Ground Proximity Warning System Converter.

The result of the POST is stored in the Built In Test Equipment (BITE) and sent to the Central Diagnostic System (CDS).

The POST checks the parameters that follows:

- Program memory
- Data memory
- Watch dog
- Power supplies Hardware (HW) monitoring
- Module Validity HW and Software (SW) logic
- Memory partitioning HW monitoring
- Time partitioning HW monitoring.

If a parameter malfunctions, it will causes the FDPS to malfunction.

The POST also checks the parameters that follow:

- ARINC 429 inputs and outputs
- ARINC 422 inputs and outputs
- ARINC 717 inputs and outputs
- Discrete inputs and outputs
- IO1, IO2 board validity
- Analogue inputs and outputs
- Discrete inputs and outputs.

They do not cause a FDPS malfunction.

The POST sequence continues for 25 seconds.

Operational State: The FDPS does the functions that follow:

- Flight Data Concentrator (FDC)
- Data Hub Concentrator (DHC)
- Data Control (DCO)
- Mismatch calculations
- Electronic Instrument System (EIS) essential monitoring
- Advisory message generation
- Aircraft configuration management
- Warning Tone Generator (WTG 1, WTG 2).

Flight Data Concentrator (FDC): The FDC is located in IOP1. It receives data in different formats from the avionics and other aircraft systems and supplies digital data to the Solid State Flight Data Recorder (SSFDR) through an ARINC 573/717 bus.

Data Hub Concentrator (DHC): The FDPS receives noncritical data from avionics and other aircraft systems and supplies their parameters to other systems through a single output.

The FDPS supplies the concentrated parameters through ARINC 429 data buses to the avionics systems that follow:

- Electronic Instruments System (EIS)
- Flight Guidance Module (FGM 1, FGM 2)
- Stall Protection Module (SPM 1, SPM 2)
- Audio and Radio Control Display Units (ARCDU 1, ARCDU 2)
- Traffic Collision Avoidance System (TCAS)
- Ground Proximity Warning System (GPWS)
- Weather Radar (WXR)
- Flight Management System (FMS 1, FMS 2).

Two specific external non–avionics General Purpose Data Buses (GPDB 1, GPDB 2) are used to transmit data to systems that are not part of the avionics suite.

Data Control (DCO): The Data Control (DCO) makes calculations for the parameters before concentration.

Refer to Figure 4.
Mismatch Calculations: The FDPS monitors different parameters. If an IOP senses a difference between its related value and the same the parameters it received from the other IOP, a mismatch message is shown by the Flight Mode Annunciators (FMA) in the opposite Primary Flight Displays (PFD).

The Flight Data Processing System (FDPS) Mismatch Messages that follow:

- PITCH MISMATCH
- HEADING MISMATCH
- IAS MISMATCH
- ALT MISMATCH
- RAD ALT MISMATCH
- GS MISMATCH
- LOC MISMATCH.

The calculations are done only when the parameters are valid. For localizer and glideslope mismatch messages, the two navigation receivers outputs must be valid and tuned to the same frequency.

Electronic Instrument System (EIS) Essential monitoring: The FDPS does essential monitoring for the systems that follow:

- Radio Altimeter
- Localizer deviation
- Glideslope deviation.

The IOPs monitors the difference between a parameter received directly and same parameter received from the opposite PFD and IOP. When the difference is more than the predetermined value, the FDPS causes to the PFD to show a malfunction condition for that parameter.

Refer to Figure 3.
The essential monitoring parameter thresholds are shown in the table that follows:

Parameter	Difference
AGL altitude, less than 1000 ft	20 ft
LOC deviation, between 50 and 1000 ft AGL	2/3 of Expanded LOC Scale
G/S deviation, between 50 and 1000 ft AGL	1/6 of G/S Scale

Refer to Figure 4.
Message Generation: The Message Generation System has the modes that follow:

- Advisory
- Flight Mode Annunciator (FMA).

Advisory: The advisory messages are shown in white letters on the Engine Display (ED). Each message has a location and is shown while the condition is present. There are two kinds of advisory messages as follows:

- First Family
- Second Family.

First Family: The first family messages relate to an applicable aircraft status, where crew awareness is required and an action may be necessary. These messages are located near its related indication.

Second Family: Second family messages relate to minor malfunctions and are located at the bottom of the ED. The messages relate to passive malfunctions do not affect a continued safe flight.

The FDPS causes the ED to show the advisory messages that follow:

- [BALANCE]
- INCR REF SPEEDS
- ICE DETECTED
- IFC messages
- Display messages.

NOTE

Note: The advisory messages are shown in order of decreasing importance. The most important IFC message is shown at the bottom left part of the ED and the most important display message is shown at the bottom right part of the ED.

[BALANCE] Message: The BALANCE message comes into view flashing for five seconds then stays on steady when a fuel imbalance is sensed by the left or right fuel gauging computers.

INCR REF SPEEDS Message: The INCR REF SPEEDS message comes into view when the Stall Protection System (SPS) stall sensing is changed for icing conditions. The FDPS sends a discrete data signal to Electronic Instruments System (EIS) from one or the other SPM.

ICE DETECTED Message: The ICE DETECTED message comes into view when one ice detector probe or the other senses ice accumulation.

IFC Messages: The most important IFC message is shown at the bottom left part of the ED. It shows the messages that follow:

- IOP1 FAIL, IOP2 FAIL, IOPS FAIL
- IOP BAD CONF
- IOM1 FAIL, IOM2 FAIL, IOMS FAIL
- WTG1 FAIL, WTG2 FAIL, WTGS FAIL
- GPWSC I/F FAIL
- WOW/IOP1 FAIL, WOW/IOP2 FAIL, WOW/IOPS FAIL
- IFC ACM1 FAIL, IFC ACM2 FAIL, IFC ACMS FAIL
- RA1 FAIL, RA2 FAIL, RAS FAIL.

IOP1 FAIL, IOP2 FAIL, IOPS FAIL: The IOP FAIL messages are shown when the IOP1 or IOP2 or the interface to the ED has malfunctioned.

IOP BAD CONF: The IOP BAD CONF message is shown when a bad aircraft configuration condition is sensed by one IOP or the other. The message comes into view when the aircraft is on the ground.

IOM1 FAIL, IOM2 FAIL, IOMS FAIL: The IOM FAIL messages are shown when IOM malfunctions are sensed.

WTG1 FAIL, WTG2 FAIL, WTGS FAIL: The WTG FAIL messages are shown when the WTG malfunctions. The message is shown when the aircraft is on the ground and engines are not running.

GPWSC I/F FAIL: The GPWSC I/F FAIL messages are shown when the IFC1 makes the GPWS inoperative when it malfunctions. The message is when the malfunction is sensed.

WOW/IOP1 FAIL, WOW/IOP2 FAIL, WOW/IOPS FAIL: The WOW/IOP FAIL messages are shown when the IOP senses a difference between the main and nose WOW signals from the PSEU. The IOP will not be able to do a POST after a power interruption. The message is shown when the aircraft is on the ground and engines are not running.

IFC ACM1 FAIL, IFC ACM2 FAIL, IFC ACMS FAIL: The IFC ACM FAIL messages are shown when a ACM malfunction is detected. A single ACM malfunction has no effect on the aircraft. When the two ACM malfunction, a bad DU BAD CONF messages is shown.

RA1 FAIL, RA2 FAIL, RAS FAIL: The RA FAIL message is shown when a dual Radar Altimeter system is installed and the RA malfunction is detected for more that ten seconds by the EIS ED. The message is displayed any time the malfunction is detected.

Display Messages: The most important display message is shown at the bottom right part of the ED. The FANS FAIL message is the only FDPS related display message indication. It comes into view when 2 or more avionics cooling fans do not operate. The FANS FAIL display message also comes into view when 2 or more avionics cooling fans do not operate and they are not inhibited by their related thermal switch while the aircraft is on the ground. The temperature switch 1 supplies data to IOP1 and temperature switch 2 supplies data to IOP2.

An IFC or display advisory message also causes the AVIONICS caution light to come on 2 minutes after the aircraft is on the ground and the air speed is less than 50 kts.

Refer to Figure 5.
Aircraft Configuration Management: The FDPS uses data that is stored in the Aircraft Configuration Modules (ACM 1, ACM 2). The

ACMs are attached to the back–panel of its related IFC1. Each IOP supplies the configuration data to the systems that follow:

- Electronic Instrument System (EIS)
- Stall Protection System (SPS), Stall Protection Modules (SPM 1, SPM 2)
- Auto Flight Control System (AFCS), Flight Guidance Modules (FGM 1, FGM 2)
- Audio and Radio Control Display Units (ARCDU 1, ARCDU 2)

Each system monitors the configuration data. If a malfunction is sensed, it is stored in the Built In Test Equipment (BITE) and sent to the Central Diagnostics System (CDS) and the ED shows an IFC message.

The Aircraft Configuration Modules (ACM 1, ACM 2) are programmed using the Portable Multipurpose Access Terminal (PMAT).

Refer to Figure 6.
Warning Tone Generators (WTG 1, WTG 2): The Warning Tone Generators (WTG 1, WTG 2) give different tones to tell the pilots of dangerous conditions or system malfunctions.

The tone that sounds depends on its priority. The most important tone always sound as follows:

PRIORITY	TONE	DESCRIPTION	SIGNAL TYPE	LEVEL, VOLTS
1	GPWS	Voice	N/A	N/A
2	TCAS, RA	Voice	N/A	N/A
3	Fire	Continuous chime until manually cancelled	A	4.8
4	Incorrect take off configuration	1000 Hz intermittent tone until cause of condition is removed	C	3.5
5	Autopilot disengage	250 Hz Intermittent tone for 1.5 seconds (manual) or until manually cancelled (automatic)	F	3.8
6	Pitch trim in motion	Continuous clicking until cause of condition is removed	D	4.2
7	Overspeed	1000 Hz intermittent tone until speed decreased	C	3.1

PRIORITY	TONE	DESCRIPTION	SIGNAL TYPE	LEVEL, VOLTS
8	Incorrect landing gear configuration	Continuous 800 Hz tone until cause of condition is removed	B	1.8
9	Altitude alert	2900 Hz tone for 1 seconds	B	2.9
10	Beta lockout warning	Continuous 1900 Hz to 2900 tone until cause of condition is removed	G	5.2
11	Warning annunciation	3 1000 Hz chimes	E	3.8
12	Caution annunciation	1 1000 Hz chime	E	5.0
13	TCAS, TA	Voice	N/A	N/A
14	SELCAL	1200 Hz tone for 3 seconds	N/A	2.7

NOTE

Note: The WTGs generate the aural tones and control the priority of the voice sound from the GPWS and TCAS.

Each type of signal modulation, is shown as follows:

Refer to Figure 7.
Aural warning type A

Refer to Figure 8.
Aural warning type B

Refer to Figure 9.
Aural warning type C

Refer to Figure 10.
Aural warning type D

Refer to Figure 11.
Aural warning type E

Refer to Figure 12.
Aural warning type F

Refer to Figure 13.
Aural warning type G

Refer to Figure 14.
Aural warning type H.

The WTG is not an independent system. It is part of the Input/Output Modules' (IOM1, IOM2) function. One WTG supplies a warning tone to the Remote Control Audio Unit (RCAU) and other monitors it. The RCAU amplifies the signal and sends the tone to the flight compartment speakers and the pilots headsets.

The WTG functions with a visual indication. It gives the different tones that follow:

- GPWS
- TCAS
- Engine fire
- Incorrect takeoff warning
- Autopilot disengagement

- Pitch trim in motion
- Overspeed warning
- Incorrect landing gear configuration
- Altitude alert
- Beta lockout warning
- Master warning tone
- Master caution tone
- SELCAL.

GPWS: The GPWS is a system that makes its own synthesized voice sounds and connects directly to the Audio Integrating System (AIS).

When the WTG senses a GPWS audio on condition, it inhibits the other tones.

The voice sounds from the GPWS is inhibited if a GPWS malfunction is sensed while the aircraft is airborne. The GPWS audio on signal is monitored to prevent an inhibit of a different WTG tone caused by a partial GPWS malfunction. If the GPWS audio on signal stays on for more than 60 seconds, the GPWS's priority status is ignored. A GPWS and a TCAS or WTG tone is allowed to be heard at the same time.

The GPWS malfunction condition and that the TCAS and WTG continues to function is easily identified by the pilots. To be able to sense this malfunction condition, the FDPS does not inhibit the GPWS during the WTG test (ADC1 or ADC2 TEST toggle switch selection).

TCAS: The TCAS is a system that makes its own synthesized voice sounds and connects directly to the Audio Integrating System (AIS). It has two types of alerts with two different priority levels that follows:

- Resolution advisories
- Traffic advisories.

Resolution Advisories: The resolution advisories are grouped as corrective advisories or preventive advisories. The vertical flight path of the aircraft must be changed when a corrective advisory condition is sensed. A preventive advisory is sensed to maintain the vertical flight path.

Traffic Advisories: The traffic advisories mode senses traffic advisory intruders only.

The WTG inhibits the TCAS voice sound when a more important tone sounds.

When the WTG senses a TCAS resolution advisory (corrective or preventive) or a traffic advisory signal, it inhibits less important tones.

The voice sounds from the TCAS is inhibited if a TCAS malfunction is sensed while the aircraft is airborne. The TCAS audio resolution advisory (corrective or preventive) and traffic advisory signals are monitored to prevent an inhibit of a less important WTG tone caused by a partial TCAS malfunction. If a TCAS signal stays on for more than 60 seconds, the TCAS's priority status is ignored. A TCAS and a GPWS or WTG tone is allowed to be heard at the same time.

The TCAS malfunction condition and that the GPWS and WTG continues to function is easily identified by the pilots. To be able to sense this malfunction condition, the FDPS does not inhibit the TCAS during the WTG test (ADC1 or ADC2 TEST toggle switch selection).

Engine Fire: The WTG inhibits the engine fire aural warnings when a more important tone sounds.

A continuous chime sounds when the WTG senses a fire bell discrete signal from the fire detection system.

Refer to Figure 15.
Incorrect Take–Off Warning: The WTG inhibits the incorrect take–off warning indication when more a more important tone sounds.

The WTG sounds an 1000 Hz Intermittent tone when the aircraft is not set to a safe for take–off configuration. It functions when the two engine torque indications are more than 50% or the TOCW maintenance switch is set to TPCW and the one or more of parameters is sensed as follows:

- Takeoff warning test
- ENG 1 torque more than 50%
- ENG 2 torque more than 50%
- Flaps set to more than 20
- CLA 1 not set to 1020
- CLA 2 not set to 1020
- Parking brake set
- Inboard spoilers extended
- Outboard spoiler extended
- Left elevator trim out of range
- Right elevator trim out of range.

The take off warning system has the interfaces that follow:

Parameter	Source
T/O WARN MAINT	T/O WARN TEST Toggle switch
ENG1 TORQUE > 50%	FADEC
ENG2 TORQUE > 50%	FADEC
NGWOFW	PESU
FLAPS ≤ 13.5°	FPIU
CLA1 ≠ MAX/1020	FADEC
CLA2 ≠ MAX/1020	FADEC
PARKING BRAKE ON	PARKING BRAKE LEVER
INBOARD SPOILERS EXTENDED	FCS ECU LEFT CHANNEL
LEFT ELEVATOR TRIM OUT OF RANGE	FCS ECU LEFT CHANNEL
OUTBOARD SPOILERS EXTENDED	FCS ECU RIGHT CHANNEL
RIGHT ELEVATOR TRIM OUT OF RANGE	FCS ECU RIGHT CHANNEL

NOTE

Note: A malfunctioned ARINC parameter causes it to set.

Autopilot Disengagement: The WTG inhibits the autopilot disengagement tones when a more important tone sounds.

The WTG sounds an intermittent tone for 1.5 seconds for a manual AFCS disengagement. It sounds a 4000 Hz Intermittent tone when the Auto Flight Control System (AFCS) automatically disconnects until it is cancelled.

The two Flight Guidance Modules (FGM1, FGM2) supply Autopilot (AP) engagement/disengagement data through data buses to the WTGs.

The autopilot disengagement tones sound if one FGM or the other supplies a disengagement signal to the WTG.

Pitch Trim in Motion: The WTG inhibits the pitch trim in motion tones when a more important tone sounds.

The WTG sounds a clicking tone when the pitch trim is in motion for more than 3 seconds.

The two Flight Control Electronic Control Units (FC ECU1, FC ECU2) supply pitch trim in motion data through data buses to the WTGs.

The clicking tone sounds if one FCS ECU or the other supplies a pitch trim in motion signal to the WTG.

Overspeed Warning: The WTG inhibits the overspeed warning tones when a more important tone sounds.

The WTG sounds an intermittent 1000 Hz tone when the aircraft's speed is more than Maximum Velocity in Operation (VMO).

The two Air Data Units (ADU1, ADU2) supply overspeed data through data buses to the WTGs.

The overspeed tone sounds if one ADU or the other supplies an overspeed signal to the WTG.

Refer to Figure 16.

Incorrect Landing Gear Configuration: The WTG inhibits the incorrect landing gear configuration tones when a more important tone sounds.

The WTG sounds a continuous 800 Hz tone when the landing gear is not set to a safe for landing configuration.

It functions when the HORN toggle switch on the landing gear panel is set to the TEST position or if the gears are not down and locked one of the three different conditions are sensed as follows:

Condition 1 (Flaps and Engine torque):

- The flap lever is set to more than 8.5 degree

  AND

- ENG1 torque OR ENG2 torque is less than 50%

  AND

- PLA 1 AND PLA 2 are not in the rating power detent.

Condition 2 (Airspeed, Radio Altitude and PLA setting):

- Indicated airspeed less than 156 Knots (288 km/h), AND RAD ALT 1 invalid

  OR

- Indicated airspeed less than 156 Knots (288 km/h), AND RAD ALT 1 valid, AND Above Ground Level (AGL) altitude is less than 1053 feet

  AND

- PLA1 AND PLA2 are less than FLIGHT IDLE plus 12 degrees.

Condition 3 (Airspeed, single engine operation and PLA setting):

– Indicated airspeed less than 156 Knots (288 km/h), AND RAD ALT 1 invalid

OR

– Indicated airspeed less than 156 Knots (288 km/h), AND RAD ALT 1 valid, AND Above Ground Level (AGL) altitude is less than 1053 feet

AND

– PLA1 OR PLA2 is less than FLIGHT IDLE plus 12 degrees

AND

– HORN MUTE switch is not latched to MUTE

AND

– PLA 1 AND PLA 2 are less than the rating power detent.

The landing gear configuration warning system has interfaces with other aircraft systems as follows:

Signal	Source
GEARS DOWN AND LOCKED 1	PSEU
GEARS DOWN AND LOCKED 2	PSEU
IAS < 156 ±5 Knots	ADU1, ADU2
AG L< 1053 feet	RA1
PLA 1 < FLIGHT IDLE + 12 degree	FADEC
PLA 2< FLIGHT IDLE + 12 degree	FADEC
HORN MUTE	LANDING GEAR CONTROL PANEL
FLAP LEVER > 8.5°	FCU
ENG 1 TORQUE < 50%	FADEC
ENG 2 TORQUE < 50%	FADEC
PLA1 ≥ RATING POWER DETENT	FADEC
PLA2 ≥ RATING POWER DETENT	FADEC
LDG WARN TEST	LANDING GEAR CONTROL PANEL

Refer to Figure 17.
Altitude Alert: The WTG inhibits the altitude alert tones when a more important tone sounds.

The WTG sounds a continuous 2900 Hz tone for 1 second when the aircraft enters an area that is less than 1000 ft (305 m) above or below a set altitude.

Refer to Figure 18.
The ALT (altitude) selector rotary knob on the Flight Guidance Control Panel is used to set a cyan preselected altitude readout and bug indication on the PFDs.

Refer to Figure 19.
The range of the preselected altitude readout is between 0 and 99,900 ft. When the preselected altitude readout is set to less than zero feet, the indication goes out of view and if the value is set to more than the shown altitude scale, one half of the indication is shown at the edge of the altitude indicator.

When the aircraft is 1000 ft (305m) above or below the preselected altitude, a tone sounds for 1 second and the indication changes to yellow until the aircraft is less than 250 ft from the preselected altitude. If the aircraft deviates from the preselected altitude by more than 250 ft, a tone sounds for 1 second and the indication changes to yellow again. The indication changes back when the aircraft is less than 250 ft above or below the preselected altitude value.

If the aircraft does not return to the preselected altitude, a new value must be set to reset the indication.

The Flight Data Processing System (FDPS) senses the barometric corrected altitude from ADU1 or ADU2 and the set altitude from the FGCP. It uses ADU1 if the HSI SEL pushbutton switch on the FGCP is set to left and ADU2 when set to right.

Refer to Figure 20.
Beta lockout warning: The WTG inhibits the beta lockout alerts when a more important tone sounds.

The Warning Tone Generator (WTG 1, WTG 2) sounds a continuous 800 Hz tone when the Power Lever Angle (PLA) is set below the IDLE position while in flight. To give a beta lockout indication, the Flight Data Processing System (FDPS) senses the parameters that follow :

- Beta lockout switch position
- Power lever angle
- Main landing gear WOW.

Master Warning: The WTG inhibits the master warning tones when a more important tone sounds. The WTG sounds three chimes when one or the other red master warning light comes on.

Master Caution: The WTG inhibits the master caution tones when a more important tone sounds. The WTG sounds a single chime when one or the other amber master caution light comes on.

SELCAL: The WTG inhibits the SELCAL tone when a more important tone sounds. A continuous 1200 Hz tone sound for three seconds when the Selective Calling (SELCAL) system senses an incoming call.

There Warning Tone Generators (WTG 1, WTG 2). The WTG1 sounds the applicable tone when necessary while WTG2 functions in the standby mode. The WTG2 functions only when it senses that WTG1 has malfunctioned.

The WTG receives inputs from other systems to make it operate.

Each WTG monitors its output. If the WTGs calculate different tones, the system will use the calculations from WTG 1. Its calculations are more important than WTG 2.

The WTGs does the monitoring that follows:

- Software
- Hardware.

Software Monitor: Each Warning Tone Generator (WTG 1, WTG 2) uses their related inputs to calculate the tone logic. The tone calculations are compared and the WTG 1 supplies the applicable tone.

Hardware Monitor: Each Input/Output Module (IOM 1, IOM 2) has two Input/Output boards (IO1, IO2). The WTG 1 sends an analogue tone to the Remote Control Audio Unit (RCAU) through IO2. It routes the same signal to IO1 to make sure that the output tone is correct. If there is a difference, the WTG 1 stops functioning and makes itself invalid. Then, the WTG 2 functions when required.

The Warning Tone Generator (WTG 1, WTG 2) will remain off until the next power interruption when it is latched off because of a malfunction. The Power−On Self Test (POST) will determine its validity.

A Warning Tone Generator (WTG 1, WTG 2) malfunction is stored in the Built In Test Equipment (BITE) and sent to the Central Diagnostic System.

The Engine Display (ED) advisory message location will show an WTG FAIL message when a WTG malfunctions.

FDPS Abnormal Modes: Most aircraft and avionics system supplies data to other systems through the two FDPS's. Each aircraft or avionics system uses data from its related FDPS. If its related FDPS malfunctions, the systems will automatically receive data from the other FDPS.

The Flight Data Recorder (FDR) receives data through the Input/Output Processor (IOP 1) from the two FDPS's.

If FDPS1 malfunctions, the Flight Data Recorder (FDR) function also malfunctions. If FDPS 2 malfunctions, the FDR will not record its parameters.

The data is supplied only to the FDPS1 from the systems that follow:

- Hydraulic quantity 1
- Hydraulic quantity 3
- Fuel inlet temperature 1
- Parking brake pressure
- Main Oil Pressure 1
- Ground Proximity Warning System Converter.

If FDPS1 malfunctions, its parameters are shown as dashes on the Engine Display (ED).

The data is supplied only to the FDPS2 from the systems that follow:

- Hydraulic quantity 2
- Fuel inlet temperature 2
- Traffic Collision Avoidance System (TCAS).

If FDPS2 malfunctions, its parameters are shown as dashes on the Engine Display (ED).

If one FDPS or the other malfunctions, the Engine Display (ED) will also show an IFC message in the advisory message area.

Maintenance Mode: The Built In Test Equipment (BITE) uses the Central Diagnosis System (CDS) to give the condition of the component. It stores faults in a Non Volatile Memory (NVM) for reporting to line and shop maintenance.

The Built In Test Equipment (BITE) allows aircraft maintenance personnel to:

- Do fault isolation and return to service testing after completing maintenance actions
- Access failure reports from last or previous flight legs
- Get the avionics status report
- Get the part number of a given part

The Built In Test Equipment (BITE) modes monitors the condition of the component as follows:

- Power−On Self Test (POST)
- Continuous Monitoring.

Power−On Self Test (POST): The Power−On Self Test (POST) checks the condition of the component at Power−Up or after a long power interruption.

Continuous Monitoring: The Continuous Monitoring checks the status of the component in flight. It records faults in a Non Volatile Memory (NVM) for later troubleshooting using the Central Diagnosis System (CDS).

A long power interruption causes the FDPS to start a Power−On Self Test (POST) again.

Teleloading: The CDS is a communication connection between the FDPS and a Portable Multipurpose Access Terminal (PMAT). The PMAT connects to a Personal Computer (PC) to download a new software version when a software upgrade is necessary.

The CDS functions in the teleloading mode when the conditions that follow is correct:

- The Calibrated Air Speed (CAS) is less than 50 kts for more than 10 seconds
- The aircraft is on the ground
- The CDS GND MAINT toggle switch on maintenance panel is set
- The MAINT Key on the ARCDU is pushed.

The PMAT is connected through an RS422 data bus to each FDPS when software is downloaded. The CDS also transmits through an ARINC 429 data bus to the PMAT all "acknowledge" words from each FDPS during the software downloading function.

The FDPS does a software configuration check before teleloading mode starts. It makes sure that the software version to be teleloaded is consistent with the equipment part number. After the teleload, the POST mode is started within 10 seconds.

The left essential bus supplies electrical power through a 10 A circuit breaker and the Prime Power Supply Module (PPSM1) to the Input/Output Processor (IOP1) and Input/Output Module (IOM1). The circuit breaker is located in position F7 on the avionics circuit breaker panel. The left main bus supplies electrical power through a 7.5 A circuit breaker and the PPSM1 to the Stall Protection Module (SPM1). The circuit breaker is located in position F2 on the avionics circuit breaker panel.

The right main bus supplies electrical power through a 10 A circuit breaker and the Prime Power Supply Module (PPSM 2) to the Input/Output Processor (IOP2) and Input/Output Module (IOM2). The circuit breaker is located in position F6 on the avionics circuit breaker panel. The right main bus supplies electrical power through a 7.5 A circuit breaker and the PPSM2 to the SPM2. The circuit breaker is located in position F5 on the avionics circuit breaker panel.

Refer to Figure 21.
The FDPS receives data from the DME, ADF and MLS (option) systems and supplies it to the ARCDUs. It supplies the data as shown in the table that follows:

Parameter	Source
DME frequency	DME1, DME2
ADF frequencies	ADF1, ADF2
MLS channel	MLS1, MLS2
MLS selected azimuth angle	MLS1, MLS2
MLS maximum selected glidepath angle	MLS1, MLS2

Refer to Figure 22.
The FDPS receives data from different non important systems and supplies it to the EIS and the opposite IOP. It supplies the data as shown in the table that follows:

Parameter	Source
Selected Heading	FGCP
Selected Course	FGCP
Navigation source	FGCP
Selected DH	ICP1, ICP2
Speed Bugs	ICP1, ICP2
Rudder position	FCSECU1, FCSECU2
Elevator position	FCSECU1, FCSECU2
Spoiler position	FCSECU1, FCSECU2
MLS channel	MLS1, MLS2
MLS AZ/EL deviation	MLS1, MLS2
Radio Altitude	RA1, RA2
Fuel Quantity	FGC / FQC
Fuel discretes	FGC / FQC
Fuel Temperature	Fuel Temperature Sensor
Fuel Flow	From Fuel Flow Sensors
VOR/ILS frequencies	VHFNAV1, VHFNAV2
LOC/GS deviation	VHFNAV1, VHFNAV2
VOR bearing	VHFNAV1, VHFNAV2
Hydraulics Pressure	Hydraulics Sensor
Hydraulics Quantity	Hydraulics Sensor
Parking brake hydraulic pressure	Hydraulics Sensor
Air/Ground status	PSEU

Doors Status	PSEU
Main Oil Pressure	FADEC1, FADEC2
Engine torques	FADEC1, FADEC2
Flap position	FPIU
Static Air Temperature	ADU1, ADU2
ED Brightness	ESCP
System page status	ESCP
Reversion status	ESCP
APU generator volt	EPGDS
AC Bus Voltage	EPGDS
AC Bus Load	EPGDS
Generator Load	EPGDS
TRU Load	EPGDS
Secondary Bus Voltage	EPGDS
Battery Load	EPGDS
Essential Bus Voltage	EPGDS
Main Bus Voltage	EPGDS
Battery Temperature	EPGDS
External Power Discrete	EPGDS
frequency	DME1, DME2
Distance	DME1, DME2
Time To Station	DME1, DME2
Ground Speed	DME1, DME2
Display Control	TCAS
Vertical Resolution	TCAS
Output Sense Level and Reply Data	TCAS
Intruder range, altitude, bearing	TCAS
Failure status	TCAS
frequency	ADF1, ADF2
Bearing	ADF1, ADF2

Refer to Figure 23.
The FDPS receives data from the Attitude and Heading Reference System (AHRS) and the EFIS Control Panels through the EIS and

supplies it to the weather radar system. It supplies the data as shown in the table that follows:

Parameter	Source
Roll	AHRS1, AHRS2
Pitch	AHRS1, AHRS2
Magnetic Heading	AHRS1, AHRS2
WXR Range	ESCP

Refer to Figure 24.
The FDPS receives data from different systems and supplies it to other non avionic systems through a General Purpose Data bus. It supplies the data as shown in the table that follows:

Parameter	Source
VHFCOM Frequency	VHFCOM1, VHFCOM2
ADF Frequency	ADF1, ADF2
Selected Heading	FGCP
Selected Altitude	FGCP
Standard Pressure Altitude	ADU1, ADU2
Barometric Corrected Altitude	ADU1, ADU2
Mach	ADU1, ADU2
Calibrated airspeed	ADU1, ADU2
VMO	ADU1, ADU2
True airspeed	ADU1, ADU2
Total Air Temperature	ADU1, ADU2
Altitude Rate	ADU1, ADU2
Static Air Temperature	ADU1, ADU2
Hydraulic pressure	Hydraulic Sensor
HP compress Exit Pres (P3.0)	FADEC1, FADEC2
Propeller speed	FADEC1, FADEC2
Fuel flow sensor	FADEC1, FADEC2
Vertical Speed	AHRS1, AHRS2
Roll	AHRS1, AHRS2

Pitch	AHRS1, AHRS2
Magnetic Heading	AHRS1, AHRS2

Refer to Figure 25.
The FDPS receives a valid discrete from each EIS indication and sends it to the DUs.

Refer to Figure 26.
The FDPS1 calculates the data from Index Control Panel 1 (ICP1) and FDPS2 calculates data from ICP2. The ICPs supply the data to the FDPS's as shown in the table that follows:

Parameter	Source
Speed Bug Selector	ICP1, ICP2
Speed Bug Index Setting	ICP1, ICP2
Decision Height	ICP1, ICP2

The ICP1 barometric altitude setting is calculated by the ADC1 and ICP2 barometric altitude setting is calculated by the ADC2.

Refer to Figure 27.
The FDPS calculates data from the FGCP and supplies it to the EIS, FGM and opposite IOP that follows:

Parameter	Source
Heading Selection	FGCP
Course Selection	FGCP
Attitude selection	FGCP
HSI SEL	FGCP
Navigation Source	FGCP

The IOP, when valid, calculates the parameters from the FGCP and transmits its result through IOP2 to the EIS.

If the data connection between IOP1 and the FGCP malfunctions, IOP1 also calculates parameters from the FGCP through IOP2.

If IOP1 malfunctions, the FDPS continues to operate with calculations from IOP2 and with parameters previously calculated by IOP1.

Refer to Figure 28.

The FDPS1 calculates with IOP1 with data from FGCP and sends the result to FDPS2. The FDPS sends the data to the EIS. When the link malfunctions between IOP1 and the FGCP, the FDPS1 will calculate data from the FGCP through FDPS2.

Refer to Figure 29.
The FDPS1 calculates the data from the ESID Control Panel (ESCP). It supplies the data to the FDPS's as shown in the table that follows:

Parameter	Source
ED Brightness	ESCP
MFD1 Reversion	ESCP
MFD2 Reversion	ESCP
ATT/HDG Reversion	ESCP
ADU Reversion	ESCP

Refer to Figure 30.
The DME is tuned by the MLS or VHF NAV. The FDPS moves an external relay contact to select a navigation source, MLS or VHF NAV.

Refer to Figure 31.
The ARCDUs control the VHF NAV or MLS audio level. The IOMS sends a discrete signal to control the Remote Control Audio Unit (RCAU) navigation source audio.

Refer to Figure 32.
The FDPS uses data from the FADEC oil pressure discrete signal to change the Passenger Address Amplifier (PAA) gain. The FDPS supplies a discrete signal to the PAA when oil pressure from the two engines are less than their threshold. This discrete signal causes the PAA to reduce its PA gain by 6 dB when the two engine are not running.

Refer to Figure 33.
The FDPS calculates data from the PSEU, PEC and FPIU to inhibit the climb and increase climb signals to the TCAS.

Refer to Figure 34.
The FDPS changes the analog fuel flow signals from the fuel flow meter to digital data for transmission to the EIS through a data bus.

The range of the fuel flow signal is 80 to 3000pph.

Refer to Figure 35.
The FDPS changes the analog fuel tank temperature signals from the HLCU signal conditioner to digital data for transmission to the EIS through a data bus.

The range of the fuel tank temperature signal is –70°C to +75°C.

Refer to Figure 36.
The FDPS changes the analog fuel inlet temperature signals from the fuel temperature sensor to digital data for transmission to the EIS through a data bus.

The range of the fuel inlet temperature signal is –55°C to +121°C.

Refer to Figure 37.
The FDPS changes the analog hydraulic pressure and quantity signals from the different hydraulic systems to digital data for transmission to the EIS through a data bus.

The hydraulics pressure data from all hydraulic systems is supplied to the two IOMs. The hydraulics quantity data from system 1 and 3 are supplied to IOM1 and hydraulic quantity 2 to IOM2.

NOTE

Note: A FDPS1 malfunction causes hydraulic quantity indication 1 and 3 to also malfunction. If FDPS2 malfunctions, hydraulic quantity 2 indication also malfunctions.

The range of the hydraulic quantity signal is 0 to 4000PSI.

Refer to Figure 38.
The FDPS changes the analog main oil pressure signals from the engine oil pressure sensors to digital data for transmission to the EIS through a data bus.

The MOP data is supplied to the two IOMs. MOP1 data is supplied through its related IOM1 to the EIS for indication. The MOP2 data is supplied through IOM2 to the EIS when the right main bus power is energized and by IOM1 when the right main bus is not energized.

NOTE

Note: A FDPS1 malfunction causes MOP1 to also malfunction.

NOTE

A FDPS2 malfunction causes MOP2 indication to also malfunction if the right main bus is energized.

The range of the main oil pressure signal is 0 to 260PSI.

Refer to Figure 39.
The FDPS changes the analog parking brake pressure signals from the parking brake hydraulic pressure transmitter to digital data for transmission to the EIS through a data bus.

The parking brake pressure data is supplied to the two IOMs and is supplied through IOM1 to the EIS for indication.

NOTE

Note: A FDPS1 malfunction causes the parking brake pressure indication to also malfunction.

The range of the parking brake pressure signal is 0 to 4000PSI.

Refer to Figure 34.
The FDPS changes the analog flap position signals from the FPIU to digital data for transmission to the EIS through a data bus.

The flap position data is supplied through the IOM1 from the right FPIU channel to the EIS for indication. The data changes to the left FPIU channel if the right malfunctions and to IOM2 if IOM1 malfunctions.

Refer to Figure 40.
The Aircraft Configuration Module stores information. The ACM stores information from the systems that follow.

- TCAS
- ACARS
- SELCAL
- MLS
- FMS
- VHF COM
- HF
- SSM
- ADF
- DME
- ATC
- RA
- IRS

- Aircraft type
- Engine type
- Temperatures
- Quantities
- Fuel used
- Ground mapping
- Aircraft serial numbers
- AOA correction
- WTG outputs
- CRC LSB indications
- CRC MSB indication.

Integrated Flight Cabinets (IFC1, IFC2)

Refer to Figure 41.
The Integrated Flight Cabinets (IFC 1, IFC 2) are Electromagnetic Containment (EMC) compartments that protect the components inside of it from High Intensity Radio Frequencies (HIRF).

It has two avionics cooling fans to remove air to cool the components inside of it.

The IFC have modules that are line replaceable. Each IFC contains the Line Replaceable Modules (LRM) that follow:

- Input/Output Processor (IOP 1, IOP 2)
- Input/Output Module (IOM 1, IOM 2)
- Aircraft Configuration Module (ACM)
- The Ground Proximity Warning System Converter (GPWSC) located in the Stall Protection Module (SPM 1, SPM 2)
- Prime Power Supply Module (PPSM 1, PPSM 2).

Each LRM is guided into the IFC using a guide rail and guide pins. The top and bottom guide pins at the Integrated Flight Cabinets (IFC 1, IFC 2) back panel makes sure that the module and the back panel come together precisely in the correct location.

The Line Replaceable Units (LRU) are held in place with front locking levers. This gives easy installation and removal of the Line Replaceable Unit (LRU) without any tools.

The IFCs are 10.75 in. (273 mm) wide, by 8.9 in. (226 mm) high, by 16.14 in. (410 mm) in depth and weighs 29.8 lb (13.1 kg).

The IFC are installed in the avionics rack.

Input/Output Processor (IOP 1, IOP 2)

Refer to Figure 42.
The Input/Output Processors (IOP 1, IOP 2) receives, calculates, and then transmits data to other avionics systems.

Each FDPS has an IOP and IOM. The FDPS and the Centralized Diagnostic System (CDS) uses the same processor located on the Input/Output Processor (IOP1) module but they function independently.

Each IOP has two electronic boards a CPU Board and an ARINC Board.

The Input/Output Processors (IOP 1, IOP 2) are 1.5 in. (38 mm) wide, by 7.2 in. (183 mm) high, 13.26 in. (337 mm) in depth, and weigh less than a pound (0.45 kilograms).

The IOPs are Line Replaceable Modules (LRM) that are located in their related IFC.

Input/Output Module (IOM 1, IOM 2)

Refer to Figure 42.
The Input/Output Modules (IOM 1, IOM 2) receive different analog and discrete signals. It changes the inputs into an ARINC 429 format and transmits them to the IOPs for calculations.

The IOM are 1.5 in. (38 mm) wide, by 7.2 in. (183 mm) high, 13.26 in. (337 MM) in depth, and weigh less than a pound (0.45 kg).

The IOMs are Line Replaceable Modules (LRM) that are located in their related IFC.

Prime Power Supply Module (PPSM 1, PPSM 2)

Refer to Figure 42.
The PPSM receives 28 Vdc from the main and essential busses. The PPSM has the functions that follow:

– Filtering
– Pre–regulation
– Energy reserve
– Power limitaion.

The modules that normally receive power are as follows:

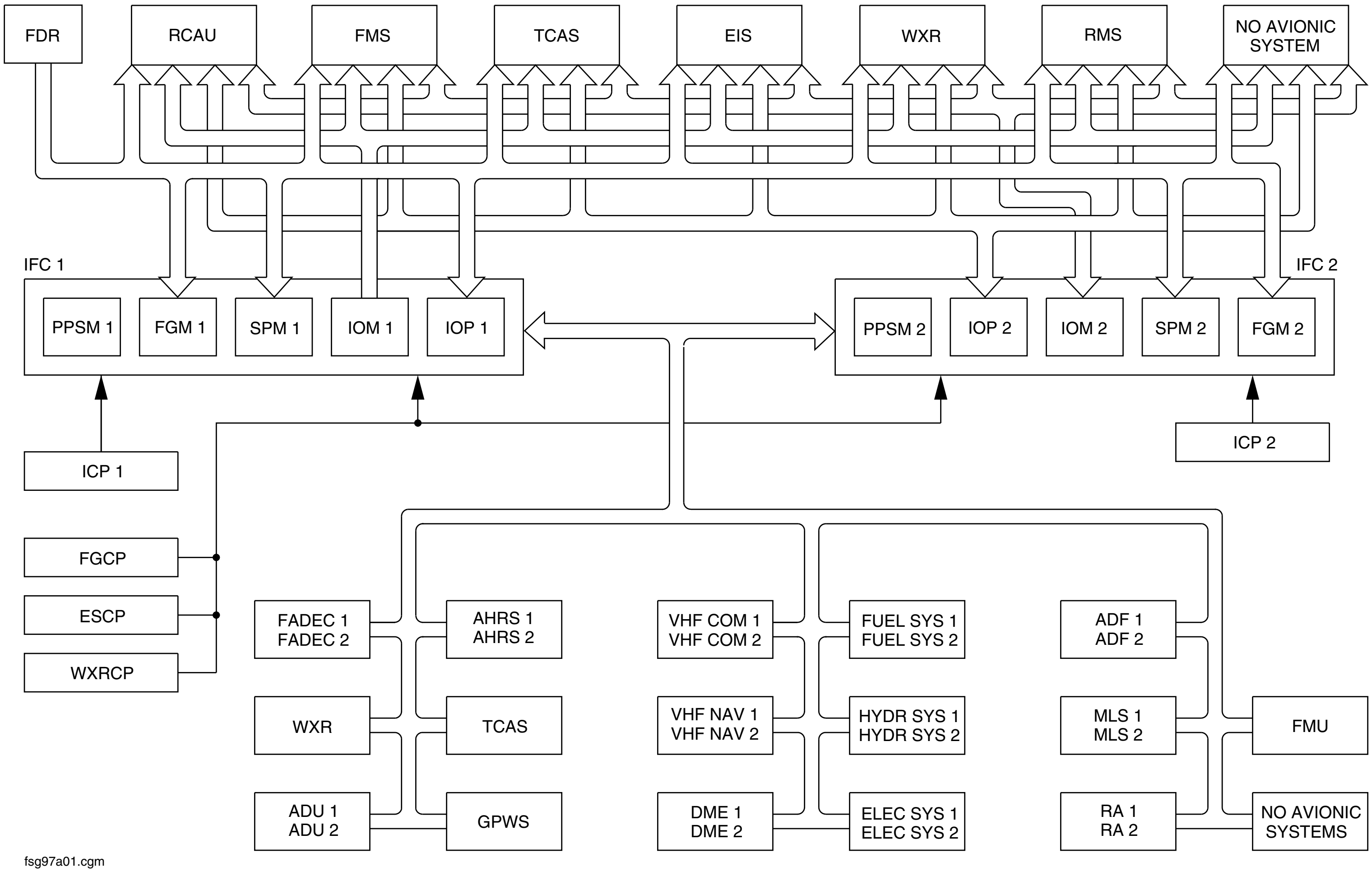
– SPARE1 LRM
– SPM1 LRM
– IOM2 LRM
– IOP2 LRM
– FGM2 LRM
– SPARE2 LRM
– SPM2 LRM.

The modules that receive emergency power are as follows:

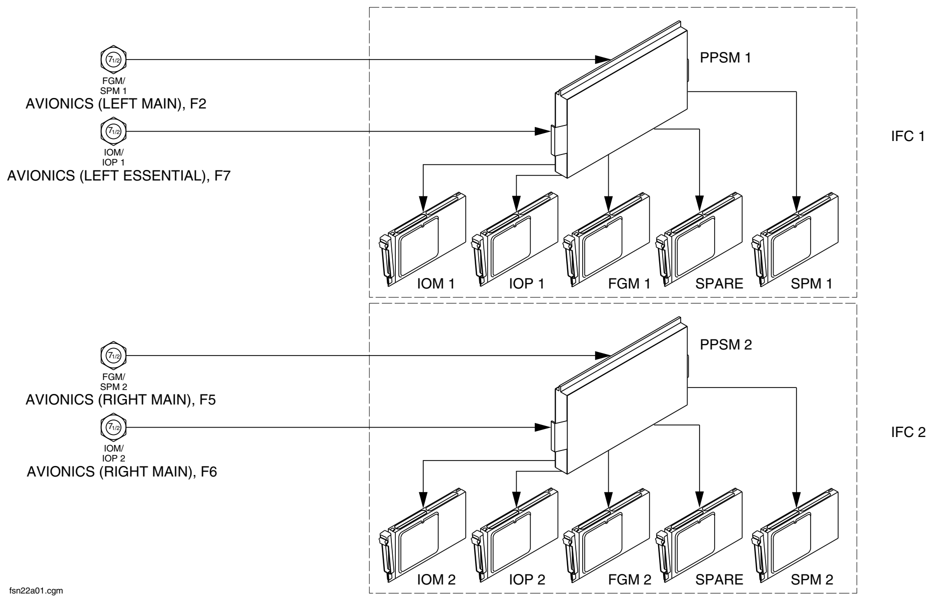
– IOM1 LRM
– IOP1 LRM
– FGM1 LRM

The Prime Power Supply Module (PPSM 1, PPSM 2) are 1 in. wide, by 7.2 in. high, 13.26 in. in depth, and weigh 1.1 lb.

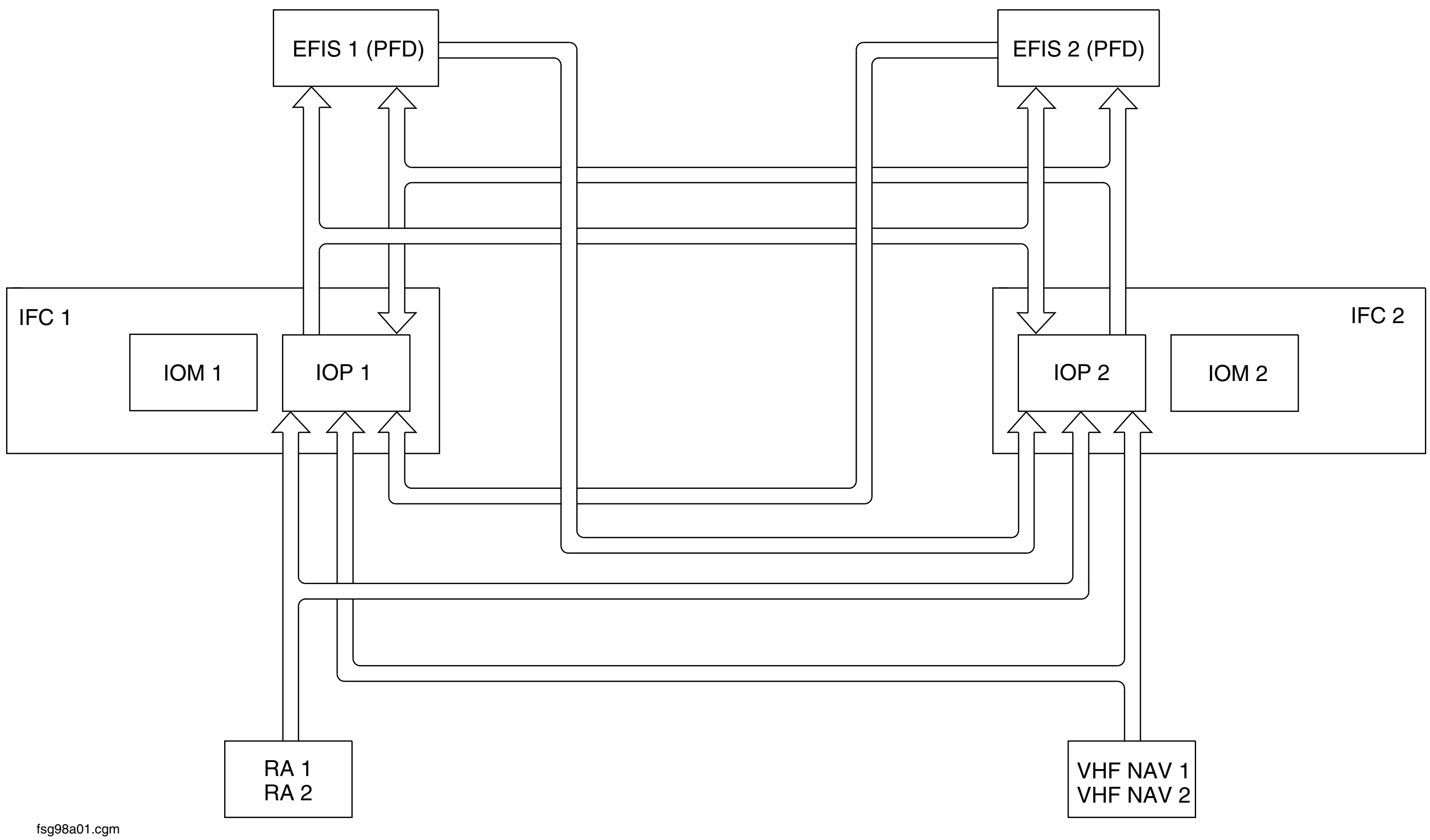
The Prime Power Supply Module (PPSM 1, PPSM 2) are Line Replaceable Modules (LRM) that are located in their related IFC.



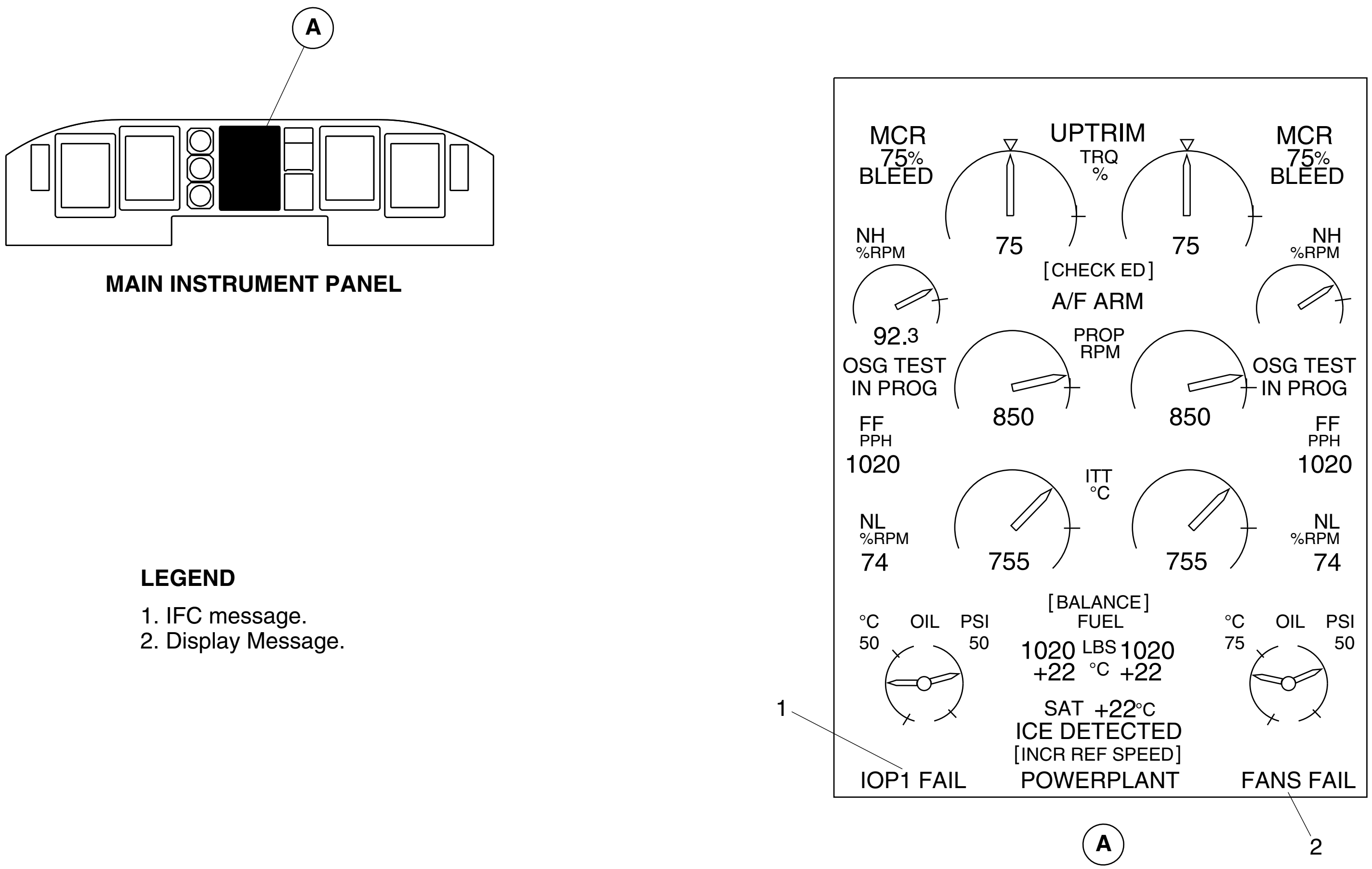
FLIGHT DATA PROCESS SYSTEM BLOCK DIAGRAM
Figure 1



FLIGHT DATA PROCESS SYSTEM BLOCK DIAGRAM, POWER
Figure 2



EIS ESSENTIAL MONITORING
Figure 3

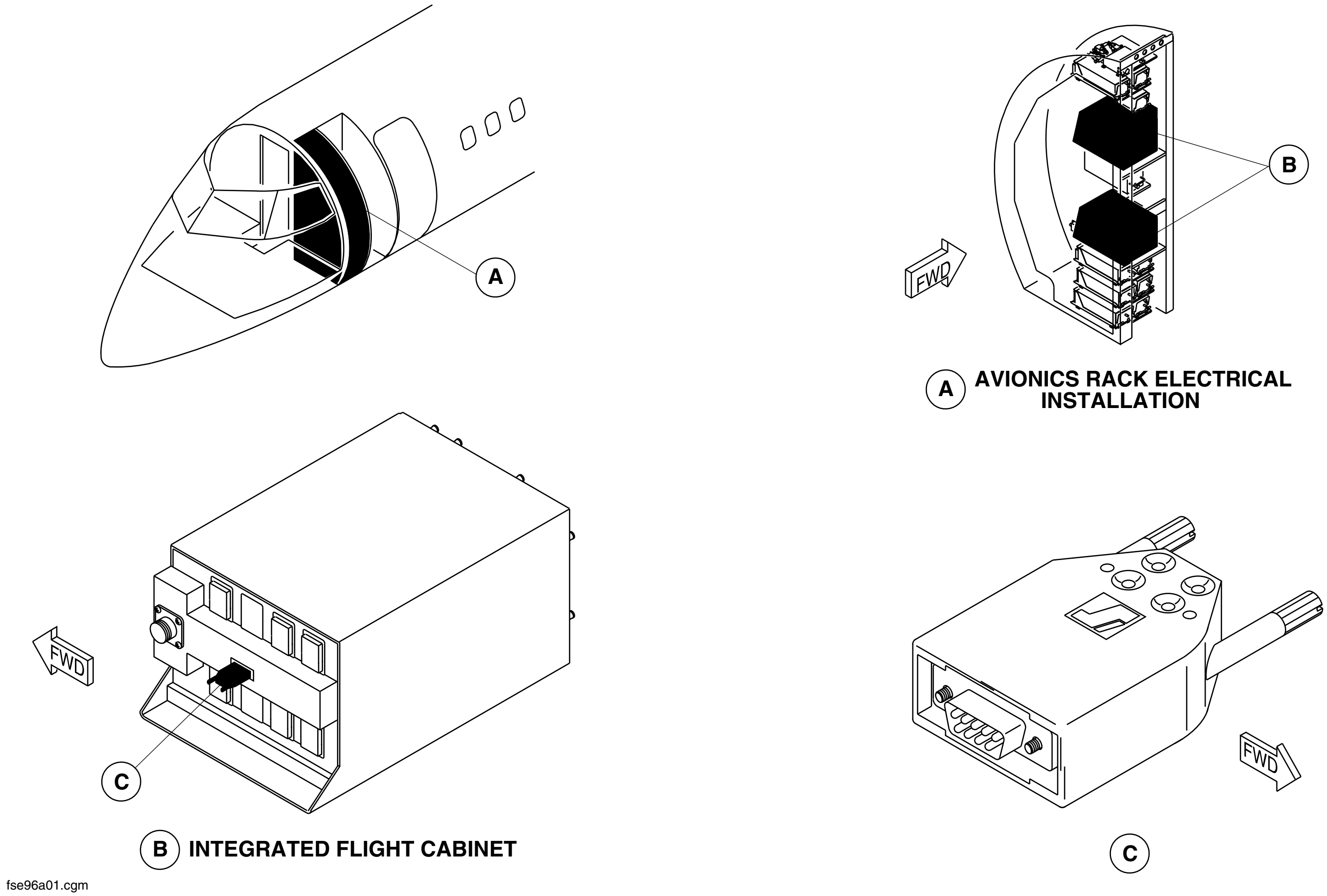


LEGEND

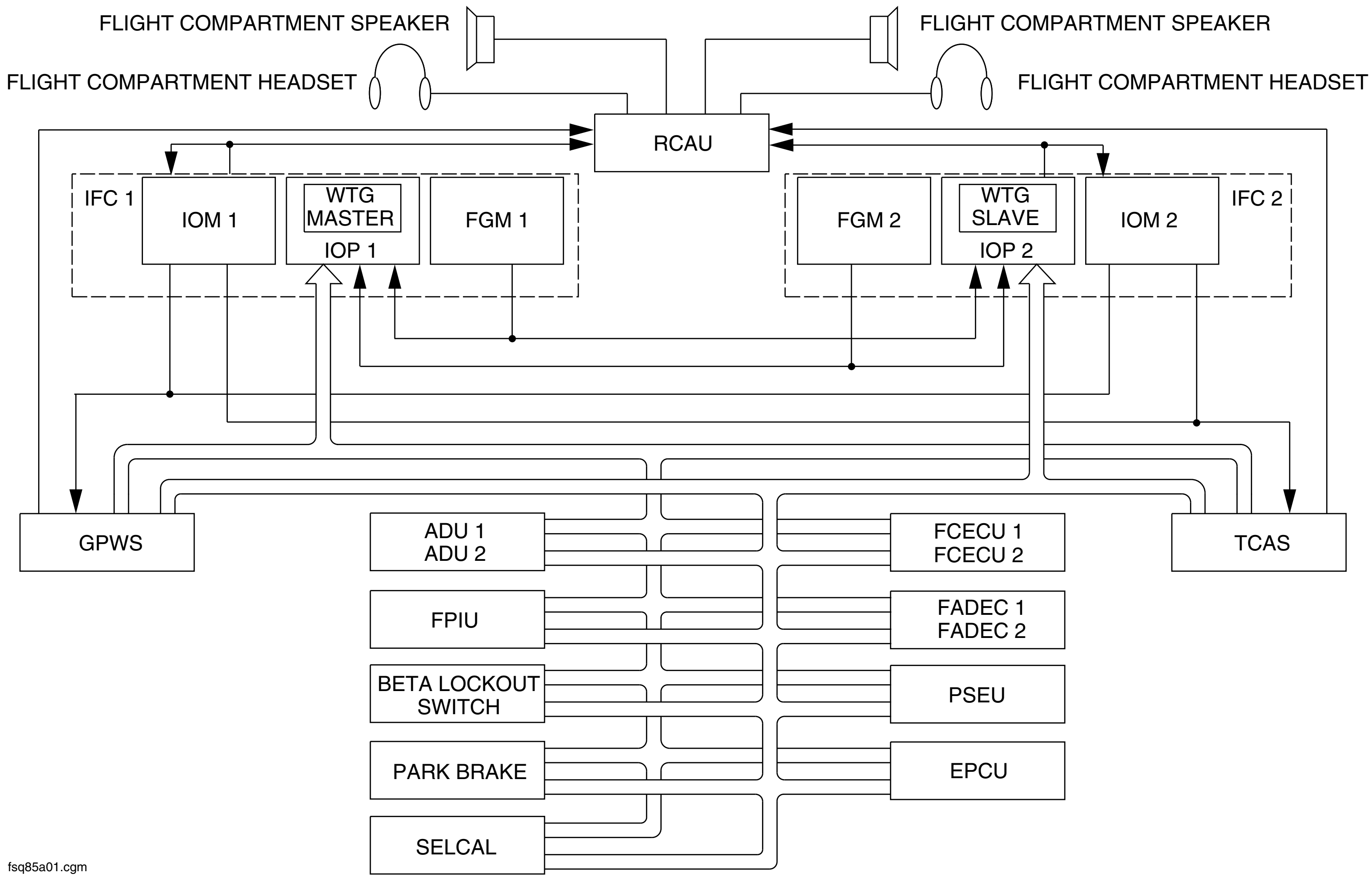
1. IFC message.
2. Display Message.

fse57a05.cgm

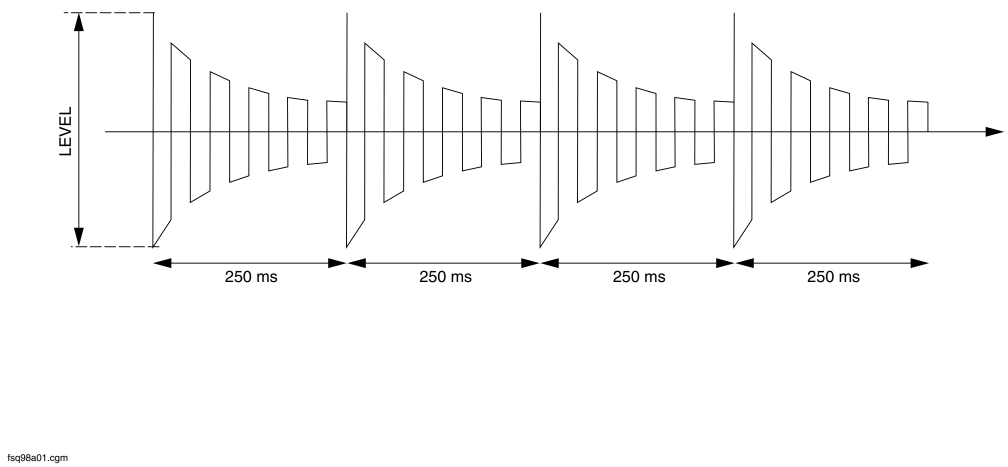
FDPS MESSAGE GENERATION
Figure 4



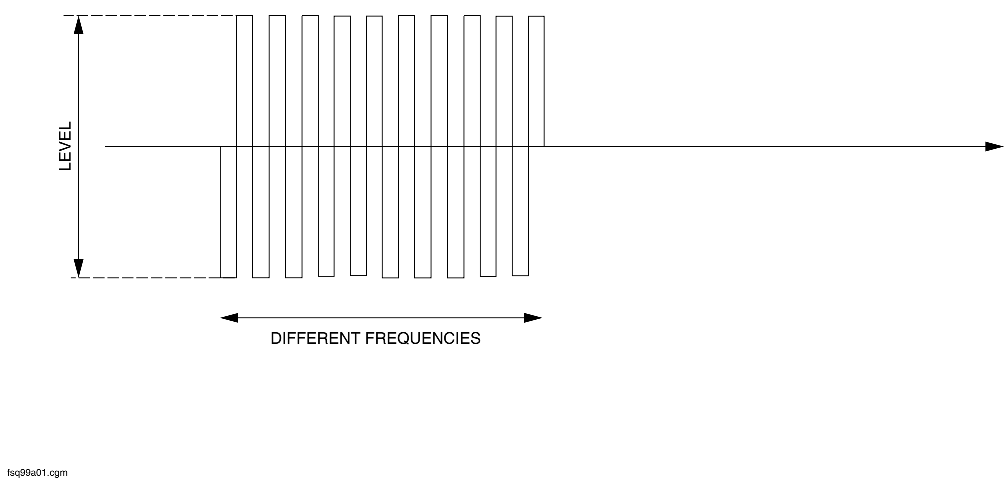
FDPS AIRCRAFT CONFIGURATION MODULES
Figure 5



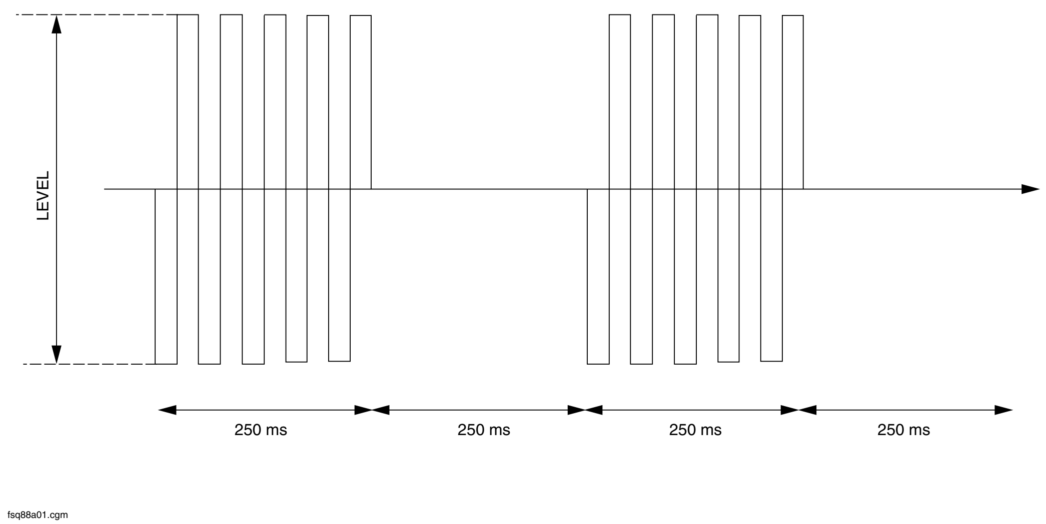
FDPS WARNING TONE GENERATORS (WTG1, WTG2)
Figure 6



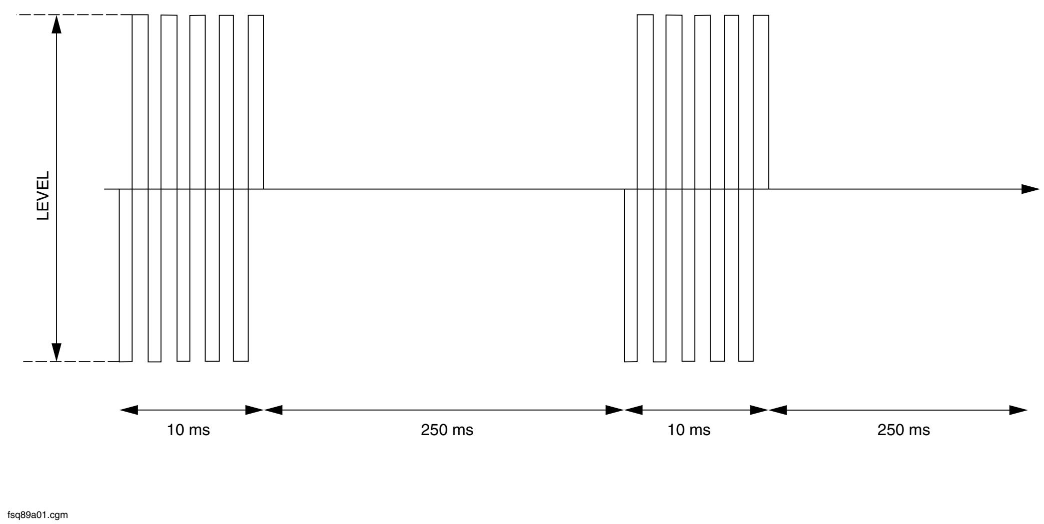
WTG AURAL WARNING TYPE A
Figure 7



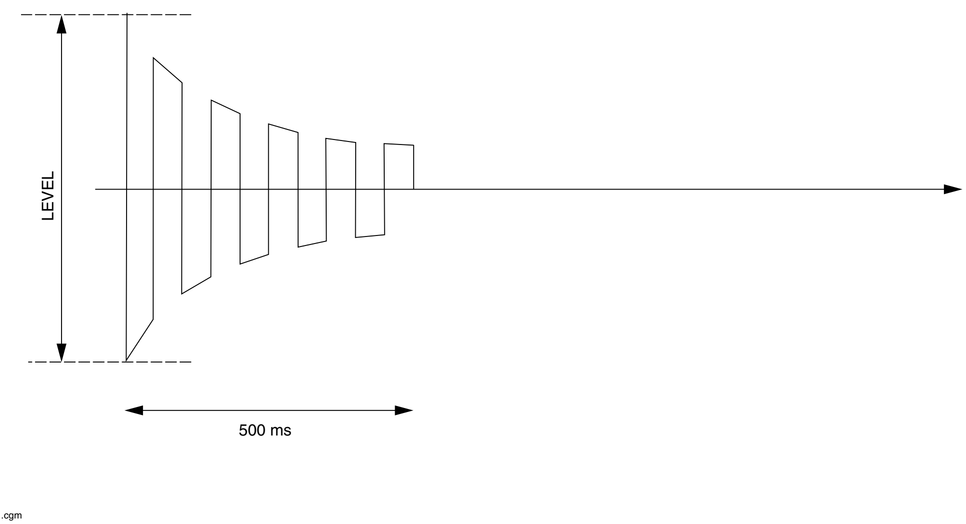
WTG AURAL WARNING TYPE B
Figure 8



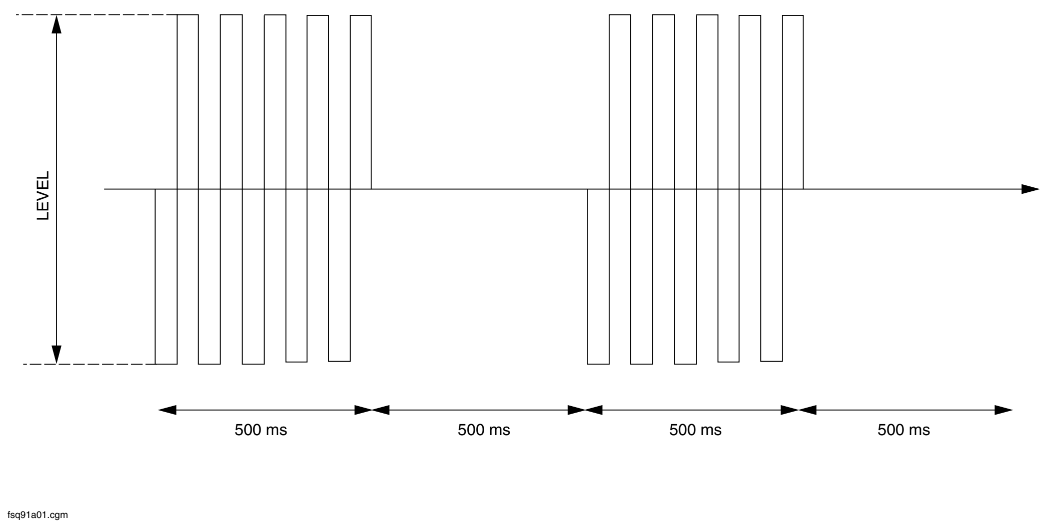
WTG AURAL WARNING TYPE C
Figure 9



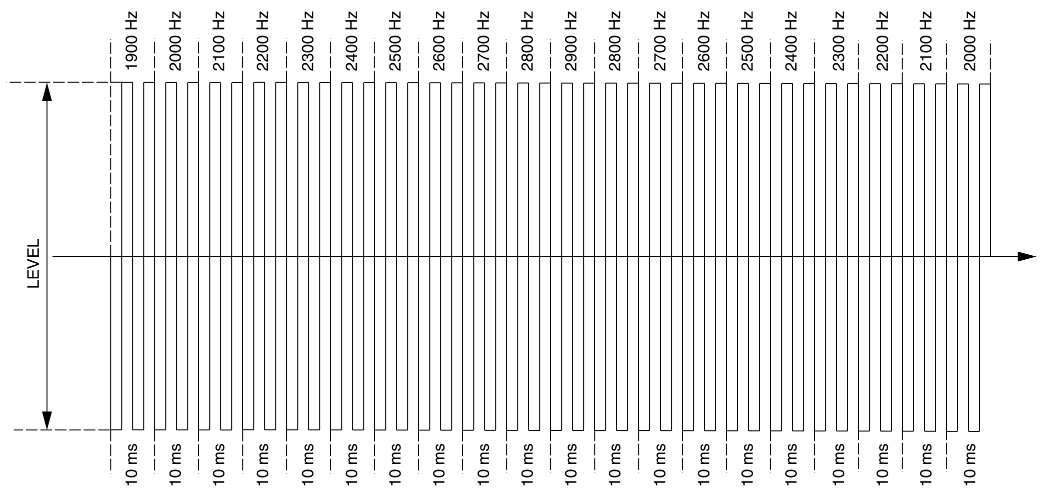
WTG AURAL WARNING TYPE D
Figure 10



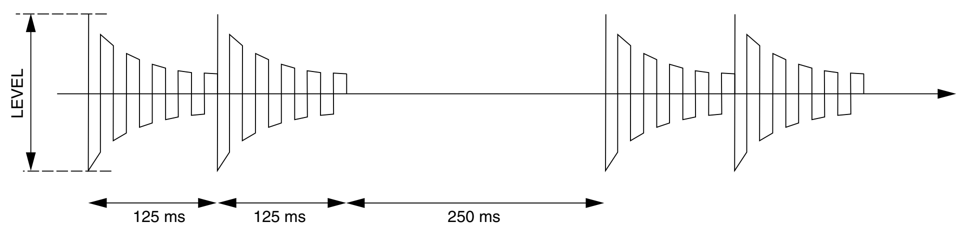
WTG AURAL WARNING TYPE E
Figure 11



WTG AURAL WARNING TYPE F
Figure 12

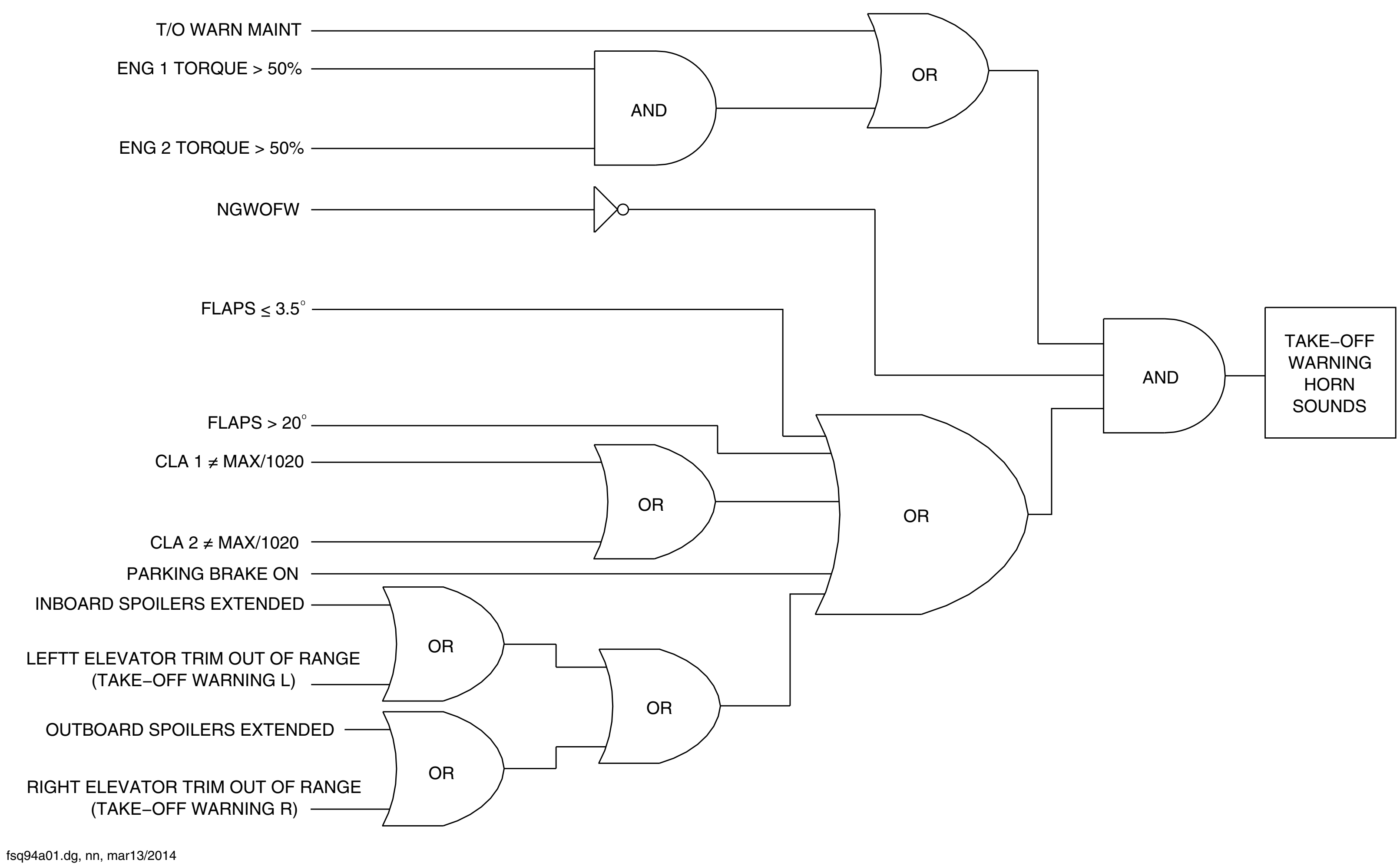


WTG AURAL WARNING TYPE G
Figure 13

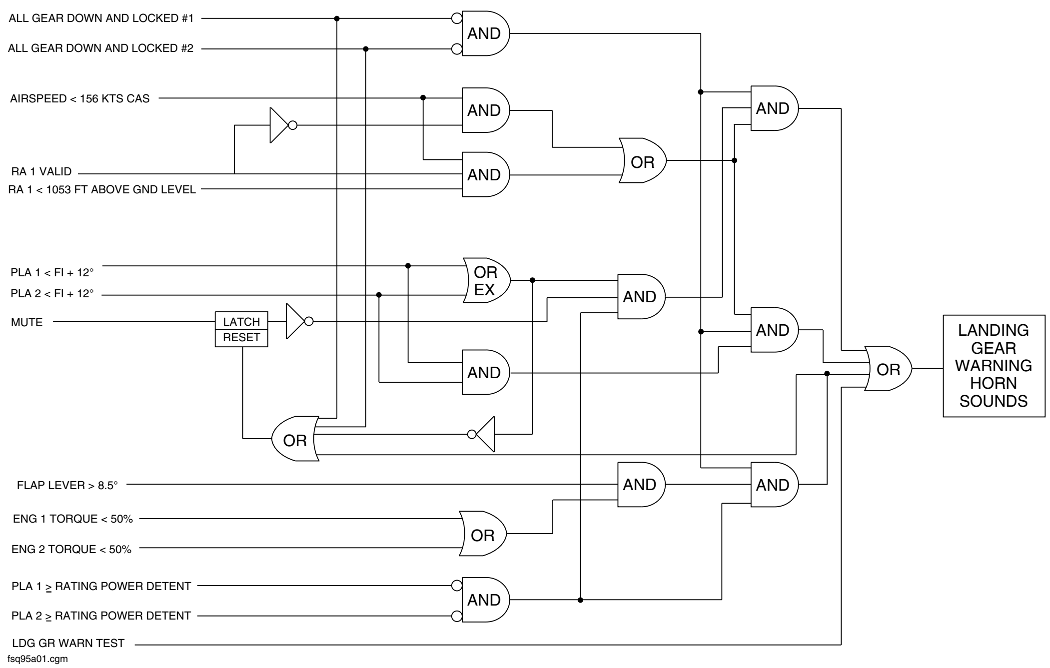


fsq93a01.cgm

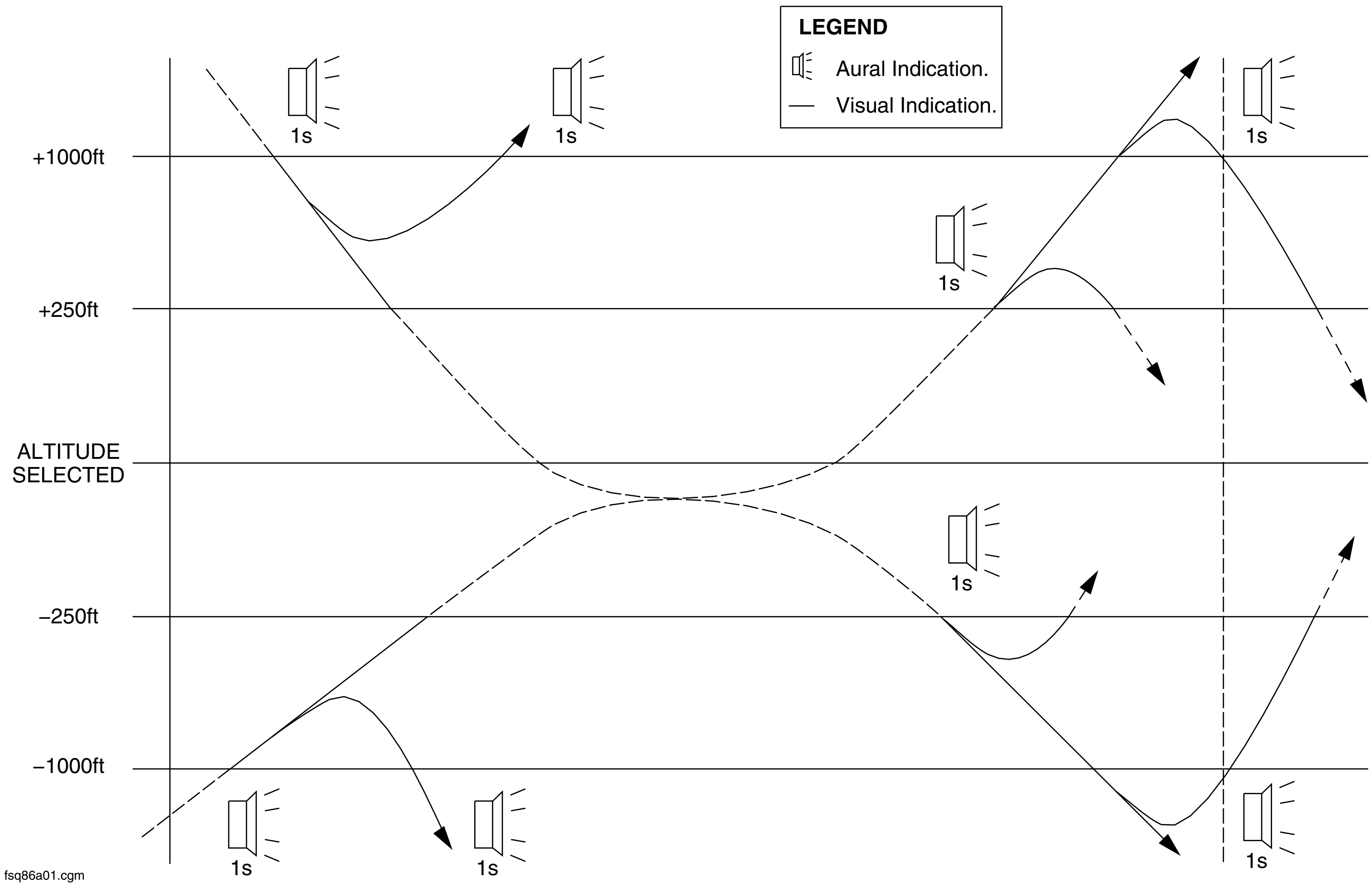
WTG AURAL WARNING TYPE H
Figure 14



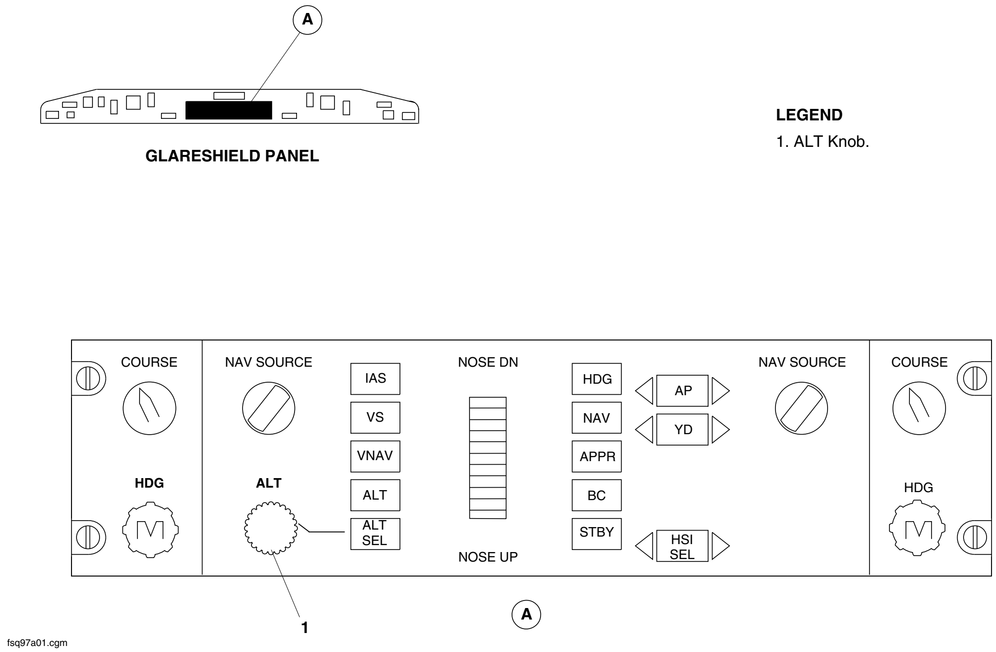
FDPS Incorrect Take–Off Configuration Warning
Figure 15



FDPS INCORRECT LANDING GEAR CONFIGURATION
Figure 16



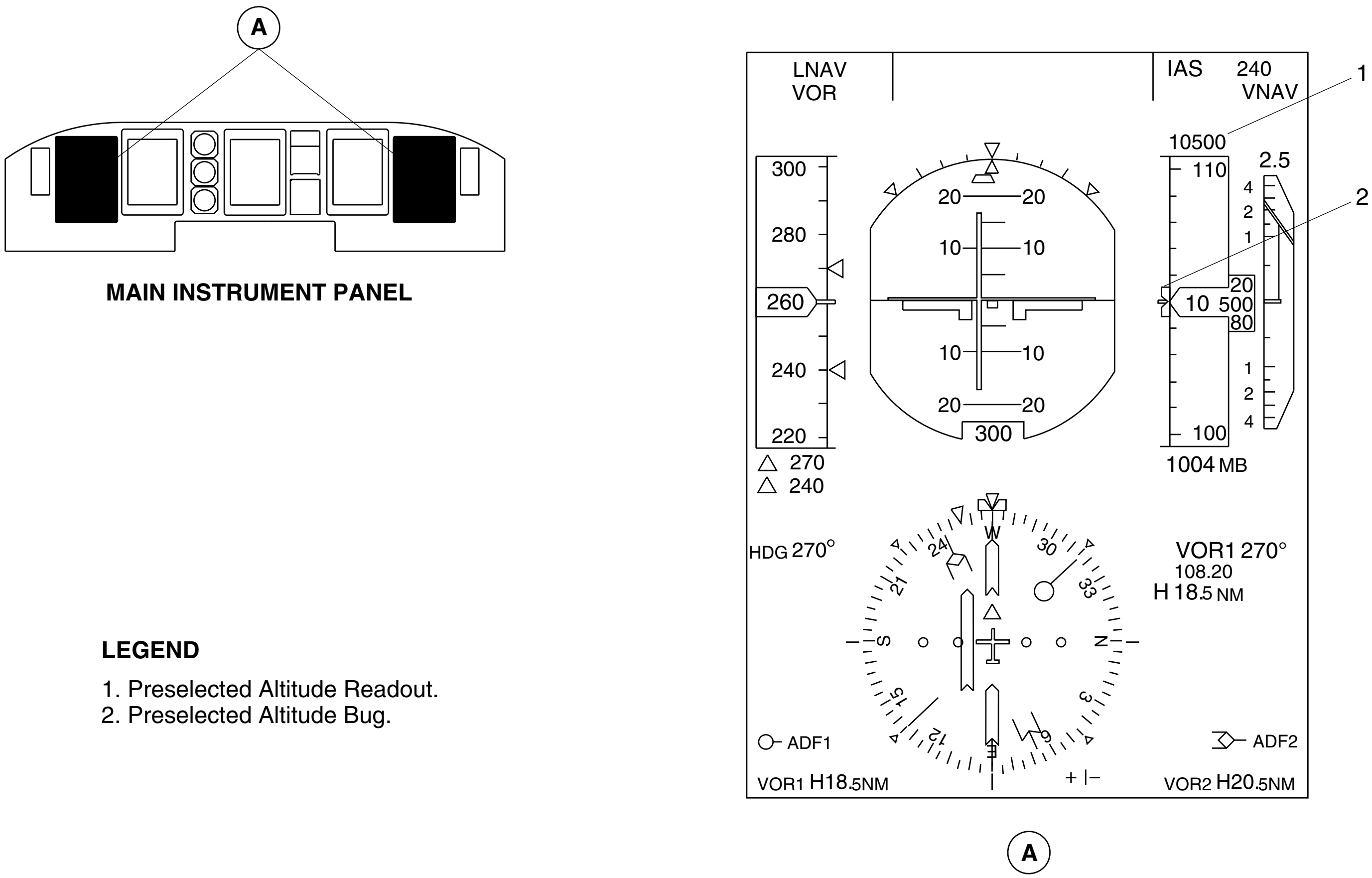
FDPS ALTITUDE ALERT
Figure 17



LEGEND

1. ALT Knob.

FDPS ALTITUDE ALERT SELECTION
Figure 18

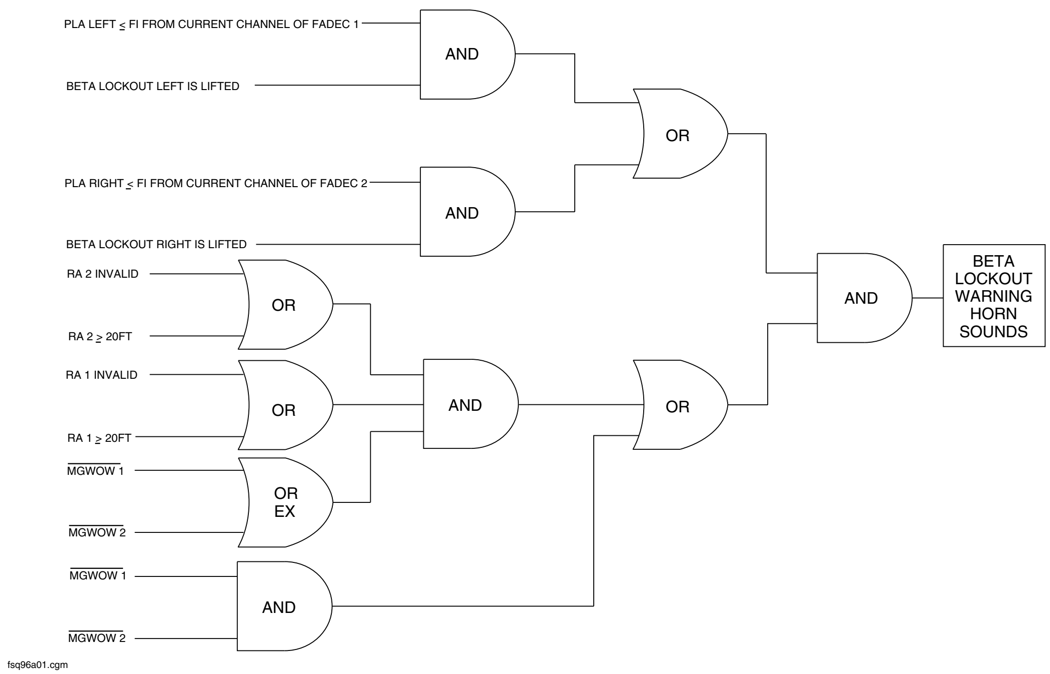


LEGEND

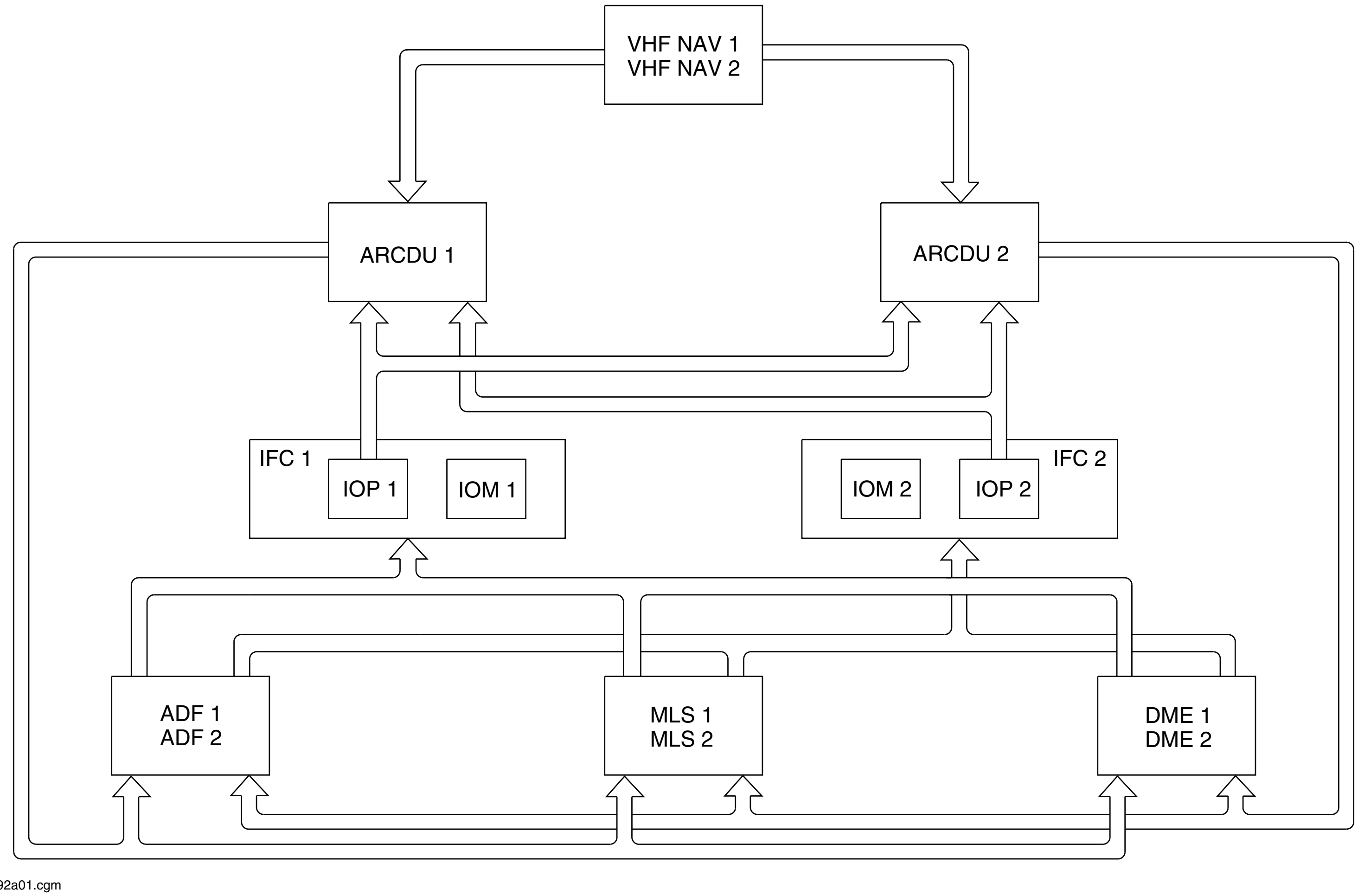
1. Preselected Altitude Readout.
2. Preselected Altitude Bug.

fae56a02.cgm

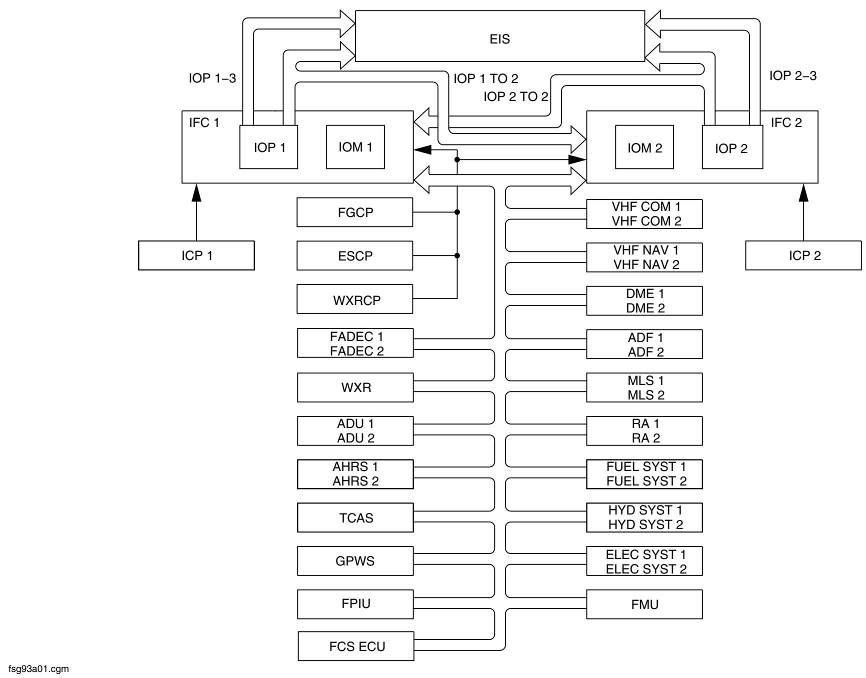
FDPS ALTITUDE ALERT INDICATION
Figure 19



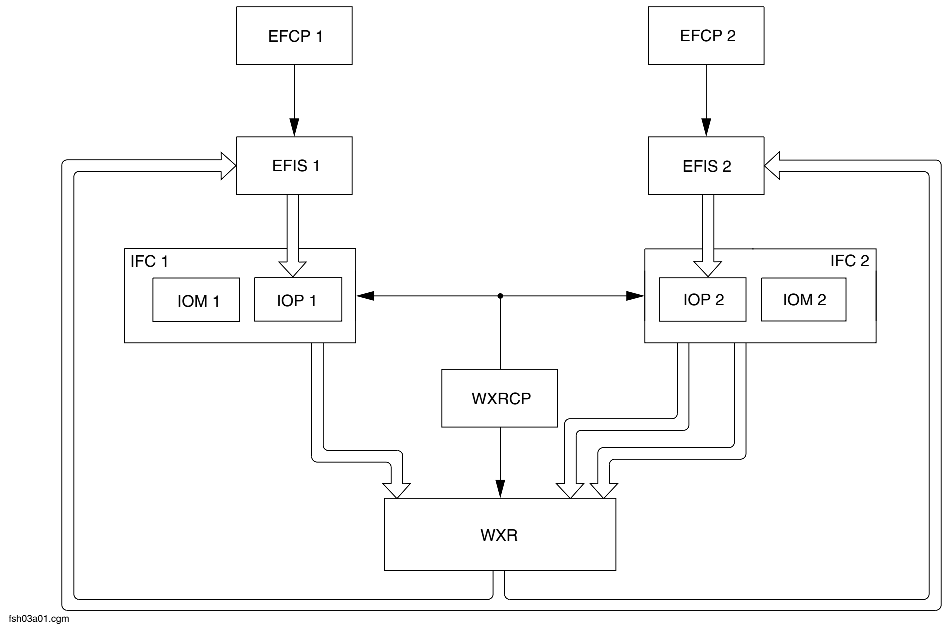
FDPS BETA LOCKOUT WARNING
Figure 20



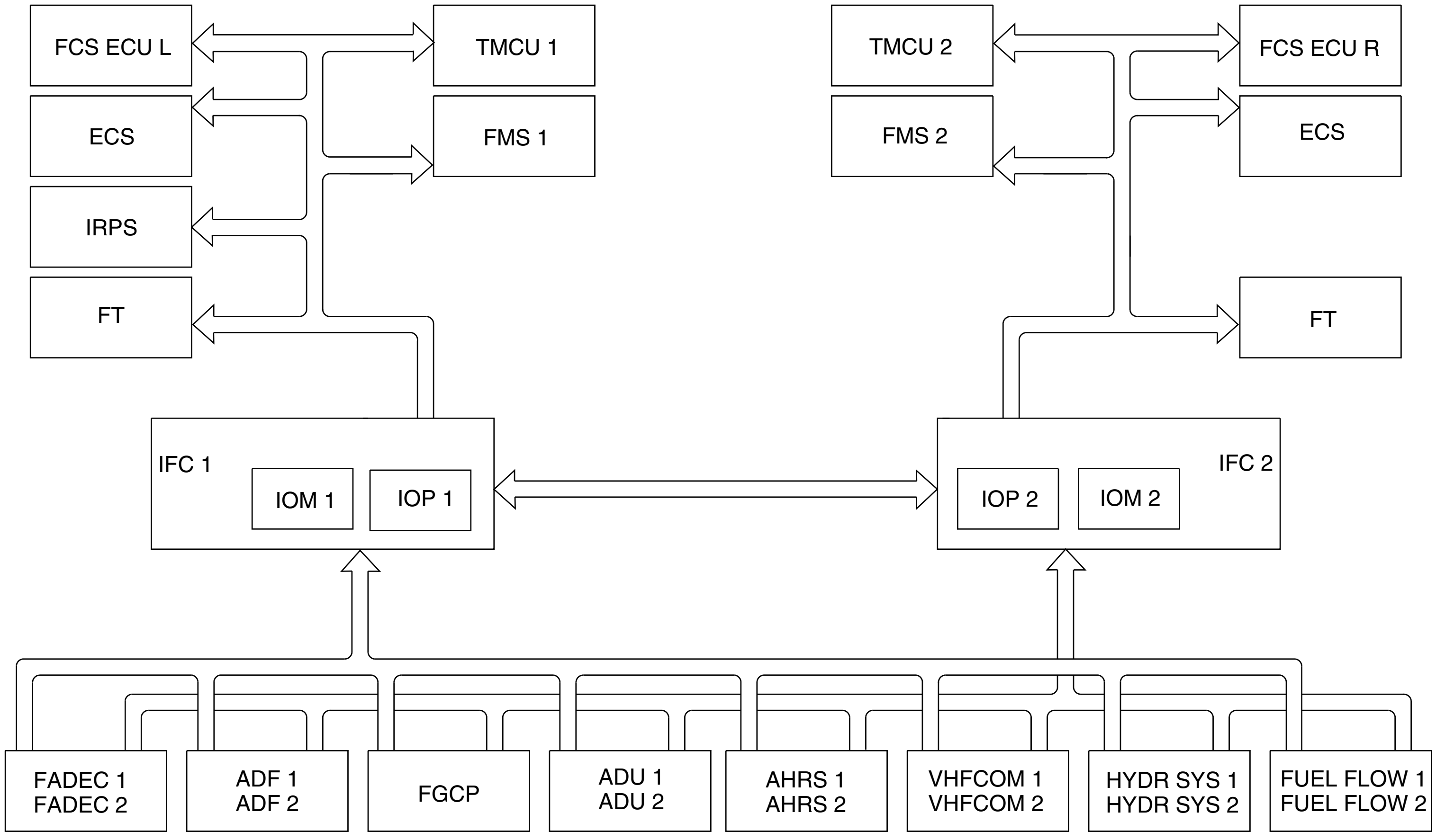
FDPS ARCDU CALCULATION AND CONCENTRATION
Figure 21



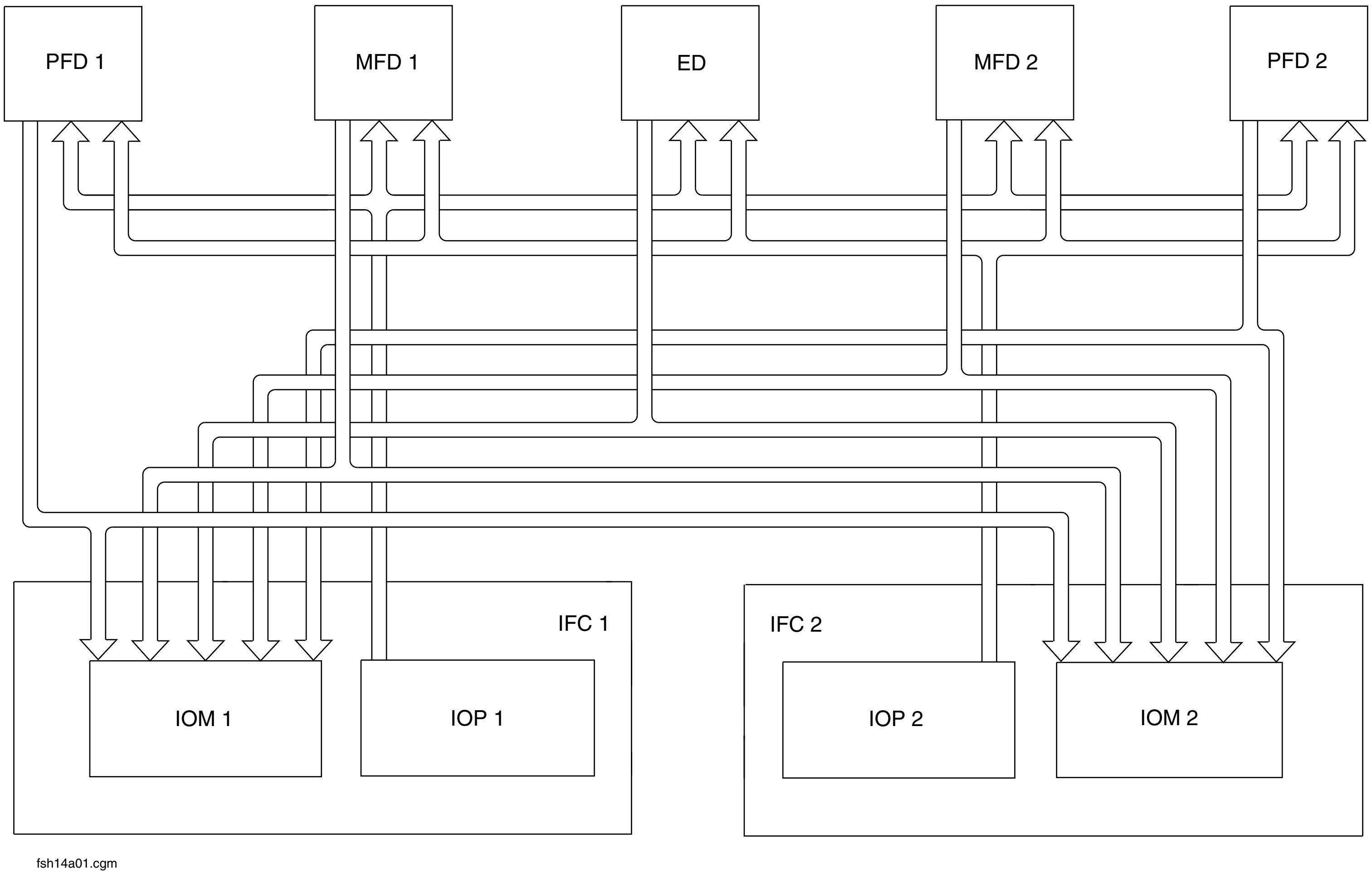
FDPS EIS AND OPPOSITE IOP CONCENTRATION
Figure 22



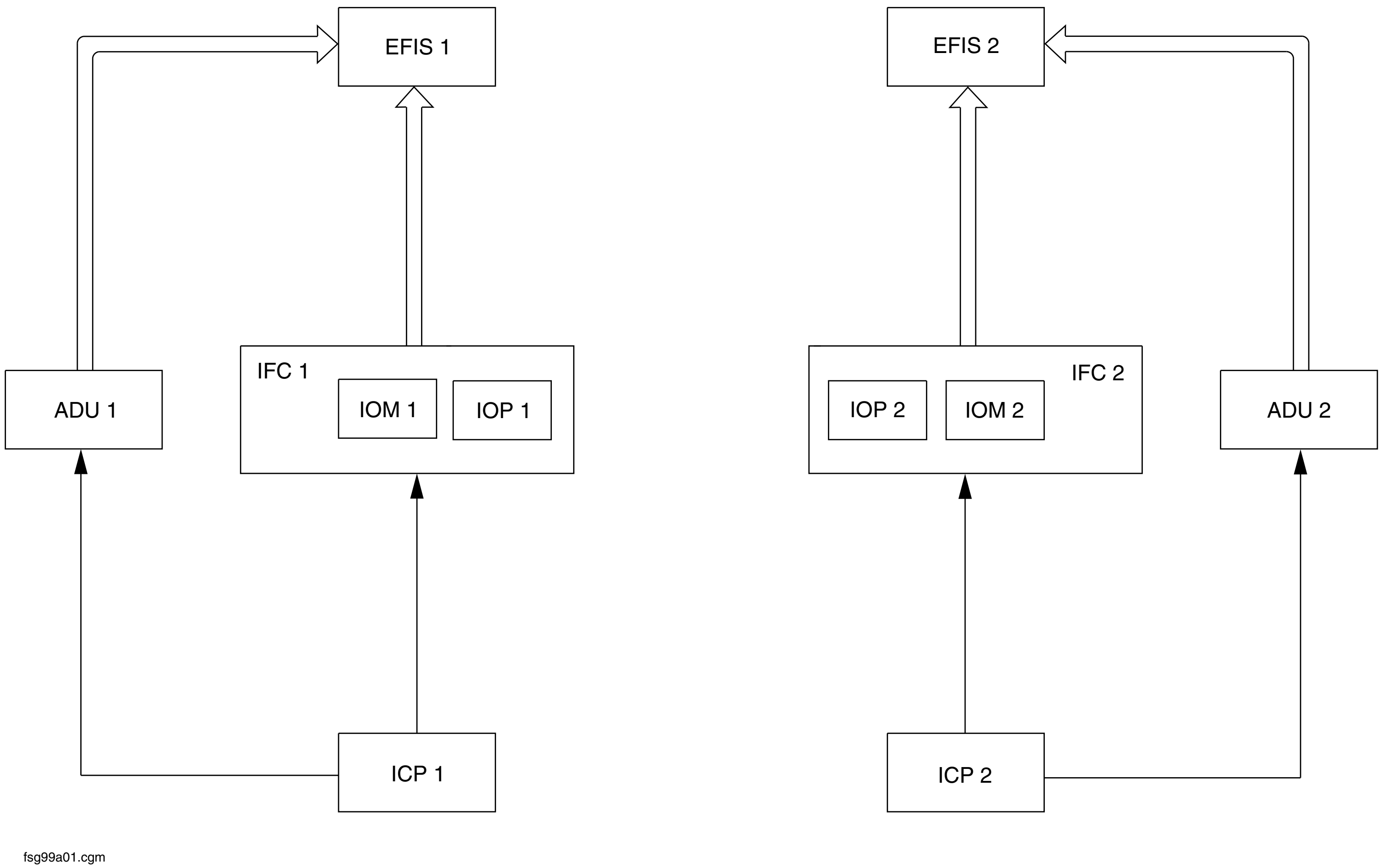
FDPS TCAS AND WXR CALCULATION AND CONCENTRATION
Figure 23



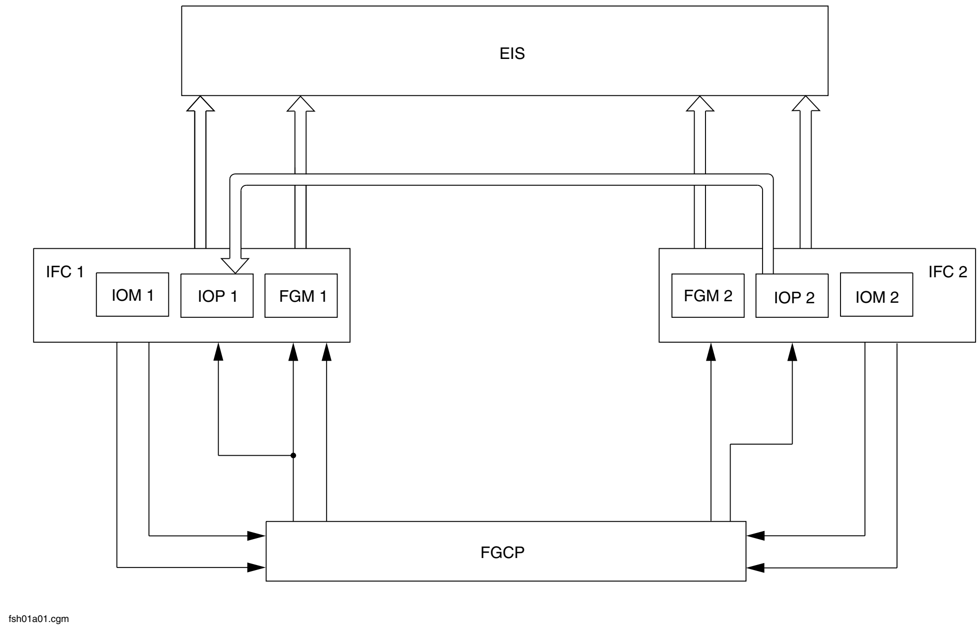
FDPS GENERAL PURPOSE DATA BUS (GPDB) CONCENTRATION
Figure 24



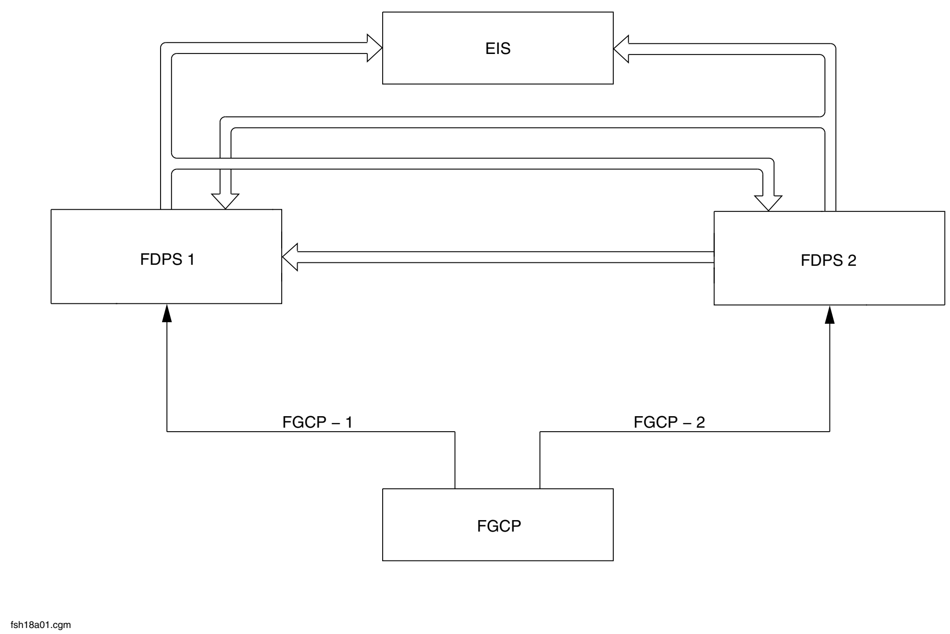
FDPS DISPLAY UNIT CALCULATION AND CONCENTRATION
Figure 25



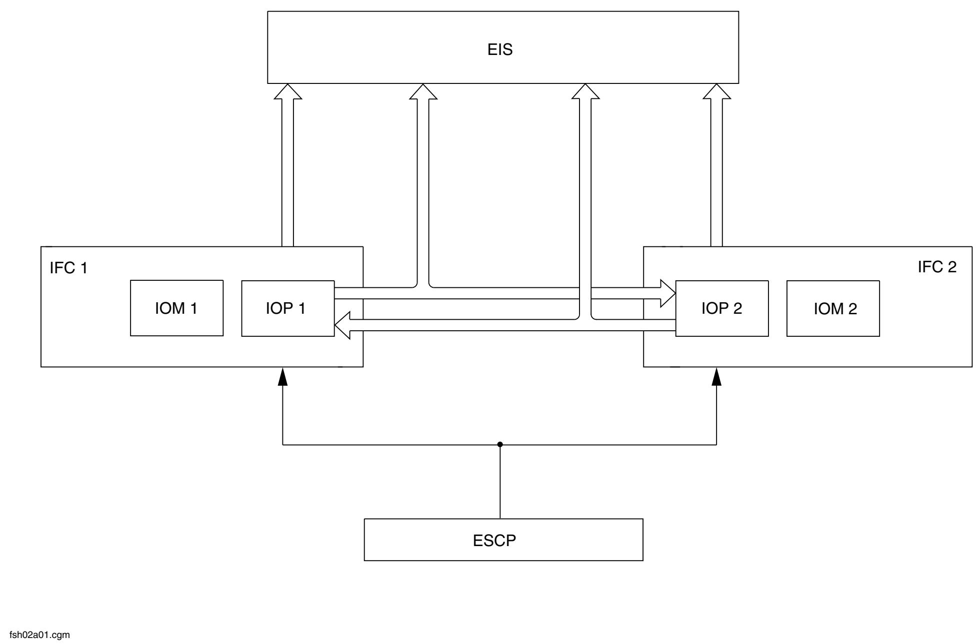
FDPS ICP CALCULATION
Figure 26



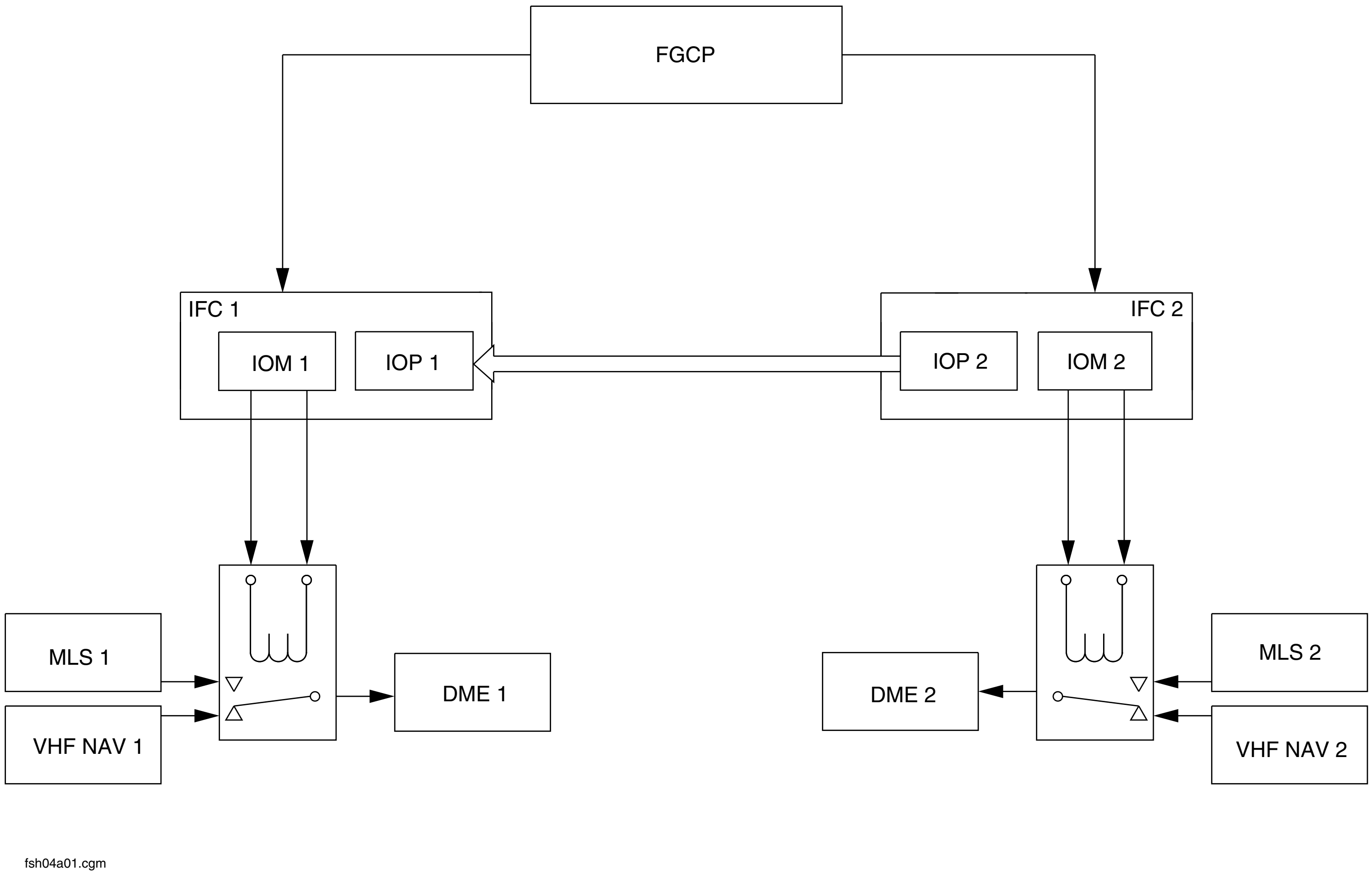
FDPS FLIGHT GUIDANCE CONTROL PANEL (FGCP) CALCULATION
Figure 27



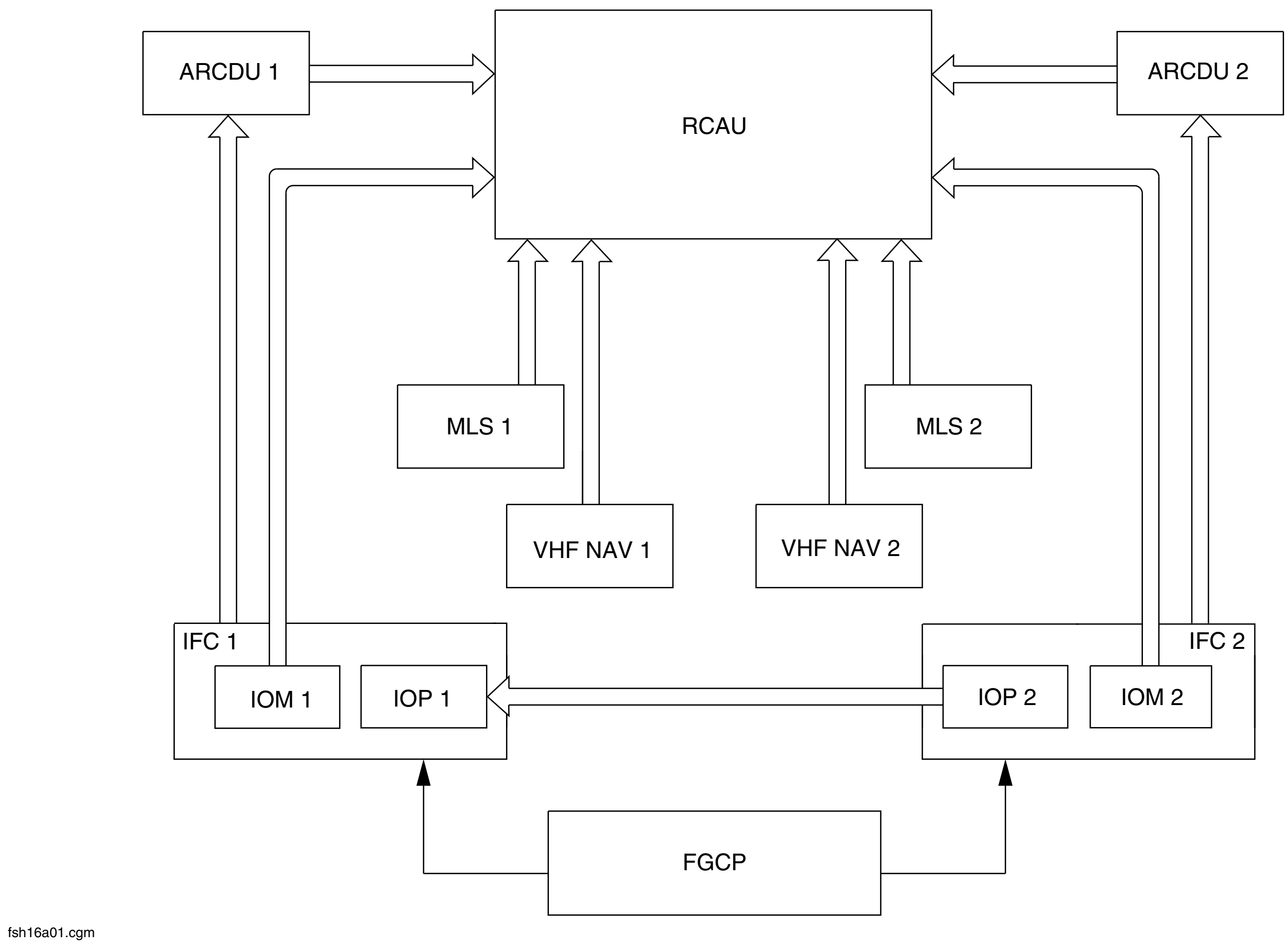
FDPS FGCP MONITORING
Figure 28



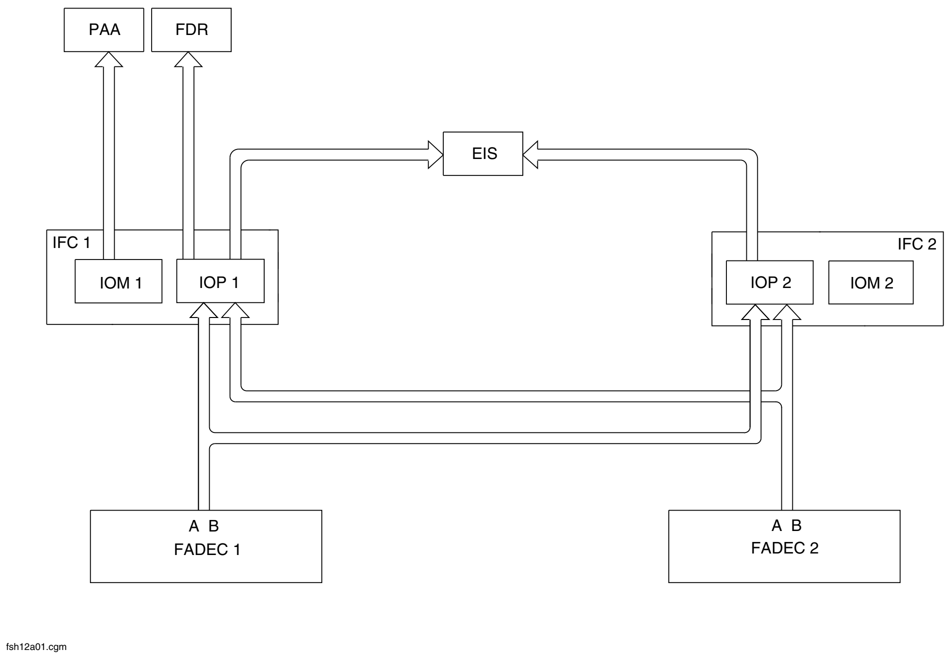
FDPS ESID CONTROL PANEL (ESCP) CALCULATION
Figure 29



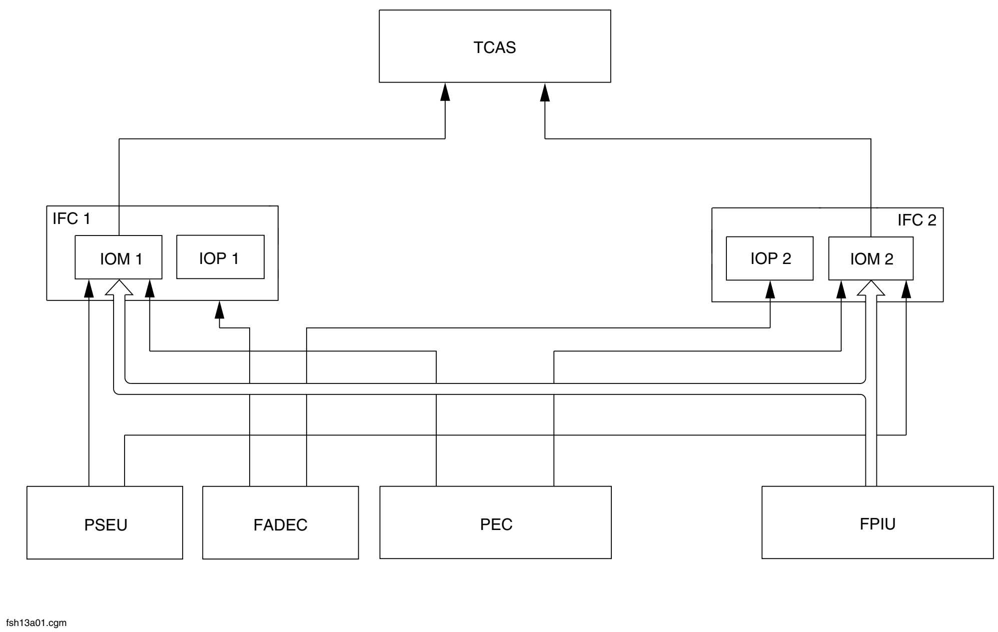
FDPS MLS/NAV TUNE SELECTION CALCULATION
Figure 30



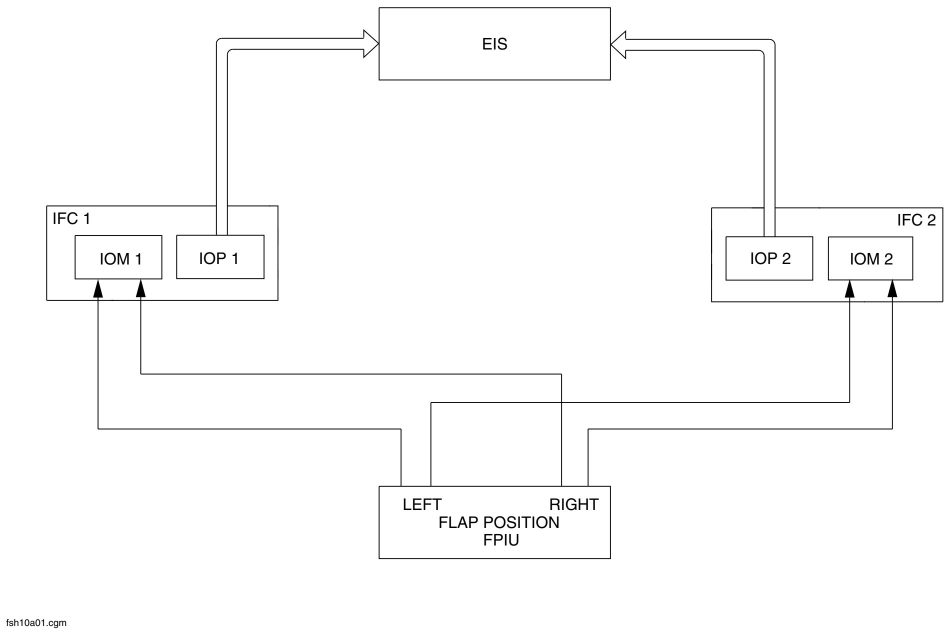
VOR/MLS AUDIO SELECTION CALCULATION
Figure 31



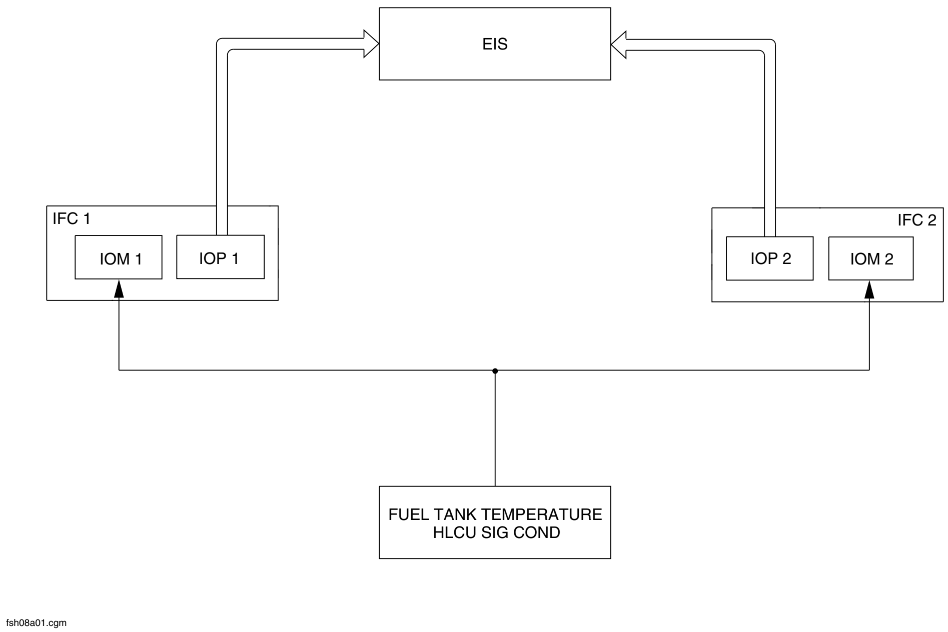
FDPS ENGINES OPERATING/STOPPED CALCULATION
Figure 32



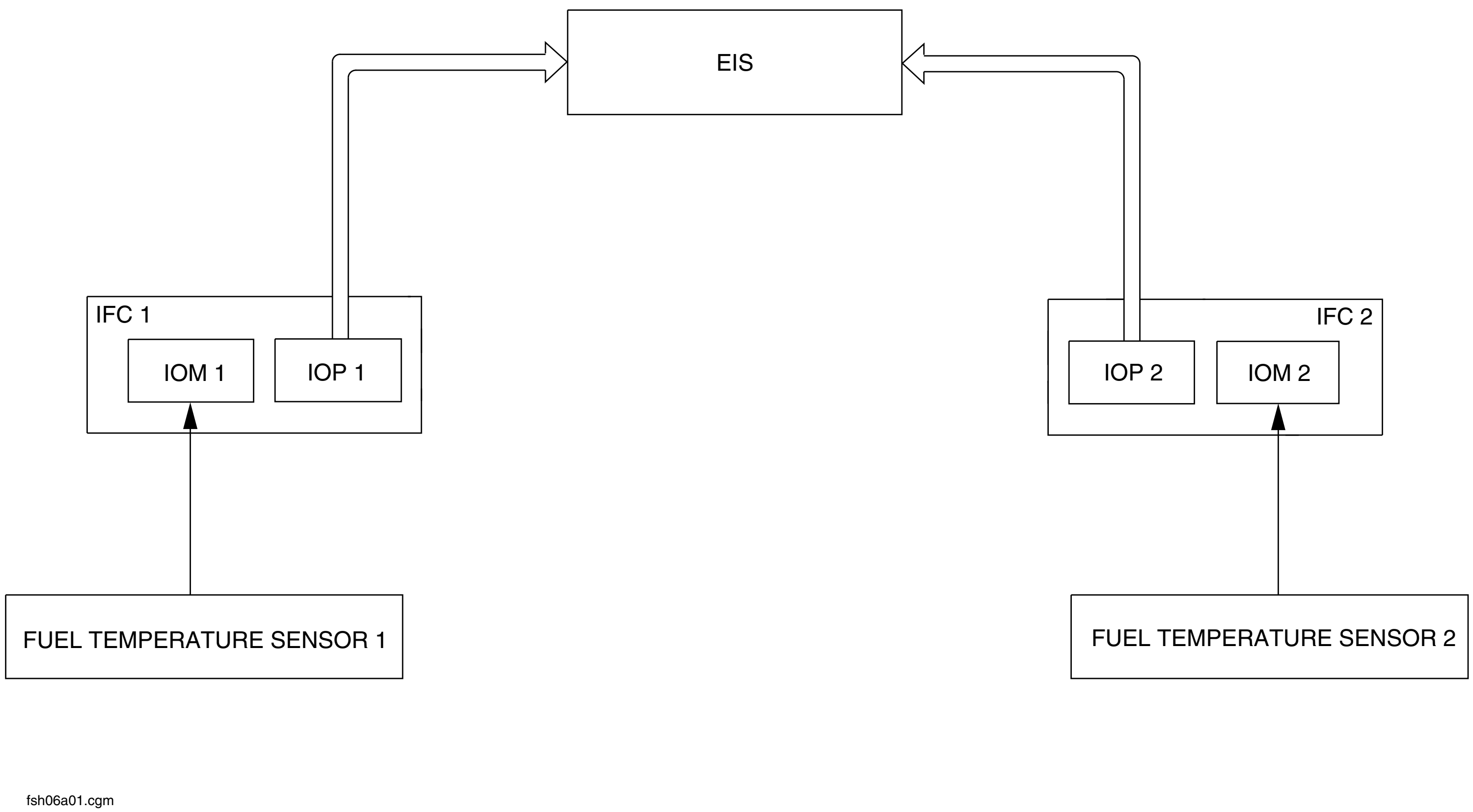
FDPS TCAS MANEUVER INHIBIT CALCULATION
Figure 33



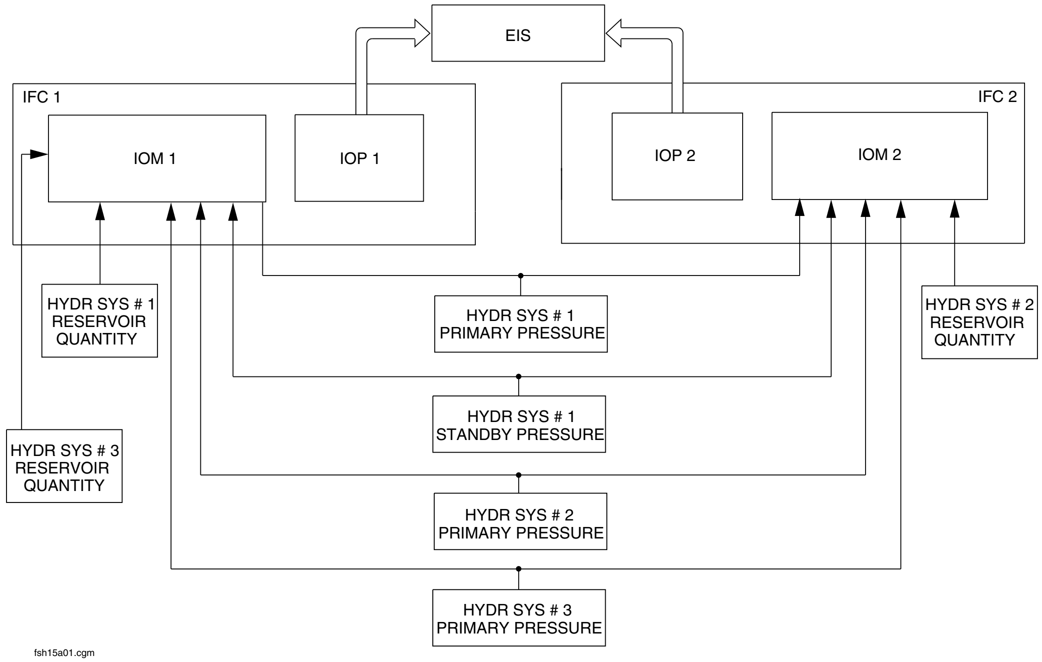
FDPS FLAP POSITION CALCULATION
Figure 34



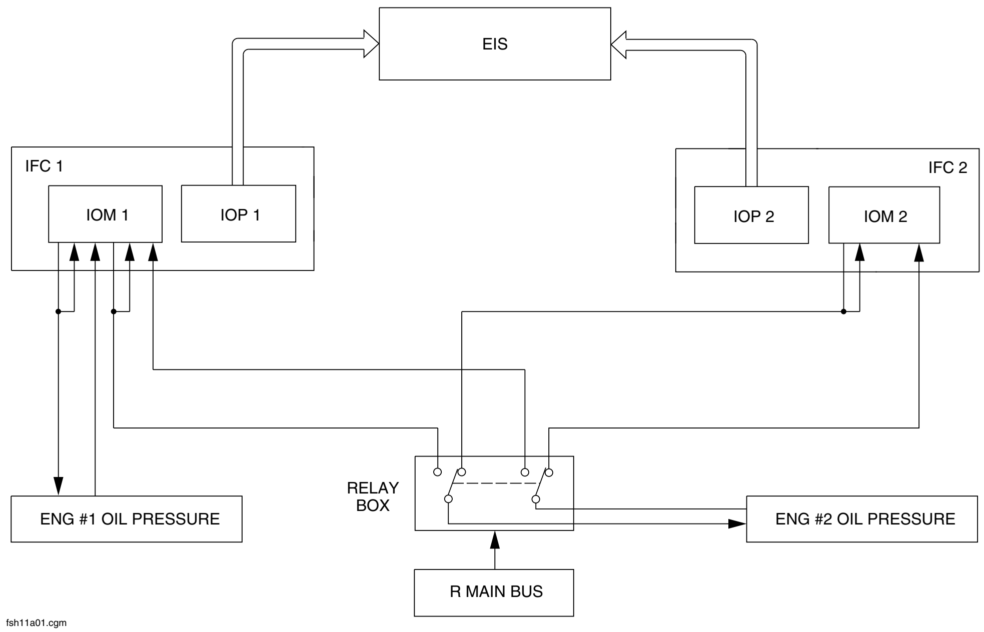
FDPS FUEL TANK TEMPERATURE CALCULATION
Figure 35



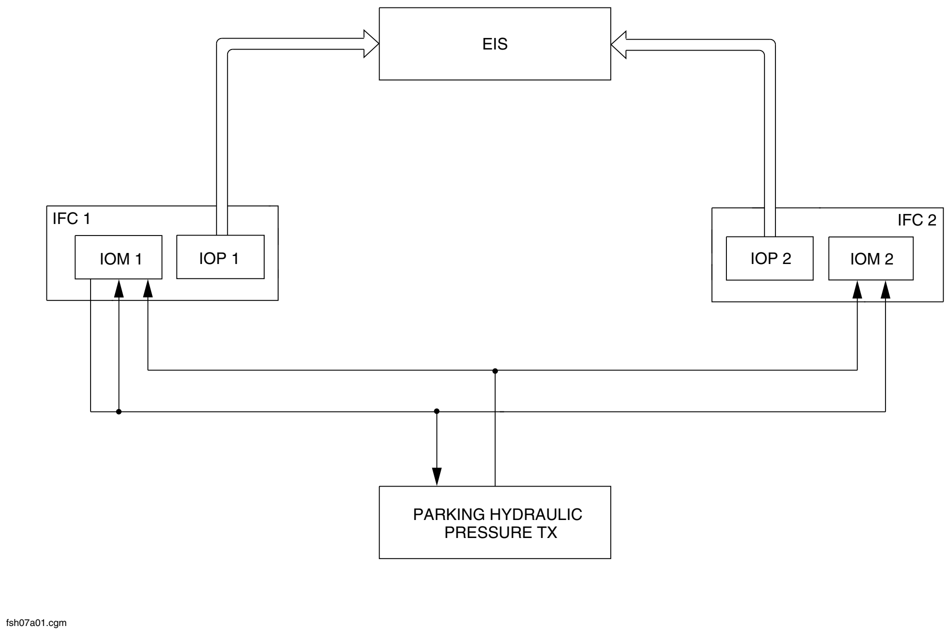
FDPS FUEL INLET TEMPERATURE CALCULATION
Figure 36



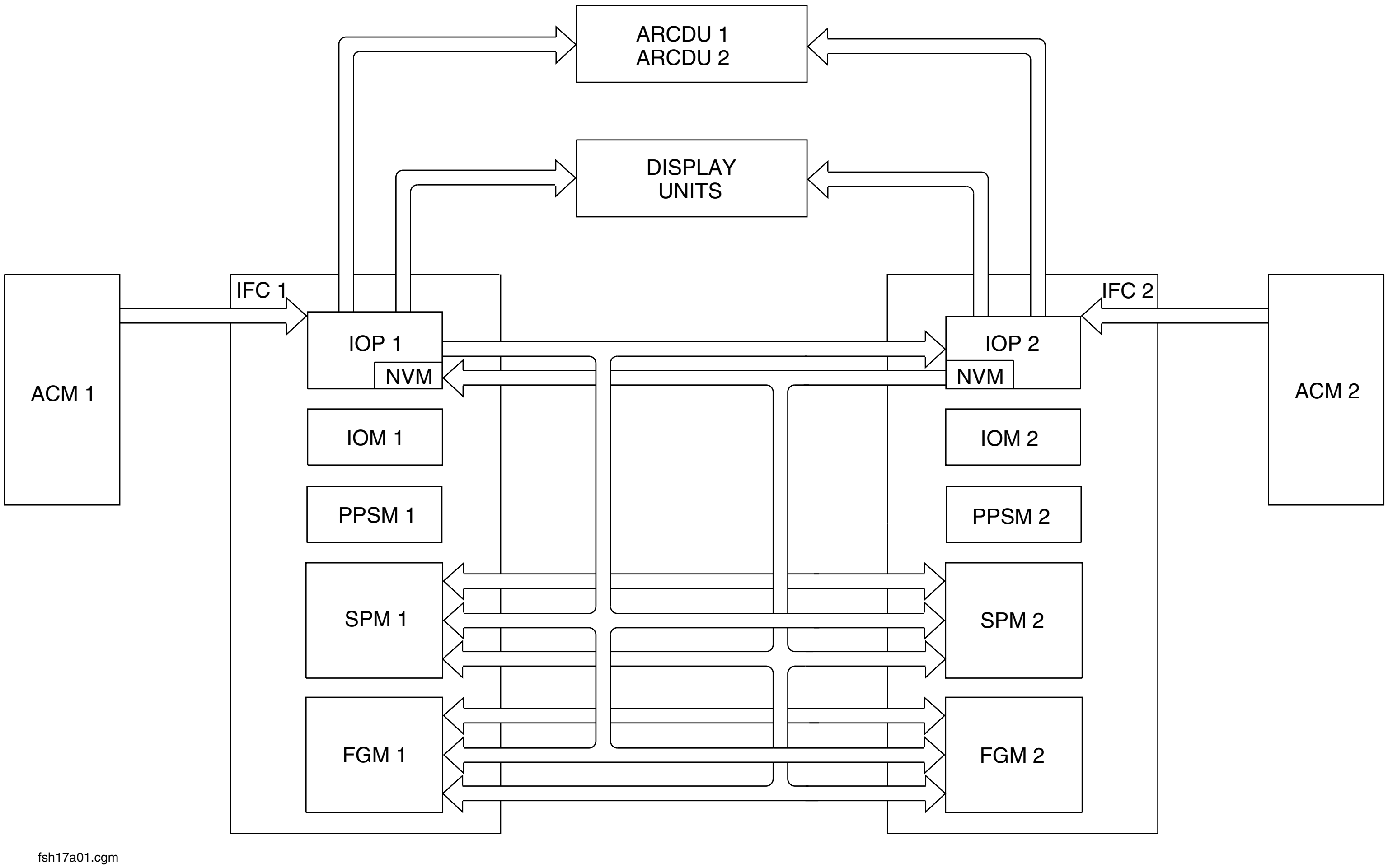
FDPS HYDRAULIC PARAMETERS CALCULATION
Figure 37



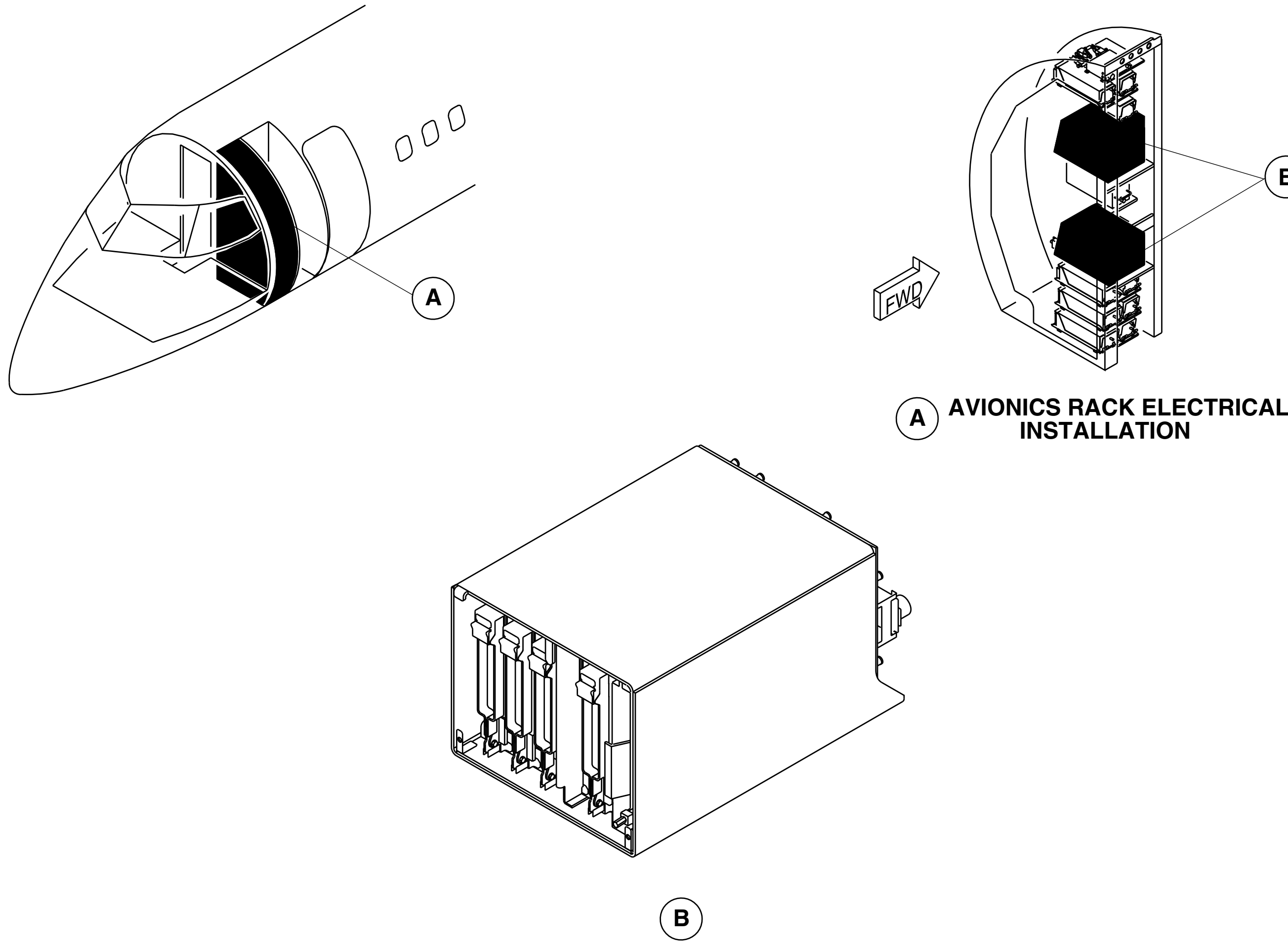
FDPS MAIN OIL PRESSURE (MOP) CALCULATION
Figure 38



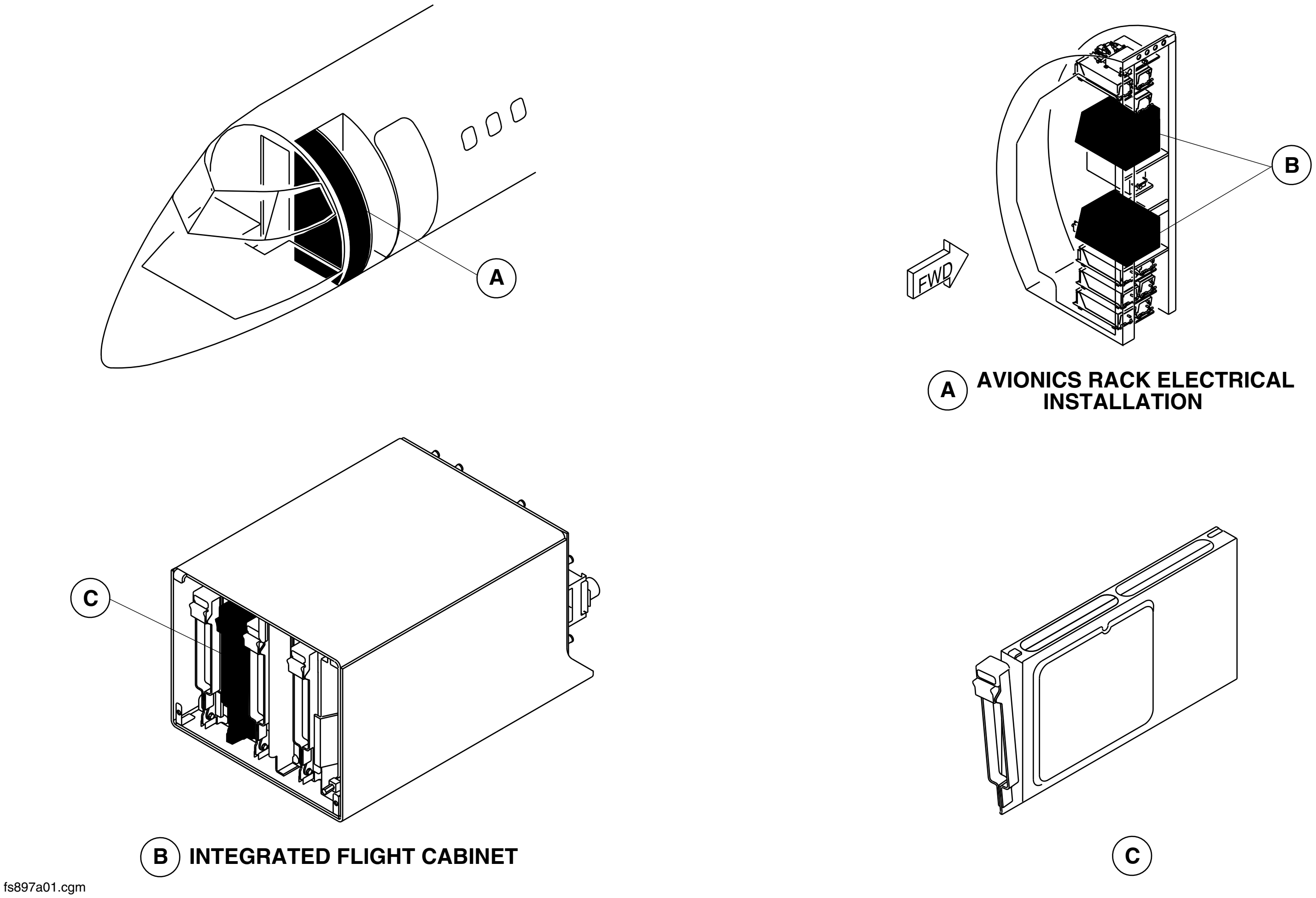
FDPS PARKING BRAKE HYDRAULIC PRESSURE CALCULATION
Figure 39



FDPS AIRCRAFT CONFIGURATION MANAGEMENT
Figure 40



INTEGRATED FLIGHT CABINETS, LOCATOR
Figure 41



INPUT/OUTPUT PROCESSORS, LOCATOR
Figure 42

31–40–00–001

CENTRAL COMPUTER (ALTERNATE SYMBOLOGY)

Introduction

The central computer has a Flight Data Processing System (FDPS) to receive data from sensors and avionics systems and supply it to other systems. The Flight Data Processing System (FDPS) also causes different warning tones to sound if an important system has malfunctioned or if the aircraft is in a dangerous condition.

General Description

Refer to Figures 43 and 44.

The two Flight Data Processing System (FDPS1, FDPS2) do the functions that follow:

- Receives and calculates aircraft and avionics parameters for the Flight Data Recorder (FDR)
- Receives data from different avionics and aircraft systems and routes their parameters to a other systems through a single output
- Receives analogue, discrete, and digital data from other systems and changes it to ARINC 429 format
- Mismatch message calculations
- Monitors the Electronic Instrument System (EIS)
- Message generation
- Aircraft Configuration Management (ACM)
- Supplies Warning Tones (WTG 1, WTG 2) and manages its priority
- Maintenance tests
- Software teleloading.

Each Flight Data Processing System (FDPS 1, FDPS 2) has the modules that follow:

- Integrated Flight Cabinet (31–41–00)
- Input/Output Processor (31–41–06)
- Input/Output Module (31–41–11)
- Prime Power Supply Module (31–41–16)
- Aircraft Configuration Module (31–41–21).

The modules are located in two Integrated Flight Cabinets (IFC 1, IFC 2) installed in the Avionics rack.

Detailed Description

The Flight Data Processing System (FDPS 1, FDPS 2) functions in the modes that follow:

- Initialization (INIT)
- Power–On Self Test (POST)
- Operational (OPER)
- Maintenance
- Teleloading.

Initialization (INIT) State: The FDPS functions in the initialization mode after an electrical power interruption. It initializes the hardware and software when it is on the ground.

If the FDPS does not receive a maintenance or teleloading request, it will then function in the Power−On Self Test (POST) mode.

Power−On Self Test (POST): The FDPS does a (POST) to make sure the hardware operates correctly before starting the operational mode.

The FDPS does a POST when the aircraft is on the ground and there is a long electrical power interruption that continues for more than 200 milliseconds.

The FDPS tests the interfaces that follow:

- Input/Output Processor (IOP 1, IOP 2)
- Input/Output Module (IOM 1, IOM 2)
- Ground Proximity Warning System Converter.

The result of the POST is stored in the Built In Test Equipment (BITE) and sent to the Central Diagnostic System (CDS).

The POST checks the parameters that follows:

- Program memory
- Data memory
- Watch dog
- Power supplies Hardware (HW) monitoring
- Module Validity HW and Software (SW) logic
- Memory partitioning HW monitoring
- Time partitioning HW monitoring.

If a parameter malfunctions, it will causes the FDPS to malfunction.

The POST also checks the parameters that follow:

- ARINC 429 inputs and outputs
- ARINC 422 inputs and outputs
- ARINC 717 inputs and outputs
- Discrete inputs and outputs
- IO1, IO2 board validity
- Analogue inputs and outputs
- Discrete inputs and outputs.

They do not cause a FDPS malfunction.

The POST sequence continues for 25 seconds.

Operational State: The FDPS does the functions that follow:

- Flight Data Concentrator (FDC)
- Data Hub Concentrator (DHC)
- Data Control (DCO)
- Mismatch calculations
- Electronic Instrument System (EIS) essential monitoring
- Advisory message generation
- Aircraft configuration management
- Warning Tone Generator (WTG 1, WTG 2).

Flight Data Concentrator (FDC): The FDC is located in IOP1. It receives data in different formats from the avionics and other aircraft systems and supplies digital data to the Solid State Flight Data Recorder (SSFDR) through an ARINC 573/717 bus.

Data Hub Concentrator (DHC): The FDPS receives noncritical data from avionics and other aircraft systems and supplies their parameters to other systems through a single output.

The FDPS supplies the concentrated parameters through ARINC 429 data buses to the avionics systems that follow:

- Electronic Instruments System (EIS)
- Flight Guidance Module (FGM 1, FGM 2)
- Stall Protection Module (SPM 1, SPM 2)
- Audio and Radio Control Display Units (ARCDU 1, ARCDU 2)
- Traffic Collision Avoidance System (TCAS)
- Ground Proximity Warning System (GPWS)
- Weather Radar (WXR)
- Flight Management System (FMS 1, FMS 2).

Two specific external non–avionics General Purpose Data Buses (GPDB 1, GPDB 2) are used to transmit data to systems that are not part of the avionics suite.

Data Control (DCO): The Data Control (DCO) makes calculations for the parameters before concentration.

Refer to Figure 45.

Mismatch Calculations: The FDPS monitors different parameters. If an IOP senses a difference between its related value and the same the parameters it received from the other IOP, a mismatch message is shown by the Flight Mode Annunciators (FMA) in the opposite Primary Flight Displays (PFD).

The Flight Data Processing System (FDPS) Mismatch Messages that follow:

- PITCH MISMATCH
- HEADING MISMATCH
- IAS MISMATCH
- ALT MISMATCH
- RAD ALT MISMATCH
- GS MISMATCH
- LOC MISMATCH.

The calculations are done only when the parameters are valid. For localizer and glideslope mismatch messages, the two navigation receivers outputs must be valid and tuned to the same frequency.

Electronic Instrument System (EIS) Essential monitoring: The FDPS does essential monitoring for the systems that follow:

- Radio Altimeter
- Localizer deviation
- Glideslope deviation.

The IOPs monitors the difference between a parameter received directly and same parameter received from the opposite PFD and IOP. When the difference is more than the predetermined value, the FDPS causes to the PFD to show a malfunction condition for that parameter.

Refer to Figure 46.

The essential monitoring parameter thresholds are shown in the table that follows:

Parameter	Difference
AGL altitude, less than 1000 ft	20 ft
LOC deviation, between 50 and 1000 ft AGL	2/3 of Expanded LOC Scale
G/S deviation, between 50 and 1000 ft AGL	1/6 of G/S Scale

Refer to Figure 47.
Message Generation: The Message Generation System has the modes that follow:

- Advisory
- Flight Mode Annunciator (FMA).

Advisory: The advisory messages are shown in white letters on the Engine Display (ED). Each message has a location and is shown while the condition is present. There are two kinds of advisory messages as follows:

- First Family
- Second Family.

First Family: The first family messages relate to an applicable aircraft status, where crew awareness is required and an action may be necessary. These messages are located near its related indication.

Second Family: Second family messages relate to minor malfunctions and are located at the bottom of the ED. The messages relate to passive malfunctions do not affect a continued safe flight.

The FDPS causes the ED to show the advisory messages that follow:

- [BALANCE]
- INCR REF SPEEDS
- ICE DETECTED
- IFC messages
- Display messages.

NOTE

Note: The advisory messages are shown in order of decreasing importance. The most important IFC message is shown at the bottom left part of the ED and the most important display message is shown at the bottom right part of the ED.

[BALANCE] Message: The BALANCE message comes into view flashing for five seconds then stays on steady when a fuel imbalance is sensed by the left or right fuel gauging computers.

INCR REF SPEEDS Message: The INCR REF SPEEDS message comes into view when the Stall Protection System (SPS) stall sensing is changed for icing conditions. The FDPS sends a discrete data signal to Electronic Instruments System (EIS) from one or the other SPM.

ICE DETECTED Message: The ICE DETECTED message comes into view when one ice detector probe or the other senses ice accumulation.

IFC Messages: The most important IFC message is shown at the bottom left part of the ED. It shows the messages that follow:

- IOP1 FAIL, IOP2 FAIL, IOPS FAIL
- IOP BAD CONF
- IOM1 FAIL, IOM2 FAIL, IOMS FAIL
- WTG1 FAIL, WTG2 FAIL, WTGS FAIL
- GPWSC I/F FAIL
- WOW/IOP1 FAIL, WOW/IOP2 FAIL, WOW/IOPS FAIL
- IFC ACM1 FAIL, IFC ACM2 FAIL, IFC ACMS FAIL
- RA1 FAIL, RA2 FAIL, RAS FAIL.

IOP1 FAIL, IOP2 FAIL, IOPS FAIL: The IOP FAIL messages are shown when the IOP1 or IOP2 or the interface to the ED has malfunctioned.

IOP BAD CONF: The IOP BAD CONF message is shown when a bad aircraft configuration condition is sensed by one IOP or the other. The message comes into view when the aircraft is on the ground.

IOM1 FAIL, IOM2 FAIL, IOMS FAIL: The IOM FAIL messages are shown when IOM malfunctions are sensed.

WTG1 FAIL, WTG2 FAIL, WTGS FAIL: The WTG FAIL messages are shown when the WTG malfunctions. The message is shown when the aircraft is on the ground and engines are not running.

GPWSC I/F FAIL: The GPWSC I/F FAIL messages are shown when the IFC1 makes the GPWS inoperative when it malfunctions. The message is when the malfunction is sensed.

WOW/IOP1 FAIL, WOW/IOP2 FAIL, WOW/IOPS FAIL: The WOW/IOP FAIL messages are shown when the IOP senses a difference between the main and nose WOW signals from the PSEU. The IOP will not be able to do a POST after a power interruption. The message is shown when the aircraft is on the ground and engines are not running.

IFC ACM1 FAIL, IFC ACM2 FAIL, IFC ACMS FAIL: The IFC ACM FAIL messages are shown when a ACM malfunction is detected. A single ACM malfunction has no effect on the aircraft. When the two ACM malfunction, a bad DU BAD CONF messages is shown.

RA1 FAIL, RA2 FAIL, RAS FAIL: The RA FAIL message is shown when a dual Radar Altimeter system is installed and the RA malfunction is detected for more that ten seconds by the EIS ED. The message is displayed any time the malfunction is detected.

Display Messages: The most important display message is shown at the bottom right part of the ED. The FANS FAIL message is the only FDPS related display message indication. It comes into view when 2 or more avionics cooling fans do not operate. The FANS FAIL display message also comes into view when 2 or more avionics cooling fans do not operate and they are not inhibited by their related thermal switch while the aircraft is on the ground. The temperature switch 1 supplies data to IOP1 and temperature switch 2 supplies data to IOP2.

An IFC or display advisory message also causes the AVIONICS caution light to come on 2 minutes after the aircraft is on the ground and the air speed is less than 50 kts.

Refer to Figure 48.
Aircraft Configuration Management: The FDPS uses data that is stored in the Aircraft Configuration Modules (ACM 1, ACM 2). The ACMs are attached to the back–panel of its related IFC1. Each IOP supplies the configuration data to the systems that follow:

– Electronic Instrument System (EIS)
– Stall Protection System (SPS), Stall Protection Modules (SPM 1, SPM 2)
– Auto Flight Control System (AFCS), Flight Guidance Modules (FGM 1, FGM 2)
– Audio and Radio Control Display Units (ARCDU 1, ARCDU 2)

Each system monitors the configuration data. If a malfunction is sensed, it is stored in the Built In Test Equipment (BITE) and sent to the Central Diagnostics System (CDS) and the ED shows an IFC message.

The Aircraft Configuration Modules (ACM 1, ACM 2) are programmed using the Portable Multipurpose Access Terminal (PMAT).

Refer to Figure 49.
Warning Tone Generators (WTG 1, WTG 2): The Warning Tone Generators (WTG 1, WTG 2) give different tones to tell the pilots of dangerous conditions or system malfunctions.

The tone that sounds depends on its priority. The most important tone always sound as follows:

PRIORITY	TONE	DESCRIPTION	SIGNAL TYPE	LEVEL, VOLTS
1	GPWS	Voice	N/A	N/A
2	TCAS, RA	Voice	N/A	N/A
3	Fire	Continuous chime until manually cancelled	A	4.8
4	Incorrect take off configuration	1000 Hz intermittent tone until cause of condition is removed	C	3.5
5	Autopilot disengage	250 Hz Intermittent tone for 1.5 seconds (manual) or until manually cancelled (automatic)	F	3.8
6	Pitch trim in motion	Continuous clicking until cause of condition is removed	D	4.2
7	Overspeed	1000 Hz intermittent tone until speed decreased	C	3.1

PRIORITY	TONE	DESCRIPTION	SIGNAL TYPE	LEVEL, VOLTS
8	Incorrect landing gear configuration	Continuous 800 Hz tone until cause of condition is removed	B	1.8
9	Altitude alert	2900 Hz tone for 1 seconds	B	2.9
10	Beta lockout warning	Continuous 1900 Hz to 2900 tone until cause of condition is removed	G	5.2
11	Warning annunciation	3 1000 Hz chimes	E	3.8
12	Caution annunciation	1 1000 Hz chime	E	5.0
13	TCAS, TA	Voice	N/A	N/A
14	SELCAL	1200 Hz tone for 3 seconds	N/A	2.7

NOTE

Note: The WTGs generate the aural tones and control the priority of the voice sound from the GPWS and TCAS.

Each type of signal modulation, is shown as follows:

Refer to Figure 50.
Aural warning type A

Refer to Figure 51.
Aural warning type B

Refer to Figure 52.
Aural warning type C

Refer to Figure 53.
Aural warning type D

Refer to Figure 54.
Aural warning type E

Refer to Figure 55.
Aural warning type F

Refer to Figure 56.
Aural warning type G

Refer to Figure 57.
Aural warning type H.

The WTG is not an independent system. It is part of the Input/Output Modules' (IOM1, IOM2) function. One WTG supplies a warning tone to the Remote Control Audio Unit (RCAU) and other monitors it. The RCAU amplifies the signal and sends the tone to the flight compartment speakers and the pilots headsets.

The WTG functions with a visual indication. It gives the different tones that follow:

– GPWS
– TCAS
– Engine fire
– Incorrect takeoff warning
– Autopilot disengagement

– Pitch trim in motion
– Overspeed warning
– Incorrect landing gear configuration
– Altitude alert
– Beta lockout warning
– Master warning tone
– Master caution tone
– SELCAL.

GPWS: The GPWS is a system that makes its own synthesized voice sounds and connects directly to the Audio Integrating System (AIS).

When the WTG senses a GPWS audio on condition, it inhibits the other tones.

The voice sounds from the GPWS is inhibited if a GPWS malfunction is sensed while the aircraft is airborne. The GPWS audio on signal is monitored to prevent an inhibit of a different WTG tone caused by a partial GPWS malfunction. If the GPWS audio on signal stays on for more than 60 seconds, the GPWS's priority status is ignored. A GPWS and a TCAS or WTG tone is allowed to be heard at the same time.

The GPWS malfunction condition and that the TCAS and WTG continues to function is easily identified by the pilots. To be able to sense this malfunction condition, the FDPS does not inhibit the GPWS during the WTG test (ADC1 or ADC2 TEST toggle switch selection).

TCAS: The TCAS is a system that makes its own synthesized voice sounds and connects directly to the Audio Integrating System (AIS). It has two types of alerts with two different priority levels that follows:

– Resolution advisories
– Traffic advisories.

Resolution Advisories: The resolution advisories are grouped as corrective advisories or preventive advisories. The vertical flight path of the aircraft must be changed when a corrective advisory condition is sensed. A preventive advisory is sensed to maintain the vertical flight path.

Traffic Advisories: The traffic advisories mode senses traffic advisory intruders only.

The WTG inhibits the TCAS voice sound when a more important tone sounds.

When the WTG senses a TCAS resolution advisory (corrective or preventive) or a traffic advisory signal, it inhibits less important tones.

The voice sounds from the TCAS is inhibited if a TCAS malfunction is sensed while the aircraft is airborne. The TCAS audio resolution advisory (corrective or preventive) and traffic advisory signals are monitored to prevent an inhibit of a less important WTG tone caused by a partial TCAS malfunction. If a TCAS signal stays on for more than 60 seconds, the TCAS's priority status is ignored. A TCAS and a GPWS or WTG tone is allowed to be heard at the same time.

The TCAS malfunction condition and that the GPWS and WTG continues to function is easily identified by the pilots. To be able to sense this malfunction condition, the FDPS does not inhibit the TCAS during the WTG test (ADC1 or ADC2 TEST toggle switch selection).

Engine Fire: The WTG inhibits the engine fire aural warnings when a more important tone sounds.

A continuous chime sounds when the WTG senses a fire bell discrete signal from the fire detection system.

Refer to Figure 58.
Incorrect Take–Off Warning: The WTG inhibits the incorrect take–off warning indication when more a more important tone sounds.

The WTG sounds an 1000 Hz Intermittent tone when the aircraft is not set to a safe for take–off configuration. It functions when the two engine torque indications are more than 50% or the TOCW maintenance switch is set to TPCW and the one or more of parameters is sensed as follows:

- Takeoff warning test
- ENG 1 torque more than 50%
- ENG 2 torque more than 50%
- Flaps set to more than 20
- CLA 1 not set to 1020
- CLA 2 not set to 1020
- Parking brake set
- Inboard spoilers extended
- Outboard spoiler extended
- Left elevator trim out of range
- Right elevator trim out of range.

The take off warning system has the interfaces that follow:

Parameter	Source
T/O WARN MAINT	T/O WARN TEST Toggle switch
ENG1 TORQUE > 50%	FADEC
ENG2 TORQUE > 50%	FADEC
NGWOFW	PESU
FLAPS ≤ 13.5°	FPIU
CLA1 ≠ MAX/1020	FADEC
CLA2 ≠ MAX/1020	FADEC
PARKING BRAKE ON	PARKING BRAKE LEVER
INBOARD SPOILERS EXTENDED	FCS ECU LEFT CHANNEL
LEFT ELEVATOR TRIM OUT OF RANGE	FCS ECU LEFT CHANNEL
OUTBOARD SPOILERS EXTENDED	FCS ECU RIGHT CHANNEL
RIGHT ELEVATOR TRIM OUT OF RANGE	FCS ECU RIGHT CHANNEL

NOTE

Note: A malfunctioned ARINC parameter causes it to set.

Autopilot Disengagement: The WTG inhibits the autopilot disengagement tones when a more important tone sounds.

The WTG sounds an intermittent tone for 1.5 seconds for a manual AFCS disengagement. It sounds a 4000 Hz Intermittent tone when the Auto Flight Control System (AFCS) automatically disconnects until it is cancelled.

The two Flight Guidance Modules (FGM1, FGM2) supply Autopilot (AP) engagement/disengagement data through data buses to the WTGs.

The autopilot disengagement tones sound if one FGM or the other supplies a disengagement signal to the WTG.

Pitch Trim in Motion: The WTG inhibits the pitch trim in motion tones when a more important tone sounds.

The WTG sounds a clicking tone when the pitch trim is in motion for more than 3 seconds.

The two Flight Control Electronic Control Units (FC ECU1, FC ECU2) supply pitch trim in motion data through data buses to the WTGs.

The clicking tone sounds if one FCS ECU or the other supplies a pitch trim in motion signal to the WTG.

Overspeed Warning: The WTG inhibits the overspeed warning tones when a more important tone sounds.

The WTG sounds an intermittent 1000 Hz tone when the aircraft's speed is more than Maximum Velocity in Operation (VMO).

The two Air Data Units (ADU1, ADU2) supply overspeed data through data buses to the WTGs.

The overspeed tone sounds if one ADU or the other supplies an overspeed signal to the WTG.

Refer to Figure 59.

Incorrect Landing Gear Configuration: The WTG inhibits the incorrect landing gear configuration tones when a more important tone sounds.

The WTG sounds a continuous 800 Hz tone when the landing gear is not set to a safe for landing configuration.

It functions when the HORN toggle switch on the landing gear panel is set to the TEST position or if the gears are not down and locked one of the three different conditions are sensed as follows:

Condition 1 (Flaps and Engine torque):

- The flap lever is set to more than 8.5 degree

  AND

- ENG1 torque OR ENG2 torque is less than 50%

  AND

- PLA 1 AND PLA 2 are not in the rating power detent.

Condition 2 (Airspeed, Radio Altitude and PLA setting):

- Indicated airspeed less than 156 Knots (288 km/h), AND RAD ALT 1 invalid

  OR

- Indicated airspeed less than 156 Knots (288 km/h), AND RAD ALT 1 valid, AND Above Ground Level (AGL) altitude is less than 1053 feet

  AND

- PLA1 AND PLA2 are less than FLIGHT IDLE plus 12 degrees.

Condition 3 (Airspeed, single engine operation and PLA setting):

- Indicated airspeed less than 156 Knots (288 km/h), AND RAD ALT 1 invalid

  OR

- Indicated airspeed less than 156 Knots (288 km/h), AND RAD ALT 1 valid, AND Above Ground Level (AGL) altitude is less than 1053 feet

  AND

- PLA1 OR PLA2 is less than FLIGHT IDLE plus 12 degrees

  AND

- HORN MUTE switch is not latched to MUTE

  AND

- PLA 1 AND PLA 2 are less than the rating power detent.

The landing gear configuration warning system has interfaces with other aircraft systems as follows:

Signal	Source
GEARS DOWN AND LOCKED 1	PSEU
GEARS DOWN AND LOCKED 2	PSEU
IAS < 156 ±5 Knots	ADU1, ADU2
AG L< 1053 feet	RA1
PLA 1 < FLIGHT IDLE + 12 degree	FADEC
PLA 2< FLIGHT IDLE + 12 degree	FADEC
HORN MUTE	LANDING GEAR CONTROL PANEL
FLAP LEVER > 8.5°	FCU
ENG 1 TORQUE < 50%	FADEC
ENG 2 TORQUE < 50%	FADEC
PLA1 ≥ RATING POWER DETENT	FADEC
PLA2 ≥ RATING POWER DETENT	FADEC
LDG WARN TEST	LANDING GEAR CONTROL PANEL

Refer to Figure 60.
Altitude Alert: The WTG inhibits the altitude alert tones when a more important tone sounds.

The WTG sounds a continuous 2900 Hz tone for 1 second when the aircraft enters an area that is less than 1000 ft (305 m) above or below a set altitude.

Refer to Figure 61.
The ALT (altitude) selector rotary knob on the Flight Guidance Control Panel is used to set a cyan preselected altitude readout and bug indication on the PFDs.

Refer to Figure 62.
The range of the preselected altitude readout is between 0 and 99,900 ft. When the preselected altitude readout is set to less than zero feet, the indication goes out of view and if the value is set to more than the shown altitude scale, one half of the indication is shown at the edge of the altitude indicator.

When the aircraft is 1000 ft (305m) above or below the preselected altitude, a tone sounds for 1 second and the indication changes to yellow until the aircraft is less than 250 ft from the preselected altitude. If the aircraft deviates from the preselected altitude by more than 250 ft, a tone sounds for 1 second and the indication changes to yellow again. The indication changes back when the aircraft is less than 250 ft above or below the preselected altitude value.

If the aircraft does not return to the preselected altitude, a new value must be set to reset the indication.

The Flight Data Processing System (FDPS) senses the barometric corrected altitude from ADU1 or ADU2 and the set altitude from the FGCP. It uses ADU1 if the HSI SEL pushbutton switch on the FGCP is set to left and ADU2 when set to right.

Refer to Figure 63.
Beta lockout warning: The WTG inhibits the beta lockout alerts when a more important tone sounds.

The Warning Tone Generator (WTG 1, WTG 2) sounds a continuous 800 Hz tone when the Power Lever Angle (PLA) is set below the IDLE position while in flight. To give a beta lockout indication, the Flight Data Processing System (FDPS) senses the parameters that follow :

– Beta lockout switch position
– Power lever angle
– Main landing gear WOW.

Master Warning: The WTG inhibits the master warning tones when a more important tone sounds. The WTG sounds three chimes when one or the other red master warning light comes on.

Master Caution: The WTG inhibits the master caution tones when a more important tone sounds. The WTG sounds a single chime when one or the other amber master caution light comes on.

SELCAL: The WTG inhibits the SELCAL tone when a more important tone sounds. A continuous 1200 Hz tone sound for three seconds when the Selective Calling (SELCAL) system senses an incoming call.

There Warning Tone Generators (WTG 1, WTG 2). The WTG1 sounds the applicable tone when necessary while WTG2 functions in the standby mode. The WTG2 functions only when it senses that WTG1 has malfunctioned.

The WTG receives inputs from other systems to make it operate.

Each WTG monitors its output. If the WTGs calculate different tones, the system will use the calculations from WTG 1. Its calculations are more important than WTG 2.

The WTGs does the monitoring that follows:

– Software
– Hardware.

Software Monitor: Each Warning Tone Generator (WTG 1, WTG 2) uses their related inputs to calculate the tone logic. The tone calculations are compared and the WTG 1 supplies the applicable tone.

Hardware Monitor: Each Input/Output Module (IOM 1, IOM 2) has two Input/Output boards (IO1, IO2). The WTG 1 sends an analogue tone to the Remote Control Audio Unit (RCAU) through IO2. It routes the same signal to IO1 to make sure that the output tone is correct. If there is a difference, the WTG 1 stops functioning and makes itself invalid. Then, the WTG 2 functions when required.

The Warning Tone Generator (WTG 1, WTG 2) will remain off until the next power interruption when it is latched off because of a malfunction. The Power–On Self Test (POST) will determine its validity.

A Warning Tone Generator (WTG 1, WTG 2) malfunction is stored in the Built In Test Equipment (BITE) and sent to the Central Diagnostic System.

The Engine Display (ED) advisory message location will show an WTG FAIL message when a WTG malfunctions.

FDPS Abnormal Modes: Most aircraft and avionics system supplies data to other systems through the two FDPS's. Each aircraft or avionics system uses data from its related FDPS. If its related FDPS malfunctions, the systems will automatically receive data from the other FDPS.

The Flight Data Recorder (FDR) receives data through the Input/Output Processor (IOP 1) from the two FDPS's.

If FDPS1 malfunctions, the Flight Data Recorder (FDR) function also malfunctions. If FDPS 2 malfunctions, the FDR will not record its parameters.

The data is supplied only to the FDPS1 from the systems that follow:

– Hydraulic quantity 1
– Hydraulic quantity 3
– Fuel inlet temperature 1
– Parking brake pressure
– Main Oil Pressure 1
– Ground Proximity Warning System Converter.

If FDPS1 malfunctions, its parameters are shown as dashes on the Engine Display (ED).

The data is supplied only to the FDPS2 from the systems that follow:

– Hydraulic quantity 2
– Fuel inlet temperature 2
– Traffic Collision Avoidance System (TCAS).

If FDPS2 malfunctions, its parameters are shown as dashes on the Engine Display (ED).

If one FDPS or the other malfunctions, the Engine Display (ED) will also show an IFC message in the advisory message area.

Maintenance Mode: The Built In Test Equipment (BITE) uses the Central Diagnosis System (CDS) to give the condition of the component. It stores faults in a Non Volatile Memory (NVM) for reporting to line and shop maintenance.

The Built In Test Equipment (BITE) allows aircraft maintenance personnel to:

– Do fault isolation and return to service testing after completing maintenance actions
– Access failure reports from last or previous flight legs
– Get the avionics status report
– Get the part number of a given part

The Built In Test Equipment (BITE) modes monitors the condition of the component as follows:

- Power–On Self Test (POST)
- Continuous Monitoring.

Power–On Self Test (POST): The Power–On Self Test (POST) checks the condition of the component at Power–Up or after a long power interruption.

Continuous Monitoring: The Continuous Monitoring checks the status of the component in flight. It records faults in a Non Volatile Memory (NVM) for later troubleshooting using the Central Diagnosis System (CDS).

A long power interruption causes the FDPS to start a Power–On Self Test (POST) again.

Teleloading: The CDS is a communication connection between the FDPS and a Portable Multipurpose Access Terminal (PMAT). The PMAT connects to a Personal Computer (PC) to download a new software version when a software upgrade is necessary.

The CDS functions in the teleloading mode when the conditions that follow is correct:

- The Calibrated Air Speed (CAS) is less than 50 kts for more than 10 seconds
- The aircraft is on the ground
- The CDS GND MAINT toggle switch on maintenance panel is set
- The MAINT Key on the ARCDU is pushed.

The PMAT is connected through an RS422 data bus to each FDPS when software is downloaded. The CDS also transmits through an ARINC 429 data bus to the PMAT all "acknowledge" words from each FDPS during the software downloading function.

The FDPS does a software configuration check before teleloading mode starts. It makes sure that the software version to be teleloaded is consistent with the equipment part number. After the teleload, the POST mode is started within 10 seconds.

The left essential bus supplies electrical power through a 10 A circuit breaker and the Prime Power Supply Module (PPSM1) to the Input/Output Processor (IOP1) and Input/Output Module (IOM1). The circuit breaker is located in position F7 on the avionics circuit breaker panel. The left main bus supplies electrical power through a 7.5 A circuit breaker and the PPSM1 to the Stall Protection Module (SPM1). The circuit breaker is located in position F2 on the avionics circuit breaker panel.

The right main bus supplies electrical power through a 10 A circuit breaker and the Prime Power Supply Module (PPSM 2) to the Input/Output Processor (IOP2) and Input/Output Module (IOM2). The circuit breaker is located in position F6 on the avionics circuit breaker panel. The right main bus supplies electrical power through a 7.5 A circuit breaker and the PPSM2 to the SPM2. The circuit breaker is located in position F5 on the avionics circuit breaker panel.

Refer to Figure 64.
The FDPS receives data from the DME, ADF and MLS (option) systems and supplies it to the ARCDUs. It supplies the data as shown in the table that follows:

Parameter	Source
DME frequency	DME1, DME2
ADF frequencies	ADF1, ADF2
MLS channel	MLS1, MLS2
MLS selected azimuth angle	MLS1, MLS2
MLS maximum selected glidepath angle	MLS1, MLS2

Refer to Figure 65.
The FDPS receives data from different non important systems and supplies it to the EIS and the opposite IOP. It supplies the data as shown in the table that follows:

Parameter	Source
Selected Heading	FGCP
Selected Course	FGCP
Navigation source	FGCP
Selected DH	ICP1, ICP2
Speed Bugs	ICP1, ICP2
Rudder position	FCSECU1, FCSECU2
Elevator position	FCSECU1, FCSECU2
Spoiler position	FCSECU1, FCSECU2
MLS channel	MLS1, MLS2
MLS AZ/EL deviation	MLS1, MLS2
Radio Altitude	RA1, RA2
Fuel Quantity	FGC / FQC
Fuel discretes	FGC / FQC
Fuel Temperature	Fuel Temperature Sensor
Fuel Flow	From Fuel Flow Sensors
VOR/ILS frequencies	VHFNAV1, VHFNAV2
LOC/GS deviation	VHFNAV1, VHFNAV2
VOR bearing	VHFNAV1, VHFNAV2
Hydraulics Pressure	Hydraulics Sensor
Hydraulics Quantity	Hydraulics Sensor
Parking brake hydraulic pressure	Hydraulics Sensor
Air/Ground status	PSEU

Doors Status	PSEU
Main Oil Pressure	FADEC1, FADEC2
Engine torques	FADEC1, FADEC2
Flap position	FPIU
Static Air Temperature	ADU1, ADU2
ED Brightness	ESCP
System page status	ESCP
Reversion status	ESCP
APU generator volt	EPGDS
AC Bus Voltage	EPGDS
AC Bus Load	EPGDS
Generator Load	EPGDS
TRU Load	EPGDS
Secondary Bus Voltage	EPGDS
Battery Load	EPGDS
Essential Bus Voltage	EPGDS
Main Bus Voltage	EPGDS
Battery Temperature	EPGDS
External Power Discrete	EPGDS
frequency	DME1, DME2
Distance	DME1, DME2
Time To Station	DME1, DME2
Ground Speed	DME1, DME2
Display Control	TCAS
Vertical Resolution	TCAS
Output Sense Level and Reply Data	TCAS
Intruder range, altitude, bearing	TCAS
Failure status	TCAS
frequency	ADF1, ADF2
Bearing	ADF1, ADF2

Refer to Figure 66.
The FDPS receives data from the Attitude and Heading Reference System (AHRS) and the EFIS Control Panels through the EIS and

supplies it to the weather radar system. It supplies the data as shown in the table that follows:

Parameter	Source
Roll	AHRS1, AHRS2
Pitch	AHRS1, AHRS2
Magnetic Heading	AHRS1, AHRS2
WXR Range	ESCP

Refer to Figure 67.
The FDPS receives data from different systems and supplies it to other non avionic systems through a General Purpose Data bus. It supplies the data as shown in the table that follows:

Parameter	Source
VHFCOM Frequency	VHFCOM1, VHFCOM2
ADF Frequency	ADF1, ADF2
Selected Heading	FGCP
Selected Altitude	FGCP
Standard Pressure Altitude	ADU1, ADU2
Barometric Corrected Altitude	ADU1, ADU2
Mach	ADU1, ADU2
Calibrated airspeed	ADU1, ADU2
VMO	ADU1, ADU2
True airspeed	ADU1, ADU2
Total Air Temperature	ADU1, ADU2
Altitude Rate	ADU1, ADU2
Static Air Temperature	ADU1, ADU2
Hydraulic pressure	Hydraulic Sensor
HP compress Exit Pres (P3.0)	FADEC1, FADEC2
Propeller speed	FADEC1, FADEC2
Fuel flow sensor	FADEC1, FADEC2
Vertical Speed	AHRS1, AHRS2
Roll	AHRS1, AHRS2

Pitch	AHRS1, AHRS2
Magnetic Heading	AHRS1, AHRS2

Refer to Figure 68.
The FDPS receives a valid discrete from each EIS indication and sends it to the DUs.

Refer to Figure 69.
The FDPS1 calculates the data from Index Control Panel 1 (ICP1) and FDPS2 calculates data from ICP2. The ICPs supply the data to the FDPS's as shown in the table that follows:

Parameter	Source
Speed Bug Selector	ICP1, ICP2
Speed Bug Index Setting	ICP1, ICP2
Decision Height	ICP1, ICP2

The ICP1 barometric altitude setting is calculated by the ADC1 and ICP2 barometric altitude setting is calculated by the ADC2.

Refer to Figure 70.
The FDPS calculates data from the FGCP and supplies it to the EIS, FGM and opposite IOP that follows:

Parameter	Source
Heading Selection	FGCP
Course Selection	FGCP
Attitude selection	FGCP
HSI SEL	FGCP
Navigation Source	FGCP

The IOP, when valid, calculates the parameters from the FGCP and transmits its result through IOP2 to the EIS.

If the data connection between IOP1 and the FGCP malfunctions, IOP1 also calculates parameters from the FGCP through IOP2.

If IOP1 malfunctions, the FDPS continues to operate with calculations from IOP2 and with parameters previously calculated by IOP1.

Refer to Figure 71.
The FDPS1 calculates with IOP1 with data from FGCP and sends the result to FDPS2. The FDPS sends the data to the EIS. When the link malfunctions between IOP1 and the FGCP, the FDPS1 will calculate data from the FGCP through FDPS2.

Refer to Figure 72.
The FDPS1 calculates the data from the ESID Control Panel (ESCP). It supplies the data to the FDPS's as shown in the table that follows:

Parameter	Source
ED Brightness	ESCP
MFD1 Reversion	ESCP
MFD2 Reversion	ESCP
ATT/HDG Reversion	ESCP
ADU Reversion	ESCP

Refer to Figure 73.
The DME is tuned by the MLS or VHF NAV. The FDPS moves an external relay contact to select a navigation source, MLS or VHF NAV.

Refer to Figure 74.
The ARCDUs control the VHF NAV or MLS audio level. The IOMS sends a discrete signal to control the Remote Control Audio Unit (RCAU) navigation source audio.

Refer to Figure 75.
The FDPS uses data from the FADEC oil pressure discrete signal to change the Passenger Address Amplifier (PAA) gain. The FDPS supplies a discrete signal to the PAA when oil pressure from the two engines are less than their threshold. This discrete signal causes the PAA to reduce its PA gain by 6 dB when the two engine are not running.

Refer to Figure 76.
The FDPS calculates data from the PSEU, PEC and FPIU to inhibit the climb and increase climb signals to the TCAS.

Refer to Figure 77.
The FDPS changes the analog fuel flow signals from the fuel flow meter to digital data for transmission to the EIS through a data bus.

The range of the fuel flow signal is 80 to 3000pph.

Refer to Figure 78.
The FDPS changes the analog fuel tank temperature signals from the HLCU signal conditioner to digital data for transmission to the EIS through a data bus.

The range of the fuel tank temperature signal is –70°C to +75°C.

Refer to Figure 79.
The FDPS changes the analog fuel inlet temperature signals from the fuel temperature sensor to digital data for transmission to the EIS through a data bus.

The range of the fuel inlet temperature signal is –55°C to +121°C.

Refer to Figure 80.
The FDPS changes the analog hydraulic pressure and quantity signals from the different hydraulic systems to digital data for transmission to the EIS through a data bus.

The hydraulics pressure data from all hydraulic systems is supplied to the two IOMs. The hydraulics quantity data from system 1 and 3 are supplied to IOM1 and hydraulic quantity 2 to IOM2.

NOTE

Note: A FDPS1 malfunction causes hydraulic quantity indication 1 and 3 to also malfunction. If FDPS2 malfunctions, hydraulic quantity 2 indication also malfunctions.

The range of the hydraulic quantity signal is 0 to 4000PSI.

Refer to Figure 81.
The FDPS changes the analog main oil pressure signals from the engine oil pressure sensors to digital data for transmission to the EIS through a data bus.

The MOP data is supplied to the two IOMs. MOP1 data is supplied through its related IOM1 to the EIS for indication. The MOP2 data is supplied through IOM2 to the EIS when the right main bus power is energized and by IOM1 when the right main bus is not energized.

NOTE

Note: A FDPS1 malfunction causes MOP1 to also malfunction.

NOTE

A FDPS2 malfunction causes MOP2 indication to also malfunction if the right main bus is energized.

The range of the main oil pressure signal is 0 to 260PSI.

Refer to Figure 82.
The FDPS changes the analog parking brake pressure signals from the parking brake hydraulic pressure transmitter to digital data for transmission to the EIS through a data bus.

The parking brake pressure data is supplied to the two IOMs and is supplied through IOM1 to the EIS for indication.

NOTE

Note: A FDPS1 malfunction causes the parking brake pressure indication to also malfunction.

The range of the parking brake pressure signal is 0 to 4000PSI.

Refer to Figure 77.
The FDPS changes the analog flap position signals from the FPIU to digital data for transmission to the EIS through a data bus.

The flap position data is supplied through the IOM1 from the right FPIU channel to the EIS for indication. The data changes to the left FPIU channel if the right malfunctions and to IOM2 if IOM1 malfunctions.

Refer to Figure 83.
The Aircraft Configuration Module stores information. The ACM stores information from the systems that follow.

- TCAS
- ACARS
- SELCAL
- MLS
- FMS
- VHF COM
- HF
- SSM
- ADF
- DME
- ATC
- RA
- IRS

- Aircraft type
- Engine type
- Temperatures
- Quantities
- Fuel used
- Ground mapping
- Aircraft serial numbers
- AOA correction
- WTG outputs
- CRC LSB indications
- CRC MSB indication.

Integrated Flight Cabinets (IFC1, IFC2)

Refer to Figure 84.
The Integrated Flight Cabinets (IFC 1, IFC 2) are Electromagnetic Containment (EMC) compartments that protect the components inside of it from High Intensity Radio Frequencies (HIRF).

It has two avionics cooling fans to remove air to cool the components inside of it.

The IFC have modules that are line replaceable. Each IFC contains the Line Replaceable Modules (LRM) that follow:

- Input/Output Processor (IOP 1, IOP 2)
- Input/Output Module (IOM 1, IOM 2)
- Aircraft Configuration Module (ACM)
- The Ground Proximity Warning System Converter (GPWSC) located in the Stall Protection Module (SPM 1, SPM 2)
- Prime Power Supply Module (PPSM 1, PPSM 2).

Each LRM is guided into the IFC using a guide rail and guide pins. The top and bottom guide pins at the Integrated Flight Cabinets (IFC 1, IFC 2) back panel makes sure that the module and the back panel come together precisely in the correct location.

The Line Replaceable Units (LRU) are held in place with front locking levers. This gives easy installation and removal of the Line Replaceable Unit (LRU) without any tools.

The IFCs are 10.75 in. (273 mm) wide, by 8.9 in. (226 mm) high, by 16.14 in. (410 mm) in depth and weighs 29.8 lb (13.1 kg).

The IFC are installed in the avionics rack.

Input/Output Processor (IOP 1, IOP 2)

Refer to Figure 85.
The Input/Output Processors (IOP 1, IOP 2) receives, calculates, and then transmits data to other avionics systems.

Each FDPS has an IOP and IOM. The FDPS and the Centralized Diagnostic System (CDS) uses the same processor located on the Input/Output Processor (IOP1) module but they function independently.

Each IOP has two electronic boards a CPU Board and an ARINC Board.

The Input/Output Processors (IOP 1, IOP 2) are 1.5 in. (38 mm) wide, by 7.2 in. (183 mm) high, 13.26 in. (337 mm) in depth, and weigh less than a pound (0.45 kilograms).

The IOPs are Line Replaceable Modules (LRM) that are located in their related IFC.

Input/Output Module (IOM 1, IOM 2)

Refer to Figure 85.

The Input/Output Modules (IOM 1, IOM 2) receive different analog and discrete signals. It changes the inputs into an ARINC 429 format and transmits them to the IOPs for calculations.

The IOM are 1.5 in. (38 mm) wide, by 7.2 in. (183 mm) high, 13.26 in. (337 MM) in depth, and weigh less than a pound (0.45 kg).

The IOMs are Line Replaceable Modules (LRM) that are located in their related IFC.

Prime Power Supply Module (PPSM 1, PPSM 2)

The PPSM receives 28 Vdc from the main and essential busses. The PPSM has the functions that follow:

- Filtering
- Pre–regulation
- Energy reserve
- Power limitaion.

The modules that normally receive power are as follows:

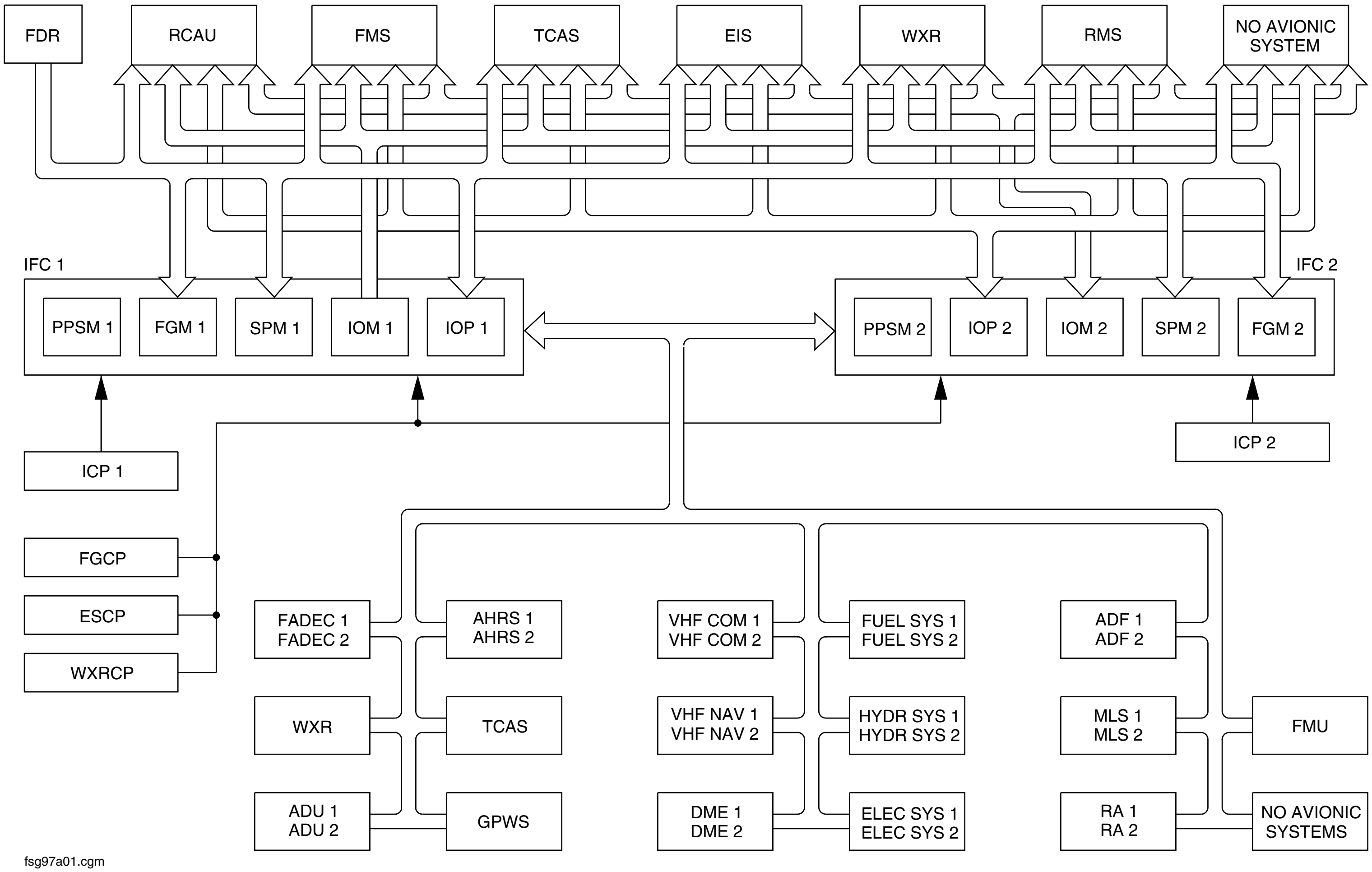
- SPARE1 LRM
- SPM1 LRM
- IOM2 LRM
- IOP2 LRM
- FGM2 LRM
- SPARE2 LRM
- SPM2 LRM.

The modules that receive emergency power are as follows:

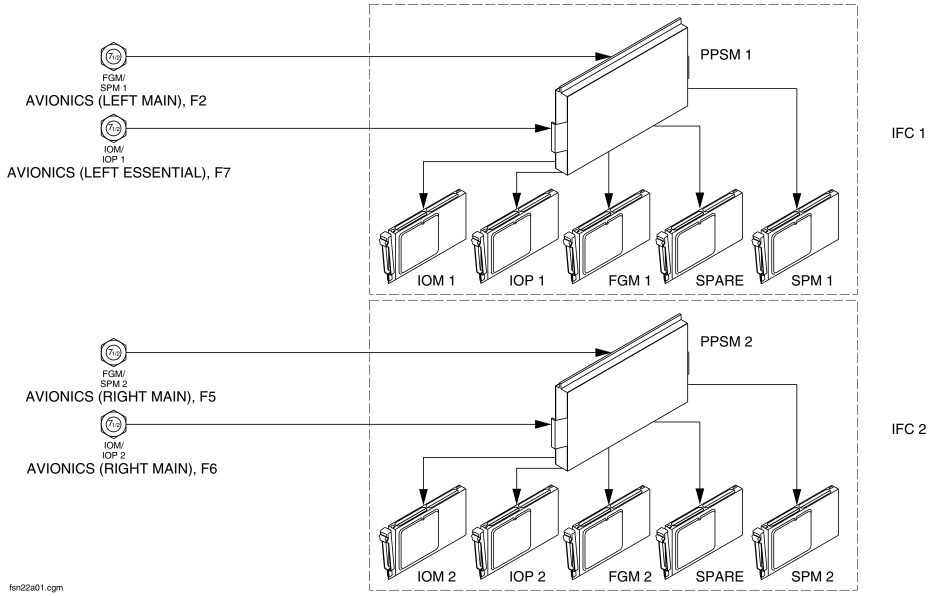
- IOM1 LRM
- IOP1 LRM
- FGM1 LRM

The Prime Power Supply Module (PPSM 1, PPSM 2) are 1 in. wide, by 7.2 in. high, 13.26 in. in depth, and weigh 1.1 lb.

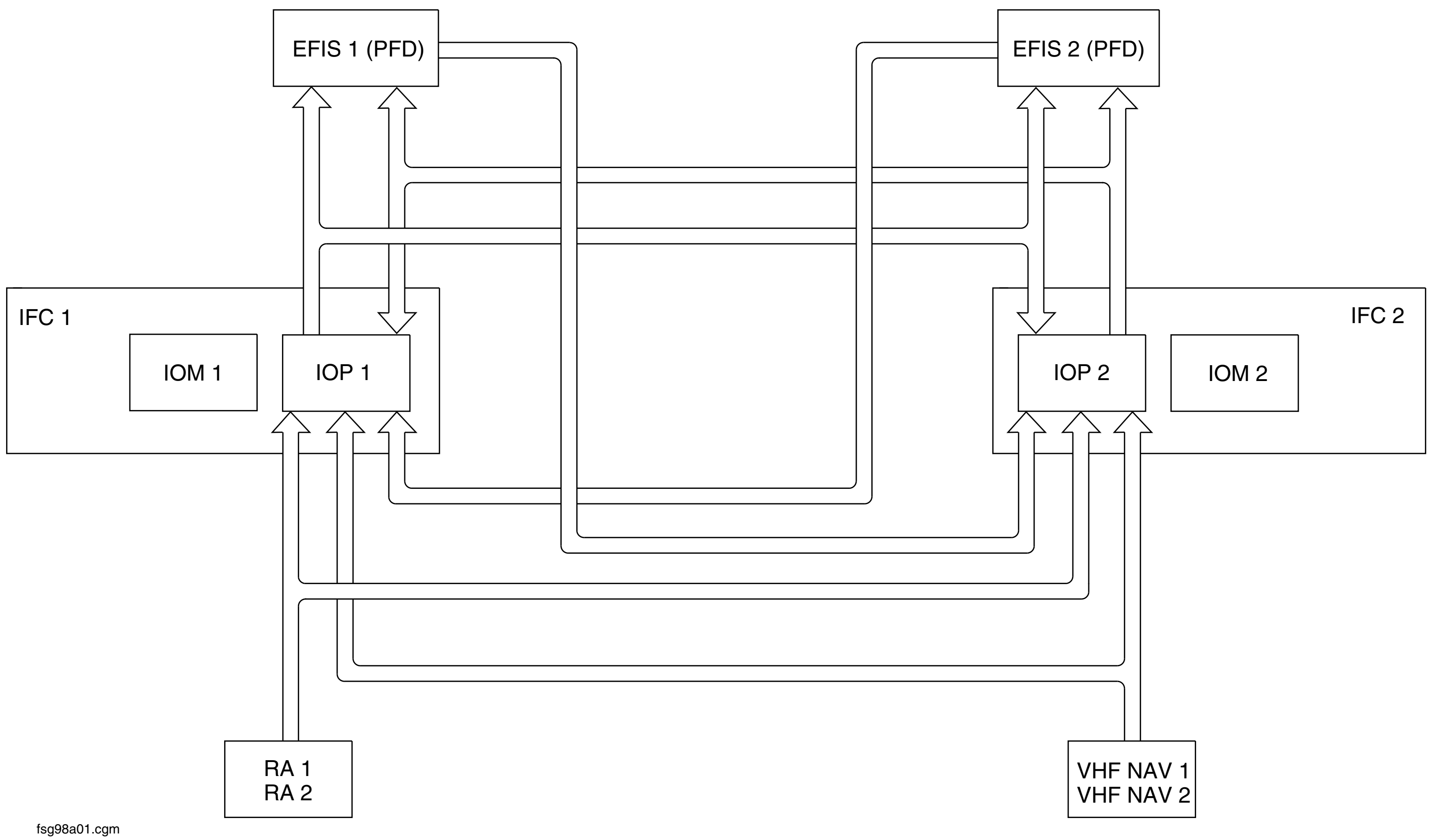
The Prime Power Supply Module (PPSM 1, PPSM 2) are Line Replaceable Modules (LRM) that are located in their related IFC.



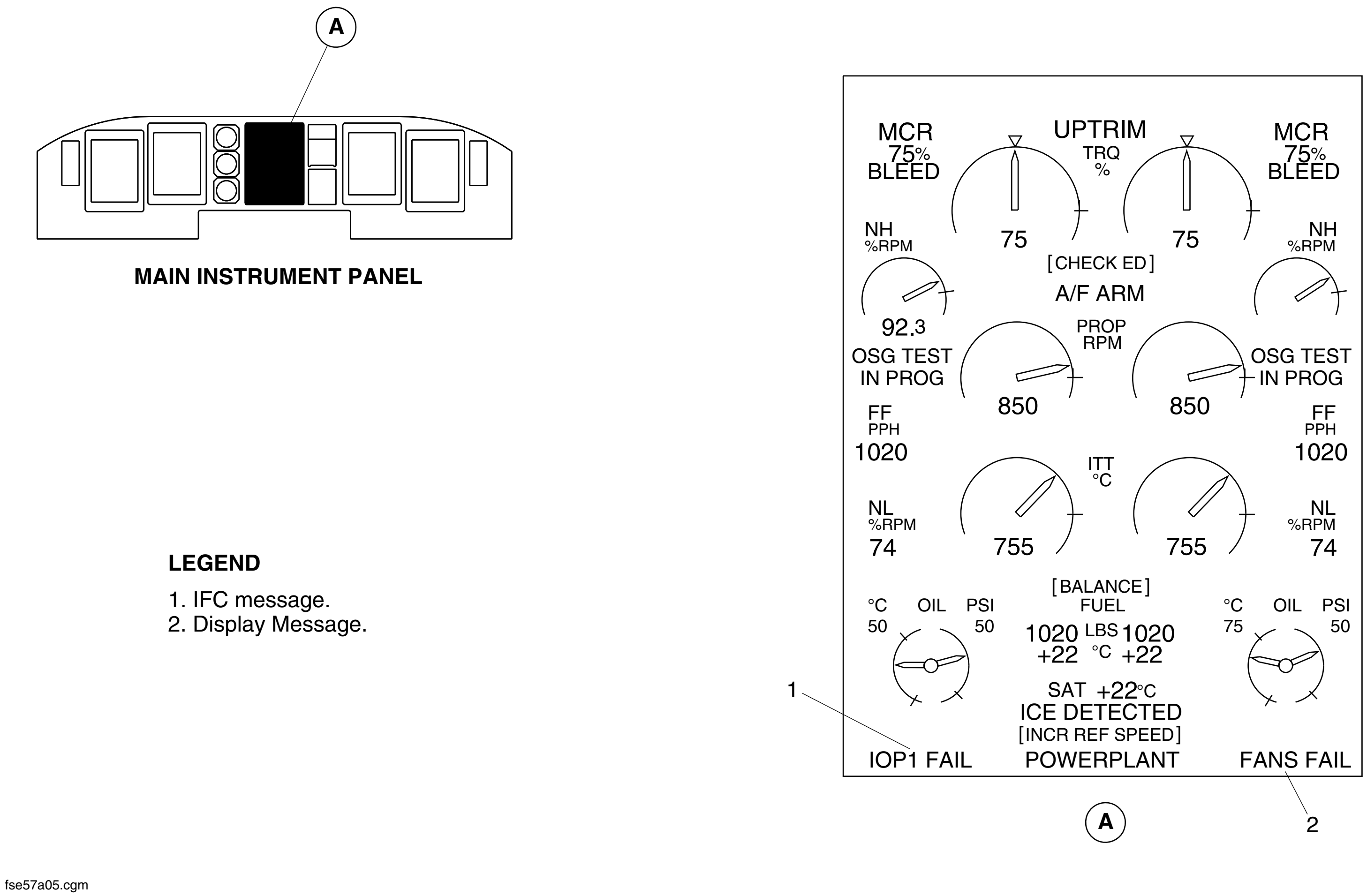
FLIGHT DATA PROCESS SYSTEM BLOCK DIAGRAM
Figure 43



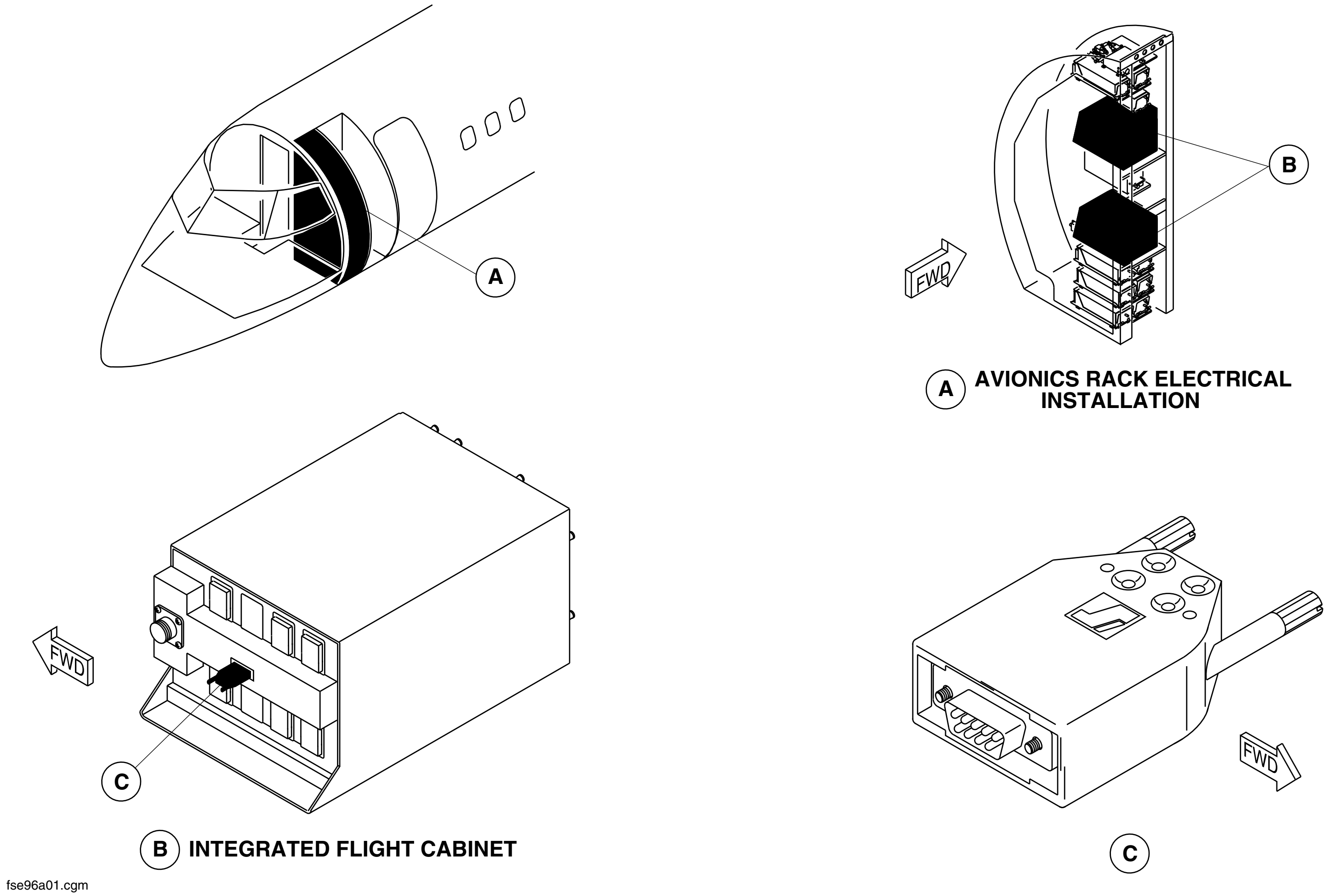
FLIGHT DATA PROCESS SYSTEM BLOCK DIAGRAM, POWER
Figure 44



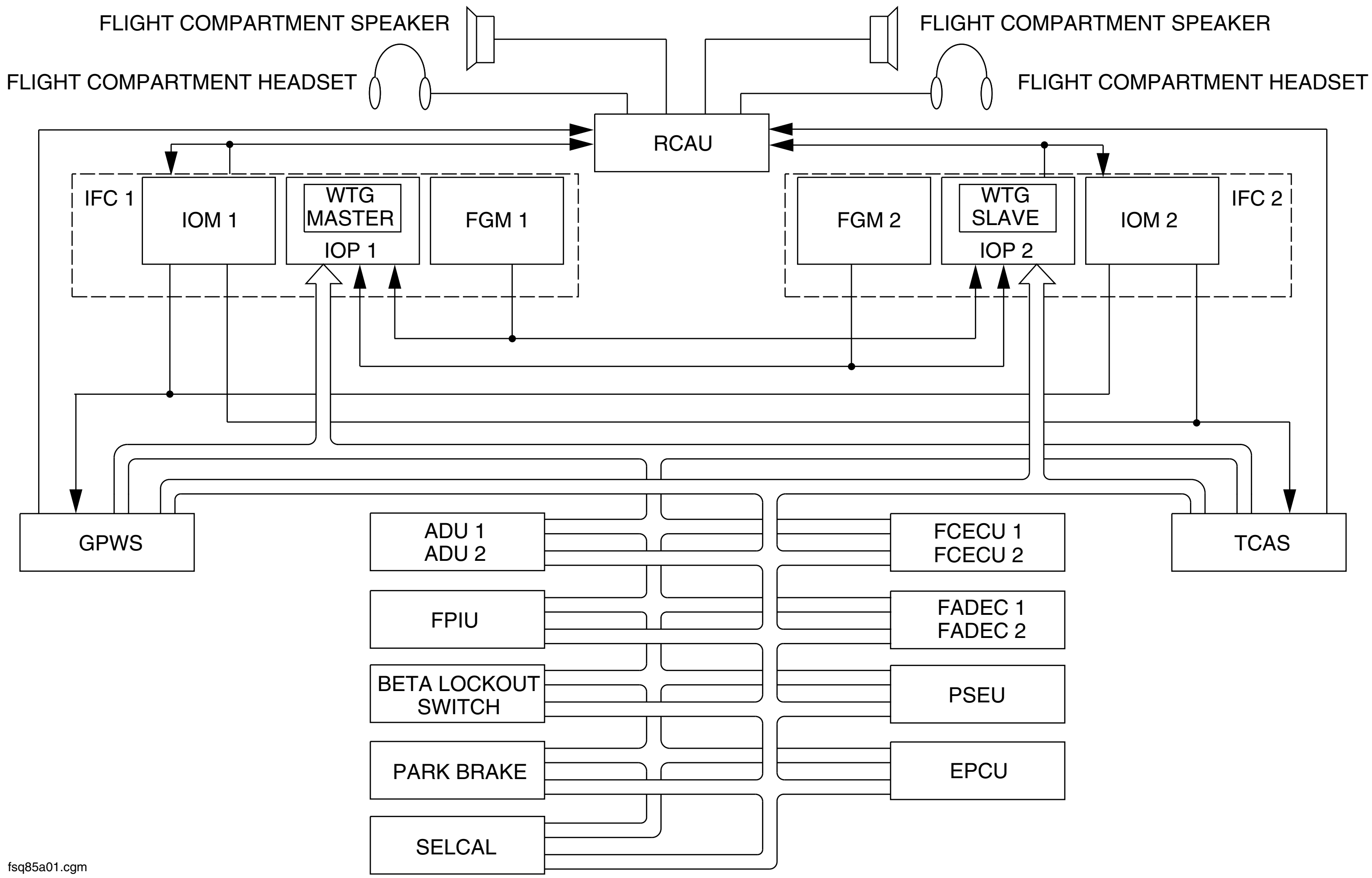
EIS ESSENTIAL MONITORING
Figure 45



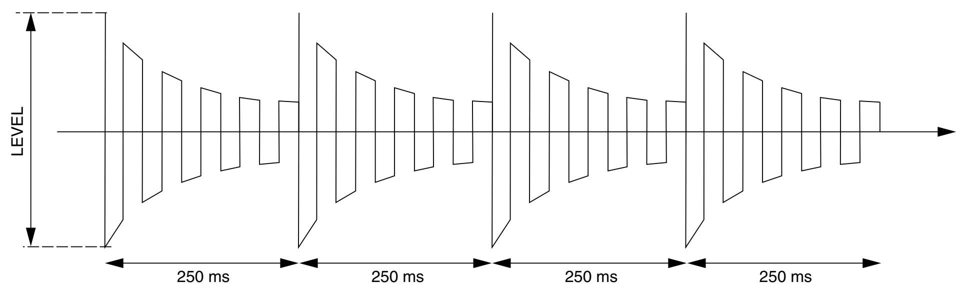
FDPS MESSAGE GENERATION
Figure 46



FDPS AIRCRAFT CONFIGURATION MODULES
Figure 47

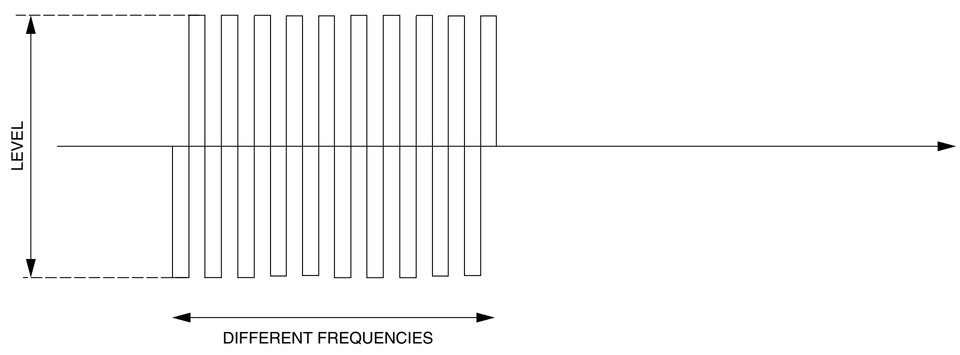


FDPS WARNING TONE GENERATORS (WTG1, WTG2)
Figure 48

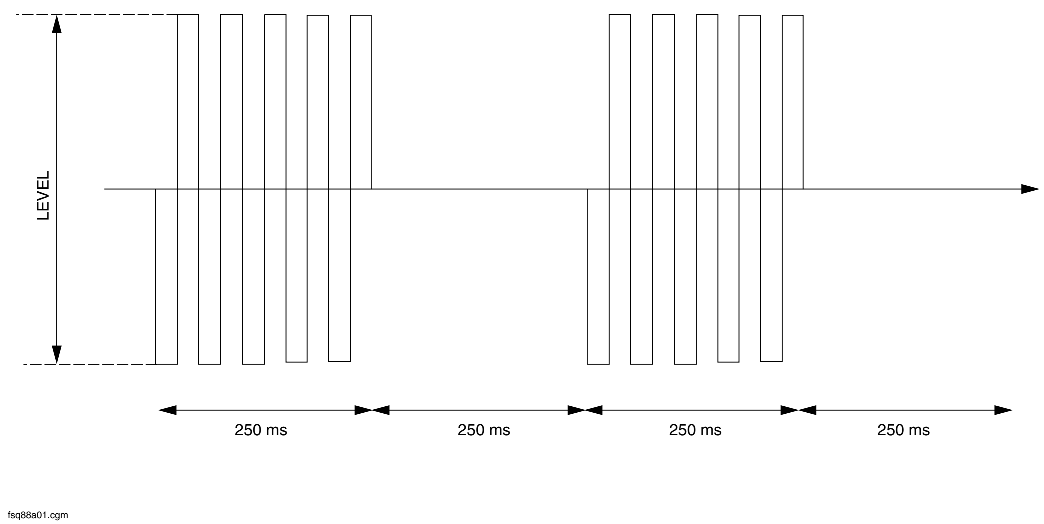


fsq98a01.cgm

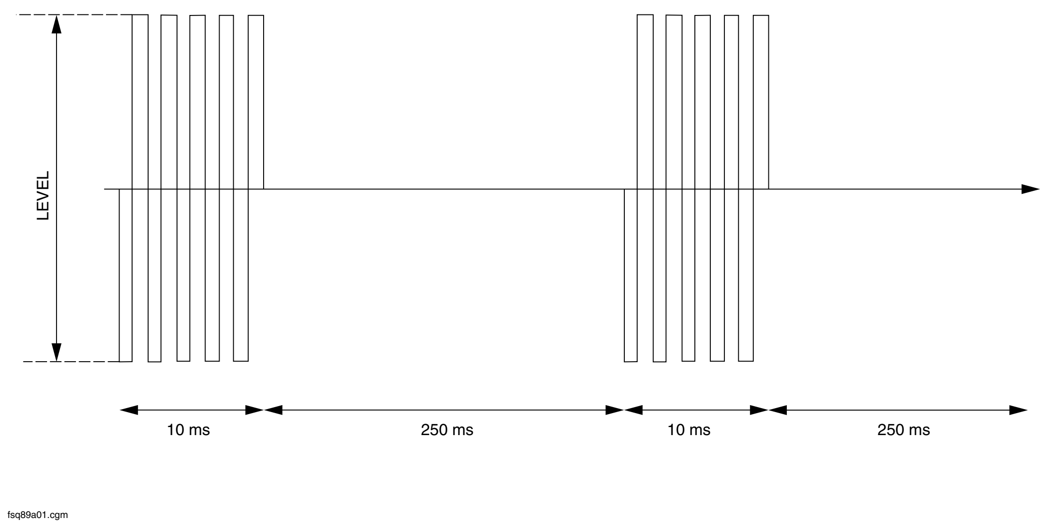
WTG AURAL WARNING TYPE A
Figure 49



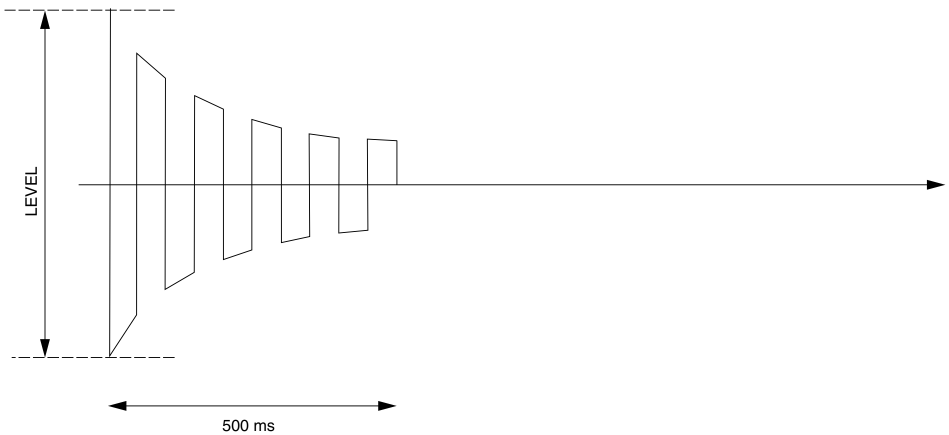
WTG AURAL WARNING TYPE B
Figure 50



WTG AURAL WARNING TYPE C
Figure 51

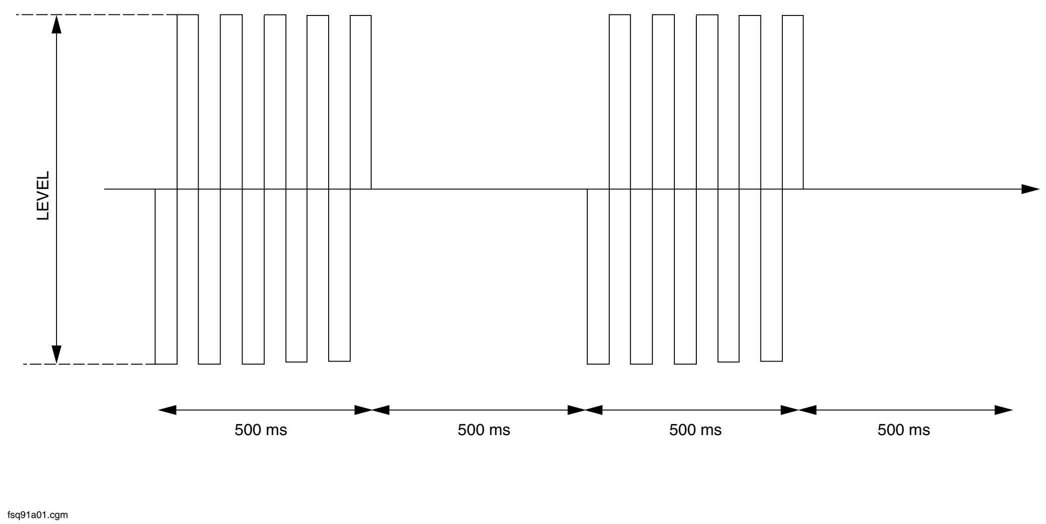


WTG AURAL WARNING TYPE D
Figure 52

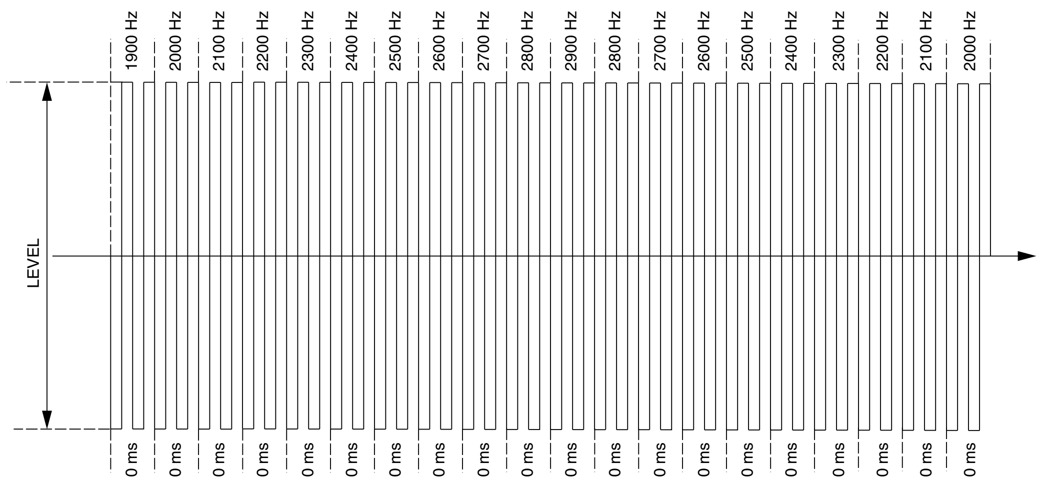


fsq90a01.cgm

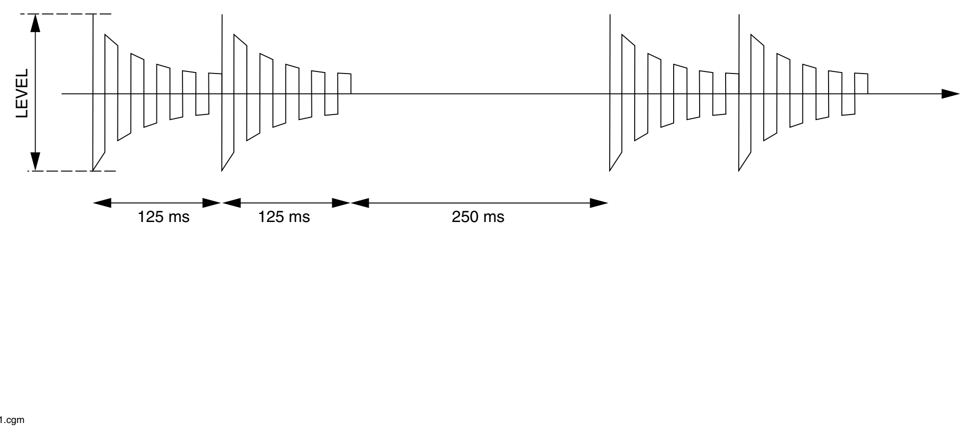
WTG AURAL WARNING TYPE E
Figure 53



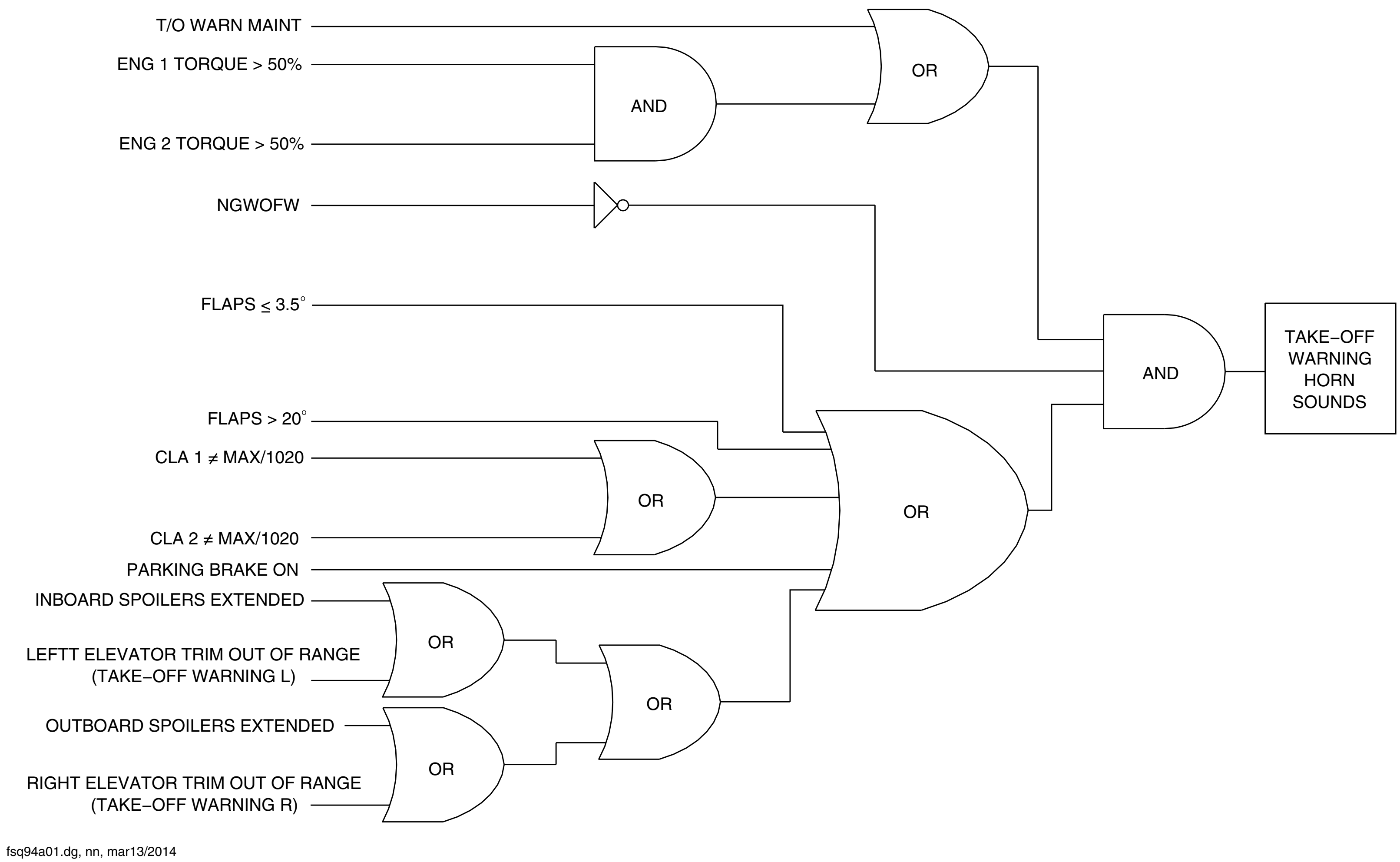
WTG AURAL WARNING TYPE F
Figure 54



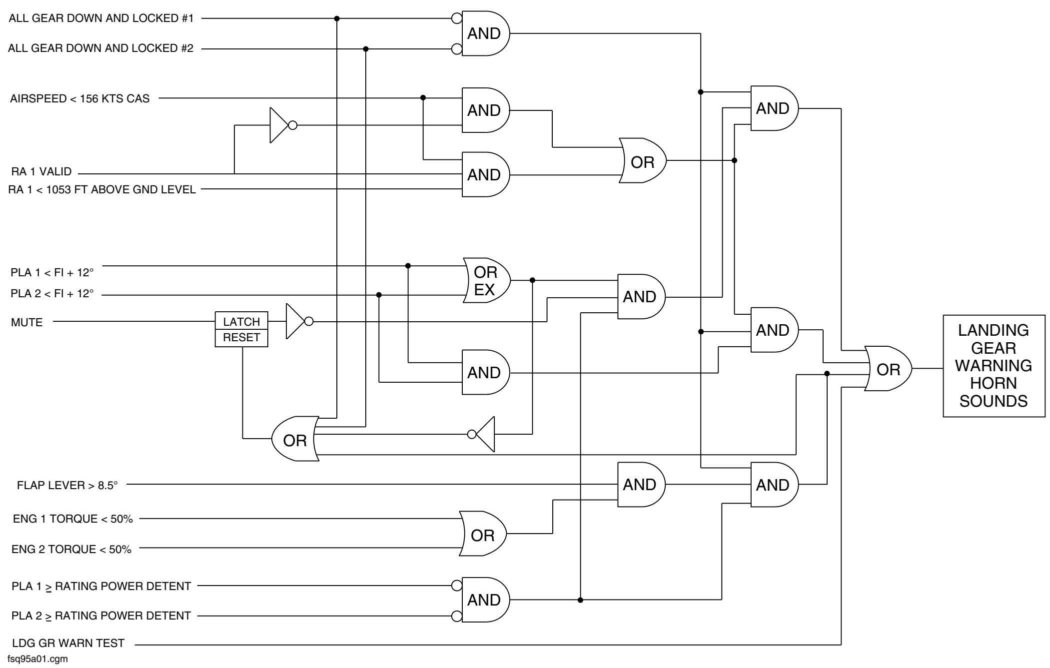
WTG AURAL WARNING TYPE G
Figure 55



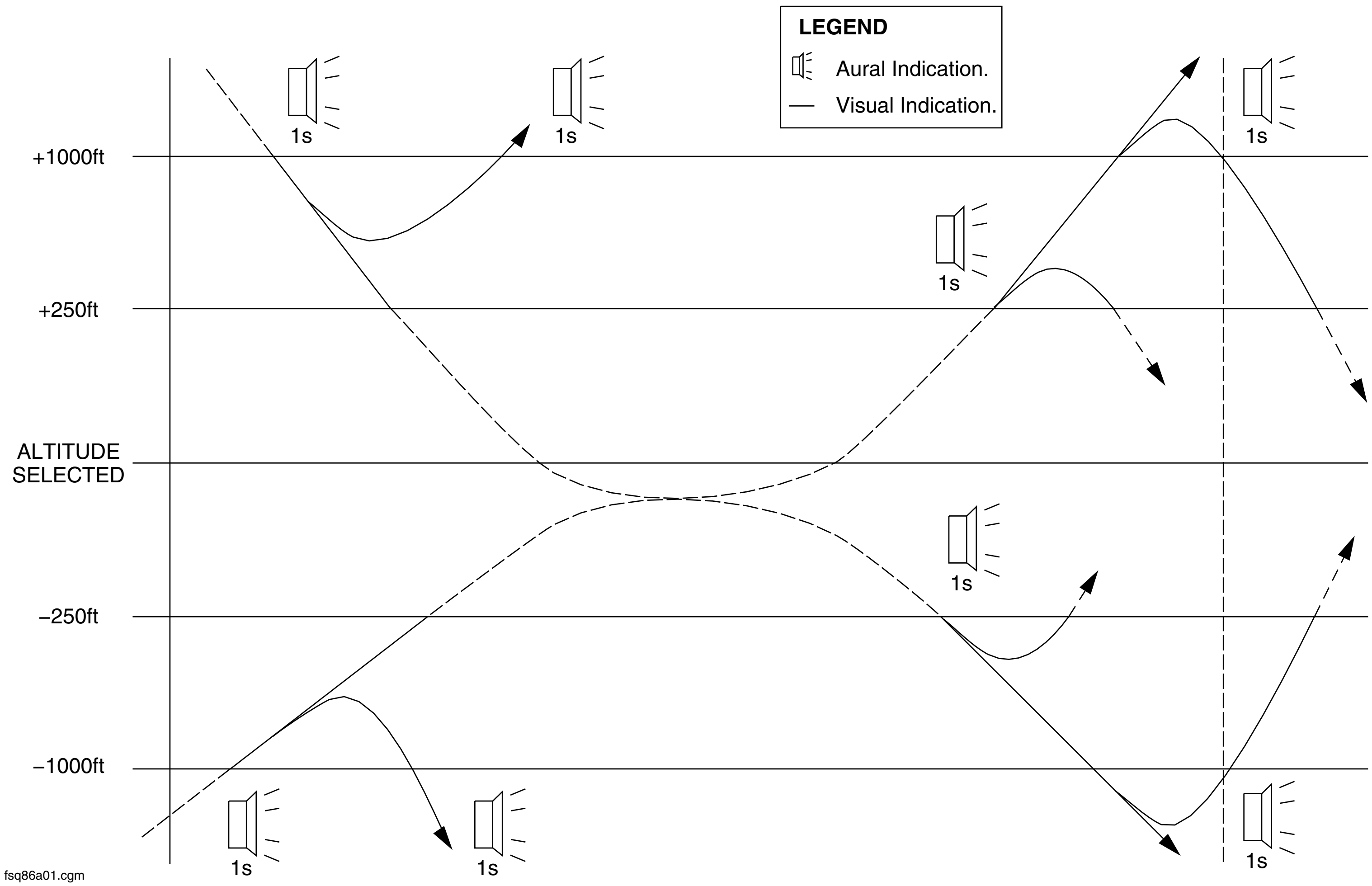
WTG AURAL WARNING TYPE H
Figure 56



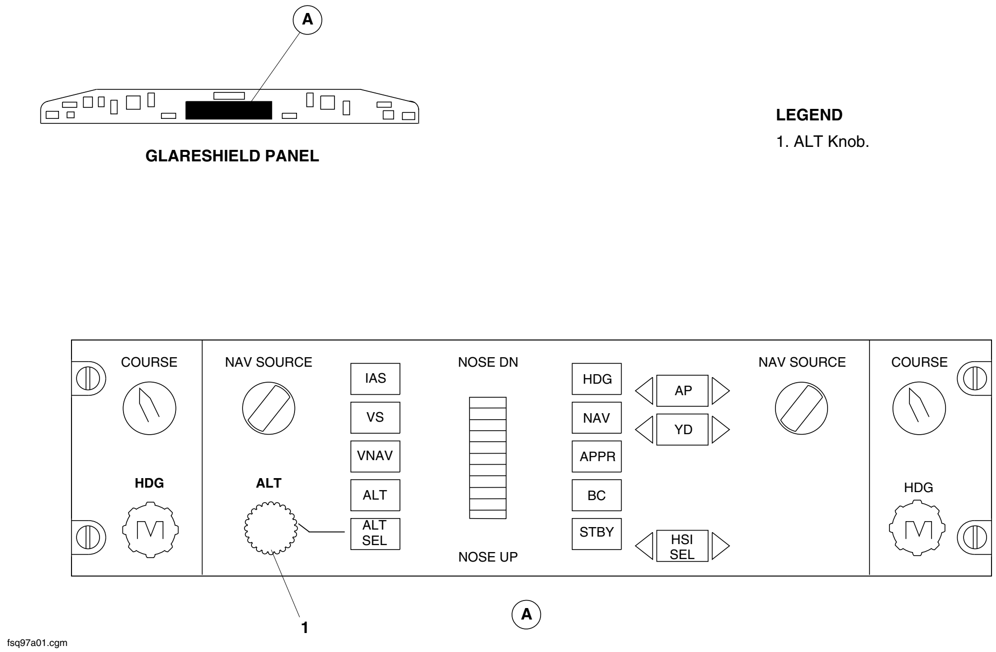
FDPS Incorrect Take–Off Configuration Warning
Figure 57



FDPS INCORRECT LANDING GEAR CONFIGURATION
Figure 58

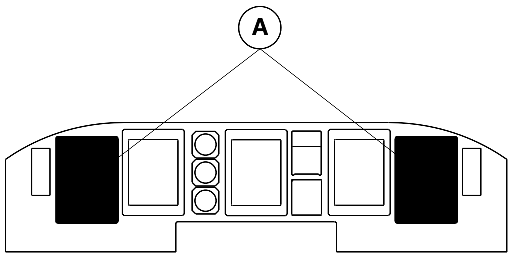


FDPS ALTITUDE ALERT
Figure 59

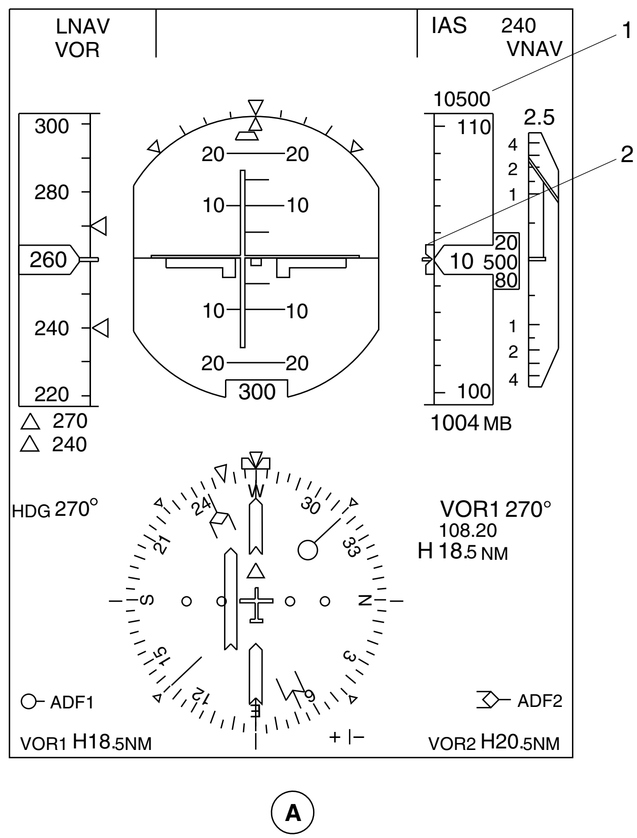


LEGEND
1. ALT Knob.

FDPS ALTITUDE ALERT SELECTION
Figure 60



MAIN INSTRUMENT PANEL

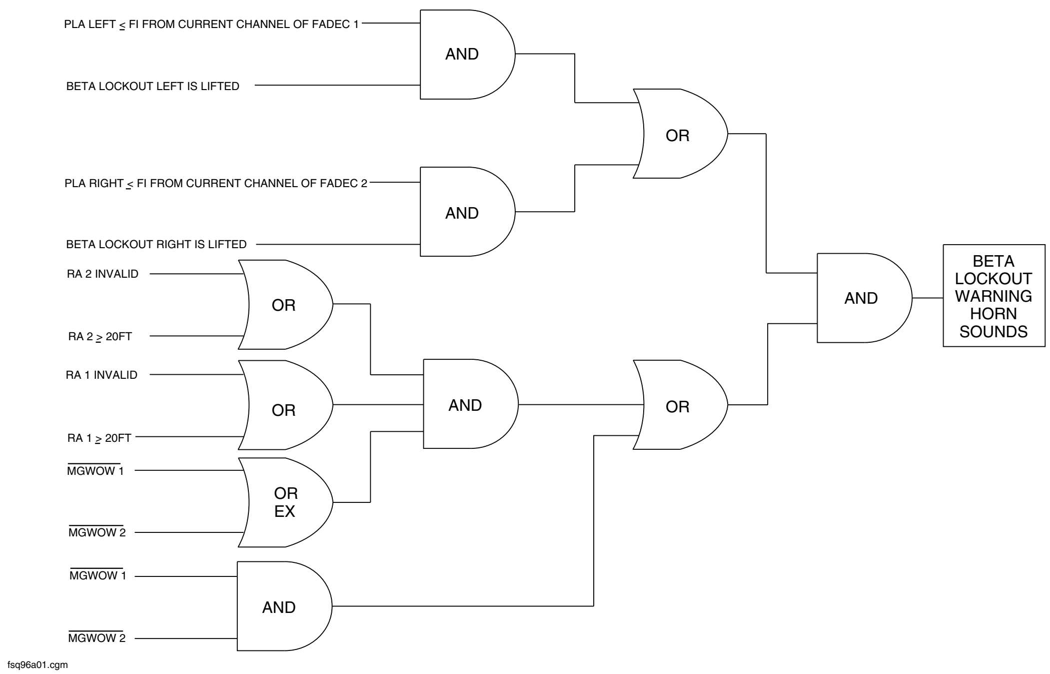


LEGEND

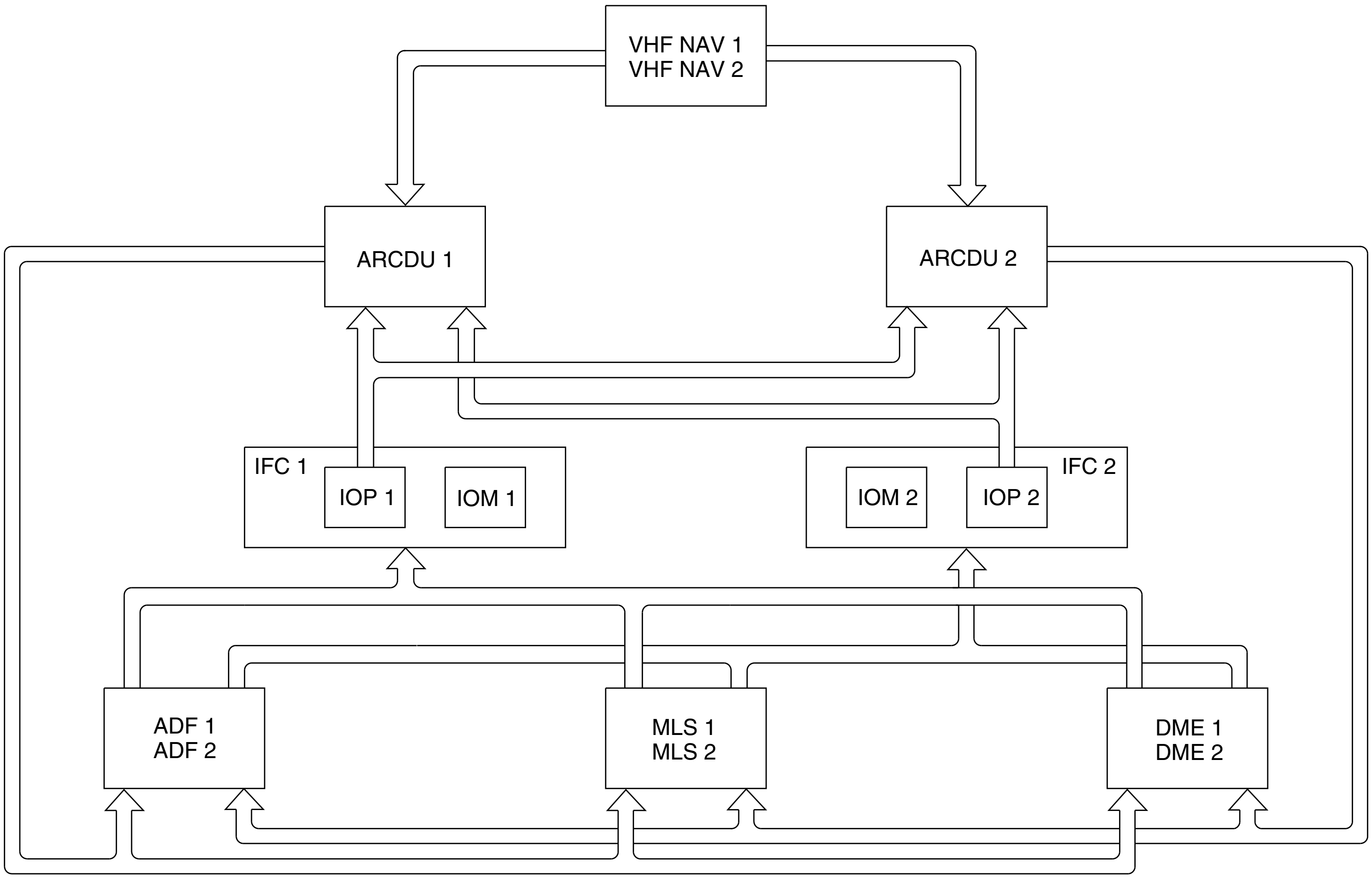
1. Preselected Altitude Readout.
2. Preselected Altitude Bug.

fae56a02.cgm

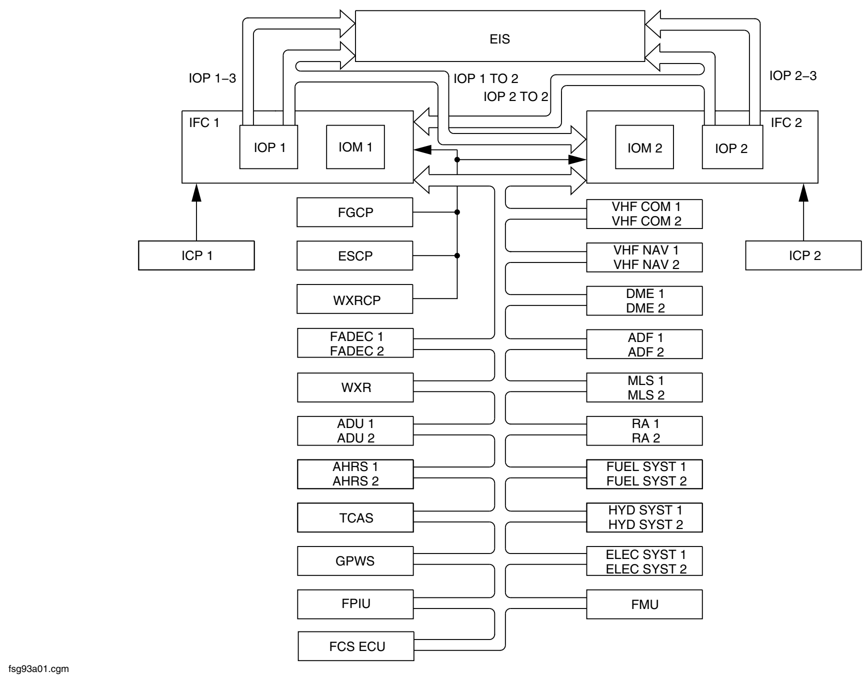
FDPS ALTITUDE ALERT INDICATION
Figure 61



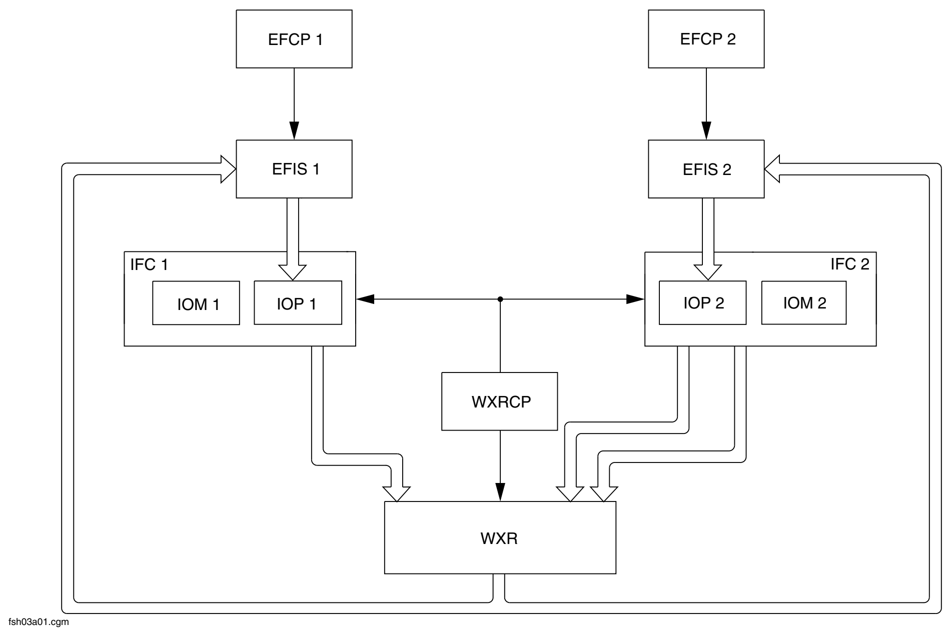
FDPS BETA LOCKOUT WARNING
Figure 62



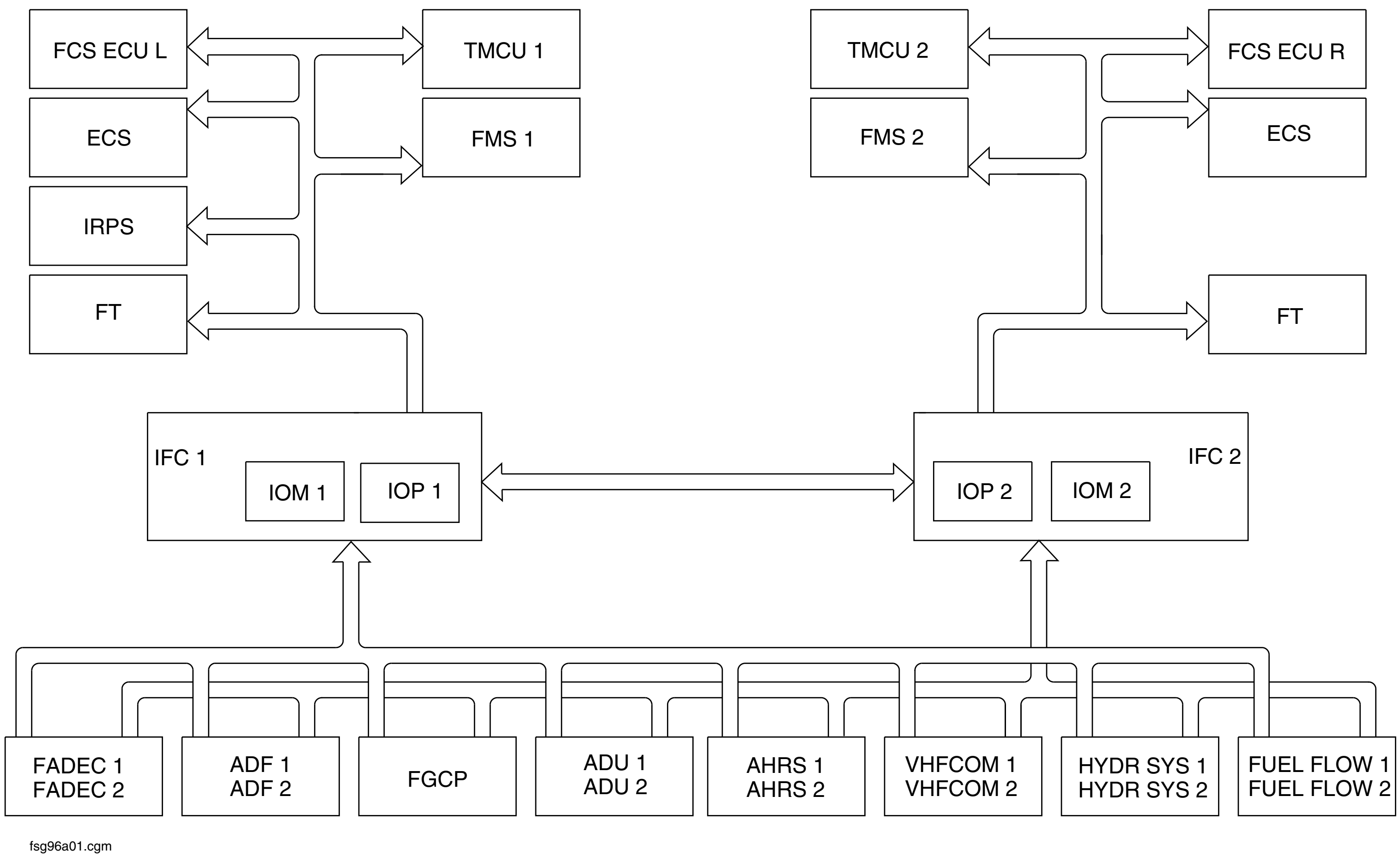
FDPS ARCDU CALCULATION AND CONCENTRATION
Figure 63



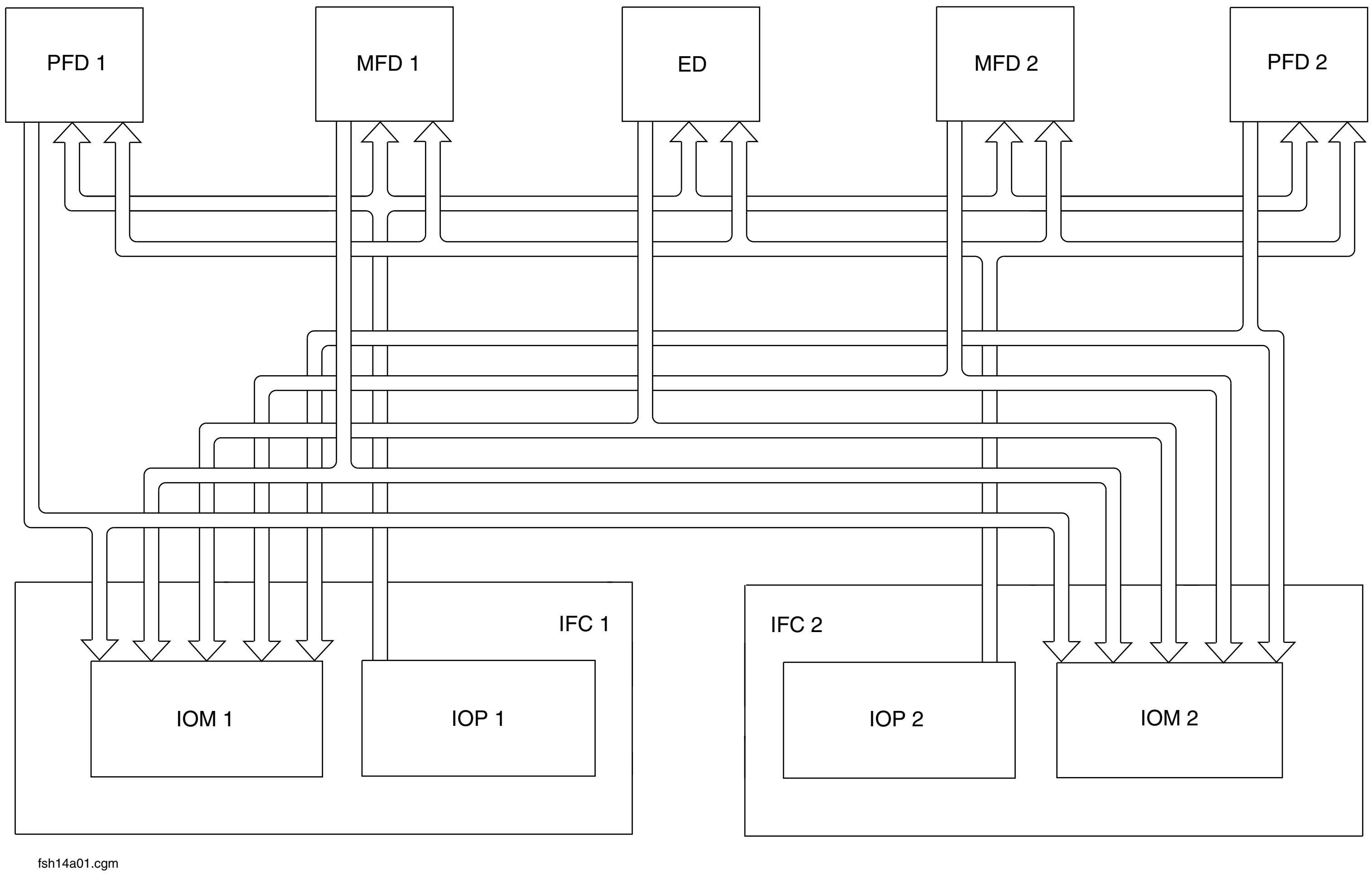
FDPS EIS AND OPPOSITE IOP CONCENTRATION
Figure 64



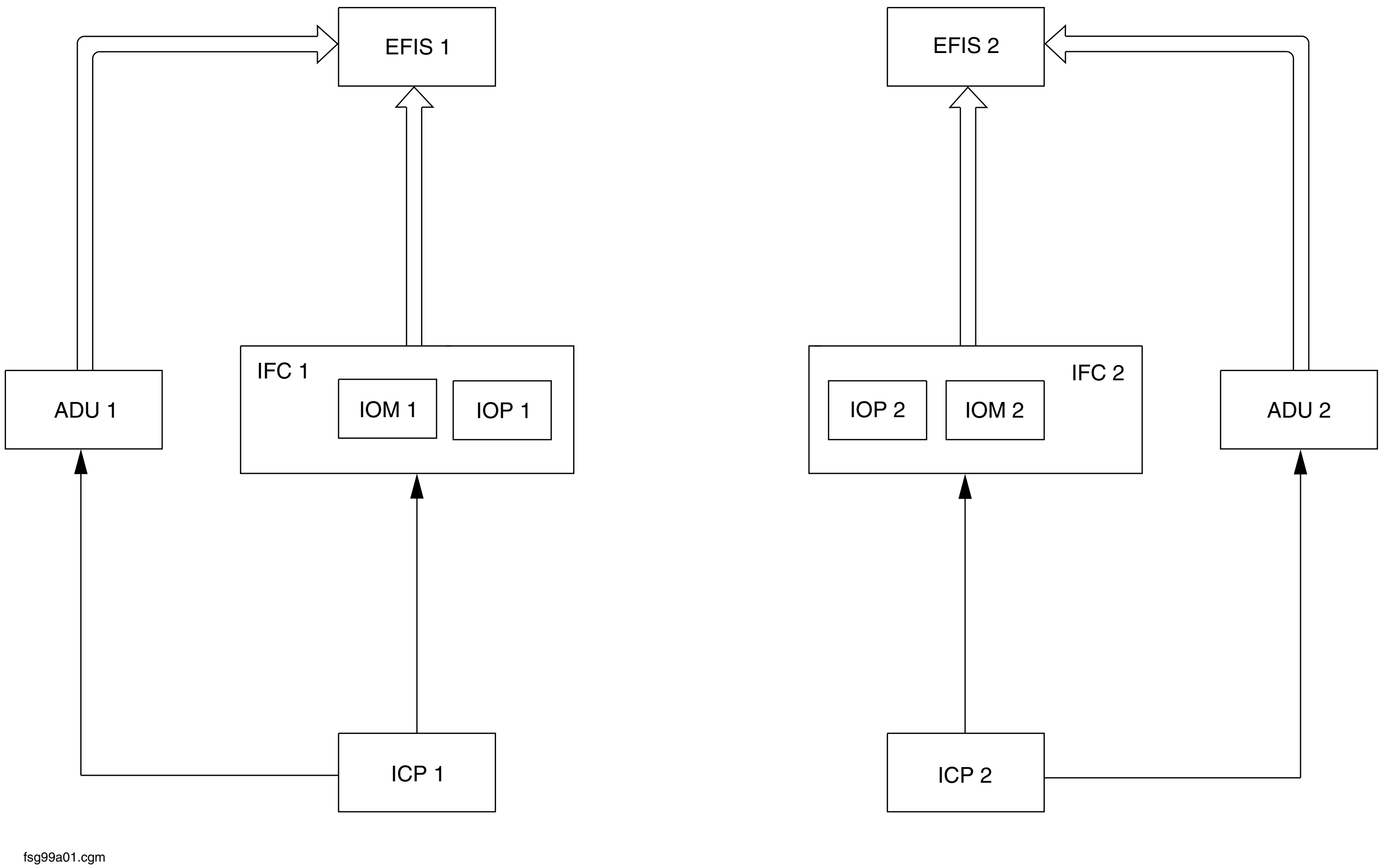
FDPS TCAS AND WXR CALCULATION AND CONCENTRATION
Figure 65



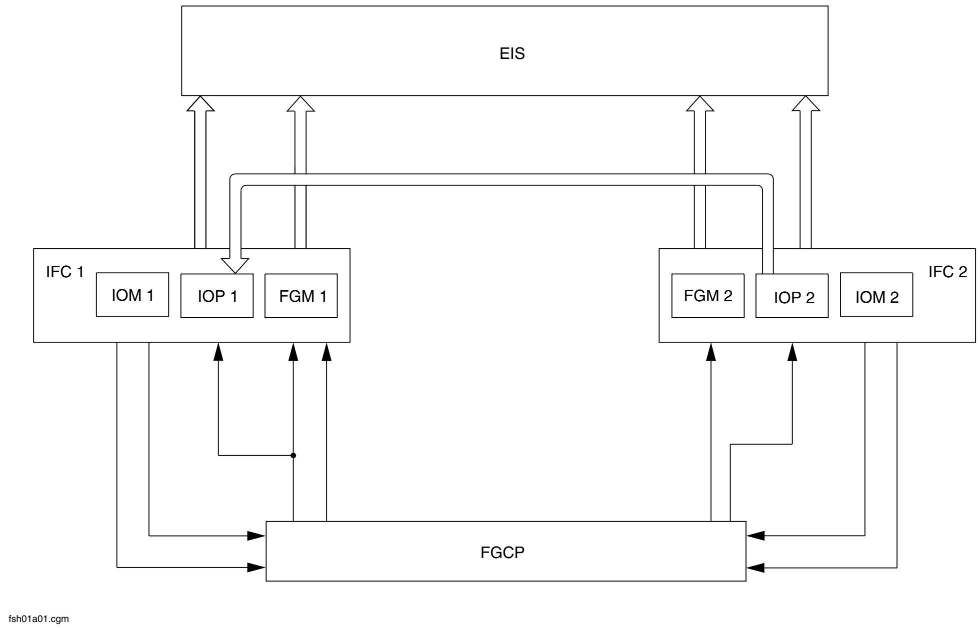
FDPS GENERAL PURPOSE DATA BUS (GPDB) CONCENTRATION
Figure 66



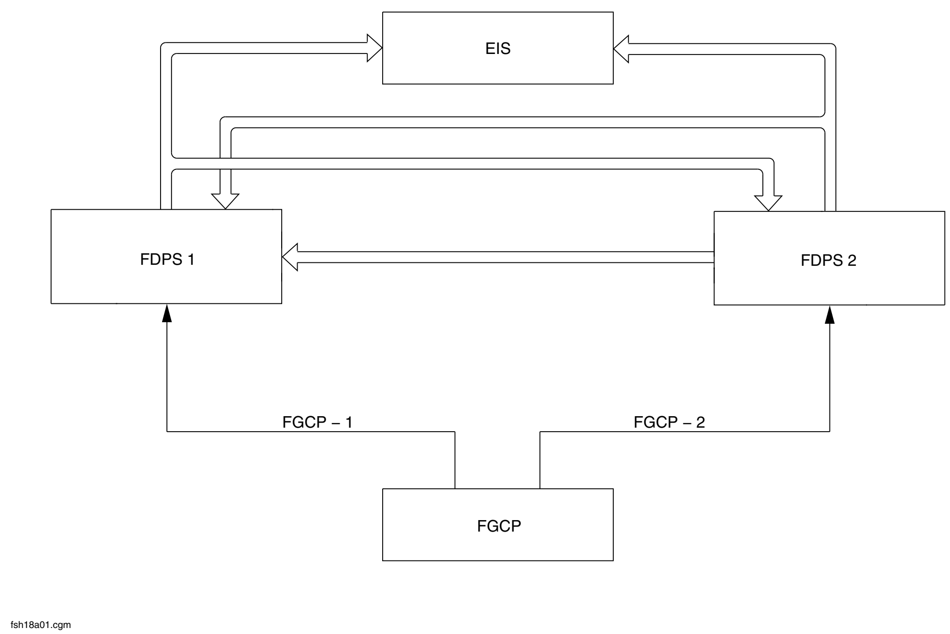
FDPS DISPLAY UNIT CALCULATION AND CONCENTRATION
Figure 67



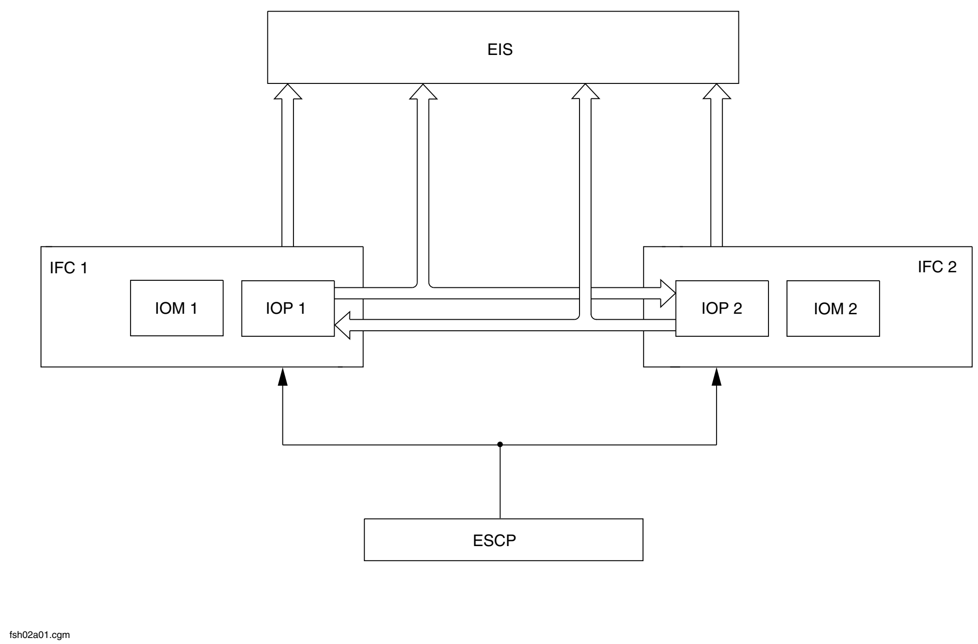
FDPS ICP CALCULATION
Figure 68



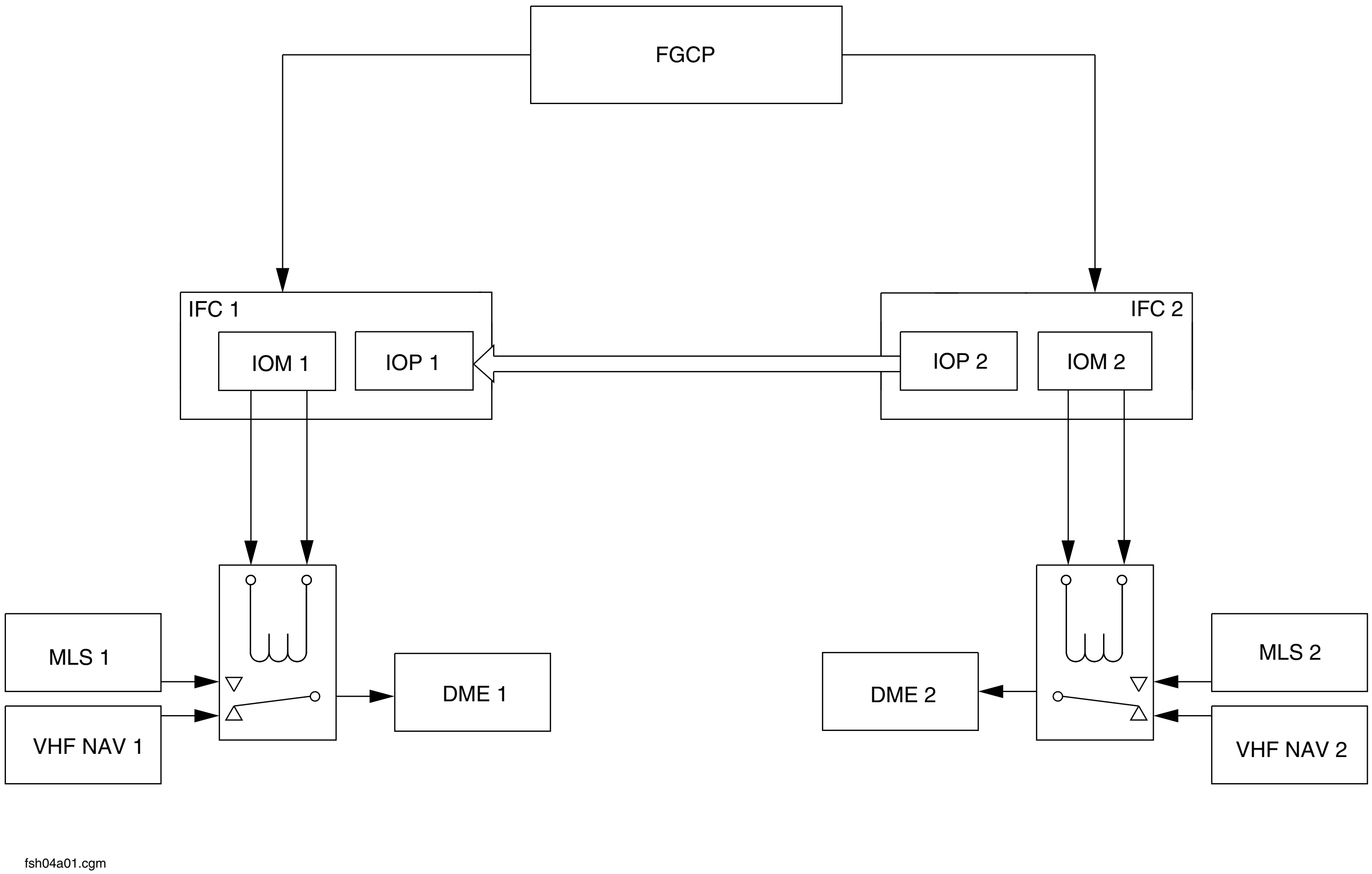
FDPS FLIGHT GUIDANCE CONTROL PANEL (FGCP) CALCULATION
Figure 69



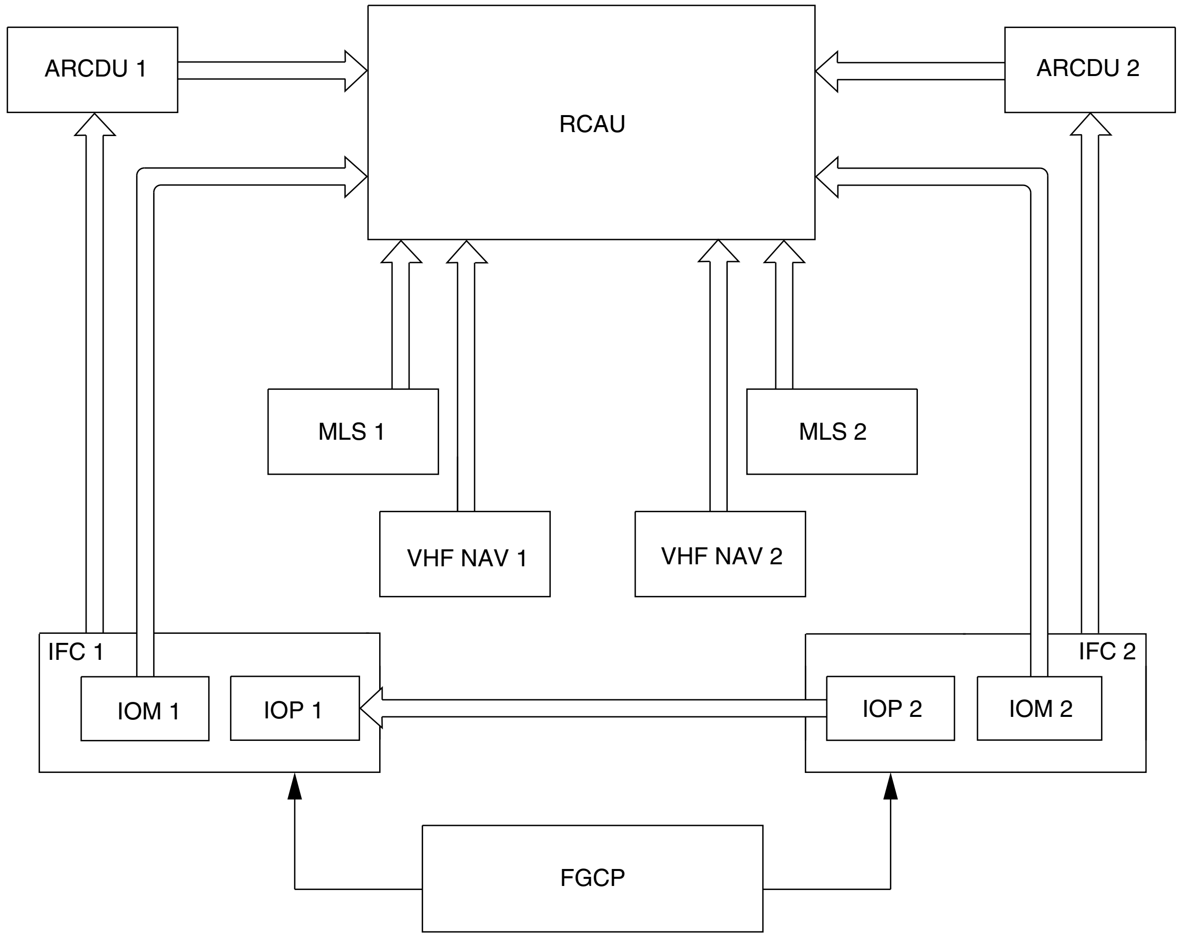
FDPS FGCP MONITORING
Figure 70



FDPS ESID CONTROL PANEL (ESCP) CALCULATION
Figure 71

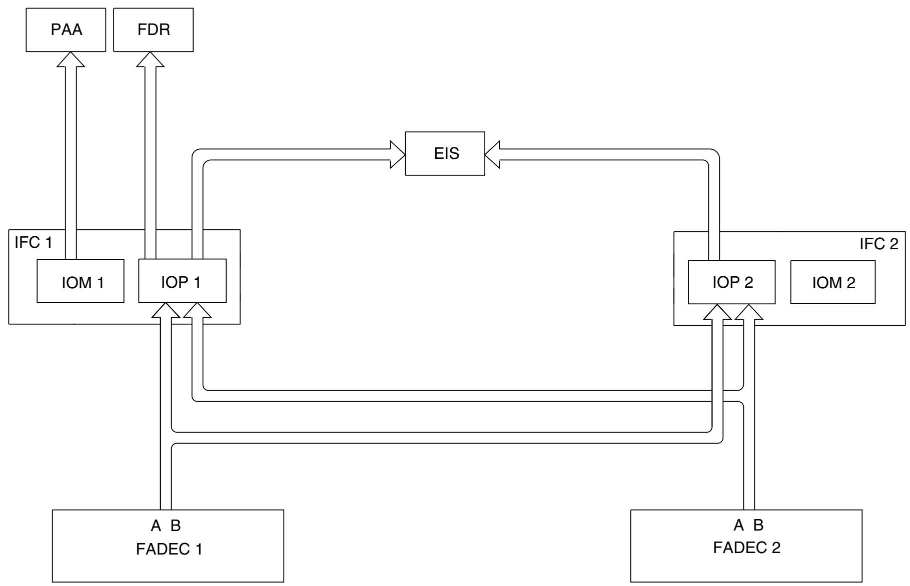


FDPS MLS/NAV TUNE SELECTION CALCULATION
Figure 72



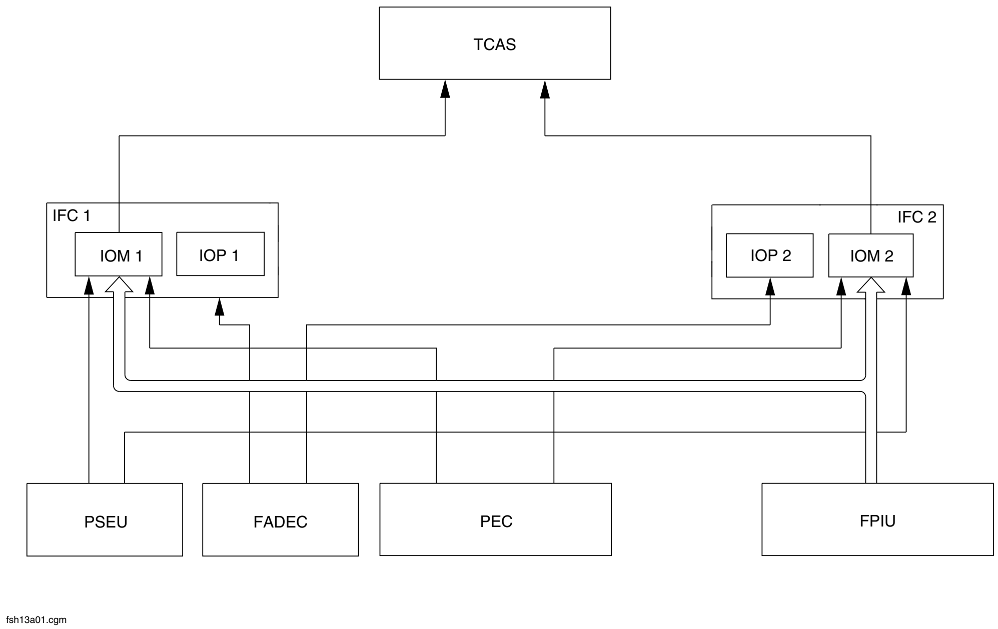
fsh16a01.cgm

VOR/MLS AUDIO SELECTION CALCULATION
Figure 73

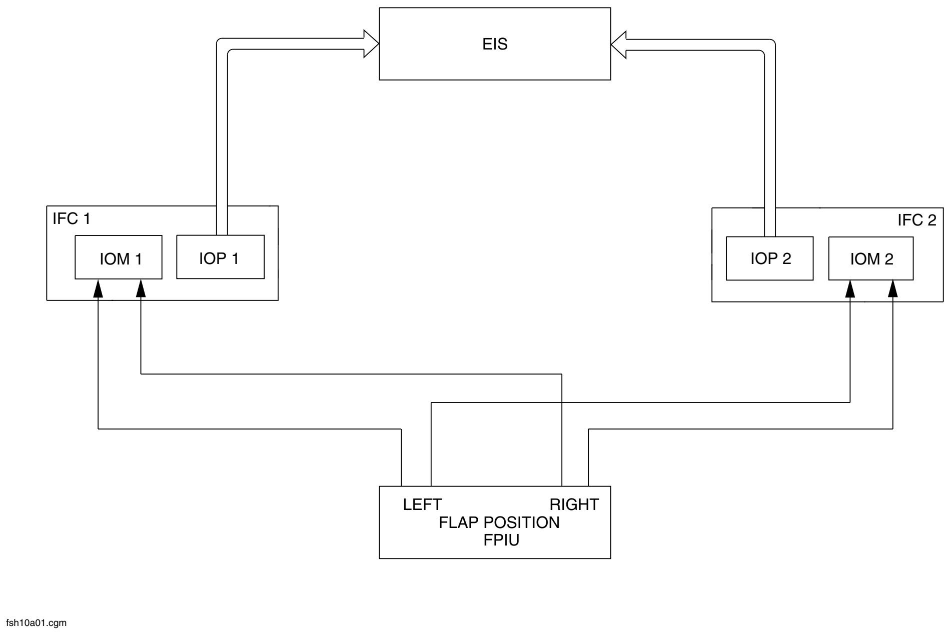


fsh12a01.cgm

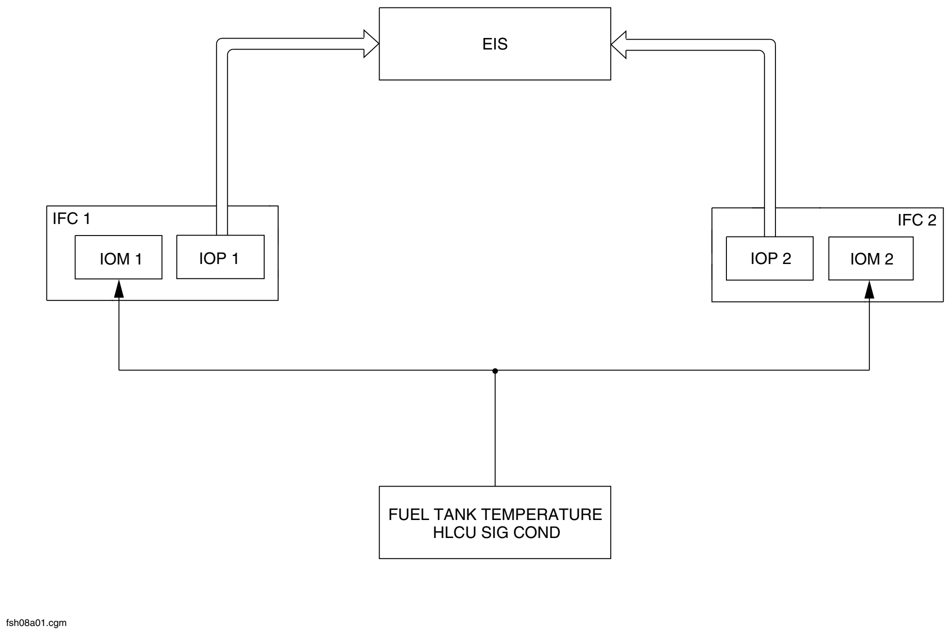
FDPS ENGINES OPERATING/STOPPED CALCULATION
Figure 74



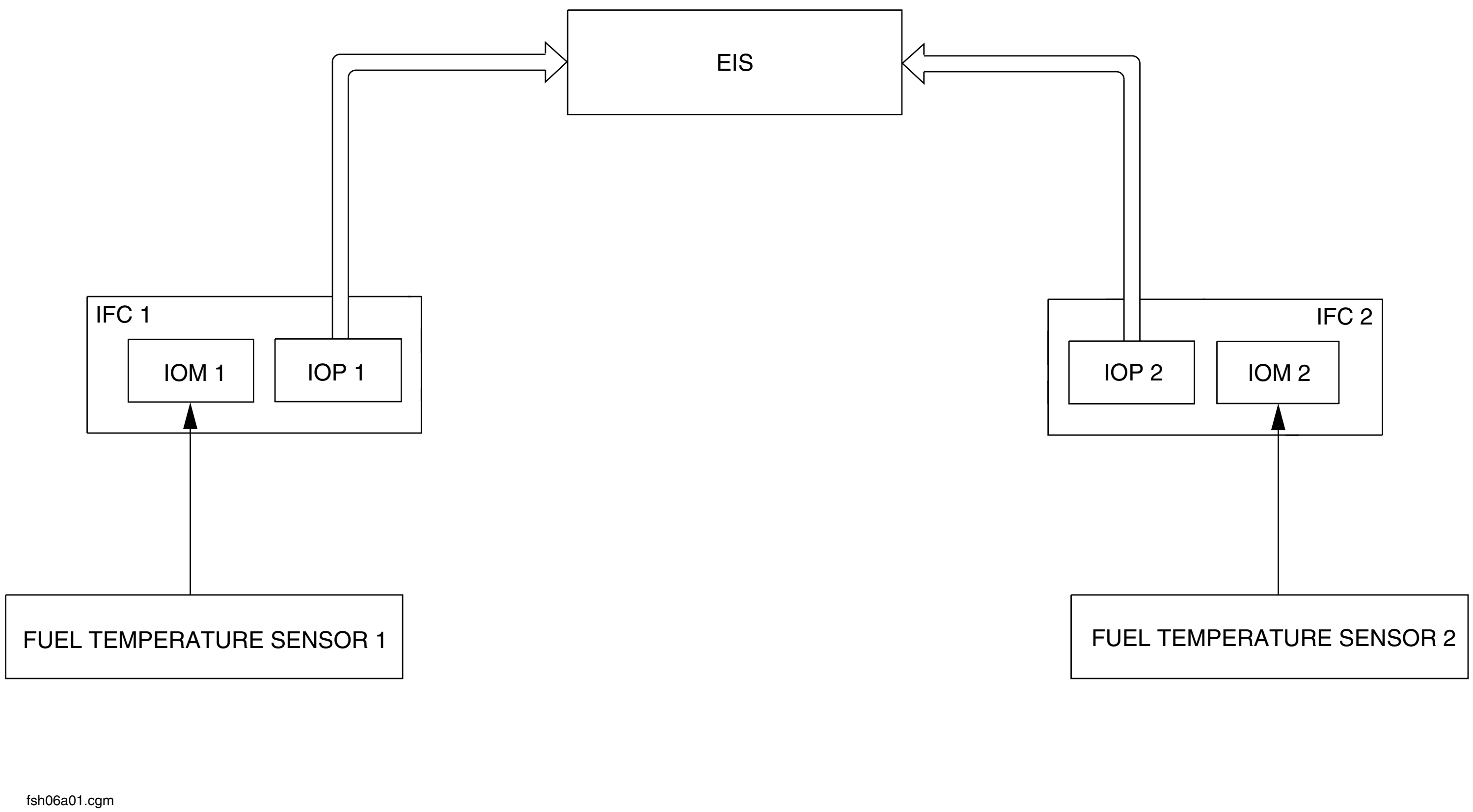
FDPS TCAS MANEUVER INHIBIT CALCULATION
Figure 75



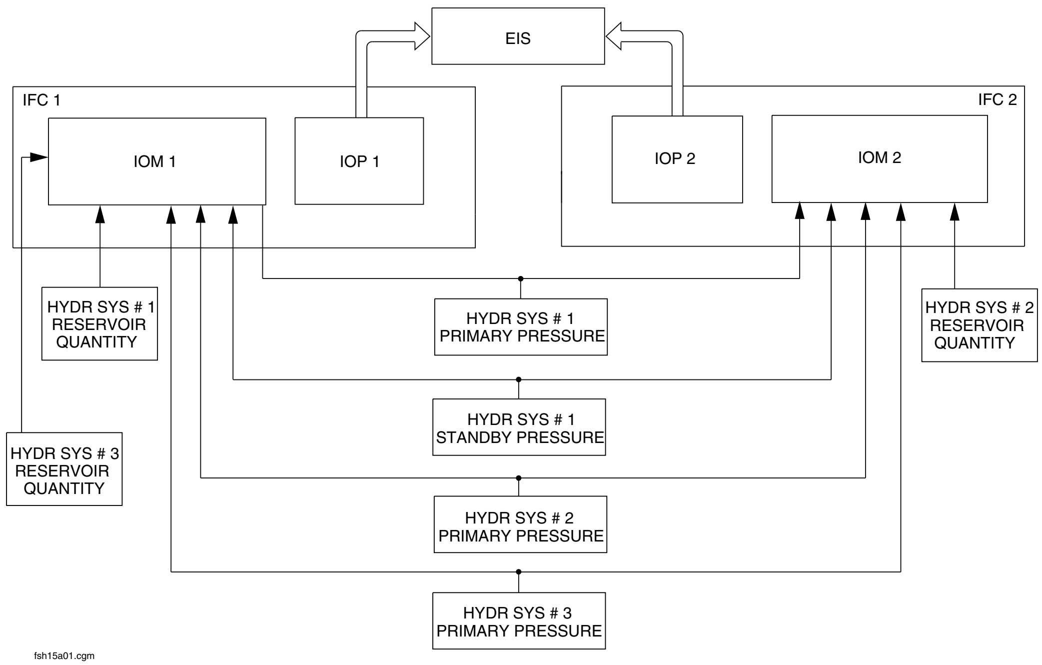
FDPS FLAP POSITION CALCULATION
Figure 76



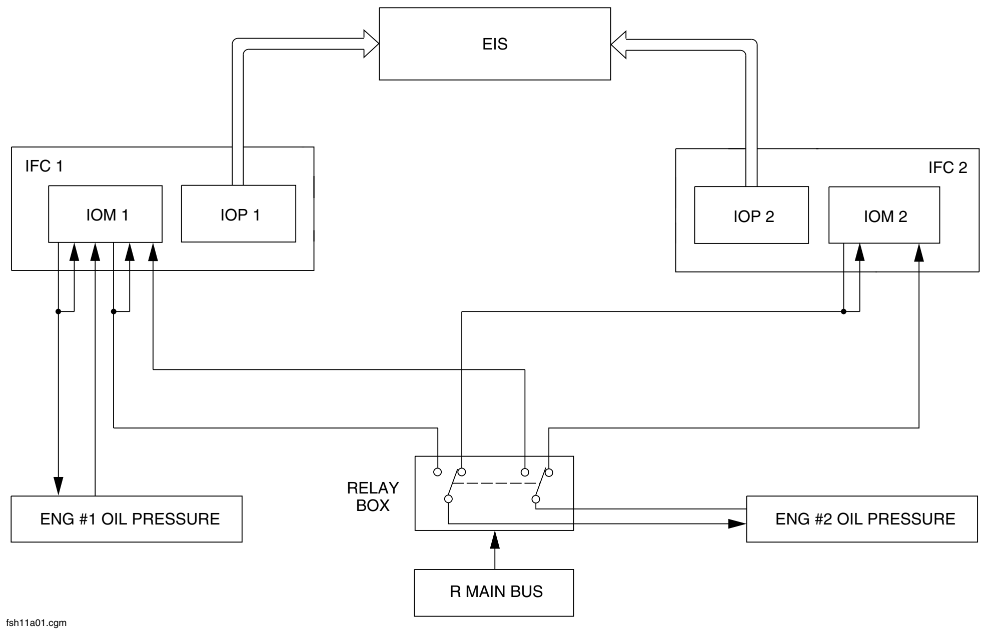
FDPS FUEL TANK TEMPERATURE CALCULATION
Figure 77



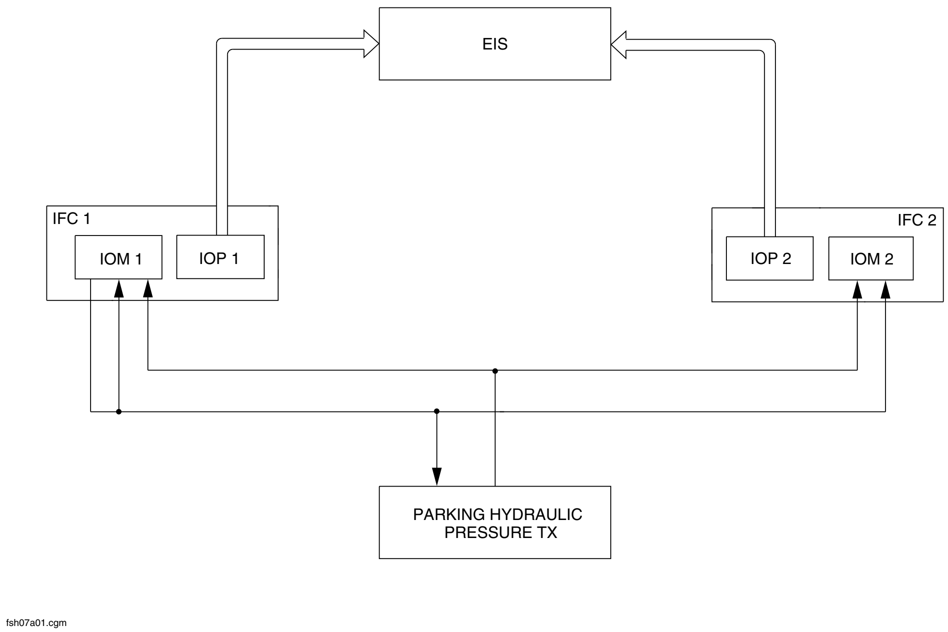
FDPS FUEL INLET TEMPERATURE CALCULATION
Figure 78



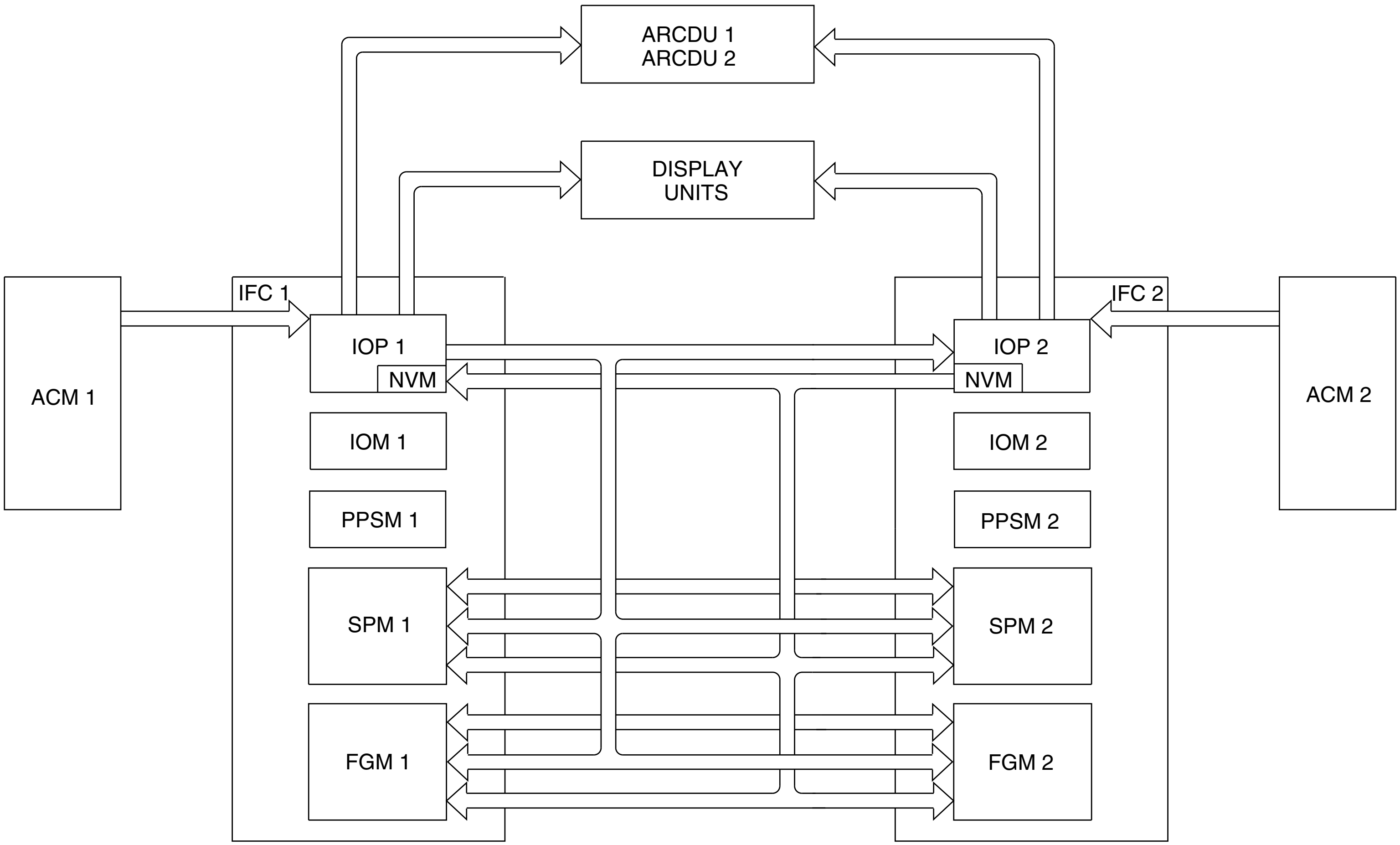
FDPS HYDRAULIC PARAMETERS CALCULATION
Figure 79



FDPS MAIN OIL PRESSURE (MOP) CALCULATION
Figure 80

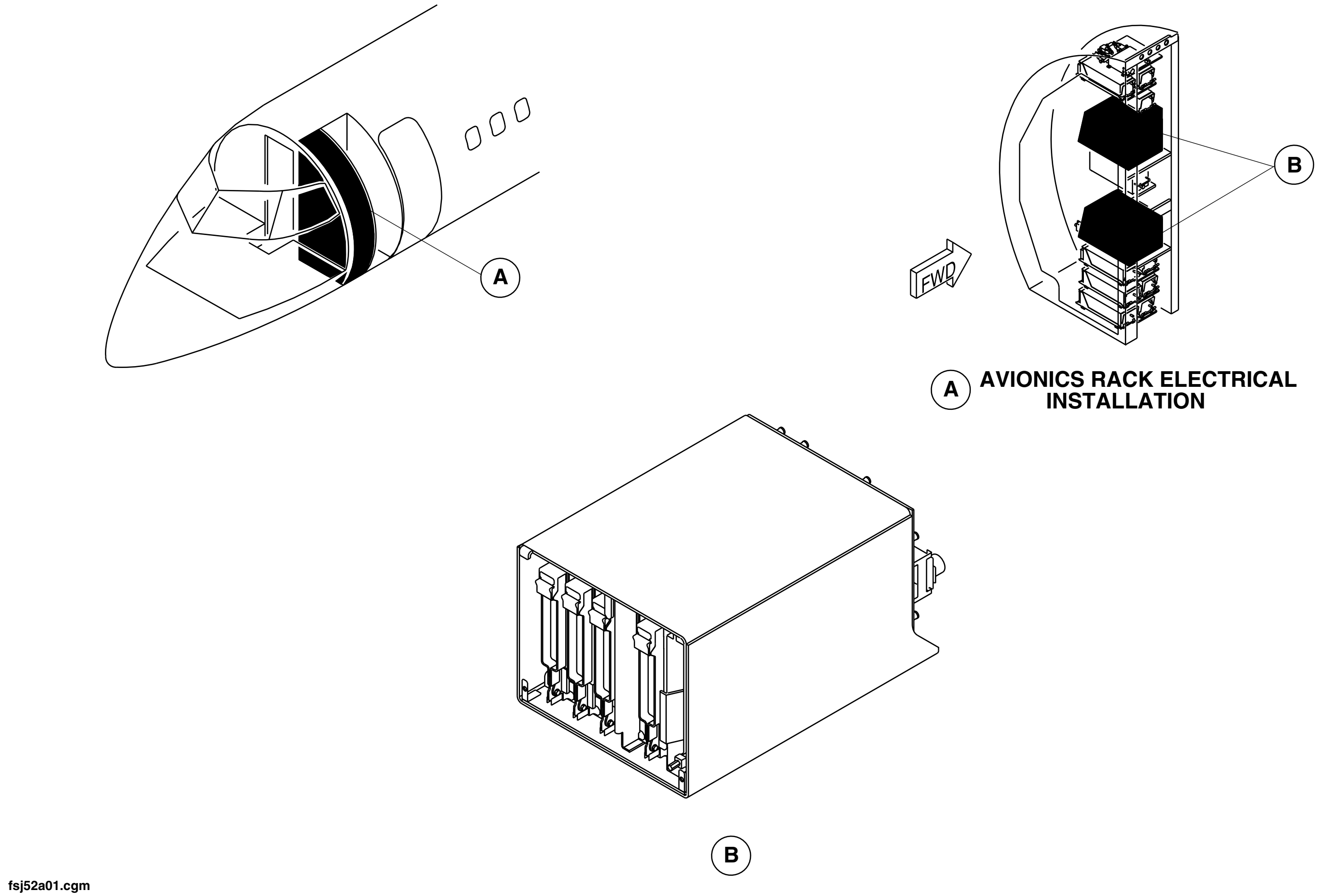


FDPS PARKING BRAKE HYDRAULIC PRESSURE CALCULATION
Figure 81

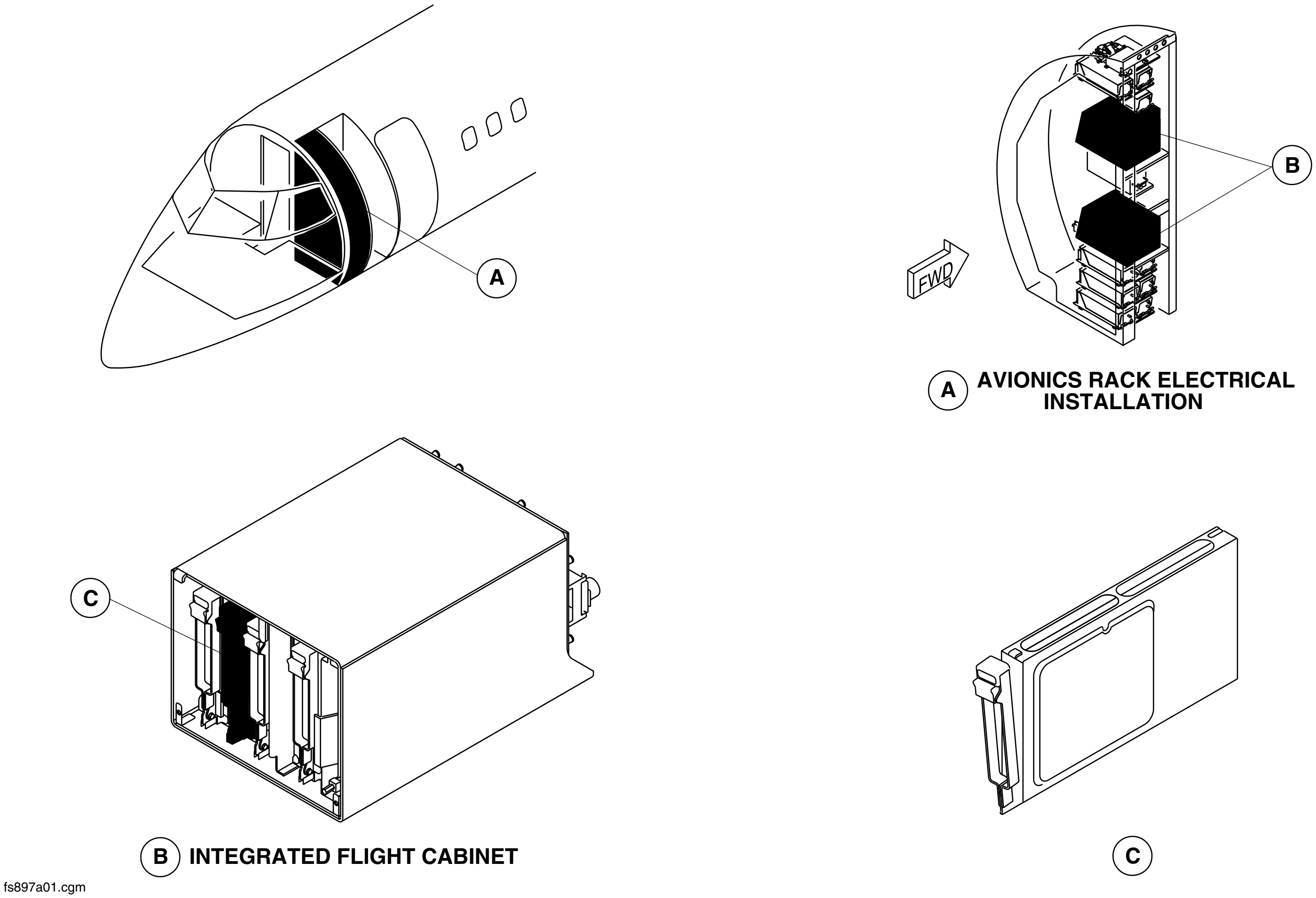


fsh17a01.cgm

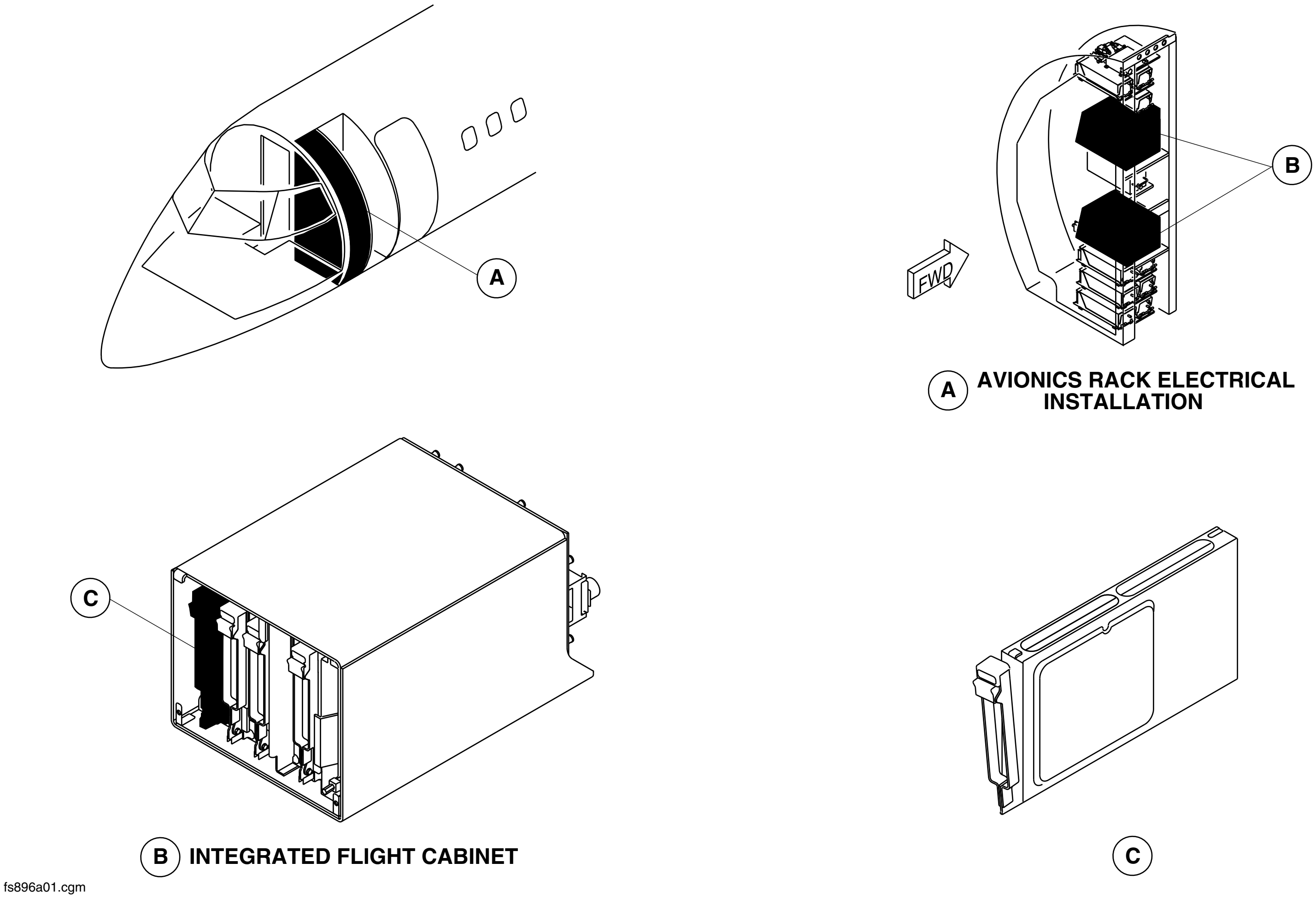
FDPS AIRCRAFT CONFIGURATION MANAGEMENT
Figure 82



INTEGRATED FLIGHT CABINETS, LOCATOR
Figure 83



INPUT/OUTPUT PROCESSORS, LOCATOR
Figure 84



INPUT/OUTPUT MODULES, LOCATOR
Figure 85

**ON A/C ALL

31–50–00–001

CENTRAL WARNING SYSTEM

Introduction

The central warning system has different sub–systems to do the functions that follows:

- Show caution and warning lights
- Show advisory lights
- Take–off warning calculation.

General Description

Refer to Figure 1.

The indicating and recording system has the sub–systems that follow:

- 34–51–00 Caution and Warning Lights System
- 34–52–00 Take–off Warning System.

Caution and Warning Lights System

The caution and warning light system is divided into the two parts that follow:

- Caution and warning lights
- Advisory lights.

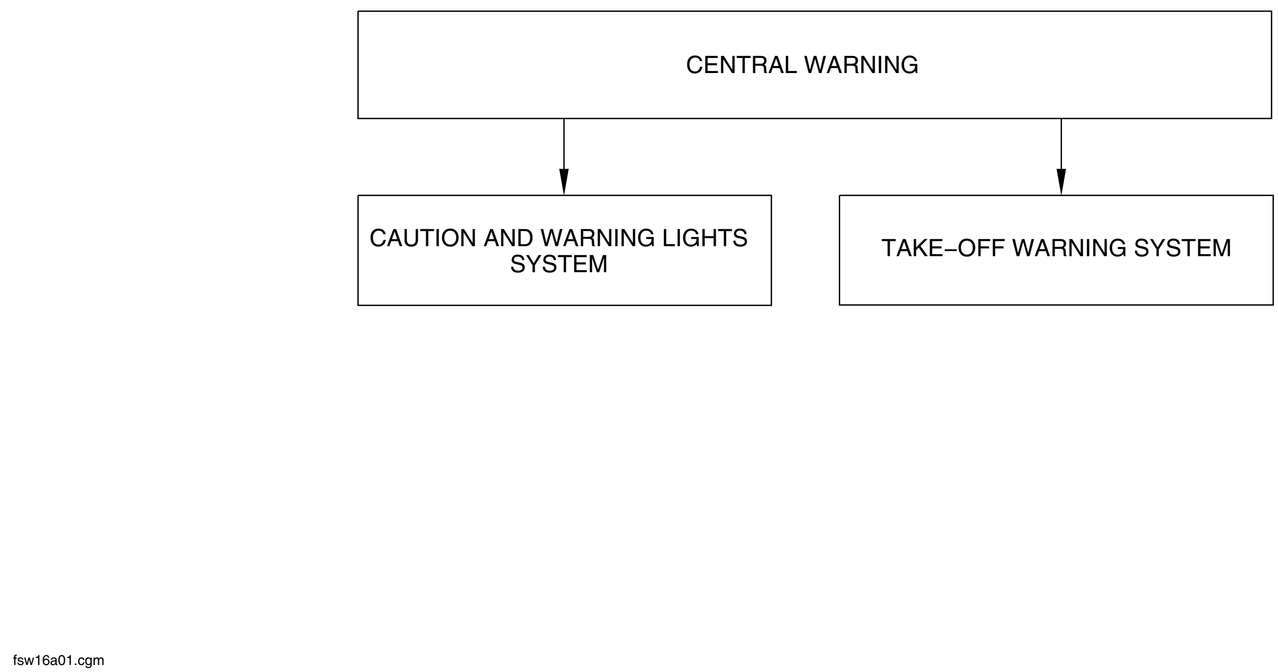
Caution and Warning Lights: The caution and warning lights system shows system malfunctions and other conditions that require a corrective action.

Advisory Lights: The advisory lights show malfunctions and other conditions that require a corrective action and safe and normal system operation.

Take–off Warning System

Refer to 31–40–00, Central Computer.

The take–off warning system supplies an aural warning when a take–off is attempted with the aircraft not in the correct take–off configuration.



CENTRAL WARNING SYSTEM BLOCK DIAGRAM
Figure 1

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**ON A/C ALL

31–51–00–001

CAUTION AND WARNING LIGHTS SYSTEM

Introduction

The caution and warning light system is divided into the two parts that follow:

– Caution and warning lights
– Advisory lights.

Caution and Warning Lights: The caution and warning lights system shows system malfunctions and other conditions that require a corrective action.

Advisory Lights: The advisory lights show malfunctions and other conditions that require a corrective action and safe and normal system operation.

General Description

The caution and warning and advisory lights system has color attributes that follow:

– Red
– Amber or yellow
– Green, white, or blue.

The red warning lights show system malfunctions or the conditions that cause dangerous flight conditions. It shows a malfunction or condition that requires an immediate corrective action.

The amber or yellow caution lights show system malfunctions or conditions that do not immediately affect the aircraft flight capabilities. It shows that a future corrective action is possibly necessary.

The green lights show safe system operation and white or blue lights show normal operation.

Refer to Figure 1.

The caution and warning panel receives input caution and warning signals from the different aircraft systems, and shows them on the caution and warning panel annunciator segments. The caution and warning panel is attached to the overhead panel frame.

Refer to Figures 2 and 3.

When the caution and warning panel senses a warning condition, it causes a related red warning annunciator to come on flashing with the pilot glareshield panel master warning switch and three chimes sounds. The master warning switch has a WARNING PRESS TO RESET label.

On aircraft with ModSum 4–410700 or SB84–33–09 incorporated, an additional master warning switch labeled WARNING PRESS TO RESET is installed on the co–pilot glareshield panel.

The caution and warning panel warning annunciator and the master warning switch indication flashes until the master warning switch is pushed. A switch selection causes the caution and warning panel warning indication to change to steady and the master warning indication to go out. When the condition is corrected, the caution and warning panel warning indication goes out.

NOTE

The No. 1 or No. 2 engine oil pressure red warning indication comes on steady. It will not flash.

Refer to Figures 2 and 3.

When the caution and warning panel senses a caution condition, it causes a related yellow caution annunciator to come on with the copilot glareshield panel master caution switch and one chime sounds. The master caution switch has a CAUTION PRESS TO RESET label.

On aircraft with ModSum 4–410700 or SB84–33–09 incorporated, an additional master caution switch labeled CAUTION PRESS TO RESET is installed on the pilot glareshield panel.

The caution and warning panel caution annunciator and the master caution switch indication stays on until the master warning switch is pushed. A switch selection causes the master caution indication to go out. When the condition is corrected, the caution and warning panel caution indication goes out.

NOTE

The caution and warning panel FUELING ON caution light will not cause the master caution indicator to come on.

Refer to Figure 4.

Advisory lights are located on the panels that follow:

– Overhead
– Glareshield
– Instrument
– Center console
– Pilot side console.

NOTE

An amber or yellow advisory indication does not cause the master caution light to come on and a red advisory indication does not cause the master warning light to come on.

An Advisory Lights Control Unit (ACU) is used to test and control the brightness of the advisory annunciators. It is controlled by the CAUTION/ADVSY LIGHTS toggle switches located on the passenger warning panel.

The signals from different aircraft systems are supplied to the their related advisory annunciator through the ACU. A loss of electrical power to the ACU will not prevent individual indicators from coming on when it receives an advisory light signal. The advisory test, dim or bright selections will have no effect.

The caution and warning light system has the components that follow:

– Panel, Caution and Warning (31–51–01)
– Unit, Advisory Control (31–51–06)
– Switch, Master Warning (31–51–11)
– Switch, Master Caution (31–51–11)
– Switch, Test Caution/Advisory (31–51–26)
– Switch, Dim/Bright (31–51–26).

Detailed Description

There are five different types of inputs to the caution and warning panel:

- Type I, supplied 28 V dc
- Type II, supplied ground
- Type III, 28 V dc removed
- Type IV, ground removed
- Type V, combination.

Type I: A malfunction condition is present if the input to the caution and warning panel is more than 10 V dc.

Type II: A condition is present if the impedance to ground potential is less than 50 Ω.

Type III: A condition is present if the input to the caution and warning panel is less than 10 V dc.

Type IV: A condition is present if the impedance to ground potential is more than 2000 Ω or if the total voltage across a switching element and a 50 Ω resistor is more than 4.5 V dc.

Type V: A condition is present if the input is a combination of type I, II, III, or IV inputs.

There are five other control inputs from the caution and advisory lights toggle switches to the caution and warning panel:

- Master warning reset, 28 V dc
- Master caution reset, 28 V dc
- Test, 28 V dc
- Dim, 28 V dc
- Bright, 28 V dc.

A system malfunction or condition causes its related advisory annunciator to come on. The annunciator stays in view until the malfunction or condition is corrected.

There are two different types of inputs from the caution and advisory lights toggle switches to the caution and warning panel:

- Type I, supplied 28 V dc
- Type II, supplied ground.

Type I: A condition is present if the input to the ACU is more than 10 V dc.

Type II: A condition is present if the impedance to ground potential is less than 50 Ω.

There are three other control inputs to the ACU:

- Test, 28 V dc
- Dim, 28 V dc
- Bright, 28 V dc.

Refer to Figure 5.

The warning light indications at the bottom row of the caution and warning panel is summarized in the table shown.

Refer to Figure 6.

The caution light indications at the top part of the caution and warning panel is summarized in the tables shown.

On aircraft with ModSum 4–309222 OR 4–309227 OR SB84–52–48 incorporated, the caution light indication INTERNAL BAGG DOOR on the caution and warning panel is changed to INTERNAL DOORS.

On aircraft with ModSum 4–458953 incorporated, the caution light indication INTERNAL DOORS on the caution and warning panel is changed to COCKPIT DOOR.

Refer to Figure 7.

The advisory light indications is summarized in the tables shown.

Refer to Figure 8.

The warning, caution, and advisory annunciator lights are initially set to bright when the systems are energized. The DIM, BRT toggle switch located on the passenger warning panel is momentarily set to the DIM position to make the advisory and caution annunciators lights and compass light change to dim.

The DIM, BRT toggle switch located on the passenger warning panel is momentarily set to the BRT position to make the advisory and caution annunciators lights and compass light change back to bright.

The TEST CAUT/ADVSY toggle switch on the passenger warning panel is set to TEST CAUT position to test the caution and warning lights on the caution warning panel and the master caution and master warning indication.

The caution and warning test gives the indications that follow:

- Warning lights come on bright and flashing
- Caution lights come on bright
- Master warning comes on flashing
- Master caution comes on flashing
- Three chimes sound.

The TEST CAUT/ADVSY toggle switch on the passenger warning panel is set to TEST ADVSY position to test the advisory annunciator lights.

The aircraft right essential and right main buses supply 28 V dc electrical power through two 7.5 A circuit breakers to the caution and warning panel. The circuit breaker that is connected to the right essential bus also supplies electrical power to the master warning and master caution switches and its related part of the DIM, BRT and TEST CAUT/ADVSY toggle switches. The circuit breakers are located in position M4 and N4 on the 28 right circuit breaker panels.

The aircraft left essential bus supplies 28 V dc electrical power through a 7.5 A circuit breaker to the ACU and its related part of the DIM, BRT and TEST CAUT/ADVSY toggle switches. The circuit breaker is located in position F3 on the 28 left circuit breaker panel.

Caution and Warning Annunciator Panel

Refer to Figure 9.

Caution and warning fault signals caused by important malfunctions or conditions are supplied to the caution and warning panel. These malfunctions or conditions are shown on the ninety six active annunciator segments on the panel. The panel is divided into 14 modules. Each module is further divided into seven segments, six yellow cautions and one red warning annunciator at the bottom. Each segment is labelled with an appropriate legend that is back lit with a LED. The legends are not readable until the LEDs come on.

Each module has a bezel and lens assembly that is keyed and marked with the module number. It is pulled removed from the module to get access to the LEDs.

The caution and warning panel weighs 6 lb (2.72 kg) and is 15.40 in. (391 mm) long, 3.35 in. (85.1 mm) high, and 3.50 in. (88.9 mm) deep. The unit is attached to the overhead panel frame of the aircraft with four dzus fasteners.

Advisory Lights Control Unit

Refer to Figure 10.

The Advisory Lights Control Unit (ACU) tests and controls the brightness of the advisory annunciators.

The advisory control unit receives 28 V dc input or ground input signals from different aircraft systems and supplies them to the related advisory lights on the aircraft.

The Advisory Control Unit (ACU) weighs 4.2 lb (1.9 kg) and is 12.55 in. (319 mm) long, 5.51 in. (140 mm) wide and 5.50 in. (139.7 mm) deep. The advisory control unit is attached to the bottom shelf of the right DC circuit breaker console with seven mounting screws.

Master Warning Switch

Refer to Figure 11.

The master warning switch has a pushbutton reset switch and a WARNING PRESS TO RESET annunciator in red letters on a black background. It is back lit by a LED when a warning signal is received. The master warning switch is located on the pilot glareshield panel and is secured by two mounting screws.

On aircraft with ModSum 4–410700 or SB84–33–09 incorporated, a master warning switch is also installed on the co–pilot glareshield panel.

Master Caution Switch

Refer to Figure 12.

The master caution switch has a pushbutton reset switch and a CAUTION PRESS TO RESET annunciator in yellow letters on a black background. It is back lit by a LED when a caution signal is received. The master caution switch is located on the copilot glareshield panel and is secured by two mounting screws.

On aircraft with ModSum 4–410700 or SB84–33–09 incorporated, a master caution switch is also installed on the pilot glareshield panel.

Dim/Bright Switch

Refer to Figure 13.

The warning, caution, and advisory annunciator lights are initially set to bright when the systems are energized. The DIM, BRT toggle switch is momentarily set to the DIM position to make the advisory and caution annunciators lights and compass light change to dim. It is momentarily set to the BRT position to make the advisory and caution annunciators lights and compass light change back to bright.

The dim/bright switch is a four pole, two position center–off toggle switch located on the right side of the passenger warning panel. It has a toggle lever that springs back to the center off position when released. The toggle lever is connected to a mechanism in the body of the switch that moves in the opposite direction as the toggle lever. The toggle switch has wire lug terminals for wiring connections.

The dim/bright switch is attached to the passenger warning panel assembly with a mounting nut and lock washer. The passenger warning panel assembly is attached to the overhead console with four dzus fasteners.

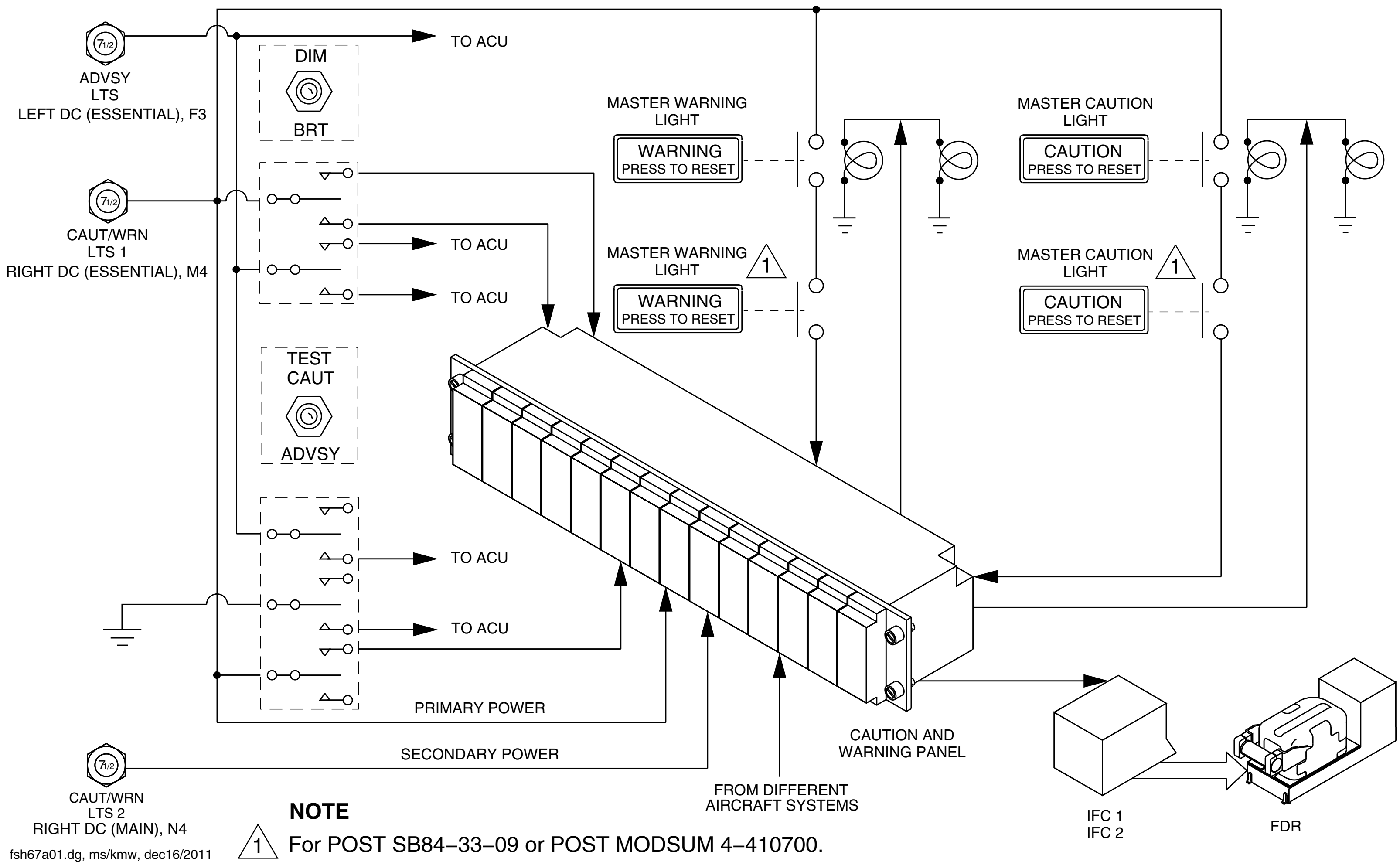
Caution/Advisory Test Switch

Refer to Figure 14.

The caution/advisory test switch is set to TEST CAUT position to test the caution and warning lights on the caution warning panel and the master caution and master warning indication. It is set to TEST ADVSY position to test the advisory annunciator lights.

The caution/advisory test switch is a 6 pole, two position center–off toggle switch located on the right side of the passenger warning panel. It has a toggle lever that springs back to the center off position when released. The toggle lever is connected to a mechanism in the body of the switch that moves in the opposite direction as the toggle lever. The toggle switch has wire lug terminals for wiring connections.

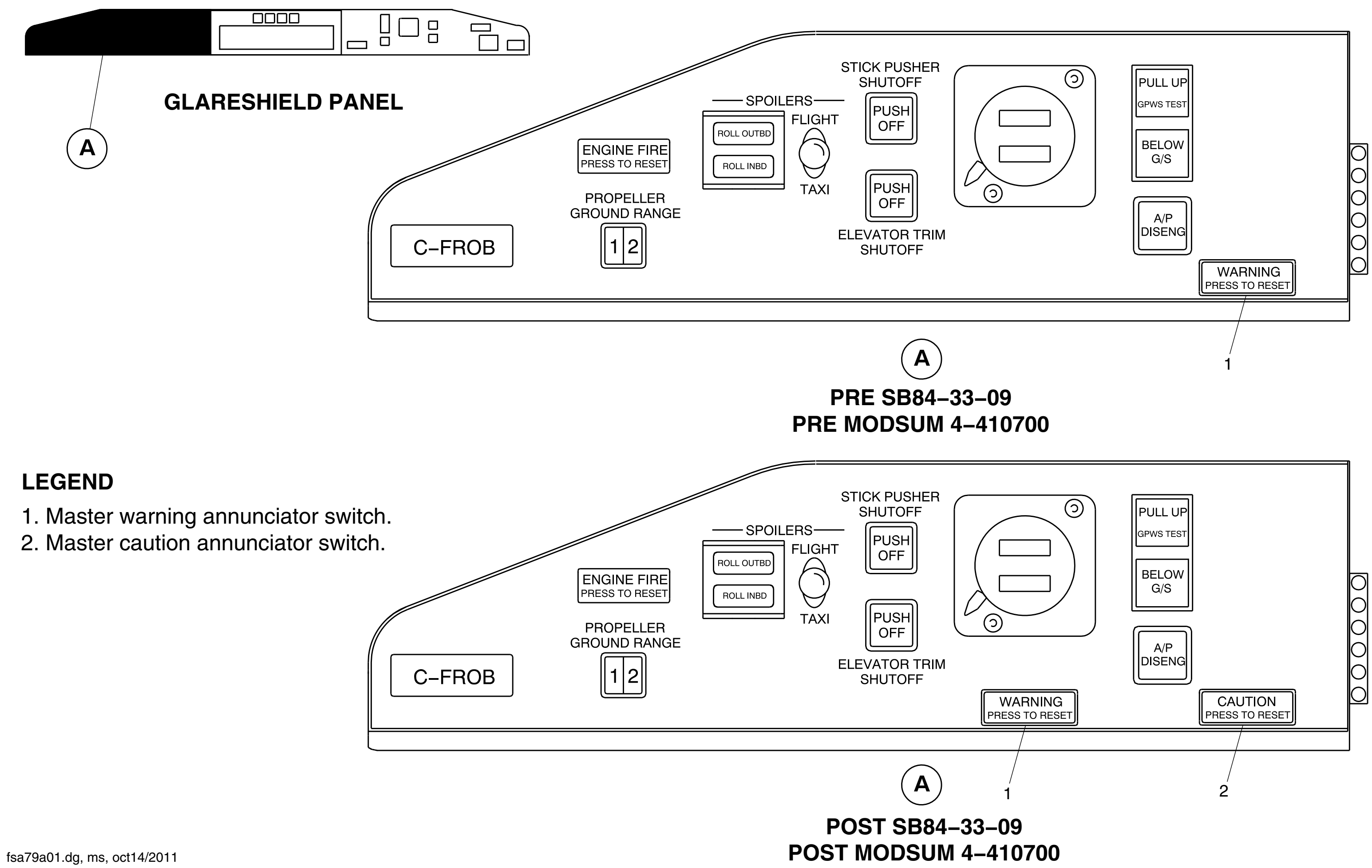
The caution/advisory test switch is attached to the passenger warning panel assembly with a mounting nut and lock washer. The passenger warning panel assembly is attached to the overhead console with four dzus fasteners.



NOTE

1 For POST SB84–33–09 or POST MODSUM 4–410700.

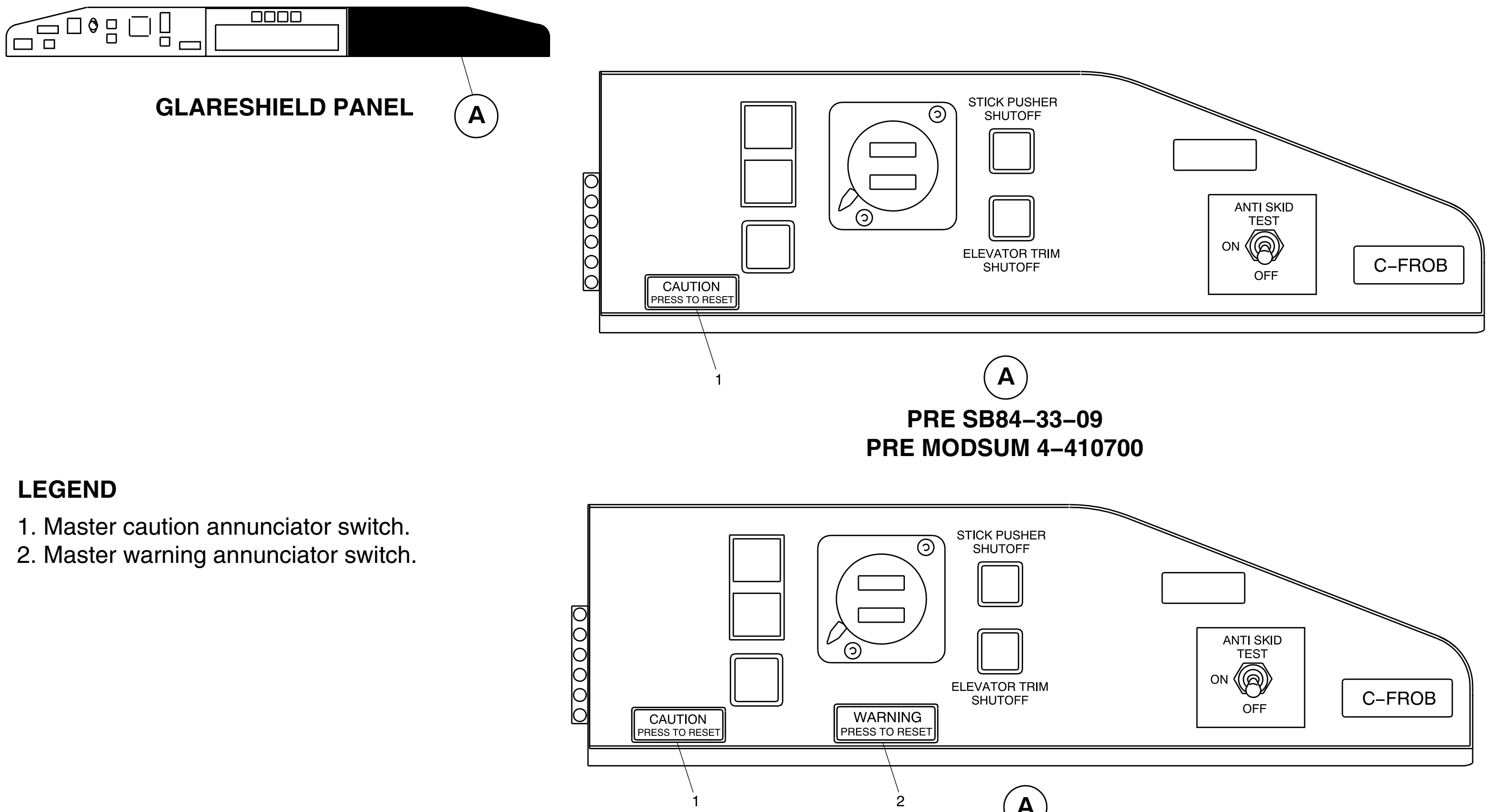
CAUTION AND WARNING LIGHTS SYSTEM BLOCK DIAGRAM
Figure 1



LEGEND

1. Master warning annunciator switch.
2. Master caution annunciator switch.

Pilot Glareshield Panel, Master Caution/Warning Indication
Figure 2



LEGEND

1. Master caution annunciator switch.
2. Master warning annunciator switch.

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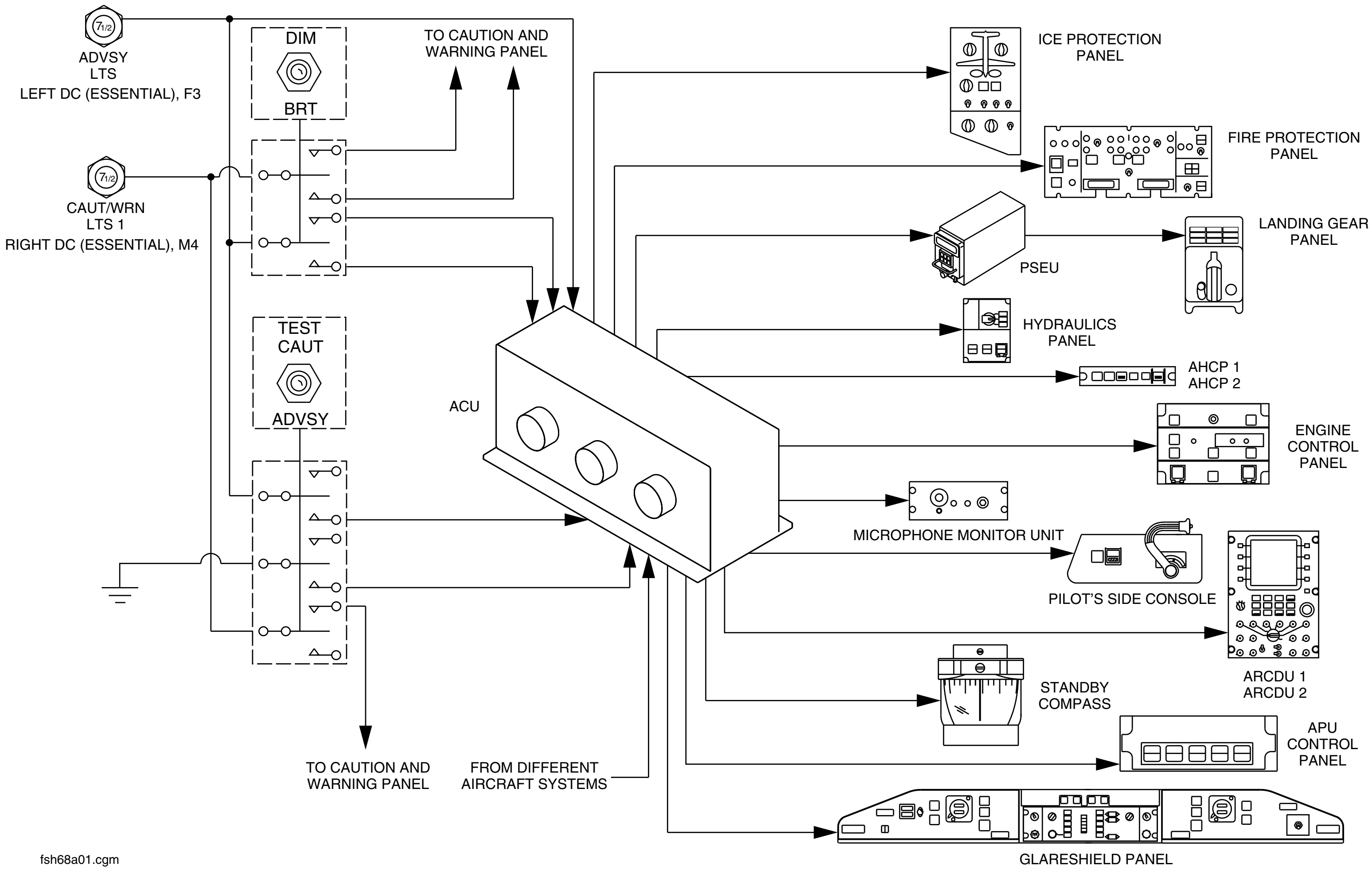
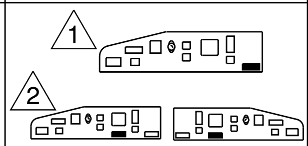
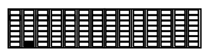
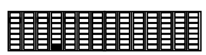
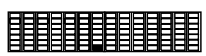
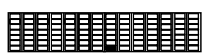
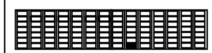
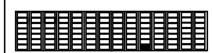
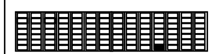
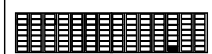
Co–pilot Glareshield Panel, Master Caution/Warning Indication
Figure 3



ADVISORY LIGHTS SYSTEM BLOCK DIAGRAM
Figure 4

WARNING INDICATIONS

Annunciator	Panel Location	Cause for Illumination
WARNING PRESS TO RESET		Comes on flashing with a flashing warning annunciator.
WARNING PRESS TO RESET		Comes on flashing with a flashing warning annunciator.
CABIN PRESS		Cabin altitude is more than 10,000 ft. (3.05 km).
CHECK FIRE DET		Engine fire, APU fire, or detector loop circuit malfunction is sensed.
SMOKE		Smoke is sensed in baggage area.
TOUCHED RUNWAY		Rear outer fuselage surface below pressure bulkhead has touched the runway during landing or take−off.
#1 ENG OIL PRESS		No. 1 engine oil pressure is less than 42 psi (289.58 kPa).
#1 ENG FADEC FAIL		No. 1 Full Authority Digital Engine Control has malfunctioned.

Annunciator	Panel Location	Cause for Illumination
#2 ENG FADEC FAIL		No. 2 full authority digital engine control has malfunctioned.
#2 ENG OIL PRESS		No. 2 engine oil pressure is less than 42 psi (289.58 kPa).
STBY BAT HOT		Battery temperature is more than 71 °C (160 °F).
AUX BAT HOT		Battery temperature is more than 71 °C (160 °F).
MAIN BAT HOT		Battery temperature is more than 71 °C (160 °F).
FUSELAGE DOORS		One or more fuselage doors are not closed.

NOTES

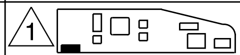
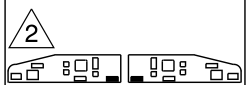
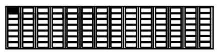
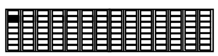
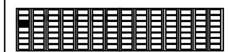
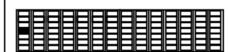
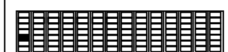
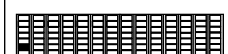
1. For PRE SB84−33−09 and PRE MODSUM 4−410700.

2. For POST SB84−33−09 or POST MODSUM 4−410700.

fsm97a01.dg, ms/kmw, dec16/2011

Warning Light Indications
Figure 5

CAUTION INDICATIONS

Annunciator	Panel Location	Cause for Illumination
CAUTION PRESS TO RESET		Comes on flashing with a flashing caution annunciator.
CAUTION PRESS TO RESET		Comes on flashing with a flashing caution annunciator.
PITCH TRIM		One of the FCECU channels is not able to control its related pitch trim actuator or pilot and copilot pitch trim input commands are different. Airspeed 1 and 2 mismatch (RUD PRESS, SPLR OUTBD, and ELEVATOR FEEL caution lights will also come on).
ELEVATOR FEEL		One of the FCECU channels is not able to control the pitch feel actuator or acceleration 1 and 2 mismatch. Airspeed 1 and 2 mismatch (RUD PRESS, SPLR OUTBD, and PITCH TRIM caution lights will also come on).
ELEVATOR ASYMMETRY		FCECU senses a left and right elevator mismatch more than 5 degrees.
ELEVATOR PRESS		Hydraulic system 1, 2 and 3 pressure switches are in the high position but if one of the switches is in the low position, the caution light will not illuminate.
#3 STBY HYD PUMP		Direct Current Motor Pump (DCMP) contactor has been energized for more than 60 seconds and the WOW condition is true or the pressure drops below 900 psi (6205.30 kPa).
EMER LTS DISARMED		Emergency lights are not armed.

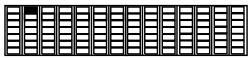
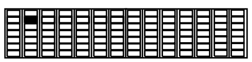
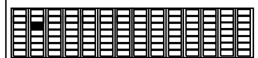
NOTES

1. For PRE SB84–33–09 and PRE MODSUM 4–410700.
2. For POST SB84–33–09 or POST MODSUM 4–410700.

fsm98a01.dg, ms/kmw, dec19/2011

Caution Light Indications
Figure 6 (Sheet 1 of 9)

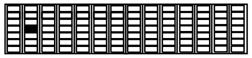
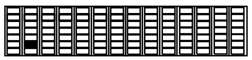
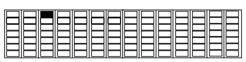
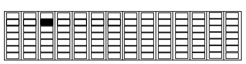
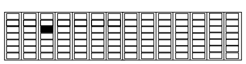
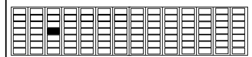
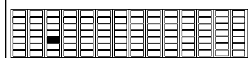
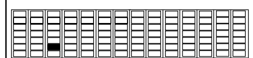
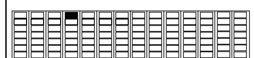
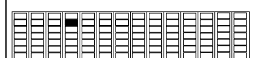
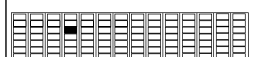
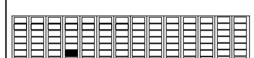
CAUTION INDICATIONS

Annunciator	Panel Location	Cause for Illumination
L AC BUS		If the related AC GCU senses an overcurrent condition for more than seven seconds. If the generator line contactor K1 is in the center position while the generator switch is in the ON position, the generator frequency is more than 300 Hz and the engine rpm value is more than 600 rpm.
R AC BUS		If the related AC GCU senses an overcurrent condition for more than seven seconds. If the generator line contactor K2 is in the center position while the generator switch is in the ON position, the generator frequency is more than 300 Hz and the engine rpm value is more than 600 rpm.
DC BUS		When the DC generator overcurrent condition is sensed by the #1 DC GCU or #2 DC GCU or #3 DC GCU (APU) for more than seven seconds. When this occurs, the GCU will send a signal to the EPCU, which opens (or inhibits closing of) the bus tie. If the fault is not cleared, the EPCU will send a command to the GCU to open the applicable channel. But if the GCU does not receive a trip signal in five seconds and still senses the overcurrent in its related generator, it will disconnect the generator from the bus. All this occurs in less than seven seconds.
L TRU		AC secondary feeder bus is not energized by its related TRU because of a source fault condition or the ac generator toggle switches are set to the OFF position.

fsm98a02.dg, hp/kmw, jul08/2013

Caution Light Indications
Figure 6 (Sheet 2 of 9)

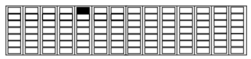
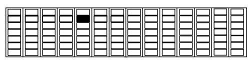
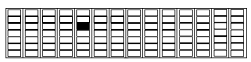
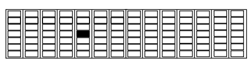
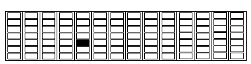
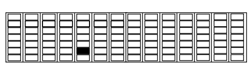
CAUTION INDICATIONS

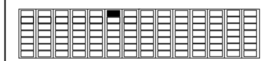
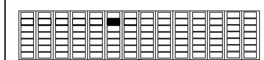
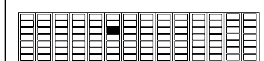
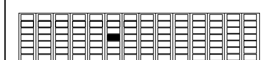
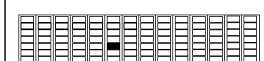
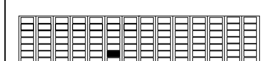
Annunciator	Panel Location	Cause for Illumination
R TRU		AC secondary feeder bus is not energized by its related TRU because of a source fault condition or the ac generator toggle switches are set to the OFF position.
INTERNAL BAGG DOOR		Forward baggage compartment internal door is open. On aircraft with MS 4–309222 OR MS 4–309227 OR SB84–52–48 incorporated, this indication is INTERNAL DOORS and with MS 4–458953 incorporated, this indication is COCKPIT DOOR – comes on when the forward baggage compartment internal door is open or the flight compartment door is unlatched.
PITOT HEAT STBY		PITOT/STATIC PORTS, STBY toggle switch set to OFF or standby pitot/static port heater malfunction.
PITOT HEAT 1		PITOT/STATIC PORTS, STBY toggle switch set to OFF or standby pitot/static port heater malfunction.
ENG ADPT HEAT 1		No. 1 engine malfunction of its 2 heating elements.
WSHLD CTRL		Left or right anti–ice control malfunction.
WSHLD HOT		Windshield temperature is more than 50 °C(122 °F).
SIDE WDO HOT		Pilot's side window temperature is more than 52 °C (126 °F).
ICE DETECT FAIL		Ice detectors have malfunctioned.
PITOT HEAT 2		PITOT/STATIC PORTS, STBY toggle switch set to OFF or standby pitot/static port heater malfunction.
ENG ADPT HEAT 2		No. 2 engine malfunction of its 2 heating elements.
PROP DEICE		Time Monitor Control Unit has malfunctioned or it senses a malfunction in the system.
DEICE TIMER		Timer and monitor unit malfunction.
DEICE PRESS		Airframe deice system pressure is too low for deicer boot operation.

fsm98a03.dg, js/vk, may13/2014

Caution Light Indications
Figure 6 (Sheet 3 of 9)

CAUTION INDICATIONS

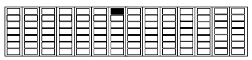
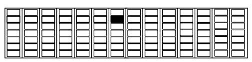
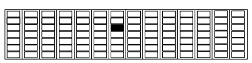
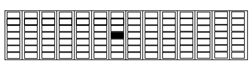
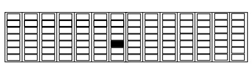
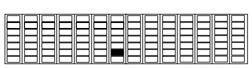
Annunciator	Panel Location	Cause for Illumination
FLT COMPT DUCT HOT		Flight compartment supply duct temperature is more than 213 °C (415 °F).
CABIN DUCT HOT		Cabin supply duct temperature is more than 213 °C (415 °F).
CABIN PACK HOT		Cabin supply duct temperature is more than 88 °C (190 °F).
FLT COMPT PACK HOT		Cabin air cycle machine compressor discharge temperature is more than 88 °C (190 °F).
L TRU HOT		Left TRU temperature is more than 71 °C (160 deg °F).
R TRU HOT		Right TRU temperature is more than 71 °C (160 °F).

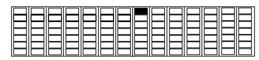
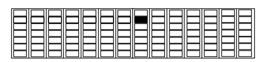
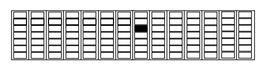
Annunciator	Panel Location	Cause for Illumination
#1 HYD ISO VLV		No.1 isolation valve switch hydro–mechanically has moved to the isolate position.
#2 HYD ISO VLV		No. 2 isolation valve switch hydro–mechanically has moved to the isolate position.
STBY BATTERY		Standby battery is not connected to its main feeder bus for charging.
MAIN BATTERY		Main battery is not connected to its main feeder bus for charging.
AUX BATTERY		Auxiliary battery is not connected to its main feeder bus for charging.
AVIONICS		TBD

fsm98a04.dg, js, nov24/2011

Caution Light Indications
Figure 6 (Sheet 4 of 9)

CAUTION INDICATIONS

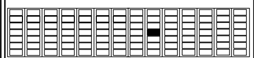
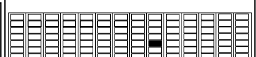
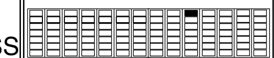
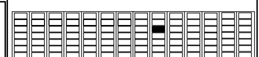
Annunciator	Panel Location	Cause for Illumination
#1 PEC		No. 1 PEC senses a Beta mode malfunction (Beta mode – PLA just above flight idle to discing).
#1 BLEED HOT		No. 1 bleed air supply duct temperature is more than 354 °C (670 °F).
#1 DC GEN HOT		No. 1 DC generator temperature is more than 166 °C (330 °F).
#1 AC GEN HOT		No. 1 AC generator temperature is more than 166 °C (330 °F).
#1 HYD FLUID HOT		No. 1 reservoir temperature switch is more than 107 °C (225 °F).
#1 FUEL FLTR BYPASS		No. 1 engine filter is about to go into bypass.

Annunciator	Panel Location	Cause for Illumination
#1 ENG FUEL PRESS		Pressure at engine–driven pump No.1 inlet is low, ejector pump failure or loss of motive flow to the ejector pump.
#1 ENG HYD PUMP		Engine Driven Pump 1 pressure switch is less than 1,800 psi (12410.56 kPa) or when the condition lever is set to the fuel cut off position.
#1 DC GEN		No. 1 dc generator is not connected to its bus because of a source fault condition, the dc starter/generator toggle switch is set to the OFF position, or the external dc power is energizing the left and right main feeder busses.

fsm98a05.dg, js, nov24/2011

Caution Light Indications
Figure 6 (Sheet 5 of 9)

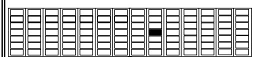
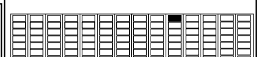
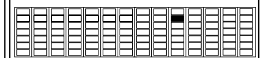
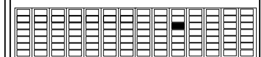
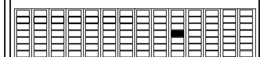
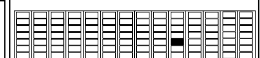
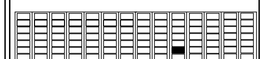
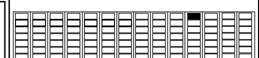
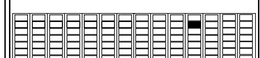
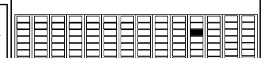
CAUTION INDICATIONS

Annunciator	Panel Location	Cause for Illumination
#1 AC GEN		The AC generator toggle switch is set to the off position, AC generator rotating frequency is less than 300 Hz and the engine RPM value is less than 600 rpm, AC generator rotating frequency is more than 300 Hz and the GCU has tripped the generator (other than the overcurrent fault), external AC power is energizing the L and R AC buses, AC GCU has issued a command to close the line contactor but the line contactor has not closed in less than 100 millisecond, or K1 status relay logic is interrupted for more than 500 millisecond.
#1 ENG FADEC		No. 1 FADEC sensed a condition that requires caution when applying control inputs.
#1 TANK FUEL LOW		No. 1 fuel tank senses 305 lbs (138.35 kg) of fuel remaining.
#2 ENG FUEL PRESS		Pressure at engine–driven pump No. 2 inlet is low, ejector pump is low, ejector pump failure or loss of motive flow to the ejector pump.
#2 ENG HYD PUMP		Engine driven pump No. 2 pressure switch is less than 1,800 psi (12410.56 kPa) or when the condition lever is set to the fuel cut off position.
#2 DC GEN		No. 2 DC generator is not connected to its bus because of a source fault condition, the DC starter/generator toggle switch is set to the OFF position, or the external DC power is energizing the left and right main feeder busses.

fsm98a06.dg, js/hp, jun14/2013

Caution Light Indications
Figure 6 (Sheet 6 of 9)

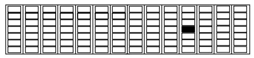
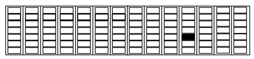
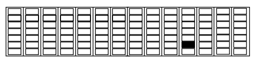
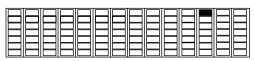
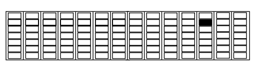
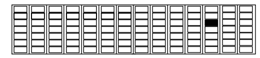
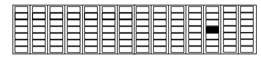
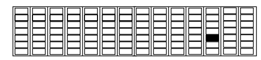
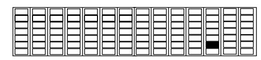
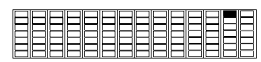
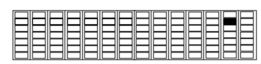
CAUTION INDICATIONS

Annunciator	Panel Location	Cause for Illumination
#2 AC GEN		The AC generator toggle switch is set to the off position, AC generator rotating frequency is less than 300 Hz and the engine RPM value is less than 600 rpm, AC generator rotating frequency is more than 300 Hz and the GCU has tripped the generator (other than the overcurrent fault), external AC power is energizing the L and R AC buses, AC GCU has issued a command to close the line contactor but the line contactor has not closed in less than 100 millisecond, or K2 status relay logic is interrupted for more than 500 millisecond.
#2 ENG FADEC		No. 2 FADEC sensed a condition that requires caution when applying control inputs.
#2 TANK FUEL LOW		No. 2 fuel tank senses 305 lbs (138.35 kg) of fuel remaining.
#2 PEC		No. 1 PEC senses a Beta mode malfunction (Beta mode – PLA just above flight idle to discing).
#2 BLEED HOT		No. 2 bleed air supply duct temperature is more than 354 °C (670 °F).
#2 DC GEN HOT		No. 2 DC generator temperature is more than 166 °C (330 °F).
#2 AC GEN HOT		No. 2 AC generator temperature is more than 166 °C (330 °F).
#2 HYD FLUID HOT		No. 1 reservoir temperature switch is more than 107 °C (225 °F).
#2 FUEL FLTR BYPASS		No. 2 engine filter is about to go into bypass.
ROLL SPLR INBD GND		FCECU is not able to control the inboard spoiler lift dump operation, the inboard spoilers do not extend at touchdown or the inboard spoilers extend after the TAXI mode selection after touchdown.
#1 STBY HYD PUMP HOT		The temperature of the electric motor windings on the No. 1 standby hydraulic pump is too hot.
ROLL SPLR INBD HYD		No hydraulic pressure to the inboard roll spoiler actuators.

fsm98a07.dg, js/hp, jun14/2013

Caution Light Indications
Figure 6 (Sheet 7 of 9)

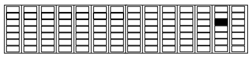
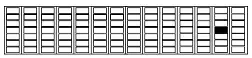
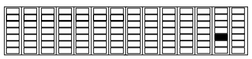
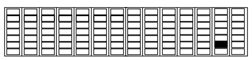
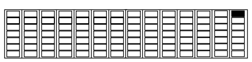
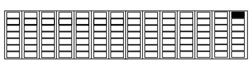
CAUTION INDICATIONS

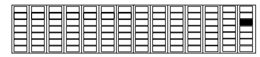
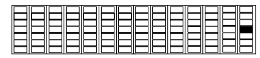
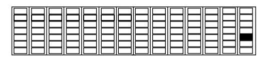
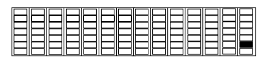
Annunciator	Panel Location	Cause for Illumination
#1 RUD HYD		Hydraulic pressure to the No. 1 rudder actuator is less than 550 ±50 psi (3792.12 ±344.74 kPa).
RUD CNTRL		No hydraulic pressure to the rudder actuators.
APU		APU FADEC has sensed a fault and shut down the APU.
ROLL SPLR OUTBD GRD		FCECU is not able to control the outboard spoiler lift dump operation, the outboard spoilers do not extend at touchdown or the outboard spoilers extend after the TAXI mode selection after touchdown.
SPLR OUTBD		FCECU has failed to inhibit at greater than 170 KIAS or has failed to re–enable below 165 KIAS.
ROLL SPLR OUTBD HYD		No. 2 hydraulic system pressure is less than 900 psi (6205.28 kPa).
#2 RUD HYD		Hydraulic pressure to the No. 2 rudder actuator is less than 550 ±50 psi (3792.12 ±344.74 kPa).
FLAP DRIVE		Flap drive system malfunction.
FLAP POWER		Low hydraulic pressure in flap power unit during flap extension or retraction.
#1 STALL SYST FAIL		No. 1 stall warning system malfunction or no vane heater current is detected.
#2 STALL SYST FAIL		No. 2 stall warning system malfunction or no vane heater current is detected.

fsm98a08dg, js/kmw, dec16/2011

Caution Light Indications
Figure 6 (Sheet 8 of 9)

CAUTION INDICATIONS

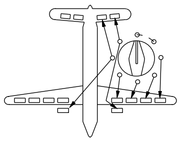
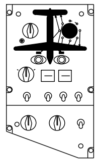
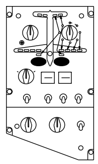
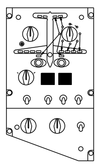
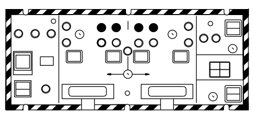
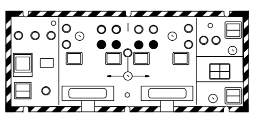
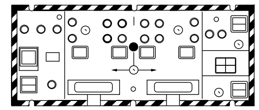
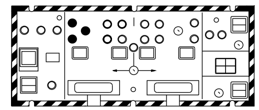
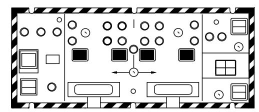
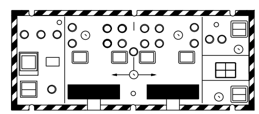
Annunciator	Panel Location	Cause for Illumination
PUSHER SYST FAIL		Stick pusher malfunction or no vane heater current is sensed.
INBD ANTISKID		Inboard antiskid control unit malfunction or set to OFF.
LDG GEAR INOP		Landing gear sequence malfunction after a gear retraction.
NOSE STEERING		Nosewheel steering ECU malfunction or nose–wheel castered beyond 70 degrees.
FLT DATA RECORDER		Flight data recorder malfunction or it is not energized.
GPWS		Ground Proximity Warning System (GPWS) becomes unserviceable.

Annunciator	Panel Location	Cause for Illumination
PARKING BRAKE		Emergency/parking brake is set on.
OUTBD ANTISKID		Outboard antiskid control unit malfunction or it is not energized.
WT ON WHEELS		PSEU weight–on–wheels sensing circuit malfunction.
FUELING ON		Refuel/Defuel panel access panel is open and power is applied to the panel.

fsm98a09.dg, js, nov24/2011

Caution Light Indications
Figure 6 (Sheet 9 of 9)

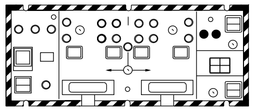
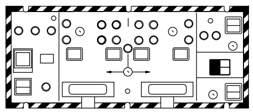
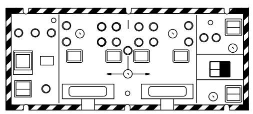
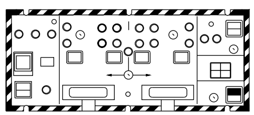
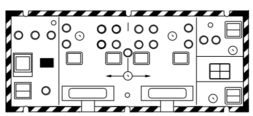
ADVISORY LIGHTS – OVERHEAD PANEL

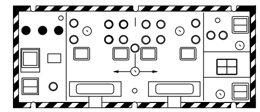
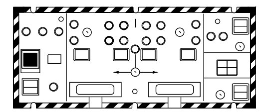
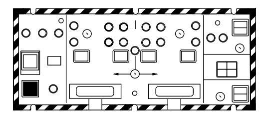
Annunciator	Panel Location	Cause for Illumination
		Boot inflation is 15 psi (103.42 kPa) or more.
PROPS PROPS		Related blade pair heaters are on.
ENGINE INTAKE OPN HTR / CLOSED OPN HTR / CLOSED		Shows door position and if heater is on.
VALVES FUEL OPEN CLOSED		Fuel valve position. Green shows open, fuel pressure present. White shows closed, no fuel pressure.
OPEN CLOSED HYD		Hydraulic valve position. Green shows open, fuel pressure present. White shows closed, no hydraulic pressure.
BTL LOW		Low extinguishing bottle pressure.
EXTG AFT BTL FWD BTL		When PULL FUEL OFF handle is pulled, lights come on to show related aft and forward extinguishing bottles have sufficient pressure for discharge into respective nacelle.
FAULT A FAULT B		Detector loop malfunction. The CHECK FIRE DET warning light also comes on.
PULL FUEL/HYD OFF		Fire condition sensed by a loop in the related nacelle. The ENGINE FIRE light and CHECK FIRE DET warning lights also comes on.

fsm96a01.dg, js, nov25/2011

Advisory Light Indications
Figure 7 (Sheet 1 of 9)

ADVISORY LIGHTS – OVERHEAD PANEL

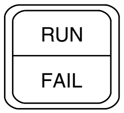
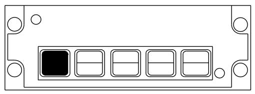
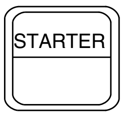
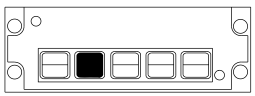
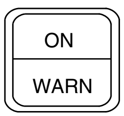
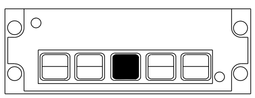
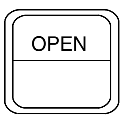
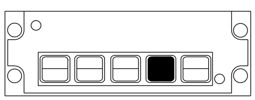
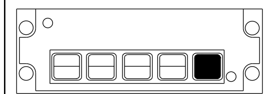
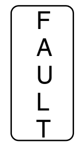
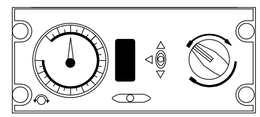
Annunciator	Panel Location	Cause for Illumination
VENT INLT / VALVE OTLT / CLOSED		Aft baggage compartment vent valve position. Green shows open, and white shows closed.
FIRE BOTTLE / AFT ARM / FWD ARM		Extinguishing bottle is armed.
FIRE BOTTLE / LOW AFT / LOW FWD		Lights come on when the extinguishing bottle press is low.
SMOKE		Smoke is sensed in forward baggage compartment.
FIRE		Fire is sensed in the APU.

Annunciator	Panel Location	Cause for Illumination
APU / FUEL OPEN / VALVE CLOSED / BTL ARM		APU fuel valve position. Green shows open, fuel pressure present. White shows closed, no fuel pressure. APU extinguishing bottle is armed.
EXTG		Fire is sensed in the aft baggage compartment. The EXTG light which illuminates when a fire is detected in the APU.
BTL LOW / FAULT		BTL LOW – Low APU fire extinguishing bottle pressure. FAULT – APU fire detection system malfunction.

fsm96a02.dg, js, nov26/2011

Advisory Light Indications
Figure 7 (Sheet 2 of 9)

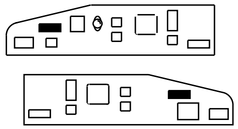
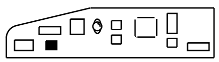
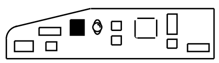
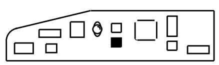
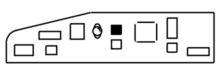
ADVISORY LIGHTS – OVERHEAD PANEL

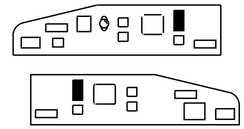
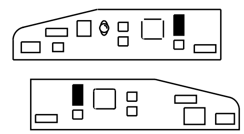
Annunciator	Panel Location	Cause for Illumination
PWR 		RUN indication shows that the APU is functioning at operating speed. FAIL indication shows an APU malfunction other than fire.
START 		STARTER indication shows APU starter engagement.
GEN 		ON indication shows that the APU generator is on line. WARN segment shows a generator malfunction.
BL AIR 		OPEN indication shows APU bleed valve is open.
		APU Generator overheat.
		Pressurization system malfunction. It also comes on for 2 seconds to show a system self test.

fsm96a03.dg, js, nov26/2011

Advisory Light Indications
Figure 7 (Sheet 3 of 9)

ADVISORY LIGHTS – GLARESHIELD PANEL

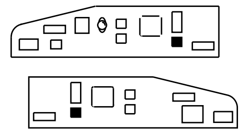
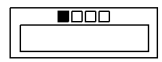
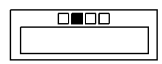
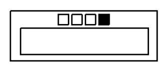
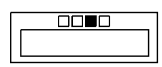
Annunciator	Panel Location	Cause for Illumination
ENGINE FIRE PRESS TO RESET		Both come on with PULL FUEL OFF, CHECK FIRE DET and master WARNING lights to warn of a fire in one nacelle or the other.
PROPELLER GROUND RANGE 1 2		Propellers are in ground operation of beta range (slightly below FLT IDLE to MAX REVERSE).
ROLL OUTBD / ROLL INBD		ROLL OUTD – outboard spoilers are at full extension. ROLL INBD – inboard spoilers are at full extension.
PUSH OFF ELEVATOR TRIM SHUTOFF		Elevator trim deactivation.
STICK PUSHER SHUTOFF PUSH OFF		Malfunctions sensed by the stick pusher actuator.

Annunciator	Panel Location	Cause for Illumination
PULL UP GPWS TEST		PULL UP Ground Proximity Warning System (GPWS) calculates any of the warnings Mode 2, Excessive closure rate to terrain. Mode 3, Descent after takeoff. Mode 4, Insufficient PULL UP lights stay on while the aircraft is in the warning area.
BELOW G/S		BELOW G/S annunciator to show a Mode 5 excessive descent below the Glideslope. The Above Ground 50 ft. (15.24 m). The Above Ground Level (AGL) is more than 1900 ft. (579.12 m). The ILS frequency is de–selected.

fsm96a04.dg, JS, nov25/2011

Advisory Light Indications
Figure 7 (Sheet 4 of 9)

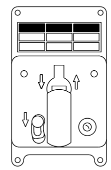
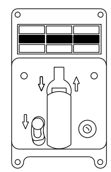
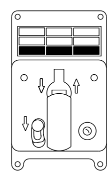
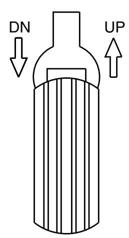
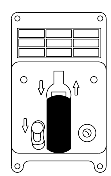
ADVISORY LIGHTS – GLARE SHIELD PANEL

Annunciator	Panel Location	Cause for Illumination
A/P DISENG		Flashes at a 1 Hz rate when AP is automatically disengaged.
RUD 1 PUSH OFF		Jam in the No. 1 actuator linkage.
SPLR 1 PUSH OFF		Roll spoiler system malfunction.
RUD 2 PUSH OFF		Jam in the No. 2 actuator linkage.
SPLR 2 PUSH OFF		Roll spoiler system malfunction.

fsm96a05.dg, js, nov26/2011

Advisory Light Indications
Figure 7 (Sheet 5 of 9)

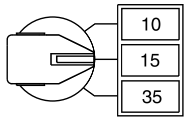
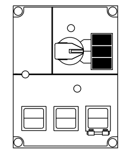
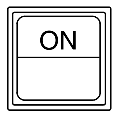
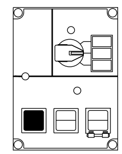
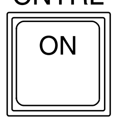
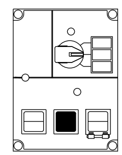
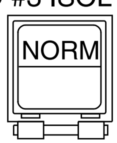
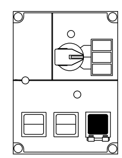
ADVISORY LIGHTS – MAIN INSTRUMENT PANEL

Annunciator	Panel Location	Cause for Illumination
L. DOOR N. DOOR R. DOOR		Doors are not in correct position.
LEFT NOSE RIGHT		Landing gears are down and locked.
LEFT NOSE RIGHT		Landing gear is not locked in position selected.
		Amber light in handle comes on with red lights to show gear in transit.

fsm96a06.dg, js/rc, feb06/2014

Advisory Light Indications
Figure 7 (Sheet 6 of 9)

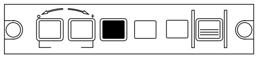
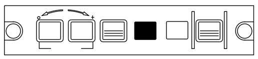
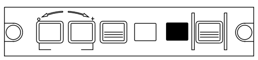
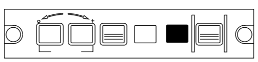
ADVISORY LIGHTS – MAIN INSTRUMENT PANEL

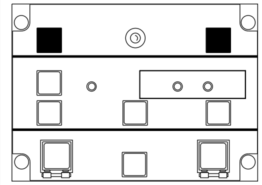
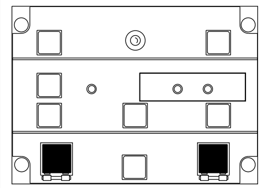
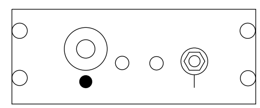
Annunciator	Panel Location	Cause for Illumination
GPWS LANDING FLAP 		GPWS landing flap position in degrees.
STBY HYD PRESS 		No. 3 hydraulic system is selected on.
PTU CNTRL  **PRE SB84–29–22** **PRE MODSUM 4–126354** PTU CNTRL  **POST SB84–29–22** **POST MODSUM 4–126354**		PTU CNTRL is set to on.
HYD #3 ISOL VLV 		Isolation valve is open.

fsm96a07.dg, js/rc, feb06/2014

Advisory Light Indications
Figure 7 (Sheet 7 of 9)

ADVISORY LIGHTS – CENTER CONSOLE

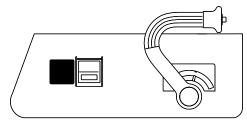
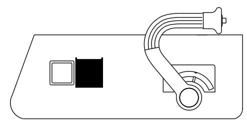
Annunciator	Panel Location	Cause for Illumination
DG		DG mode is engaged.
SLAVE		Magnetic Detector Unit malfunction
BASIC		Degraded Basic operation. Starts when the True Airspeed Reference is not available. AHRS operates as a vertical gyro (VG) system in this mode.
ATT/HDG ALIGN		Comes on during the alignment mode for 60 seconds on the ground and 90 seconds in flight.

Annunciator	Panel Location	Cause for Illumination
TANK 1 AUX PUMP / TANK 2 AUX PUMP		Auxiliary boost pump is operating.
FTHR		Auxiliary feathering pump is operating.
STATUS		SSCVR malfunction.

fsm96a08.dg, js/rc, feb06/2014

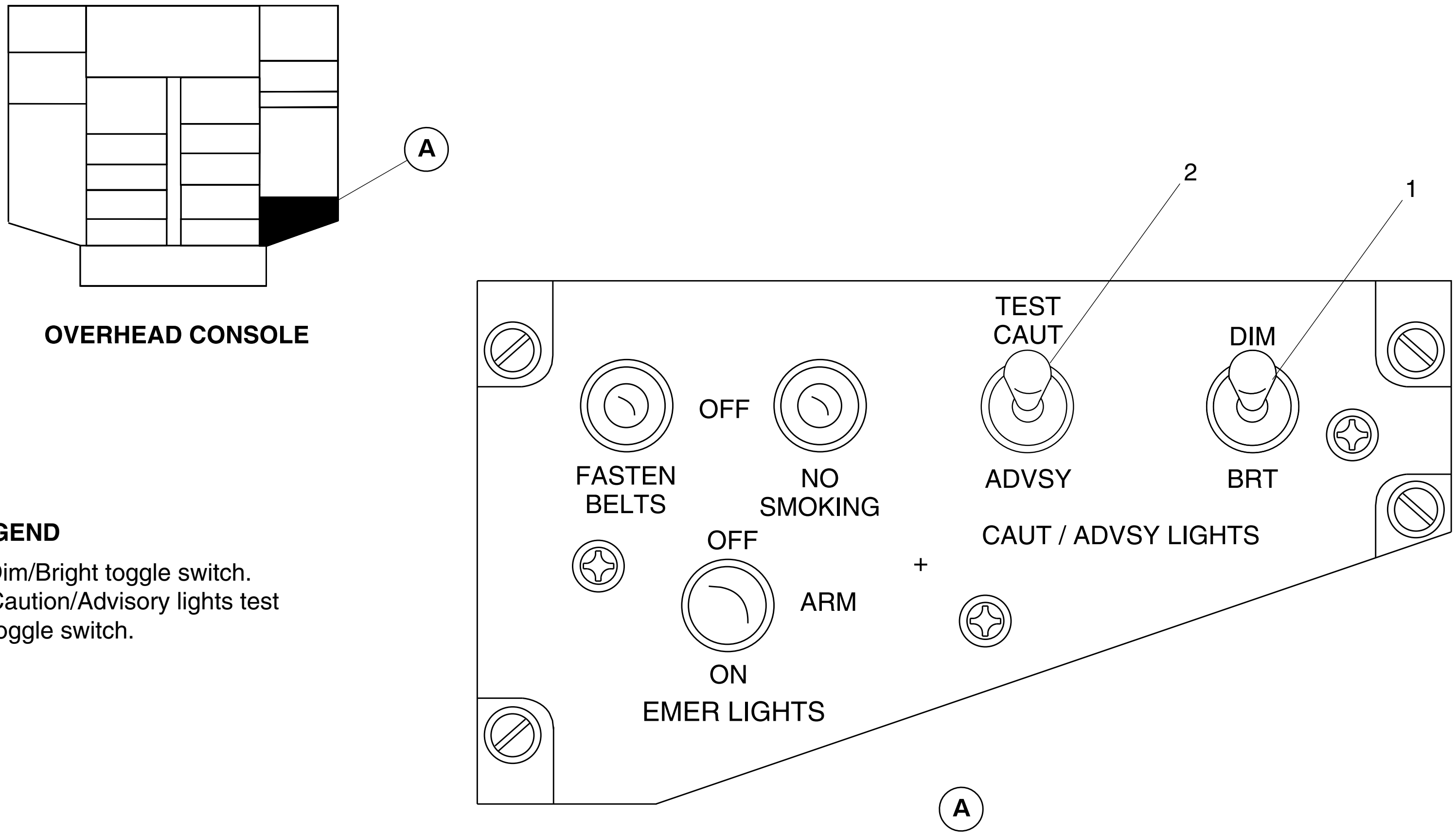
Advisory Light Indications
Figure 7 (Sheet 8 of 9)

ADVISORY LIGHTS – PILOT’S SIDE CONSOLE

Annunciator	Panel Location	Cause for Illumination
		Ground crew is connected to the interphone system. FWD/AFT shows the location.
		GPWS FLAP OVERRIDE selection.

fsm96a09.dg, rc, feb06/2014

Advisory Light Indications
Figure 7 (Sheet 9 of 9)

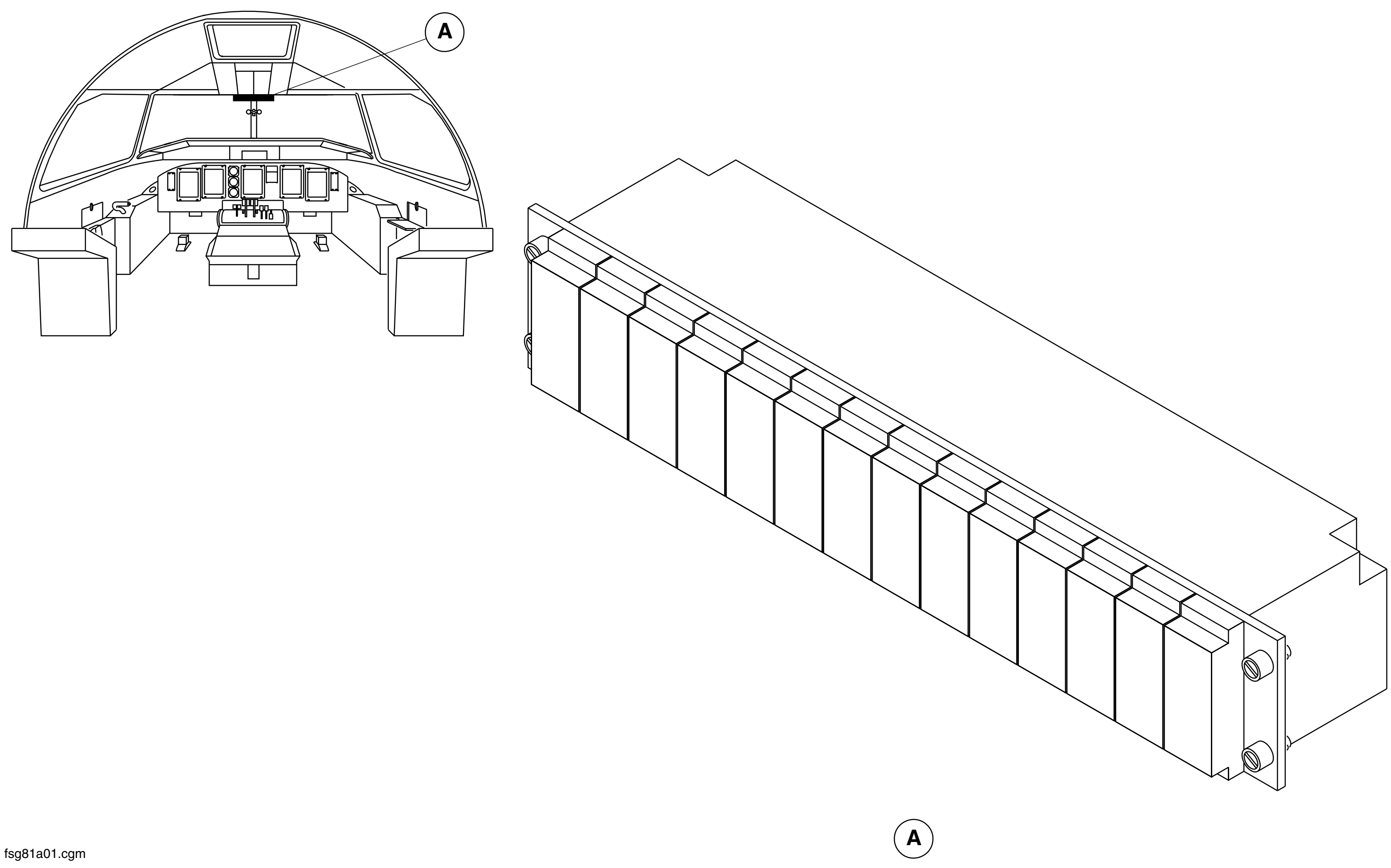


LEGEND

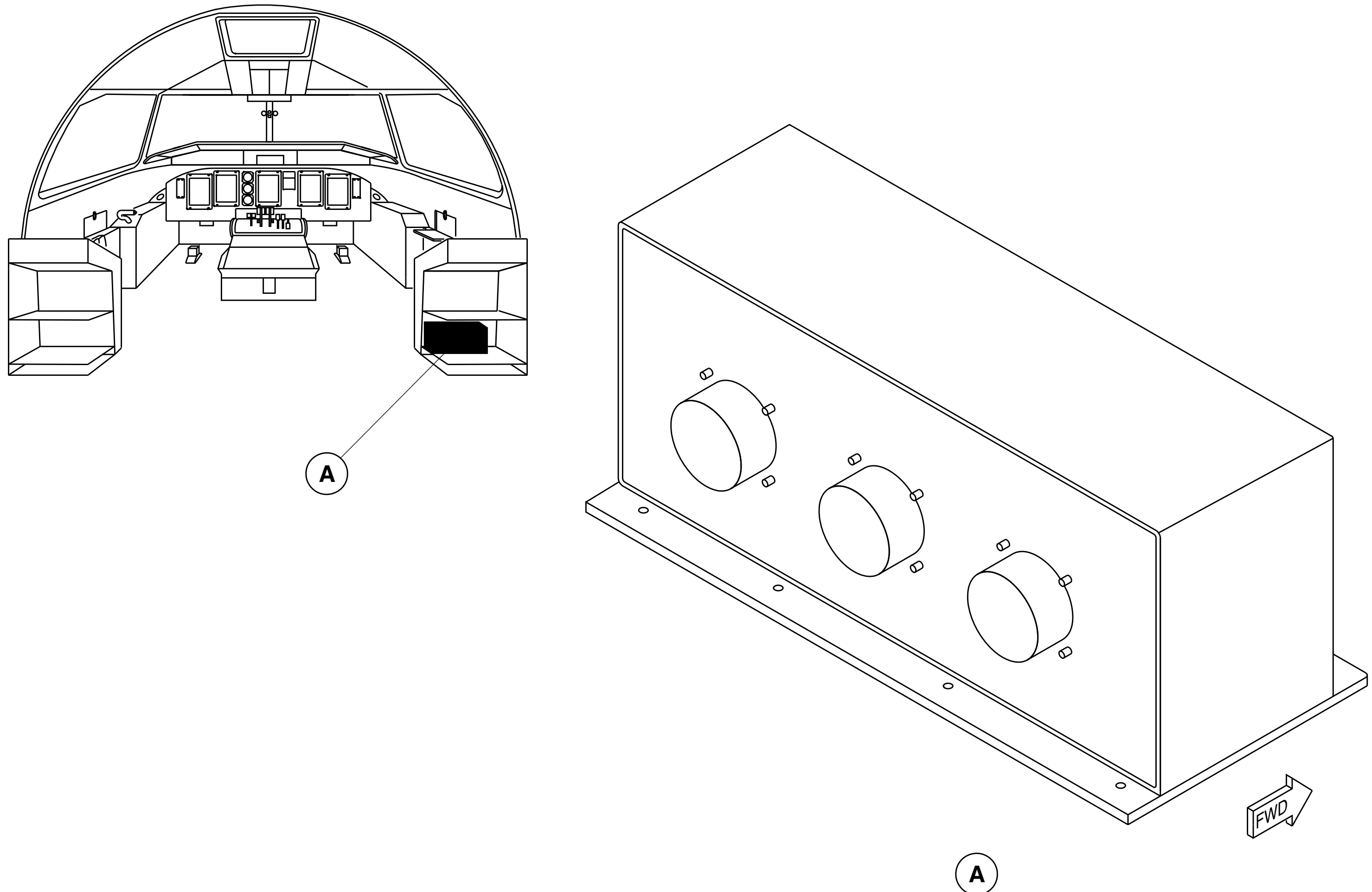
1. Dim/Bright toggle switch.
2. Caution/Advisory lights test toggle switch.

fsb41a08.dg, js nov25/2011

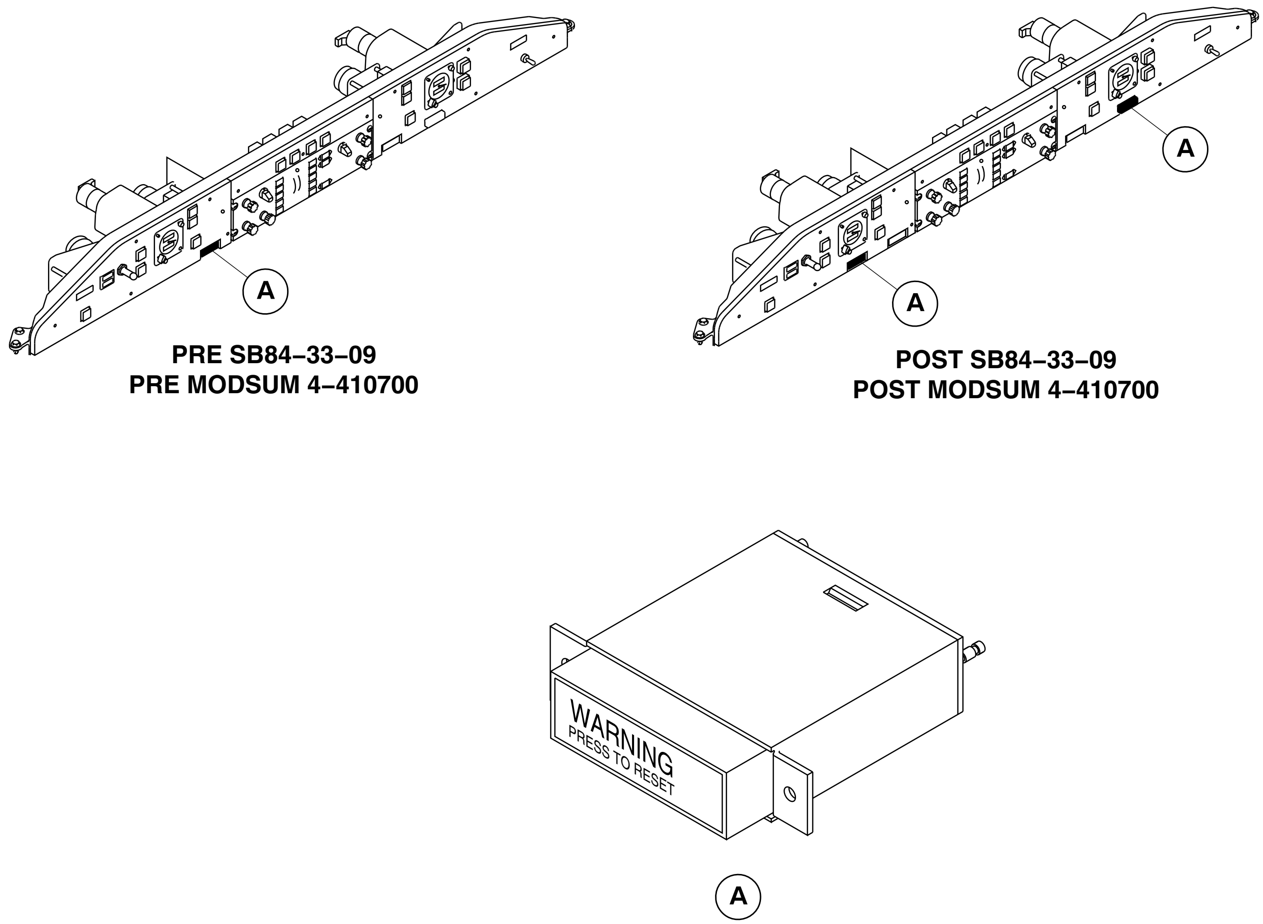
Passenger Warning Panel
Figure 8



CAUTION AND WARNING PANEL LOCATOR
Figure 9

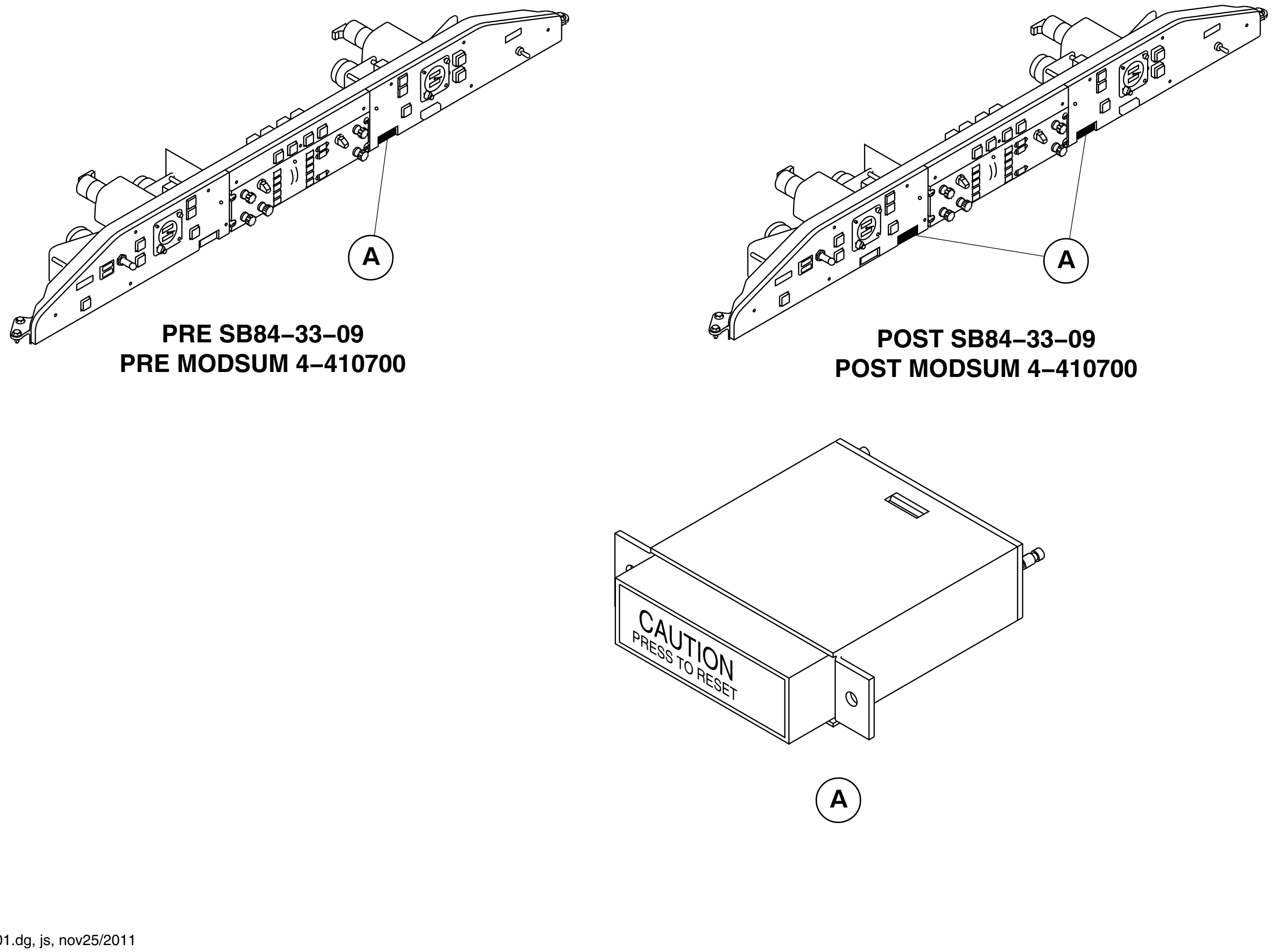


ADVISORY LIGHTS CONTROL UNIT LOCATOR
Figure 10

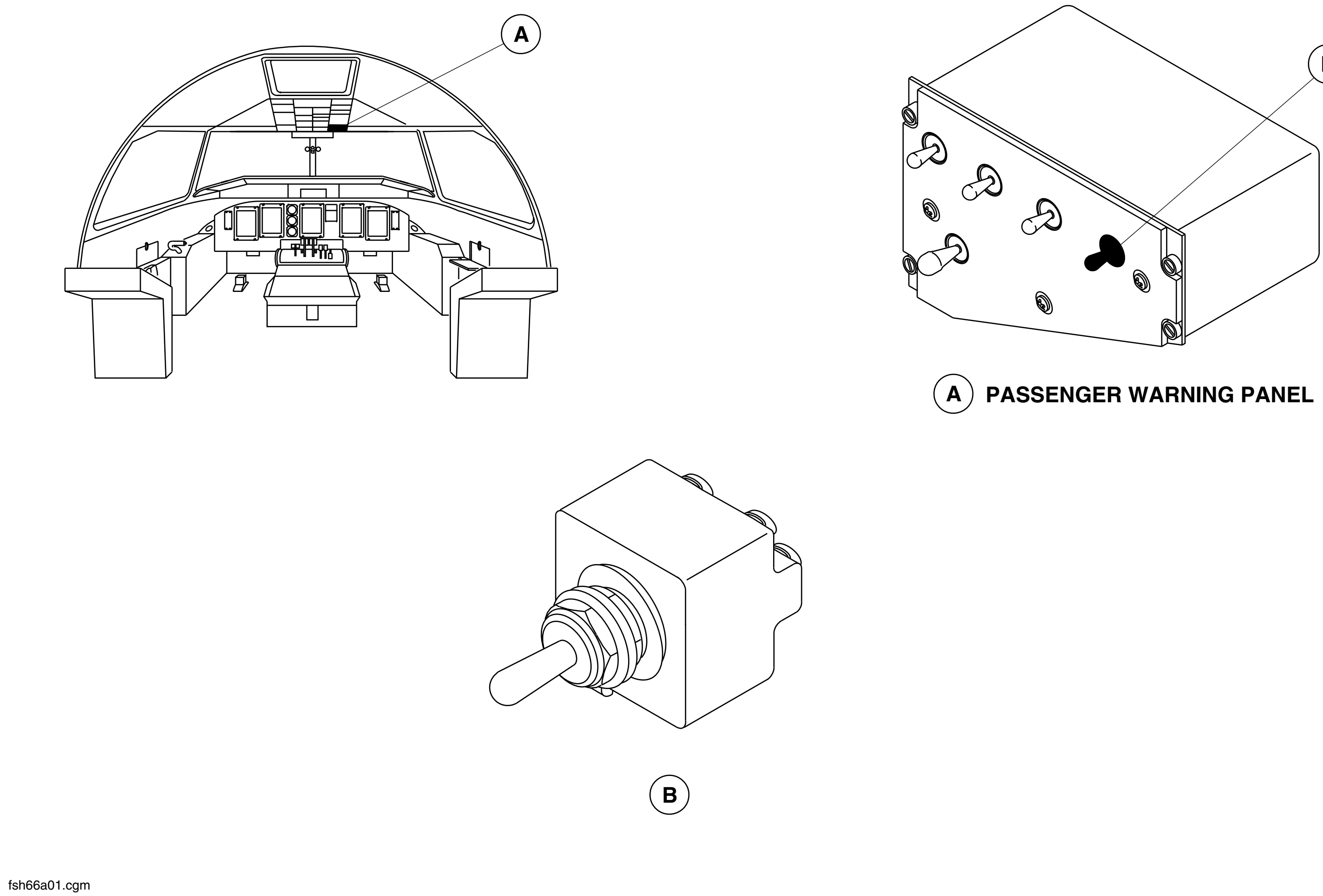


fsk41a01.dg, js, nov25/2011

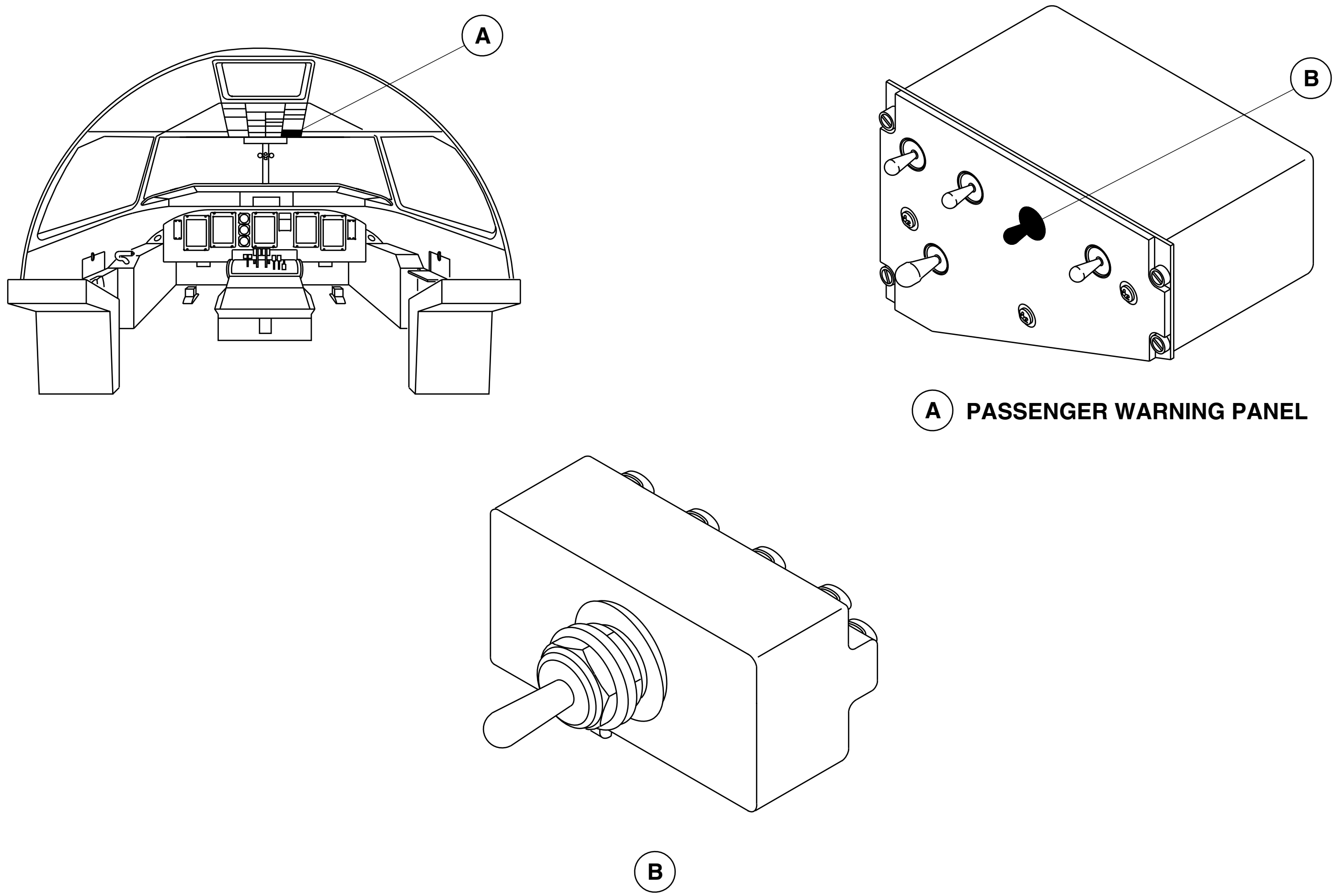
Master Warning Switch Locator
Figure 11



Master Caution Switch Locator
Figure 12



DIM/BRIGHT SWITCH LOCATOR
Figure 13



fsh65a01.cgm

CAUTION/ADVISORY TEST SWITCH LOCATOR
Figure 14

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**ON A/C ALL

31–52–00–001

TAKE–OFF WARNING SYSTEM

Introduction

Refer to 31–40–00, Central Computer.

The take–off warning system supplies an aural warning when a take–off is attempted with the aircraft not in the correct take–off configuration.

General Description

Refer to Figures 1 and 2.

The Warning Tone Generator (WTG) sounds an 1000 Hz intermittent tone when the aircraft is not in a configuration which is safe for take–off. It functions when the two engine torque indications are more than 50% or the T/O WARN TEST Toggle switch on the pilot's side console is set to TEST and one or more of the parameters is sensed as follows:

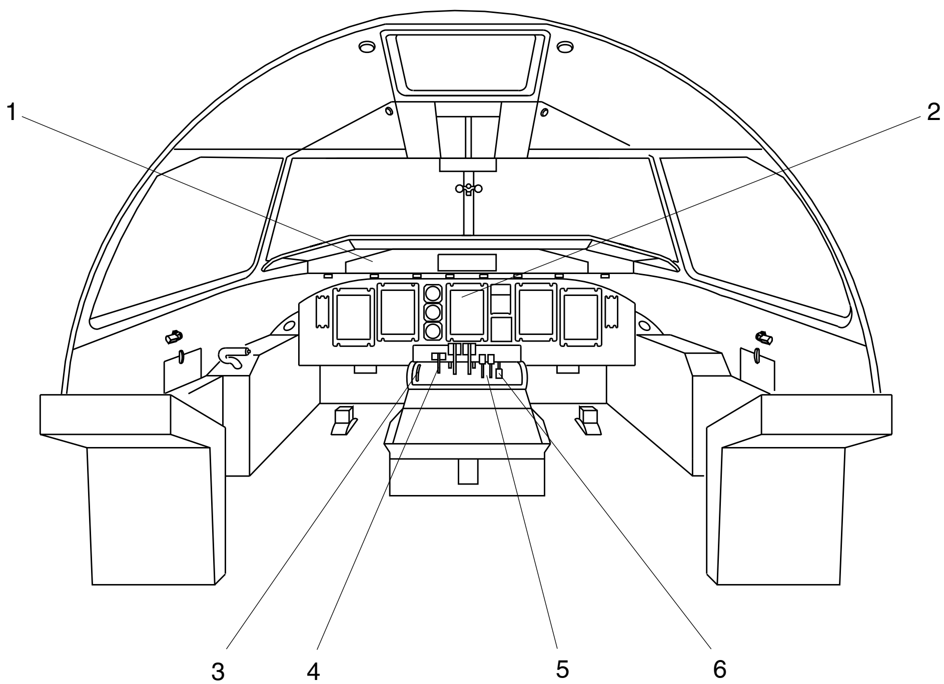
- Flaps set to more than 20
- CLA 1 not set to 1020
- CLA 2 not set to 1020
- Parking brake set
- Inboard spoilers extended
- Outboard spoiler extended
- Left elevator trim out of range
- Right elevator trim out of range.

DETAILED DESCRIPTION

The take off warning system has the interfaces that follow:

Parameter	Source
T/O WARN MAINT	T/O WARN TEST Toggle switch
ENG1 TORQUE > 50%	FADEC
ENG2 TORQUE > 50%	FADEC
NGWOFW	PSEU
FLAPS ≤ 13.5°	FPIU
CLA1 ≠ MAX/1020	FADEC
CLA2 ≠ MAX/1020	FADEC
PARKING BRAKE ON	PARKING BRAKE LEVER
INBOARD SPOILERS EXTENDED	FCS ECU LEFT CHANNEL
LEFT ELEVATOR TRIM OUT OF RANGE	FCS ECU LEFT CHANNEL

OUTBOARD SPOILERS EXTENDED	FCS ECU RIGHT CHANNEL
RIGHT ELEVATOR TRIM OUT OF RANGE	FCS ECU RIGHT CHANNEL

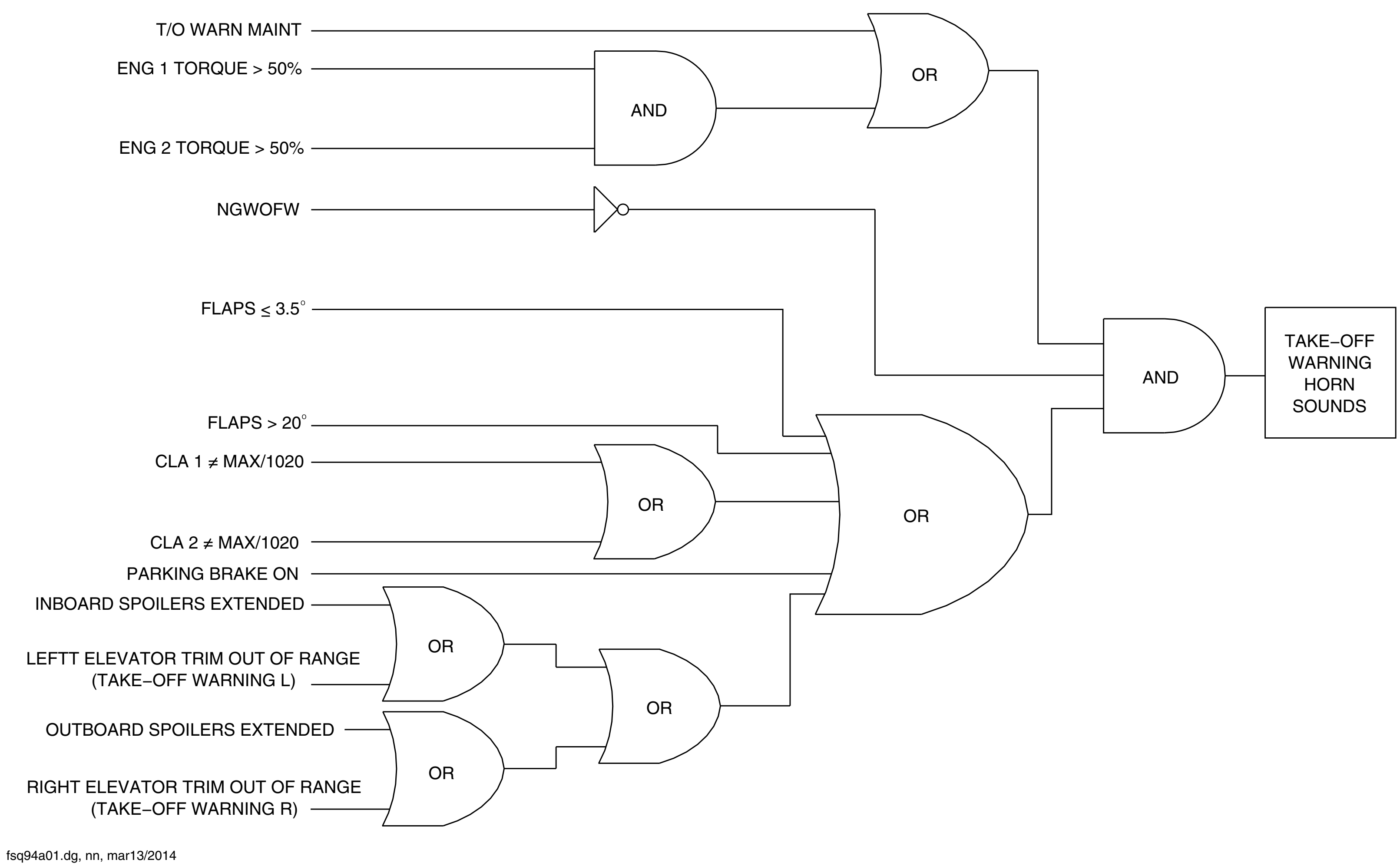


LEGEND

1. Spoiler advisory lights.
2. Engine torque indication.
3. Elevator trim indicator.
4. Parking brake lever.
5. Condition levers.
6. Flap selector lever.

fsr01a01.dg, nn, mar13/2014

Flight Compartment Take–Off Configuration
Figure 1



FDPS Incorrect Take–Off Configuration Warning
Figure 2

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**ON A/C ALL

31–61–00–001

ELECTRONIC INSTRUMENTS SYSTEM

Introduction

The Electronic Instrument System (EIS) shows navigation, engine, and system parameters. It interfaces with other systems to calculate, make, and show their images. The Electronic Instrument System (EIS) also monitors is calculations to prevent misleading information from being shown.

General Description

Refer to Figures 1 and 2.

The electronic instruments system has five identical and interchangeable active matrix liquid crystal Display Units (DUs) with the same hardware and software part numbers. Different pin programming at each DU location gives the functions that follow:

- Pilot's Primary Flight Display, PFD1
- Pilot's Multi–Function Display, MFD1
- Engine Display, ED
- Copilot's Primary Flight Display, PFD2
- Copilot's Multi–Function Display, MFD2.

The Electronic Instrument System (EIS) is divided into the two subsystems that follow:

- Electronic Flight Instrument System (EFIS) (31–61–01)
- Engine and System Integrated Display (ESID) (31–61–02).

The Electronic Flight Instrument System (EFIS) is further subdivided into two systems, one for the pilot and one for the copilot. Each EFIS has the components that follow:

- Primary Flight Display (PFD) (31–61–01)
- Multi–Function Display (MFD) (31–61–01)
- Panels, EFIS Control (EFCP) (31–61–06)
- Panel, ESID Control (ESCP) (31–61–11)
- Index Control Panels (ICP1) (31–61–16).

The Engine System Integrated Display (ESID) shares the MFDs with EFIS. It has the components that follow:

- Engine Display (ED) (31–61–02)
- Panel, ESID Control (ESCP) (31–61–11).

Detailed Description

The DUs function in the modes that follow:

- Power–On Self Test (POST)
- Line operational
- Maintenance
- Teleloading.

Power−On Self Test (POST): The Power−On Self Test (POST) mode automatically continues for 2 seconds after power up to ensure correct operation before operational mode is started. The Power−On Self−Test (POST) does the checks that follow:

- Hardware program pin parity
- CPU/ASIC self test
- RAM/EEPROM memory
- Input/Output (I/O) circuits and ports
- Hardware/Consistency.

During the self−test, the output data from DUs cannot be used. They show a green T during the Power−On Self−Test (POST) mode. The POST mode continues for 30 seconds.

During the POST mode, the DU checks the compatibility between the DU part number, hardware and software version. It will not start the operational mode if the checks are not correct.

The Display Units (DU) sense their internal temperature. They automatically turn an internal heater on while the temperature is too low to make sure the correct display brightness. It shows a WARMING UP message in white in the center of the otherwise blank screen.

The normal display replaces the message as soon as the internal temperature reaches the minimum operational requirement.

The WARMING UP message is shown after the Power−On Self−Test (POST) mode.

From −55 to −25 °C (−67 to −13 °F), the display can be of less than the nominal quality but shows the correct data. These images are available in less than 15 minutes after the POWER ON.

Nominal performance is available in less than 20 minutes of POWER ON.

From −25 °C (−13 °F) and above, the display can be of less than the nominal quality but shows the correct data. These images are available in less than 5 minutes after the POWER ON.

Nominal performance is available in less than 15 minutes of POWER ON.

Line Operational: Each DU calculates the position, size and the value of all parameters that is shown.

The DUs makes images similar to existing conventional symbology. These images have line strokes and raster to display critical flight data and guidance commands. Raster is a scan pattern in which an area is scanned from side to side in lines from top to bottom.

A circuit independent of channel that makes the images does a monitoring calculation to make sure that incorrect parameters are not shown. The DUs shows warnings when it senses a malfunction by continuous monitoring.

The indication brightness is automatically and manually controlled.

The DU brightness changes automatically with the ambient lighting conditions in the flight compartment. A light detector located on the front face of each Display Unit (DU) provides an ambient light input for automatic brightness control.

The manual brightness controls enable the pilot and copilot to change the brightness of each DU from a minimum to a maximum level. The brightness level is stays the same after a power interruption.

The manual brightness control can change the automatic control level in normal and failure conditions.

Maintenance Mode: The Built In Test Equipment (BITE) uses the Central Diagnostic System (CDS) to give the condition of the component. It saves faults in a Non Volatile Memory (NVM) for reporting to line and shop maintenance.

The maintenance mode is selected from one or the other Audio and Radio Control Display Units (ARCDUs) during maintenance operations. The DUs do the tests when they receive a maintenance discrete command from IOM1 while the air speed is less than 50 kt and the aircraft is on the ground.

A short or medium power interruption causes the DUs to go off and then start an initialisation of the hardware. It does not start the Power–On Self Test (POST) mode or show a malfunction indication. The DUs operates in the line operational mode after the power interruption. (The display flickers).

A long power interruption while the aircraft is on the ground causes a POST that continues for 30 seconds.

A long power interruption while the aircraft is airborne starts a short hardware and RAM initialisation that continues for 2 seconds.

When a DU senses an internal malfunction, it will stay in the POST mode for the duration of the malfunction.

The BITE allows aircraft maintenance personnel to:

- Do fault isolation and return to service testing after completing maintenance actions
- Access failure reports from last or previous flight legs
- Get the avionics status report
- Get the part number of an LRU.

The BITE mode monitors the condition of the component as follows:

- Power–On Self Test (POST)
- Continuous monitoring.

Power–On Self Test (POST): The POST checks the condition of the component at power–up or after a long power interruption more than 200 milliseconds.

Refer to Figure 3.

Continuous Monitoring: The continuous monitoring checks the status of the components in flight. It records faults in a Non Volatile Memory (NVM) for later troubleshooting using the Central Diagnostic System (CDS).

The DUs monitor the critical indication parameters that follow:

- Pitch
- Roll
- Altitude
- Airspeed
- Heading
- Torque
- ITT
- NH
- NP.

A mismatch condition between inputs from critical parameters is monitored by the Flight Data Processing System (FDPS). When a difference is sensed by the FDPS, it causes the PFDs to show a mismatch message.

Display monitoring makes sure that the symbols are shown correctly. A feedback bus is used to compare the shown indication with the direct input.

The monitoring is done by one or two DUs as follows:

- The PFD is monitored by its adjacent MFD (if not set to PFD) and opposite PFD. If the opposite PFD is de–energized, it uses the opposite MFD.
- The ED is monitored by its adjacent MFDs. If one MFD is set to PFD, it uses the other MFD.

The threshold for PFD monitoring is shown in the table that follows:

Table 1: Table 1.

PARAMETER	THRESHOLD
Roll	4 degrees
Pitch	4 degrees
Heading	6 degrees
Indicated Air Speed	10 knots
Corrected Barometric Altitude	100 feet

The threshold for ED monitoring is shown in the table that follows:

Table 2: Table 2.

PARAMETER	THRESHOLD
Torque	5 percent
NH	5 percent
NP	50 rpm
ITT	50° C

When a difference between the shown indication and the direct input is more than the threshold, the EIS shows the messages that follow:

- CHECK PFD on PFDs and MFDs
- CHECK ED on ED and MFDs.

An incorrect response from a switch selection is immediately sensed by the the pilot but a malfunction causes set parameters to change to dashes.

Teleloading: The operational software serial data is supplied to the DUs through a Portable Maintenance Access Terminal (PMAT).

The aircraft must be on the ground and it must receive a data load request signal.

The DUs do a software configuration check before the teleloading mode starts. It makes sure that the software version to be teleloaded is the same as the equipment part number.

The teleload time is less than 5 minutes for each DU. The DUs show a green L during the Teleloading mode.

The Display Unit (DU) restarts after a complete and successful Teleload. It initiates a complete Power–On Self Test (POST) within 10 seconds.

The DU starts a POST less than 10 seconds after the teleload mode is complete.

There are programming pins located on the DU connectors to supply data to the DUs that follows:

- Aircraft type
- Location of the DU on the aircraft
- Selection of the DU
- Shop maintenance
- Teleloading.

Refer to Figure 4.

When the pilot is flying the aircraft, the MFD1 reversion selector is set to the NAV position and the MFD2 reversion selector is set to the SYS position on the ESID Control Panel (ESCP).

Refer to Figure 5.

The DUs show the Electronic Instruments System (EIS) data that follows:

- Pilot's Primary Flight Display (PFD1)
- Copilot's Primary Flight Display (PFD2)
- Engine Display (ED).
- Pilot's Multi–Function Displays (MFD1)
- Copilot's Multi–Function Displays (MFD2).

Refer to Figure 6.

Pilot's Primary Flight Displays (PFD1, PFD2): The Primary Flight Displays (PFD) are the outer displays of the Electronic Instrument System (EIS). Each display shows the primary navigation information that follows:

- Flight Mode Annunciator (FMA)
- Indicated Air Speed (IAS)
- Altimeter (ALT)
- Electronic Attitude Direction Indicator (EADI)
- Electronic Horizontal Situation Indicator (EHSI)
- Inertial Vertical Speed Indicator (IVSI)
- Traffic Collision Avoidance System II (TCAS II).

The MFDs are located inboard of the PFDs. Their indication is divided into parts when they are not showing a Primary Flight Display (PFD) or Engine Display (ED) reversionary indication.

Refer to Figure 7.

The pilot's MFD1 is usually set to show the Electronic Flight Instrument System (EFIS) navigation page at the top part of the indication and the Engine and System Integrated Display (ESID) permanent system data area at the bottom. Its permanent system data area usually shows the Powered Flight Control Surface Position (PFCS) parameters.

Refer to Figure 8.

The copilot's MFD2 is usually set to show the Engine and System Integrated Display (ESID) electrical system page at the top part of the indication and permanent system data area at the bottom. Its permanent system data area usually shows the parameters that follow:

- Flap position
- Hydraulic pressure
- Hydraulic quantity.

Refer to Figures 9 , 10 and 11.

The Engine and System Integrated Display (ESID) system shows the additional system pages that follows:

- Engine, or
- Fuel, or
- Doors.

Refer to Figure 12.

The Engine Display (ED) is located at the center of the instrument panel. It shows primary engine and aircraft systems data that follows:

- Torque
- NH High Pressure Compressor Rotational Speed
- NP Propeller Rotational Speed
- ITT Inter–Turbine Temperature
- Fuel Flow
- NL Low Pressure Compressor Rotational Speed
- Engine rating mode
- Oil Pressure
- Oil Temperature
- Tank Fuel Quantity
- Fuel inlet temperature
- Powerplant messages
- Advisory messages
- Static Air Temperature.
- System Page (Mono Mode).

The Electronic Instruments System (EIS) has the control panels that follow to control the DU indications:

- Pilot’s EFIS Control Panels (EFCP1)
- Copilot’s EFIS Control Panels (EFCP2)
- ESID Control Panel (ESCP).
- Pilot’s Index Control Panel (ICP1)
- Copilot’s Index Control Panel (ICP2).

Refer to Figure 13.

EFIS Control Panels (EFCP1, EFCP2): To control some related EFIS PFD and navigation page indications, the EFCPs have the controls that follow:

- Bearing 1 selector knob
- Bearing 2 selector knob
- FULL/ARC mode pushbutton switch
- WX mode pushbutton switch
- FMS MAP mode pushbutton switch
- TCAS pushbutton switch

- RANGE selector knob
- PFD OFF, BRT knob
- MFD OFF, BRT knob
- WXBRT knob.

Bearing 1, 2 Selector Knob: The bearing 1 or 2 selector knob is turned to make an AUX, VOR, ADF, or FMS bearing data selection for indication on the EFIS PFD.

FULL/ARC Mode Pushbutton Switch: The FULL/ARC mode pushbutton switch is pushed to make a full mode indication selection on the navigation page.

WX mode pushbutton switch: The WX mode pushbutton switch is pushed to show the weather radar indication on the navigation page.

FMS MAP mode pushbutton switch: The MAP mode pushbutton switch is pushed to show the Flight Management System (FMS) map indications on the navigation page.

TCAS pushbutton switch: The TCAS pushbutton switch is pushed to change the TCAS range indications on the navigation page.

RANGE selector knob: The RANGE pushbutton switch is pushed to change the weather radar range indications on the navigation page.

PFD OFF, BRT Knob: The PFD OFF, BRT Knob is turned to energize or de–energize the PFD and to control its brightness.

MFD OFF, BRT Knob: The MFD OFF, BRT Knob is turned to energize or de–energize the MFD and to control its brightness.

WXBRT knob: The WXBRT knob is turned to adjust the weather radar image brightness.

Refer to Figure 14.

ESID Control Panels (ESCP1, ESCP2): To control the Engine and System Integrated Display ESID) indications and some EFIS indications, the ESCP has the controls that follow:

- MFD1 and MFD2 Reversion Switches
- ELEC SYS Pushbutton Switch
- ENG SYS Pushbutton Switch
- FUEL SYS Pushbutton Switch
- DOORS SYS Pushbutton Switch
- ALL Pushbutton Switch
- EFIS ATT/HDG SOURCE reversion selector
- EFIS ADC SOURCE Reversion Selector
- ED OFF, BRT Knob.

MFD1 and MFD2 Reversion Switches: The MFD1 and MFD2 reversion switches are used to make indication selections on the ESID DUs.

Refer to Figures 15 and 16.

ELEC SYS Pushbutton Switch: The ELEC SYS pushbutton switch is pushed to show the electrical system page on the ESID DUs.

Refer to Figures 17 and 18.

ENG SYS Pushbutton Switch: The ENG SYS pushbutton switch is pushed to show the engine system page on the ESID DUs.

Refer to Figures 19 and 20.

FUEL SYS Pushbutton Switch: The FUEL SYS pushbutton switch is pushed to show the fuel system page on the ESID DUs.

Refer to Figures 21 and 22.

DOORS SYS Pushbutton Switch: The DOORS SYS pushbutton switch is pushed to show the doors system page on the ESID DUs.

ALL Pushbutton Switch: The ALL SYS pushbutton switch is pushed to sequence through all the system pages on the ESID DUs.

EFIS ATT/HDG SOURCE Reversion Selector: The EFIS ATT/HDG SOURCE reversion selector is used to make an Attitude and Heading Reference Unit (AHRU 1, AHRU 2) indication selection on the EFIS DUs.

EFIS ADC SOURCE Reversion Selector: The EFIS ADC SOURCE reversion selector is used to make Attitude and Heading Reference Units, ADU 1 or ADU 2 indication selections on the EFIS DUs.

ED OFF, BRT Knob: The ED OFF, BRT Knob is turned to energize or de–energize the ED and to control its brightness.

Refer to Figure 23.

The MFD1 permanent system data area shows the Powered Flight Control Surface Position (PFCS) parameters and the MFD 2 permanent system data area shows flap position, hydraulic pressure, and hydraulic quantity indications.

Refer to Figures 24 and 25.

When a Multi–Function Display (MFD) malfunctions, its permanent system data moves to the other MFD and the permanent system data indication changes to a composite indication to show the two permanent system data together.

When a Primary Flight Display (PFD) or Engine Display (ED) malfunctions, the MFD1 or MFD2 reversion switch located on the ESID Control Panel (ESCP) is used to manually move the indications from the defective DU to a Multi–Function Display (MFD). As part of this reversionary mode, the permanent system data moves to the other MFD and changes to a composite indication to show the two MFD1 and MFD2 permanent system data together.

Refer to Figure 26.

There is one automatic DU reversionary mode. When the Engine Display (ED) malfunctions while airborne, the ED images automatically move to the pilot's Multi–Function Display (MFD1) and the the copilot's Multi–Function Display (MFD2) shows a composite permanent system data indication. This automatic reversionary mode occurs only if the pilot's Multi–Function Display (MFD1) is not set to show the pilot's Primary Flight Display (PFD1) images.

Refer to Figure 27.

After the automatic reversion, the MFD1 reversion switch is manually set to the ENG position to make the DU indications agree with the switch selection.

Refer to Figures 28 , 29 and 30.

If the Engine Display (ED) and pilot's Primary Flight Display (PFD1) malfunction at the same time while airborne, the ED images automatically move to the pilot's Multi–Function Display (MFD1) and the the copilot's Multi–Function Display (MFD2) shows a composite permanent system data indication. The pilot's the MFD1 reversion switch is set to PFD to manually move the PFD indications from the defective PFD1 to its related Multi–Function Display (MFD1). The ENG SYS Pushbutton switch is then pushed to show the engine system page and a composite permanent system data indication.

Refer to Figure 31.

If the pilot and copilot's Multi–Function Displays (MFD1 and MFD2) malfunction at the same time, the navigation page, system page, and the permanent system data indication is not in view.

Refer to Figures 32 and 33.

The Engine and System Integrated Display (ESID) system has a mono mode to let the pilots view the system page with a composite permanent system data indication when the two MFDs malfunction or are set to show PFD or ED indications. A system pushbutton switch is pushed to show its applicable system page with a composite permanent system data indication on the Engine Display (ED). The ED images are shown again when when the push button switch is released.

Refer to Figure 34.

Index Control Panels (ICP1, ICP2): To control some related EFIS PFD indications, the ICPs have the controls that follow:

- SPEED BUG index switch
- SPEED BUG rotary selector
- BARO SET knob
- DH knob.

SPEED BUG index switch: The SPEED BUG index switch located on the Index Control Panel (ICP1, ICP2) is turned to select the solid speed bug or outline speed bug on the airspeed indicator.

SPEED BUG rotary selector: The SPEED BUG rotary selector is turned to adjust the speed bug value.

BARO SET knob: The BARO SET knob is turned to supply the related Air Data Unit (ADU1, ADU2) with a barometric correction.

DH knob: The DH knob is turned to set a DH altitude for Decision Height (DH) calculations.

The Electronic Instrument System (EIS) components are energized by the main 28 Vdc generation system. The electrical power source for each component is shown in the table that follows:

Table 3: Table 3.

PART	LEFT MAIN	LEFT ESS	RIGHT MAIN	RIGHT ESS
PFD1	X			
MFD1		X		
ED			X	X
MFD2			X	
PFD2			X	
EFCP1	X			
ICP1		X		
ESCP	X			
EFCP 2			X	
ICP2			X	

Each Display Unit (DU) dissipates 170 W of power when the display heater is on and the brightness is set to a maximum level. It is 110 W when the heater turns off. The DUs dissipates 90 W of power when its heater is off and the brightness is set to a maximum level with low ambient flight compartment illumination.

The left main bus supplies electrical power through a 10 A circuit breaker and two 5 A circuit breakers to the PFD1, EFCP1, and ESCP. The circuit breakers are located in position A2, C2, and B2 on the avionics circuit breaker panel.

The left essential bus supplies electrical power through a 10 A circuit breaker and a 5 A circuit breaker to the MFD1 and ICP1. The circuit breakers are located in position B7 and C7 on the avionics circuit breaker panel.

The right main bus supplies electrical power through three 10 A circuit breakers and two 5 A circuit breakers to the ED, MFD2, PFD2, EFCP2, and ICP2. The circuit breakers are located in position A5, B5, B6, C5, and C6 on the avionics circuit breaker panel.

The right essential bus also supplies electrical power through a 10 A circuit breaker to the ED. The circuit breakers is located in position B11 on the avionics circuit breaker panel.

Display Units (DU)

Refer to Figure 35.

Each Display Unit has a 6 by 8 in. (152 by 203 mm) active matrix liquid crystal display with a display resolution of 167 Dots Per Inch (DPI). It uses the colour rules shown in the table that follows:

Table 4: Table 4.

COLOUR	INDICATION
Red	Warning
Yellow	Caution
White	Advisory, labels, scales
Green	Active modes, passed test
Cyan	Selectable parameters
Magenta	FMS
Brown	Attitude sphere
Blue	Attitude shere, ESID units
Grey	Indicator background
Black	Aircraft symbol

The DU shows messages with the attributes that follows:

– Flashing
– Reverse video
– Brackets.

Flashing: Flashing is used when a new message that requires pilot attention comes into view. It is time limited to 5 seconds or stays in view until the pilot action is completed. The flashing frequencny is 1 Hz.

Reverse Video: Reverse video is used to show a change in the operating mode that was not pilot set. It is time limited to 5 seconds.

Brackets: The message in the brackets are pilot instructions.

The dynamic data of the images are received and updated on the screen at a refresh frequency of 20 Hz. The stroke and raster images are updated at a nominal 60 Hz rate.

The DU has the sub–assemblies that follow:

- Interface Connection Module (ICM)
- Digital Processing Module (DPM)
- Backlighting Module (BLM)
- LCD Assembley Module (LAM)
- Power Supply Module (PSM)
- Housing Assembly Module (HAM).

The display units weigh 12.32 lb (5.59 kg). They are 7.56 in. (195 mm) wide, 10.486 in. (266.3 mm) high and 7.46 in. (190 mm) long.

The DUs are attached to the instrument panel with four mounting screws. They are electrical bonded to the instrument panel through wires in the rear connector and by surface contact between the DU chassis and the instrument panel mounting frame.

EFIS Control Panels (EFCP)

Refer to Figure 36.

To control some related EFIS PFD and navigation page indications, the EFCPs have the controls that follow:

- Bearing 1 selector knob
- Bearing 2 selector knob
- FULL/ARC mode pushbutton switch
- WX mode pushbutton switch
- FMS MAP mode pushbutton switch
- TCAS pushbutton switch
- RANGE selector knob
- PFD OFF, BRT knob
- MFD OFF, BRT knob
- WXBRT knob.

Refer to Figure 37.

The EFCP has the sub–assemblies that follow:

- Front face module
- Interface board
- Outer box.

The Interface board has the functions that follow:

- Power supply
- Parallel/serial converter

- ARINC emmiter/receiver
- EMI, lightning protection.

The EFIS selection and the type of interface signal is shown in the table that follows:

Table 5: Table 5.

SELECTION	SIGNAL TYPE
Bearing 1 selector knob	ARINC 429
Bearing 2 selector knob	ARINC 429
FULL/ARC mode pushbutton switch	ARINC 429
WX mode pushbutton switch	ARINC 429
FMS MAP mode pushbutton switch	ARINC 429
TCAS pushbutton switch	ARINC 429
RANGE selector knob	ARINC 429
PFD OFF, BRT knob	Discrete, ARINC 429
MFD OFF, BRT knob	Discrete, ARINC 429
WXBRT knob	ARINC 429

NOTE

Note: The EFCP PFD and MFD OFF, BRT knobs supply a discrete signal to the DUs for ON/OFF control and ARINC 429 data for brightness control.

The EFCPs weigh 2.2 lb (1.0 kg). The EFCP is 2.625 in. (66.68 mm) wide, 5.75 in. (146 mm) high and 5.9 in. (150 mm) deep. The EFIS Control Panels (EFCPs) are attached to the centre console with four DZUS fasteners. The fasteners and a bonding wire directly connect to the chassis to make a ground continuity conection between control panel chassis and the aircraft structure.

ESID Control Panel (ESCP)

Refer to Figure 38.

To control some related EFIS PFD and navigation page indications, the ESCP has the controls that follow:

- MFD1 and MFD2 Reversion Switches
- ELEC SYS Pushbutton Switch
- ENG SYS Pushbutton Switch
- FUEL SYS Pushbutton Switch
- DOORS SYS Pushbutton Switch
- ALL Pushbutton Switch

- EFIS ATT/HDG SOURCE reversion selector
- EFIS ADC SOURCE Reversion Selector
- ED OFF, BRT Knob.

The ESCP has the sub–assemblies that follow:

- Front face module
- Interface board
- Outer box.

The Interface board has the functions that follow:

- Power supply
- Parallel/serial converter
- ARINC emmiter/receiver
- EMI, lightning protection.

The ESID selection and the type of interface signal is shown in the table that follows:

Table 6: Table 6.

SELECTION	SIGNAL TYPE
MFD1 and MFD2 Reversion Switches	ARINC 429
ELEC SYS Pushbutton Switch	ARINC 429
ENG SYS Pushbutton Switch	ARINC 429
FUEL SYS Pushbutton Switch	ARINC 429
DOORS SYS Pushbutton Switch	ARINC 429
ALL Pushbutton Switch	Discrete
EFIS ATT/HDG SOURCE reversion selector	Discrete
EFIS ADC SOURCE Reversion Selector	Discrete
ED OFF, BRT Knob	Discrete, ARINC 429

NOTE

Note: ED OFF, BRT knob supplies a discrete signal to the DU for ON/OFF control and ARINC 429 data for brightness control.

The ESCP weighs 2.2 lb (1.0 kg). The ESCP is 2.625 in. (66.68 mm) wide, 5.75 in. (146 mm) high and 5.9 in. (150 mm) deep. The ESIS Control Panel (ESCP) is attached to the centre console with four DZUS fasteners. The fasteners and a bonding wire directly connect to the chassis to make a ground continuity conection between control panel chassis and the aircraft structure.

Index Control Panels (ICP)

Refer to Figure 39.

To control some related EFIS PFD indications, the ICPs have the controls that follow:

- SPEED BUG index switch

- SPEED BUG rotary selector
- BARO SET knob
- DH knob.

The ICP has the sub−assemblies that follow:

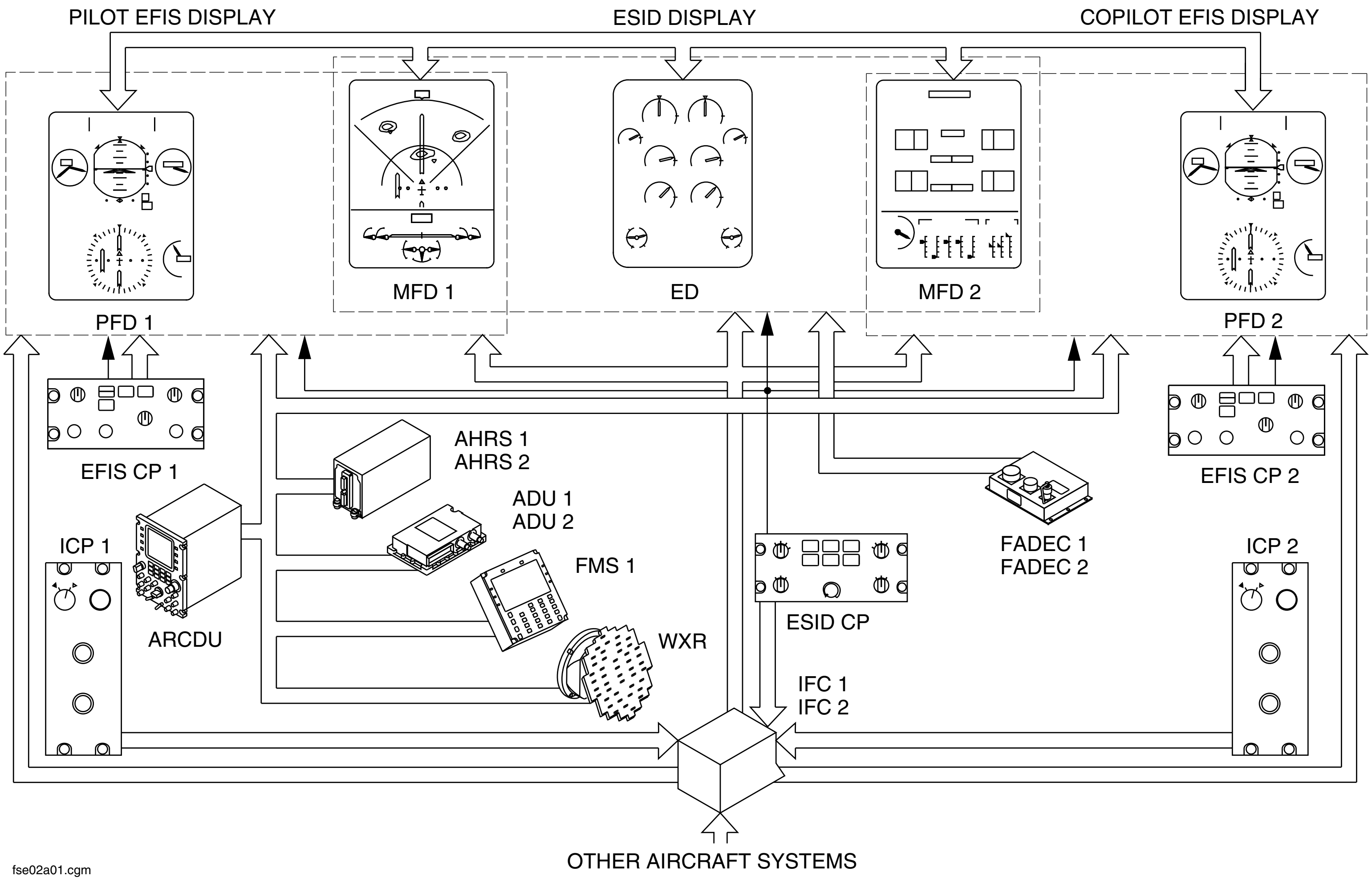
- Front face module
- Interface board
- Outer box.

The Interface board has the functions that follow:

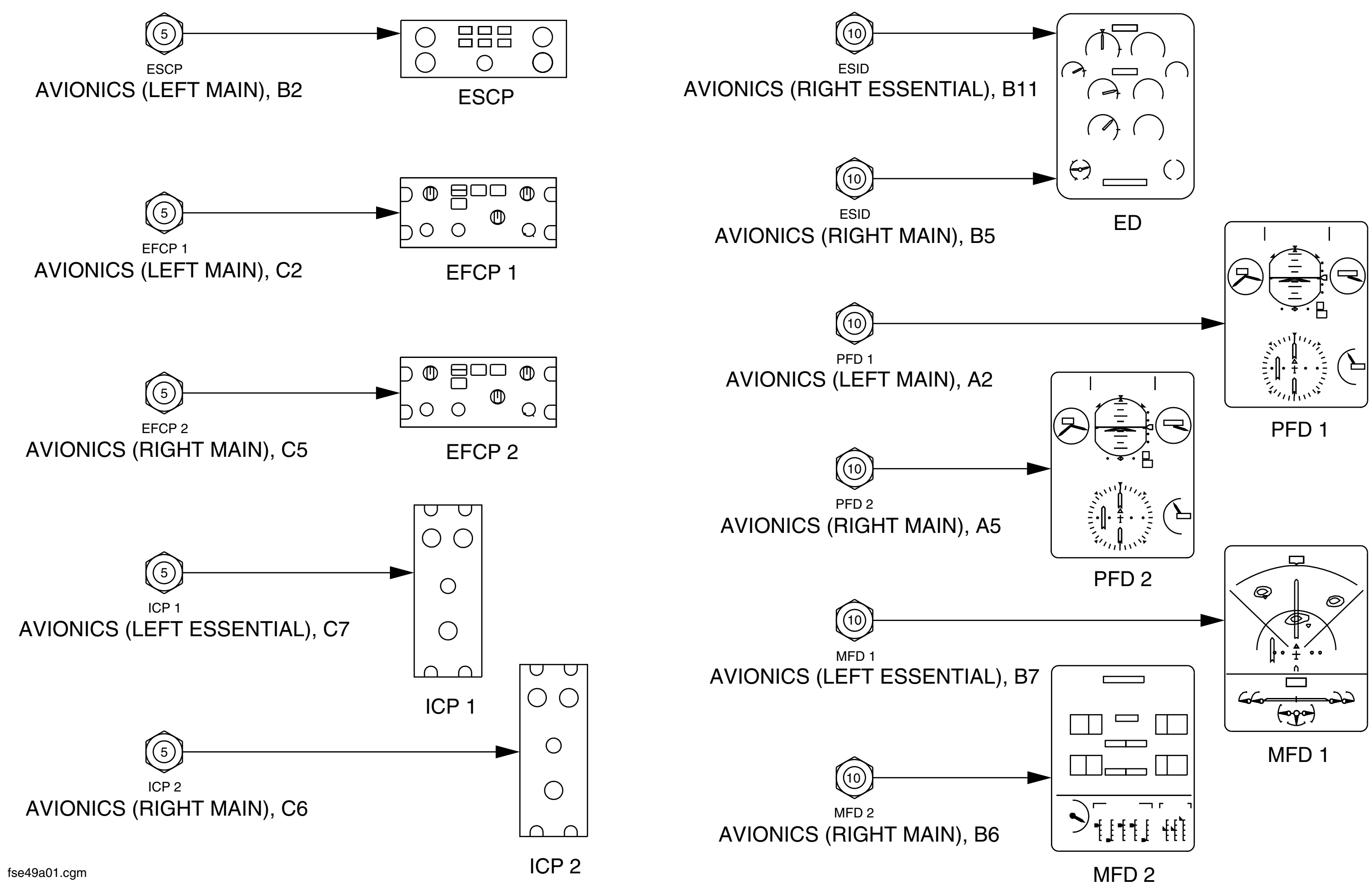
- Power supply
- Parallel/serial converter
- ARINC emmiter/receiver
- EMI, lightning protection.

Each ICP supplies data to its related Input/Output Modules (IOP) and Air Data Unit ADU through ARINC 429 data buses.

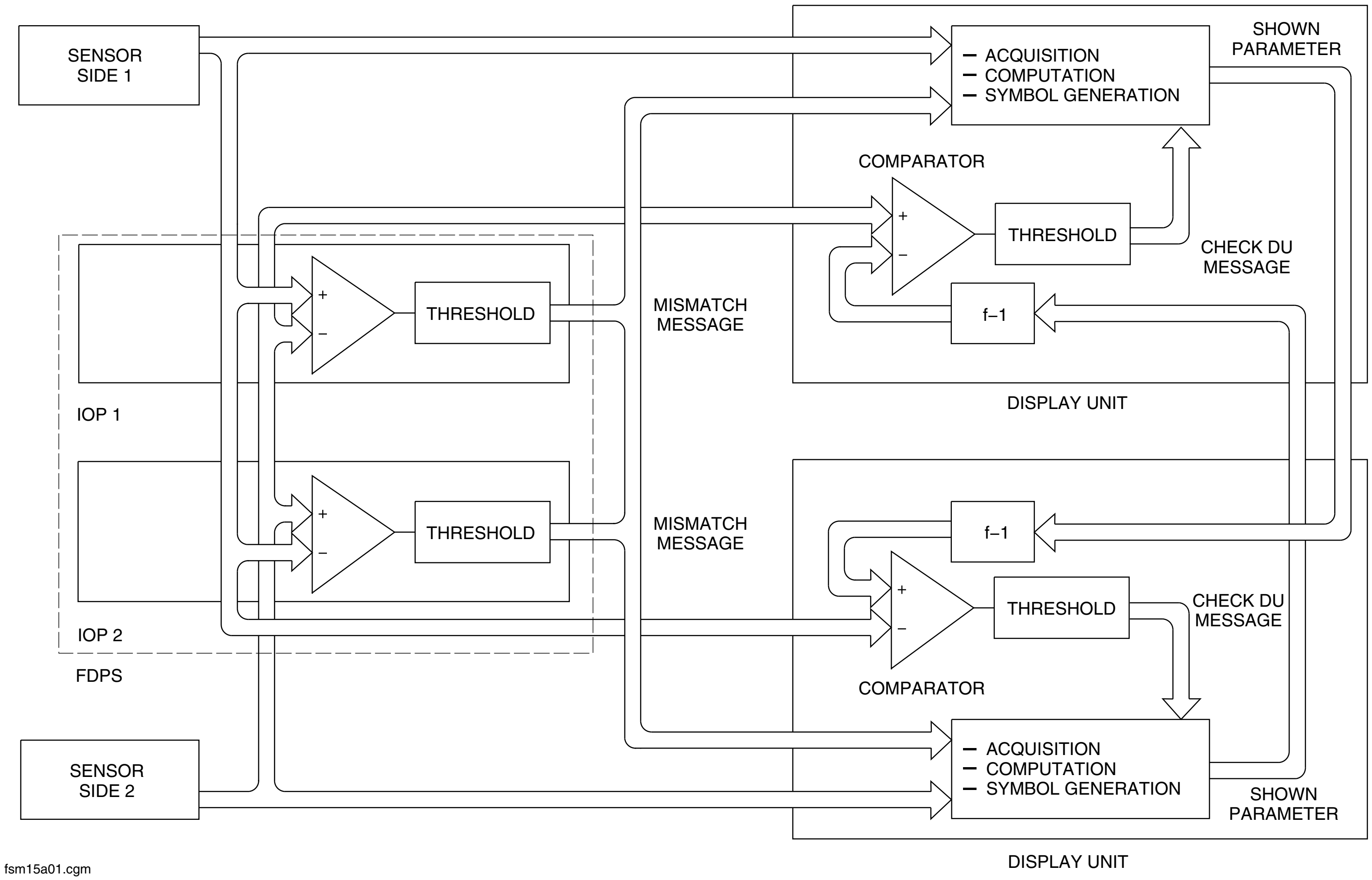
The ICP weighs 2 lb (0.91 kg). It is 2.25 in. (57.2 mm) wide, 5.75 in. (146 mm) high and 5.5 in. (140 mm) deep. The Index Control Panels (ICPs) are attached to the instrument panel with four mounting screws. The fasteners and a bonding wire directly connect to the chassis to make a ground continuity conection between control panel chassis and the aircraft structure.



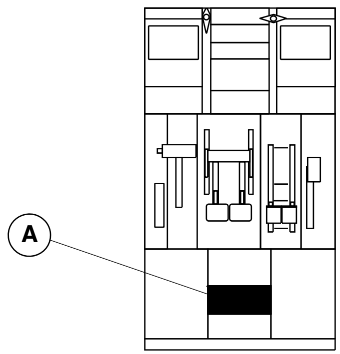
ELECTRONIC INSTRUMENT SYSTEM BLOCK DIAGRAM
Figure 1



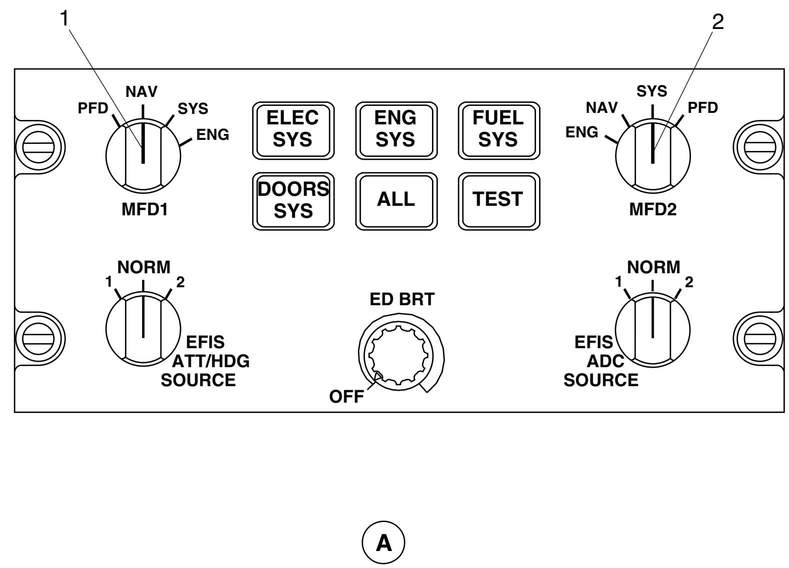
ELECTRONIC INSTRUMENT SYSTEM BLOCK DIAGRAM, POWER
Figure 2



ELECTRONIC INSTRUMENT SYSTEM DISPLAY MONITORING
Figure 3



CENTRE CONSOLE



LEGEND

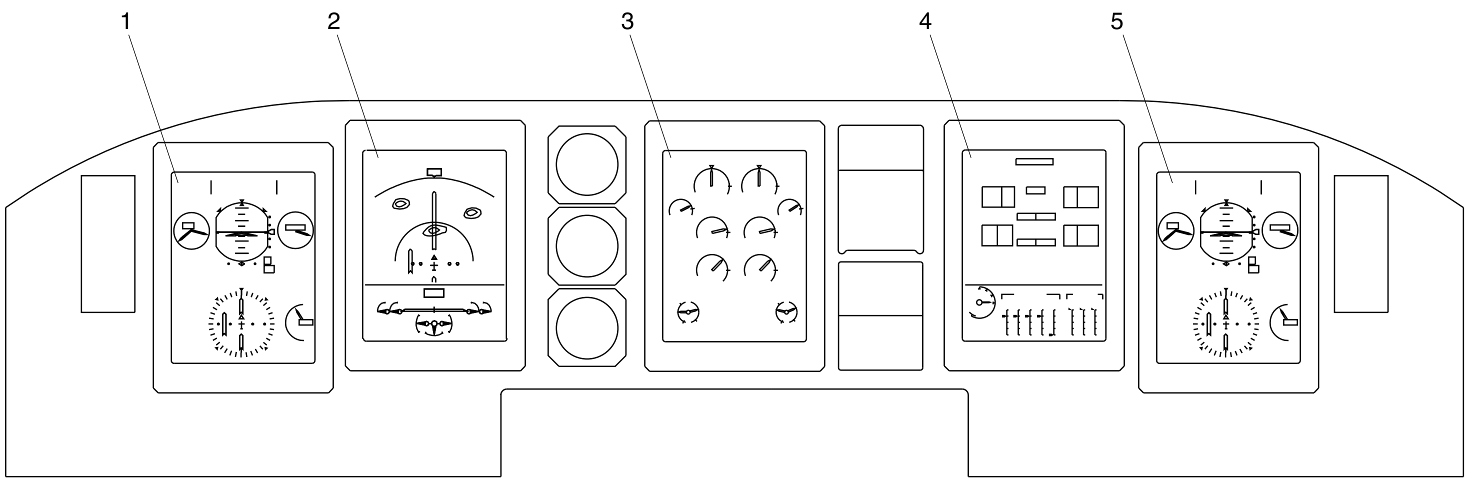
1. MFD 1 Reversion Selector.
2. MFD 2 Reversion Selector.

fs750a01.cgm

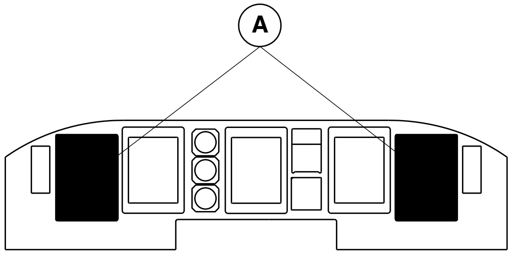
ELECTRONIC INSTRUMENT SYSTEM INDICATION SELECTION
Figure 4

LEGEND

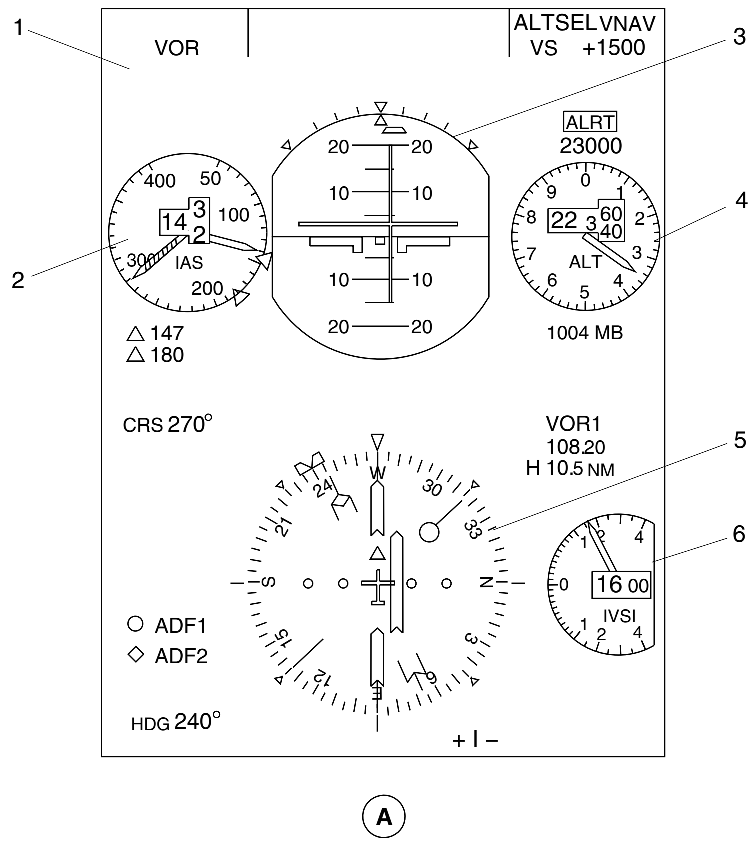
1. PFD1.
2. MFD1.
3. ED.
4. MFD2.
5. PFD2.



ELECTRONIC INSTRUMENT SYSTEM INDICATIONS PAGE 1
Figure 5



MAIN INSTRUMENT PANEL

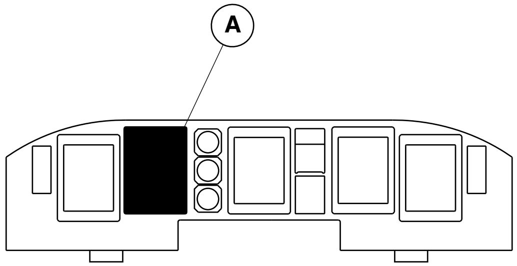


LEGEND

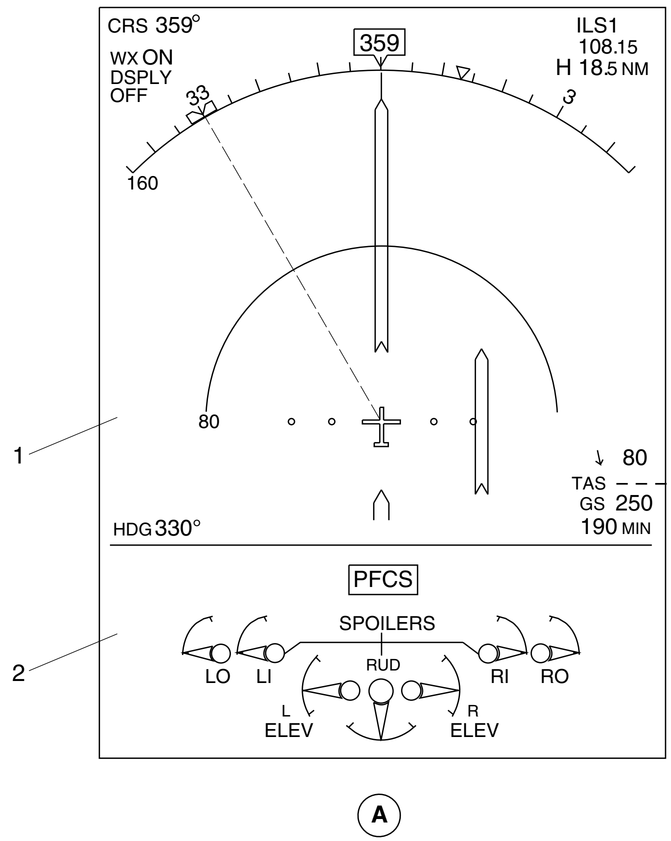
1. Flight Mode Annunciator (FMA).
2. Air Speed Indicator (IAS).
3. Electronic Attitude Direction Indicator (EADI).
4. Altimeter (ALT).
5. Electronic Horizontal Situation Indicator (EHSI).
6. Inertial Vertical Speed Indicator (IVSI), TCAS.

fse25a01.cgm

ELECTRONIC INSTRUMENT SYSTEM PFD INDICATIONS
Figure 6



MAIN INSTRUMENT PANEL

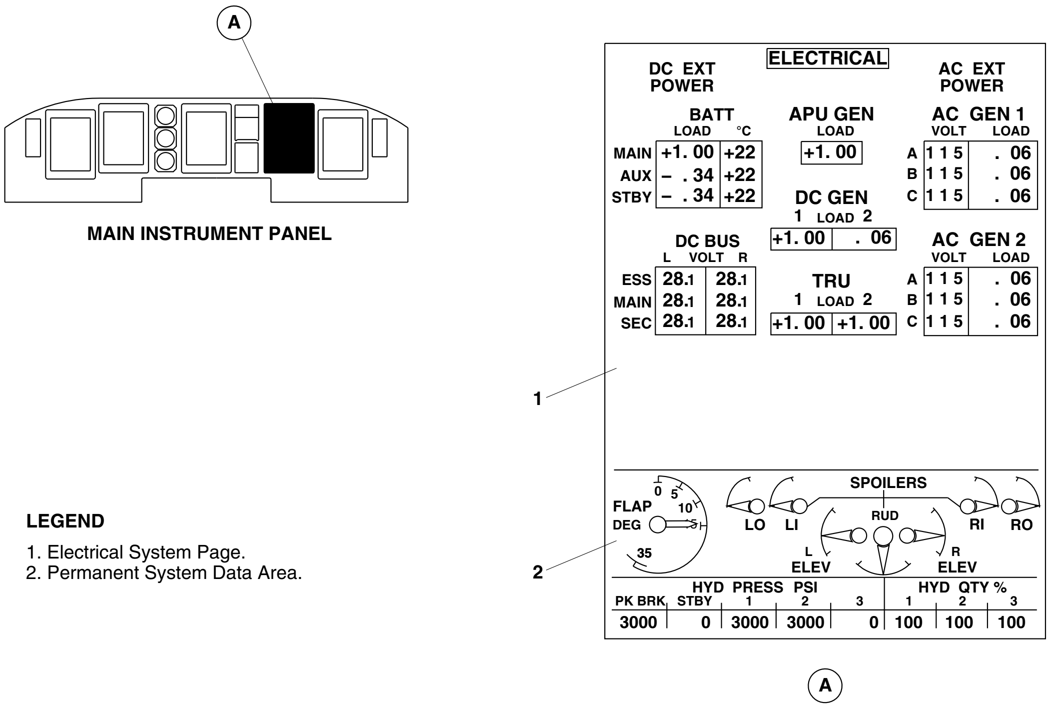


LEGEND

1. System Page or Navigation Page.
2. Permanent System Data Area.

fse31a01.cgm

ELECTRONIC INSTRUMENT SYSTEM MFD1 INDICATIONS
Figure 7

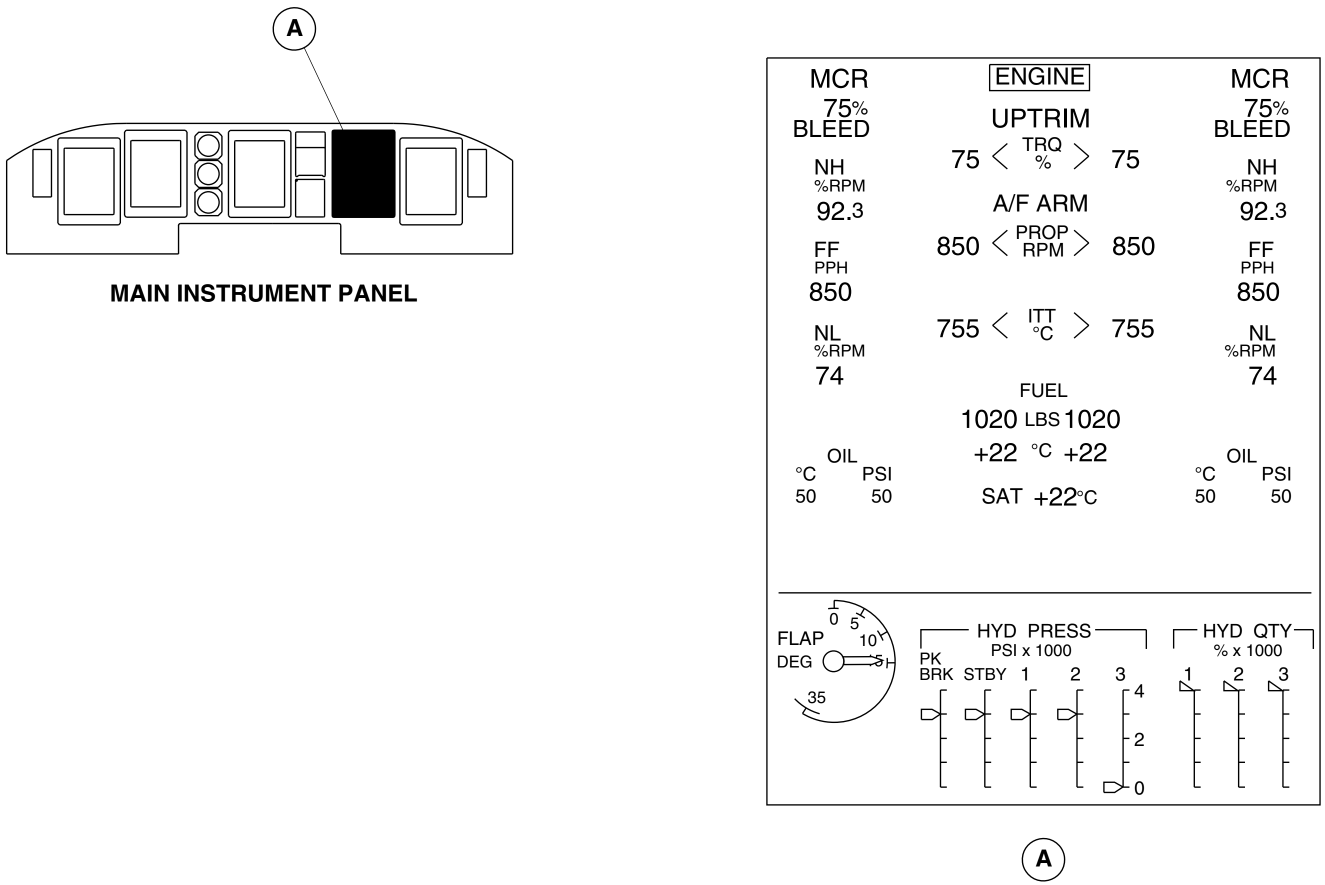


LEGEND

1. Electrical System Page.
2. Permanent System Data Area.

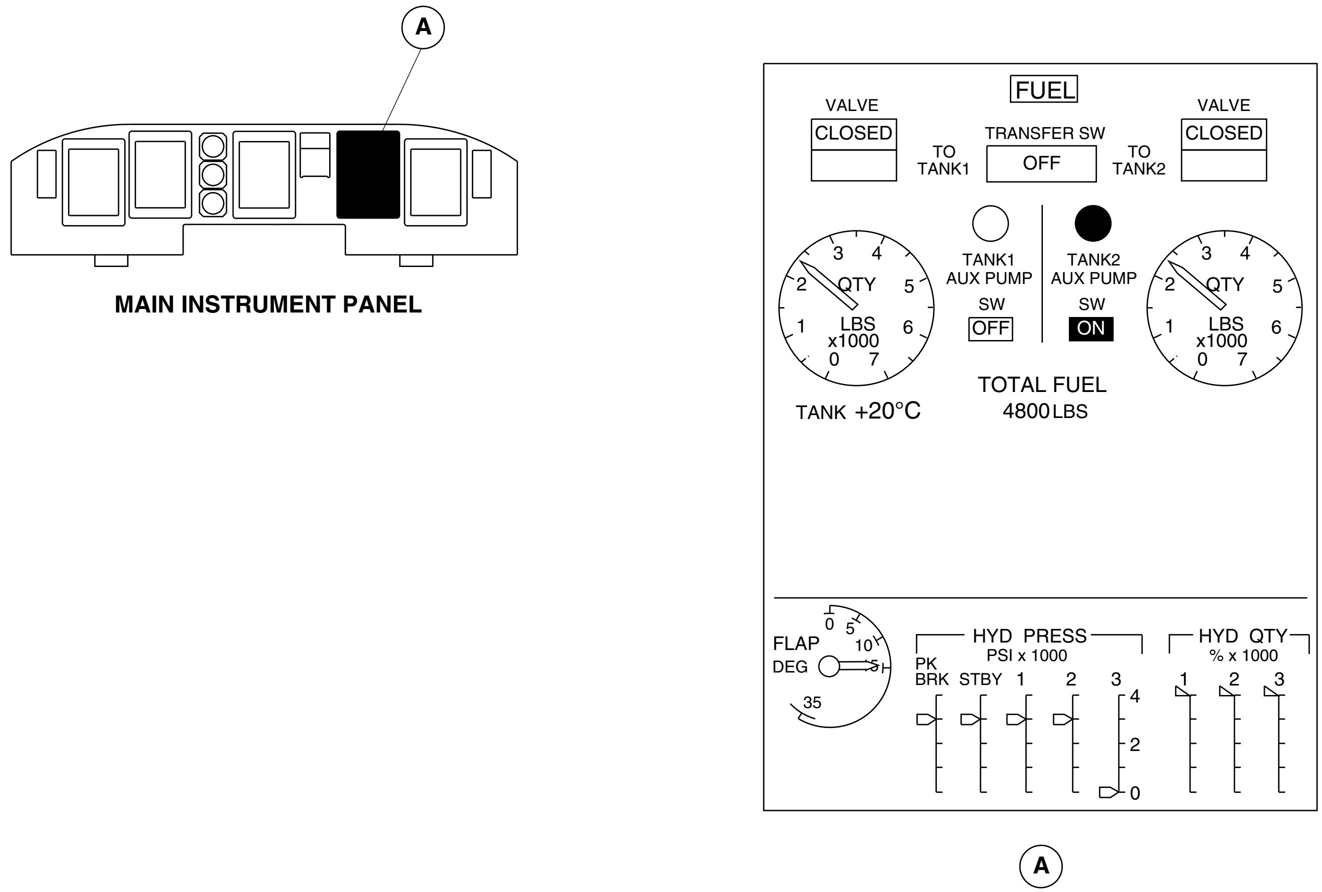
fse32a01.cgm

ELECTRONIC INSTRUMENT SYSTEM MFD2 INDICATIONS
Figure 8



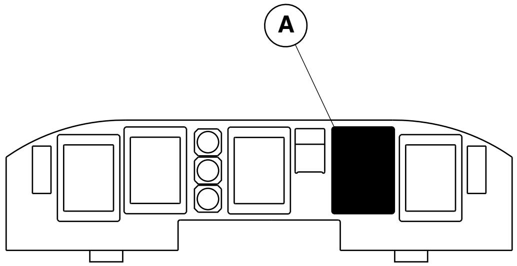
fse61a01.cgm

ELECTRONIC INSTRUMENT SYSTEM ENGINE SYSTEM PAGE
Figure 9

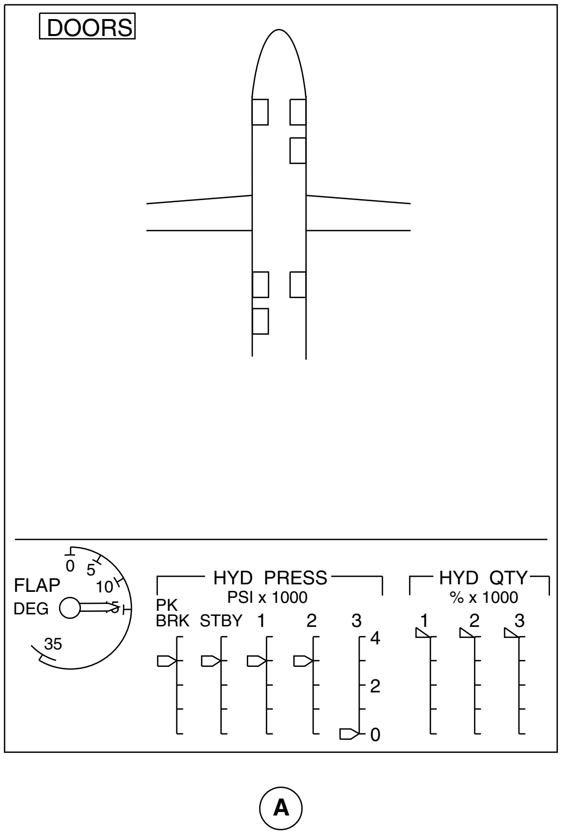


fse30a02.cgm

ELECTRONIC INSTRUMENT SYSTEM FUEL SYSTEM PAGE
Figure 10

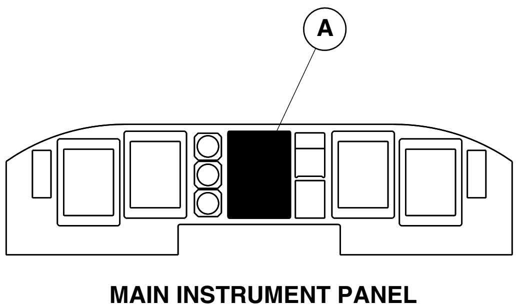


MAIN INSTRUMENT PANEL



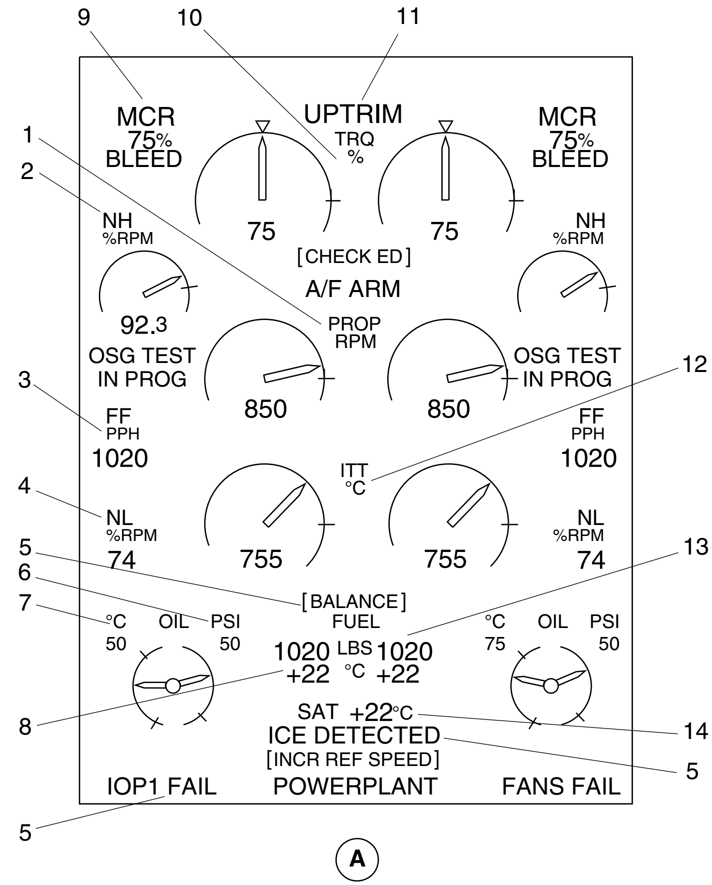
fs748a01.cgm

ELECTRONIC INSTRUMENT SYSTEM DOORS SYSTEM PAGE
Figure 11



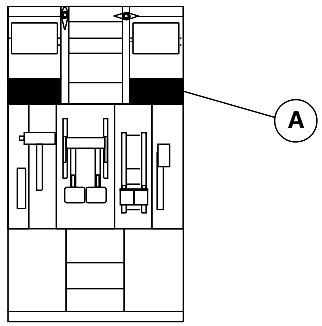
LEGEND

1. NP Propeller Rotational Speed Indicator.
2. NH High Pressure Compressor Rotational Speed Indicator.
3. Fuel Flow Indicator.
4. NL Low Pressure Compressor Rotational Speed Indicator.
5. Advisory Messages.
6. Oil Pressure Indication.
8. Fuel Temperature Indication.
9. Engine Rating Mode Indication.
10. Torque Indication.
11. Powerplant Messages.
12. ITT Inter–Turbine Temperature.
13. Tank Fuel Quantity Indication.
14. Static Air Temperature Indication.

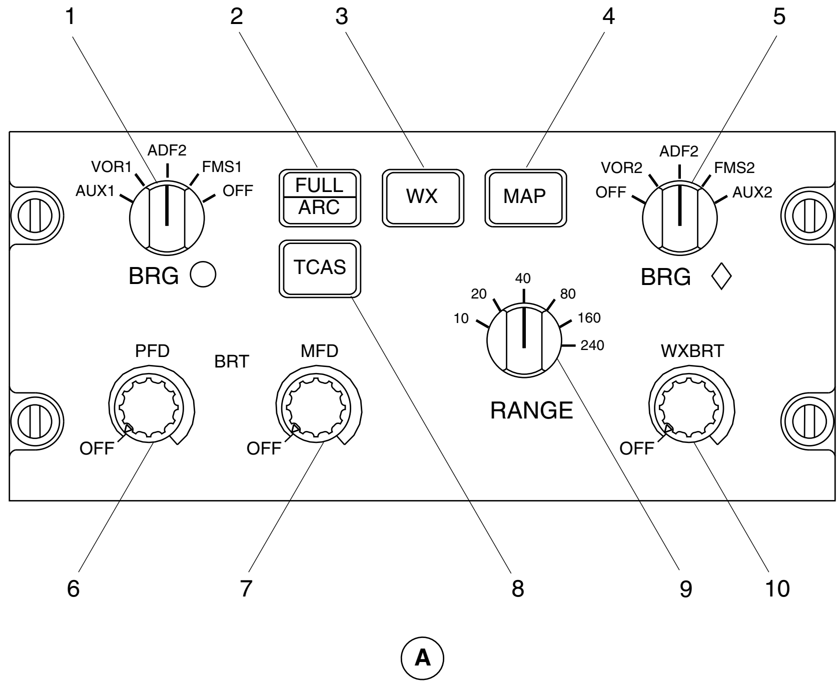


fse57a02.cgm

ELECTRONIC INSTRUMENT SYSTEM ED INDICATIONS
Figure 12



CENTRE CONSOLE

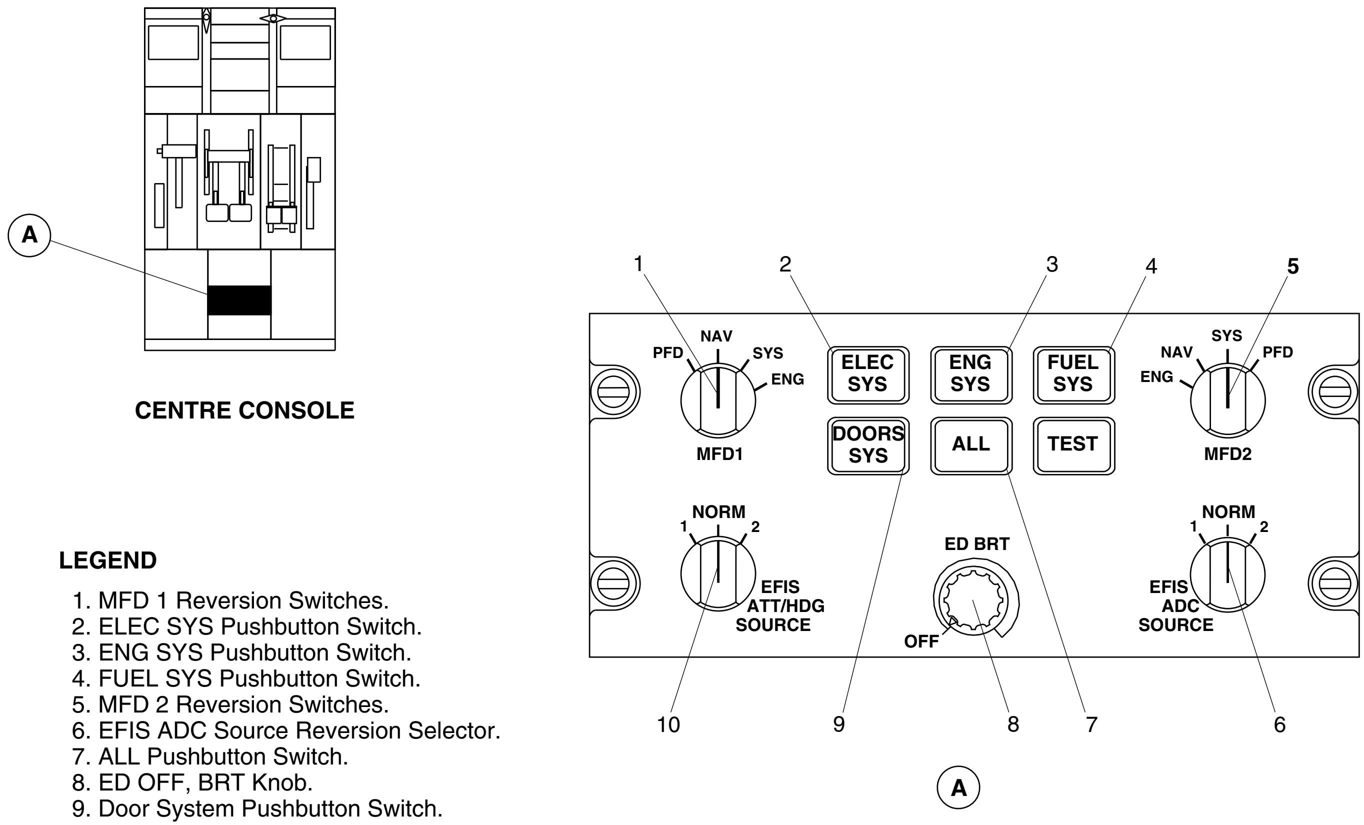


LEGEND

1. BRG 1 Selector Knob.
2. FULL/ARC Mode Pushbutton Switch.
3. WX Mode Pushbutton Switch.
4. FMS MAP Mode Pushbutton Switch.
5. BRG 2 Selector Knob.
6. PFD OFF, BRT Knob .
7. MFD OFF, BRT Knob.
8. TCAS Pushbutton Switch.
9. RANGE Selector Knob.
10. WXBRT Knob.

fsg30a02.cgm

ELECTRONIC INSTRUMENT SYSTEM EFCP PAGE 1
Figure 13

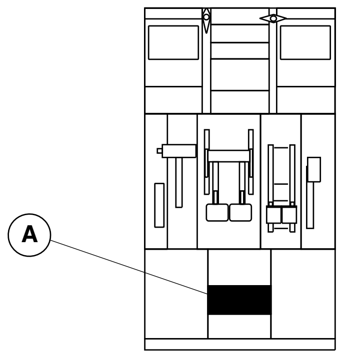


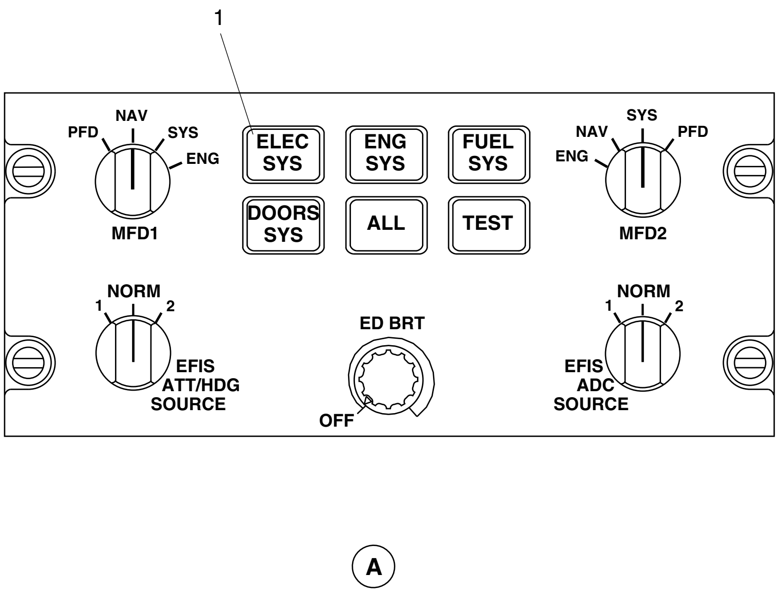
LEGEND

1. MFD 1 Reversion Switches.
2. ELEC SYS Pushbutton Switch.
3. ENG SYS Pushbutton Switch.
4. FUEL SYS Pushbutton Switch.
5. MFD 2 Reversion Switches.
6. EFIS ADC Source Reversion Selector.
7. ALL Pushbutton Switch.
8. ED OFF, BRT Knob.
9. Door System Pushbutton Switch.
10. EFIS ATT/HDG Source Reversion Selector.

fsa73a12.cgm

ELECTRONIC INSTRUMENT SYSTEM ESCP PAGE 2
Figure 14





LEGEND

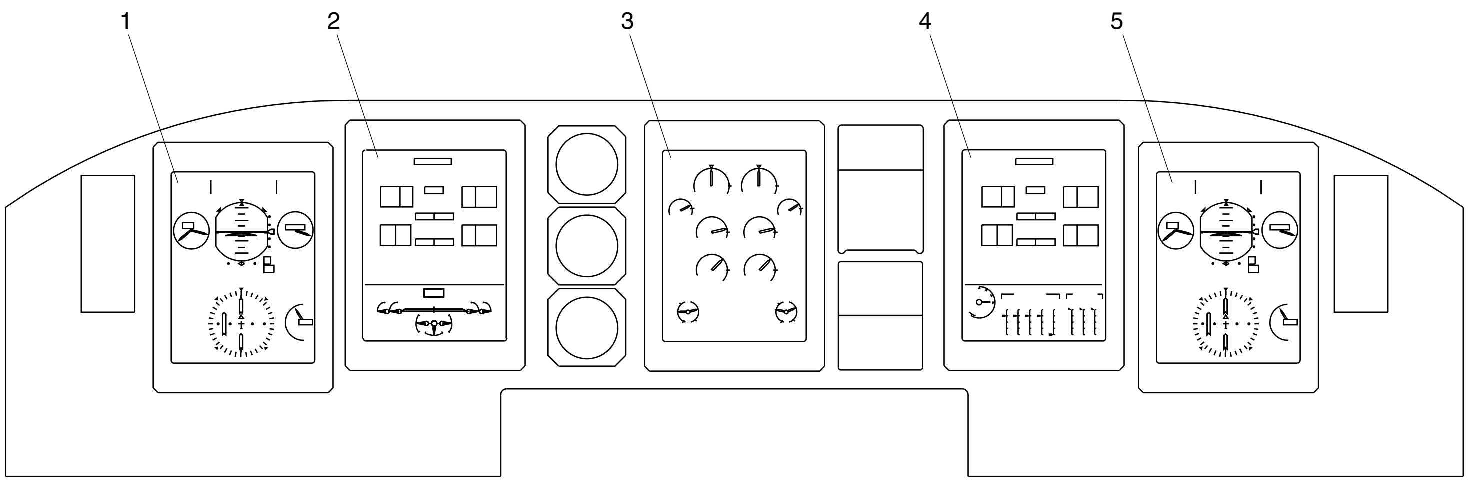
1. Electrical System Pushbutton Switch.

fsb78a01.cgm

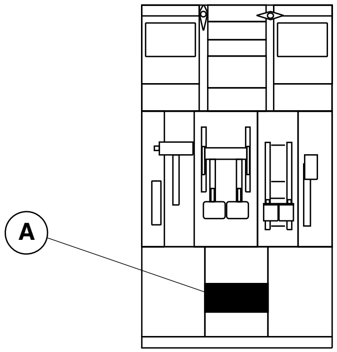
ELECTRONIC INSTRUMENT SYSTEM ESCP, ELECTRICAL SYSTEM PAGE SELECTION
Figure 15

LEGEND

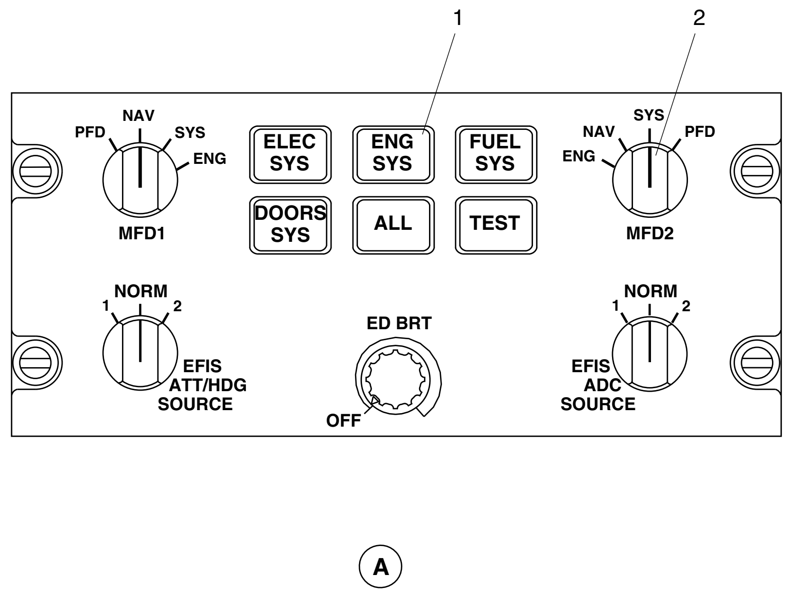
1. PFD1.
2. MFD1.
3. ED.
4. MFD2.
5. PFD2



ELECTRONIC INSTRUMENT SYSTEM ESCP, ELECTRICAL SYSTEM PAGE INDICATION
Figure 16



CENTRE CONSOLE



LEGEND

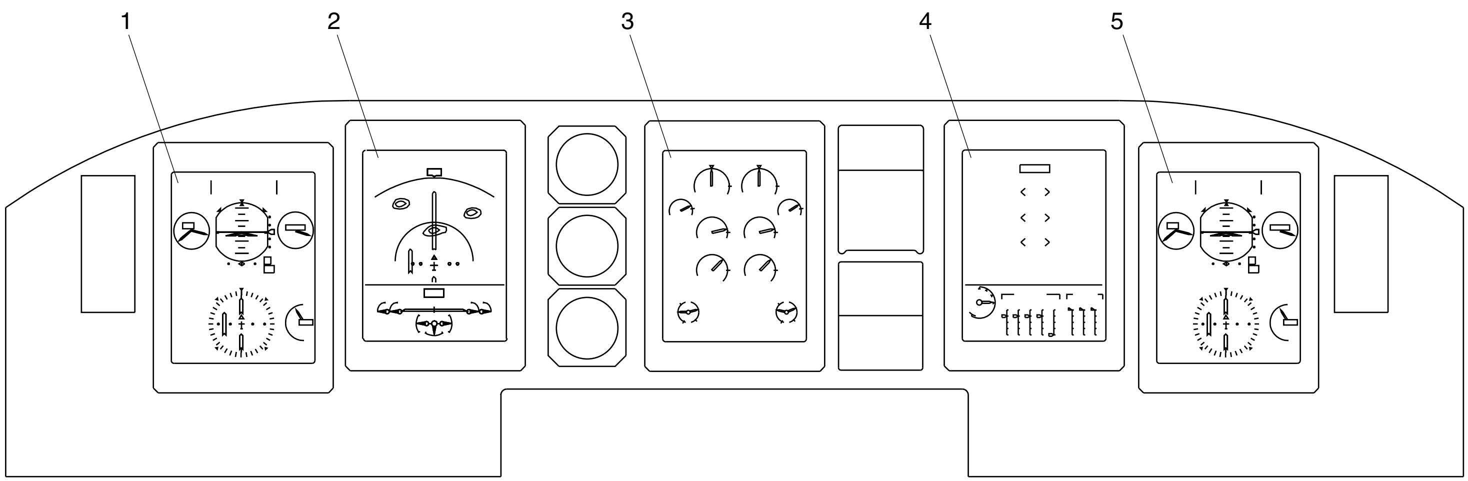
1. Engine System Pushbutton Switch.
2. MFD2 Reversion Selector.

fsb80a03.cgm

ELECTRONIC INSTRUMENT SYSTEM ESCP, ENGINE SYSTEM PAGE SELECTION
Figure 17

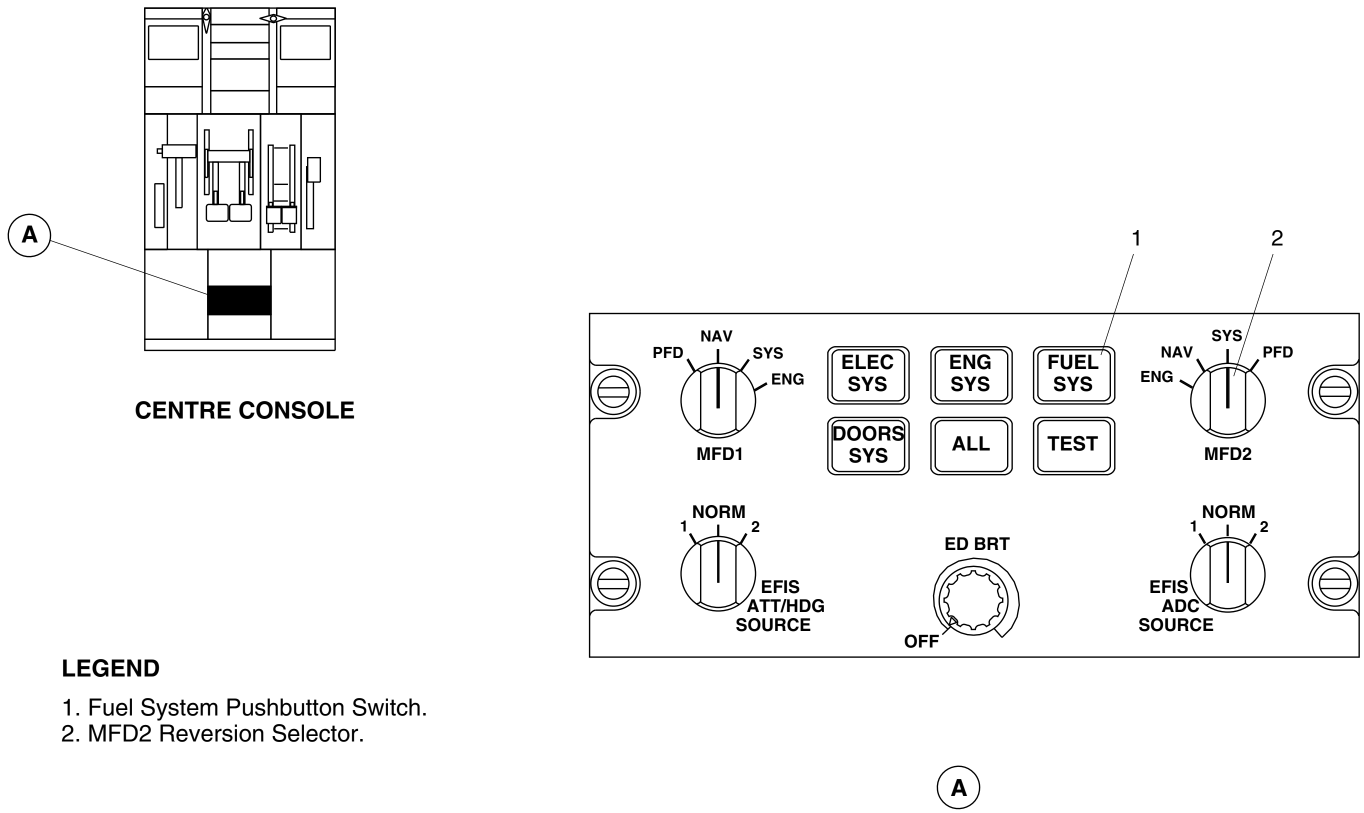
LEGEND

1. PFD1.
2. MFD1.
3. ED.
4. MFD2.
5. PFD2.



fsa97a09.cgm

ELECTRONIC INSTRUMENT SYSTEM ESCP, ENGINE SYSTEM PAGE INDICATION
Figure 18

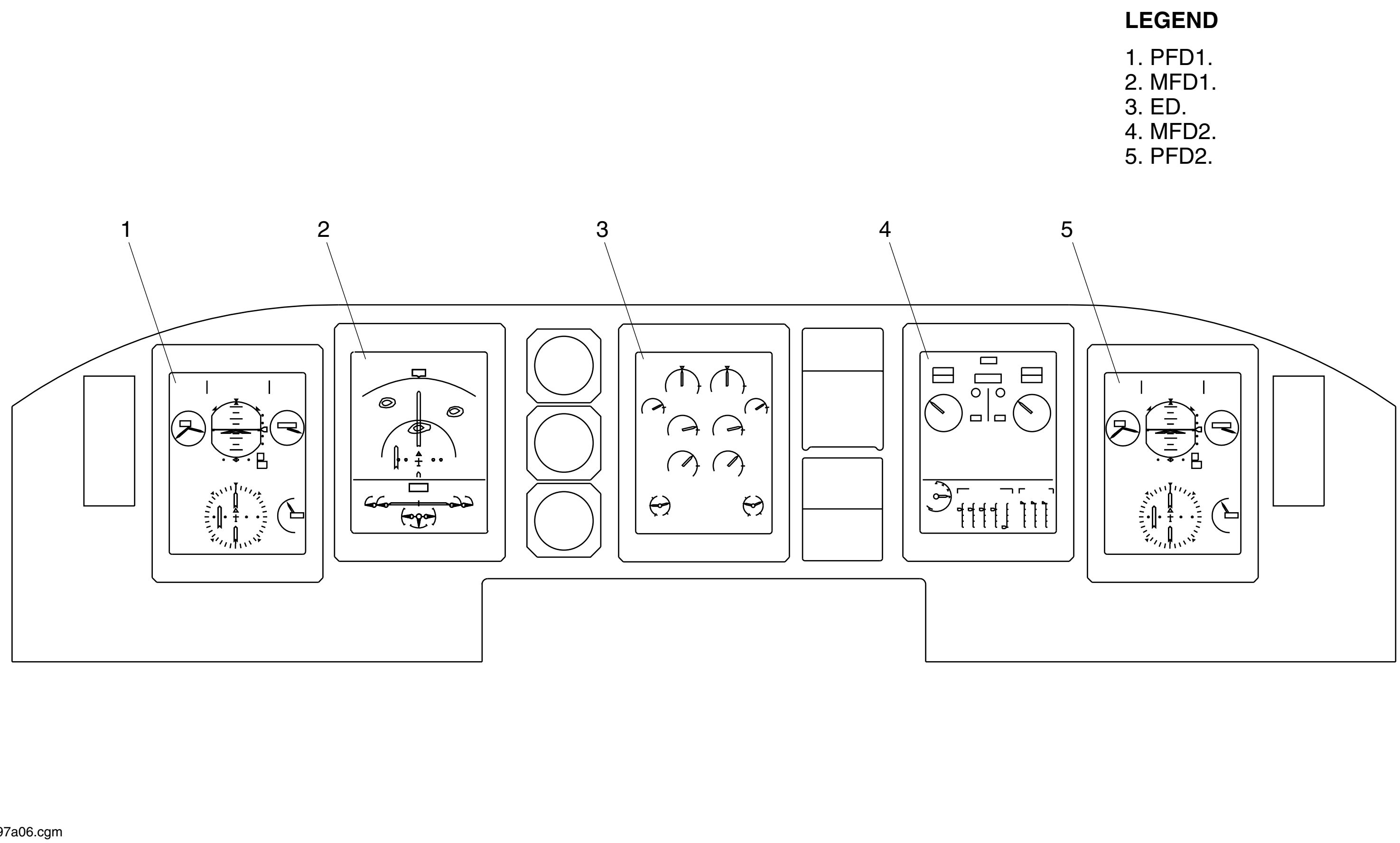


LEGEND

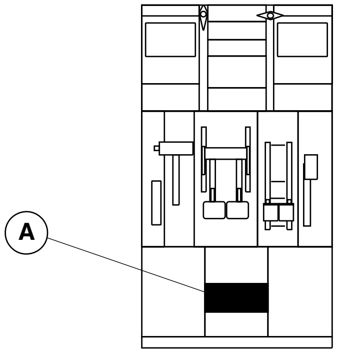
1. Fuel System Pushbutton Switch.
2. MFD2 Reversion Selector.

fsb80a01.cgm

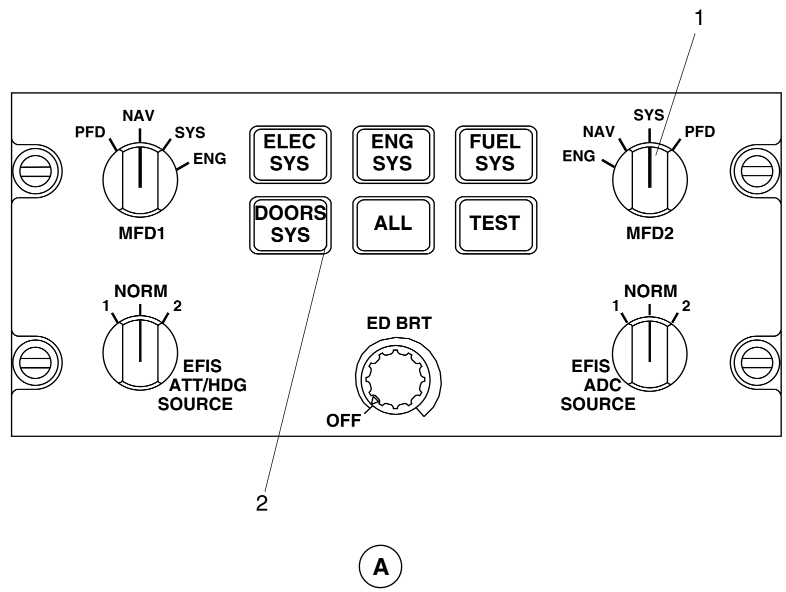
ELECTRONIC INSTRUMENT SYSTEM ESCP, FUEL SYSTEM PAGE SELECTION
Figure 19



ELECTRONIC INSTRUMENT SYSTEM ESCP, FUEL SYSTEM PAGE INDICATION
Figure 20



CENTRE CONSOLE



LEGEND

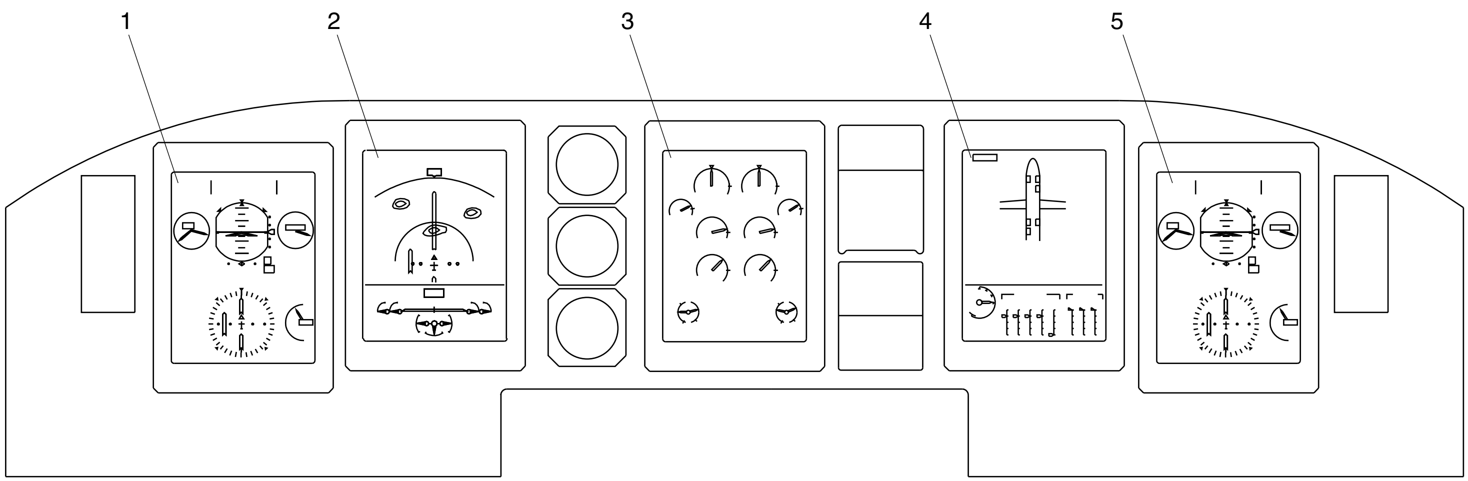
1. MFD2 Reversion Selector.
2. Doors System Pushbutton Switch

fsb80a04.cgm

ELECTRONIC INSTRUMENT SYSTEM ESCP, DOORS SYSTEM PAGE SELECTION
Figure 21

LEGEND

1. PFD1.
2. MFD1.
3. ED.
4. MFD2.
5. PFD2.

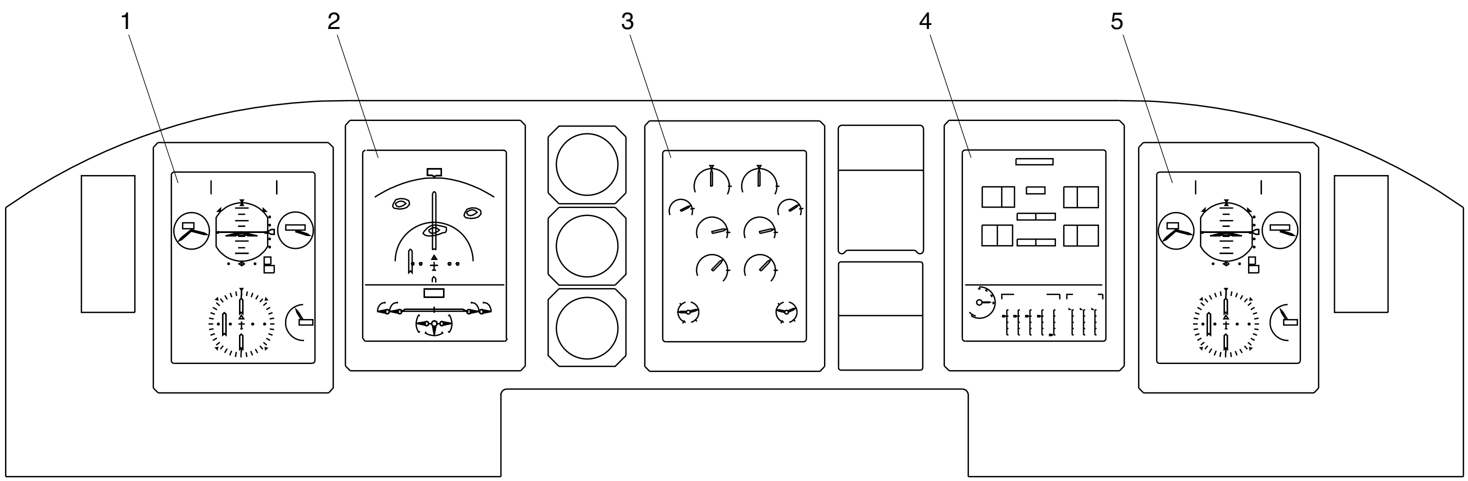


fsa97a10.cgm

ELECTRONIC INSTRUMENT SYSTEM ESCP, DOORS SYSTEM PAGE INDICATION
Figure 22

LEGEND

1. PFD1.
2. MFD1.
3. ED.
4. MFD2.
5. PFD2.

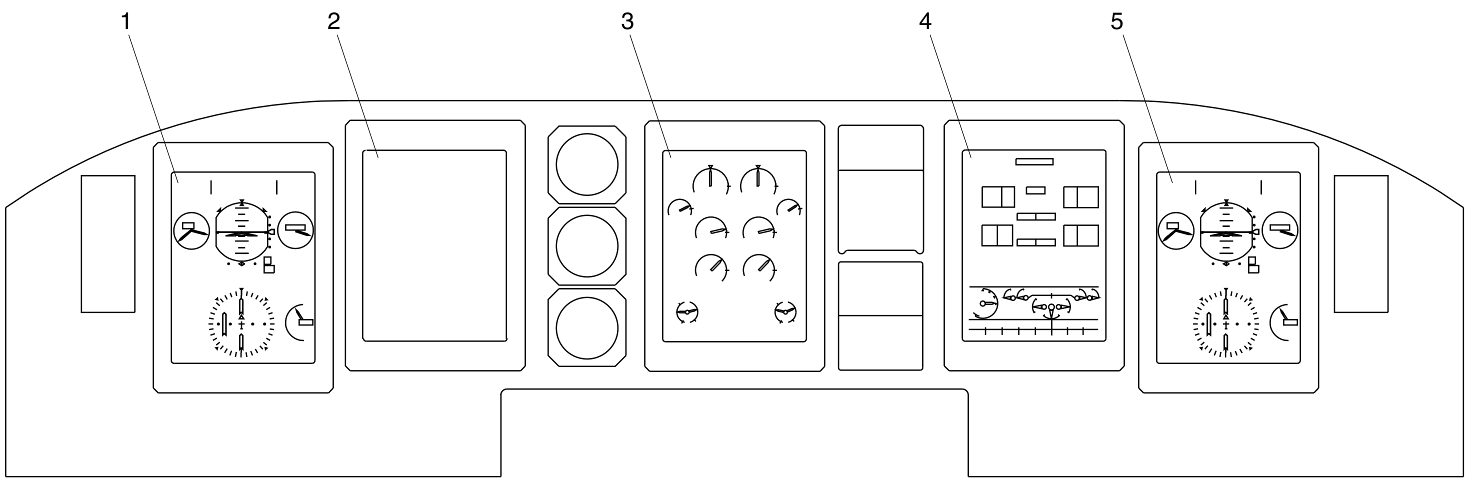


fsa97a08.cgm

ELECTRONIC INSTRUMENT SYSTEM INDICATIONS PAGE 2
Figure 23

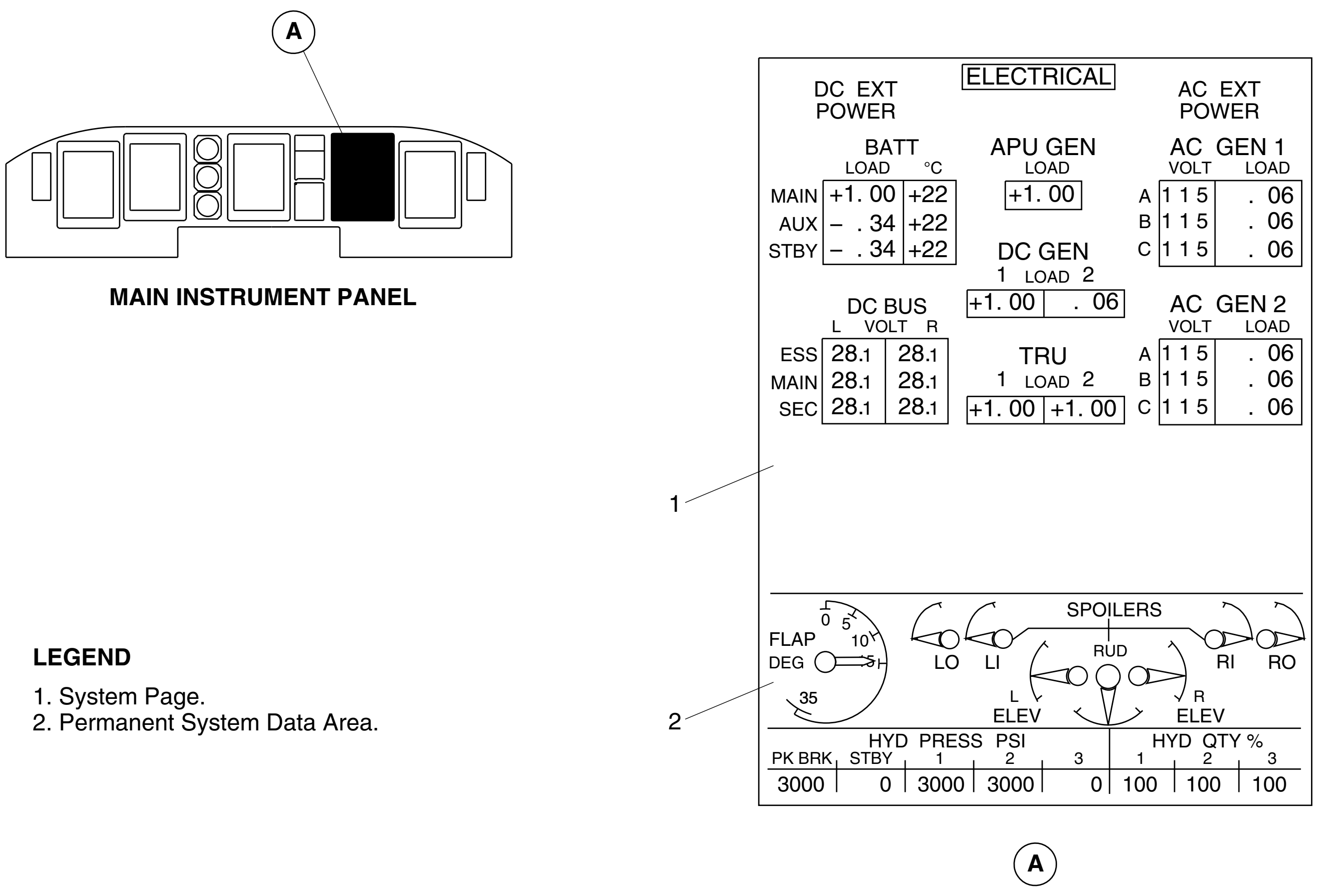
LEGEND

1. PFD1.
2. MFD1 Malfunction.
3. ED.
4. MFD2.
5. PFD2



fsa97a11.cgm

ELECTRONIC INSTRUMENT SYSTEM MFD1 MALFUNCTION
Figure 24



LEGEND

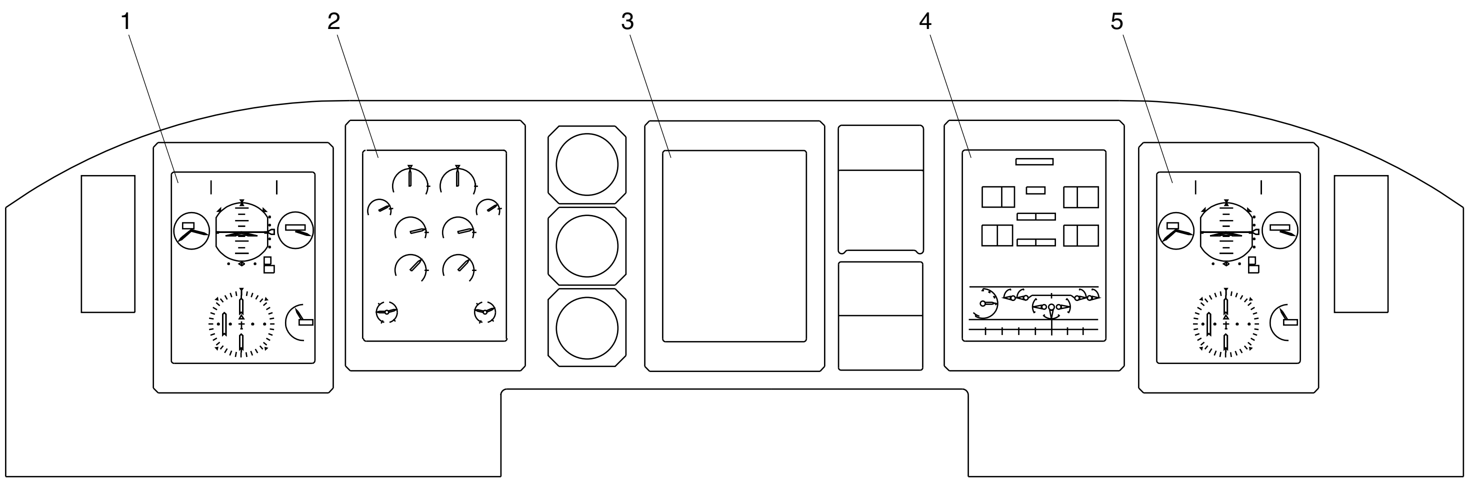
1. System Page.
2. Permanent System Data Area.

fse33a01.cgm

ELECTRONIC INSTRUMENT SYSTEM PERMANENT SYSTEM DATA AREA, COMPOSITE VIEW
Figure 25

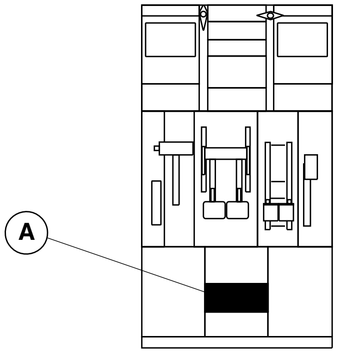
LEGEND

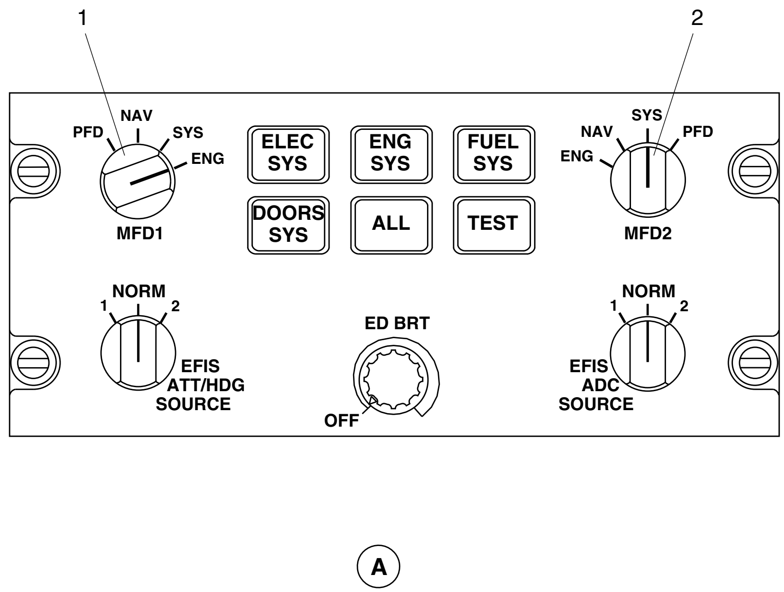
1. PFD1.
2. MFD1 Showing ED.
3. ED Malfunction.
4. MFD2.
5. PFD2



fsa97a02.cgm

ELECTRONIC INSTRUMENT SYSTEM ED MALFUNCTION
Figure 26





LEGEND

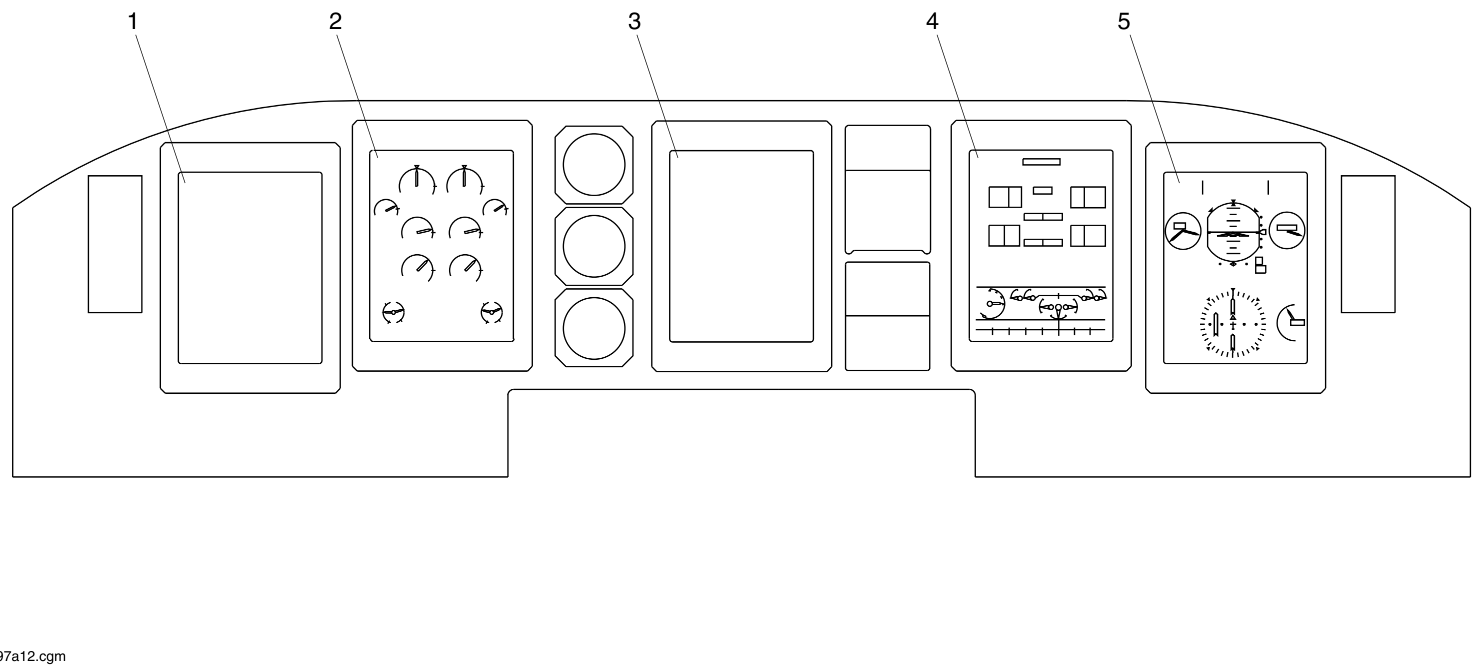
1. MFD 1 Reversion Selector.
2. MFD 2 Reversion Selector.

fsb76a01.cgm

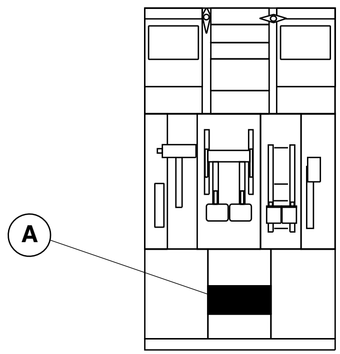
ELECTRONIC INSTRUMENT SYSTEM ESCP, ED MFD1 REVERSION SELECTION
Figure 27

LEGEND

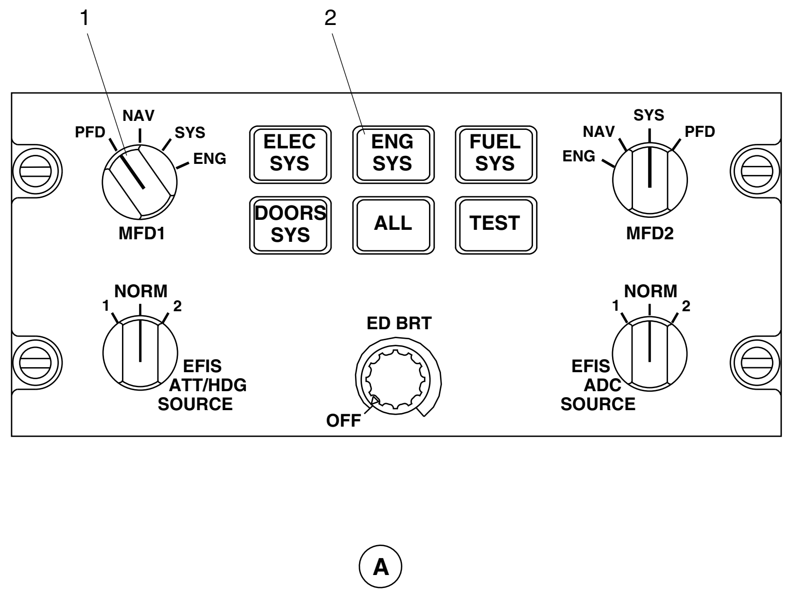
1. PFD1 Malfunction.
2. MFD1 Showing ED.
3. ED Malfunction.
4. MFD2.
5. PFD2.



ELECTRONIC INSTRUMENT SYSTEM PFD1 AND ED MALFUNCTION
Figure 28



CENTRE CONSOLE



LEGEND

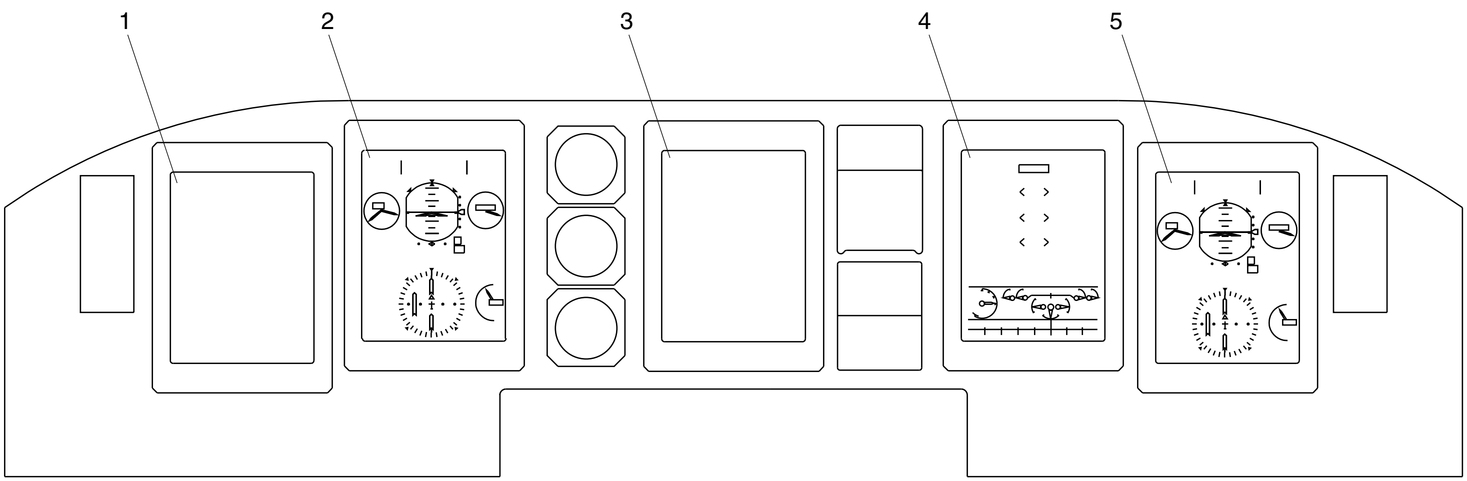
1. MFD1 Reversion Selector.
2. Engine System Pushbutton Switch.

fsb77a01.cgm

ELECTRONIC INSTRUMENT SYSTEM ESCP, PFD MFD1 REVERSION AND ENGINE SYSTEM PAGE SELECTIONS
Figure 29

LEGEND

1. PFD1 Malfunction.
2. MFD1 Showing PFD.
3. ED Malfunction.
4. MFD2.
5. PFD2.

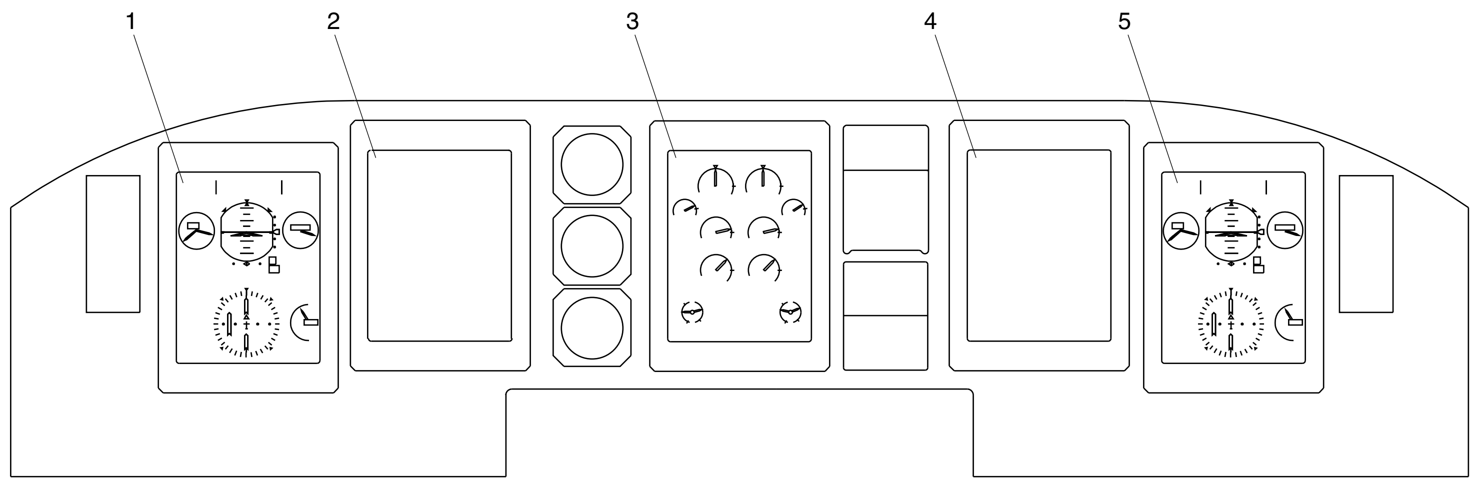


fsa97a04.cgm

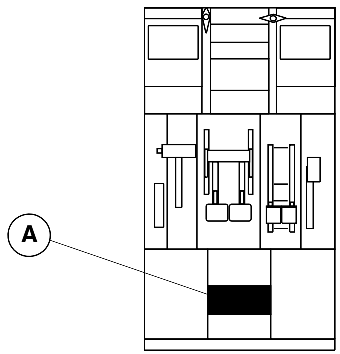
ELECTRONIC INSTRUMENT SYSTEM, PFD1 AND ED REVERSIONARY INDICATION
Figure 30

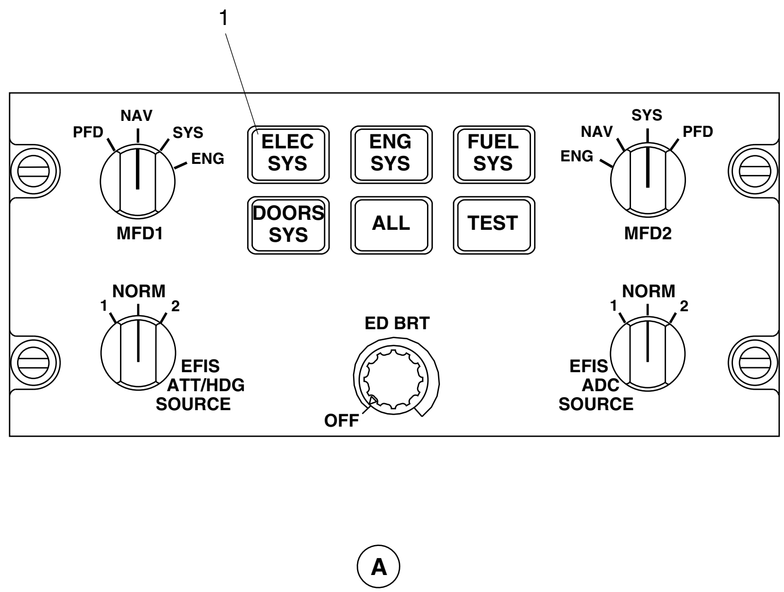
LEGEND

1. PFD1.
2. MFD1 Malfunction.
3. ED.
4. MFD2 Malfunction.
5. PFD2.



ELECTRONIC INSTRUMENT SYSTEM MFD1 AND MFD2 MALFUNCTION
Figure 31





LEGEND

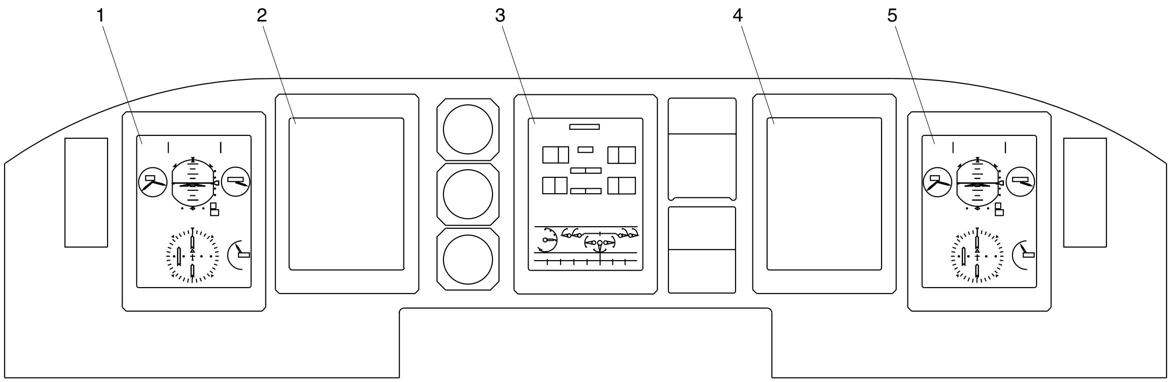
1. Electrical System Pushbutton Switch.

fsb78a01.cgm

ELECTRONIC INSTRUMENT SYSTEM ESCP, MONO ELECTRICAL SYSTEM PAGE SELECTION
Figure 32

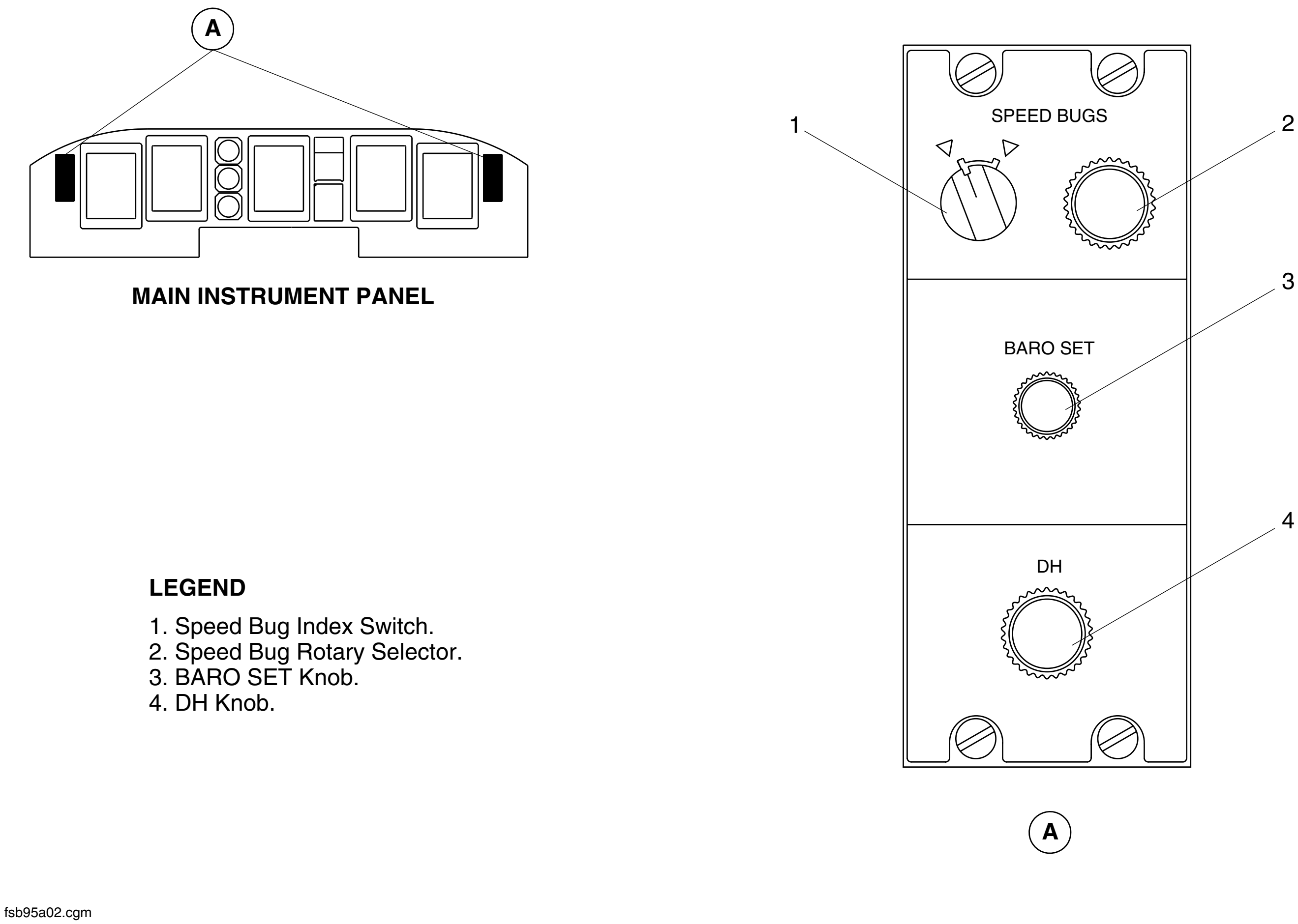
LEGEND

1. PFD1.
2. MFD1 Malfunction.
3. ED Showing System Page.
4. MFD2 Malfunction.
5. PFD2.



fsa97a13.cgm

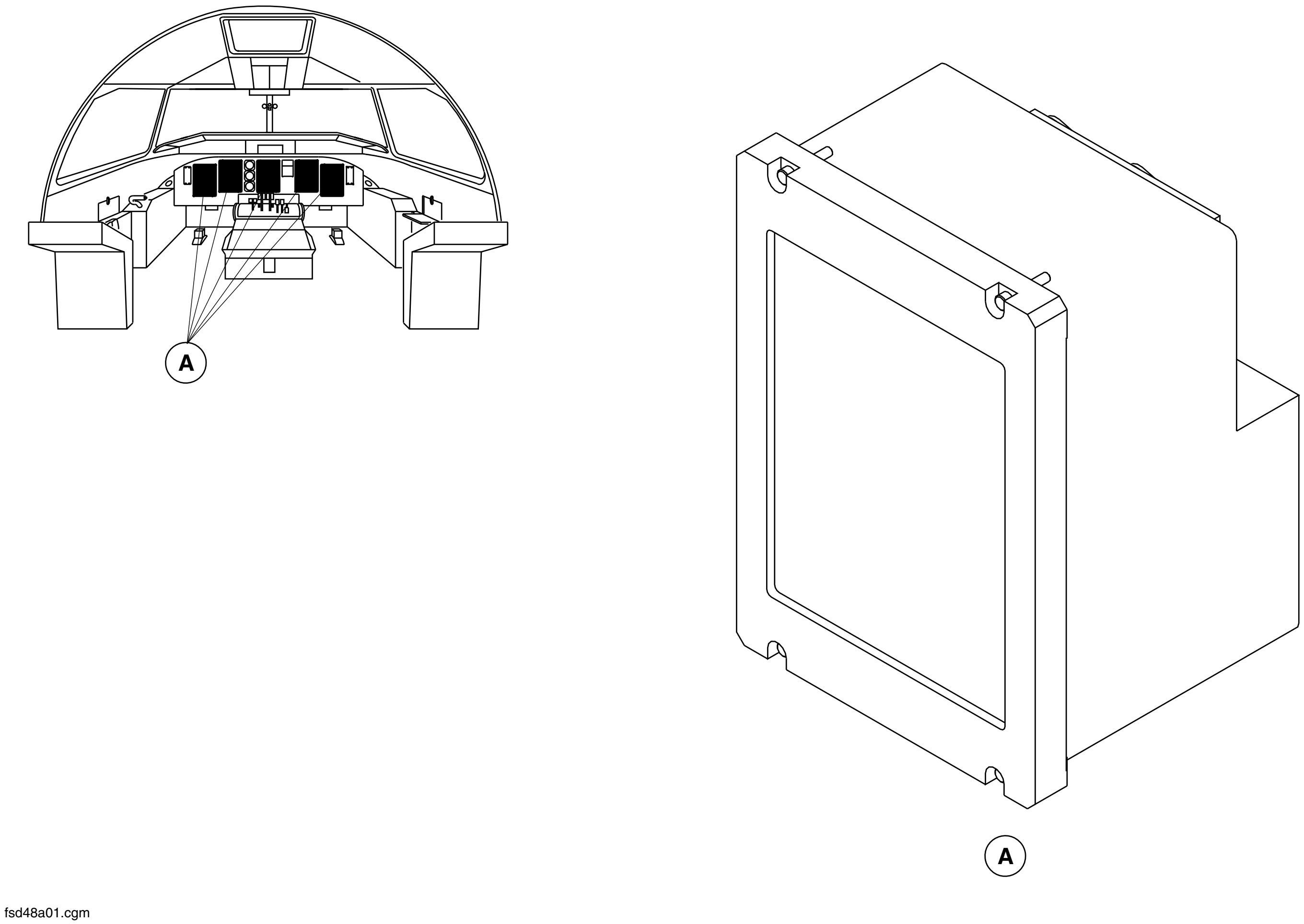
ELECTRONIC INSTRUMENT SYSTEM ESCP, MONO ELECTRICAL SYSTEM PAGE INDICATION
Figure 33



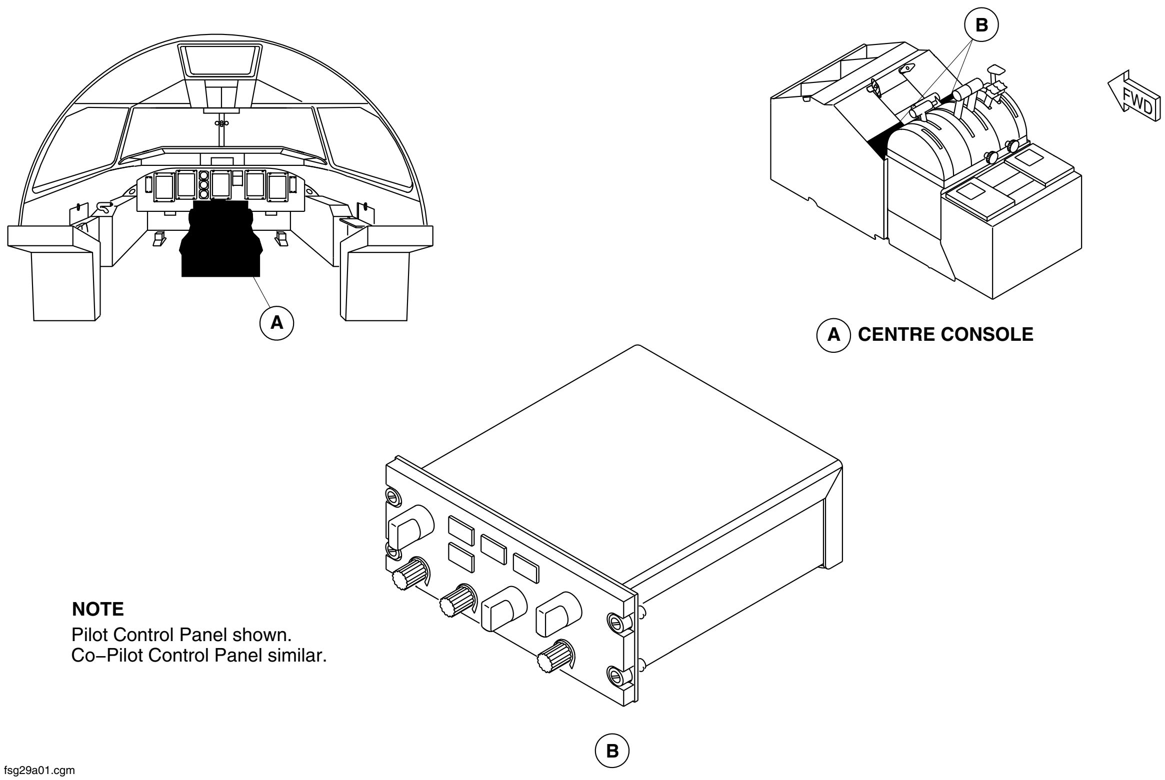
LEGEND

1. Speed Bug Index Switch.
2. Speed Bug Rotary Selector.
3. BARO SET Knob.
4. DH Knob.

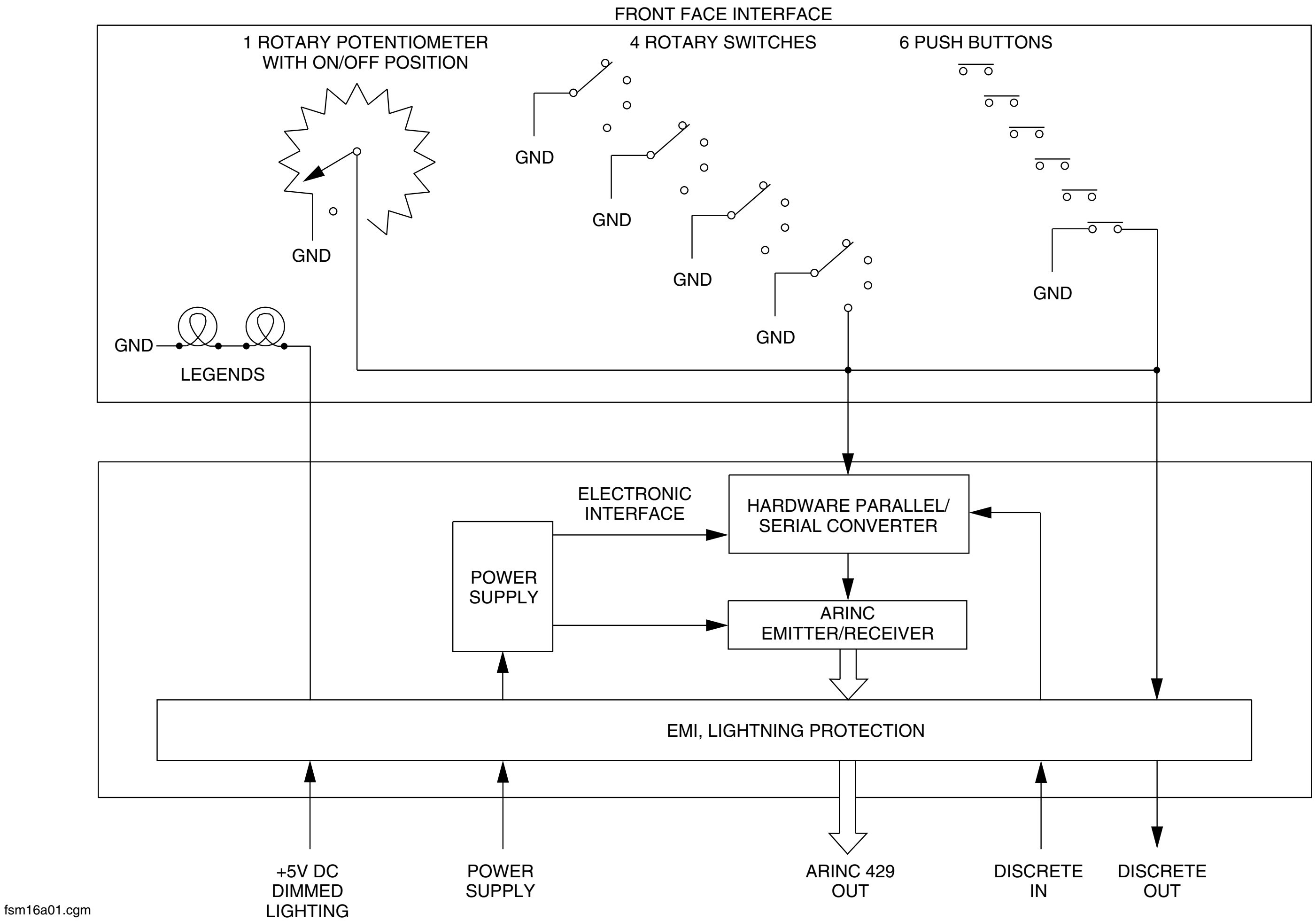
ELECTRONIC INSTRUMENT SYSTEM ICP
Figure 34



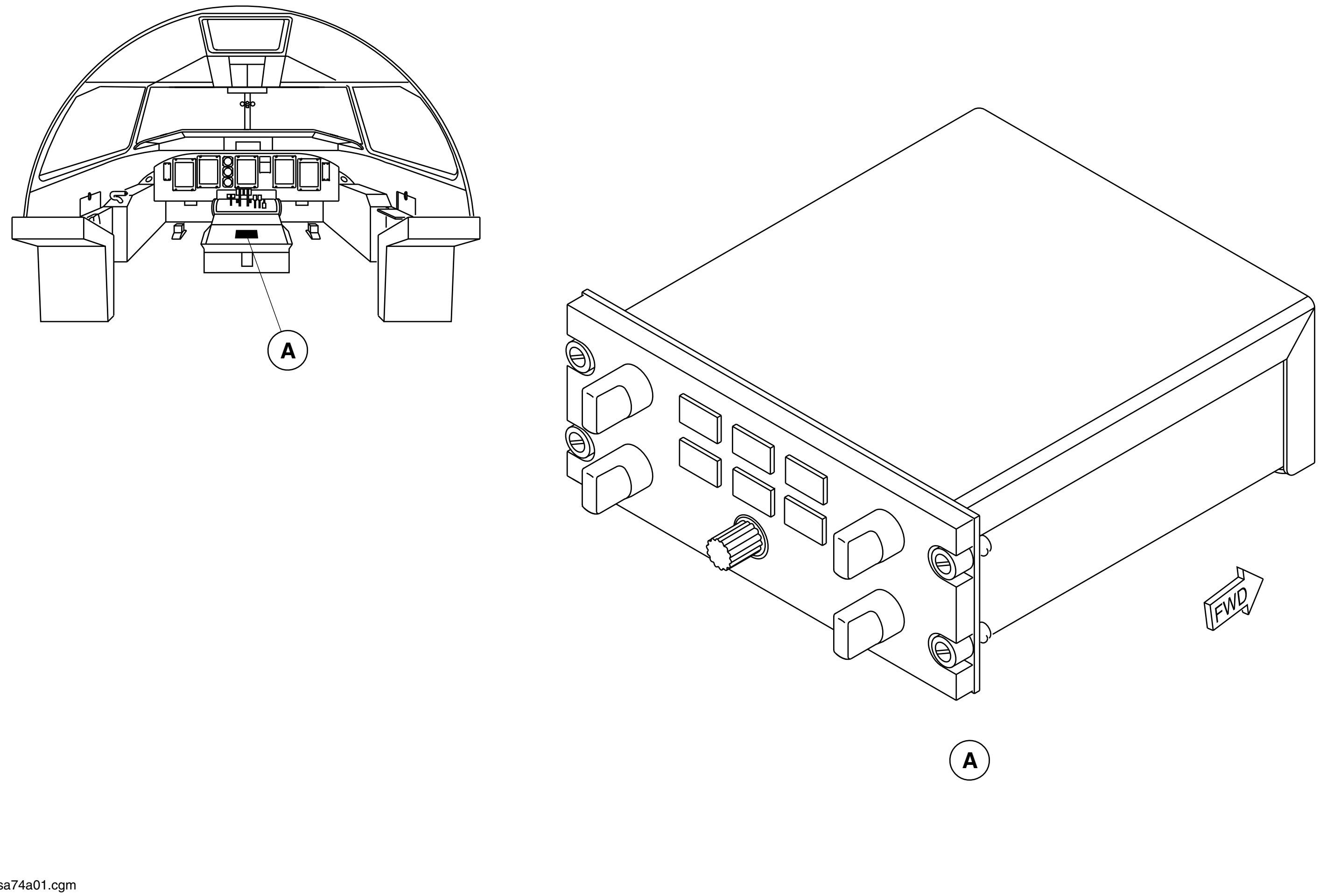
DISPLAY UNIT LOCATOR
Figure 35



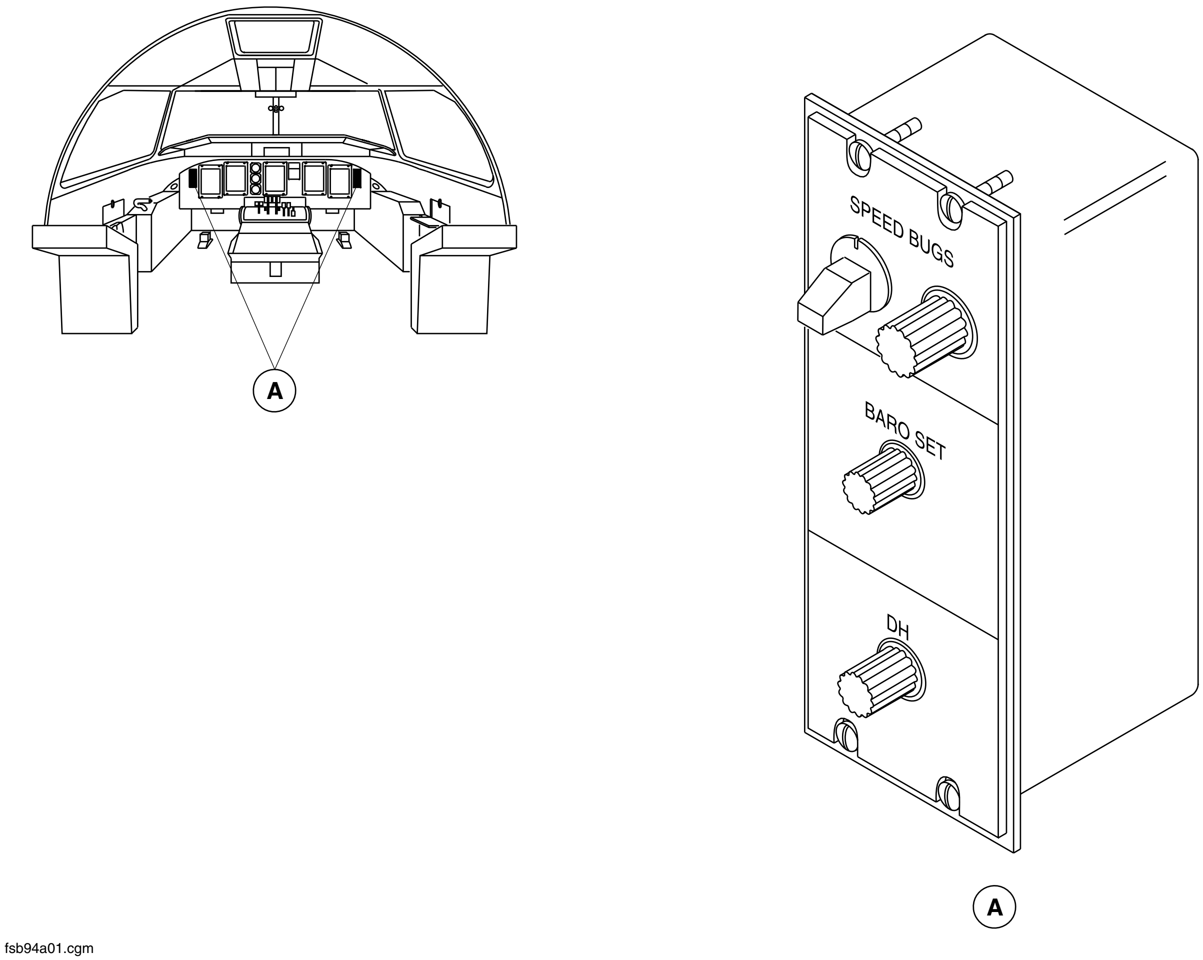
EFIS CONTROL PANEL (EFCP) LOCATOR
Figure 36



CONTROL PANEL BLOCK DIAGRAM
Figure 37



ESID CONTROL PANEL (ESCP) LOCATOR
Figure 38



INDEX CONTROL PANEL (ICP) LOCATOR
Figure 39

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31–61–00–001

ELECTRONIC INSTRUMENTS SYSTEM (ALTERNATE SYMBOLOGY)

Introduction

The Electronic Instrument System (EIS) shows navigation, engine, and system parameters. It interfaces with other systems to calculate, make, and show their images. The Electronic Instrument System (EIS) also monitors its calculations to prevent misleading information from being shown.

General Description

Refer to Figures 40 and 41.

The electronic instruments system has five identical and interchangeable active matrix liquid crystal Display Units (DUs) with the same hardware and software part numbers. Different pin programming at each DU location gives the functions that follow:

– Pilot's Primary Flight Display, PFD1
– Pilot's Multi–Function Display, MFD1
– Engine Display, ED
– Copilot's Primary Flight Display, PFD2
– Copilot's Multi–Function Display, MFD2.

The Electronic Instrument System (EIS) is divided into the two subsystems that follow:

– Electronic Flight Instrument System (EFIS) (31–61–01)
– Engine and System Integrated Display (ESID) (31–61–02).

The Electronic Flight Instrument System (EFIS) is further subdivided into two systems, one for the pilot and one for the copilot. Each EFIS has the components that follow:

– Primary Flight Display (PFD) (31–61–01)
– Multi–Function Display (MFD) (31–61–01)
– Panels, EFIS Control (EFCP1, EFCP2) (31–61–06)
– Panel, ESID Control (ESCP) (31–61–11)
– Index Control Panels (ICP1, ICP2) (31–61–16).

The Engine System Integrated Display (ESID) shares the MFDs with EFIS. It has the components that follow:

– Engine Display (ED) (31–61–02)
– Panel, ESID Control (ESCP) (31–61–11).

Detailed Description

The DUs function in the modes that follow:

– Power–On Self Test (POST)
– Line operational
– Maintenance
– Teleloading.

Power–On Self Test (POST): The Power–On Self Test (POST) mode automatically continues for 2 seconds after power up to ensure correct operation before operational mode is started. The Power−On Self−Test (POST) does the checks that follow:

- Hardware program pin parity
- CPU/ASIC self test
- RAM/EEPROM memory
- Input/Output (I/O) circuits and ports
- Hardware/Consistency.

During the self−test, the output data from DUs cannot be used. They show a green T during the Power−On Self−Test (POST) mode. The POST mode continues for 30 seconds.

During the POST mode, the DU checks the compatibility between the DU part number, hardware and software version. It will not start the operational mode if the checks are not correct.

The Display Units (DU) sense their internal temperature. They automatically turn an internal heater on while the temperature is too low to make sure the correct display brightness. It shows a WARMING UP message in white in the center of the otherwise blank screen.

The normal display replaces the message as soon as the internal temperature reaches the minimum operational requirement.

The WARMING UP message is shown after the Power−On Self−Test (POST) mode.

From −55 to −25 °C (−67 to −13 °F), the display can be of less than the nominal quality but shows the correct data. These images are available in less than 15 minutes after the POWER ON.

Nominal performance is available in less than 20 minutes of POWER ON.

From −25 °C (−13 °F) and above, the display can be of less than the nominal quality but shows the correct data. These images are available in less than 5 minutes after the POWER ON.

Nominal performance is available in less than 15 minutes of POWER ON.

Line Operational: Each DU calculates the position, size and the value of all parameters that is shown.

The DUs makes images similar to existing conventional symbology. These images have line strokes and raster to display critical flight data and guidance commands. Raster is a scan pattern in which an area is scanned from side to side in lines from top to bottom.

A circuit independent of channel that makes the images does a monitoring calculation to make sure that incorrect parameters are not shown. The DUs shows warnings when it senses a malfunction by continuous monitoring.

The indication brightness is automatically and manually controlled.

The DU brightness changes automatically with the ambient lighting conditions in the flight compartment. A light detector located on the front face of each Display Unit (DU) provides an ambient light input for automatic brightness control.

The manual brightness controls enable the pilot and copilot to change the brightness of each DU from a minimum to a maximum level. The brightness level is stays the same after a power interruption.

The manual brightness control can change the automatic control level in normal and failure conditions.

Maintenance Mode: The Built In Test Equipment (BITE) uses the Central Diagnostic System (CDS) to give the condition of the component. It saves faults in a Non Volatile Memory (NVM) for reporting to line and shop maintenance.

The maintenance mode is selected from one or the other Audio and Radio Control Display Units (ARCDUs) during maintenance operations. The DUs do the tests when they receive a maintenance discrete command from IOM1 while the air speed is less than 50 kt and the aircraft is on the ground.

A short or medium power interruption causes the DUs to go off and then start an initialization of the hardware. It does not start the Power–On Self Test (POST) mode or show a malfunction indication. The DUs operates in the line operational mode after the power interruption. (The display flickers).

A long power interruption while the aircraft is on the ground causes a POST that continues for 30 seconds.

A long power interruption while the aircraft is airborne starts a short hardware and RAM initialization that continues for 2 seconds.

When a DU senses an internal malfunction, it will stay in the POST mode for the duration of the malfunction.

The BITE allows aircraft maintenance personnel to:

- Do fault isolation and return to service testing after completing maintenance actions
- Access failure reports from last or previous flight legs
- Get the avionics status report
- Get the part number of an LRU.

The BITE mode monitors the condition of the component as follows:

- Power–On Self Test (POST)
- Continuous monitoring.

Power–On Self Test (POST): The POST checks the condition of the component at power–up or after a long power interruption more than 200 milliseconds.

Refer to Figure 42.

Continuous Monitoring: The continuous monitoring checks the status of the components in flight. It records faults in a Non Volatile Memory (NVM) for later troubleshooting using the Central Diagnostic System (CDS).

The DUs monitor the critical indication parameters that follow:

- Pitch
- Roll
- Altitude
- Airspeed
- Heading
- Torque
- ITT
- NH
- NP.

A mismatch condition between inputs from critical parameters is monitored by the Flight Data Processing System (FDPS). When a difference is sensed by the FDPS, it causes the PFDs to show a mismatch message.

Display monitoring makes sure that the symbols are shown correctly. A feedback bus is used to compare the shown indication with the direct input.

The monitoring is done by one or two DUs as follows:

– The PFD is monitored by its adjacent MFD (if not set to PFD) and opposite PFD. If the opposite PFD is de–energized, it uses the opposite MFD.

– The ED is monitored by its adjacent MFDs. If one MFD is set to PFD, it uses the other MFD.

The threshold for PFD monitoring is shown in the table that follows:

Table 1: Table 1.

PARAMETER	THRESHOLD
Roll	4 degrees
Pitch	4 degrees
Heading	6 degrees
Indicated Air Speed	10 knots
Corrected Barometric Altitude	100 feet

The threshold for ED monitoring is shown in the table that follows:

Table 2: Table 2.

PARAMETER	THRESHOLD
Torque	5 percent
NH	5 percent
NP	50 rpm
ITT	50 °C

When a difference between the shown indication and the direct input is more than the threshold, the EIS shows the messages that follow:

– CHECK PFD on PFDs and MFDs

– CHECK ED on ED and MFDs.

An incorrect response from a switch selection is immediately sensed by the pilot but a malfunction causes set parameters to change to dashes.

Teleloading: The operational software serial data is supplied to the DUs.

NOTE

Note: The DU must be removed from the aircraft to receive a teleload.

The DUs do a software configuration check before the teleloading mode starts. It makes sure that the software version to be teleloaded is the same as the equipment part number.

The teleload time is less than 5 minutes for each DU. The DUs show a green L during the Teleloading mode.

The Display Unit (DU) restarts after a complete and successful teleload. It initiates a complete Power–On Self Test (POST) within 10 seconds.

The DU starts a POST less than 10 seconds after the teleload mode is complete.

There are programming pins located on the DU connectors to supply data to the DUs that follows:

- Aircraft type
- Location of the DU on the aircraft
- Selection of the DU
- Shop maintenance
- Teleloading.

Refer to Figure 43.

When the pilot is flying the aircraft, the MFD1 reversion selector is set to the NAV position and the MFD2 reversion selector is set to the SYS position on the ESCP.

Refer to Figure 44.

The DUs show the Electronic Instruments System (EIS) data that follows:

- Primary Flight Display (PFD1)
- Primary Flight Display (PFD2)
- Engine Display (ED).
- Multi–Function Displays (MFD1)
- Multi–Function Displays (MFD2).

Refer to Figure 45.

Primary Flight Displays (PFD1, PFD2): The Primary Flight Displays (PFD) are the outer displays of the Electronic Instrument System (EIS). Each display shows the primary navigation information that follows:

- Flight Mode Annunciator (FMA)
- Indicated Air Speed (IAS)
- Altimeter (ALT)
- Electronic Attitude Direction Indicator (EADI)
- Electronic Horizontal Situation Indicator (EHSI)
- Inertial Vertical Speed Indicator (IVSI)
- Traffic Collision Avoidance System II (TCAS II).

The MFDs are located inboard of the PFDs. Their indication is divided into parts when they are not showing a Primary Flight Display (PFD) or Engine Display (ED) reversionary indication.

Refer to Figure 46.

The MFD1 is usually set to show the Electronic Flight Instrument System (EFIS) navigation page, map format at the top part of the indication and the Engine and System Integrated Display (ESID) permanent system data area at the bottom. Its permanent system data area usually shows the Powered Flight Control Surface Position (PFCS) parameters.

Refer to Figure 47.

The navigation page shows also shows the arc format.

Refer to Figure 48.

The navigation page shows also shows the plan format.

Refer to Figure 49.

The MFD2 is usually set to show the Engine and System Integrated Display (ESID) electrical system page at the top part of the indication and permanent system data area at the bottom. Its permanent system data area usually shows the parameters that follow:

- Flap position
- Hydraulic pressure
- Hydraulic quantity.

Refer to Figures 50 , 51 and 52.

The Engine and System Integrated Display (ESID) system shows the additional system pages that follows:

- Engine, or
- Fuel, or
- Doors.

Refer to Figure 53.

The Engine Display (ED) is located at the center of the instrument panel. It shows primary engine and aircraft systems data that follows:

- Torque
- NH High Pressure Compressor Rotational Speed
- NP Propeller Rotational Speed
- ITT Inter–Turbine Temperature
- Fuel Flow
- NL Low Pressure Compressor Rotational Speed
- Engine rating mode
- Oil Pressure
- Oil Temperature
- Tank Fuel Quantity
- Fuel inlet temperature
- Advisory messages
- Static Air Temperature.
- System Page (Mono Mode).

Three different types of advisory messages are shown in white letters at the bottom part of the Engine Display (ED). An IFC, powerplant, or display related message is shown at a different location while the condition is present.

The messages are shown in order of decreasing importance. The most important display message is shown at the bottom right part of the ED. The DUs cause the ED to show the advisory messages that follow:

- DU BAD CONF
- ED MON FAIL
- PFD1,PFD2 MON FAIL

- HOT PFD1, MFD1, ED, MFD2 or PFD2
- HOT DISPLAYS
- PFD1, MFD1, MFD2 or PFD2 LINK FAIL.

DU BAD CONF Message: The DU BAD CONF message comes into view when a bad or incorrect aircraft configuration is sensed by a display unit.

NOTE

Note: The DU BAD CONF message comes into view after a POST while the aircraft is on the ground.

ED MON FAIL Message: The ED MON FAIL message comes into view when the ED critical parameters are not monitored another display.

PFD1, PFD2 MON FAIL Message: The PFD1 or PFD2 MON FAIL message comes into view when the PFD critical parameters are not monitored another display unit.

HOT PFD1, MFD1, ED, MFD2 or PFD2 Message: The HOT display unit message comes into view when one display unit senses an overheat condition.

NOTE

Note: The related display unit is automatically dimmed to decrease its temperature.

HOT DISPLAYS Message: The HOT DISPLAYS message comes into view when two or more display units sense an overheat condition.

PFD1, MFD1, MFD2 or PFD2 LINK FAIL Message: The PFD1, MFD1, MFD2 or PFD2 LINK FAIL message comes into view to tell the pilots that the ED will not show an advisory message from the related display unit.

The Electronic Instruments System (EIS) has the control panels that follow to control the DU indications:

- EFIS Control Panels (EFCP1)
- EFIS Control Panels (EFCP2)
- ESID Control Panel (ESCP).
- Index Control Panel (ICP1)
- Index Control Panel (ICP2).

Refer to Figure 54.

EFIS Control Panels (EFCP1, EFCP2): On aircraft with the MFD navigation source selection independent of the PFD navigation source selection (823CH00015): The EFCPs have the controls that follow:

- Bearing 1 selector knob
- Bearing 2 selector knob
- TCAS pushbutton switch
- WX/TERR pushbutton switch
- DATA pushbutton switch
- FORMAT pushbutton switch
- RANGE selector knob
- PFD OFF, BRT knob
- MFD OFF, BRT knob
- WX/TERR BRT knob.

Bearing 1, 2 Selector Knob: The bearing 1 or 2 selector knob is turned to make an AUX, VOR, ADF, or FMS bearing data selection for indication on the EFIS PFD.

TCAS pushbutton Switch: The TCAS pushbutton switch is pushed to change the TCAS indication from automatic (pop−up) to on. It is pushed again to show the automatic indication. The automatic selection, only shows Traffic Advisory (TA) and Resolution Advisory (RA) indications. The on selection shows all TCAS traffic indications.

NOTE

Note: A TCAS pushbutton switch selection has no effect when the navigation page is not in view.

WX/TERR Pushbutton Switch: The WX/TERR pushbutton switch is pushed to change the weather radar indication from on to off. It is pushed again to show the EGPWS terrain indication. A third WX/TERR pushbutton switch selection causes the MFD to show the weather radar indication again.

If the navigation page is set to show a weather radar indication (ARC or MAP format) and an EGPWS terrain alert is sensed, the terrain indication automatically comes into view. If the navigation page is set to the PLAN format, an EGPWS terrain alert causes it to change the ARC format. An EGPWS terrain alert always causes the weather radar range indication to change to 10 NM.

NOTE

Note: A WX/TERR pushbutton switch selection has no effect when the navigation page is not in view.

DATA Pushbutton Switch: The DATA pushbutton switch is pushed to show FMS navigation aids and airports when the navigation page is set to MAP or PLAN format. The first selection causes the navigation page to show the nearest navigation aids. It is pushed again to shown the nearest airports. The third selection causes the navigation page to show the nearest navigation aids and airports. The fourth selection causes the data indications to go out of view.

NOTE

Note: A DATA pushbutton switch selection has no effect when the navigation page is set to show an ARC VOR/ILS or ARC MLS format.

FORMAT Pushbutton Switch: The FORMAT pushbutton switch is pushed to change to the navigation page indication from a map (FMS) to arc (VOR/ILS) indication. It is pushed again to show the map indication.

If the FORMAT pushbutton switch is pushed for more than one second, the plan format is shown. It is pushed again to show the map indication.

RANGE selector knob: The RANGE pushbutton switch is pushed to change the weather radar range indications on the navigation page.

PFD OFF, BRT Knob: The PFD OFF, BRT Knob is turned to energize or de−energize the PFD and to control its brightness.

MFD OFF, BRT Knob: The MFD OFF, BRT Knob is turned to energize or de−energize the MFD and to control its brightness.

WX/TERR BRT knob: The WX/TERR BRT knob is turned to adjust the weather radar image brightness.

Refer to Figure 55.

ESID Control Panels (ESCP): To control the Engine and System Integrated Display ESID) indications and some EFIS indications, the ESCP has the controls that follow:

- MFD1 and MFD2 Reversion Switches

- ELEC SYS Pushbutton Switch
- ENG SYS Pushbutton Switch
- FUEL SYS Pushbutton Switch
- DOORS SYS Pushbutton Switch
- ALL Pushbutton Switch
- EFIS ATT/HDG SOURCE reversion selector
- EFIS ADC SOURCE Reversion Selector
- ED OFF, BRT Knob.

MFD1 and MFD2 Reversion Switches: The MFD1 and MFD2 reversion switches are used to make indication selections on the ESID DUs.

Refer to Figures 56 and 57.

ELEC SYS Pushbutton Switch: The ELEC SYS pushbutton switch is pushed to show the electrical system page on the ESID DUs.

Refer to Figures 58 and 59.

ENG SYS Pushbutton Switch: The ENG SYS pushbutton switch is pushed to show the engine system page on the ESID DUs.

Refer to Figures 60 and 61.

FUEL SYS Pushbutton Switch:The FUEL SYS pushbutton switch is pushed to show the fuel system page on the ESID DUs.

Refer to Figures 62 and 63.

DOORS SYS Pushbutton Switch: The DOORS SYS pushbutton switch is pushed to show the doors system page on the ESID DUs.

ALL Pushbutton Switch: The ALL SYS pushbutton switch is pushed to sequence through all the system pages on the ESID DUs.

EFIS ATT/HDG SOURCE Reversion Selector: The EFIS ATT/HDG SOURCE reversion selector is used to make an Attitude and Heading Reference Unit (AHRU 1, AHRU2) indication selection on the EFIS DUs.

EFIS ADC SOURCE Reversion Selector: The EFIS ADC SOURCE reversion selector is used to make Attitude and Heading Reference Units, ADU 1 or ADU 2 indication selections on the EFIS DUs.

ED OFF, BRT Knob: The ED OFF, BRT Knob is turned to energize or de–energize the ED and to control its brightness.

Refer to Figure 64.

The MFD1 permanent system data area shows the Powered Flight Control Surface Position (PFCS) parameters and the MFD 2 permanent system data area shows flap position, hydraulic pressure, and hydraulic quantity indications.

Refer to Figures 65 and 66.

When a MFD malfunctions, its permanent system data moves to the other MFD and the permanent system data indication changes to a composite indication to show the two permanent system data together.

When a PFD or ED malfunctions, the MFD1 or MFD2 reversion switch located on the ESCP is used to manually move the indications from the defective DU to a MFD. As part of this reversionary mode, the permanent system data moves to the other MFD and changes to a composite indication to show the two MFD1 and MFD2 permanent system data together.

Refer to Figure 67.

There is one automatic DU reversionary mode. When the Engine Display (ED) malfunctions while airborne, the ED images automatically move to the MFD1 and the MFD2 shows a composite permanent system data indication. This automatic reversionary mode occurs only if the MFD1 is not set to show the PFD images.

Refer to Figure 68.

After the automatic reversion, the MFD1 reversion switch is manually set to the ENG position to make the DU indications agree with the switch selection.

Refer to Figure 69.

If the Engine Display (ED) and Primary Flight Display (PFD1) malfunction at the same time while the TAS is more than 50 kts and the MFD 1 reversion selector is not set to PFD, the ED images automatically move to the MFD.

The MFD2 shows a composite permanent system data indication.

Refer to Figure 70.

The MFD1 reversion switch is set to PFD and the ENG SYS pushbutton switch is then pushed.

Refer to Figure 71.

The PFD indications are manually moved from the defective PFD1 to its related MFD and the other MFD will show the engine system page with a composite permanent system data indication.

Refer to Figure 72.

If the MFD1and MFD2 malfunction at the same time, the navigation page, system page, and the permanent system data indication is not in view.

Refer to Figure 73.

The Engine and System Integrated Display (ESID) system has a mono mode to let the pilots view the system page with a composite permanent system data indication when the two MFDs malfunction or are set to show PFD, ED, or navigation page indications.

A system pushbutton switch is pushed to show its applicable system page with a composite permanent system data indication on the Engine Display (ED).

Refer to Figure 74.

The system page is shown while the pushbutton switch is pushed. The ED images are shown again when the push button switch is released.

Refer to Figure 75.

Index Control Panels (ICP1, ICP2): On aircraft with the alternate ICPs (823CH00013): The ICPs have the controls that follow:

- SPEED BUG SEL pushbutton switch
- SPEED BUG rotary selector
- BARO SET knob/PUSH TO STD switch
- DH/MDA rotary switch
- DH/MDA knob.

SPEED BUG SEL Pushbutton Switch: The SPEED BUG SEL pushbutton switch located on the ICP is pushed to set the speed bugs that follow:

- V1, take off decision speed
- VR, rotation speed
- V2, take off safety speed
- No. 1 (Solid)
- No. 2 (Outline).

The SPEED BUG SEL pushbutton switch is pushed to set speed bugs V1, VR, and V2 when the aircraft is on the ground and speed bugs No. 1 and No. 2 when the aircraft is airborne.

The SPEED BUG SEL pushbutton switch is pushed to change speed bug V1 or No. 1. It is pushed again to set V1 or No. 1 and to change VR or No. 2. The SPEED BUG SEL pushbutton switch is pushed again to set V2. Each speed bug is also set five seconds after the last SPEED BUG SEL pushbutton switch or SPEED BUG rotary selection.

SPEED BUG rotary selector: The SPEED BUG rotary selector is turned to change the speed bug value.

BARO SET Knob/PUSH TO STD Switch: The knob is turned to supply the related Air Data Unit (ADU1, ADU2) with a barometric correction value and it is pushed to supply a standard barometric correction value.

DH/MDA rotary switch: The DH/MDA rotary switch is set to DH to let the DH/MDA knob change the DH value. It is set to MDA to let the DH/MDA knob change the MDA value.

DH/MDA knob: The DH/MDA knob is turned to set a DH altitude for Decision Height (DH) calculations or MDA for Minimum Descent Altitude calculations.

The Electronic Instrument System (EIS) units are energized by the main 28 V dc generation system. The electrical power source for each unit is shown in the table that follows:

Table 3: Table 3.

PART	LEFT MAIN	LEFT ESS	RIGHT MAIN	RIGHT ESS
PFD1	X			
MFD1		X		
ED			X	X
MFD2			X	
PFD2			X	
EFCP1	X			
ICP1		X		
ESCP	X			
EFCP 2			X	
ICP2			X	

Each Display Unit (DU) dissipates 170 W of power when the display heater is on and the brightness is set to a maximum level. It is 110 W when the heater turns off. The DUs dissipates 90 W of power when its heater is off and the brightness is set to a maximum level with low ambient flight compartment illumination.

The left main bus supplies electrical power through a 10 A circuit breaker and two 5 A circuit breakers to the PFD, EFCP, and ESCP. The circuit breakers are located in position A2, C2, and B2 on the avionics circuit breaker panel.

The left essential bus supplies electrical power through a 10 A circuit breaker and a 5 A circuit breaker to the MFD1 and ICP1. The circuit breakers are located in position B7 and C7 on the avionics circuit breaker panel.

The right main bus supplies electrical power through three 10 A circuit breakers and two 5 A circuit breakers to the ED, MFD2, PFD2, EFCP2, and ICP2.The circuit breakers are located in position A5, B5, B6, C5, and C6 on the avionics circuit breaker panel.

The right essential bus also supplies electrical power through a 10A circuit breaker to the ED. The circuit breaker is located in position B11 on the avionics circuit breaker panel.

Display Units (DU)

Refer to Figure 76.

Each display unit has a 6 by 8 in. (152 by 203 mm) active matrix liquid crystal display with a resolution of 167 Dots Per Inch (DPI). It uses the color rules shown in the table that follows:

Table 4: Table 4.

COLOUR	INDICATION
Red	Warning
Yellow	Caution
White	Advisory, labels, scales

COLOUR	INDICATION
Green	Active modes, passed test
Cyan	Selectable parameters
Magenta	FMS
Brown	Attitude sphere
Blue	Attitude sphere, ESID units
Grey	Indicator background
Black	Aircraft symbol

The DU shows messages with the attributes that follows:

- Flashing
- Reverse video
- Brackets.

Flashing: Flashing is used when a new message that requires pilot attention comes into view. It is time limited to 5 seconds or stays in view until the pilot action is completed. The flashing frequency is 1 Hz.

Reverse Video: Reverse video is used to show a change in the operating mode that was not pilot set. It is time limited to 5 seconds.

Brackets: The message in the brackets are pilot instructions.

The dynamic data of the images are received and updated on the screen at a refresh frequency of 20 Hz. The stroke and raster images are updated at a nominal 60 Hz rate.

The DU has the sub–assemblies that follow:

- Interface Connection Module (ICM)
- Digital Processing Module (DPM)
- Backlighting Module (BLM)
- LCD Assembly Module (LAM)
- Power Supply Module (PSM)
- Housing Assembly Module (HAM).

The display units weigh 12.32 lb (5.59 kg). They are 7.56 in. (195 mm) wide, 10.486 in. (266.3 mm) high and 7.46 in. (190 mm) long.

The DUs are attached to the instrument panel with four mounting screws. They are electrical bonded to the instrument panel through wires in the rear connector and by surface contact between the DU chassis and the instrument panel mounting frame.

EFIS Control Panels (EFCP)

Refer to Figure 77.

To control some related EFIS PFD and navigation page indications, the EFCPs have the controls that follow:

- Bearing 1 selector knob
- Bearing 2 selector knob
- TCAS pushbutton switch
- WX/TERR pushbutton switch

- DATA pushbutton switch
- FORMAT pushbutton switch
- RANGE selector knob
- PFD OFF, BRT knob
- MFD OFF, BRT knob
- WX/TERR BRT knob.

Refer to Figure 78.

The EFCP has the sub–assemblies that follow:

- Front face module
- Interface board
- Outer box.

The Interface board has the functions that follow:

- Power supply
- Parallel/serial converter
- ARINC emitter/receiver
- EMI, lightning protection.

The EFIS selection and the type of interface signal is shown in the table that follows:

Table 5: Table 5.

SELECTION	SIGNAL TYPE
Bearing 1 selector knob	ARINC 429
Bearing 2 selector knob	ARINC 429
FULL/ARC mode pushbutton switch	ARINC 429
WX mode pushbutton switch	ARINC 429
FMS MAP mode pushbutton switch	ARINC 429
TCAS pushbutton switch	ARINC 429
RANGE selector knob	ARINC 429
PFD OFF, BRT knob	Discrete, ARINC 429
MFD OFF, BRT knob	Discrete, ARINC 429
WX/TERR BRT knob	ARINC 429

NOTE

Note: The EFCP PFD and MFD OFF, BRT knobs supply a discrete signalto the DUs for ON/OFF control and ARINC 429 data for brightness control.

The EFCPs weigh 2.2 lb (1.0 kg). The EFCP is 2.625 in. (66.68 mm) wide, 5.75 in. (146 mm) high and 5.9 in. (150 mm) deep. The EFCPs are attached tothe centre console with four DZUS fasteners. The fasteners and a bonding wiredirectly connect to the chassis to make a ground continuity connection between control panel chassis and the aircraft structure.

ESID Control Panel (ESCP)

Refer to Figure 79.

To control some related EFIS PFD and navigation page indications, the ESCP has the controls that follow:

- MFD1 and MFD2 Reversion Switches
- ELEC SYS Pushbutton Switch
- ENG SYS Pushbutton Switch
- FUEL SYS Pushbutton Switch
- DOORS SYS Pushbutton Switch
- ALL Pushbutton Switch
- EFIS ATT/HDG SOURCE reversion selector
- EFIS ADC SOURCE Reversion Selector
- ED OFF, BRT Knob.

The ESCP has the sub–assemblies that follow:

- Front face module
- Interface board
- Outer box.

The Interface board has the functions that follow:

- Power supply
- Parallel/serial converter
- ARINC emitter/receiver
- EMI, lightning protection.

The ESID selection and the type of interface signal is shown in the table that follows:

Table 6: Table 6.

SELECTION	SIGNAL TYPE
MFD1 and MFD2 Reversion Switches	ARINC 429
ELEC SYS Pushbutton Switch	ARINC 429
ENG SYS Pushbutton Switch	ARINC 429
FUEL SYS Pushbutton Switch	ARINC 429
DOORS SYS Pushbutton Switch	ARINC 429
ALL Pushbutton Switch	Discrete
EFIS ATT/HDG SOURCE reversion selector	Discrete

SELECTION	SIGNAL TYPE
EFIS ADC SOURCE Reversion Selector	Discrete
ED OFF, BRT Knob	Discrete, ARINC 429

NOTE

Note: ED OFF, BRT knob supplies a discrete signal to the DU for ON/OFF control and ARINC 429 data for brightness control.

The ESCP weighs 2.2 lb (1.0 kg). The ESCP is 2.625 in. (66.68 mm) wide, 5.75 in. (146 mm) high and 5.9 in. (150 mm) deep. The ESIS Control Panel (ESCP) is attached to the centre console with four DZUS fasteners. The fasteners and a bonding wire directly connect to the chassis to make a ground continuity connection between control panel chassis and the aircraft structure.

Index Control Panels (ICP)

Refer to Figure 80.

To control some related EFIS PFD indications, the ICPs have the controls that follow:

- SPEED BUG index switch
- SPEED BUG rotary selector
- BARO SET knob
- DH knob.

The ICP has the sub–assemblies that follow:

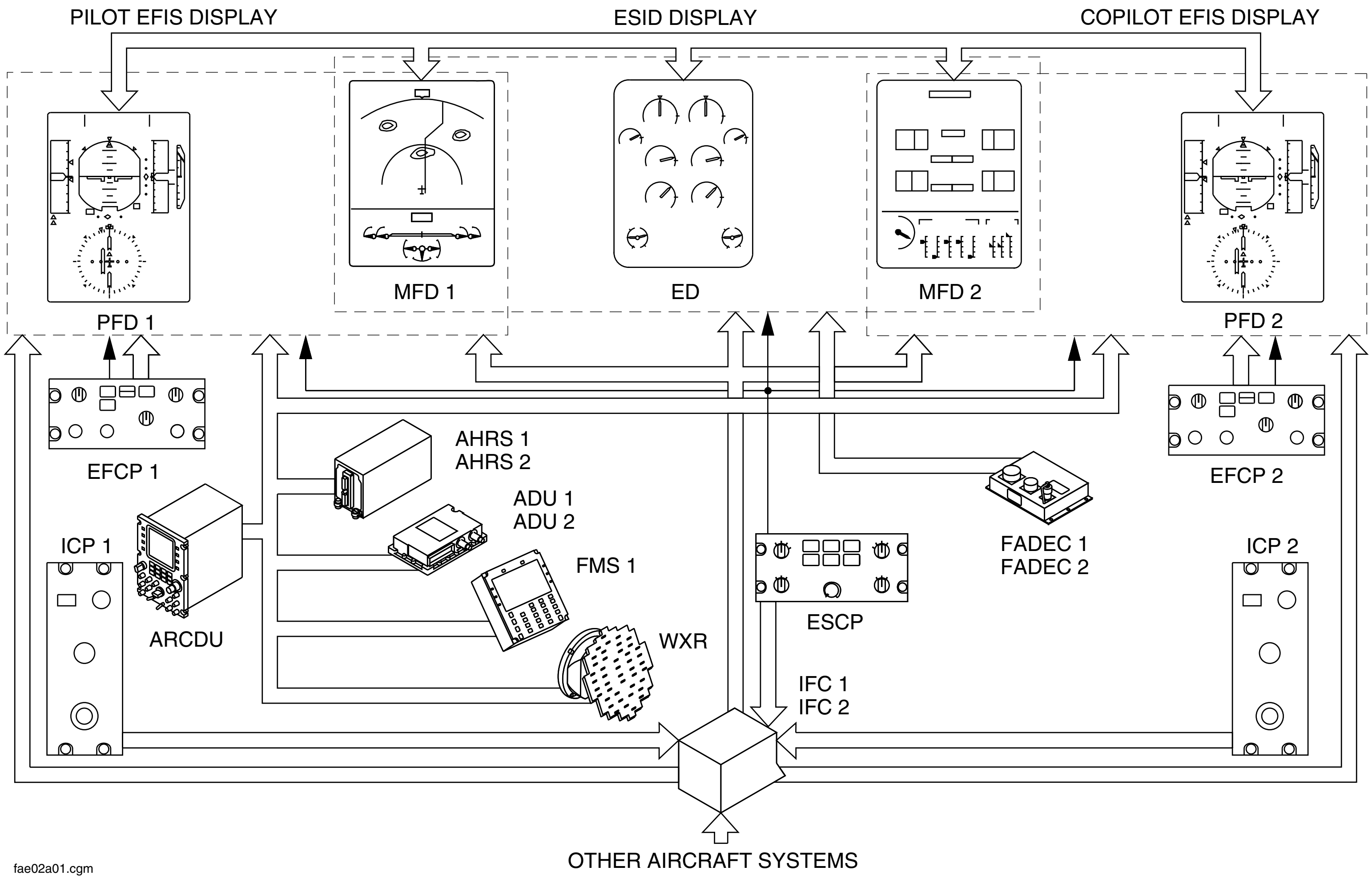
- Front face module
- Interface board
- Outer box.

The Interface board has the functions that follow:

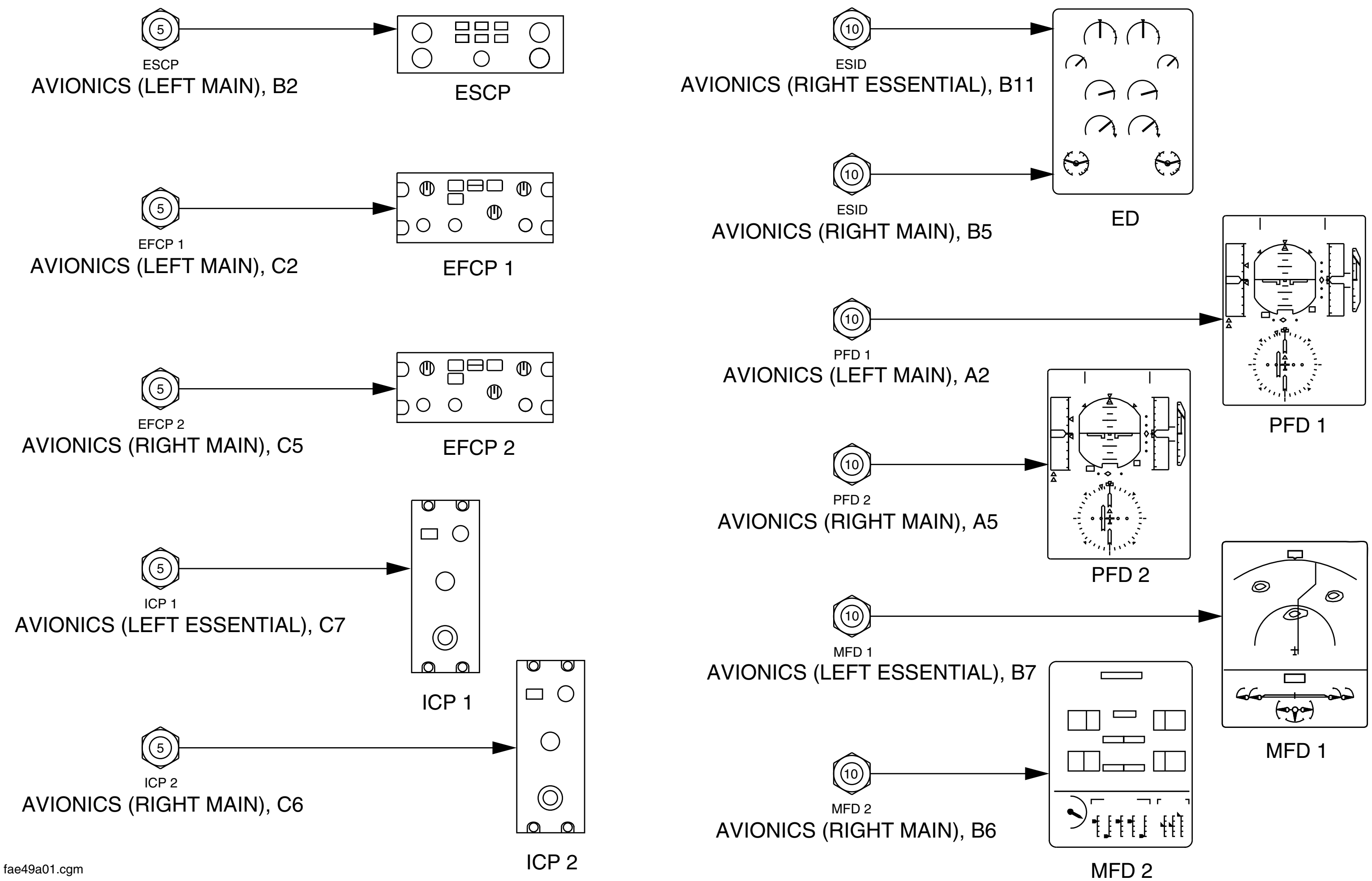
- Power supply
- Parallel/serial converter
- ARINC emitter/receiver
- EMI, lightning protection.

Each ICP supplies data to its related IOP and ADU through ARINC 429 data buses.

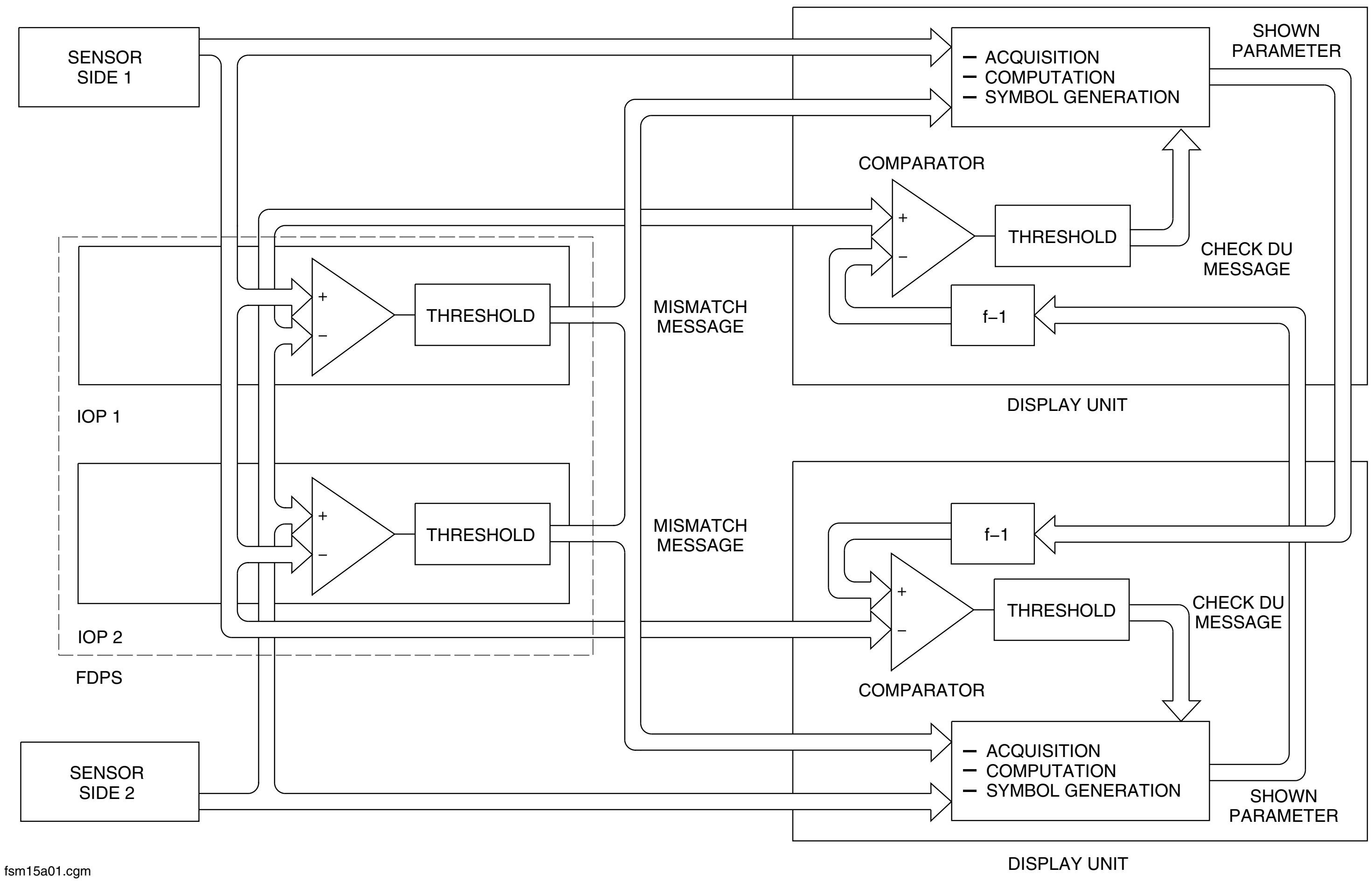
The ICP weighs 2 lb (0.91 kg). It is 2.25 in. (57.2 mm) wide, 5.75 in. (146 mm) high and 5.5 in. (140 mm) deep. The ICPs are attached to the instrument panel with four mounting screws. The fasteners and a bonding wire directly connect to the chassis to make a ground continuity connection between control panel chassis and the aircraft structure.



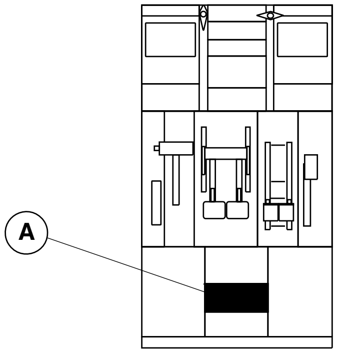
ELECTRONIC INSTRUMENT SYSTEM BLOCK DIAGRAM
Figure 40



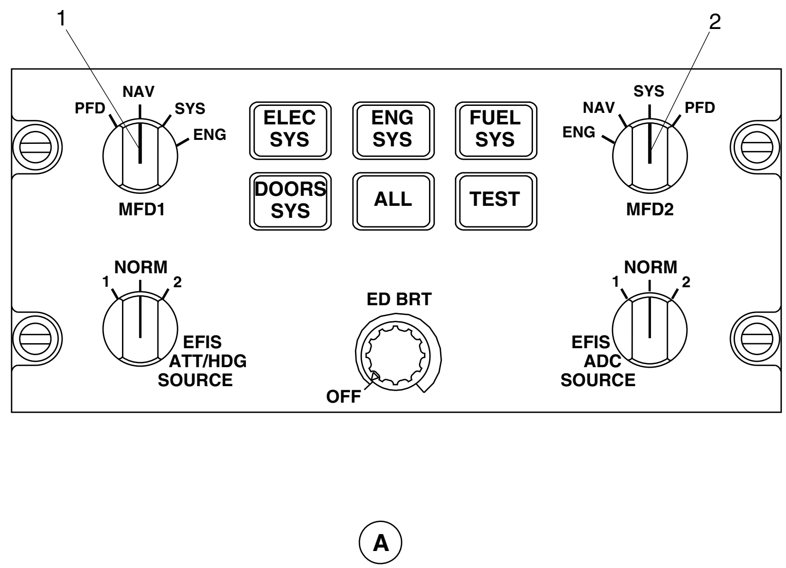
ELECTRONIC INSTRUMENT SYSTEM BLOCK DIAGRAM POWER
Figure 41



ELECTRONIC INSTRUMENT SYSTEM DISPLAY MONITORING
Figure 42



CENTRE CONSOLE



LEGEND

1. MFD 1 Reversion Selector.
2. MFD 2 Reversion Selector.

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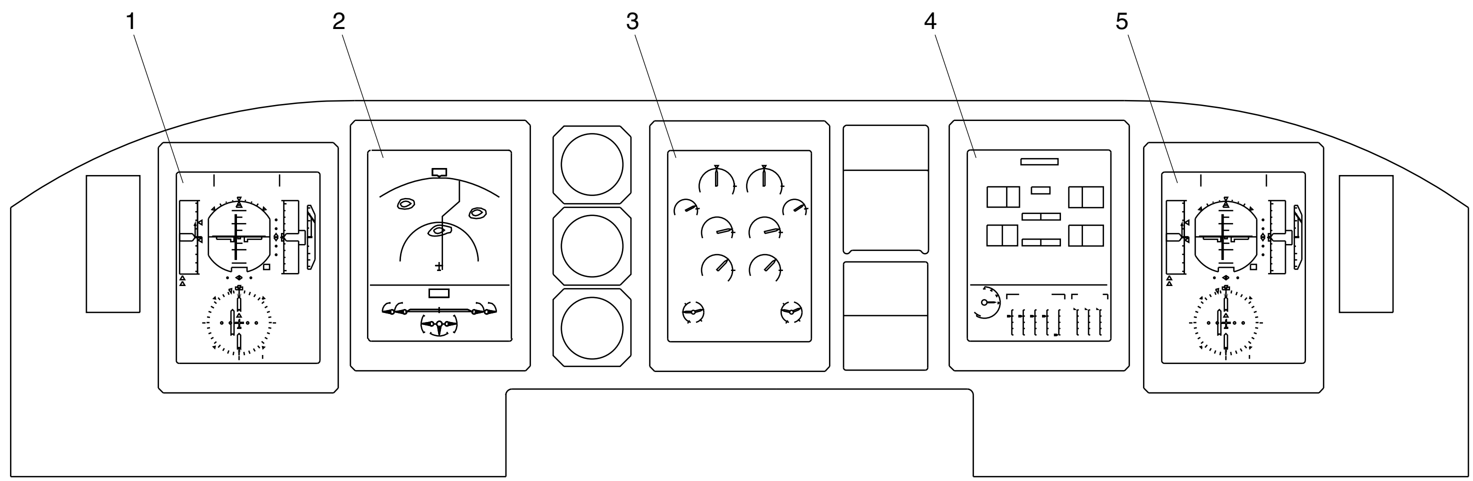
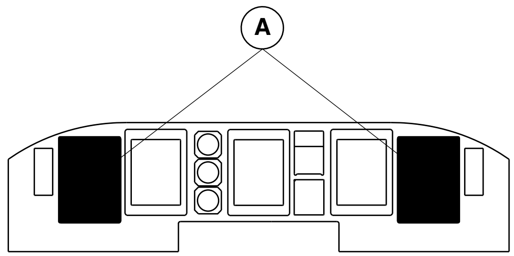
ELECTRONIC INSTRUMENT SYSTEM INDICATION SELECTION
Figure 43

LEGEND

1. PFD1.
2. MFD1.
3. ED.
4. MFD2.
5. PFD2.



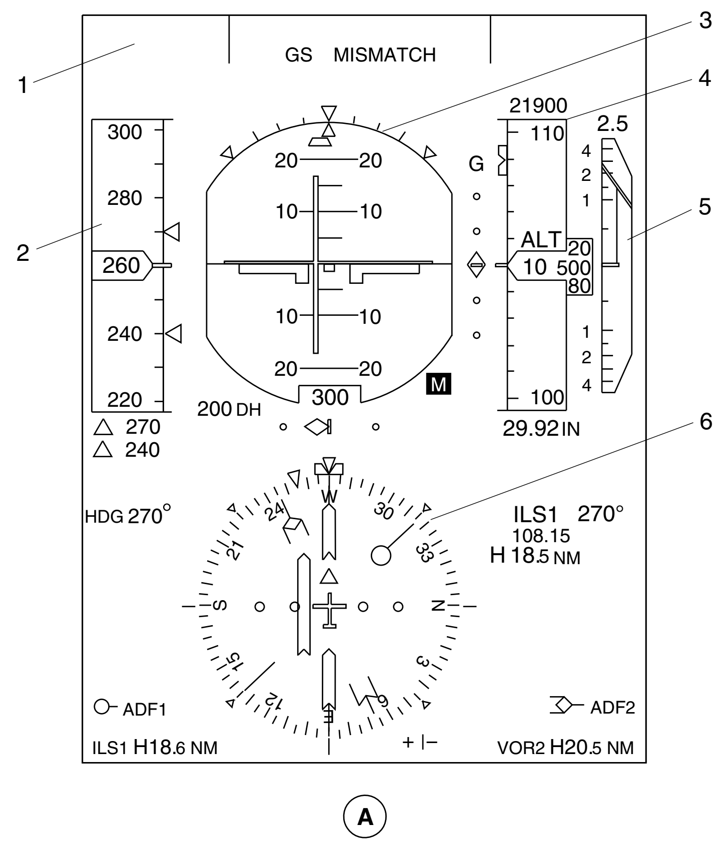
ELECTRONIC INSTRUMENT SYSTEM INDICATIONS PAGE 1
Figure 44



MAIN INSTRUMENT PANEL

LEGEND

1. Flight Mode Annunciator (FMA).
2. Air Speed Indicator (IAS).
3. Electronic Attitude Direction Indicator (EADI).
4. Altimeter (ALT).
5. Inertial Vertical Speed Indicator (IVSI), TCAS.
6. Electronic Horizontal Situation Indicator (EHSI).



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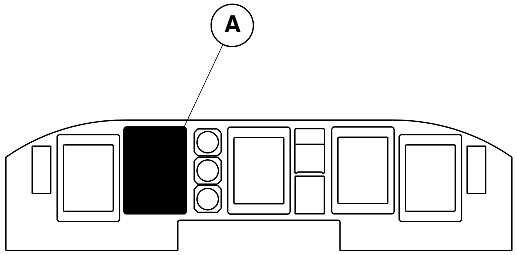
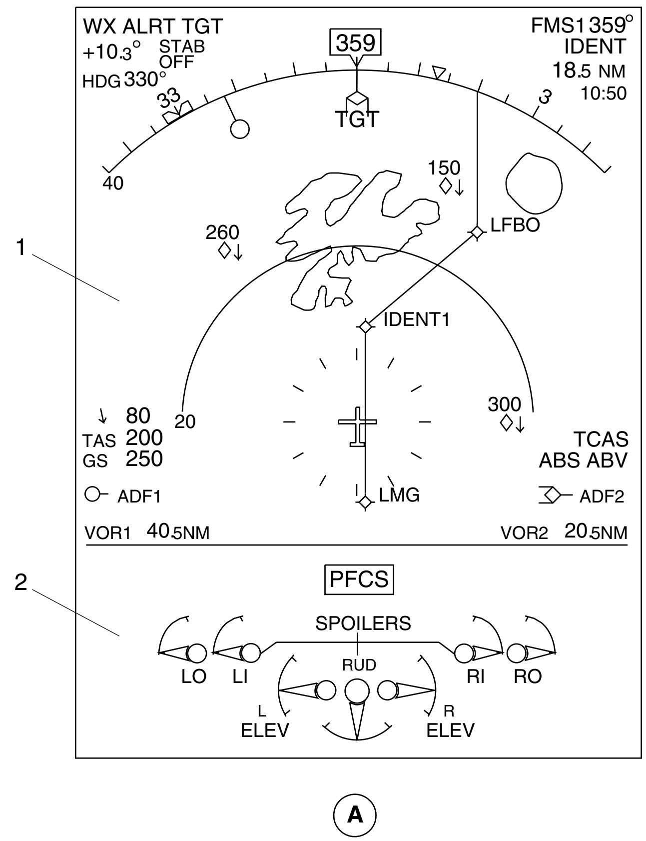
ELECTRONIC INSTRUMENT SYSTEM PFD INDICATIONS
Figure 45



MAIN INSTRUMENT PANEL



LEGEND

1. System Page or Navigation Page.
2. Permanent System Data Area.

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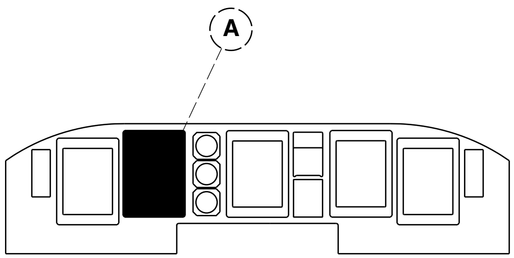
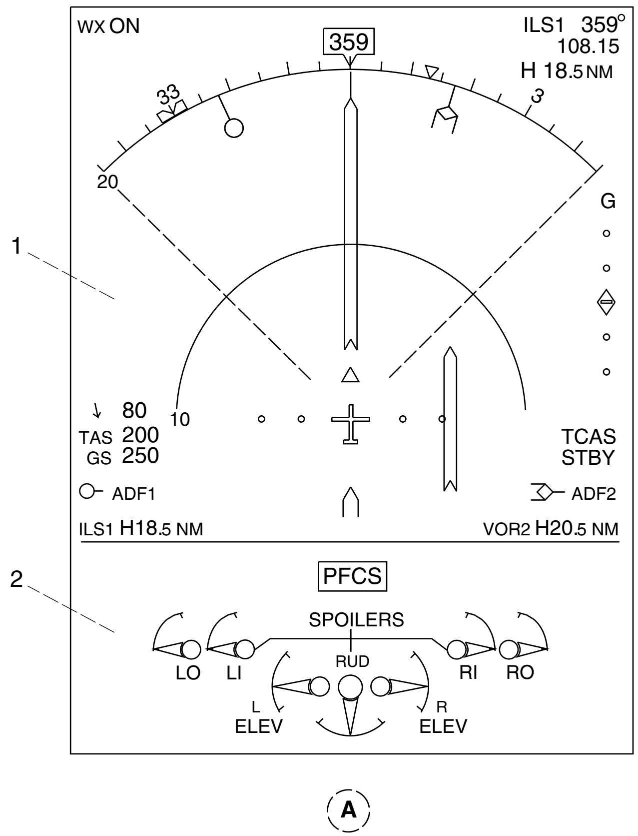
ELECTRONIC INSTRUMENT SYSTEM MFD1 INDICATION, MAP FORMAT
Figure 46



MAIN INSTRUMENT PANEL



LEGEND

1. System Page or Navigation Page.
2. Permanent System Data Area.

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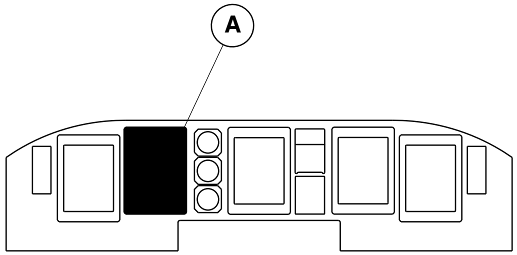
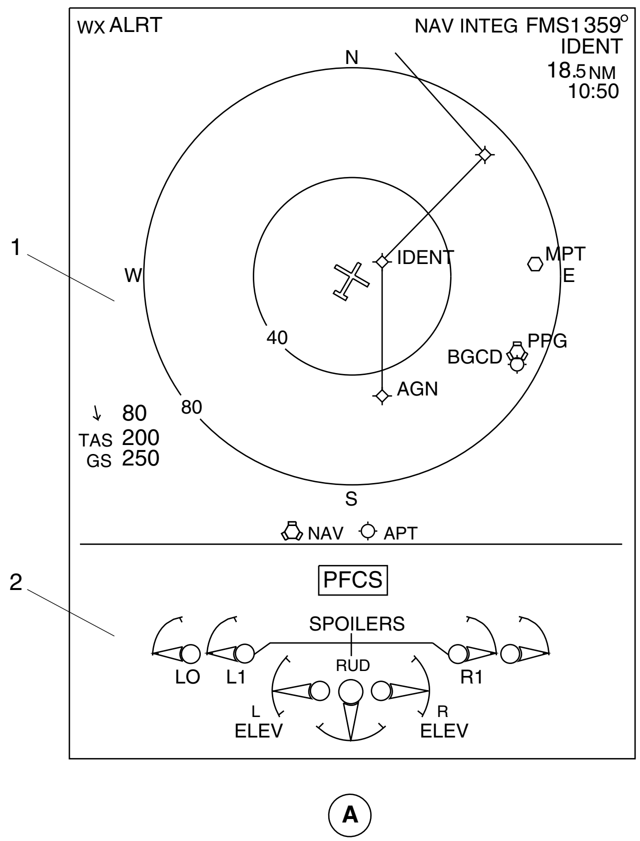
ELECTRONIC INSTRUMENT SYSTEM MFD1 INDICATION, ARC FORMAT
Figure 47



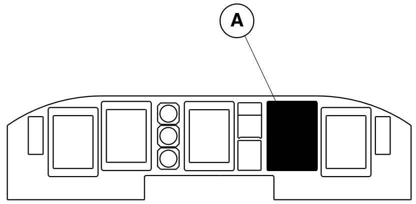


LEGEND

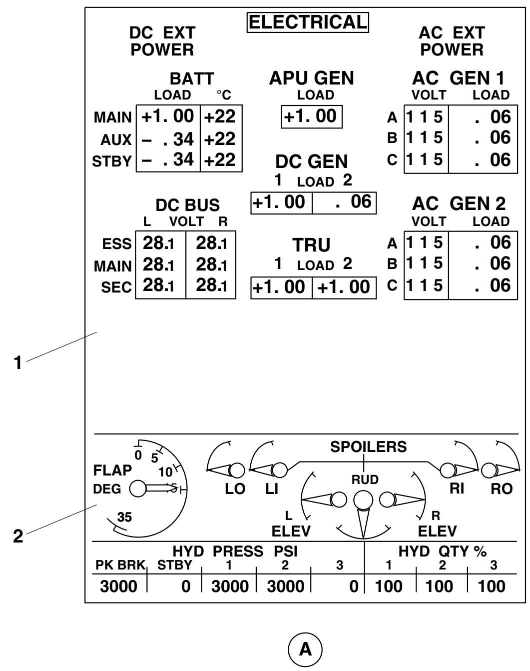
1. System Page or Navigation Page.
2. Permanent System Data Area.

fae31a02.cgm

ELECTRONIC INSTRUMENT SYSTEM MFD1 INDICATION, PLAN FORMAT
Figure 48



MAIN INSTRUMENT PANEL

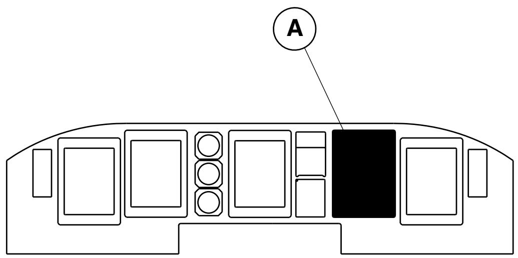


LEGEND

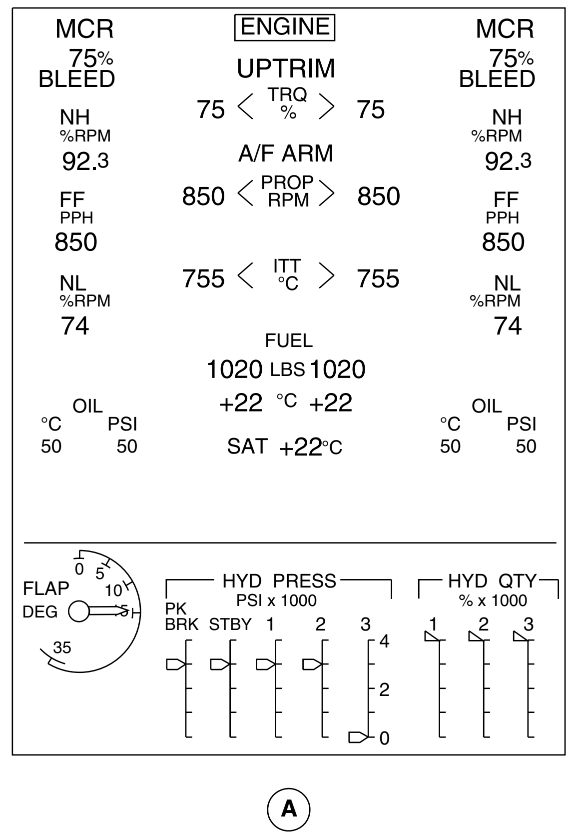
1. Electrical System Page.
2. Permanent System Data Area.

fse32a01.cgm

ELECTRONIC INSTRUMENT SYSTEM MFD2 INDICATIONS
Figure 49

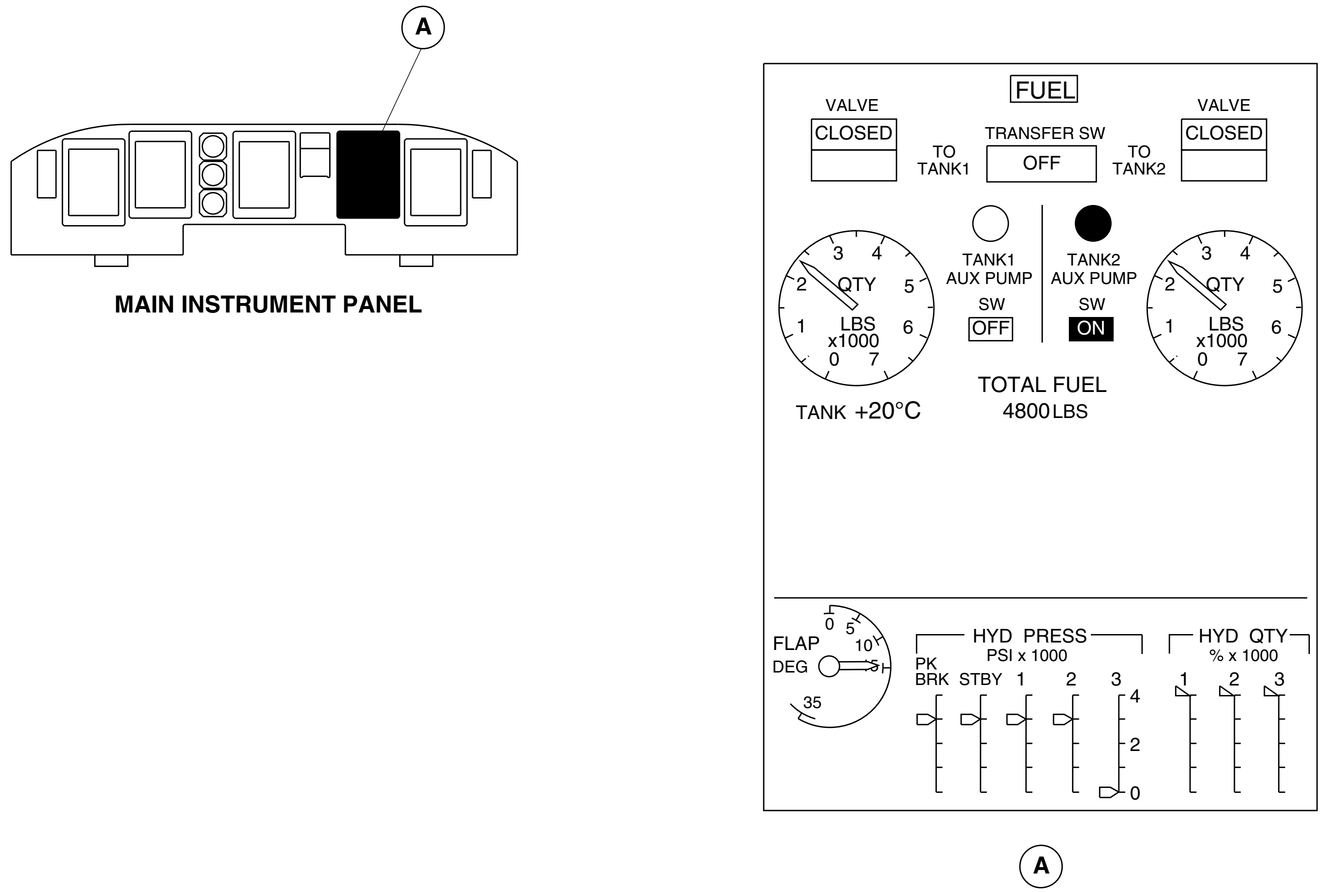


MAIN INSTRUMENT PANEL



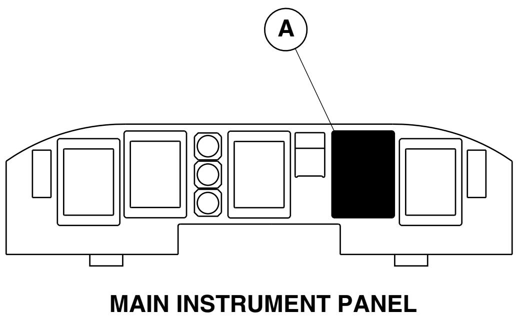
fse61a01.cgm

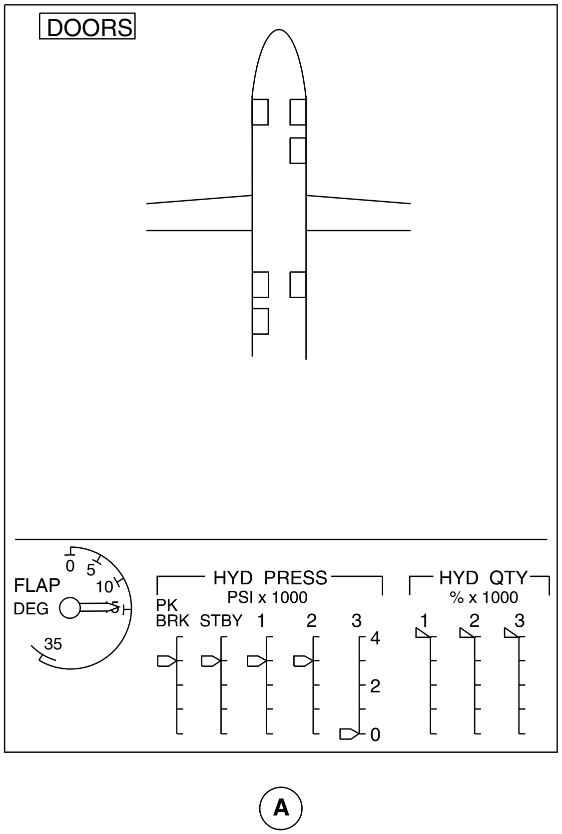
ELECTRONIC INSTRUMENT SYSTEM ENGINE SYSTEM PAGE
Figure 50



fse30a02.cgm

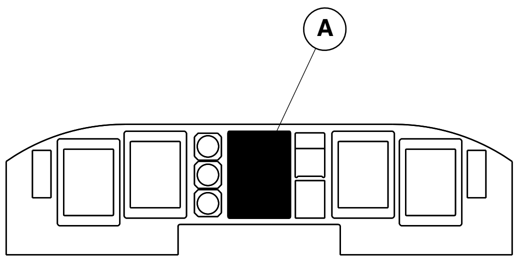
ELECTRONIC INSTRUMENT SYSTEM FUEL SYSTEM PAGE
Figure 51



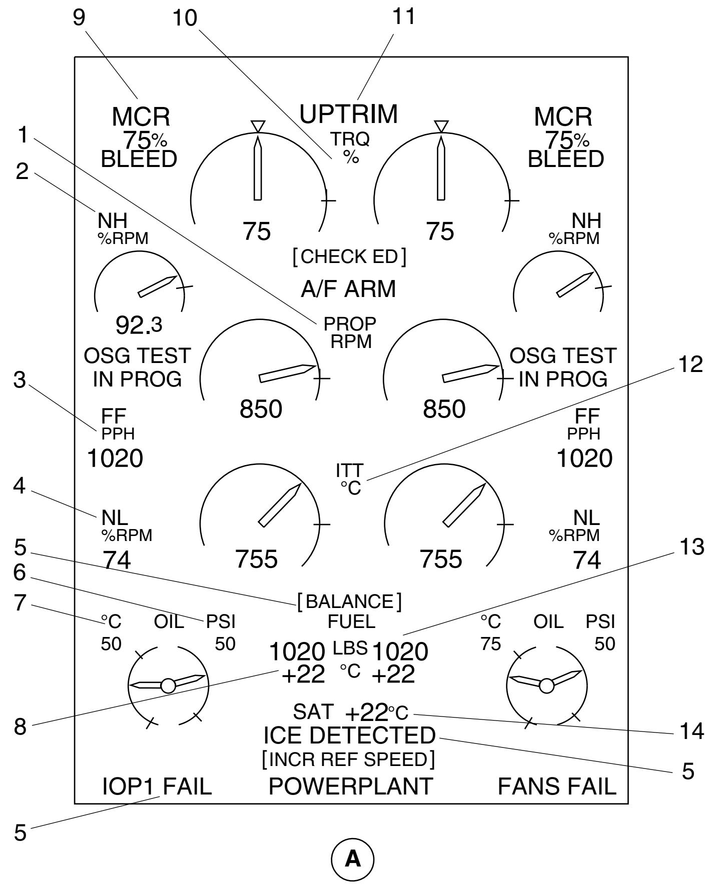


fs748a01.cgm

ELECTRONIC INSTRUMENT SYSTEM DOORS SYSTEM PAGE
Figure 52



MAIN INSTRUMENT PANEL

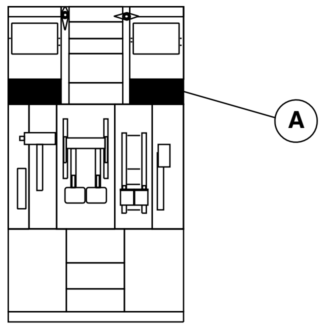


LEGEND

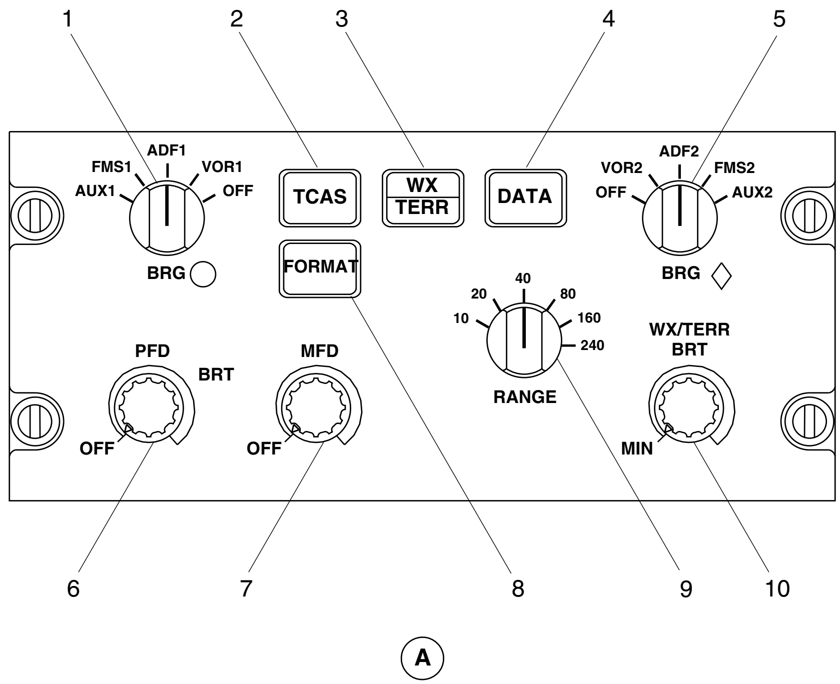
1. NP Propeller Rotational Speed Indicator.
2. NH High Pressure Compressor Rotational Speed Indicator.
3. Fuel Flow Indicator.
4. NL Low Pressure Compressor Rotational Speed Indicator.
5. Advisory Messages.
6. Oil Pressure Indication.
8. Fuel Temperature Indication.
9. Engine Rating Mode Indication.
10. Torque Indication.
11. Powerplant Messages.
12. ITT Inter–Turbine Temperature.
13. Tank Fuel Quantity Indication.
14. Static Air Temperature Indication.

fse57a02.cgm

ELECTRONIC INSTRUMENT SYSTEM ED INDICATIONS
Figure 53



CENTRE CONSOLE

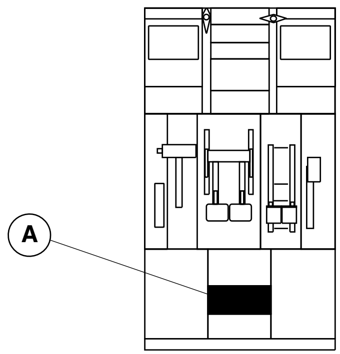


LEGEND

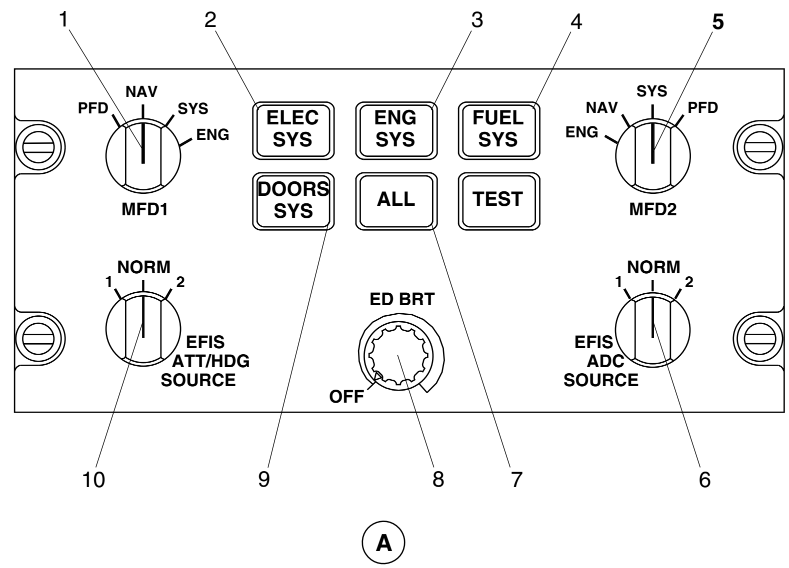
1. BRG 1 Selector Knob.
2. TCAS Pushbutton Switch.
3. WX/TERR Pushbutton Switch.
4. DATA Pushbutton Switch.
5. BRG 2 Selector Knob.
6. PFD OFF, BRT Knob .
7. MFD OFF, BRT Knob.
8. FORMAT Pushbutton Switch.
9. RANGE Selector Knob.
10. WX/TERR BRT Knob.

fag30a02.cgm

ELECTRONIC INSTRUMENT SYSTEM EFCP PAGE 1
Figure 54



CENTRE CONSOLE

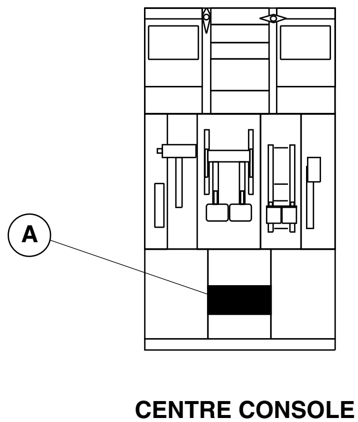


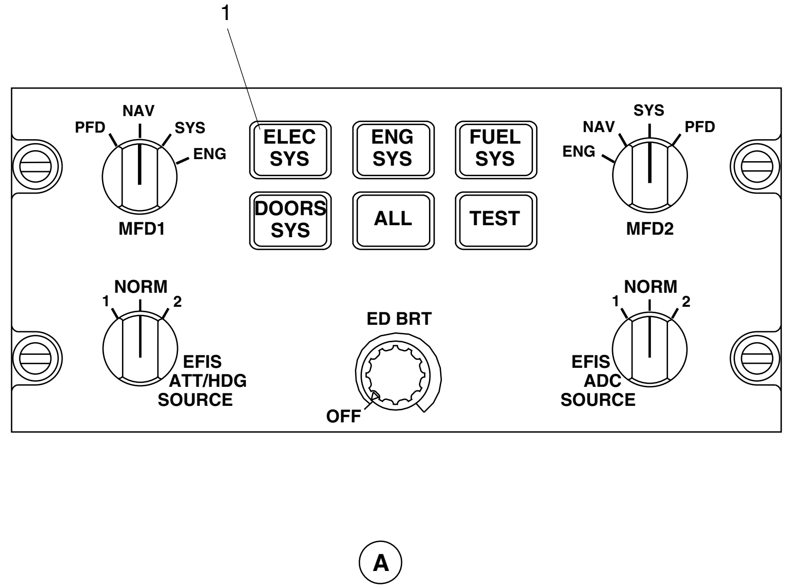
LEGEND

1. MFD 1 Reversion Switches.
2. ELEC SYS Pushbutton Switch.
3. ENG SYS Pushbutton Switch.
4. FUEL SYS Pushbutton Switch.
5. MFD 2 Reversion Switches.
6. EFIS ADC Source Reversion Selector.
7. ALL Pushbutton Switch.
8. ED OFF, BRT Knob.
9. Door System Pushbutton Switch.
10. EFIS ATT/HDG Source Reversion Selector.

fsa73a12.cgm

ELECTRONIC INSTRUMENT SYSTEM ESCP PAGE 2
Figure 55



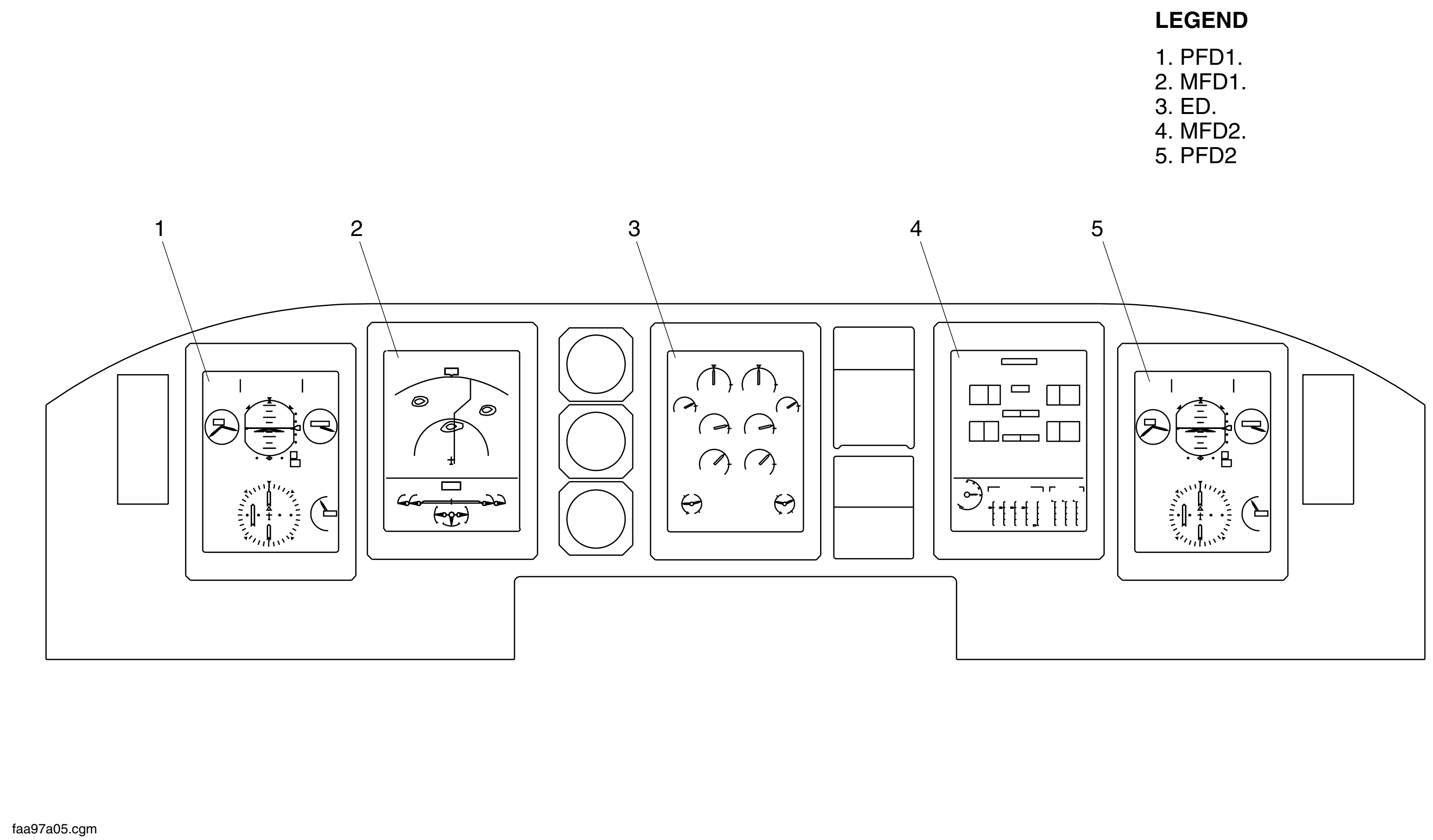


LEGEND

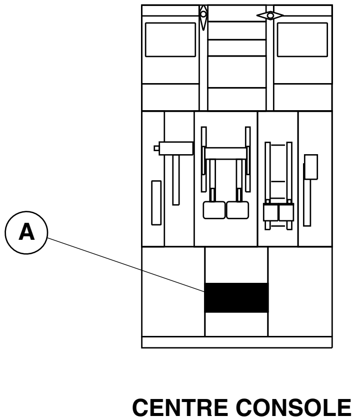
1. Electrical System Pushbutton Switch.

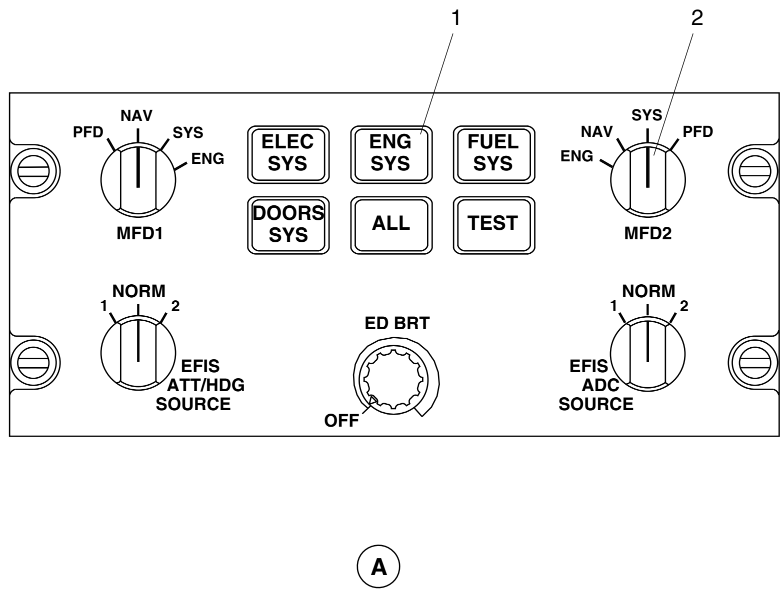
fsb78a01.cgm

ELECTRONIC INSTRUMENT SYSTEM ESCP, ELECTRICAL SYSTEM PAGE SELECTION
Figure 56



ELECTRONIC INSTRUMENT SYSTEM ESCP, ELECTRICAL SYSTEM PAGE INDICATION
Figure 57



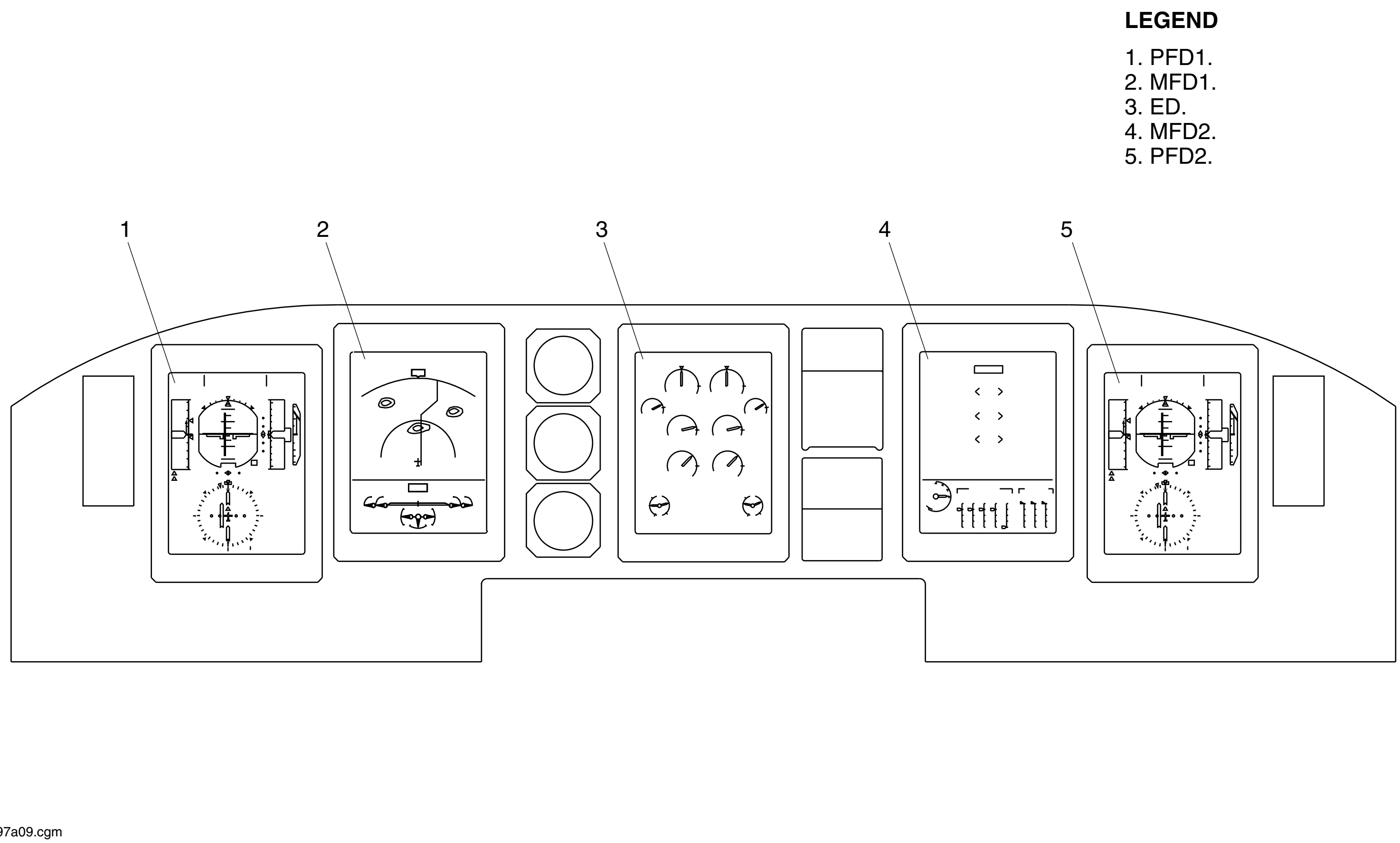


LEGEND

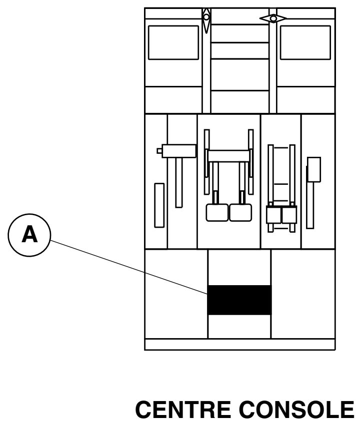
1. Engine System Pushbutton Switch.
2. MFD2 Reversion Selector.

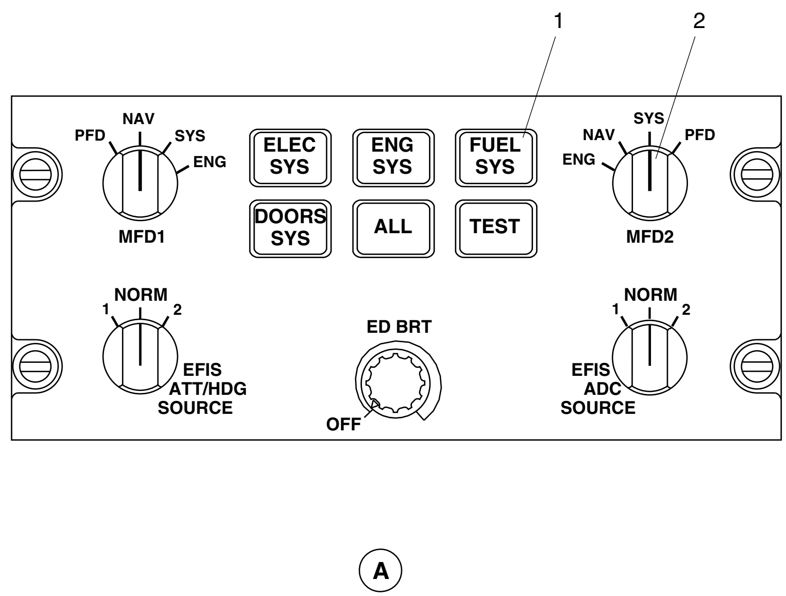
fsb80a03.cgm

ELECTRONIC INSTRUMENT SYSTEM ESCP, ENGINE SYSTEM PAGE SELECTION
Figure 58



ELECTRONIC INSTRUMENT SYSTEM ESCP, ENGINE SYSTEM PAGE INDICATION
Figure 59





LEGEND

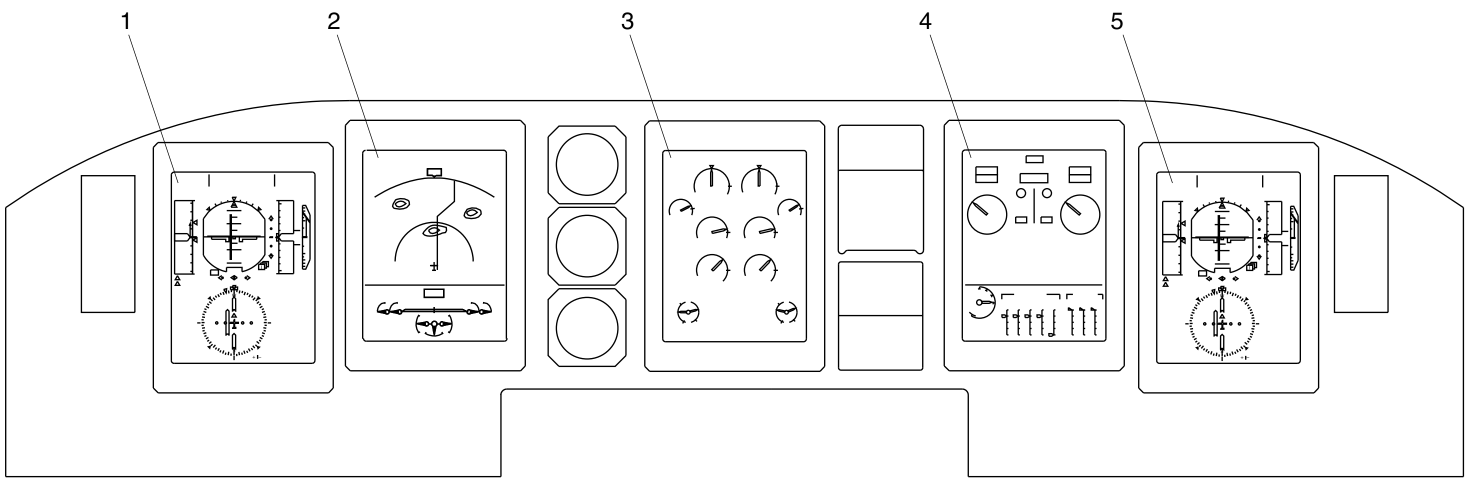
1. Fuel System Pushbutton Switch.
2. MFD2 Reversion Selector.

fsb80a01.cgm

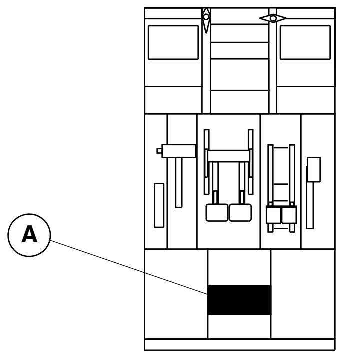
ELECTRONIC INSTRUMENT SYSTEM ESCP, FUEL SYSTEM PAGE SELECTION
Figure 60

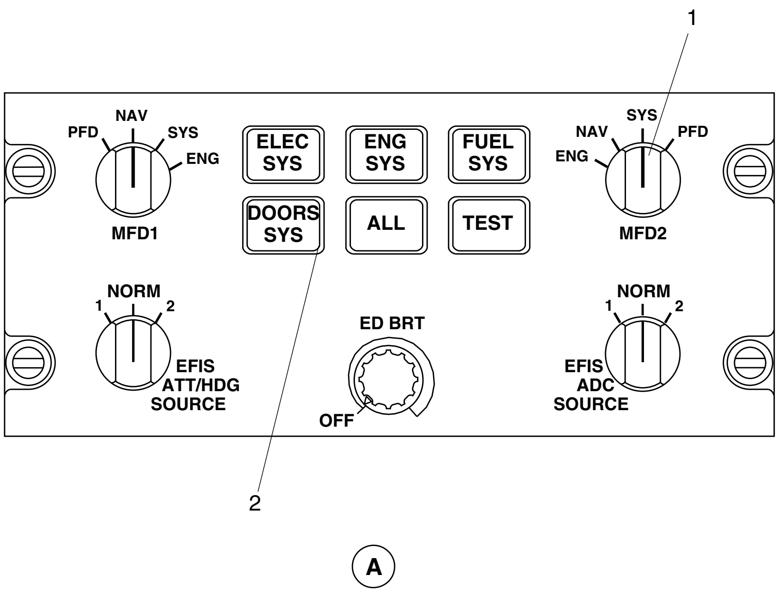
LEGEND

1. PFD1.
2. MFD1.
3. ED.
4. MFD2.
5. PFD2.



ELECTRONIC INSTRUMENT SYSTEM ESCP, FUEL SYSTEM PAGE INDICATION
Figure 61



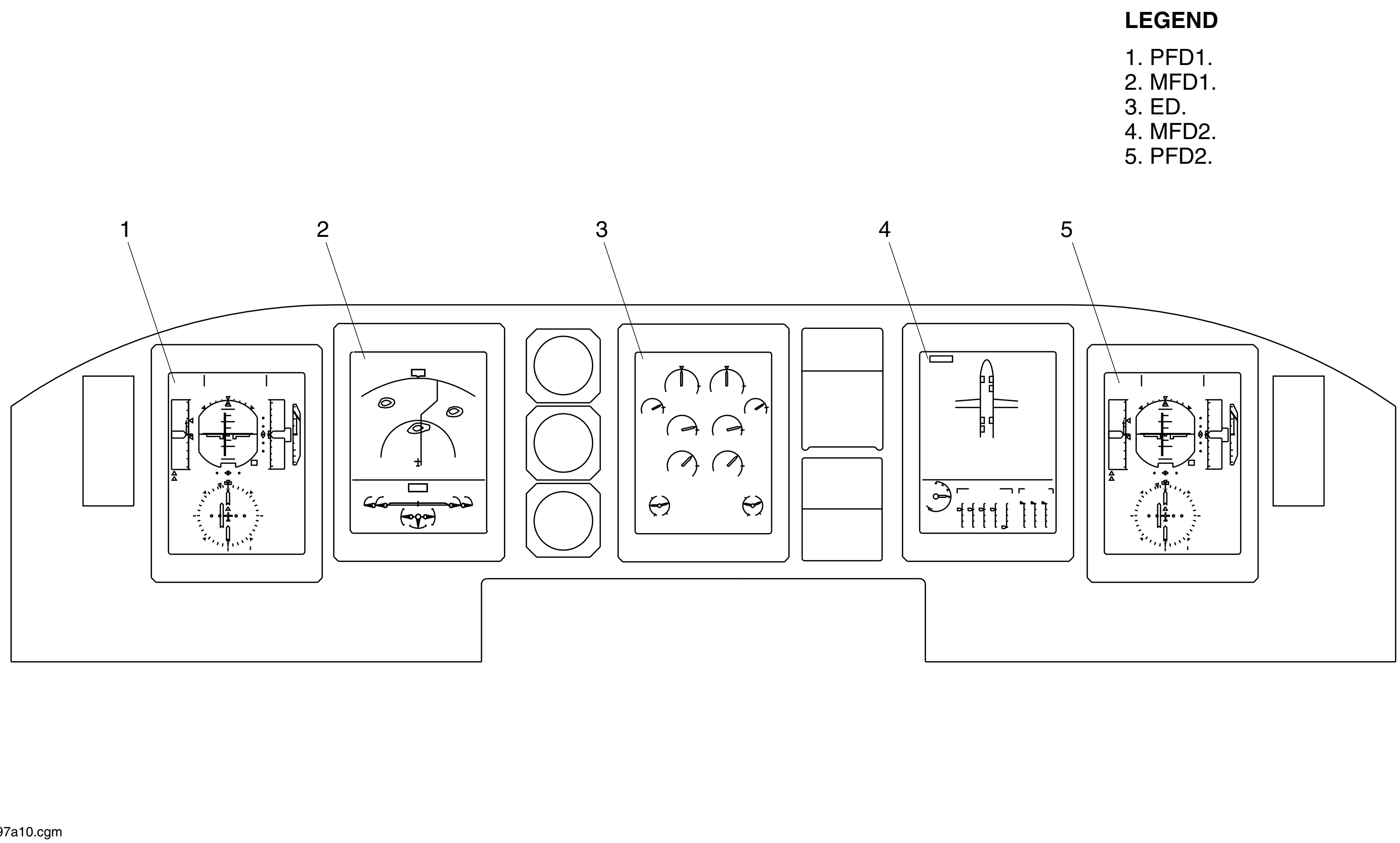


LEGEND

1. MFD2 Reversion Selector.
2. Doors System Pushbutton Switch

fsb80a04.cgm

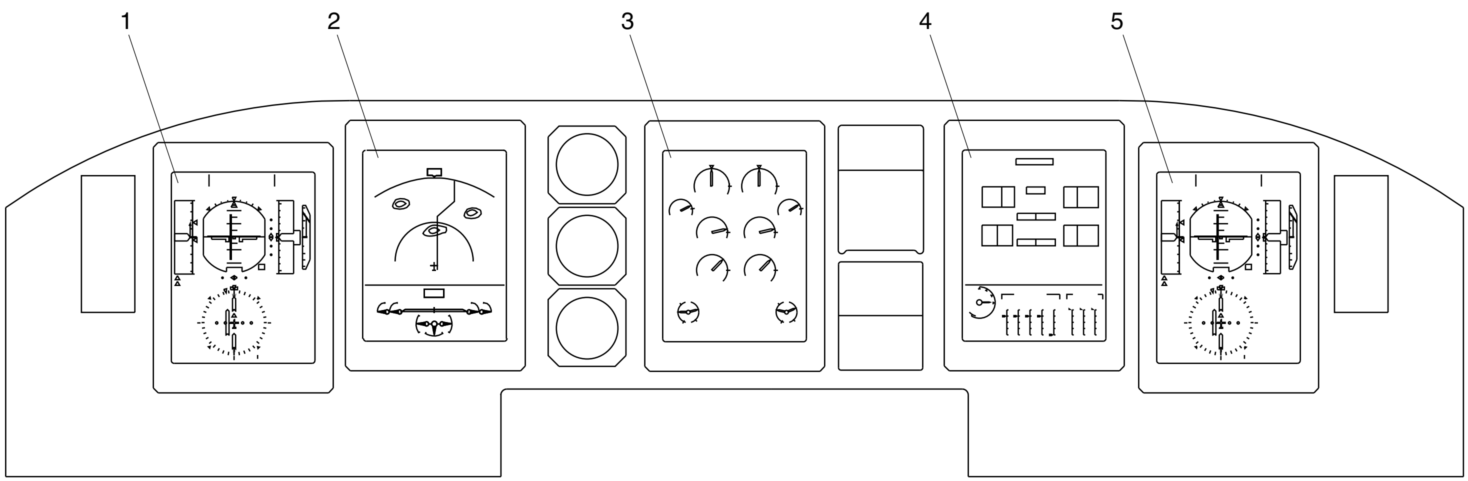
ELECTRONIC INSTRUMENT SYSTEM ESCP, DOORS SYSTEM PAGE SELECTION
Figure 62



ELECTRONIC INSTRUMENT SYSTEM ESCP, DOORS SYSTEM PAGE INDICATION
Figure 63

LEGEND

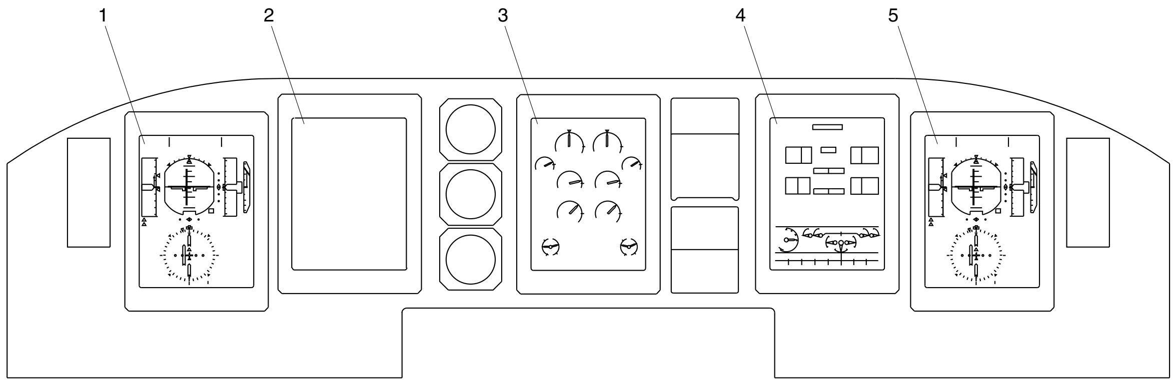
1. PFD1.
2. MFD1.
3. ED.
4. MFD2.
5. PFD2.



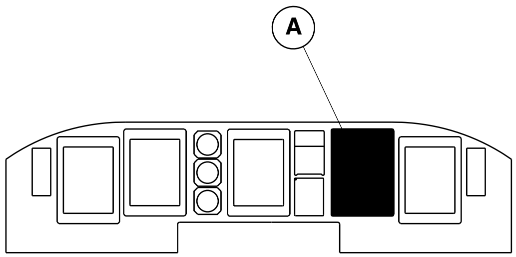
ELECTRONIC INSTRUMENT SYSTEM INDICATIONS PAGE 2
Figure 64

LEGEND

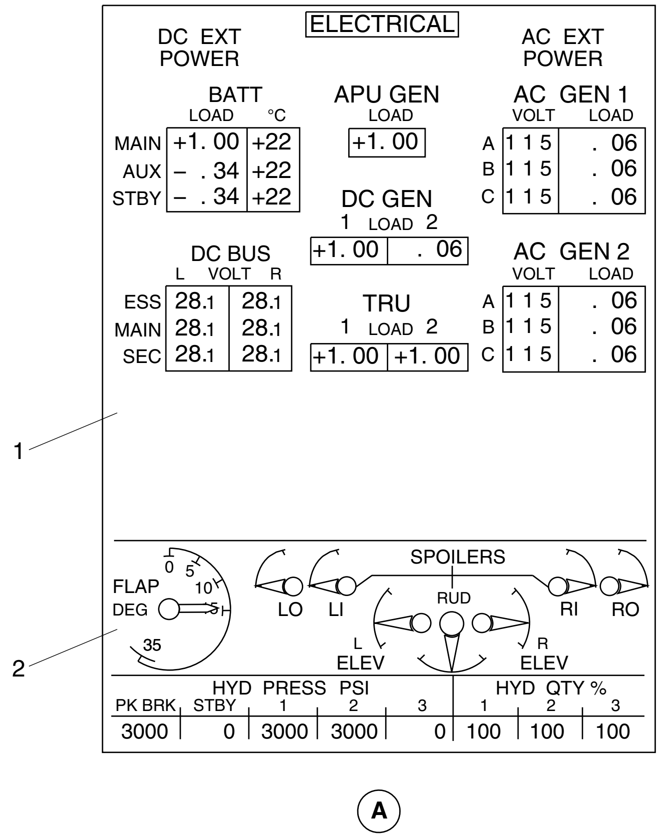
1. PFD1.
2. MFD1 Malfunction.
3. ED.
4. MFD2.
5. PFD2



ELECTRONIC INSTRUMENT SYSTEM MFD1 MALFUNCTION
Figure 65



MAIN INSTRUMENT PANEL



LEGEND

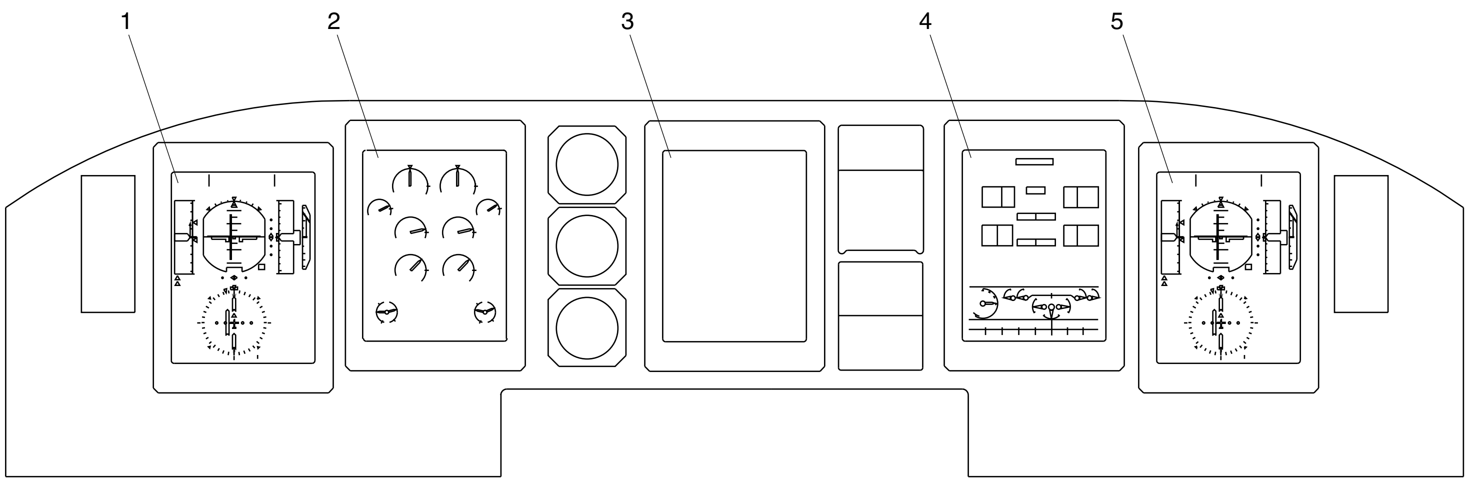
1. System Page.
2. Permanent System Data Area.

fse33a01.cgm

ELECTRONIC INSTRUMENT SYSTEM PERMANENT SYSTEM DATA AREA,COMPOSITE VIEW
Figure 66

LEGEND

1. PFD1.
2. MFD1 Showing ED.
3. ED Malfunction.
4. MFD2.
5. PFD2



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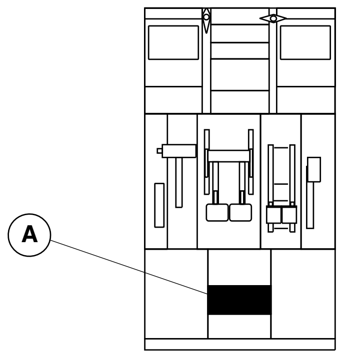
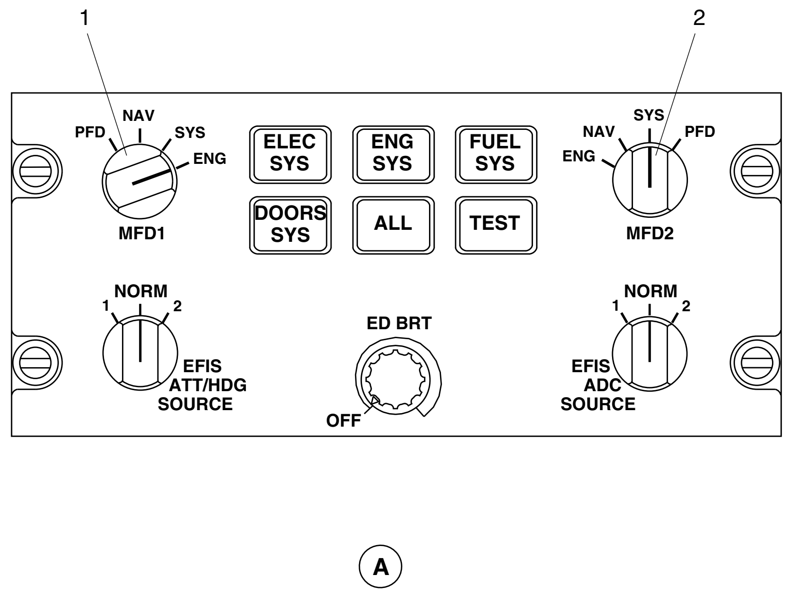
ELECTRONIC INSTRUMENT SYSTEM ED MALFUNCTION
Figure 67



CENTRE CONSOLE

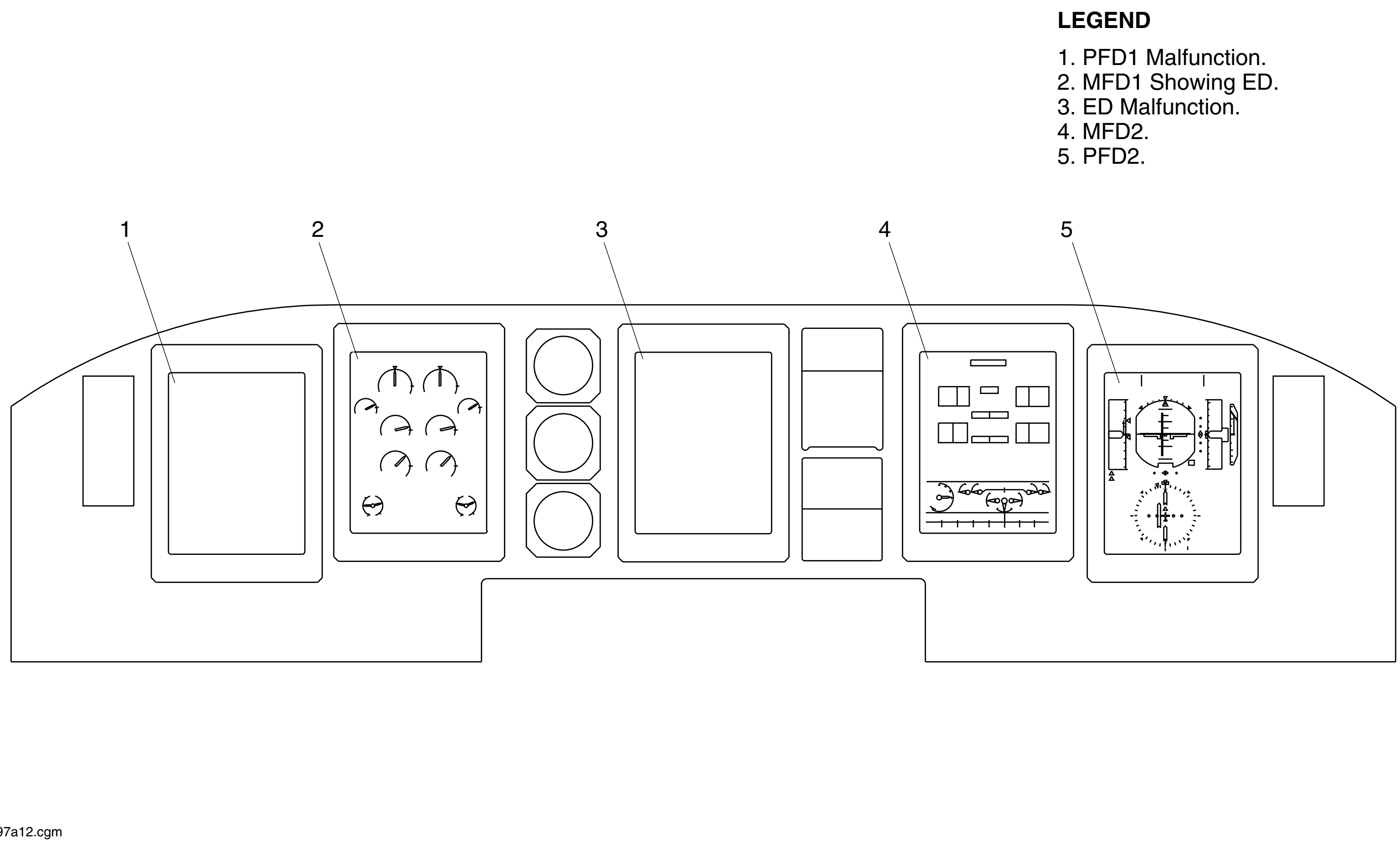


LEGEND

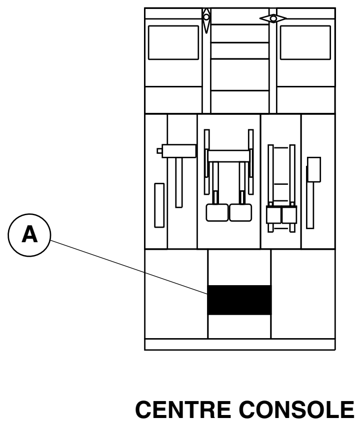
1. MFD 1 Reversion Selector.
2. MFD 2 Reversion Selector.

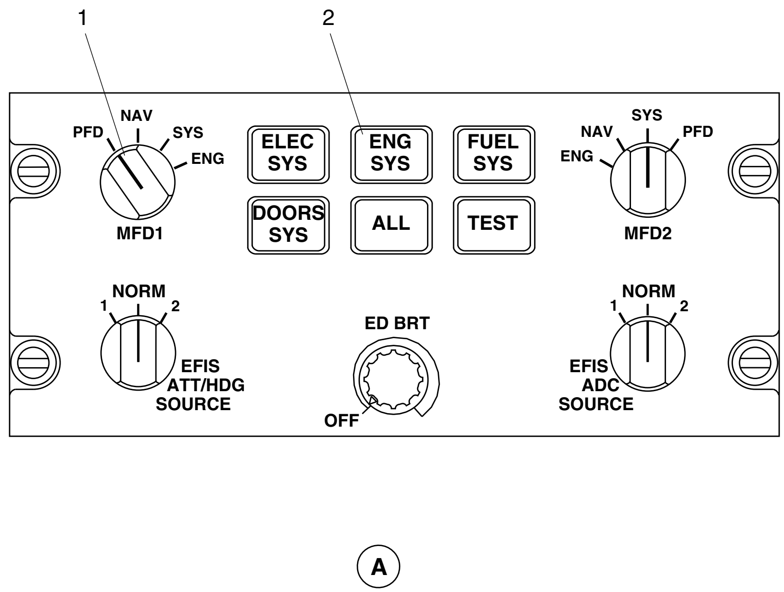
fsb76a01.cgm

ELECTRONIC INSTRUMENT SYSTEM ESCP, ED MFD1 REVERSION SELECTION
Figure 68



ELECTRONIC INSTRUMENT SYSTEM PFD1 AND ED MALFUNCTION
Figure 69



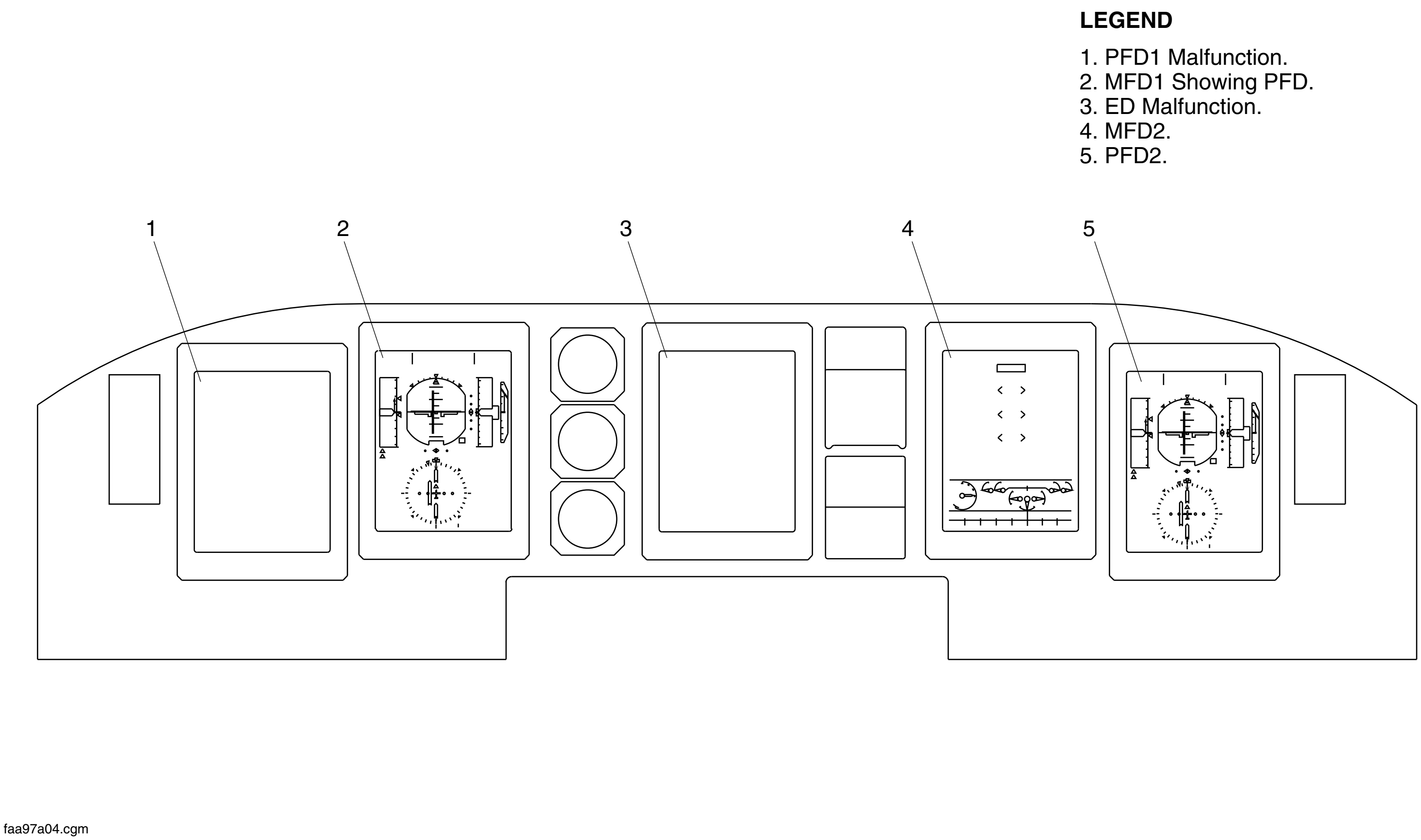


LEGEND

1. MFD1 Reversion Selector.
2. Engine System Pushbutton Switch.

fsb77a01.cgm

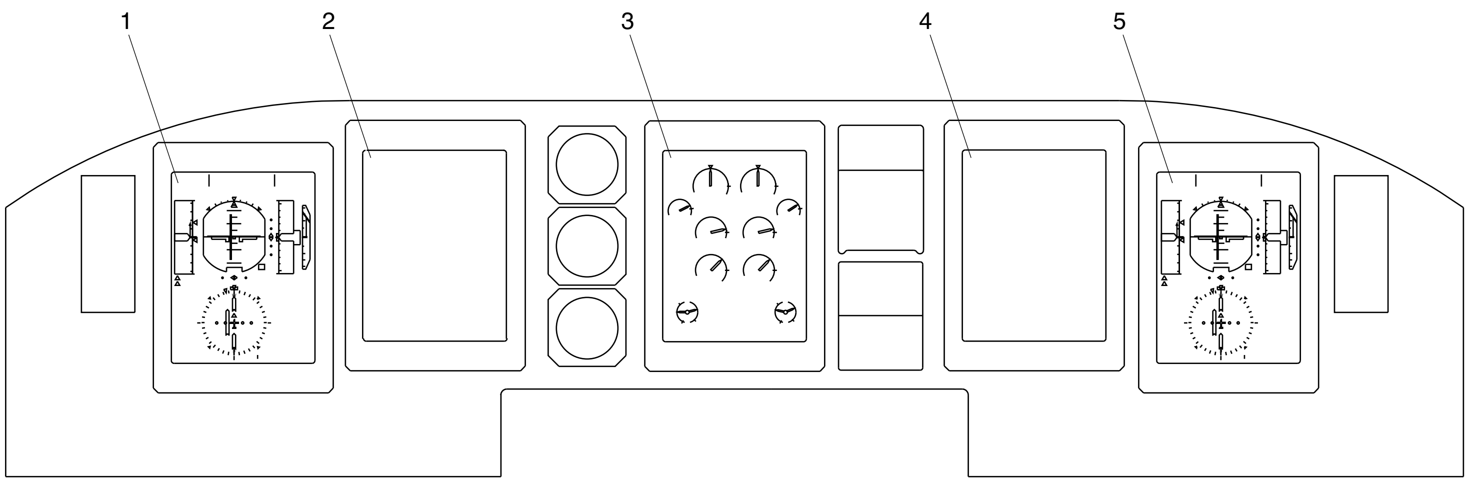
ELECTRONIC INSTRUMENT SYSTEM ESCP, PFD MFD1 REVERSION AND ENGINE SYSTEM PAGE SELECTIONS
Figure 70



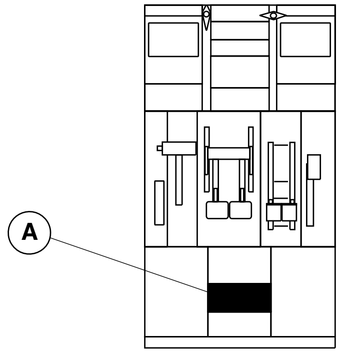
ELECTRONIC INSTRUMENT SYSTEM, PFD1 AND ED REVERSIONARY INDICATION
Figure 71

LEGEND

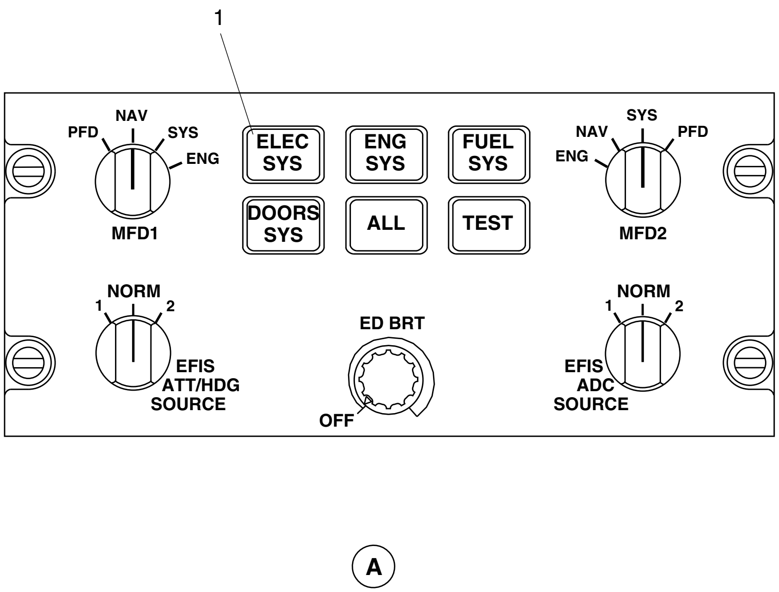
1. PFD1.
2. MFD1 Malfunction.
3. ED.
4. MFD2 Malfunction.
5. PFD2.



ELECTRONIC INSTRUMENT SYSTEM MFD1 AND MFD2 MALFUNCTION
Figure 72



CENTRE CONSOLE



LEGEND

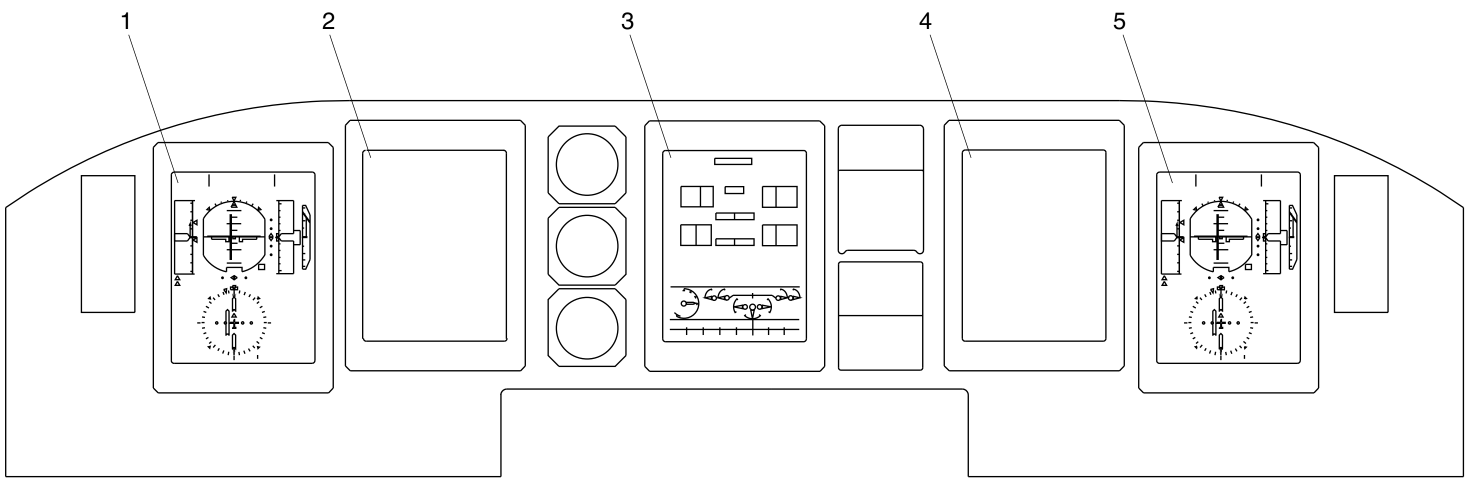
1. Electrical System Pushbutton Switch.

fsb78a01.cgm

ELECTRONIC INSTRUMENT SYSTEM ESCP, MONO ELECTRICAL SYSTEM PAGE SELECTION
Figure 73

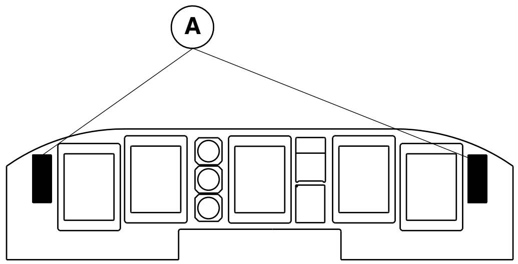
LEGEND

1. PFD1.
2. MFD1 Malfunction.
3. ED Showing System Page.
4. MFD2 Malfunction.
5. PFD2.

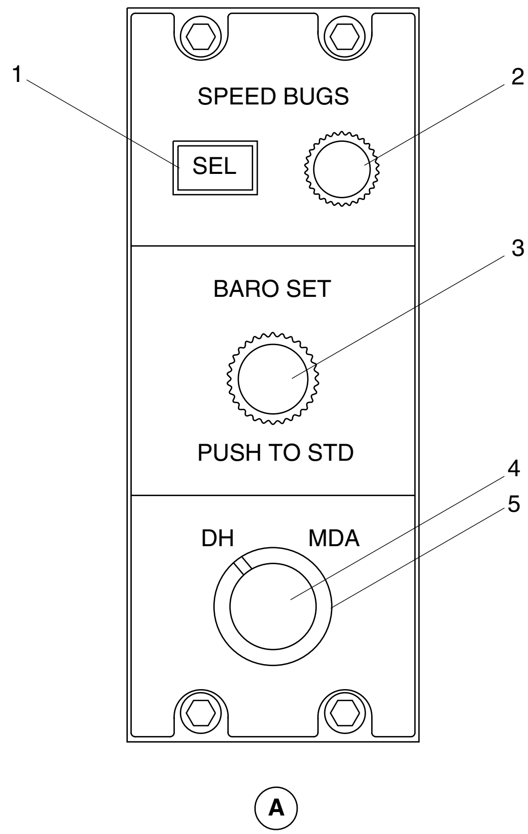


faa97a13.cgm

ELECTRONIC INSTRUMENT SYSTEM ESCP, MONO ELECTRICAL SYSTEM PAGE INDICATION
Figure 74



MAIN INSTRUMENT PANEL

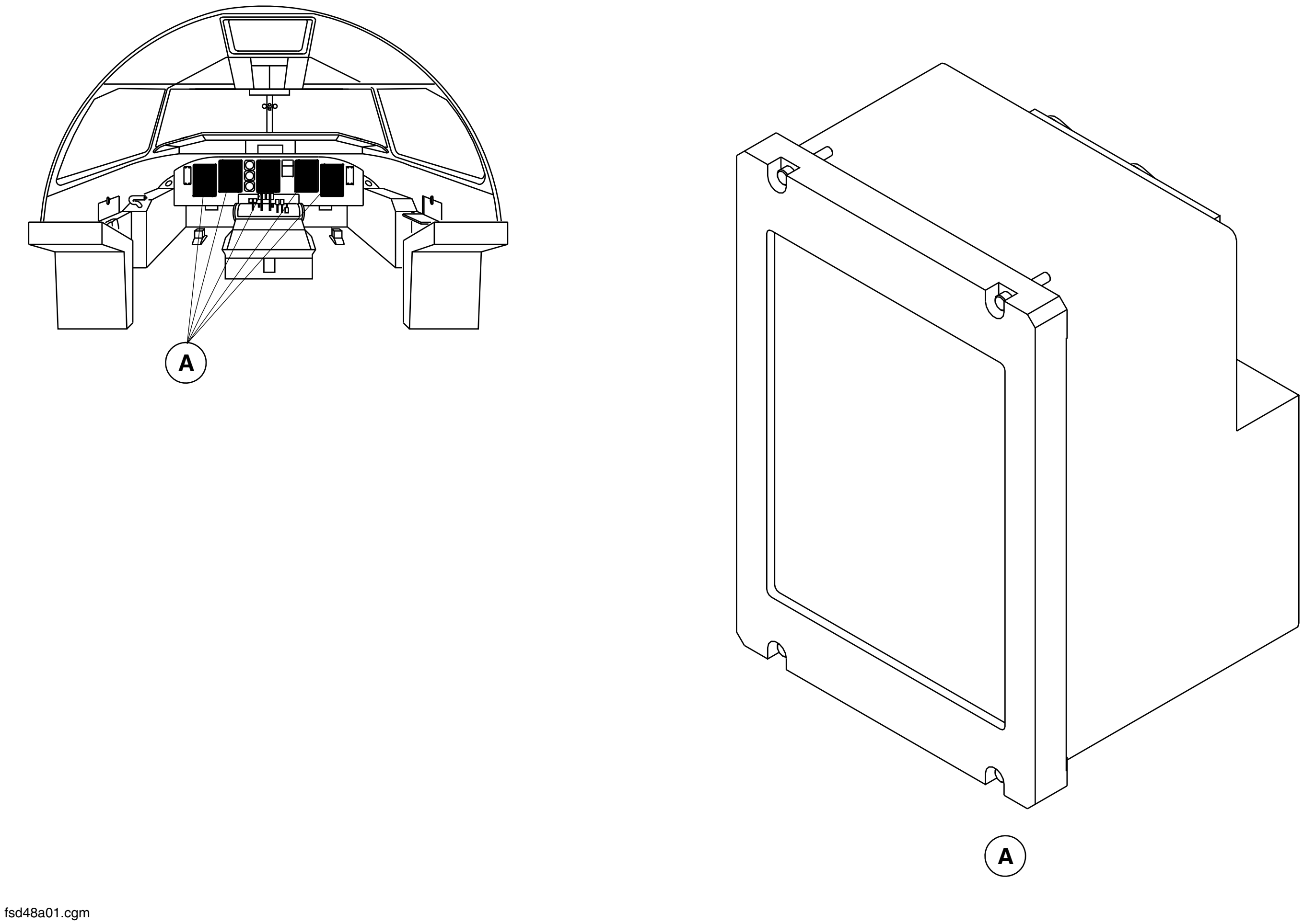


LEGEND

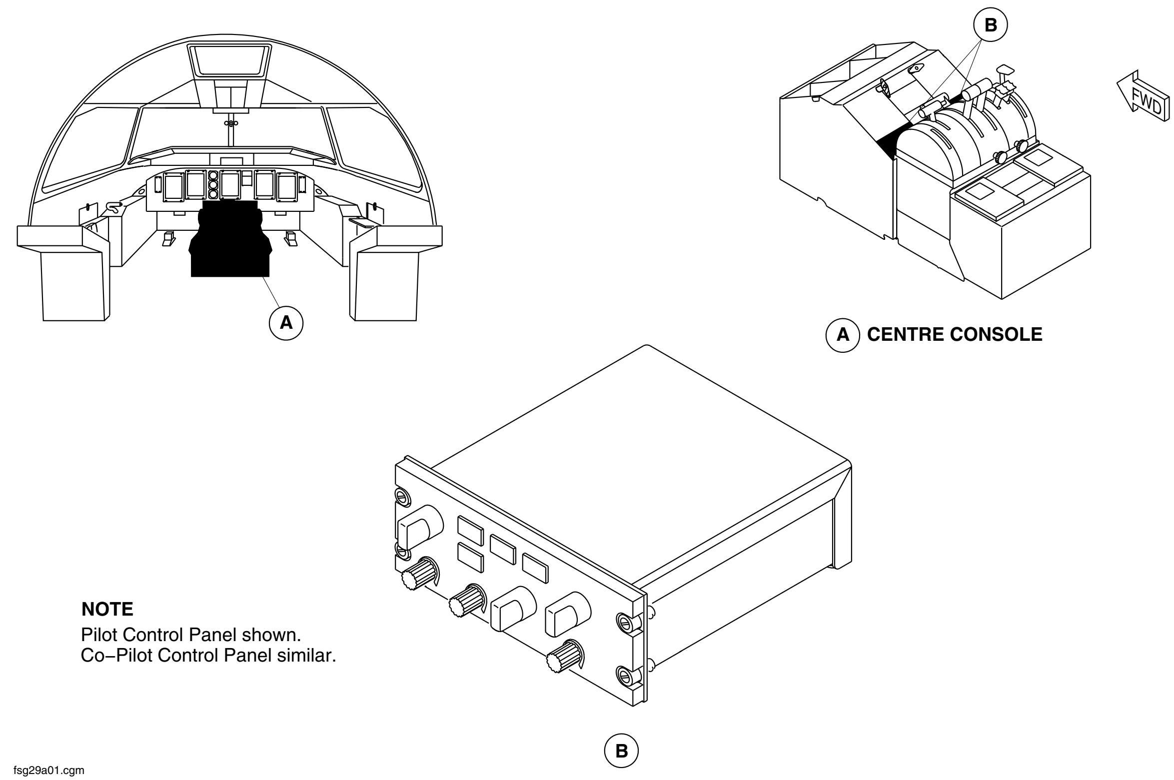
1. Speed Bug Index SEL Pushbutton Switch.
2. Speed Bug Rotary Selector.
3. BARO SET Knob / PUSH TO STD Switch.
4. DH Knob.
5. DH/MDA Switch.

fab95a02.cgm

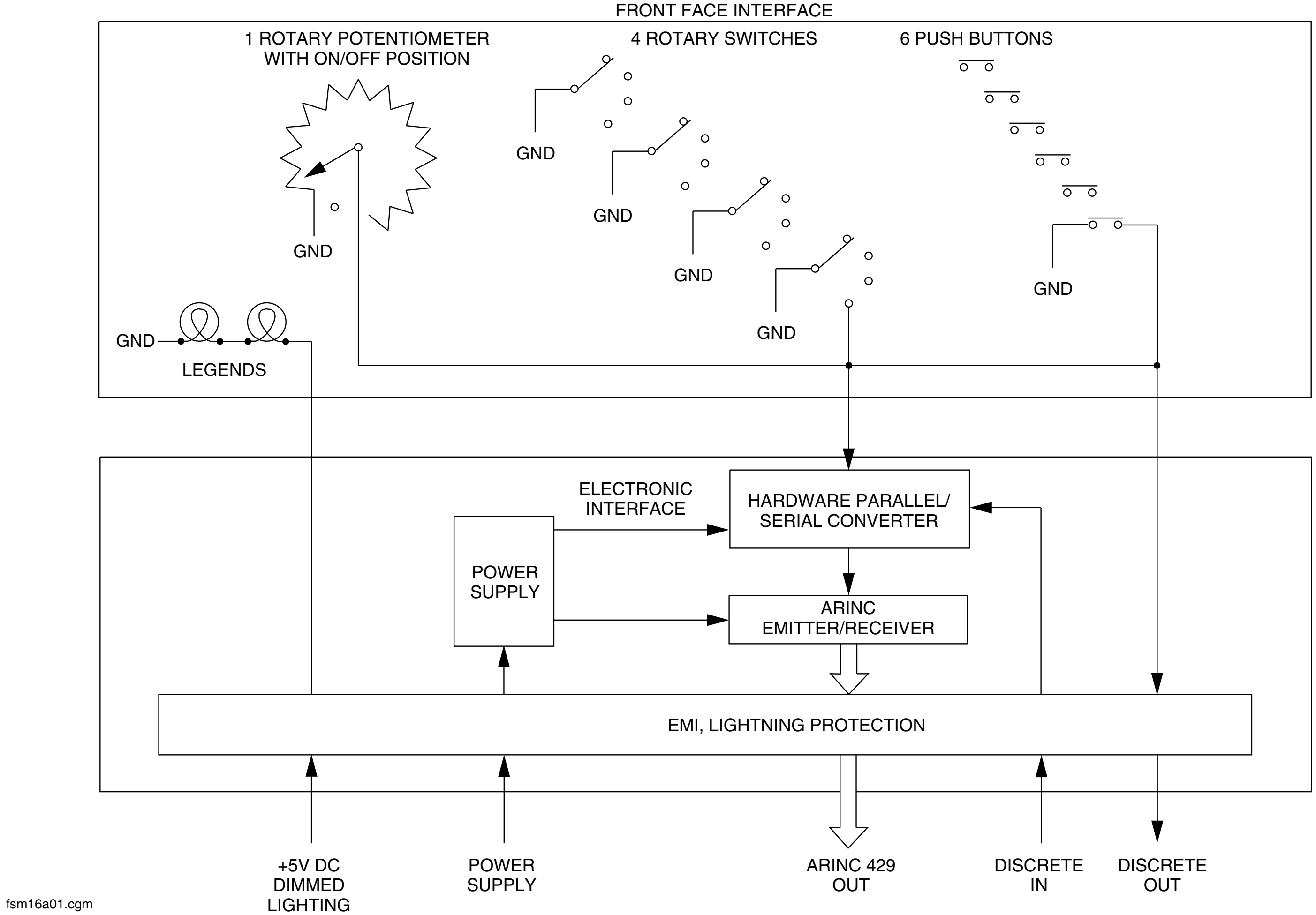
ELECTRONIC INSTRUMENT SYSTEM ICP
Figure 75



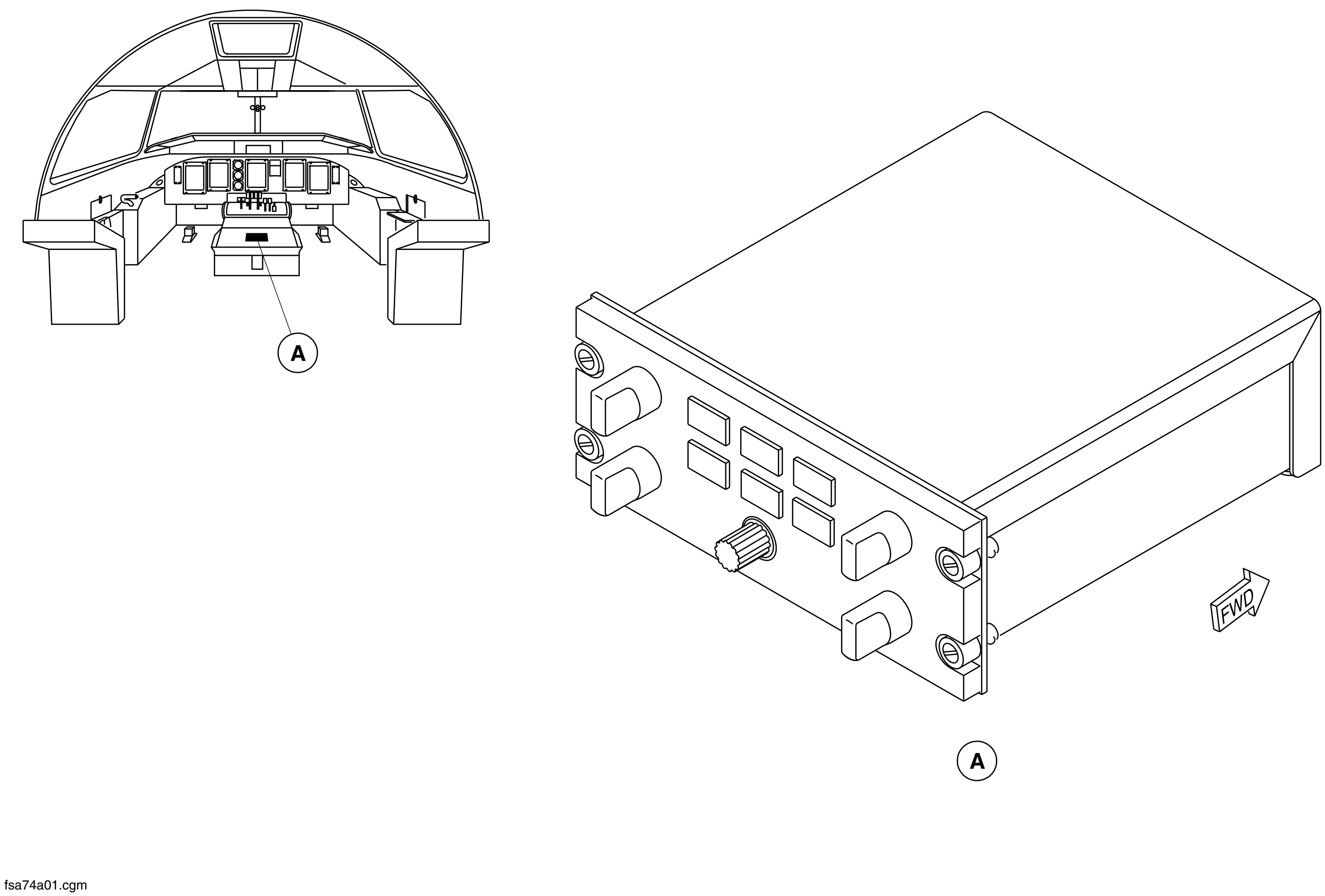
DISPLAY UNIT LOCATOR
Figure 76



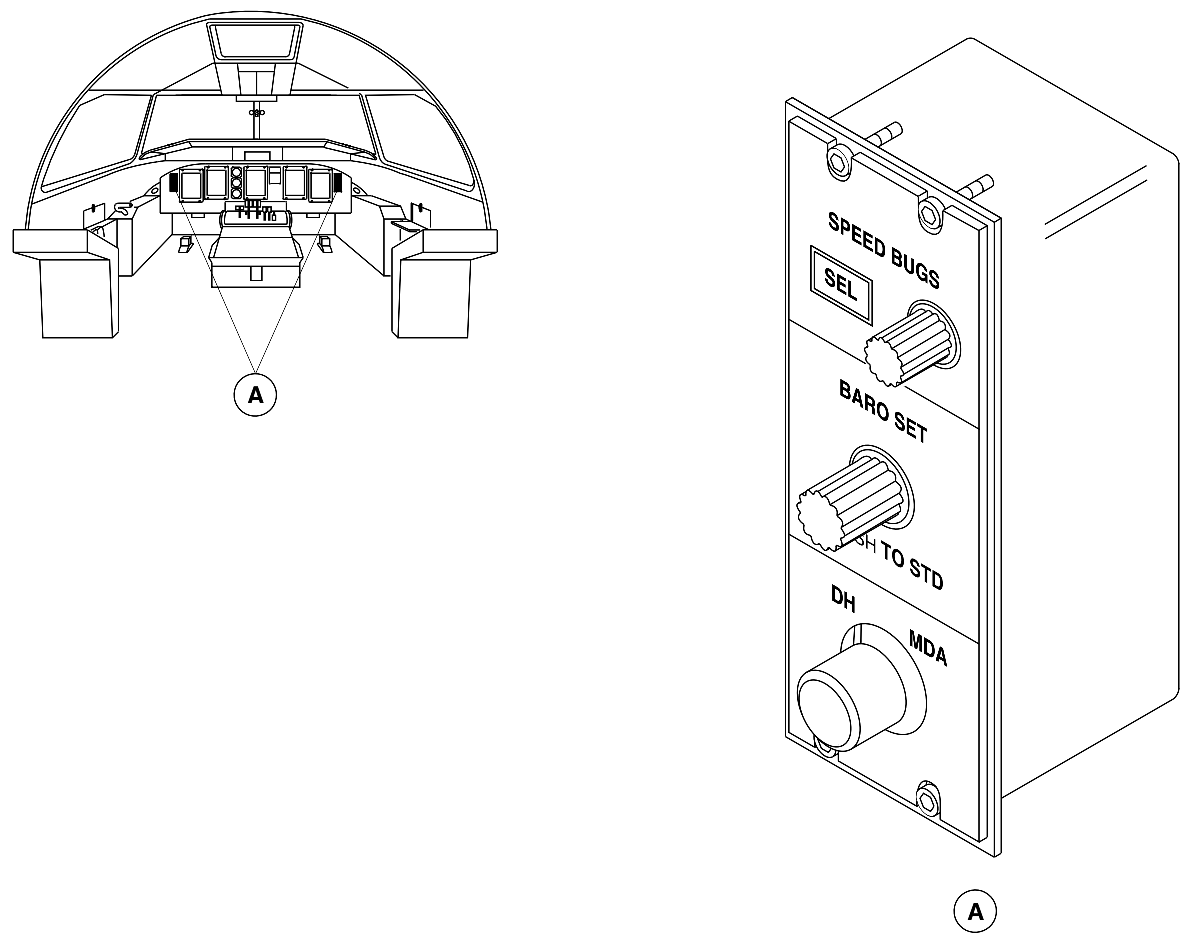
EFIS CONTROL PANEL (EFCP) LOCATOR
Figure 77



CONTROL PANEL BLOCK DIAGRAM
Figure 78



ESID CONTROL PANEL (ESCP) LOCATOR
Figure 79



INDEX CONTROL PANEL (ICP) LOCATOR
Figure 80