C Series

BD500 SERIES (PW PW1500G) INITIAL MAINTENANCE TRAINING COURSE

TECHNICAL TRAINING MANUAL

MODULE 5: POWER PLANT

CMUI: CS130-21.07-06-00-121418

VERSION v6.00

BD-500-1A10

BD-500-1A11

BOMBARDIER

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ACRONYMS AND ABBREVIATIONS

A	
A/T	autothrottle
ABIT	automatic BIT
ABS	absolute
ABS	autobrake system
ACARS	aircraft communications addressing and reporting system
ACC	active clearance control
ACES	avionics cooling and extraction system
ACL	access control list
ACM	air cycle machine
ACM	aircraft condition monitoring
ACMF	aircraft condition monitoring function
ACMP	AC motor pump
ACP	audio control panel
ACU	audio conditioning unit
ADC	air data computer
ADEF	aircraft data exchange function
ADF	automatic direction finder
ADI	attitude direction indicator
ADLS	airborne data link system
ADMF	aircraft data management function
ADRF	aircraft data recording function
ADS	air data system
ADSP	air data smart probe
ADSP	air data system probe
AES	alternate extension system
AEV	avionics exhaust valve
AEMF	aft engine mount fitting
AFCS	automatic flight control system
AFCU	alternate flight control unit
AFDA	adaptive flight display application
AFDT	adaptive flight display table
AFDX	avionics full duplex switched Ethernet
AFM	Airplane Flight Manual
AFP	automated fiber placement
AGB	accessory gearbox
AGL	above ground level
AHC	attitude heading computer
AHMS	aircraft health management system
AIS	aircraft information server
AIS	audio integrating system
AIM	axle interface module
AIM	aircraft identification module
AIM	align-in-motion
AL	autoland
ALC	APU line contactor
ALI	airworthiness limitation item
Al-Li	aluminum-lithium
ALM	application license manager
ALT	altitude
ALTN FLAP	alternate flap
AM	amplitude modulation
AMCU	advanced monitor control unit
AMP	Aircraft Maintenance Publication
ANR	archive noise reduction
ANS	aircraft network switch

AOA	angle-of-attack
AOC	air/oil cooler
AOC	airport operational communications
AODV	active oil damper valve
AOHX	air/oil heat exchanger
AP	autopilot
APM	aircraft personality module
APR	automatic power reserve
APU	auxiliary power unit
AR	automatic realignment
ARTCC	air route traffic control center
ASA	autoland status annunciator
ASC	APU starting contactor
ASM	air separation module
ASRP	aircraft structure repair publication
AT	autothrottle
ATC	air traffic control
ATIS	air traffic information services
ATP	acceptance test procedure
ATS	air turbine starter
ATS	autothrottle system
ATS	air traffic services
AV-VENTS	avionics ventilated temperature sensor
B	
BALODS	bleed air leak and overheat detection system
BAP	buffer air pressure
BAS	bleed air system
BAV	bleed air valve
BACV	buffer air check valve
BAHX	buffer air heat exchanger
BAPS	buffer air pressure sensor
BAVS	buffer air valve solenoid
BAVSOV	buffer air shutoff valve
BCCC	base coat clear coat
BCS	brake control system
BDCU	brake data concentrator unit
BFO	beat frequency oscillator
BGM	boarding music
BIT	built-in test
BITE	built-in test equipment
BL	buttock line
BLC	battery line contactor
BLS	bifurcation latch system
BMPS	bleed monitoring pressure sensor
BPCU	bus power control unit
BPMS	bleed pressure monitoring sensor
BSC	battery start contactor
BTC	bus tie contactor
BTS	base transceiver station
BTMS	brake temperature monitoring system
BTS	brake temperature sensor
BTS	bleed temperature sensor
BVID	barely visible impact damage
C	
CADTS	cargo duct temperature sensor
CAI	cowl anti-ice
CAIS	cowl anti-ice system
CAIV	cowl anti-ice valve

CAM	cockpit area microphone
CAN	controller air network
CAS	calibrated airspeed
CAS	crew alerting system
CATS	cargo temperature sensor
CB	circuit breaker
CBIT	continuous built-in test
CBP	circuit breaker panel
CBV	cross-bleed valve
CC	cabin controller
CCDL	cross-channel data link
CCM	common computing module
CCMR	common computing module runtime
CCP	cursor control panel
CCU	camera control unit
CCW	counterclockwise
CDC	control and distribution cabinet
CDI	course deviation indicator
CDTS	compressor discharge temperature sensor
CEM	cover and environmental module
CF	configuration file
CFIT	controlled flight into terrain
CFRP	carbon fiber reinforced polymer
CIC	compressor intermediate case
CIC	corrosion inhibiting compound
CLAWS	control laws
CM	configuration manager
CMS	cabin management system
CMU	communication management unit

CMUI	configuration management unique identifier
CNS	communications, navigation and surveillance
COM	communication
CPCS	cabin pressure control system
CPCV	compensator pressure check valve
CPD	circuit protection device
CPDD	circuit protection device detector
CPDLC	controller-pilot data link communications
CPN	Collins part number
CPU	central processing unit
CRC	cyclical redundancy check
CRES	corrosion resistant steel
CSD	cabin service display
CSD	customer service display
CSMU	crash-survivable memory unit
CSOV	cargo shutoff valve
CT	crew terminal
CT	current transformer
CTP	control tuning panel
CVR	cockpit voice recorder
CW	clockwise
CWB	center wing box
D	
D/I	discrete input
D/O	discrete output
DBM	database manager
DCM	data concentrator module
DCMR	data concentration module runtime
DCS	data concentration system

DCU	directional control unit
DCV	directional control valve
DFSOV	dual-flow shutoff valve
DLCA	data link communications application
DMA	display manager application
DMC	data concentrator unit module cabinet
DME	distance measuring equipment
DMM	data memory module
DP	differential protection
DPCT	differential protection current transformer
DPI	differential pressure indicator
DPLY	deploy
DRA	diagnostic and reporting application
DSK	double-stack knob
DSM	digital switching module
DSU	data storage unit
DSPU	diode shunt protection unit
DTC	DC tie contactor
DTE	damage tolerance evaluation
DTI	damage tolerance inspection
DTS	duct temperature sensor
DU	display unit
E	
EBC	essential bus contactor
ECL	electronic checklist
ECS	environmental control system
ECU	electronic control unit
ECU	external compensation unit
EDCM	electronic door control module

EFAN	extraction fan
EDM	emergency descent mode
EDP	engine-driven hydraulic pump
EDP	engine-driven pump
EDU	electronic display unit
EEC	electronic engine control
EEGS	emergency electrical power generation
EEPROM	electrical erasable programmable read only memory
EESS	emergency escape slide system
EFB	electronic flight bag
EEGS	emergency electrical power generation
EFCS	electronic flight control system
eFIM	electronic fault isolation manual
EFIS	electronic flight instrument system
EGT	exhaust gas temperature
EHSV	electrohydraulic servovalve
EIC	engine inlet cowl
EICAS	engine indication and crew alerting system
ELC	external power line contactor
ELT	emergency locator transmitter
EMA	electric motor actuator
EMU	expansion module unit
EMAC	electric motor actuator controller
EMCU	electric motor control unit
EMER	emergency
EMPC	emergency power control
EOAM	emergency opening assist means
EOF	end-of-flight
EPC	electrical power center

EPC	electronic power center
EPGD	electrical power generation and distribution
EPGDS	electrical power generation and distribution system
EPGS	electrical power generation system
EPP	engine programming plug
EPSU	emergency power supply unit
EPTRU	external power transformer rectifier unit
ERAV	emergency ram air valve
ESD	electrostatic discharge
ETC	essential tie contactor
ETOPS	extended-range twin-engine operational performance standards
F	
FA	flight attendant
FAA	Federal Aviation Authority
FADEC	full authority digital engine control
FANS	future air navigation system
FAV	fan air valve
FBC	front bearing compartment
FBW	fly-by-wire
FBWPC	fly-by-wire power converter
FC	full close
FCP	flight control panel
FCS	flight control system
FCBS	fatigue critical baseline structure
FCEE	flight crew emergency exit
FCSB	fan cowling support beam
FCU	flush control unit
FCU	fuel control unit
FCV	flow control valve
FD	flight director
FDDSS	flight deck door surveillance system
FDE	flight deck effect
FDG	fan drive gearbox
FDGS	fan drive gear system
FDR	flight data recorder
FDRAS	flight deck remote access system
FDV	flow divider valve
FEGV	fan exit guide vane
FEMB	forward engine mount bulkhead
FF	fuel flow
FFDP	fuel flow differential pressure
FFSV	free fall selector valve
FG	flight guidance
FGS	flight guidance system
FIC	fan intermediate case
FIDEX	fire detection and extinguishing
FIM	fault isolation manual
FLC	flight level change
FLS	fast load-shed
FLTA	forward-looking terrain avoidance
FMA	flight mode annunciator
FMS	flight management system
FMSA	flight management system application
FMV	fuel metering valve
FO	full open
FOD	foreign object debris
FOHX	fuel/oil heat exchanger

FOHXBV	fuel/oil heat exchanger bypass valve
FPS	feedback position sensor
FPV	flight path vector
FQC	fuel quality computer
FS	fixed structure
FS	flight station
FS	fuselage station
FSA	file server application
FSV	flow sensor venturi
FSB	fasten seat belt
FSCL	flight spoiler control lever
FTIS	fuel tank inerting system
FW	failure warning
FWSOV	firewall shutoff valve
G	
GA	go-around
GCF	ground cooling fan
GCR	generator control relay
GCS	global connectivity suite
GCU	generator control unit
GFP	graphical flight planning
GFRP	glass fiber reinforced polymer
GHTS	galley heater temperature sensor
GLC	generator line contactor
GMT	Greenwich mean time
GNSS	global navigation satellite system
GPWS	ground proximity warning system
GS	glideslope
GS	ground spoiler
GSA	ground spoiler actuator
GSCM	ground spoiler control module
GSE	ground support equipment
GUI	graphical user interface
H	
HAAO	high altitude airport operation
HDG	heading
HF	high frequency
HID	high-intensity discharge
HLEIF	high load event indication function
HLSL	high lift selector lever
HMU	health management unit
HOR	hold open rod
HP	high-pressure
HPC	high-pressure compressor
HPD	hydraulic pump depressurization
HPGC	high-pressure ground connection
HPSOV	high-pressure shutoff valve
HPT	high-pressure turbine
HPV	high-pressure valve
HRD	high-rate discharge
HRTDb	high-resolution terrain database
HS	handset
HS	high solid
HSI	horizontal situation indicator
HSTA	horizontal stabilizer trim actuator
HSTS	horizontal stabilizer trim system
HUD	head-up display
HUDS	head-up display system

I	
I/O	input/output
IAMS	integrated air management system
IAS	indicated airspeed
IASC	integrated air system controller
IBIT	initiated built-in test
IBR	integral bladed rotor
ICAO	International Civil Aviation Organization
ICCP	integrated cockpit control panel
ICDU	integrated control display unit
ICU	inerting control unit
ICU	isolation control unit
ICV	isolation control valve
IFE	in-flight entertainment system
IFEC	in-flight entertainment and connectivity system
IFIS	integrated flight information system
IFPC	integrated fuel pump and control
IFS	information landing system
IFS	inner fixed structure
IGN	ignition
IGV	inlet guide vane
IGVA	inlet guide vane actuator
IIM	inceptor interface module
IIV	inlet isolation valve
ILS	instrument landing system
IMA	integrated modular avionics
IMS	information management system
INT	intermittent
IOC	input/output concentrator
IOM	input/output module
IP	information provider
IPC	integrated processing cabinet
IPCKV	intermediate pressure check valve
IPS	inches per second
IPS	integrated processing system
IRCV	inlet return check valve
IRS	inertial reference system
IRU	inertial reference unit
ISI	integrated standby instrument
ISM	input signal management
ISPS	in-seat power supply
ISPSS	in-seat power supply system
ITT	interturbine temperature
J	
JOSV	journal oil shuttle valve
L	
L/S	lube/scavenge
LAN	local area network
LBIT	landing built-in test
LCD	life cycle data
LCT	line current transformer
LED	light-emitting diode
LLU	LED lighting unit
LGCL	landing gear control lever
LGCV	landing gear control valve
LGIS	landing gear indicating system
LGSCU	landing gear and steering control unit
LGSV	landing gear selector valve

LOC	localizer
LOP	low oil pressure
LOPA	layout of passenger area
LP	low-pressure
LPC	low-pressure compressor
LPSOV	low-pressure shutoff valve
LPT	low-pressure turbine
LRD	low-rate discharge
LRM	line replaceable module
LRU	line replaceable unit
LSK	line select key
LSOP	lubrication and scavenge oil pump
LV	lower sideband voice
LVDS	low-voltage differential signaling
LVDT	linear variable differential transformer
M	
MAX	maximum
MB	marker beacon
MCDL	motor control data link
MCE	motor control electronic
MCR	minimum control requirement
MCV	mode control valve
MDU	manual drive unit
MEL	minimum equipment list
MES	main engine start
MFK	multifunction keyboard panel
MFP	multifunction probe
MFS	multifunction spoiler
MFW	multifunction window
MIXTS	mix manifold temperature sensor
MKP	multifunction keyboard panel
MLG	main landing gear
MLW	maximum landing weight
MMEL	Master Minimum Equipment List
MOF	main oil filter
MOT	main oil temperature
MPP	maintenance planning publication
MPSOV	minimum pressure-shutoff valve
MRW	maximum ramp weight
MSV	mode select valve
MTD	master time and date
MTO	maximum rated takeoff
MTOW	maximum takeoff weight
MWW	main wheel well
MZFW	maximum zero fuel weight
N	
NA	not activated
NACA	National Advisory Committee for Aeronautics
ND	nosedown
NCD	no computed data
NCG	network communication gap
NCU	network control unit
NDB	non-directional beacon
NDO	network data object
NEA	nitrogen-enriched air
NEADS	nitrogen-enriched air distribution system
NLG	nose landing gear
NO PED	no personal electronic device

NPRV	negative pressure-relief valve
NU	noseup
NWS	nosewheel steering
NVM	non-volatile memory
O	
OAT	outside air temperature
OBB	outboard brake
OBIGGS	onboard inlet gas generation system
OC	overcurrent
OCM	oil control module
OCM	option control module
ODI	overboard discharge indicator
ODL	onboard data loader
ODM	oil debris monitor
OEA	oxygen-enriched air
OEM	original equipment manufacturer
OF	overfrequency
OFV	outflow valve
OMS	onboard maintenance system
OMS IMA	OMS interactive maintenance application
OMSA	onboard maintenance system application
OMST	onboard maintenance system table
OPAS	outboard position asymmetry sensor
OPU	overvoltage protection unit
OSP	opposite-side pressure
OSS	overspeed/shutdown solenoid
OT	other traffic
OV	overvoltage
OWEE	overwing emergency exit

P	
P&W	Pratt and Whitney
P2	inlet pressure
PA	passenger address
PAX	passenger
PBA	pushbutton annunciator
PBE	protective breathing equipment
PBIT	power-up built-in test
PCE	precooler exhaust
PCE	precooler exit
PCU	power control unit
PDF	portable document format
PDOS	power door operating system
PDPS	pack pressure differential sensor
PDL	permitted damage limits
PDS	power distribution system
PDTS	pack discharge temperature sensor
PDU	power drive unit
PED	personal electronic device
PEM	power environment module
PEV	pressure equalization valve
PFCC	primary flight control computer
PFD	primary flight display
PFS	post flight summary
PHMU	prognostics and health management unit
PIC	peripheral interface controller
PIC	processor-in-command
PIFS	pack inlet flow sensor
PIM	panel interface module

PIPS	pack inlet pressure sensor
PLD	programmable logic device
PLD	proportional lift dump
PMA	permanent magnet alternator
PMA	program manager application
PMAG	permanent magnet alternator generator
PMG	permanent magnet generator
POB	power off brake
POB	pressure off brake
POR	point of regulation
PPM	power producing module
PPT	pedal position transducer
PRAM	prerecorded announcement and message
PRSOV	pressure-regulating shutoff valve
PRV	pressure-regulating valve
PRV	pressure-relief valve
PS	passenger service
PS	pressure sensor
PSA	print server application
PSE	principal structural element
PSU	passenger service unit
PSUC	passenger service unit controller
PT	proximate traffic
PT	pressure transducer
P_t	total pressure
PTS	pack temperature sensor
PTT	push-to-talk
PTU	power transfer unit
PTY	priority

PVT	position, velocity, time
PWM	pulse width modulation
Q	
QAD	quick attach/detach
QEC	quick engine change
R	
RA	radio altimeter
RA	resolution advisory
RAM	receiver autonomous integrity monitoring
RAD	radio altitude
RARV	ram air regulating valve
RAT	ram air turbine
RDC	remote data concentrator
RDCP	refuel/defuel control panel
REL	relative altitude
REO	repair engineering order
RET	retracted
REU	remote electronic unit
RF	radio frequency
RFAN	recirculation fan
RGA	rotary geared actuator
RGC	RAT generator control
RIPS	recorder independent power supply
RIU	radio interface unit
RLC	RAT line contactor
RMA	remote maintenance access
RMS	radio management system
ROLS	remote oil lever sensor
ROV	redundant overvoltage

RPA	rudder pedal assembly
RPM	revolutions per minute
RSA	report server application
RSP	reversion switch panel
RTA	receiver-transmitter antenna
RTD	resistance temperature device
RTD	resistive thermal device
RTL	ready-to-load
RTO	rejected takeoff
RTS	return to service
RTSA	radio tuning system application
RVDT	rotary variable differential transformer
S	
SAL	specific airworthiness limitation
SAT	static air temperature
SATCOM	satellite communication
SAV	starter air valve
SB	service bulletin
SBAS	satellite-based augmentation system
SBIT	start-up BIT
SCV	surge control valve
SCV	steering control valve
SEB	seat electronics box
SELCAL	selective calling
SFCC	slat/flap control computer
SFCL	slat/flap control lever
SFCP	slat/flap control panel
SFECU	slat/flap electronic control unit
SFIS	standby flight instrument system
SFV	safety valve
SLS	slow load-shed
SMS	surface management system
SOV	solenoid operated valve
SOV	shutoff valve
SPCV	supply pressure check valve
SPDS	secondary power distribution system
SPDT	single pole double throw
SPKR	speaker
SPM	seat power module
SSC	sidestick controller
SSD OML	solid-state onboard media loader
SSI	structural significant item
SSEC	static source error connection
SSPC	solid-state power controller
SSPC-CB	solid-state power controller circuit breaker
SSRPC	solid-state remote power controller
SUA	special use airspace
SVA	stator vane actuator
SVS	synthetic vision system
T	
T/M	torque motor
T/R	thrust reverser
T2	inlet temperature
TA	traffic advisory
TACKV	trim air check valve
TAPRV	trim air pressure-regulating valve
TASOV	trim air shutoff valve
TAT	total air temperature

TAV	trim air valve
TAWS	terrain awareness and warning system
TAWSDb	terrain awareness and warning system database
TCA	turbine cooling air
TCAS	traffic alert and collision avoidance system
TCB	thermal circuit breaker
TCDS	type certificate data sheet
TCF	terrain control valve
TDR	transponder
TCV	temperature control valve
TEC	turbine exhaust case
TED	trailing edge down
TEU	trailing edge up
TFTP	trivial file transfer protocol
TIC	turbine inlet case
TIC	turbine intermediate case
TIV	temperature inlet valve
TIV	temperature isolation valve
TLA	throttle lever angle
TLC	TRU line contactor
TLD	time limited dispatch
TOGA	takeoff/go-around
TPIS	tire pressure indicating system
TPM	TAWS processing module
TPM	tire pressure module
TPMA	terrain processing module application
TPMU	tire pressure monitoring unit
TPS	tire pressure sensor
TPSA	terrain processing system application

TQA	throttle quadrant assembly
TRAS	thrust reverser actuation system
TRU	transformer rectifier unit
TSC	TRU start contactor
TSFC	thrust specific fuel consumption
TSM	trip status monitor
TSO	technical standard order
TSS	traffic surveillance system
TTG	time-to-go
TTP	time-triggered protocol
TWIP	terminal weather information for pilot
U	
UART	universal asynchronous receiver transmitter
UBMF	usage-based monitoring function
UF	underfrequency
ULB	underwater locator beacon
UPLS	ultrasonic point level sensors
USB	universal serial bus
UTC	universal time coordinated
UV	upper sideband voice
UV	ultraviolet
UV	undervoltage
V	
VAC	volts alternating current
VDC	voltage direct current
VDL	VHF data link
VDLM	VHF data link mode
VENTS	ventilated temperature sensor
VFG	variable frequency generator

VFGOOHX	variable frequency generator oil/oil heat exchanger
VGMD	vacuum generator motor drive unit
VHF-NAV	VHF navigation
VID	visible impact damage
VL	virtual link
VLAN	virtual local area network
VNAV	vertical navigation
VOC	volatile organic compounds
VOR-VHF	VHF omnidirectional radio
VORV	variable oil reduction valve
VPA	video passenger announcement
VSD	vertical situation display
VSPD	V-speed
VSWR	voltage standing-wave radio
VTU	video transmission unit
W	
WAI	wing anti-ice
WAP	wireless access point
WAIS	wing anti-ice system
WAITS	wing anti-ice temperature sensor
WAIV	wing anti-ice valve
WBV	windmill bypass valve
WIPC	windshield ice protection controller
WL	waterline
WOFFW	weight-off-wheels
WOW	weight-on-wheels
WPS	words-per-second
WS	wing situation
WSA	web server application
WST	wheel speed transducer
WTBF	wing-to-body fairing
WWHS	windshield and side window heating system
WWS	waste water system
WWSC	water and waste system controller
WXR	weather radar
Z	
ZB	zone box
ΔP	differential pressure

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MODULE 5: POWER PLANT - LIST OF CHANGES

The following table details the changes applied to this revision:

ATA NAME AND NUMBER	CMUI NUMBER	CHANGES APPLIED
Front Page	CS130-21.07-06-00-121418	Version number and CMUI number
Acronyms and Abbreviations	CS130-21.07-06-00-121418	Deleted BCT (bus tie contactor current transformer) and added MKP (multifunction keyboard panel) (pges ii and viii)
List of Changes	CS130-21.07-06-00-121418	New CMUI
21 Environmental Control	CS130-21.07-06-00-121418	No change
28 Fuel	CS130-21.07-06-00-121418	Added new Fuel Safety section (pges 28-2 to 28-13)
		Updated collector tank area (pge 28-21)
		Updated graphic (pges 28-31, 28-33, 28-35)
		Added maximum suction pressure (pge 28-36)
		Updated graphic (pges 28-37, 28-39, 28-71, 28-87, 28-101, 28-103, 28-105)
		Added caution (pge 28-106)
		Updated graphic (pges 28-107, 28-109, 28-111)
30 Ice and Rain Protection	CS130-21.07-06-00-121418	Moved System Test before CAS Messages (pges 30-30 to 30-35)
36 Pneumatic	CS130-21.07-06-00-121418	No change
47 Liquid Nitrogen (Fuel Tank Inerting System)	CS130-21.07-06-00-121418	No change
49 Airborne Auxiliary Power	CS130-21.07-06-00-121418	Updated graphic (pges 49-9, 49-13)
		Added note and caution (pge 49-10)
		Updated text (pge 49-14)
		Updated graphic (pges 49-15, 49-23, 49-97)
71 Power Plant	CS130-21.07-06-00-121418	No change
72 Engine	CS130-21.07-06-00-121418	Updated graphic (pge 72-3)
73 Engine Fuel and Control	CS130-21.07-06-00-121418	Changed throttle lever angle to thrust lever angle (pge 73-43)
74-80 Ignition and Starting	CS130-21.07-06-00-121418	No change

ATA NAME AND NUMBER	CMUI NUMBER	CHANGES APPLIED
75 Engine Air Systems	CS130-21.07-06-00-121418	No change
76 Engine Controls	CS130-21.07-06-00-121418	No change
77 Engine Indicating	CS130-21.07-06-00-121418	Updated graphic; added indicator for fan drive shaft dimple location, and view forward looking aft (pge 77-43)
		Added new Fan Trim Balance Using the PW 1500G Trim Balance Helper section with graphics (pges 77-58 to 77-69)
78 Exhaust	CS130-21.07-06-00-121418	Updated graphic (pge 78-41)
		Updated text (pge 78-42)
		Changed ICV control solenoid to ICU isolation valve control solenoid (pge 78-44)
		Changed valve to unit in figure title (pge 78-45)
		Changed DCV energized to DCU energized in heading and figure title (pges 78-48, 78-49)
		Changed DCV control solenoid to DCU control solenoid (pge 78-52)
		Changed ICV control solenoid to ICU control solenoid (pge 78-54)
		Changed ICV and DCV enable solenoids to ICU and DCU control solenoids; ICV solenoid to ICU solenoid; DCV solenoid to DCU solenoid; added information for maximum reverse thrust (pge 78-56)
79 Oil	CS130-21.07-06-00-121418	Added text for oil level and updated caution (pge 79-4)
		Updated graphic (pge 79-23)
		Updated tables in graphic (pge 79-37)
		Changed 47°C (117°F) to 49°C (120°F) (pge 79-38)
		Updated EICAS on graphic (pge 79-39)
		Updated graphic (pge 79-41)

ATA 21 - Environmental Control

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ENVIRONMENTAL CONTROL - CHAPTER BREAKDOWN

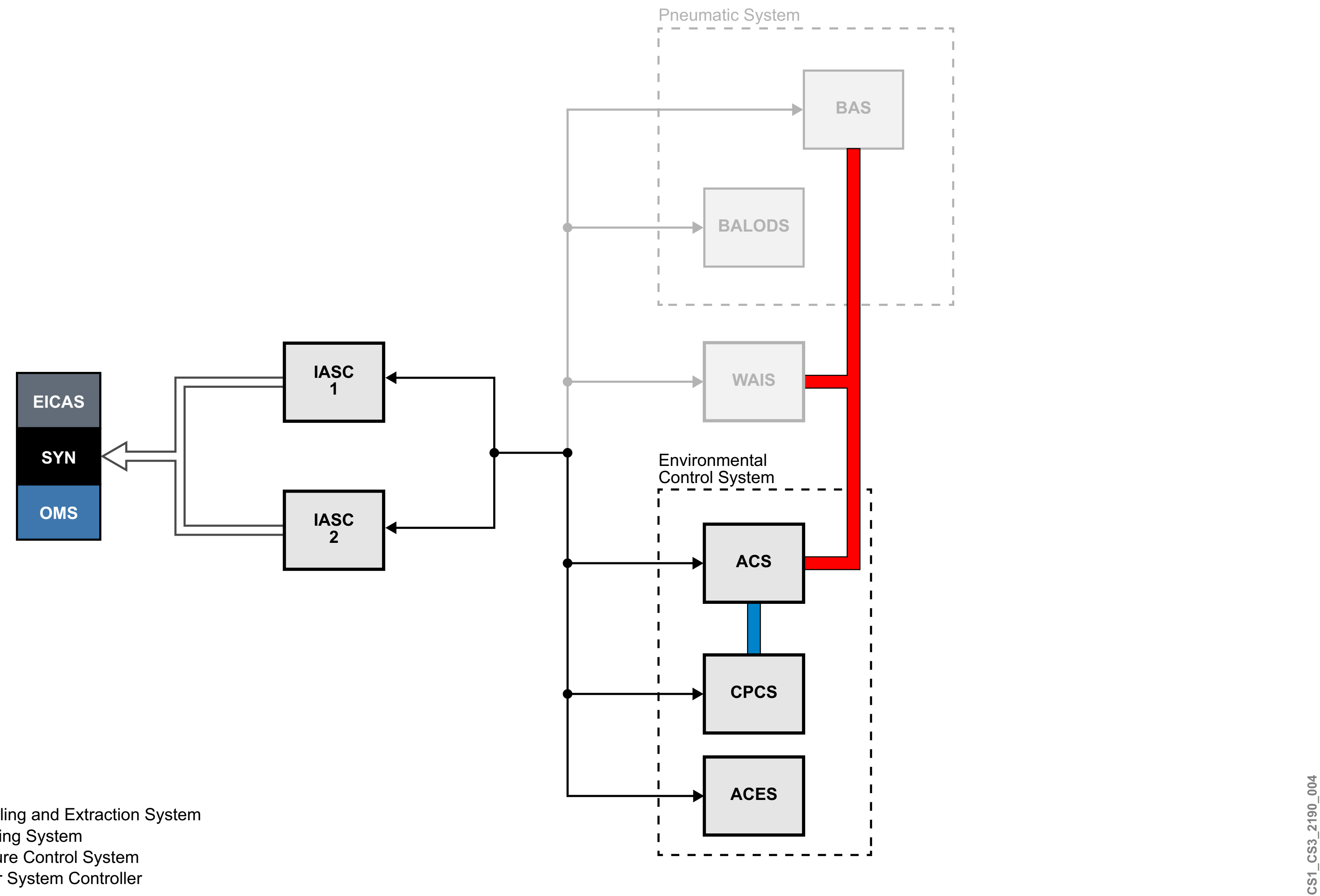
21-90 INTEGRATED AIR SYSTEM CONTROLLER

GENERAL DESCRIPTION

Control and monitoring of the environmental control system (ECS) is provided by two integrated air system controllers (IASCs). The environmental control system includes:

- Air conditioning system
- Avionics cooling and extraction system
- Cabin pressure control system

The IASCs provide system information to the engine indication and crew alerting system (EICAS), the AIR synoptic page, and the onboard maintenance system (OMS).



LEGEND

Bleed Air
Cold Air
ACES Avionics Cooling and Extraction System
ACS Air Conditioning System
CPCS Cabin Pressure Control System
IASC Integrated Air System Controller

Figure 1: Integrated Air System Controller

COMPONENT LOCATION

The IASCs are located in the mid equipment bay on the lower forward shelf.

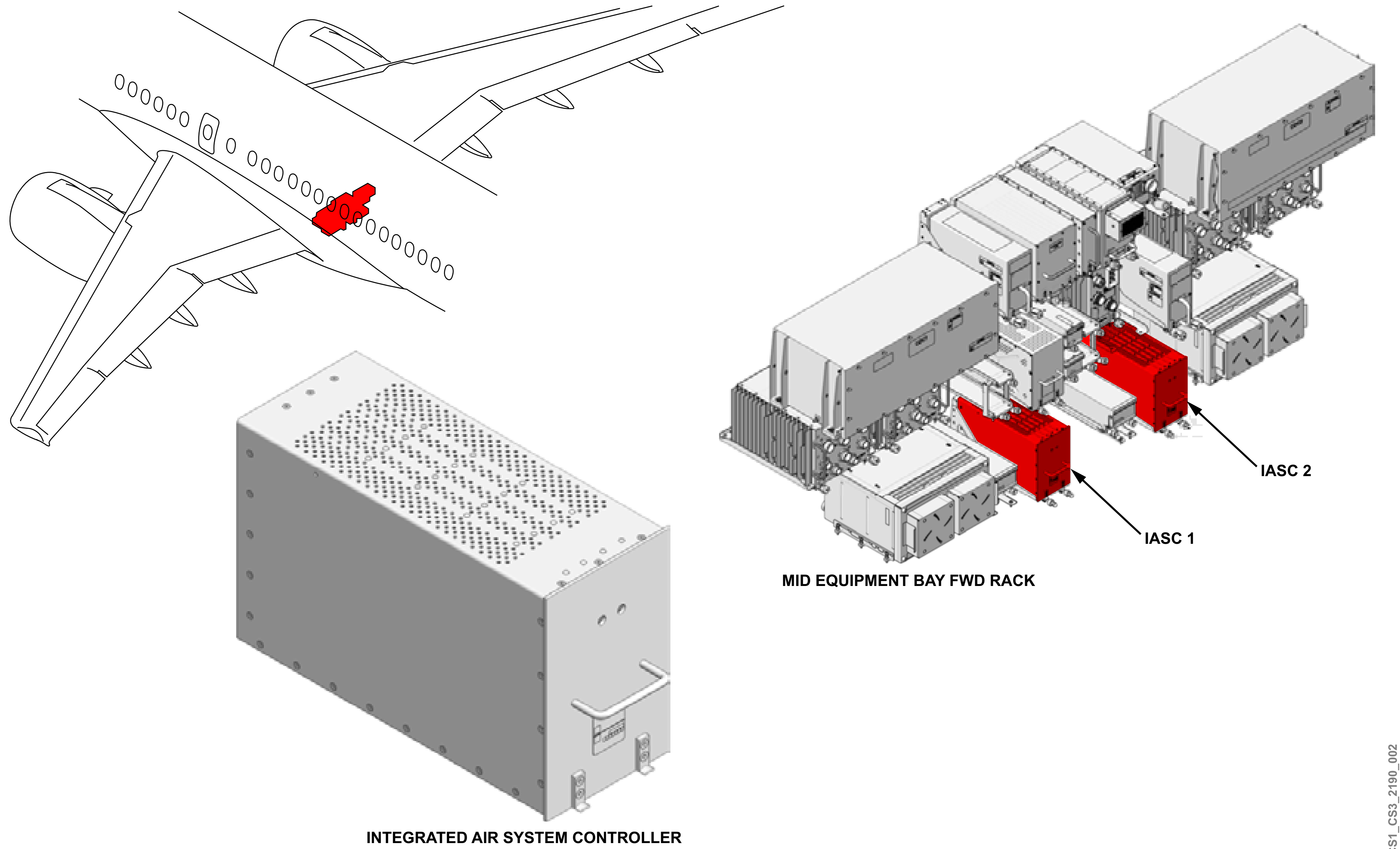
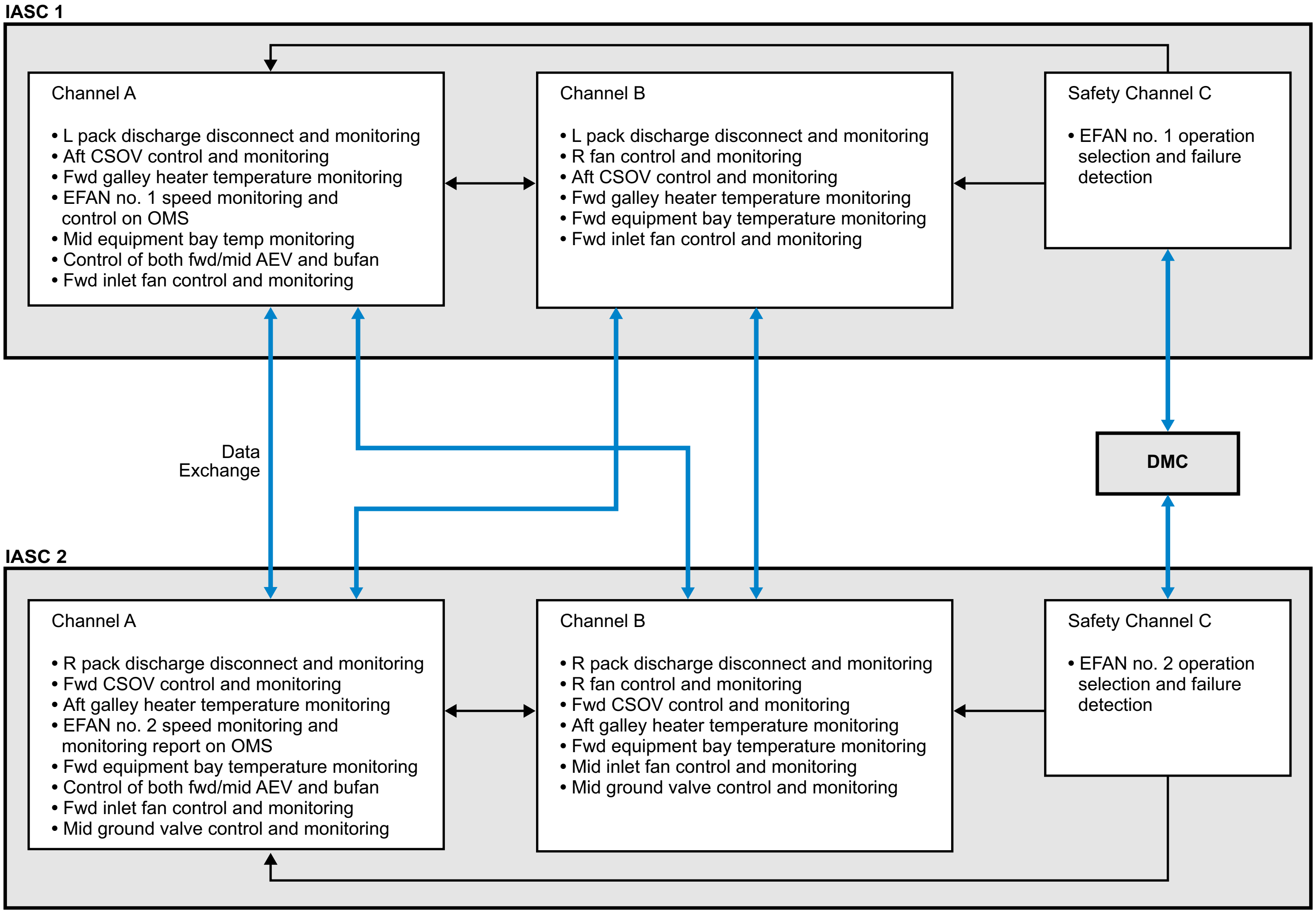


Figure 2: Integrated Air System Controller Location

DETAILED DESCRIPTION

The IASC is a fully redundant controller. In case of failure of one channel, the other channel can control the ECS without loss of any function. The AIR SYSTEM FAULT advisory message is displayed for the failure of a channel.

The IASC safety channel does an additional monitoring of all the most critical functions of the air systems. In the cooling and temperature control system, it monitors for overheat on temperature sensors and reports to the data concentrator unit module cabinet (DMC). In the avionics cooling and extraction system, it controls the extraction fans (EFAN) for the air extraction system, and reports to the DMC.



LEGEND

AEV	Avionics Exhaust Valve
CSOV	Cargo Shutoff Valve
DMC	Data Concentrator Module Cabinet
EFAN	Exhaust Fan
IASC	Integrated Air System Controller
OMS	Onboard Maintenance System
↔	ARINC 429

CS1_CS3_2190_006

Figure 3: Integrated Air System Controller Detailed Description

MONITORING AND TESTS

The following page provides the INFO messages associated with the integrated air system controller:

INFO MESSAGES

Table 1: INFO Messages

MESSAGE	LOGIC
21 AIR SYSTEM FAULT - IASC 1A INOP	No data received from IASC1A confirmed 30s and channel power supply is available.
21 AIR SYSTEM FAULT - IASC 1B INOP	No data received from IASC1B confirmed 30s and channel power supply is available.
21 AIR SYSTEM FAULT - IASC 2A INOP	No data received from IASC2B confirmed 30s and channel power supply is available.
21 AIR SYSTEM FAULT - IASC 2B INOP	No data received from IASC2B confirmed 30s and channel power supply is available.
21 AIR SYSTEM FAULT - IASC 1C INOP	No data received from IASC1C confirmed 30s and channel power supply is available.
21 AIR SYSTEM FAULT - IASC 2C INOP	No data received from IASC2C confirmed 30s and channel power supply is available.
21 AIR SYSTEM FAULT - L IASC ARINC INPUT LOSS	No data from DMC1 or from DMC2 received by IASC1.
21 AIR SYSTEM FAULT - R IASC ARINC INPUT LOSS	No data from DMC1 or from DMC2 received by IASC2.

PRACTICAL ASPECTS

ONBOARD MAINTENANCE SYSTEM TEST

The electrical environmental system components can be tested using the IBIT-ECS test on the onboard maintenance system (OMS). Follow the preconditions before starting the test.

CS1_CS3_2190_003

Figure 4: Environmental Control System OMS Page

21-53 FLOW CONTROL SYSTEM

GENERAL DESCRIPTION

The flow control system controls the airflow, airflow rate, and the temperature of bleed air provided into each air conditioning unit. The flow control system is installed on the system feed line, upstream of the air conditioning units. A trim air system is supplied with bleed air downstream of the flow control system. There is one system associated with each of the air conditioning units, installed on the aircraft.

Each flow control system provides:

- Measurement and control of the air flow supplied to the air conditioning packs
- Pack shutoff control
- Pack overheat protection

An optional ozone converter may be installed in the bleed air ducting, upstream of the flow sensing venturi. The ozone converters convert ozone concentrations into oxygen, which prevents high ozone concentrations in the cabin. This improves the quality of air for high altitude or long duration flights. A second option adds a combined ozone and volatile organic compounds (VOC) converter.

The mass airflow into the air conditioning pack is measured by a flow sensor venturi (FSV) and associated sensors. The FSV is used to calibrate and measure the flow into the pack and trim air system. A pack inlet flow sensor (PIFS) provides differential pressure measurement across the flow sensor venturi to the IASC for mass flow computation and flow control valve (FCV) control.

The pack inlet pressure sensor (PIPS) provides pack inlet static pressure to the IASC for mass flow computation and bleed air manifold pressure monitoring. The bleed air manifold pressure is displayed on the AIR synoptic page.

The FCV is powered and controlled by the IASC. The left IASC controls the left FCV and the right IASC controls the right FCV. The IASC positions the FCV butterfly, using a torque motor, to limit the pressure and temperature of the air entering the pack. The FCV is spring-loaded closed. In the event of a power loss, the valve closes. A minimum 15 psi pressure is required to drive the valve in the open position. The maximum regulated pressure is limited to 43.5 +/- 3 psi.

Each FCV is controlled by the associated PACK PBA on the AIR panel in the flight deck. The FCV is normally automatically controlled by the IASC but can be selected closed the L or R PACK PBA.

During flight with reduced passenger loads, the system reduces the flow schedule in order to reduce bleed air consumption. Using the FMS data to capture passenger load, the IASC reduces the flow schedule based on the actual quantity of passengers onboard. The PACK FLOW PBA on the AIR panel, can be used to adjust the flow control to the high flow mode, which is used for smoke and odor clearance as well as increased pack performance.

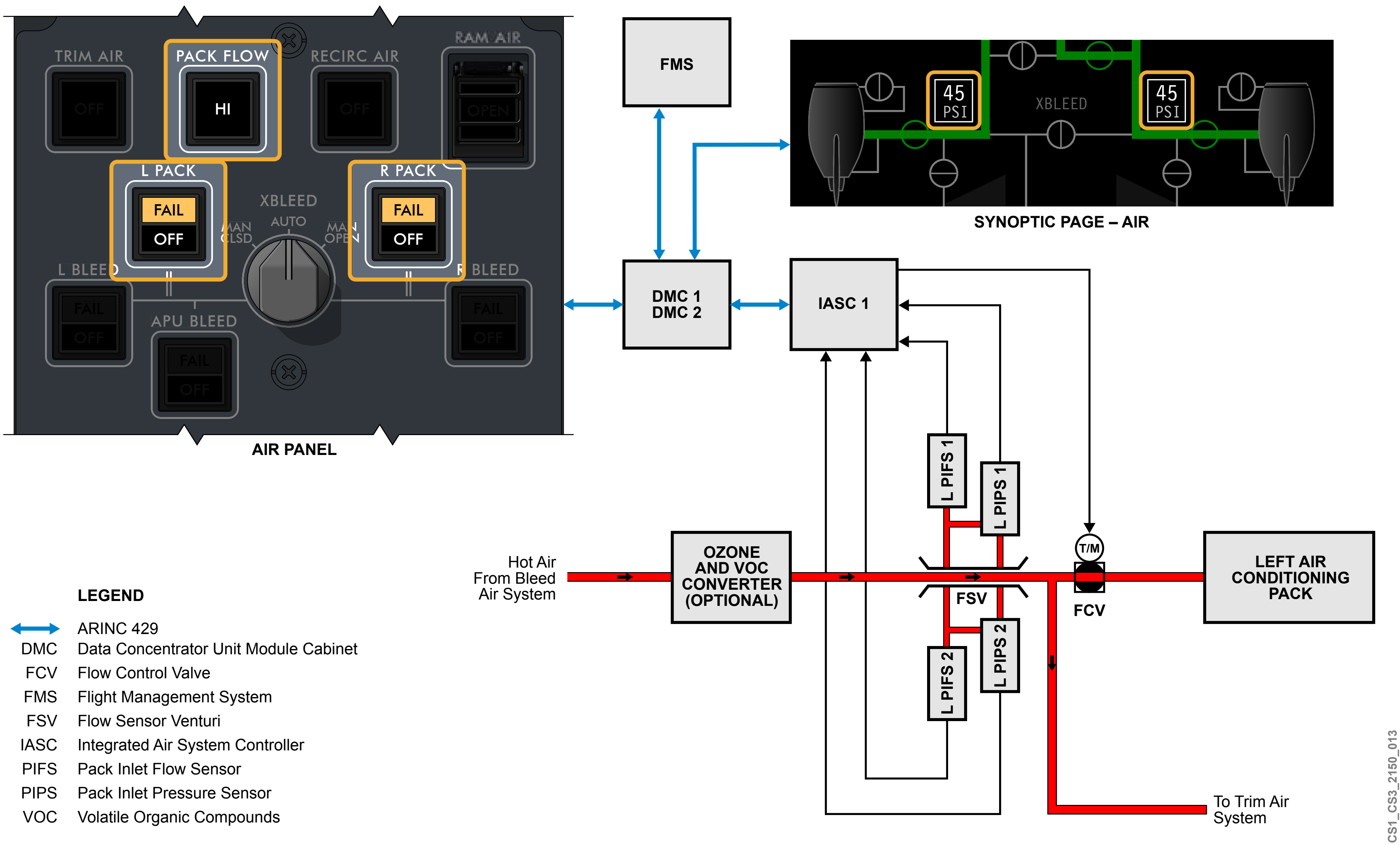


Figure 5: Flow Control System

COMPONENT LOCATION

The flow control system has the following components installed in the wing-to-body fairing (WTBF):

- Flow sensor venturi (FSV)
- Flow control valve (FCV)
- Ozone converter or combined ozone and volatile organic compounds (VOC) converter (Optional)
- Pack inlet pressure sensor (PIPS) (Refer to figure 7)
- Pack inlet flow sensor (PIFS) (Refer to figure 7)

FLOW SENSOR VENTURI

The flow sensor venturi is located in the WTBF.

FLOW CONTROL VALVE

The flow control valve is located in the WTBF.

OZONE CONVERTER/OZONE AND VOC CONVERTER (OPTIONAL)

The ozone converter is installed upstream of the flow sensor venturi.

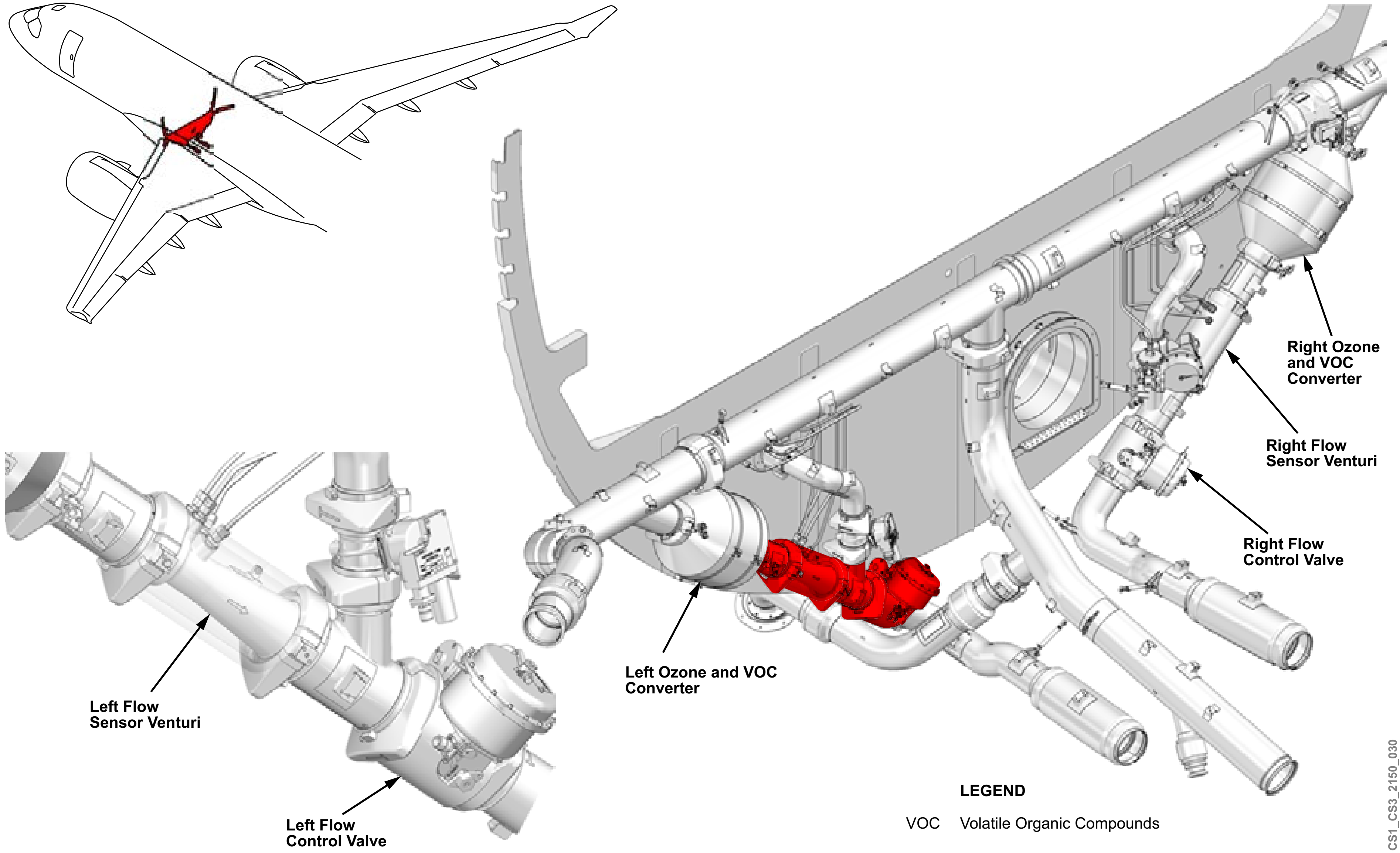


Figure 6: Flow Sensor Venturi, Flow Control Valve, and Ozone Converter Location

PACK INLET PRESSURE SENSORS

The pack inlet pressure sensors (PIPSs) are mounted on the pressure bulkhead in the wing-to-body fairing (WTBF).

PACK INLET FLOW SENSORS

The pack inlet flow sensors (PIFSs) are mounted on the pressure bulkhead in the WTBF.

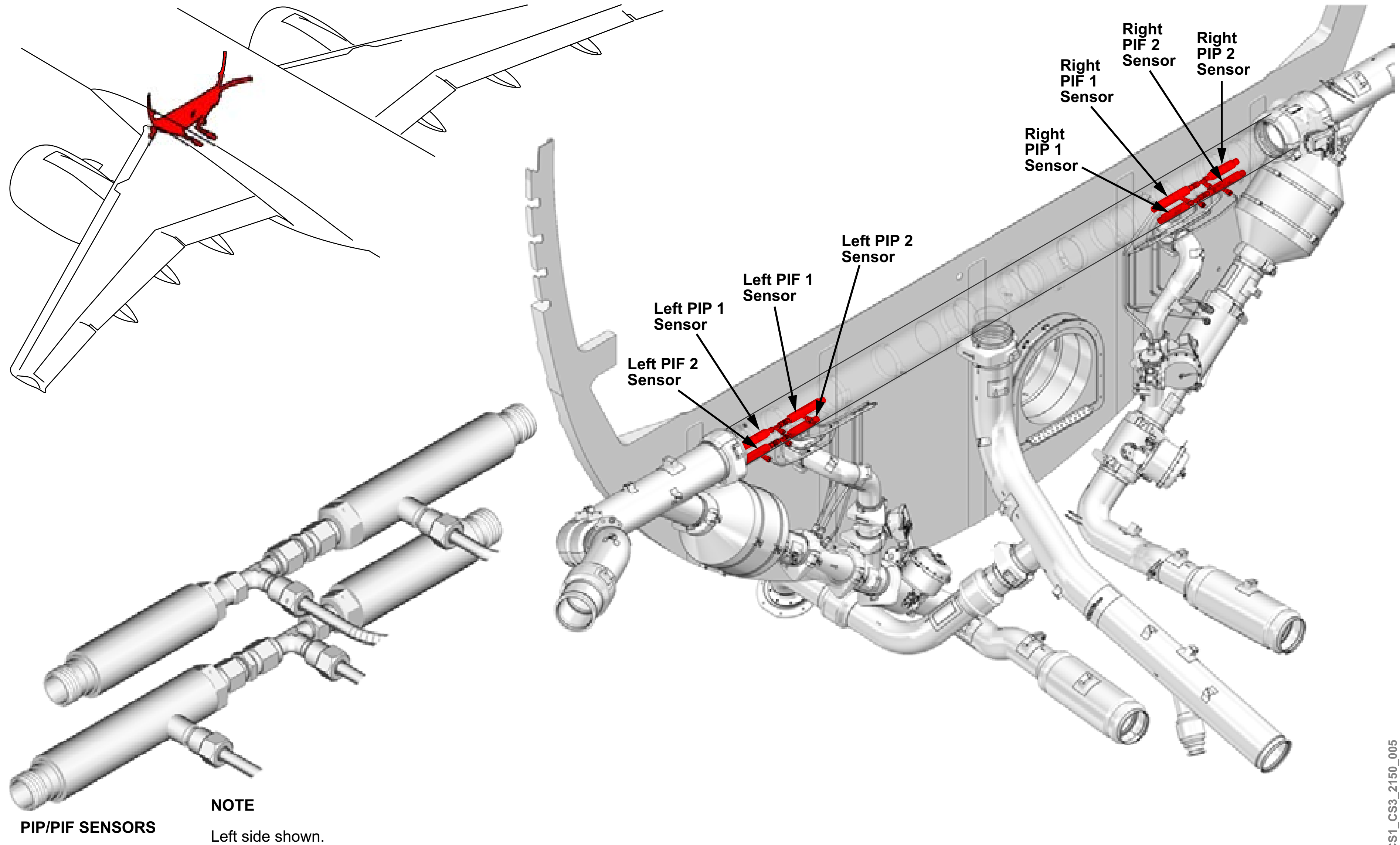


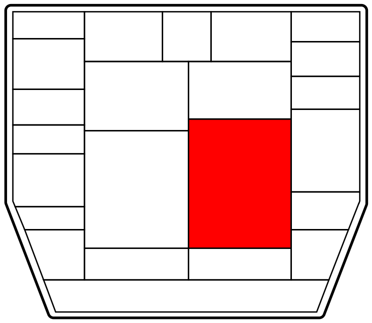
Figure 7: PIP Sensor and PIF Sensor Locations

CONTROLS AND INDICATIONS

CONTROLS

The L and R PACK PBA are used to select the flow control valves closed. An OFF legend on the PACK PBA indicates the valve is closed. The amber FAIL light indicates a pack fault.

The HI FLOW PBA is used to increase the pack flow manually.


OVERHEAD PANEL

AIR PANEL

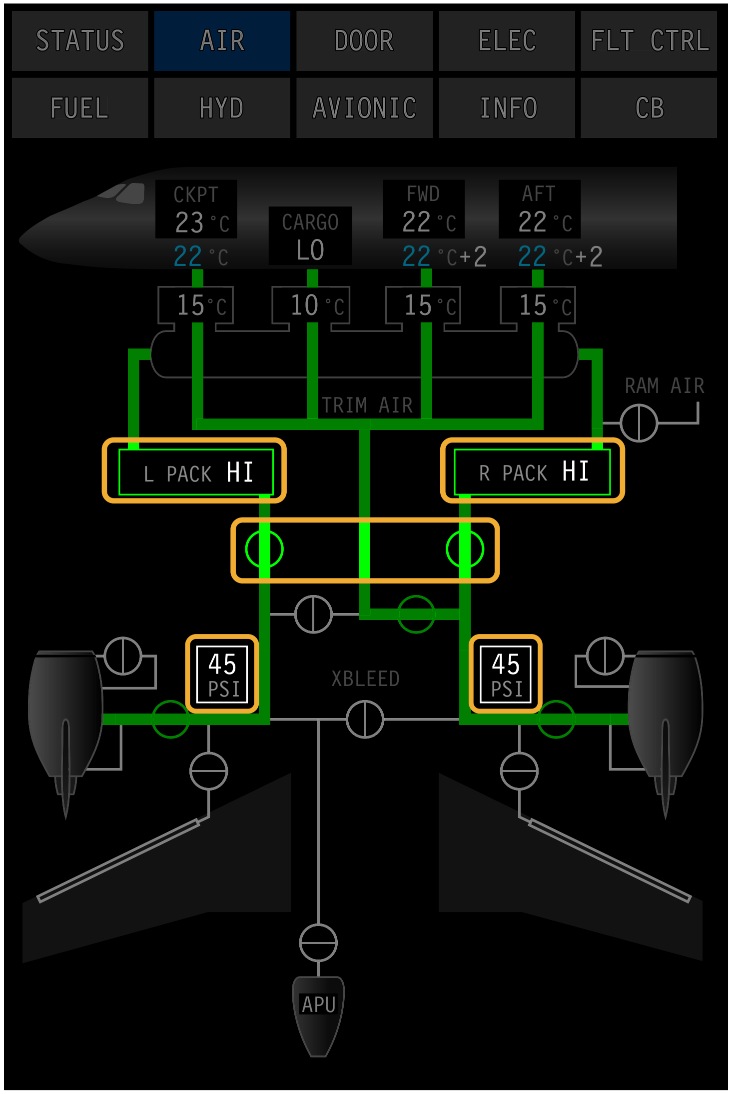
CS1_CS3_2160_006

Figure 8: Flow Control Controls

INDICATIONS

A HI indication appears in the pack symbol on the AIR synoptic page indicating the packs are in HI FLOW mode.

The bleed air manifold pressure is supplied by the PIPS.



SYNOPTIC PAGE – AIR

VALVE POSITION NORMAL OPERATION	
Symbol	**Condition**
	Closed
	Open with flow
	Open with no flow

VALVE POSITION FAIL OPERATION	
Symbol	**Condition**
	Closed
	Open with flow
	Open with no flow
	Invalid

FLOW LINES	
Symbol	**Condition**
	Normal flow
	No flow
	Abnormal flow (overheat, out of range pressure, leak)

CS1_CS3_2120_007

Figure 9: Flow Control Indications

DETAILED DESCRIPTION

FLOW CONTROL NORMAL OPERATION

The flow is automatically controlled by the integrated air system controller (IASC) depending on the system configuration according to preset flow schedules. The FCV opens when the IASC provides a signal to the torque motor on the FCV and there is a minimum of 15 psi available from the bleed air system.

The IASCs use the actual flow measured at the flow sensing venturi and the bleed temperature sensor (BTS) to modulate the FCV to obtain the required flow schedule. During flight with reduced passenger loads, the system automatically reduces the flow schedule in order to reduce bleed air consumption. Using the flight management system (FMS) passenger load data, the IASC reduces the flow schedule based on the actual quantity of passengers onboard.

In the auto mode, the bleed air system uses auxiliary power unit (APU) bleed on the ground and during takeoff. The system switches to engine bleed air based on aircraft configuration. For additional information, refer to ATA 36 - Pneumatic.

On the ground, in dual pack or single pack operation, the APU flow demand is set to 100%, and the flow control logic uses a flow schedule based on:

- Aircraft version (CS100 or CS300)
- Altitude (ranging from -2,000 ft to 14,500 ft)
- Outside air temperature (-50°C to 50°C) (-58°F to 122°F)

In flight, the APU flow demand is set to 100%, and the system regulates the FCV according to the engine flow schedule.

The main operating modes for the system are the following:

- Two bleed sources dual-pack operation
- Two bleed sources single-pack operation
- One bleed source dual-pack operation
- Trim air only during emergency ram air operation

Pull-Up and Pull-Down Sequence

When the APU is supplying bleed air on the ground and the actual zone temperatures vary by more than 10°C from the required temperature, the system automatically goes into pull-up or pull-down mode. Pull-up or pull-down modes use an increased flow schedule.

During pull-up mode, additional heating is supplied by the trim air system to improve the pull-up efficiency.

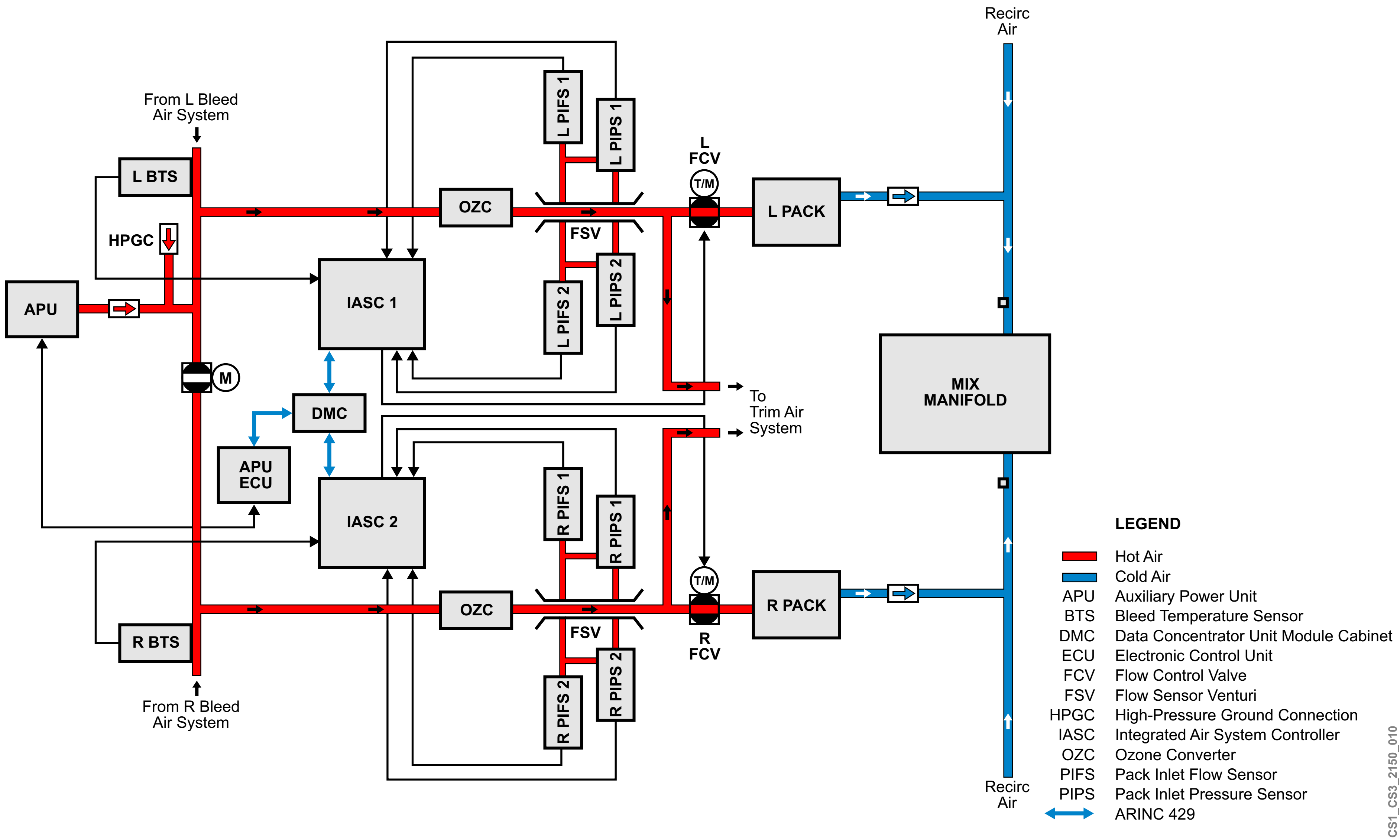


Figure 10: Flow Control Normal Operation

FLOW CONTROL

To perform the flow control, the IASCs use the actual flow measured by:

- Pack inlet pressure sensor (PIPS)
- Pack inlet flow sensor (PIFS)
- Bleed temperature sensor (BTS)

The flow reference is function of:

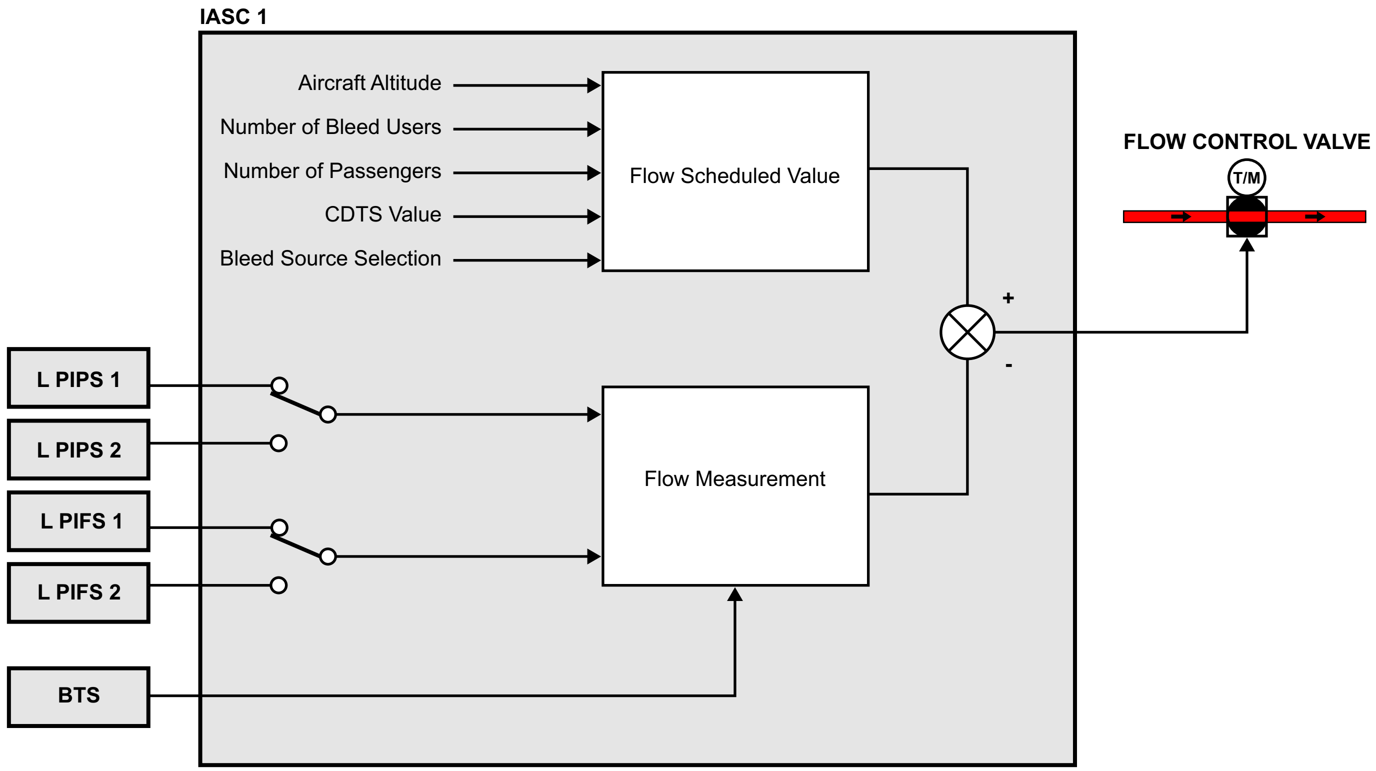
- Number of passengers
- Number of bleed air users
- Aircraft altitude
- Compressor discharge temperature sensor value
- Bleed source selection

The flow requirements take into account the total passenger loads as well as the three cockpit members and five flight attendants.

NOTE

1. In dual-pack operation, with FWD cargo heat in either the LO or HI setting, and the recirculation air on, the flow schedule on the right side is increased to account for the added FWD cargo trim air. When the recirculation air system is off, but the FWD cargo heating is on, then both left and right FCVs are supplied with the same flow.
2. During single pack operation, the aircraft is altitude limited.

The loss of a PIPS, PIFS, or an IASC control channel results in a PACK FAULT advisory message. Normal operation continues with the remaining sensors or the working channel. The loss of both PIPs, PIFs, or IASC control channels for either FCV is indicated by a L or R PACK FAIL caution message and an L(R) PACK PBA FAIL light. The pack must be shutdown by selecting the PACK PBA to OFF.



LEGEND

(red bar)	Bleed Air
BTS	Bleed Temperature Sensor
CDTS	Compressor Discharge Temperature Sensor
IASC	Integrated Air Systems Controller
PIFS	Pack Inlet Flow Sensor
PIPS	Pack Inlet Pressure Sensor

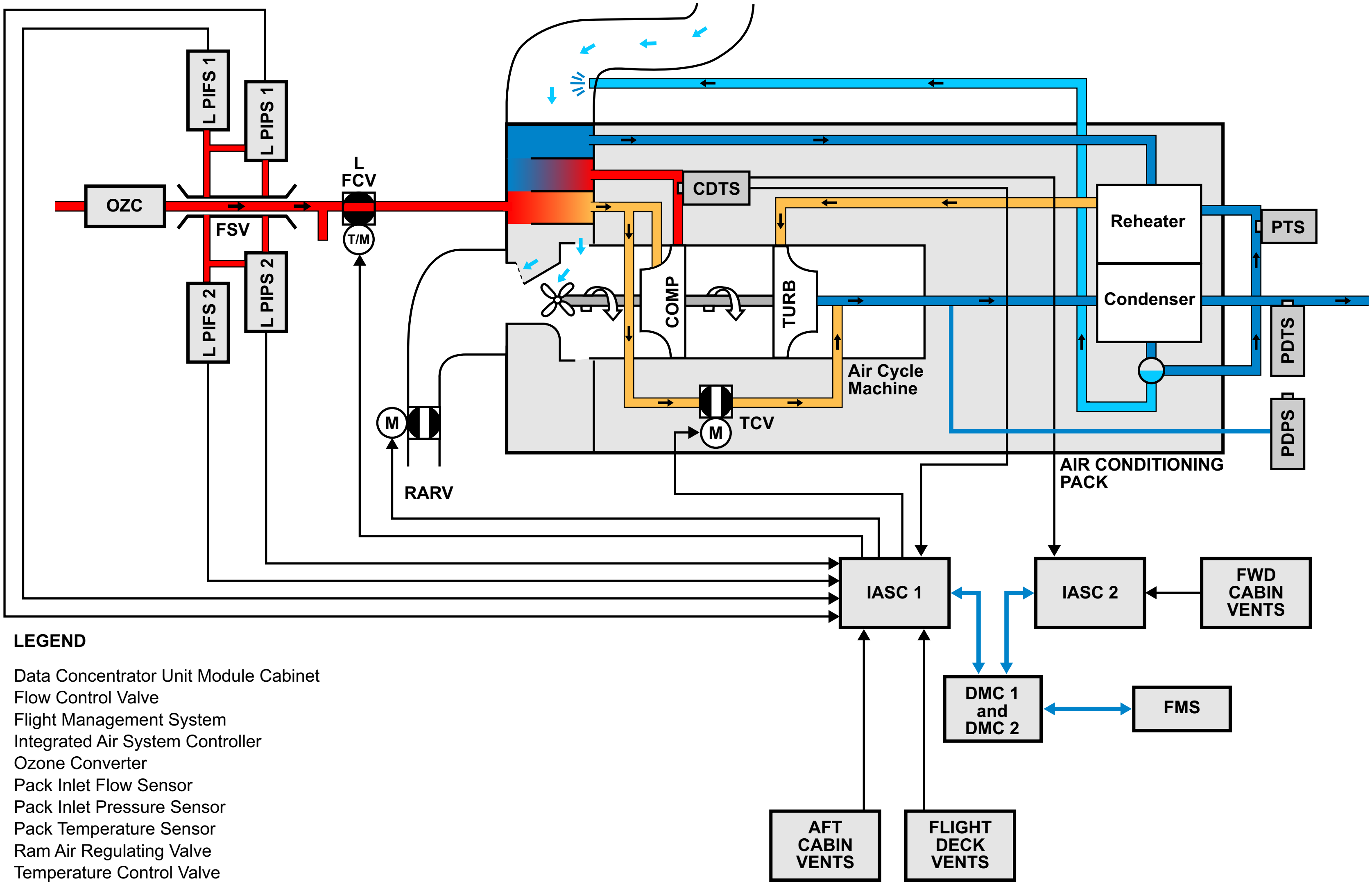
Figure 11: Flow Control Valve

FLOW CONTROL MODES

Flow Priority

When the flow measured is less than 80% of the minimum required flow for 3 minutes in single pack operation, or 90% in dual-pack operation, the flow priority mode is enabled. The flow priority circuit monitors for a low flow supply condition based on operating mode and passenger load as input in the FMS.

The corresponding pack temperature control valve (TCV) is commanded open by the IASC. This allows increased air flow either until the flow target is achieved, or until the flight deck and cabin zone temperatures reach 35°C (95°F). The ram air regulating valve (RARV) opens to maintain the pack temperature. In this mode of operation, the flight deck and cabin zone temperature control are not available as the pack outlet temperature may be higher than the selected value. If the IASC detects a low flow condition for at least 3 minutes, an engine indication and crew alerting system (EICAS) L or R PACK FAIL advisory message is displayed.



LEGEND

DMC	Data Concentrator Unit Module Cabinet
FCV	Flow Control Valve
FMS	Flight Management System
IASC	Integrated Air System Controller
OZC	Ozone Converter
PIFS	Pack Inlet Flow Sensor
PIPS	Pack Inlet Pressure Sensor
PTS	Pack Temperature Sensor
RARV	Ram Air Regulating Valve
TCV	Temperature Control Valve
VENTS	Ventilated Temperature Sensor
↔	ARINC 429

Figure 12: Flow Control Priority

Hi Flow

Whenever the PACK FLOW PBA is selected on the AIR panel, the flow control system switches to the high flow mode. The high flow mode corresponds to the maximum allowable flow for the system and it overrides all other flow control schedules. This mode is used to help evacuate cabin odors, smoke, or to increase the system air conditioning performance. The high flow mode can exchange the cabin air in less than 5 minutes in either dual and single PACK operating modes.

The HI flow mode is also automatically enabled when the auxiliary power unit (APU) is operating on the ground or when the required mix manifold temperatures decrease below 12°C (54°F) or increase above 60°C (140°F).

In flight, if the EMER DEPRESS PBA is selected, or if the cabin altitude increases above 15,000 ft, the flow control automatically switches to high flow mode.

NOTE

During single engine bleed with:

Wing anti-ice on (WAIS)

Approach

Steep approach (CS100 only)

The flow control is limited to the reference flow, and HI flow is not available.

FLOW CONTROL SHUTOFF

The flow control valve is closed when:

- The associated PACK PBA is selected OFF
- Engine start in flight
- Engine start on ground plus 30 seconds
- Ditching PBA pressed
- Bleed leak in pack zone as reported by the bleed air leak detection and overheat system

FLOW CONTROL VALVE FAILURE

When the FCV is commanded open with maximum torque motor current and the PIPS pressure is at least 15 psi and a low flow has been detected, a L or R PACK FAIL caution message is displayed.

The FCV is detected failed open when it is commanded closed and the corresponding flow computed by the IASC is greater than 30 lb/min for more than 30 seconds. The L or R PACK FAIL caution message is displayed. This message is inhibited during emergency ram air mode with trim air ON since the flow through the FSV includes both the FCV and trim air system flow, which can exceed 30 lb/min in this mode.

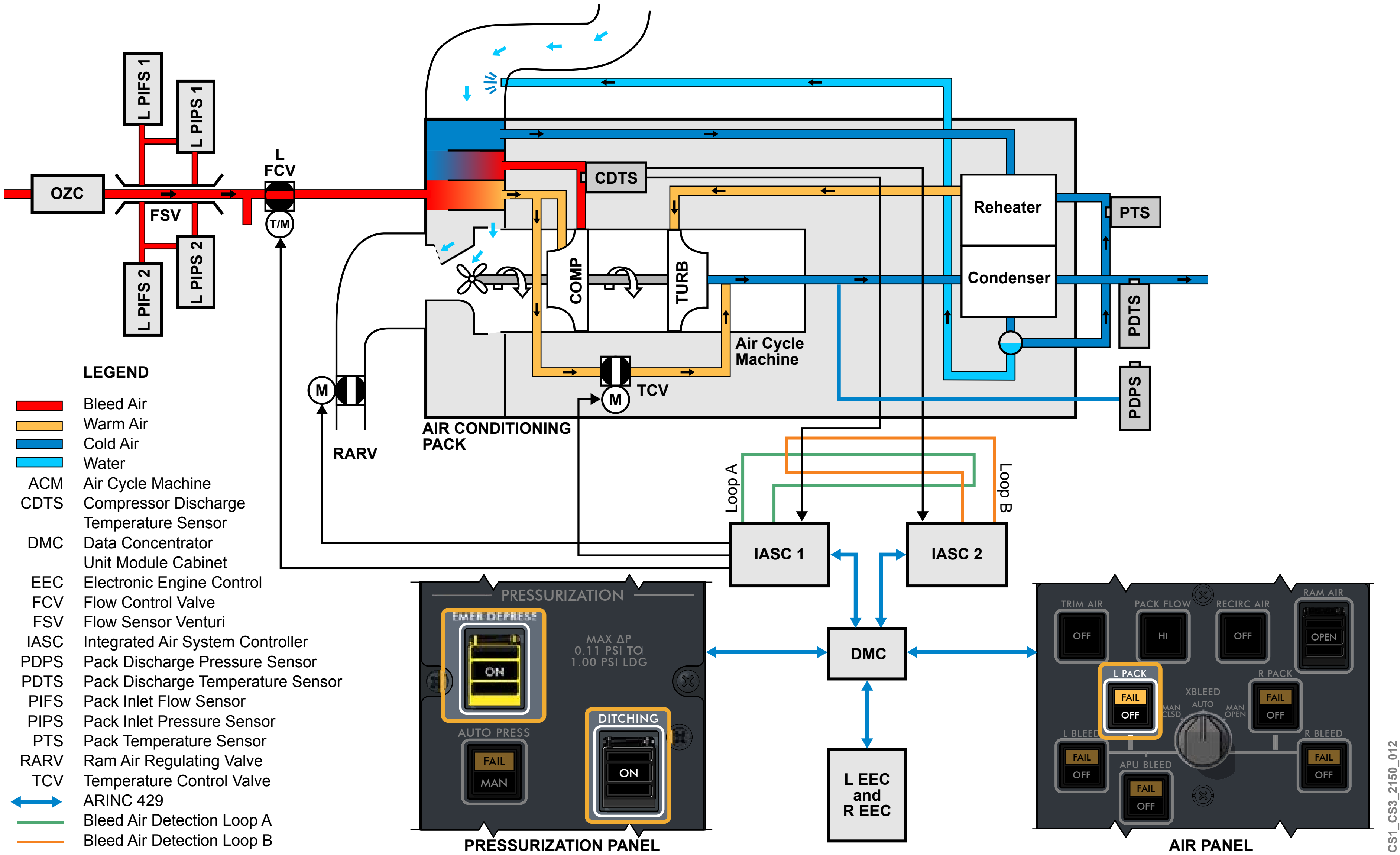


Figure 13: Flow Control Shutoff

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages associated with the flow control system:

CAS MESSAGES

Table 2: CAUTION Message

MESSAGE	LOGIC
L/R PACK FAIL	Left/right pack inoperative (loss of control or monitoring [CDTS or PDTS failure] or pack low flow detected).

Table 3: ADVISORY Message

MESSAGE	LOGIC
PACK FAULT	Loss of redundant or non-critical function for the air conditioning system.

Table 4: STATUS Message

MESSAGE	LOGIC
PACK FLOW HI	CABIN AIR FLOW selected HI and confirmed HI by the controllers.

Table 5: INFO Messages

MESSAGE	LOGIC
21 L PACK FAIL - L FLOW CTRL VLV INOP	Left FCV failed open.
21 R PACK FAIL - R FLOW CTRL VLV INOP	Right FCV failed open.
21 AIR SYSTEM FAULT - L PACK PRESS SNSR REDUND LOSS	Left pack pressure sensor redundant loss (PIFS or PIPS).
21 AIR SYSTEM FAULT - R PACK PRESS SNSR REDUND LOSS	Right pack pressure sensor redundant loss (PIFS or PIPS).
36 L BLEED FAIL - L PACK INLET PRESS SNSR INOP	Left PIPS 1 and PIPS 2 loss.
36 R BLEED FAIL - R PACK INLET PRESS SNSR INOP	Right PIPS 1 and PIPS 2 loss.

PRACTICAL ASPECTS

The flow control valve can be dispatched in the closed position by capping and stowing the electrical connector and then removing the deactivation screw and installing it in the manual position indicator. If the valve is open, the manual position indicator can be used to close the valve using a wrench.

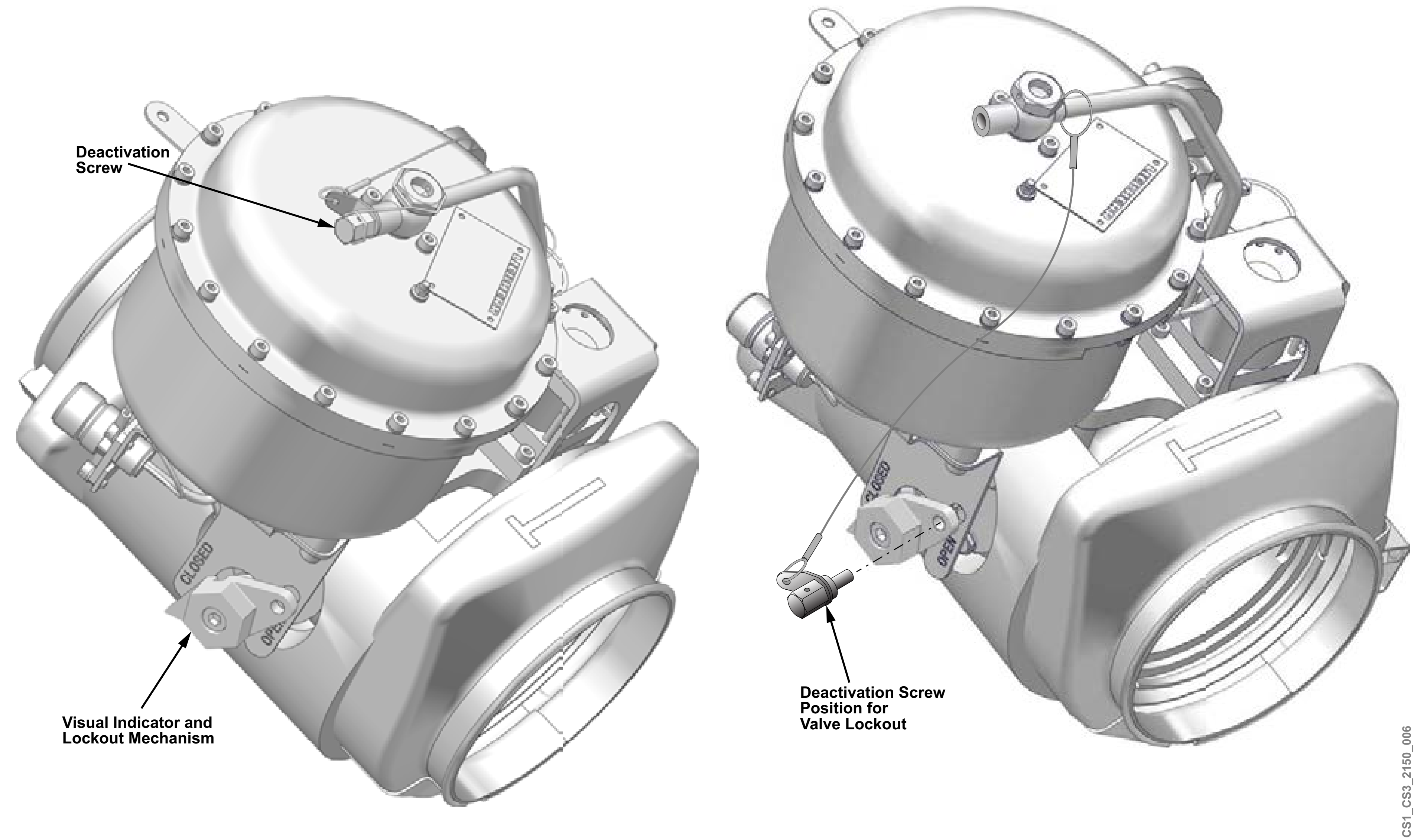


Figure 14: Flow Control Valve Lockout

21-51 COOLING

GENERAL DESCRIPTION

Cooling air is provided from two sources:

- Air conditioning units
- Low-pressure ground cart

The air conditioning units, or packs are available on the ground or in flight. Each pack consists of:

- Air cycle machine
- Dual-heat exchanger
- High-pressure water extraction system
- Temperature control valve

The air cycle machine (ACM) consists of a turbine, compressor, and a fan connected by a common shaft. The ACM shaft is supported by air bearings.

The dual-heat exchanger consists of a primary heat exchanger used to cool air from the bleed air system, and a main heat exchanger used to cool air from the air cycle machine (ACM) compressor discharge.

The high-pressure water extraction system consists of a reheater, condenser, and a water separator. The extracted water is sprayed across the heat exchanger inlet to improve the cooling performance.

The temperature control valve (TCV) bypasses bleed air around the ACM and exhausts it at the turbine outlet. This controls the pack outlet temperature and prevents pack icing.

Each pack is controlled by an integrated air system controller (IASC). IASC 1 controls the left pack, and IASC 2 controls the right pack.

Bleed air enters the pack through the flow control system. The bleed air is cooled by the primary heat exchanger before entering the ACM. The ACM compresses and heats the air. The air passes through the main heat exchanger for further cooling before entering the reheater. The reheater heats the air entering the turbine inlet, where the remaining water is vaporized in order to prevent damage to the ACM.

The air in the reheater starts cooling and any water vapor starts condensing. This air enters the condenser where it is cooled by the cold turbine outlet air. The water and air pass through the water extractor where the water is slung against the walls and is discharged through a line that runs to a spray tube upstream of the dual-heat exchanger. The water sprays across the face of the heat exchanger, increasing its cooling performance.

Air exiting the water extractor enters the turbine inlet where it expands and cools. This air is fed to the mix manifold for distribution to the aircraft.

The water extraction line is monitored by the pack temperature sensor (PTS). If the air is too cold, there is a possibility of ice forming and damaging the ACM. To prevent this, the temperature control valve (TCV) is opened to allow some hot air to bypass the ACM, raising the temperature of the air.

The dual-heat exchanger is cooled by ram air in flight. On the ground, the ACM fan draws air across the heat exchanger when the ACM is operating. The amount of cooling air drawn across the heat exchanger depends on the position of the ram air regulating valve (RARV). The pack temperature is controlled by modulating the RARV and the TCV.

Pack overheat protection is provided by the compressor discharge temperature sensor and the pack discharge temperature sensor. If either sensor detects an overheat condition, a caution message is displayed and the pack is turned off using the PACK PBA which closes the flow control valve.

On the ground, the low-pressure ground connection allows the use of an external ground cart to cool the aircraft. The low-pressure ground connection allows conditioned air to enter the left pack discharge duct downstream of the pack.

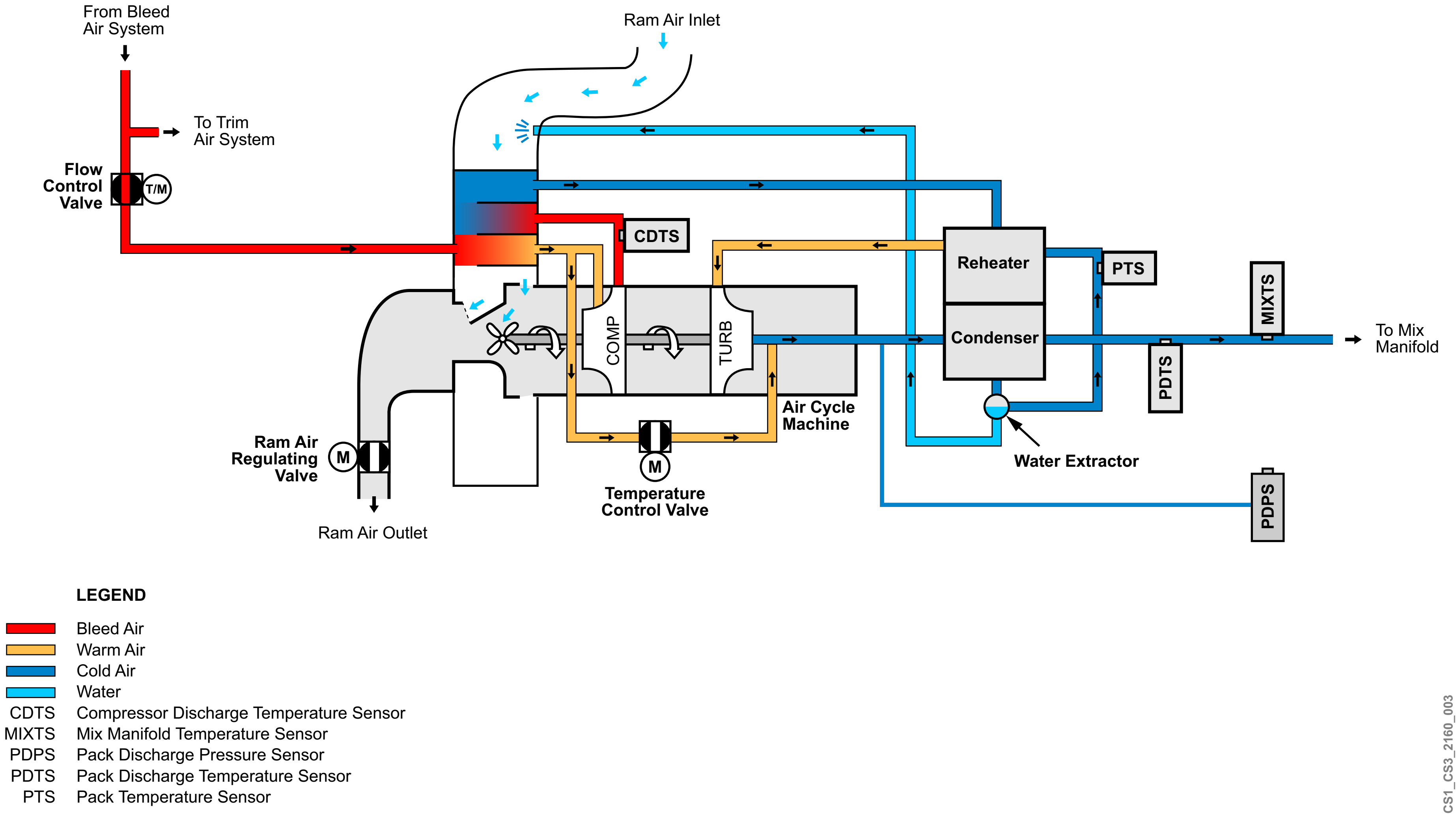


Figure 15: Air Conditioning System

COMPONENT LOCATION

The following cooling system components are located in the wing-to-body fairing (WTBF):

- Air cycle machine
- Dual-heat exchanger
- Reheater condenser
- Temperature control valve
- Compressor discharge temperature sensor (CDTS)
- Pack temperature sensor (PTS)
- Plenum
- Pack discharge temperature sensor (PDTS)
- Ram air regulating valve (RARV)
- RAM air duct
- Water sprayer (Refer to figure 17)
- Water extractor (Refer to figure 17)

The following cooling system component is located in the environmental control system bay:

- Pack discharge pressure sensor (Refer to figure 18)

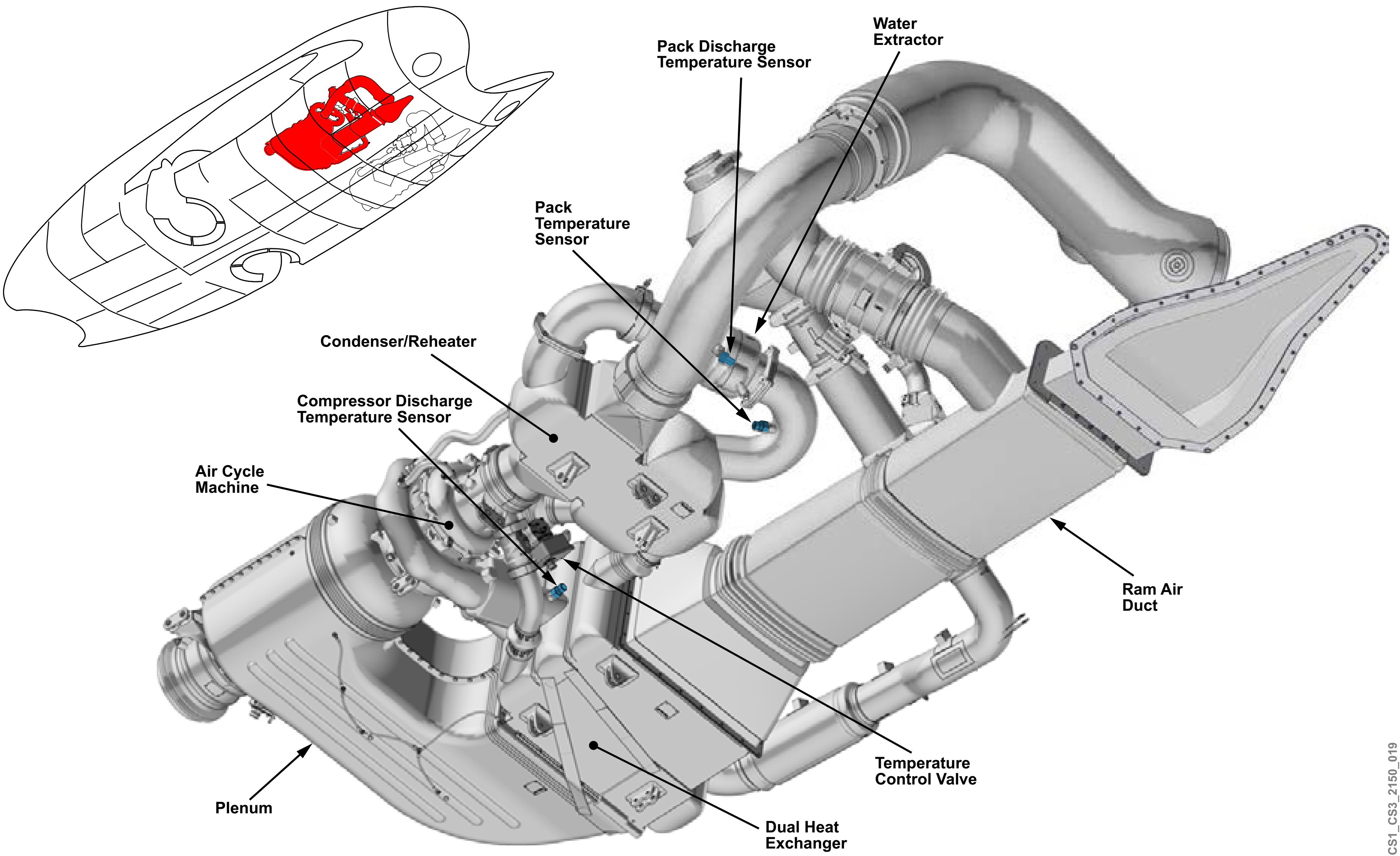


Figure 16: Air Conditioning Components Location

WATER EXTRACTOR

The water extractor is mounted in front of the condenser/reheater.

WATER SPRAYER

The water sprayer is located in the ram air duct forward of the dual-heat exchanger. The sprayer discharge is oriented forward to optimize distribution across the face of the heat exchanger.

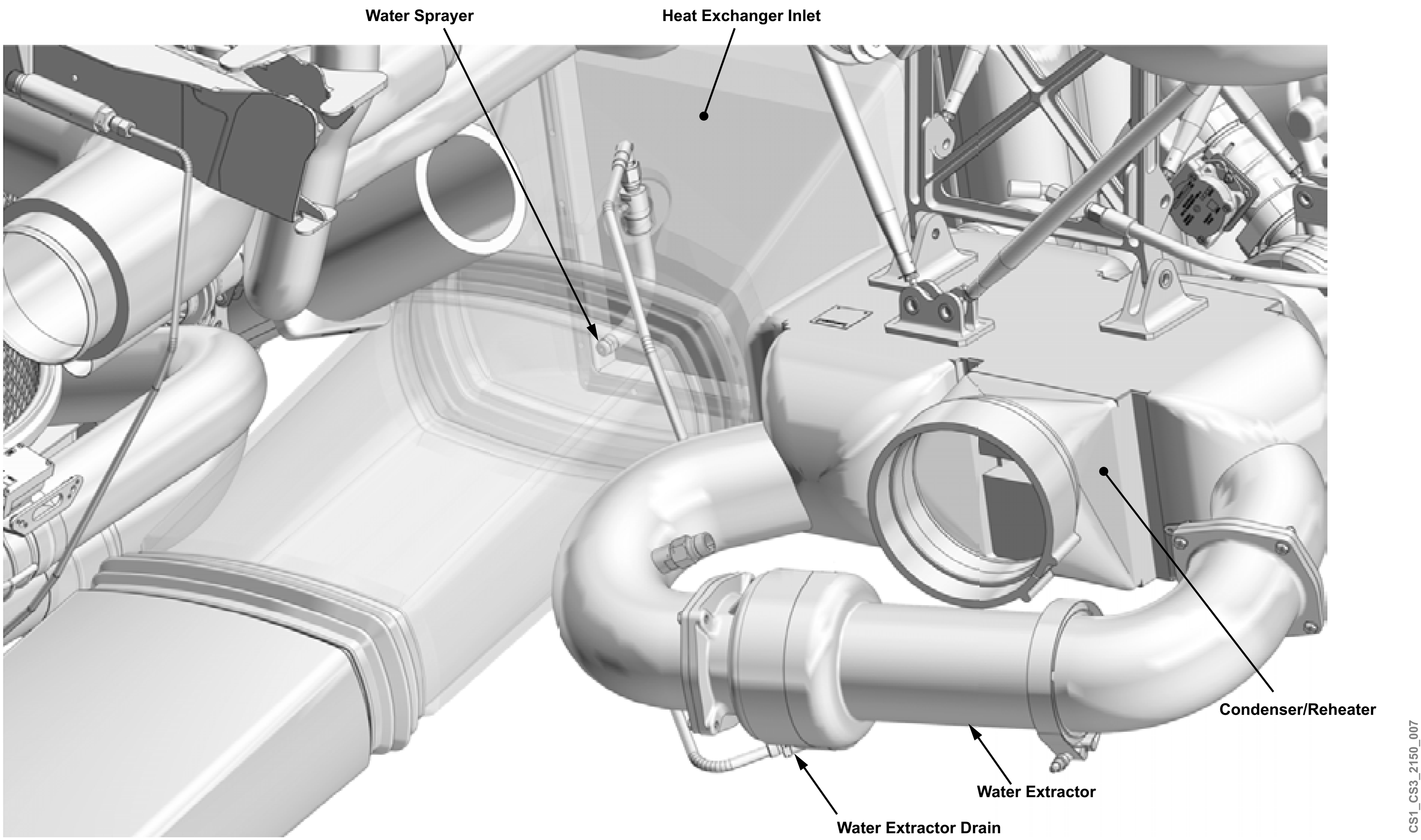


Figure 17: Water Extractor and Water Sprayer

PACK DISCHARGE PRESSURE SENSOR

The pack discharge pressure sensor is located in the environmental control system bay.

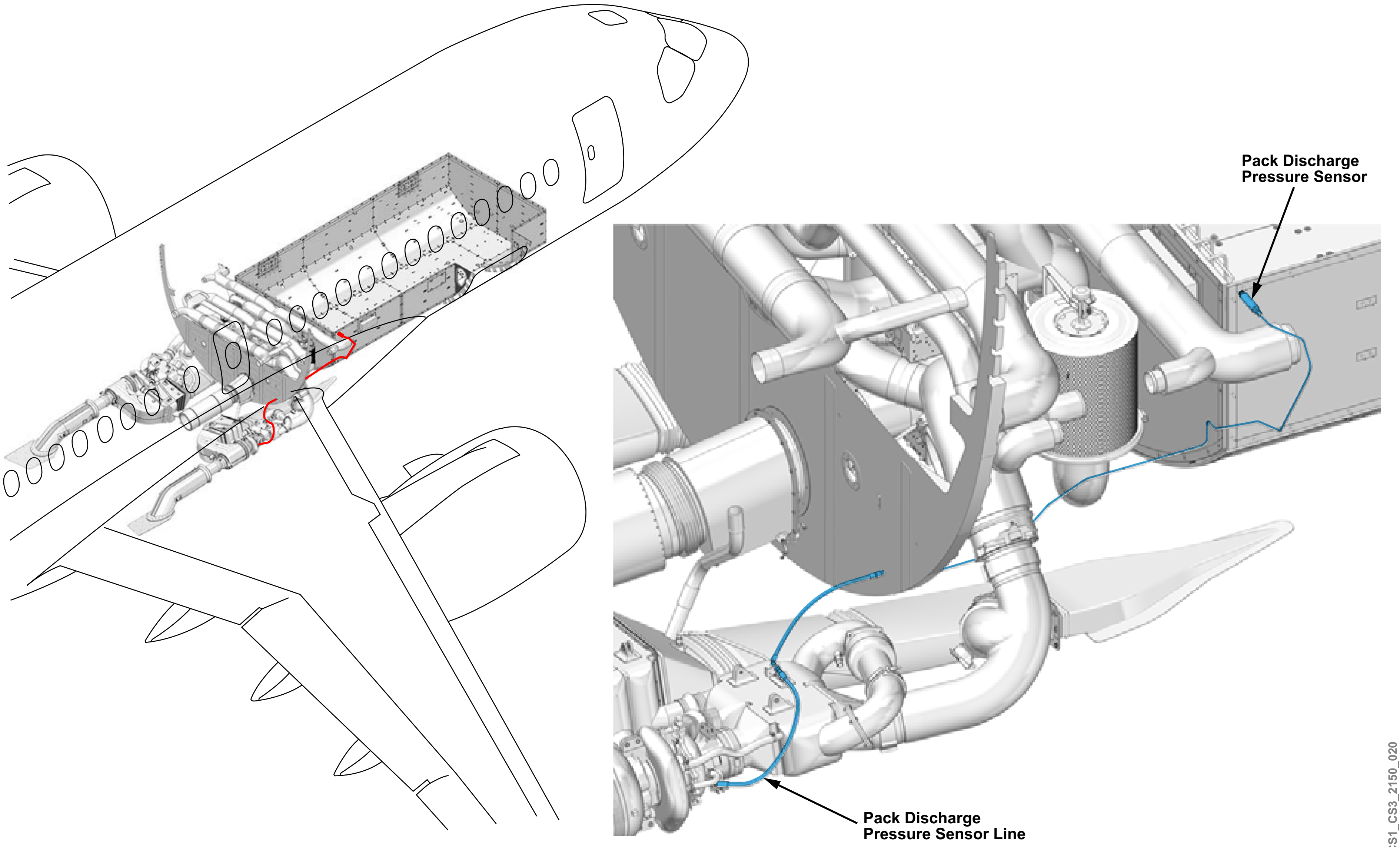


Figure 18: Pack Discharge Pressure Sensor Location

COMPONENT INFORMATION

TEMPERATURE CONTROL VALVE

The temperature control valve (TCV) controls the pack outlet temperature, by extracting hot air downstream from the primary heat exchanger and reinjecting it at the turbine.

The TCV is driven by a stepper motor. Microswitches provide position feedback to the IASC. In case of a power loss, the TCV remains in the last position. The TCV has a manual override lever to open or close the valve.

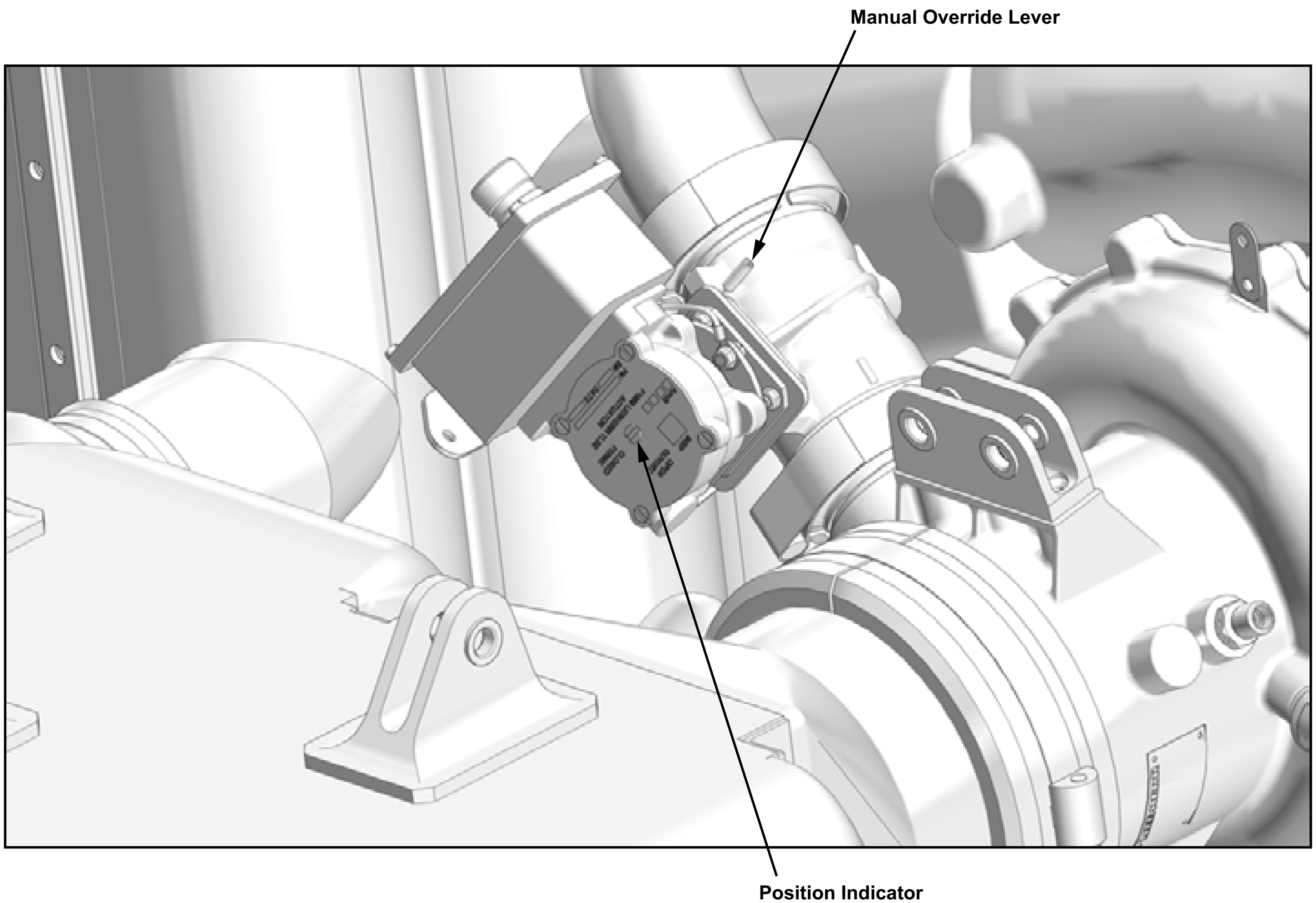


Figure 19: Temperature Control Valve

RAM AIR REGULATING VALVE

The ram air regulating valve (RARV) is used to regulate the ram air flow used on cold side of the PACK dual-heat exchanger.

The RARV is driven by a stepper motor. Microswitches provide open and close position feedback. There is always flow through the valve, even when it is closed due to the fact that the butterfly valve is smaller than the valve housing.

In case of power loss on the actuator, the RARV remains in its last position.

The RARV has a manual override to open the valve for dispatch.

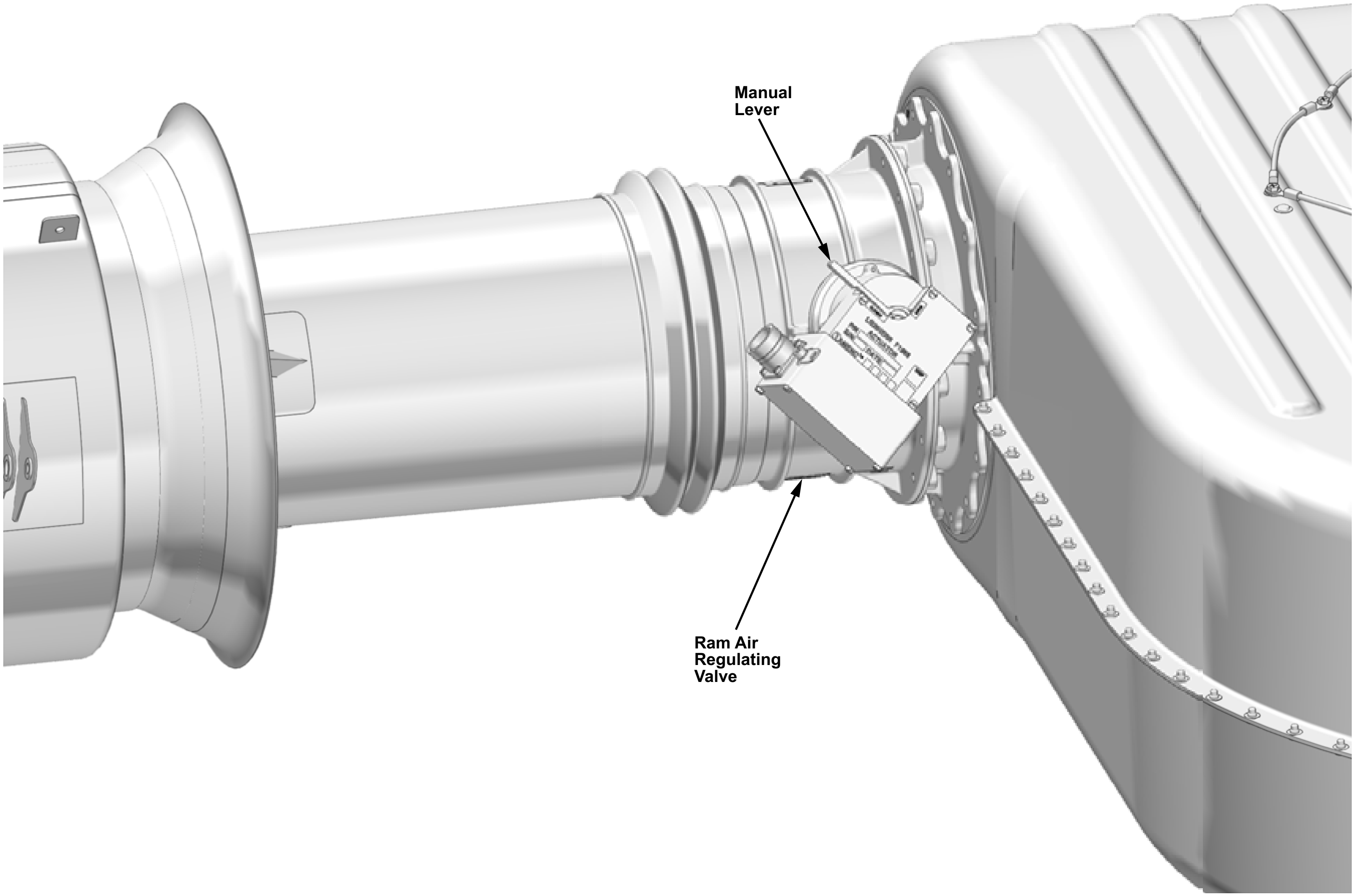


Figure 20: Ram Air Regulating Valve

CONTROLS AND INDICATIONS

CONTROLS

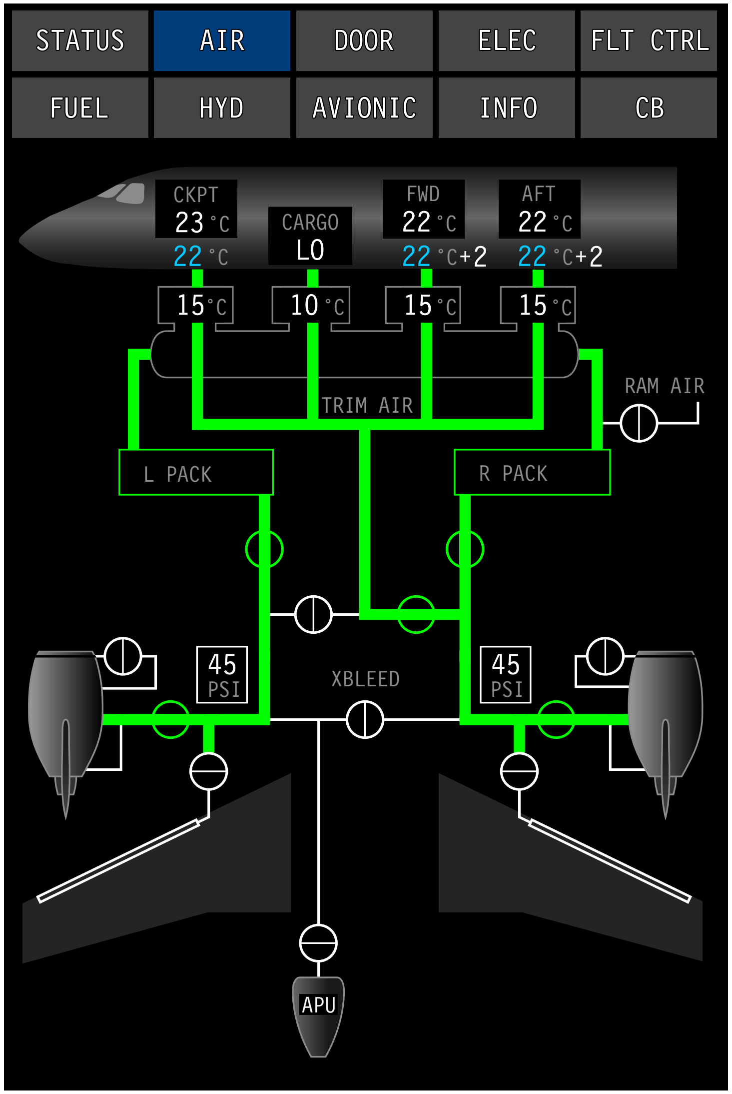
The PACK PBA turns off the pack. An OFF legend on the PACK PBA indicates the pack is off.

The PACK PBA FAIL legend indicates a fault has occurred that requires the pack to be shutdown.

Figure 21: Air Conditioning Pack Controls

INDICATIONS

The pack status is shown on the AIR synoptic page.



SYNOPTIC PAGE – AIR

PACKS	
Symbol	**Condition**
L PACK	Fail or leaked
L PACK	Active
L PACK	Invalid
L PACK	Normal OFF

CS1_CS3_2150_026

Figure 22: Air Conditioning Pack Indications

DETAILED DESCRIPTION

PACK TEMPERATURE REGULATION

The pack temperature control is performed by two regulation loops within the IASC. The system controls the RARV and TCV simultaneously. The TCV is kept as close to the fully closed position as possible, minimizing the ram air flow, and the resulting induced drag penalty.

On the ground, with the PACK selected ON, the RARV is commanded fully open above -19°C. Below this temperature the normal RARV/TCV control applies.

In flight, when the PACK is OFF, the RARV is in its closed position. When the PACK is selected ON, and the flaps or the landing gear are not fully retracted, the RARV is commanded fully open.

The TCV regulation loop compares mix manifold temperature sensor (MIXTS) values or PDTS if MIXTS is not available to a pack reference temperature.

The RARV regulation loop compares compressor discharge temperature sensor (CDTS) value to a CDTS reference function of pack reference temperature.

When the TCV or RARV fails, pack temperature control is maintained with the remaining TCV or RARV. An L or R PACK FAIL advisory message is displayed on EICAS for a TCV or RARV failure.

PACK ICING PROTECTION

To prevent ice buildup in the pack discharge outlet, the IASC opens the temperature control valve (TCV) to melt the ice. Icing protection is only available below 30,000 ft when the MIXTS is below 15°C (59°F). The ice formation is detected by the pack discharge pressure sensor (PDPS). The PDPS senses the difference between the pack discharge outlet and the cabin pressure

.

The TCV provides de-icing when the PDPS senses the differential pressure in the distribution duct is greater than 1.59 psi for more than 5 seconds. This mode is deactivated when the PDPS is less than 0.72 psi or after 30 seconds.

In the event of a total loss of the PDPS, the pack discharge temperature is limited to 5°C (41°F) to prevent the formation of ice.

PACK OVERHEAT PROTECTION

The IASCs monitor the packs for an overheat condition using a compressor discharge temperature sensor (CDTS). The CDTS supplies an output to each IASC. The IACS safety channels monitor the CDTS and if the temperature exceeds 232°C (450°F) for 30 seconds or 260°C (500°F) for 5 seconds, an L or R PACK OVHT caution message is displayed. The pack must be shutdown manually using the PACK PBA. Selecting the PACK PBA FAIL light to OFF closes the flow control valve (FCV).

To prevent a pack overheat condition, the system reduces the FCV flow as soon as the CDTS exceeds 220°C (428°F) and maintains the reduced flow until the temperature drops below this value.

If the CDTS fails completely, the pack has no overheat protection and the same L or R PACK FAIL caution message is displayed.

PACK DISCHARGE OVERHEAT PROTECTION

In the normal mode, the pack discharge temperature is limited to 70°C (158°F). The pack discharge is monitored by the pack discharge temperature sensor (PDTS). An overheat occurs when the pack discharge temperature exceeds 85°C (185°F) for 30 seconds. When IASC safety channel detects a pack discharge overheat, a L or R PACK OVHT caution message is displayed. The FCV must be closed by selecting the PACK PBA FAIL light to OFF.

If the PDTS fails completely, pack discharge overheat protection is not available, a L or R PACK FAIL caution message is displayed.

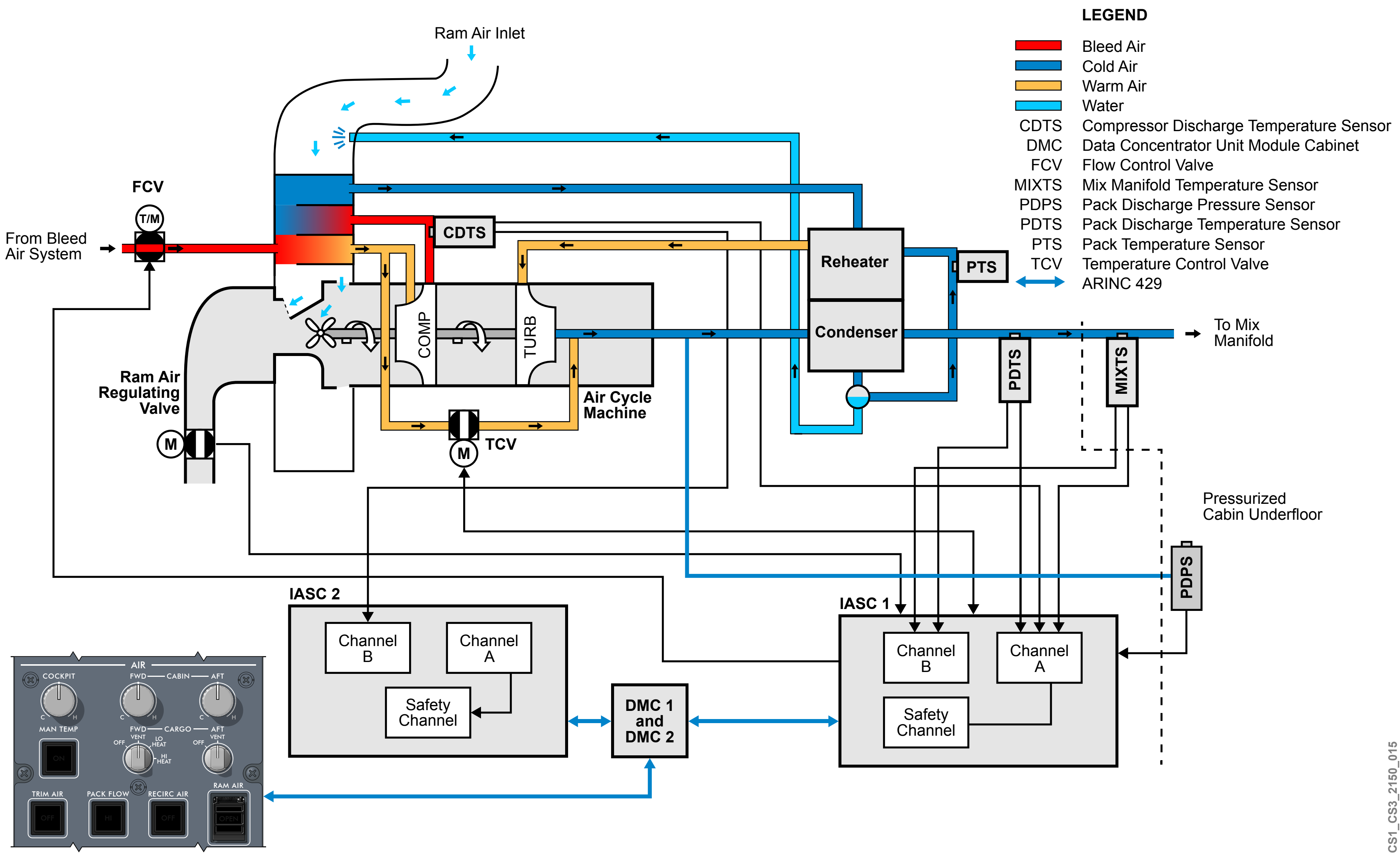


Figure 23: Air Conditioning System Detailed Description

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages associated with the air conditioning system.

CAS MESSAGES

Table 6: CAUTION Messages

MESSAGE	LOGIC
L/R PACK FAIL	Left pack inoperative (loss of control or monitoring). Loss of L PACK leak detection capability. Loss of both IASC1 channels A and B.
L/R PACK FAIL	Right pack inoperative (loss of control or monitoring). Loss of R PACK leak detection capability. Loss of both IASC2 channels A and B.
L PACK OVHT	Left compressor discharge overheat or left pack discharge overheat.
R PACK OVHT	Right compressor discharge overheat or left pack discharge overheat.

Table 7: ADVISORY Message

MESSAGE	LOGIC
PACK FAULT	Loss of redundant or non-critical function for the air conditioning system.

Table 8: STATUS Message

MESSAGE	LOGIC
L/R PACK OFF	Left/right pack selected OFF and confirmed OFF.

Table 9: INFO Messages

MESSAGE	LOGIC
21 PACK FAULT - MIX MANF TEMP SNSR TOTAL LOSS	MMTS from both packs are lost.

Table 9: INFO Messages

MESSAGE	LOGIC
21 PACK FAULT - MIX MANF TEMP SNSR REDUND LOSS	Up to 3 MMTS sensing elements redundant loss or any sensing element drift.
21 PACK FAULT - L BYPASS VLV INOP	Left TCV failed in position or position unknown.
21 PACK FAULT - R BYPASS VLV INOP	Right TCV failed in position or position unknown.
21 L PACK FAIL - L FLOW CTRL VLV INOP	Left FCV failed open.
21 R PACK FAIL - R FLOW CTRL VLV INOP	Right FCV failed open.
21 L PACK FAIL - L PACK INOP	L Pack failed for any reason except FCV failed open case.
21 R PACK FAIL - R PACK INOP	R Pack failed for any reason except FCV failed open case.
21 PACK FAULT - L PACK TEMP SNSR REDUND LOSS	Left CDTS or PDTS drift or single-channel failure.
21 PACK FAULT - R PACK TEMP SNSR REDUND LOSS	Right CDTS or PDTS drift or single-channel failure.
21 PACK FAULT - L PACK DISCH PRESS SNSR INOP	Left PDPS loss.

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Table 9: INFO Messages

MESSAGE	LOGIC
21 PACK FAULT - R PACK DISCH PRESS SNSR INOP	Right PDPS loss.
36 L BLEED FAIL - L PACK INLET PRESS SNSR INOP	Left PIPS 1 and PIPS 2 loss.
36 R BLEED FAIL - R PACK INLET PRESS SNSR INOP	Right PIPS 1 and PIPS 2 loss.
21 AIR SYSTEM FAULT - L PACK PRESS SNSR REDUND LOSS	Left pack pressure sensor redundant loss (PIFS or PIPS).
21 AIR SYSTEM FAULT - R PACK PRESS SNSR REDUND LOSS	Right pack pressure sensor redundant loss (PIFS or PIPS).
21 PACK FAULT - L RAM AIR REG VLV INOP	Left RARV failed in position or unknown position.
21 PACK FAULT - R RAM AIR REG VLV INOP	Right RARV failed in position or unknown position.
21 L PACK OVHT - L RAM AIR REG VLV INOP	Left RARV failed in position leading to pack overheat.
21 R PACK OVHT - R RAM AIR REG VLV INOP	Right RARV failed in position leading to pack overheat.

Table 9: INFO Messages

MESSAGE	LOGIC
21 L PACK OVHT - L PACK INOP	L pack overheat triggered for any reason except RARV failed in position case.
21 R PACK OVHT - R PACK INOP	R pack overheat triggered for any reason except RARV failed in position case.
21 R PACK OVHT - L PACK TEMP SNSR INOP	Left PTS drift or both channels failure.
21 R PACK OVHT - R PACK TEMP SNSR INOP	Right PTS drift or both channels failure.
21 L PACK FAIL - L PACK DISCHARGE DUCT DISCONNECT	L pack discharge duct disconnected.
21 R PACK FAIL - R PACK DISCHARGE DUCT DISCONNECT	R pack discharge duct disconnected.

21-52 RAM AIR SYSTEM

GENERAL DESCRIPTION

In case of dual pack loss, the emergency ram air valve provides a backup fresh air supply for unpressurized flight below 10,000 ft. The emergency ram air valve is powered by DC ESS BUS 3 and opens when the RAM AIR PBA is pressed. The emergency ram air valve connects the ram air inlet to the right pack discharge duct. If bleed air is available, the ram air can be mixed with hot air from the trim air system.

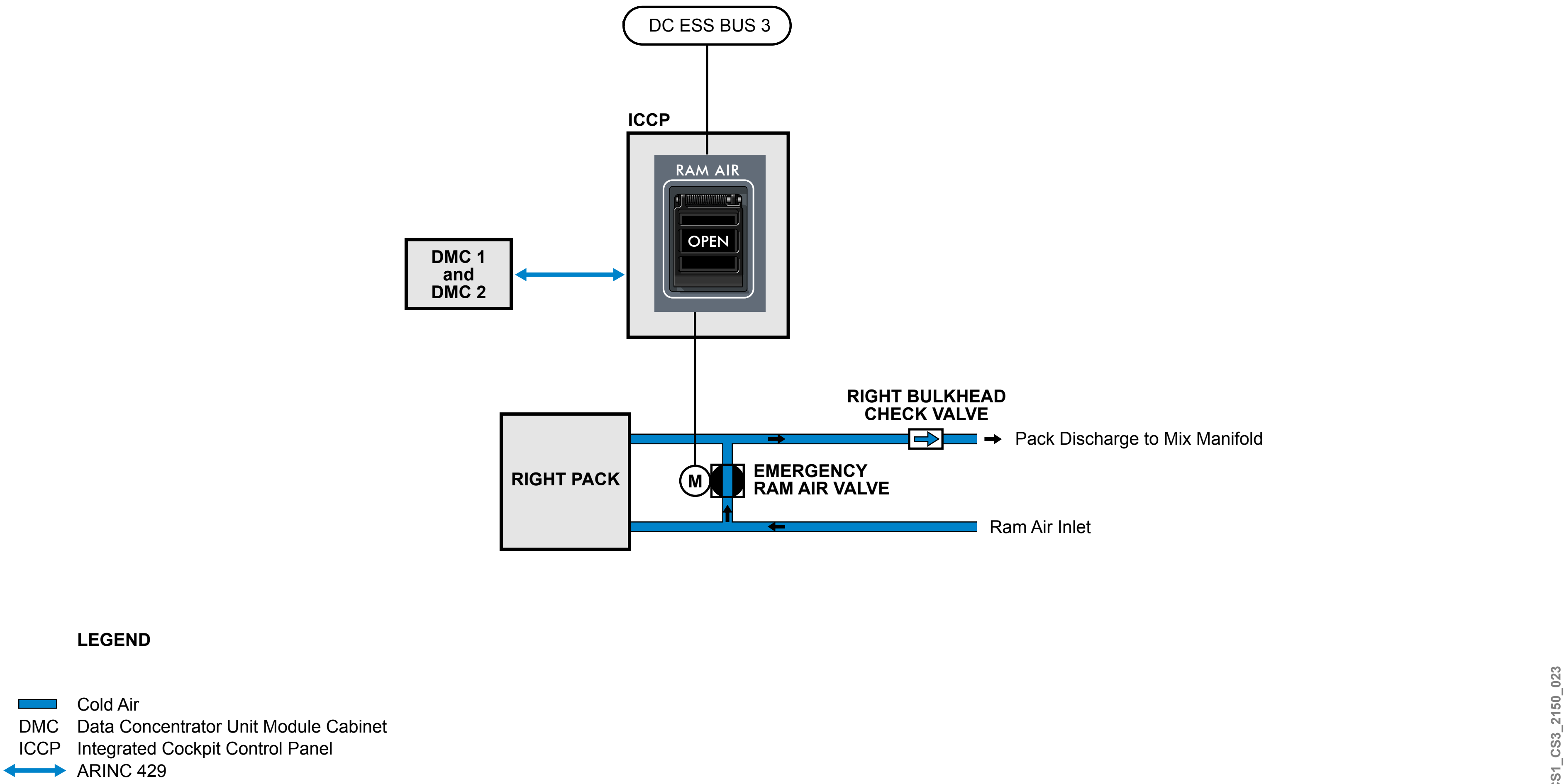


Figure 24: Ram Air System

COMPONENT LOCATION

The following components are located in the ram air system:

- Emergency ram air valve

EMERGENCY RAM AIR VALVE

The emergency ram air valve is installed between the right pack discharge duct and the ram air inlet duct.

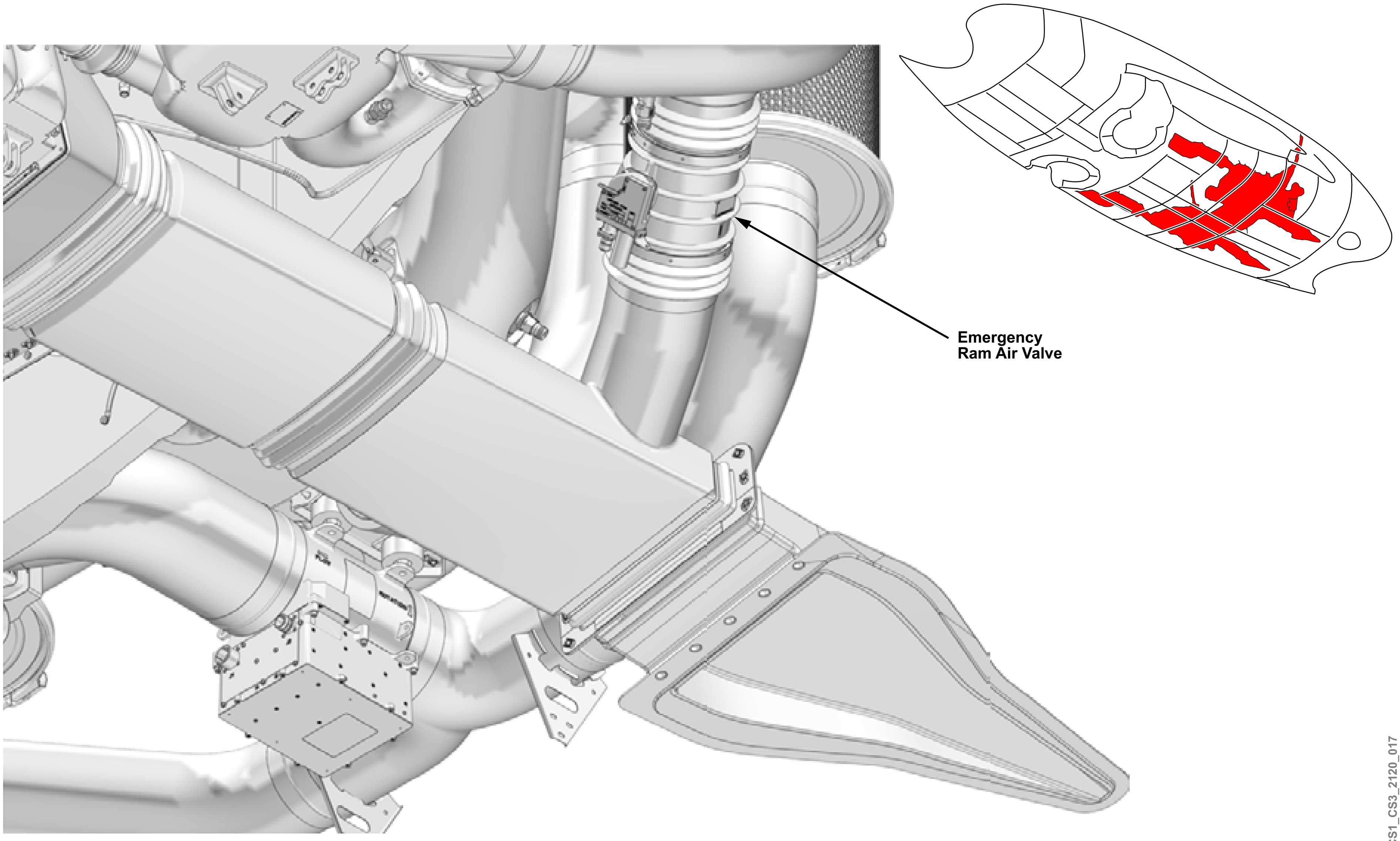


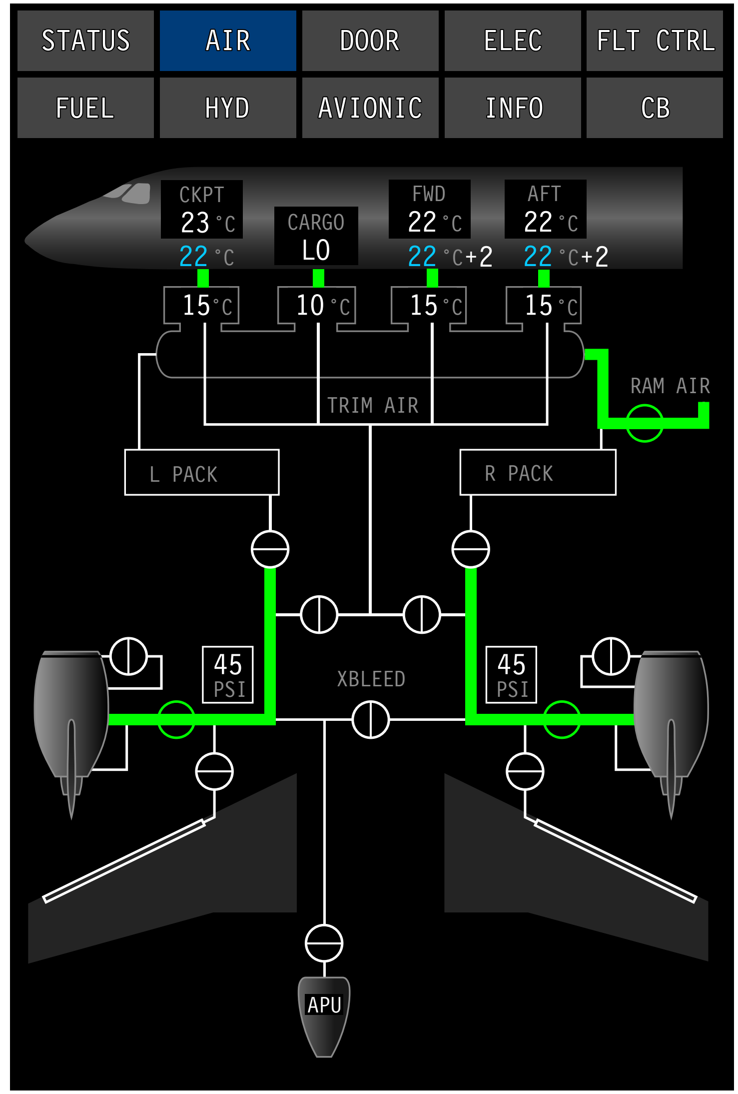
Figure 25: Emergency Ram Air Valve

CONTROLS AND INDICATIONS

The RAM AIR PBA opens the emergency ram air valve. An OPEN legend on the RAM AIR PBA indicates the valve is open.

The RAM AIR valve and flowbar on the AIR synoptic page illuminate green when the ram air system is selected ON.

AIR PANEL



SYNOPTIC PAGE – AIR

CS1_CS3_2150_024

Figure 26: Ram Air System Controls and Indications

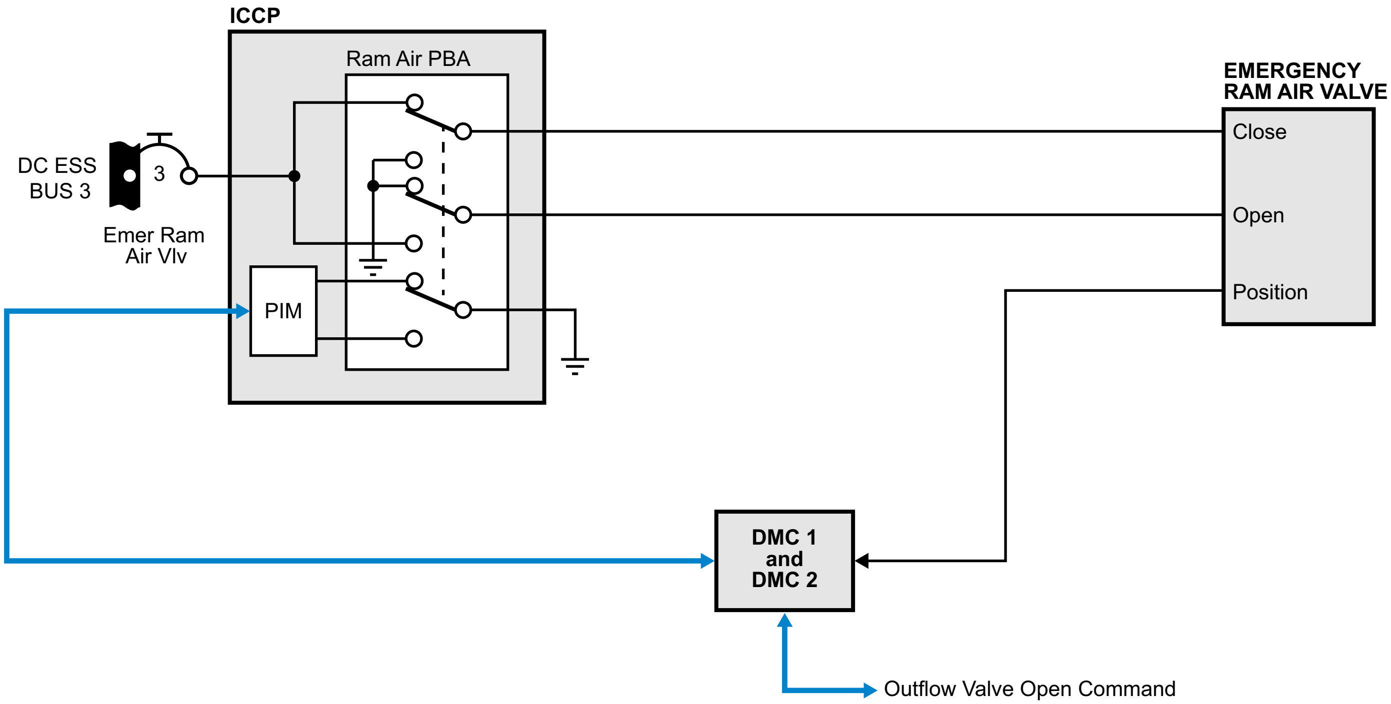
DETAILED DESCRIPTION

The emergency ram air ventilation system supplies fresh airflow for unpressurized operation. The system supplies fresh air at altitudes up to 10,000 ft. The aircraft speed needs to be maintained above 0.5 Mach to ensure that enough dynamic air pressure is available at the ram air inlet.

The emergency ram air valve connects the right hand pack ram air inlet diffuser to the right hand pack discharge duct. Air flows into the mix manifold for distribution throughout the aircraft. The valve has a manual override with a visual position indication. Microswitches provide position feedback to the data concentrator unit module cabinets (DMCs).

The emergency ram air valve is powered from DC ESSENTIAL BUS 3 supplied through the hardwired RAM AIR guarded PBA on the AIR panel. An OPEN legend on the RAM AIR PBA, and a RAM AIR OPEN status message indicate the valve is open.

When the RAM AIR PBA is selected, the outflow valve is commanded open to prevent excessive cabin pressure differential pressure.



LEGEND

DMC	Data Concentrator Unit Module Cabinet
ICCP	Integrated Cockpit Control Panel
PIM	Panel Interface Module
↔	ARINC 429

Figure 27: Ram Air System Schematic

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages associated with the ram air system:

CAS MESSAGES

Table 10: CAUTION Message

MESSAGE	LOGIC
RAM AIR FAIL	RAM AIR selected OPEN/CLOSED and ERAV not detected full open/full closed.

Table 11: STATUS Message

MESSAGE	LOGIC
RAM AIR OPEN	RAM AIR selected and confirmed OPEN.

Table 12: INFO Messages

MESSAGE	LOGIC
21 AIR SYSTEM FAULT - ERAV INOP	ERAV in unknown position.

21-60 TRIM AIR SYSTEM

GENERAL DESCRIPTION

The trim air system is used for independent zone temperature control. The aircraft zones serviced by the trim air are the flight deck, forward cabin, aft cabin, and forward cargo compartment.

Bleed air is supplied from each pack inlet line upstream of the flow control valves. The trim air shutoff valve (TASOV) on the left side, and the trim air pressure-regulating valve (TAPRV) on the right side supply bleed air to the trim air system. The trim system takes hot air upstream the flow control valve and ducts it to the distribution system downstream of the mix manifold. The trim air is injected in the individual zone distribution air supply ducting, and mixed with cold air from packs to provide the required temperature for the zone.

Each trim air line is fitted with a trim air check valve (TACKV), installed in line at the pressure bulkhead. In case of a trim air line disconnect it isolates the pressurized cabin from the unpressurized wing-to-body faring (WTBF) bay of the aircraft to prevent cabin depressurization.

The TASOV and trim air valves (TAVs) are powered through their respective integrated air system controller (IASC).

IASC 1 powers:

- TASOV
- Flight deck TAV
- Aft cabin TAV

IASC 2 powers:

- Forward cabin TAV
- Forward cargo TAV

The TAPRV is powered by DC ESS BUS 1, and reports its position to IACSC 2.

In normal operation, the system is on and controlled automatically by the IASCs. The TAPRV modulates to provide 12.8 psi pressure, to the TAVs. The TASOV remains closed during normal operation. If the TAPRV fails closed, the TASOV is driven open by IASC 1 and used to supply the trim air system.

To improve heating performance on the ground during the flow control pull up mode, the TASOV is opened by IASC 1.

If the trim air system is selected off using the TRIM AIR PBA on the AIR panel or commanded closed when the flow control valves (FCVs) close, the TASOV, TAPRV, and TAVs close. The flight deck TAV remains slightly open to prevent a pressure build-up in the trim air distribution lines.

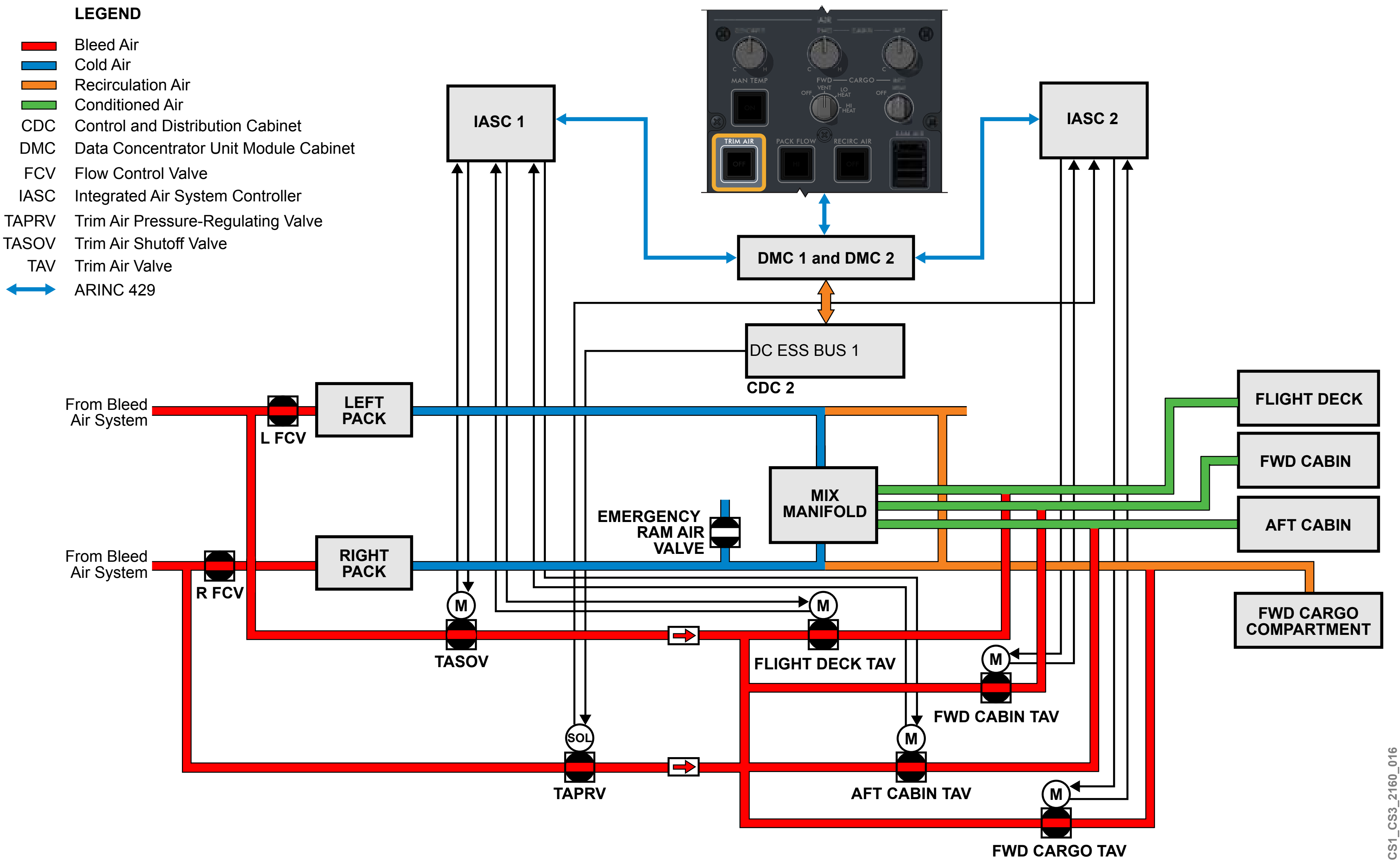


Figure 28: Trim Air System

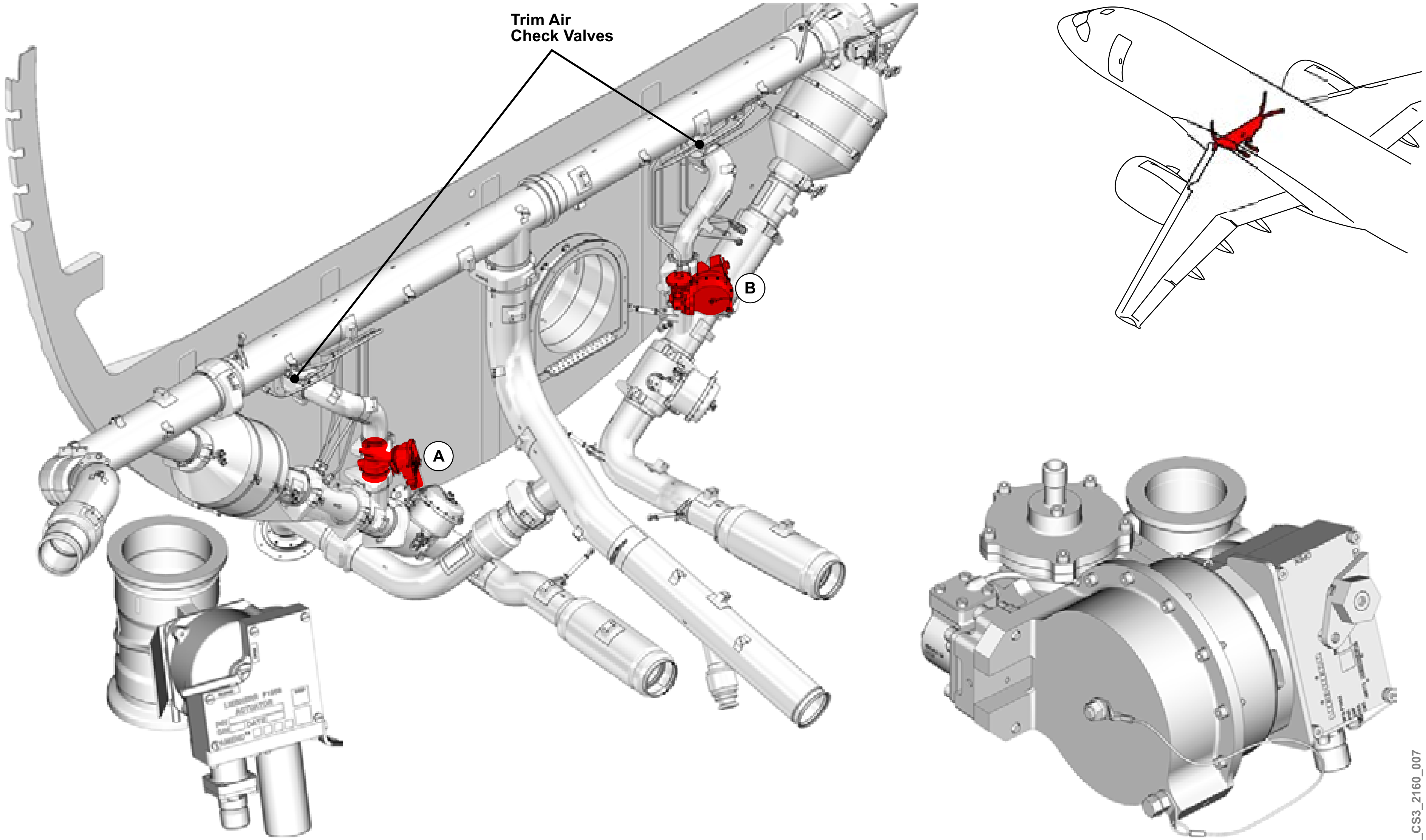
COMPONENT LOCATION

The following components are located in the WTBF:

- Trim air pressure-regulating valve (TAPRV)
- Trim air shutoff valve (TASOV)
- Trim air check valve (TACKV)

The following components are located in the environmental control system bay:

- Trim air valves (TAVs) (Refer to figure 30)



A TRIM AIR SHUTOFF VALVE

B TRIM AIR PRESSURE-REGULATING VALVE

Figure 29: Trim Air Shutoff Valves, Trim Air Check Valves, and Trim Air Pressure-Regulating Valve Location

TRIM AIR VALVES

The trim air valves (TAVs) are located along the trim air distribution line in the environmental control system bay.

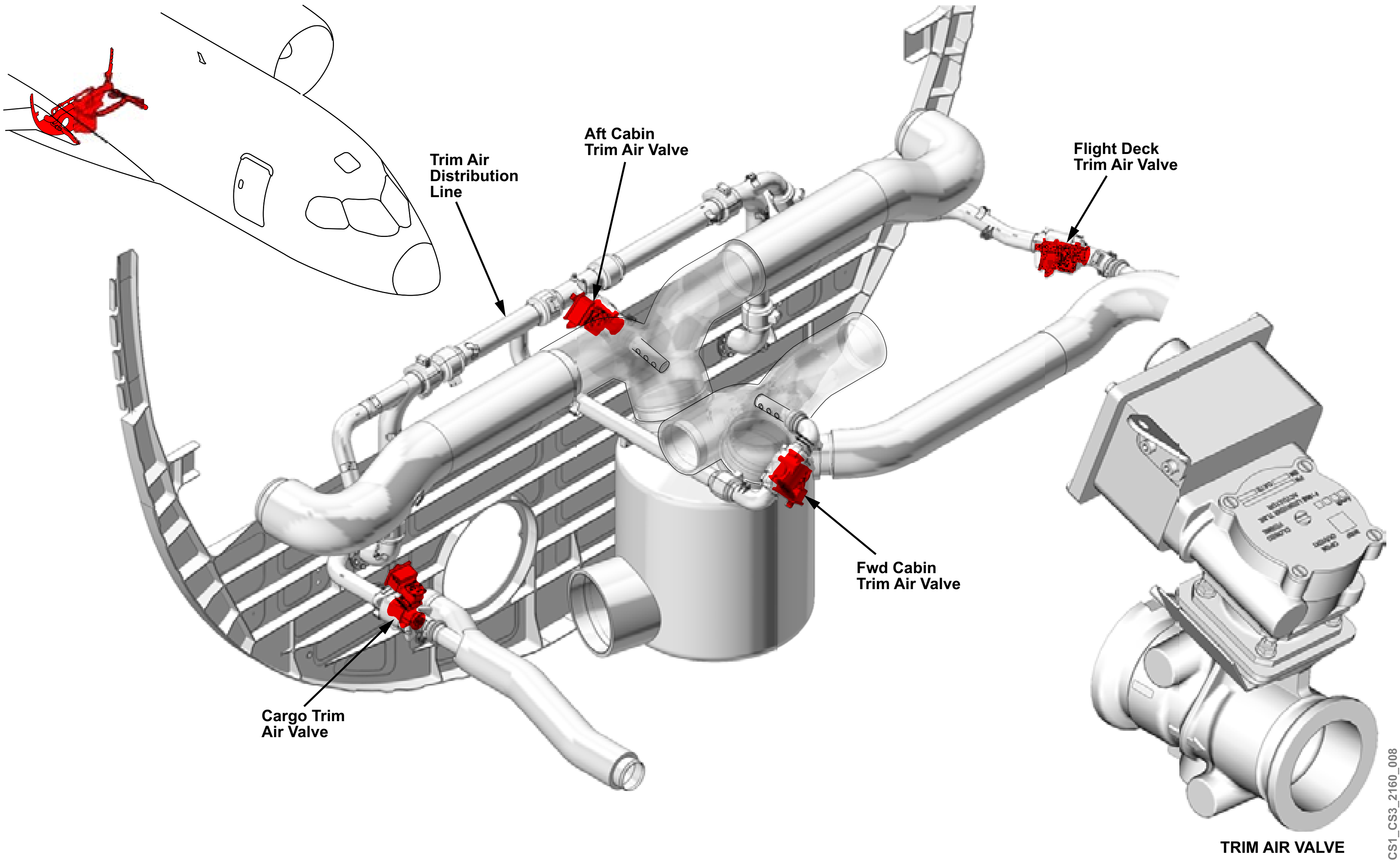


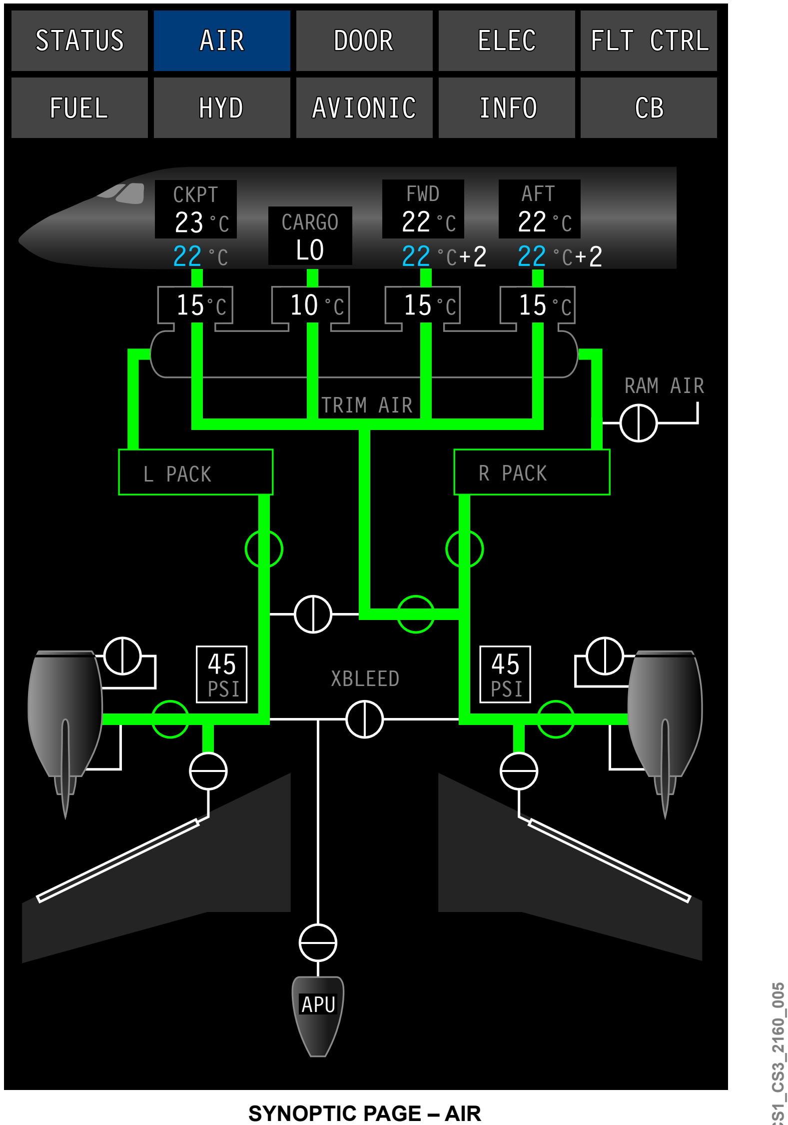
Figure 30: Trim Air Valve Locations

CONTROLS AND INDICATIONS

The TRIM AIR PBA turns off the trim air system. An OFF legend on the TRIM AIR PBA indicates the trim air system is off.

A TRIM AIR OFF message appears on the AIR synoptic page to indicate the system is off.

AIR PANEL



SYNOPTIC PAGE – AIR

Figure 31: Trim Air System Controls and Indications

DETAILED DESCRIPTION

TRIM AIR CONTROL

Each trim air valve (TAV) is commanded by the IASC according to the duct temperature sensor (DTS) measurement compared with a reference temperature, which is determined by the error between the selected temperature from the AIR panel in the flight deck and the temperature measured by the associated zone ventilated temperature sensor (VENTS).

When the trim air system is selected OFF, the TAVs are also commanded closed by the IASCs. The flight deck TAV is commanded slightly open to avoid a pressure buildup in the trim air lines.

The flight deck, forward cabin, aft cabin, and the FWD cargo TAV are automatically closed whenever the trim air system is closed, and either the TAPRV or the TASOV fails open.

NOTE

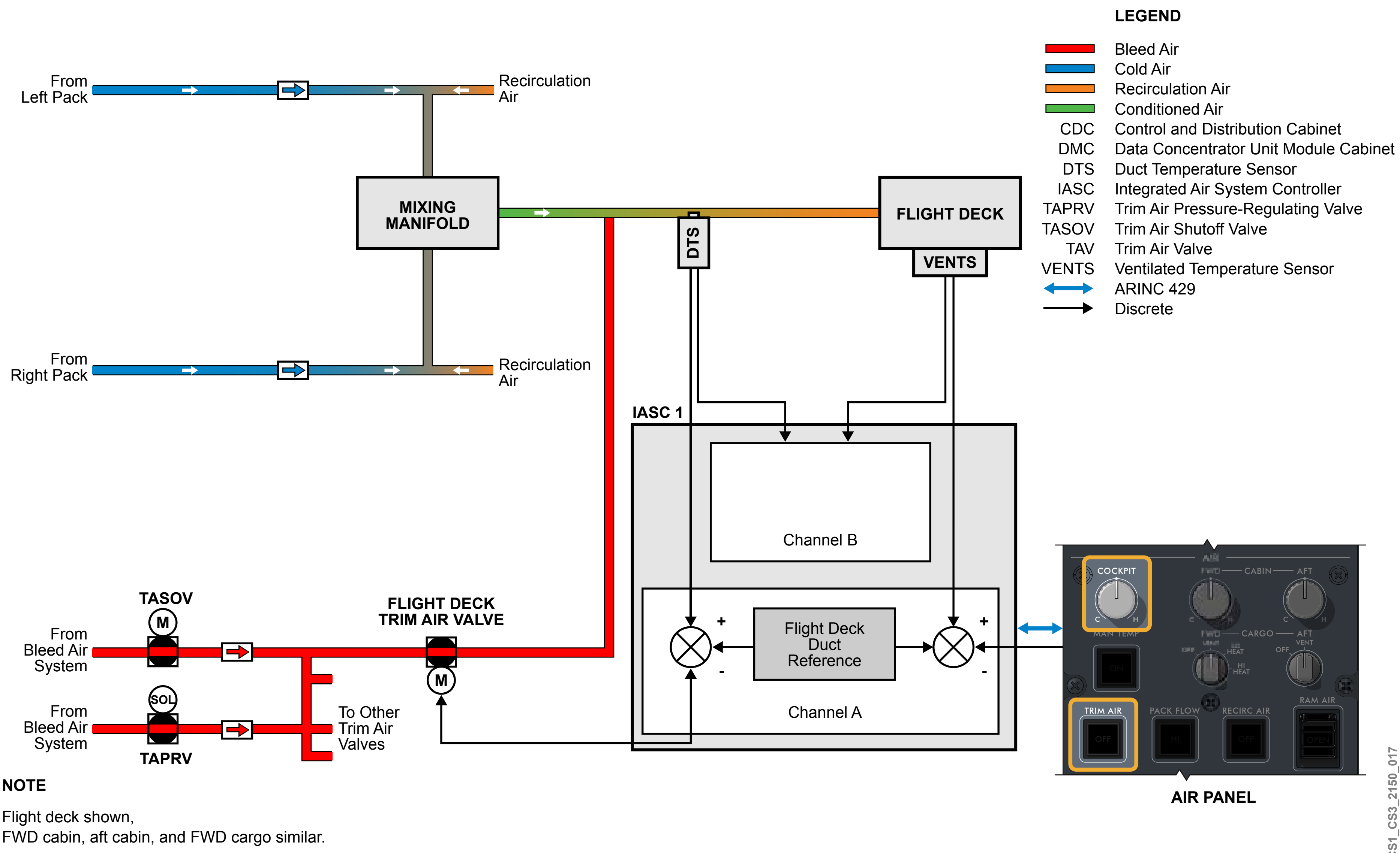
When the trim system is shutoff and all TAVs are closed, the flight deck TAV is commanded to open for 2.7 seconds, to prevent pressure build-up in the trim air distribution line.

The TASOV and TAPRV are automatically closed whenever:

- DTS overheat is detected
- Bleed leak is detected affecting the TAPRV and TASOV downstream ducting
- Both flow control valves (FCVs) are commanded closed, and the emergency ram air valve (ERAV) is not selected open
- TRIM AIR PBA is selected OFF
- The TAPRV or the TASOV are commanded closed when its associated FCV is closed, and ERAV is not selected open.
- Pack start up
- Engine start + 30 seconds
- Ditching

The cargo TAV is automatically closed when:

- The cargo heating system is selected OFF by the flight crew FWD CARGO switch set to OFF or VENT
- The cargo SOVs are commanded closed due to forward cargo compartment smoke detection
- TRIM AIR PBA OFF
- Cargo DTS overheat
- Total loss of cargo DTS



NOTE

Flight deck shown,
FWD cabin, aft cabin, and FWD cargo similar.

Figure 32: Trim Air Control

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages associated with the trim air system:

CAS MESSAGES

Table 13: CAUTION Messages

MESSAGE	LOGIC
TRIM AIR FAIL	Trim air failure or loss of trim air leak detection or loss of both IASC 1channels.
FWD CARGO HEAT FAIL	LO HEAT AND HI HEAT mode not available.
FWD CARGO LO TEMP	Low temperature <4°C in fwd cargo when FWD CARGO selected to LO/HI HEAT

Table 14: ADVISORY Message

MESSAGE	LOGIC
AIR SYSTEM FAULT	Fault not requiring immediate attention. (Total loss of VENTS, IASC single-channel failure, loss of cockpit, FWD, AFT cabin DTS, TAV stuck, TAPRV/TASOV failed closed, single channel CAR DTS, or CATS failed).

Table 15: STATUS Messages

MESSAGE	LOGIC
TRIM AIR OFF	TRIM AIR selected OFF and both TAPRV and TASOV confirmed closed.
FWD CARGO AIR OFF	FWD cargo heating selection selected and confirmed OFF.
AFT CARGO AIR OFF	AFT cargo heating selection selected and confirmed OFF.

Table 16: INFO Messages

MESSAGE	LOGIC
21 AIR SYSTEM FAULT - TRIM AIR PRV FAIL CLSD	TAPRV failed closed.
21 AIR SYSTEM FAULT - TRIM AIR PRV FAIL OPEN	TAPRV failed open.
21 AIR SYSTEM FAULT - TRIM AIR SOV FAIL CLSD	TASOV failed closed in full closed position.
21 AIR SYSTEM FAULT - TRIM AIR SOV FAIL OPEN	TASOV failed not closed or unknown position.
21 AIR SYSTEM FAULT - FWD CARGO SOV INOP	Upstream or downstream fwd cargo SOV failed in position or in unknown position.
21 AIR SYSTEM FAULT - AFT CARGO SOV INOP	Upstream or downstream aft cargo SOV failed.
21 AIR SYSTEM FAULT - FWD CAR TAV FAIL CLSD	CAR TAV failed closed.
21 AIR SYSTEM FAULT - TAV INOP	Loss of CKPT or FWD CAB or AFT CAB TAV or any TAV unknown position

21-60 TEMPERATURE CONTROL SYSTEM

GENERAL DESCRIPTION

The temperature control system controls the flight deck, forward and aft cabin, and the forward cargo compartment. Individual automatic temperature controls for the flight deck and forward and aft cabins are available on the AIR panel. The forward and aft cabin temperature is also controlled from the cabin management system (CMS) crew terminal. The forward cargo heat is selectable from the AIR panel.

The system temperature control for the flight deck, forward and aft cabins, and forward cargo compartment is automatically controlled via the integrated air system controllers (IASCs) in order to meet the flight deck temperature selections. The IASCs receive temperature inputs from:

- Ventilated temperature sensor for actual zone temperature
- Duct temperature sensors for duct temperature
- Mix manifold temperature sensors for pack temperature control
- Cargo temperature sensor for forward cargo actual temperature
- Cargo duct temperature sensor for cargo duct temperature

The packs are controlled to provide the lowest temperature output required by the flight deck, forward, or aft cabin zones. The IASCs control the trim air valve (TAV) to add hot air to each of the distribution ducts that require additional heating.

In automatic mode, the required zone temperature for the flight deck, forward, and aft cabin can be set from 18°C (64°F) to 30°C (86°F) using the AIR panel temperature selectors.

In manual mode, the flight crew adjust the zone temperatures by adjusting the duct temperature from full cold 5°C (41°F) to full hot 65°C (149°F) using the AIR panel temperature selectors.

The forward and aft cabin temperature can be adjusted at the cabin management system crew terminal. The crew terminal provides the capability to adjust the temperature by ± 3°C (5.4°F) from the temperature selection made on the AIR panel FWD, or AFT CABIN temperature selectors.

The forward cargo LO HEAT provides a temperature range of 15°C (59°F) to 20°C (68°F), and HI HEAT provides a temperature range of 20°C (68°F) to 25°C (77°F).

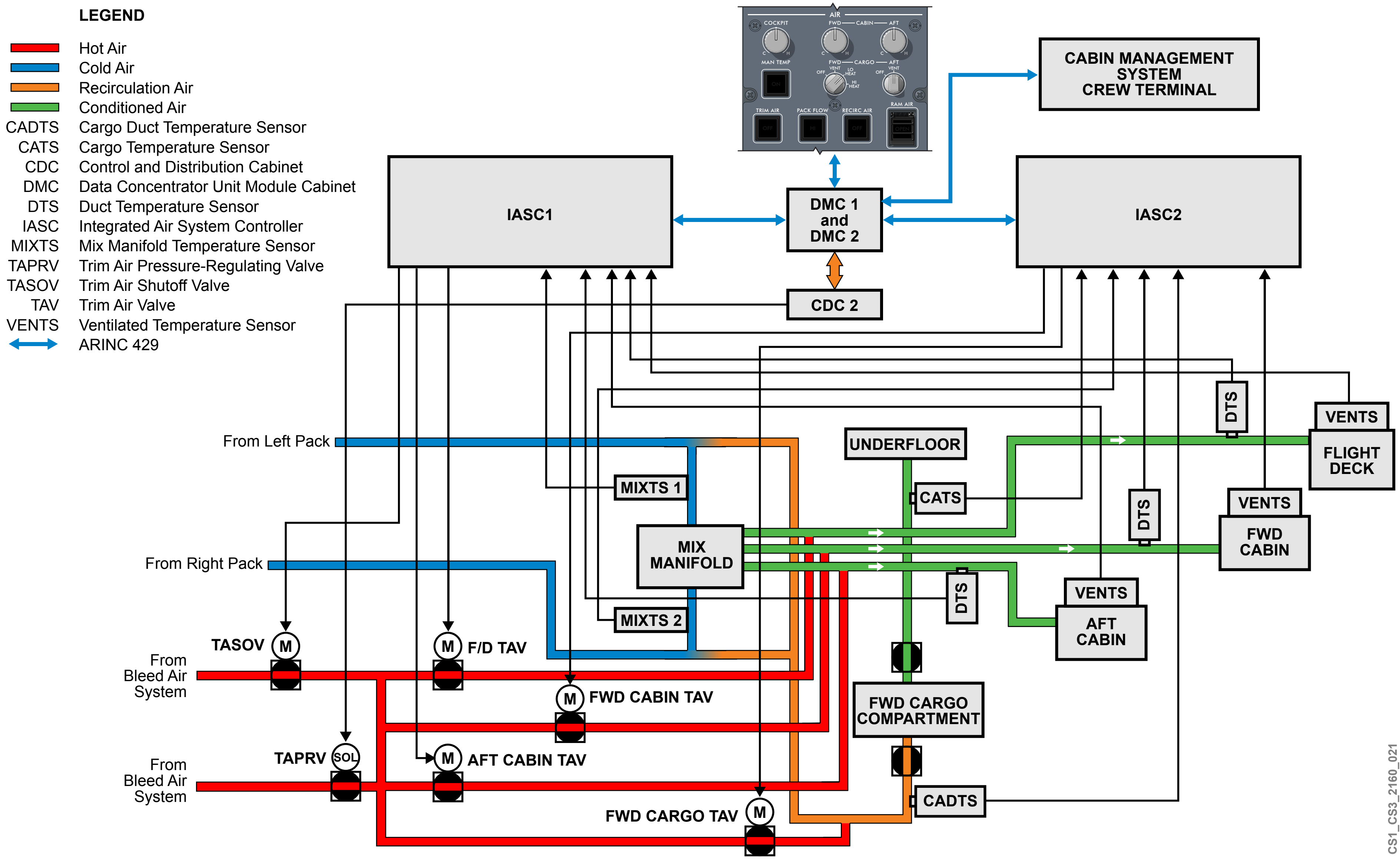


Figure 33: Temperature Control System

COMPONENT LOCATION

The following are major components of the temperature control system:

- Mix manifold temperature sensors (MIXTS)
- Duct temperature sensor (DTS) (Refer to figure 35)
- Ventilated temperature sensors (VENTS) (Refer to figure 36)
- Cargo duct temperature sensor (CADTS) (Refer to figure 37)
- Cargo temperature sensor (CATS) (Refer to figure 37)

MIX MANIFOLD TEMPERATURE SENSORS

The mix manifold temperature sensors are located in the pack discharge lines near the inlet to the mix manifold.

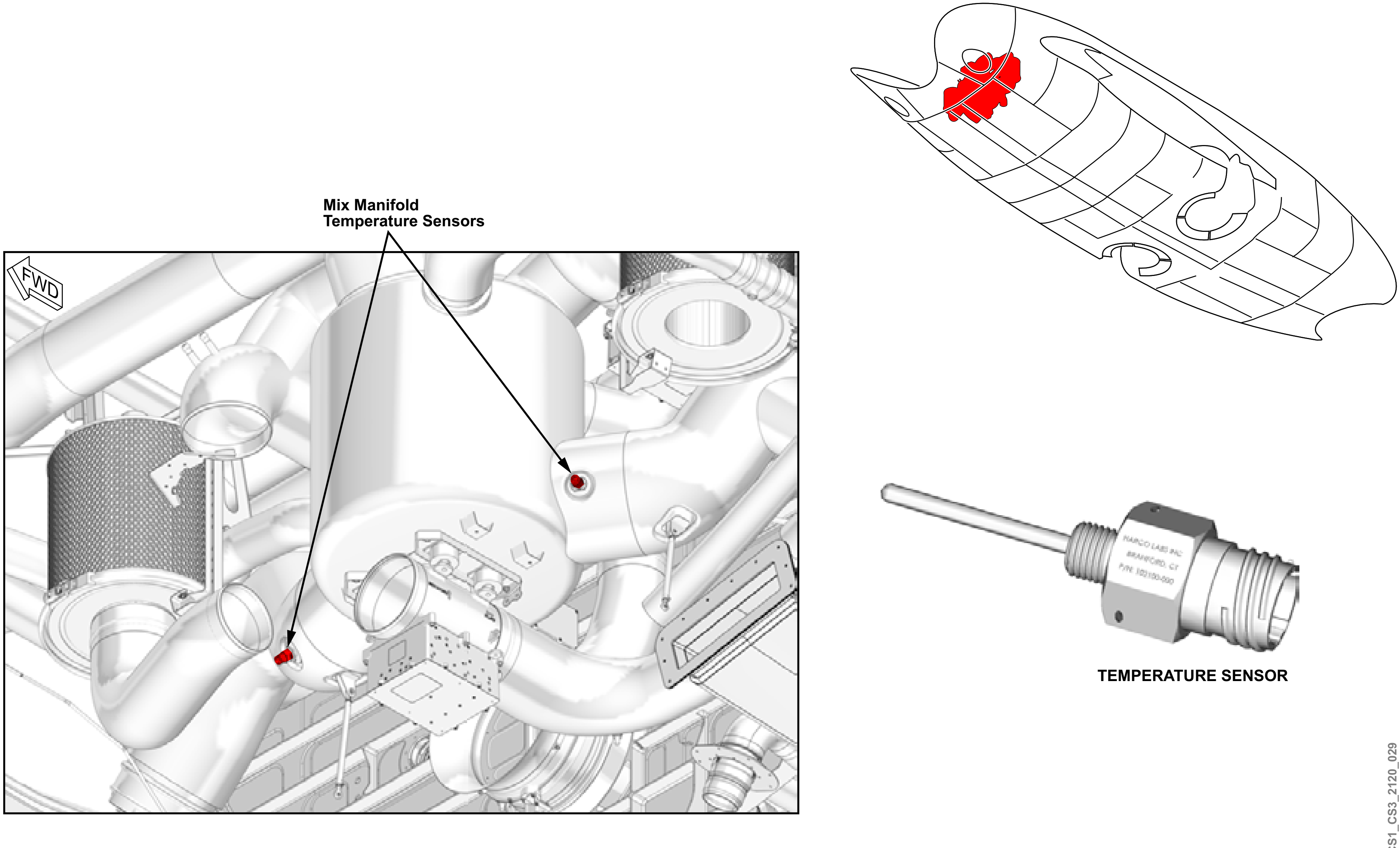


Figure 34: Mix Manifold Temperature Sensors

DUCT TEMPERATURE SENSORS

The duct temperature sensors (DTS) for the forward cabin, aft cabin, and flight deck are located in the environmental control system bay.

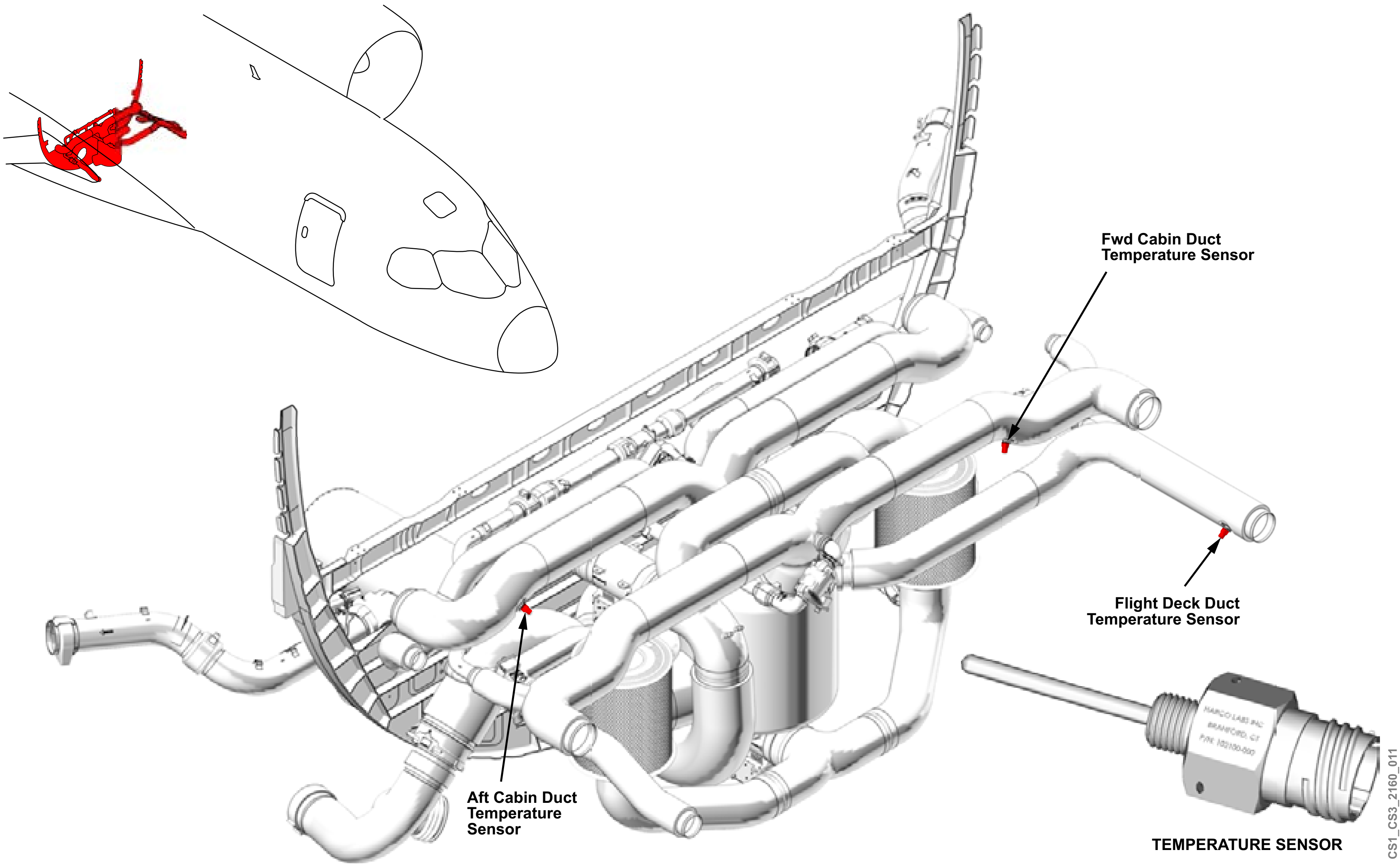


Figure 35: Duct Temperature Sensors

VENTILATED TEMPERATURE SENSORS

Flight Deck Ventilated Temperature Sensor

The flight deck ventilated temperature sensor is installed on the pilots bulkhead.

Forward Cabin Ventilated Temperature Sensor

The forward cabin ventilated temperature sensor is located on the right side of the forward cabin. The sensor is mounted in the passenger service unit located between frame 34 and frame 35.

Aft Cabin Ventilated Temperature Sensor

The aft cabin ventilated temperature sensor is located on the left side of the aft cabin. The sensor is mounted in the passenger service unit located between frame 62 and frame 64.

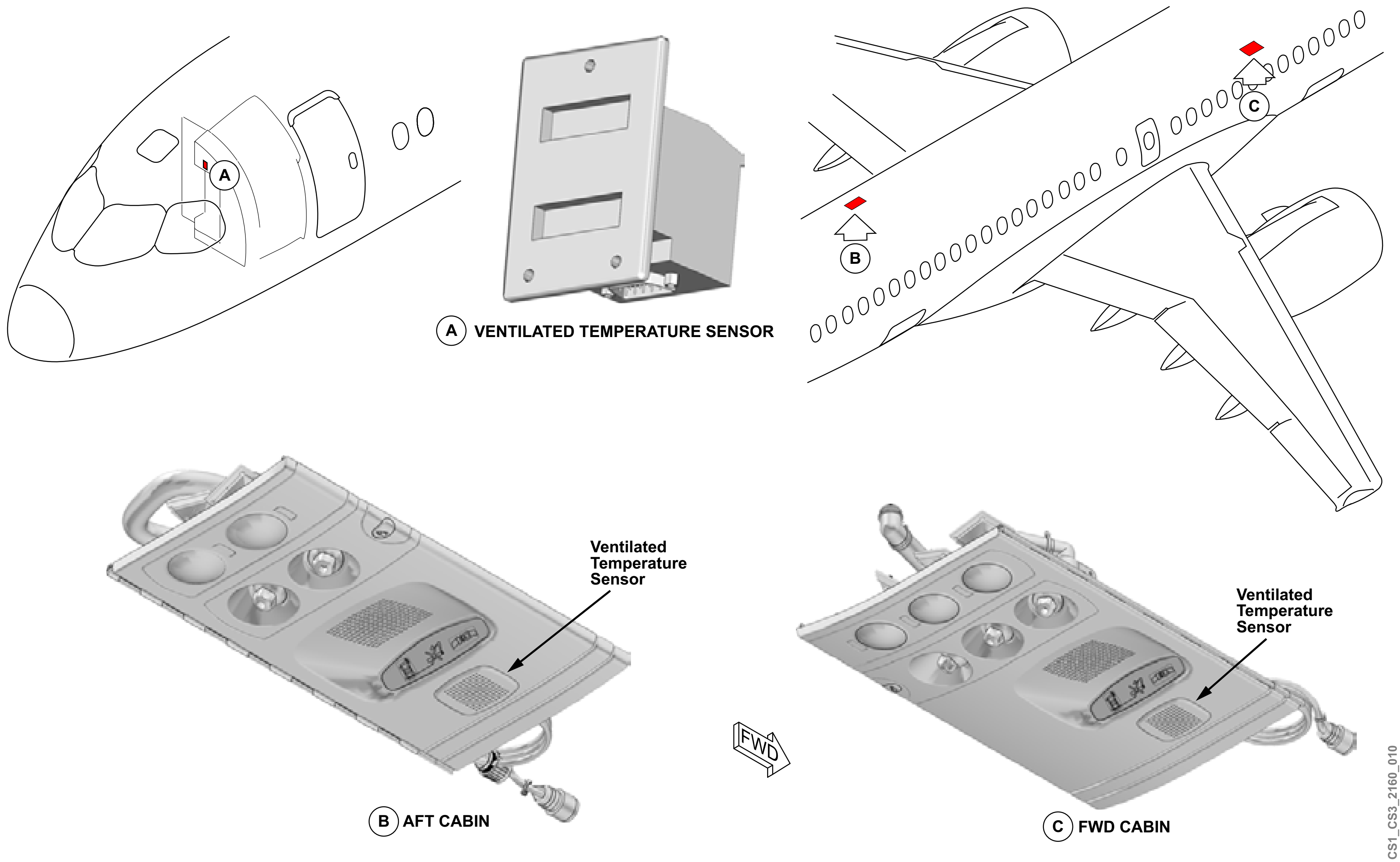


Figure 36: Flight Deck and Cabin Ventilated Temperature Sensors

CARGO DUCT TEMPERATURE SENSOR

The cargo duct temperature sensor (CADTS) is located in the forward cargo ventilation inlet duct in the environmental control system (ECS) bay.

CARGO TEMPERATURE SENSOR

The cargo temperature sensor (CATS) is located in the forward cargo ventilation exhaust duct in the environmental control system bay.

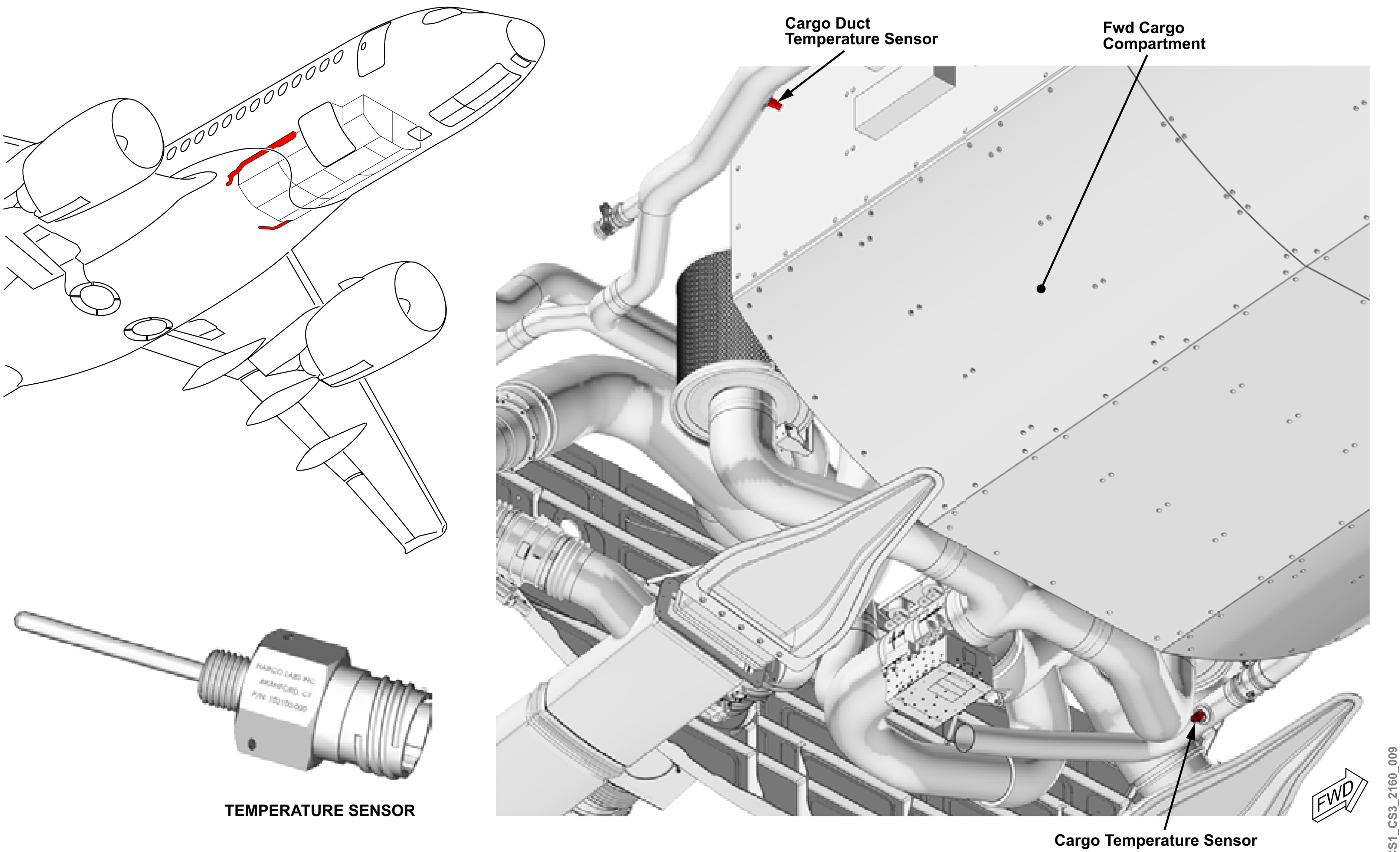


Figure 37: Cargo Duct Temperature Sensor and Cargo Temperature Sensor

COMPONENT INFORMATION

VENTILATED TEMPERATURE SENSOR

The ventilated temperature sensor (VENTS) supply temperature information to the IASC.

The vents consist of two temperature sensing elements installed on a circuit board. The circuit board is housed in a plastic box along with a small fan. The fan draws air across the temperature sensor. A filter is installed on the cover to ensure the temperature sensors stay clean.

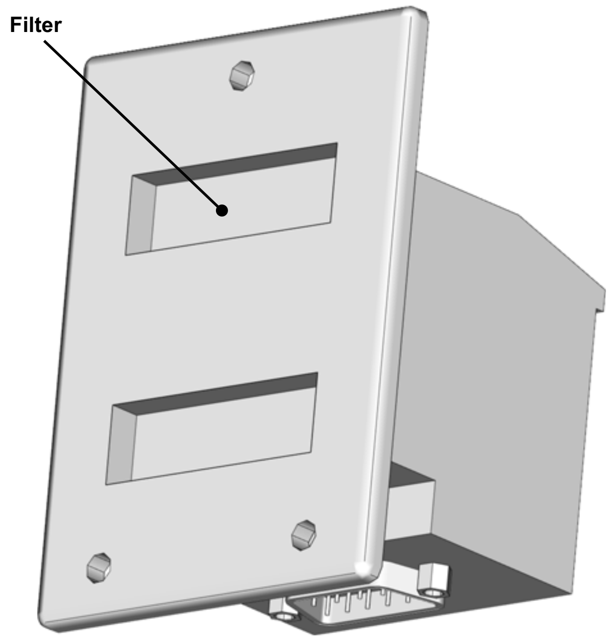


Figure 38: Ventilated Temperature Sensor

CONTROLS AND INDICATIONS

AIR PANEL

The COCKPIT, FWD and AFT CABIN temperature control selectors are used to set the desired temperature in automatic and manual modes.

A MAN TEMP PBA is to be used to select manual control of the temperature control system. An ON legend indicates the system is in manual control.

The FWD CARGO rotary switch is used to select LO HEAT or HI HEAT in the forward cargo compartment.

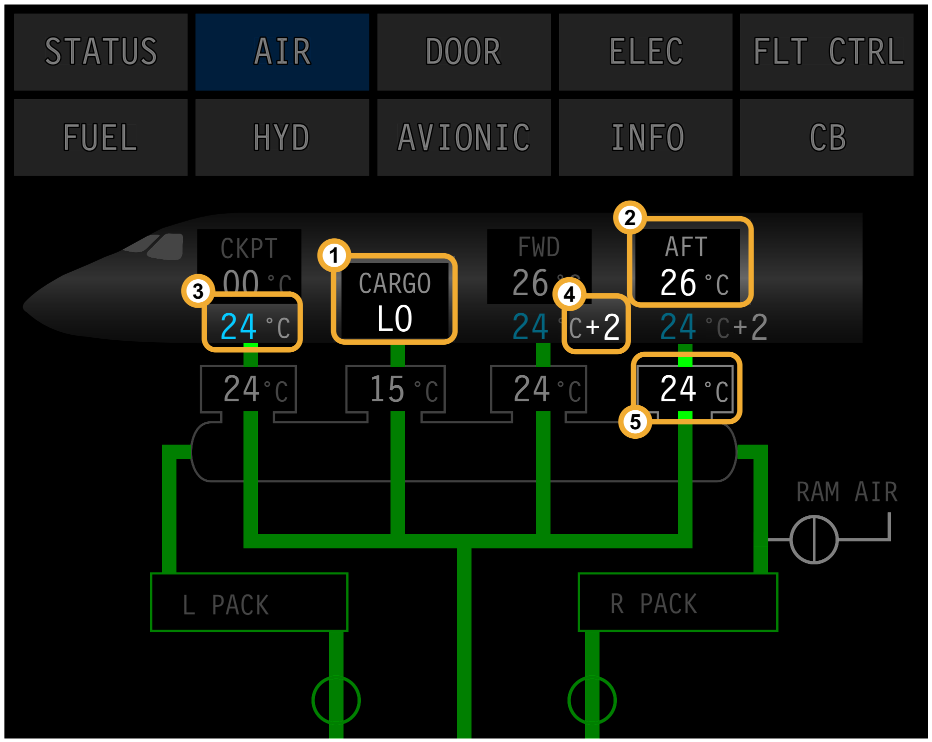
Figure 39: Temperature System Controls

AIR SYNOPTIC PAGE

The AIR synoptic page indicates the actual temperature for the flight deck, forward, and aft cabins on top. The selected temperature from the AIR panel is shown below the actual temperature.

If the selected temperature has been modified by the cabin management system (CMS) crew terminal temperature page selection, that information is displayed beside the selected temperature.

The duct temperature is indicated at the outlet of the mix manifold representation on the synoptic page.



SYNOPTIC PAGE – AIR

AIR SYNOPTIC PAGE INDICATIONS		
1	CARGO LO	Fwd Cargo Selection Indicator: • OFF – no heating, no ventilation • VENT – cargo ventilation only • LO HEAT – vent and low heating level (15-20°C) • HI HEAT – vent and warmer heating level (20-25°C)
2	AFT 26°C	Actual Cabin Temperature: • Dashed line "– – –" represents a temperature sensor failure or operating in manual mode
3	24°C	Selected Temperature: • Automatic mode – desired zone temperature • Manual mode – desired duct temperature
4	+2	Cabin Management System Input (+/-3°C)
5	24°C	Duct Temperature

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Figure 40: Temperature System Indications

CABIN MANAGEMENT SYSTEM TEMPERATURE CONTROLS

The current temperature levels, and the temperature adjustment for each cabin zone are displayed on the temperature screen. The allowable range is displayed for each cabin zone. The current temperature in each cabin is displayed on a layout of passenger area (LOPA) screen. If the temperature adjustment is not available, the temperature controls are disabled and an error message is displayed on the screen.

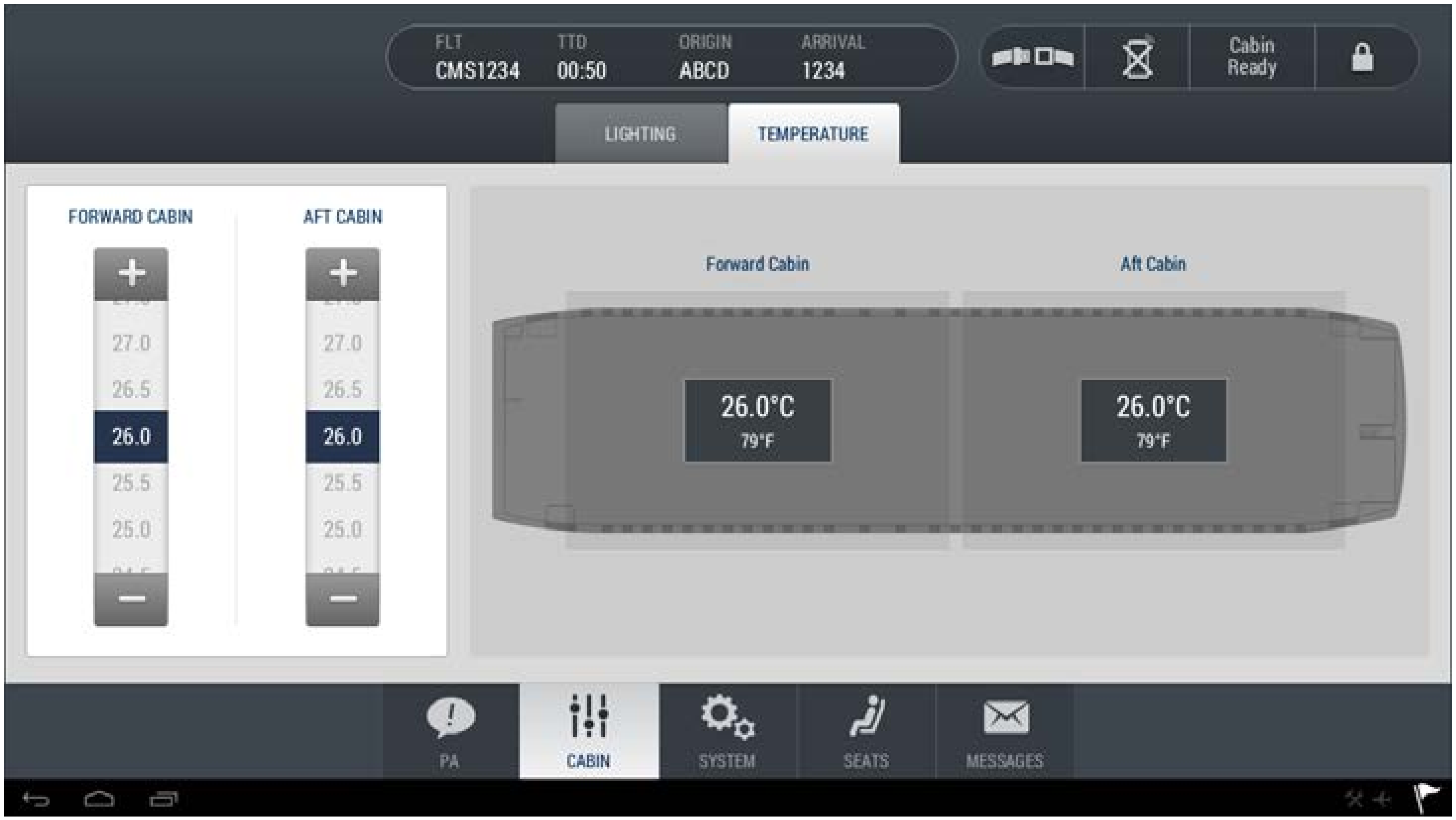

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Figure 41: Cabin Management System Temperature Controls

DETAILED DESCRIPTION

AUTOMATIC TEMPERATURE CONTROL

The automatic temperature control system controls the air conditioning packs, ram air regulating valve (RARV), recirculation system, and the trim air system to provide the selected temperature for the flight deck, the forward, and aft cabins, and the forward cargo compartment.

For zone temperature control, the integrated air system controllers (IASCs) use the following sensors:

- The ventilated temperature sensors (VENTS), located in the flight deck, forward cabin and aft cabin zones sense the actual zone temperature
- The duct temperature sensors (DTS), located in the main zone distribution lines downstream of the TAV, sense the temperature of the air supplied to each zone
- The mix temperature sensors (MIXTS) sense the temperature of the pack discharge and recirculation mixed air for the pack temperature control valve (TCV)
- The pack discharge temperature sensors (PDTS), located on pack discharge ducts, sense each pack discharge temperature. The PDTS is mainly used as a monitoring means, such as pack discharge overheat protection or as a backup to the MIXTS, the PTS or the PDPS for redundant icing protection.

The IASC modulates the TCV and RARV position to achieve the reference mix manifold temperature. The mix manifold reference temperature is set equal to the lowest reference temperature demand computed for each of the three independent zones.

The IASC controls each TAV in a closed loop against the duct temperature sensor measurement. The DTS reference temperature is computed from the difference between the ventilated temperature sensor measurement and the zone temperature selections made on the AIR panel.

Each TAV is commanded by its IASC according to the duct temperature sensor (DTS) measurement compared with a reference temperature determined by the error between the selected temperature from the AIR panel in the flight deck, and the temperature measured by the associated zone ventilated temperature sensor (VENTS).

- If the forward or aft cabin VENTS fails, the remaining VENTS is used for temperature control in both cabins
- If the flight deck VENTS or both the forward and aft cabin VENTS fails, the affected zones are controlled using the supply duct temperature as a reference similar to the manual temperature control
- If a temperature selector on the AIR panel fails, the system assumes a default value of 24°C (75°F) for that zone
- If the forward or aft DTS fails, the associated TAV is closed and the remaining zone provides temperature control for the entire cabin
- If the flight deck DTS fails, the TRIM AIR PBA must be selected OFF

If the flight deck, forward or aft cabin DTS measurement exceeds 85°C (185°F) for over 30 seconds, the trim air system is closed automatically, and a TRIM AIR FAIL caution message is posted and latched until the TRIM AIR PBA is selected OFF, and the overheat condition disappears.

MANUAL TEMPERATURE CONTROL

In manual mode, the flight crew can manually adjust the zone temperatures from the AIR panel by selecting the MAN TEMP PBA to ON.

The duct temperature sensors (DTS) are used for the reference temperature. The temperature in the duct can be controlled from 5°C (41°F) to 65°C (149°F).

When in manual temperature control, the selected temperature reading on the synoptic page becomes the DTS target, and the actual temperature displayed on the synoptic page becomes dashed.

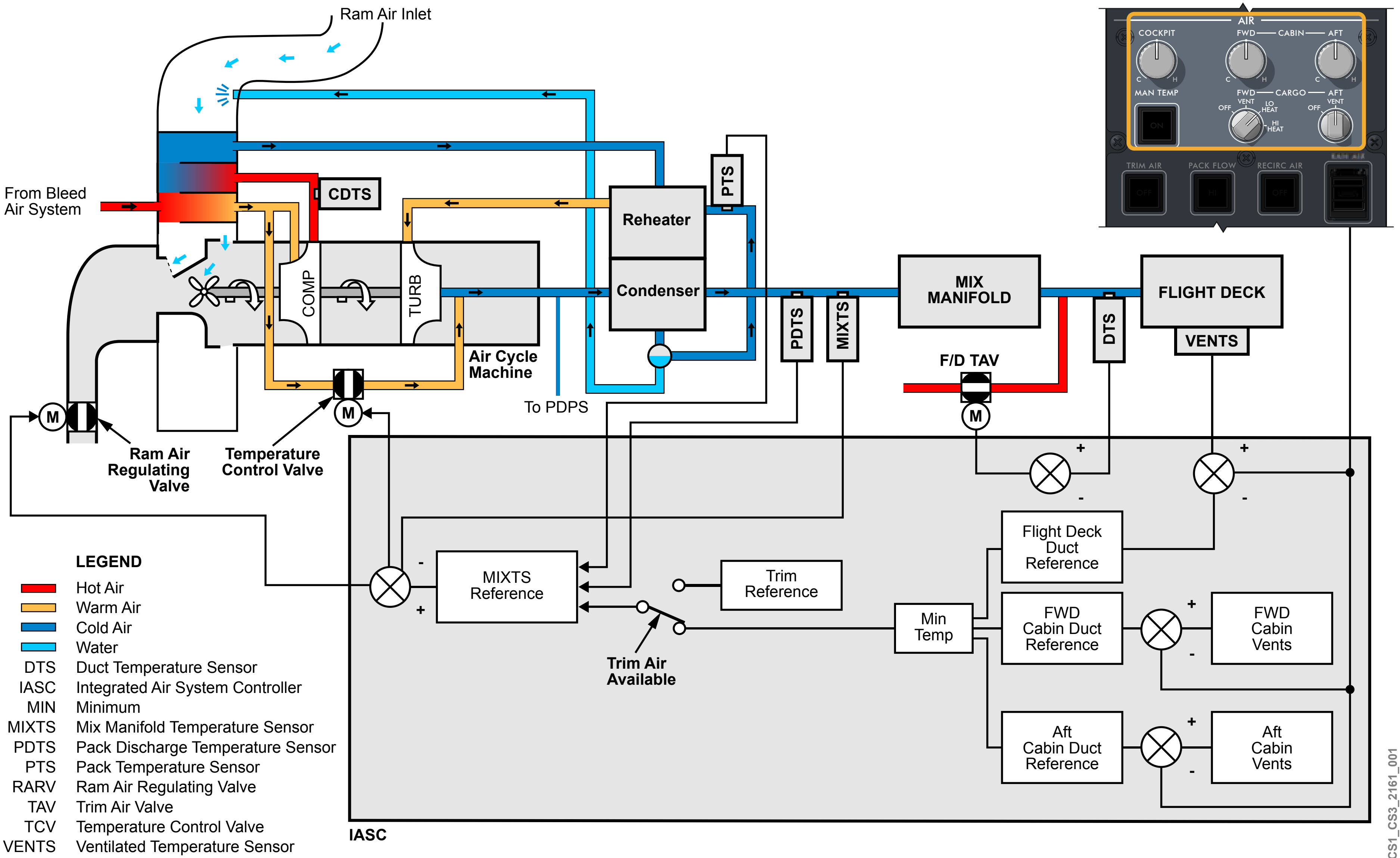


Figure 42: Flight Deck, Forward Cabin, and Aft Cabin Temperature Control System

FWD CARGO TEMPERATURE CONTROL

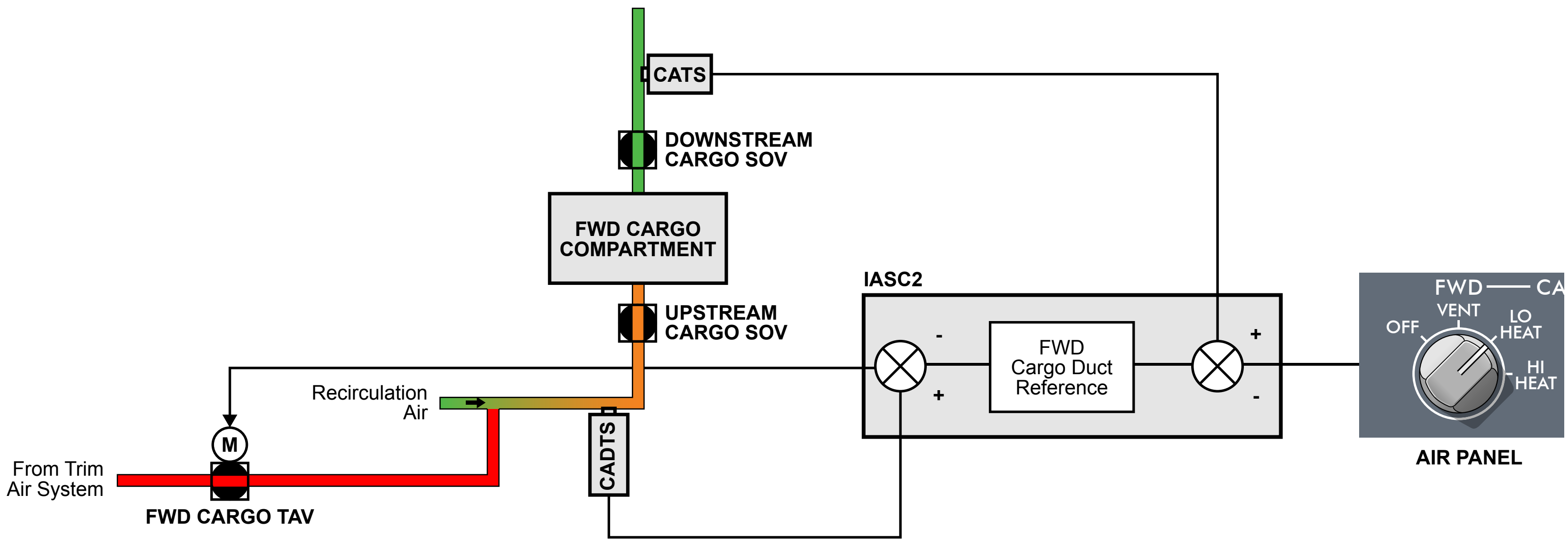
The TAV mixes hot air to the flow distributed from the recirculation system to the forward cargo in order to provide the required temperature.

The temperature is controlled in a closed loop by measurement feedback from the cargo temperature sensor (CATS) to IASC 2. The cargo duct temperature sensor (CADTS) provides temperature measurement to IASC 2 for overheat monitoring and protection. A zone target temperature is set, based on LO or HI heat selection and the duct temperature reference, based on the error between this target and the measured temperature.

If the cargo duct temperature exceeds 70°C (158°F) for 30 seconds, the trim air system is closed automatically. A TRIM AIR FAIL caution message is displayed until the TRIM AIR PBA is selected OFF, and the overheat condition disappears.

If the CADTS or CATS fails, the FWD CARGO HEAT is selected OFF.

When the FWD cargo selector is set to OFF or VENT, and the CARGO TAV full closed switch is not received, a TRIM AIR FAIL caution message is posted and latched until the TRIM AIR PBA is selected OFF.



LEGEND

	Hot Air
	Warm Air
	Cold Air
	Water
CADTS	Cargo Duct Temperature Sensor
CATS	Cargo Temperature Sensor
IASC	Integrated Air System Controller
SOV	Shutoff valve
TAV	Trim Air Valve

CS1_CS3_2161_002

Figure 43: Cargo Temperature Control

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages associated with the temperature control system:

CAS MESSAGES

Table 17: CAUTION Messages

MESSAGE	LOGIC
FWD HEAT CARGO FAIL	Total loss of CADTS or CATS.
FWD CARGO LO TEMP	In the event the FWD cargo is selected to LO or HI HEAT, and the CATS reads below 10°C for at least 2 minutes during cruise.

Table 18: ADVISORY Messages

MESSAGE	LOGIC
AIR SYSTEM FAULT	Fault not requiring immediate attention. (Total loss of VENTS, IASC single channel failure, loss of cockpit, FWD, AFT cabin DTS, TAV stuck, TAPRV/TASOV failed closed, single-channel CADTS, or CATS failed).
PACK FAULT	Loss of redundant or non-critical function for the air conditioning system.

Table 19: STATUS Messages

MESSAGE	LOGIC
FWD CARGO AIR OFF	FWD cargo heating selection selected and confirmed OFF.
AFT CARGO AIR OFF	AFT cargo heating selection selected and confirmed OFF.

Table 20: INFO Messages

MESSAGE	LOGIC
21 AIR SYSTEM FAULT - ZONE TEMP SNSR INOP	Cockpit VENTS total loss (both channels) OR Aft Cabin VENTS total loss (both channels) OR FWD Cabin VENTS total loss (both channels).
21 AIR SYSTEM FAULT - DUCT TEMP SNSR INOP	Total loss or drift of FWD CABIN DTS or AFT CABIN DTS or redundant loss or drift of CKPT DTS.
21 PACK FAULT - MIX MANF TEMP SNSR TOTAL LOSS	MMTS from both packs are lost.
21 PACK FAULT - MIX MANF TEMP SNSR REDUND LOSS	Up to three MMTS sensing elements redundant loss.

21-22 RECIRCULATION

GENERAL DESCRIPTION

Cabin air is recycled into the mixing manifold by mean of a single recirculation fan (RFAN), which draws air from the cabin underfloor across two air filters. The RFAN supplies the air into each pack discharge line for initial mixing with the pack discharge air, downstream of the bulkhead check valve.

Integrated air system controller (IASC) 1 controls the RFAN. If IASC 1 fails, IASC 2 controls the recirculation fan. The recirculation fan has a motor controller that the IASC signals to adjust the fan speed based mainly on cabin altitude. The fan is powered by AC BUS 2, and the controller is powered by DC BUS 2. The recirculation fan operation is fully automatic and runs when power is available, no faults exist, and there is no smoke in the equipment bays. The recirculation fan can be selected OFF using the RECIRC AIR PBA on the AIR panel. The RFAN is normally selected OFF when the fire detection and extinguishing (FIDEX) control unit reports smoke in the equipment bays.

The recirculation fan is commanded so that the recirculation flow ratio remains between 25% and 40% of the total air flow for normal operation, and between 35% and 50% of the total airflow for single pack operation. The recirculation flow also ensures the mixing manifolds temperature remains above freezing.

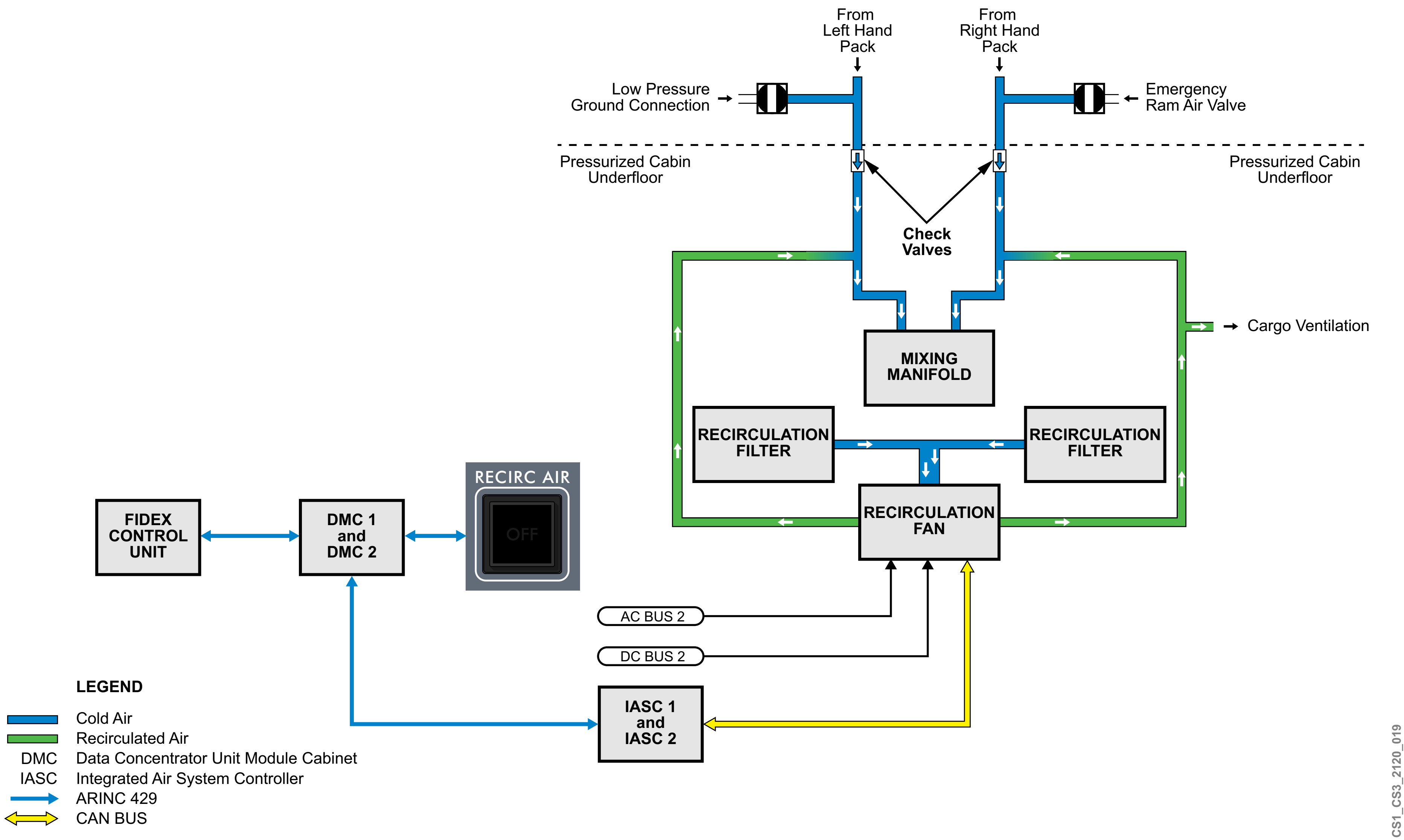


Figure 44: Recirculation System

COMPONENT LOCATION

The following components are located in the environmental control system bay, aft of the forward cargo compartment:

- Recirculation fan filters
- Recirculation fan and motor controller

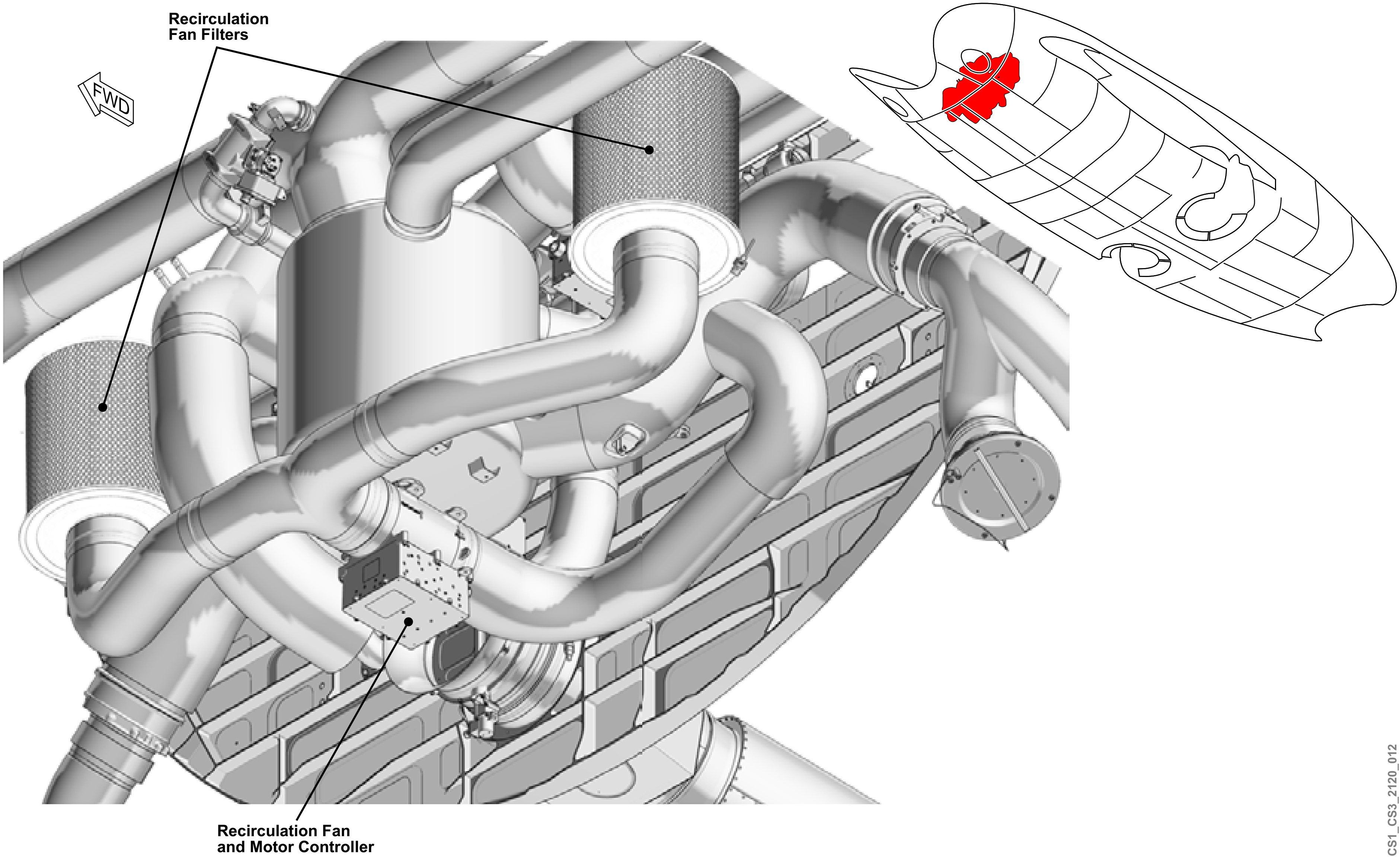


Figure 45: Recirculation System Component Location

COMPONENT INFORMATION

RECIRCULATION FILTERS

The disposable recirculation filter ensures cleaned air is recycled to the cabin. It is composed of an HEPA matrix, and an active charcoal core containing chemical absorbers used to control odors.

The recirculation filter is supported by a structure at the bottom and at the top. A V-band clamp holds the filter at the bottom, and a plunger centers the filter at the top.

WARNING

DO NOT TOUCH THE FILTER WITH BARE HANDS. USE GLOVES AND PERSONAL PROTECTIVE EQUIPMENT TO HANDLE THE FILTER. THE FILTER SHOULD BE CONSIDERED A BIOHAZARD. THE FILTER CAN CONTAIN GERMS AND VIRUSES THAT CAN CAUSE SICKNESS.

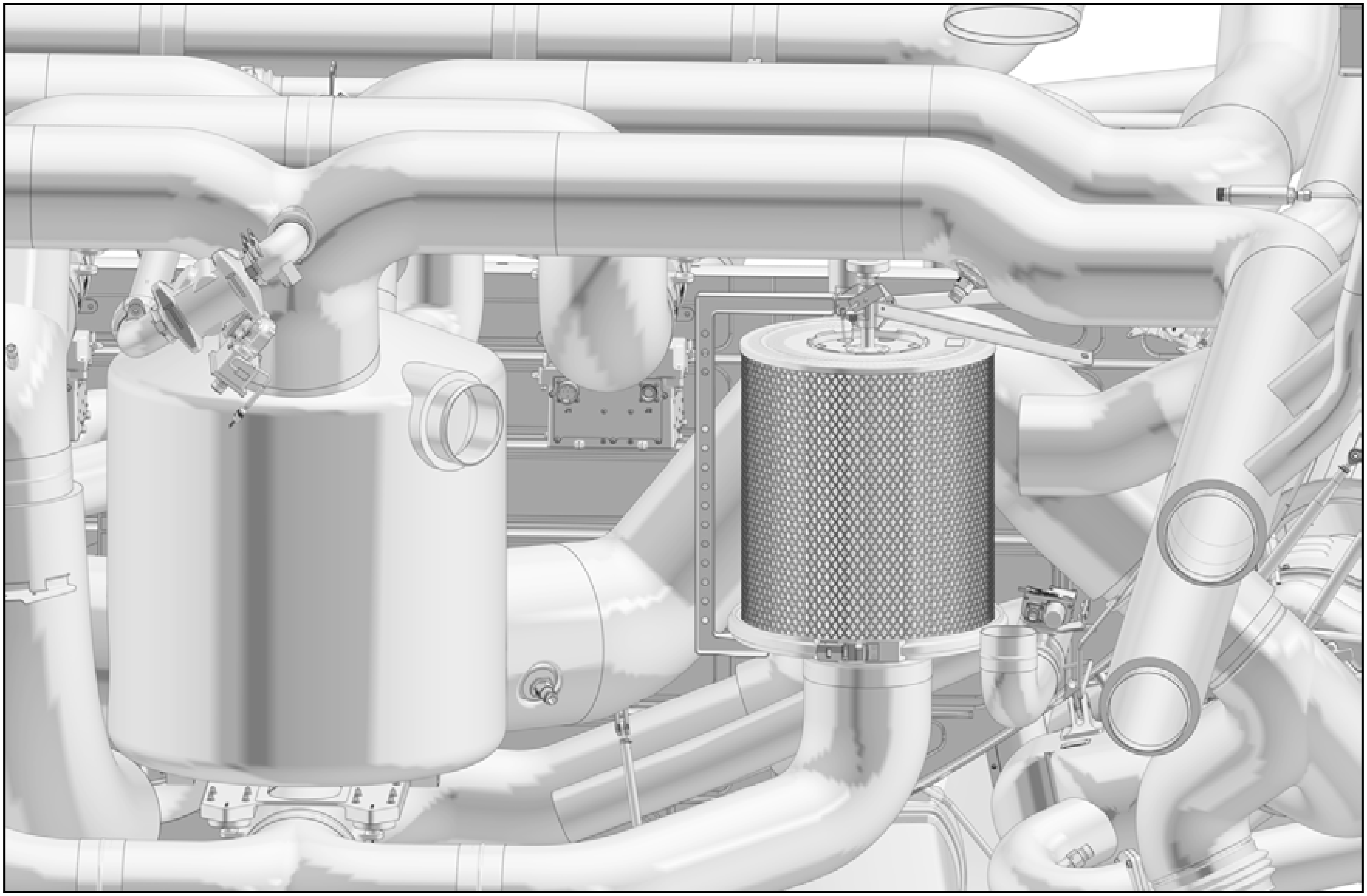


Figure 46: Recirculation Filter

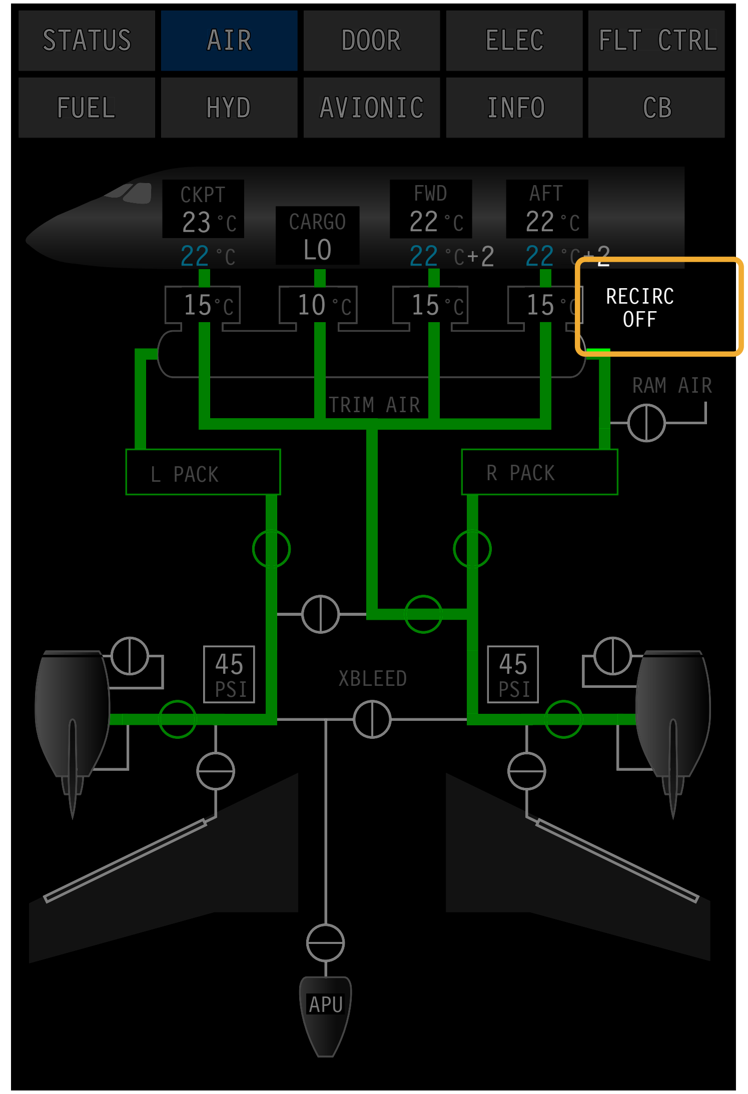
CONTROLS AND INDICATIONS

The RECIRC AIR PBA turns off the recirculation fan. An OFF legend on the RECIRC AIR PBA indicates the system is shutdown.

The RECIRC AIR PBA is used when there is a recirculation fan failure or smoke is detected in the avionics cooling air extraction system. The PBA is selected to match the system configuration.

A RECIRC OFF message appears on the AIR synoptic page to indicate the system is shutdown.

AIR PANEL



SYNOPTIC PAGE – AIR

CS1_CS3_2120_025

Figure 47: Recirculation System Controls and Indications

DETAILED DESCRIPTION

The RFAN 115 VAC power is supplied from AC BUS 2 through a relay in electrical power center (EPC 2). The relay is controlled by control and distribution cabinet 2 (CDC 2). CDC 2 monitors the relay position using the discrete feedback from a relay contact. The position information is sent to the IASCs through the data concentrator unit module cabinet (DMC). The recirculation system operation is fully automatic.

The RECIRC AIR PBA is used only in case of recirculation fan failure or in case of smoke detected in the avionics extraction system, and confirms shutdown of the recirculation system. The PBA commands the CDC 2 to isolate the 115 VAC power from the RFAN, bypassing IASC commands.

Each IASC communicates with the RFAN controller through a CAN BUS to automatically adjust the fan speed. The IASC controls the RFAN speed according to a schedule as a function of system configuration

- Number of packs operating
- Pull-up or pull-down sequence
- Cabin altitude
- Weight-on-wheels (WOW) signal
- Cargo ventilation status
- Aircraft version (CS100 or CS300)

NOTE

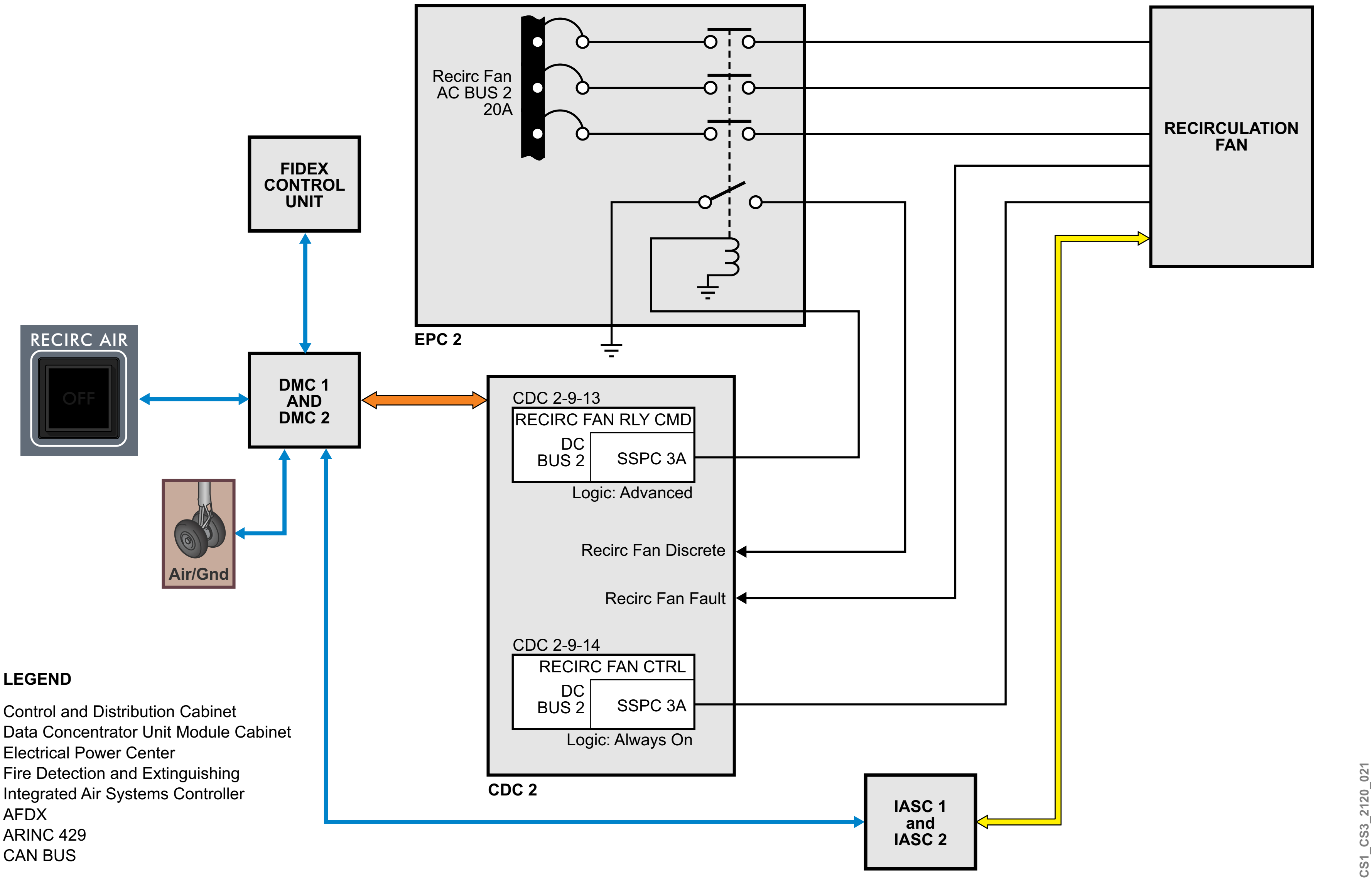
1. On the CS100, the recirculation fan is shutoff during the pull-up sequence.
2. On the CS300, the recirculation fan runs at maximum speed during the pull-up sequence.

If the FWD and AFT CARGO switches are selected OFF, with the aircraft operating on ground at a field elevation greater than 12,000 ft, the recirculation fan is automatically turned off by the IACSs to prevent excessive distribution noise.

The normal maximum fan speed is 15,800 rpm. If the cabin altitude is out of its normal range, the RFAN speed increases to 16,000 rpm.

The RFAN provides protection against internal overtemperature, overcurrent, and overvoltage. In case of a major internal failure, the RFAN is automatically turned off, and latched until the RFAN power is recycled.

Once the RFAN fault is detected by the IASCs either from the controller or from the solid-state power controller (SSPC) feedback, an RECIRC AIR FAIL caution message is displayed.



LEGEND

CDC	Control and Distribution Cabinet
DMC	Data Concentrator Unit Module Cabinet
EPC	Electrical Power Center
FIDEX	Fire Detection and Extinguishing
IASC	Integrated Air Systems Controller
⟷ (orange)	AFDX
⟷ (blue)	ARINC 429
⟷ (yellow)	CAN BUS

Figure 48: Recirculation Fan Detailed Description

MONITORING AND TEST

The following page provides the crew alerting system (CAS) and INFO messages associated with the recirculation system:

CAS MESSAGES

Table 21: CAUTION Message

MESSAGE	LOGIC
RECIRC AIR FAIL	RFAN internal failure detected. RFAN digital bus communication failure detected. Loss of both IASC B channels with RFAN powered and IASC safety channels remain functional.

Table 22: STATUS Message

MESSAGE	LOGIC
RECIRC AIR OFF	Recirculation air selected OFF.

21-22 DISTRIBUTION SYSTEM

GENERAL DESCRIPTION

The low-pressure distribution system consists of a network of rigid and flexible composite ducts, which collects air from the air supply sources, mixes fresh conditioned air with recirculation air in a mixing manifold unit.

Air conditioning is normally supplied from two air conditioning packs through low-pressure distribution lines. Air conditioning can also be supplied from a low-pressure ground cart through the low-pressure ground connection.

An emergency ram air valve connects the ram air inlet to the low-pressure distribution system just upstream the bulkhead check valves in flight. This provides ventilation to the cabin if both air conditioning packs fail.

Two bulkhead check valves isolate the pressurized cabin from the bleed air and air conditioning packs installed in the unpressurized wing-to-body fairing (WTBF). In the event of a distribution line rupture outside of the pressurized compartment, the bulkhead check valves close to minimize pressure loss in the aircraft.

Air from the air conditioning packs is mixed with recirculation air just before entering the mix manifold.

The mix manifold has three main outlets which split the flow for distribution. The air is distributed to three zones; flight deck, forward cabin, and aft cabin through dedicated distribution lines.

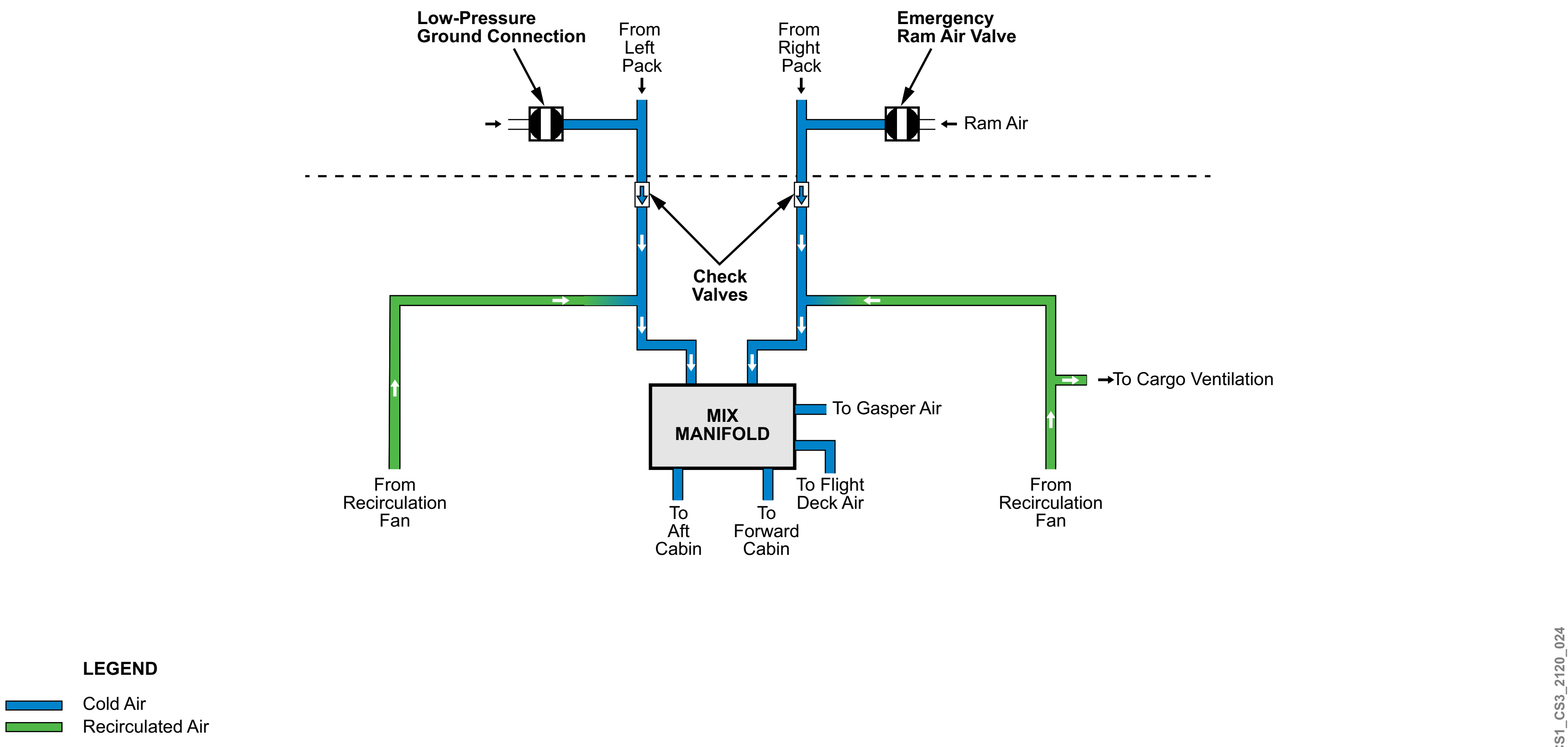


Figure 49: Distribution System General Description

ZONE DISTRIBUTION

Both forward and aft cabin zones and the flight deck are supplied independently from the mixing manifold, allowing for three independent zone temperature controls in the aircraft.

The air is ducted by risers into overhead stowage bins on both sides of the cabin, and then discharged into a plenum that distributes air at each cabin row from the top, the bottom, and the side of the bins.

On each side of the cabin, an underfloor distribution plenum connects the risers for the flight deck, the forward cabin, and the aft cabin with the mixing manifold. The end of each plenum connects with the galley and lavatory areas located at both ends of the cabin.

Conditioned air is distributed via a dedicated gasper air line. The line runs from the mix manifold to risers on the right and left sides of the aircraft. The risers connect to the overhead stowage bins for distribution to the gasper air outlets on the passenger service units.

CS100 Cabin Distribution

There are 10 risers on each side for the forward cabin zone. The farthest forward riser on each side is dedicated to the forward galley area. The aft riser on each side provides extra flow for the overwing emergency exit (OWEE) doors area ventilation.

There are 12 risers on each side for the aft cabin zone. The farthest aft riser on each side is dedicated to the aft galley area.

CS100 Gasper Air

The CS100 has five gasper risers. The three on the right and two on the left connect the mix manifold gasper air outlet to the overhead stowage bins.

Lavatory and Galley Areas

The air distribution is provided by air outlets located in the galley and lavatory areas. The ventilation flow for the galleys and lavatories is provided at a lower rate than the extraction flow to prevent odors from the galleys and lavatories flowing toward the passenger seating area.

Flight attendant gasper air outlets in the galley and entryway areas are connected to the distribution lines. There are provisions for three gasper outlets for the forward zone, and four gasper outlets for the aft zone.

Flight Deck

The flight deck air is distributed by a dedicated duct line that runs along the left side underfloor of the cabin and supplies the flight deck. For the CS100, the flight deck/cabin flow split is 14%/86% respectively.

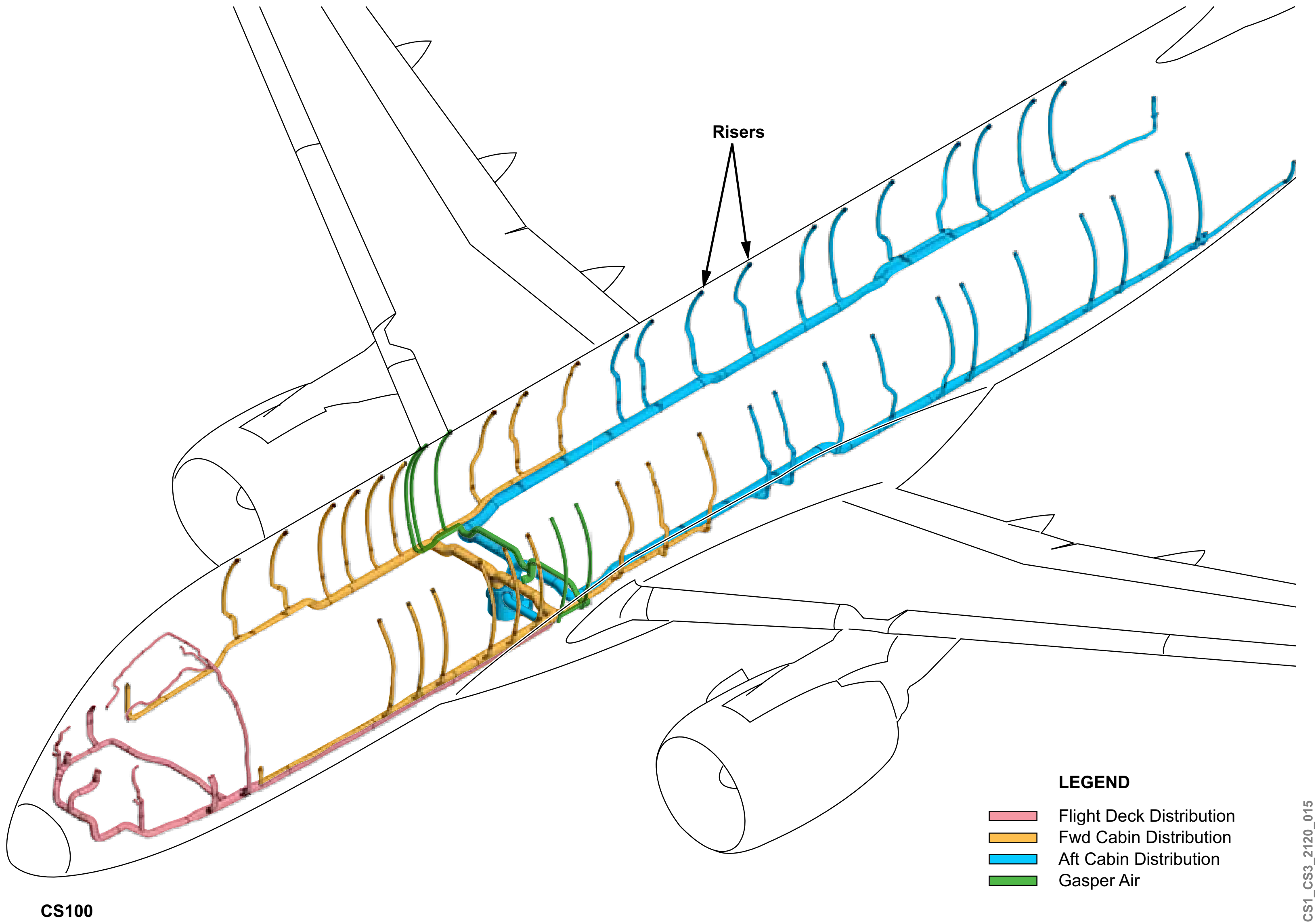


Figure 50: Zone Distribution CS100

CS300 Cabin Distribution

There are 12 risers on each side for the farthest forward cabin zone. The forward riser on each side is dedicated to the forward galley area. The most aft riser on each side provides extra flow for the overwing emergency exit (OWEE) doors area ventilation.

There are 12 risers on each side for the aft cabin zone. The farthest aft riser on each side is dedicated to the aft galley area.

For the CS300, the flight deck/cabin flow split is set at 12.5%/87.5% respectively.

CS300 Gasper Air

The CS300 has seven gasper risers. The four on the right and three on the left connect the mix manifold gasper air outlet to the overhead stowage bins.

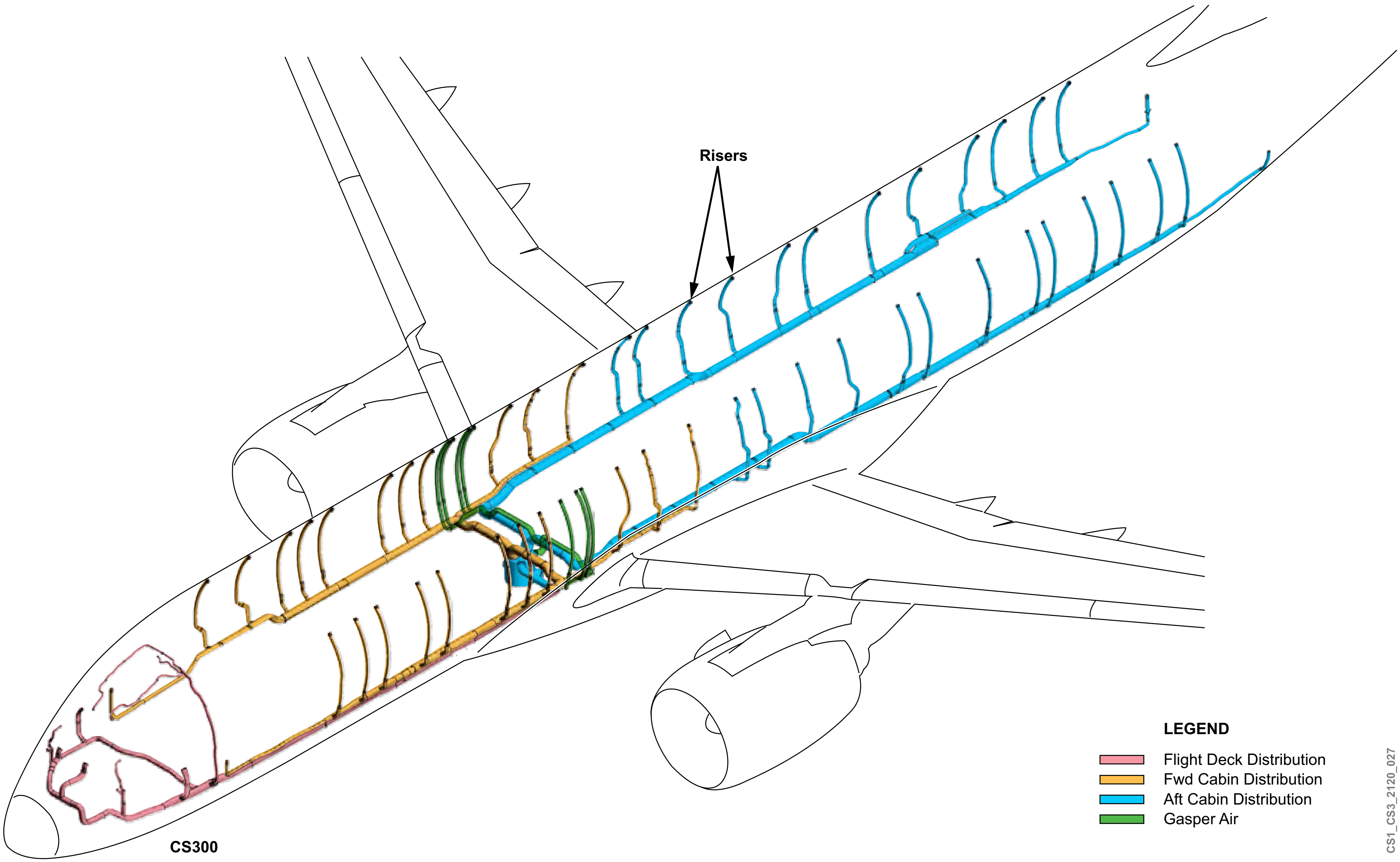


Figure 51: Zone Distribution CS300

OVERHEAD BIN DISTRIBUTION

The forward and aft cabin air is distributed from the top of the cabin through airflow channels integral to the passenger baggage overhead bins. The low-pressure distribution network connects to the bins by means of duct risers located of both side of the inner fuselage.

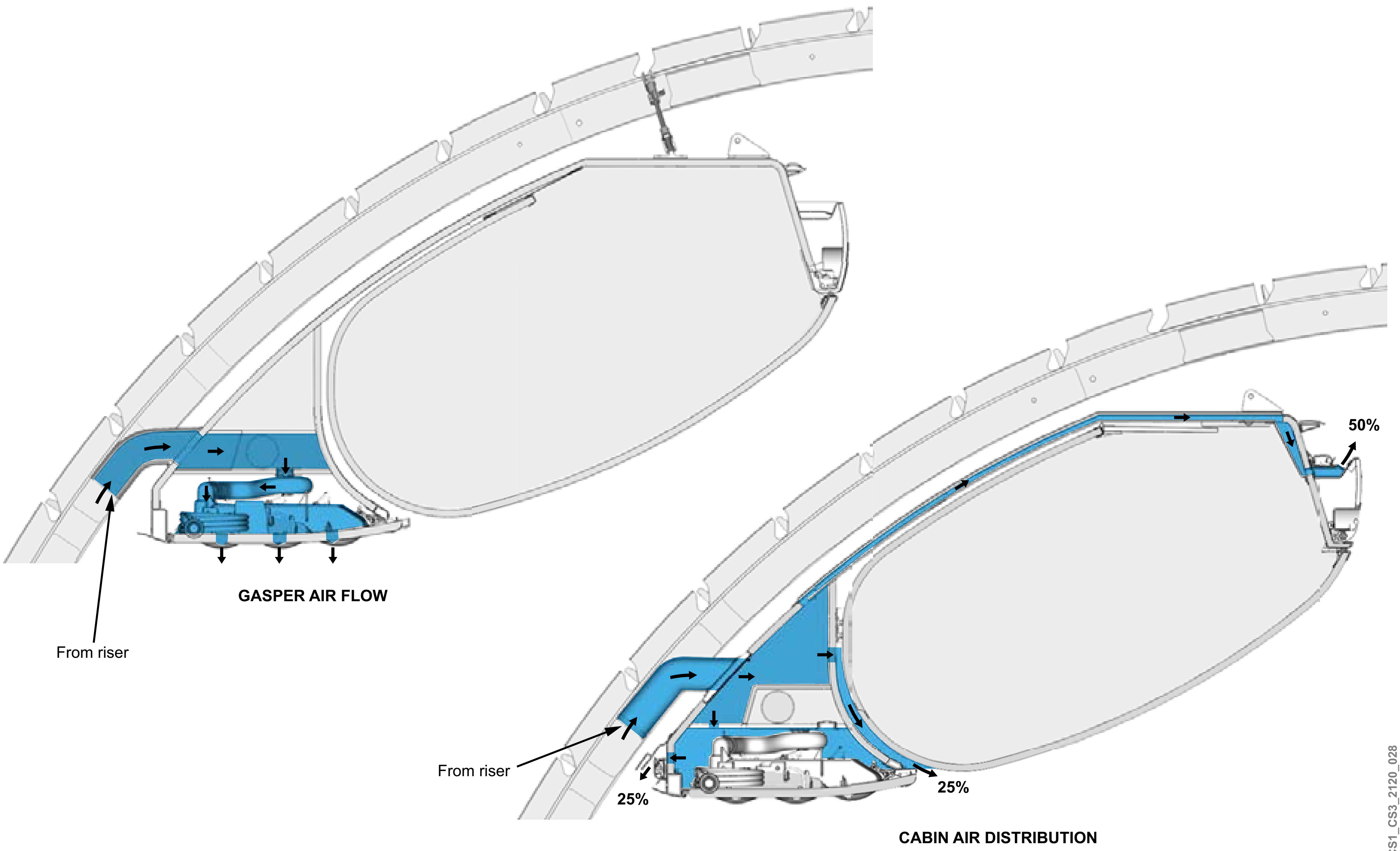


Figure 52: Overhead Bin Distribution

FLIGHT DECK DISTRIBUTION

From the mix manifold, air is routed to the flight deck via underfloor low-pressure ducting, passing into the flight deck within a single duct to provide ventilation for the crew via gaspers as well as general ventilation via diffusers in the side panels. Each gasper outlet is adjustable for flow and direction. The ventilation air is also passed into the flight deck through the glareshield.

The air supplied to the flight deck is ducted up through the floor at the center console, and routed to provide additional ventilation behind the control panels.

In the flight deck the following air outlets are provided:

- Large upward facing outlet in each side console
- Gasper outlets above the pilot, copilot, and observer head ventilation
- Gasper outlet on each side of the forward instrument panel
- Gasper outlet on each side of the floor pan at foot level

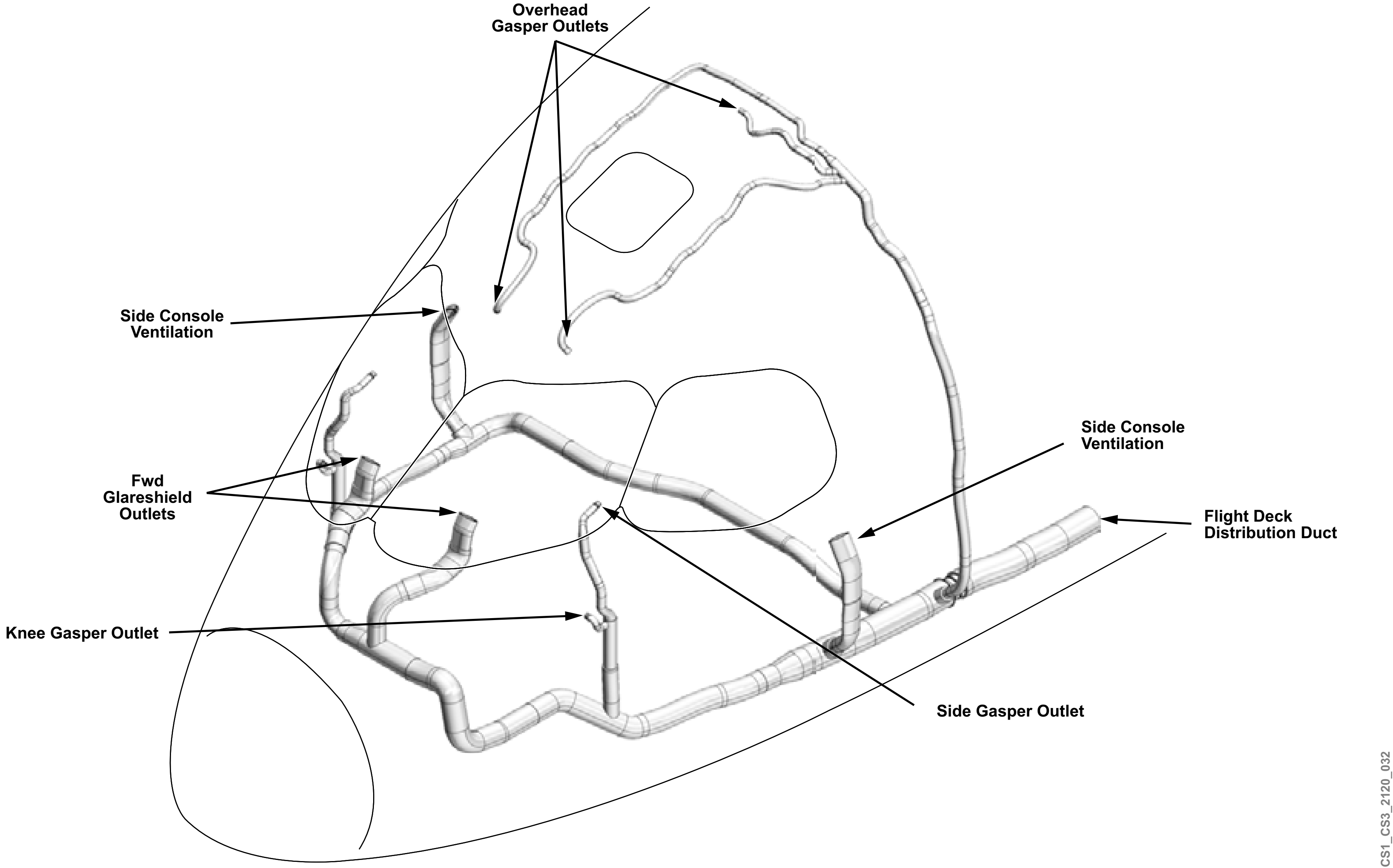


Figure 53: Flight Deck Distribution

COMPONENT LOCATION

The following distribution system components are located in the environmental control system compartment, aft of the forward cargo compartment.

- Mix manifold
- Bulkhead check valve
- Low-pressure ground connection

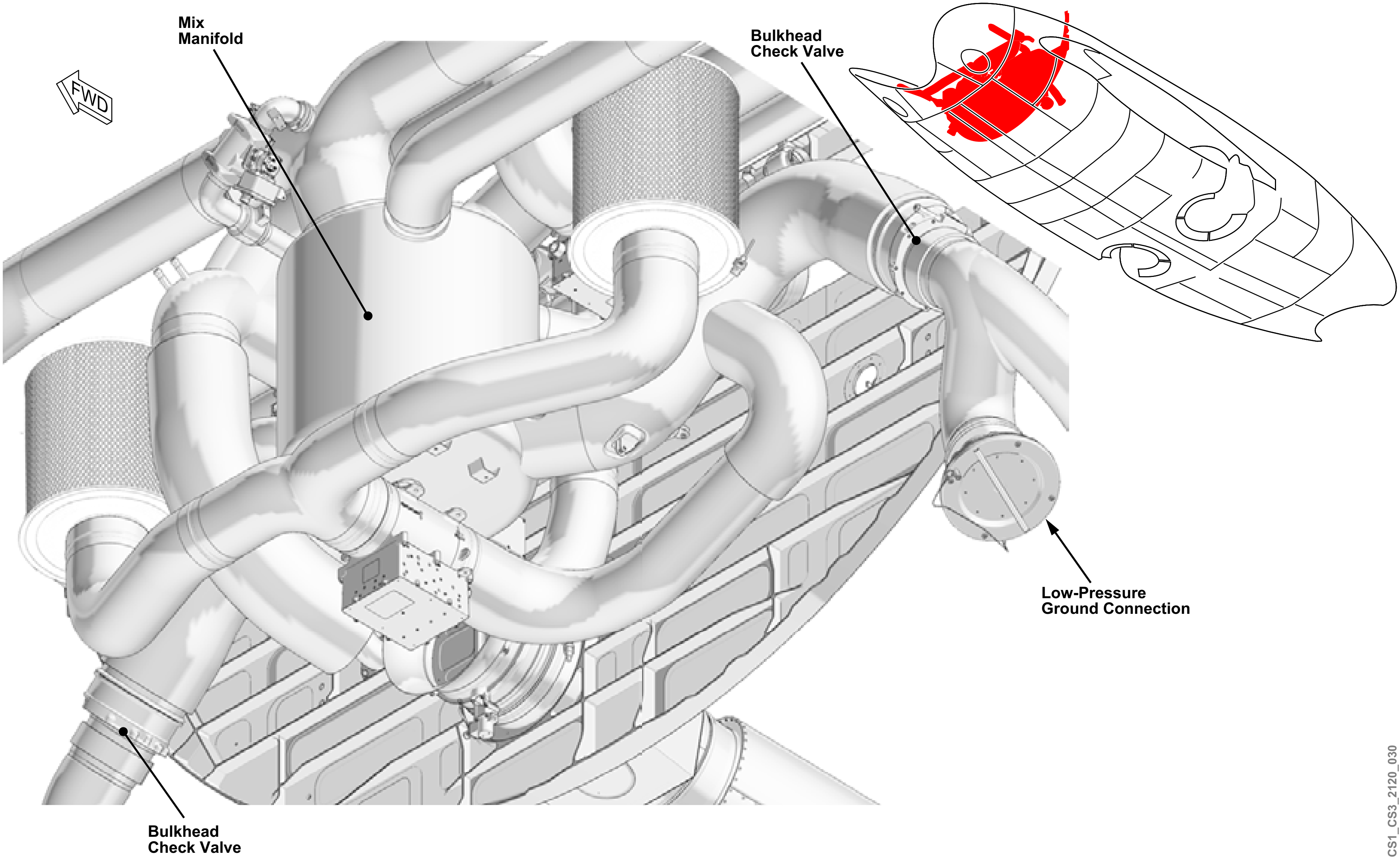


Figure 54: Distribution System Component Location

COMPONENT INFORMATION

LOW-PRESSURE GROUND CONNECTION

The low-pressure ground connection allows the connection of an external low-pressure ground cart to the aircraft during ground operations. The low-pressure ground cart supplies the aircraft cabin with conditioned air using the aircraft low-pressure distribution system. The low-pressure ground connection consists of an adapter, flapper check valve, and a cover.

In flight, when the cabin is pressurized, the flapper check valve and adapter are designed to withstand differential pressure of 8.9 psid, which is slightly higher than the differential pressure seen by the pressurized vessel.

A cover provides protection from water entering the main distribution system in the event of an aircraft ditching. The cover is held in place by a pin that remains attached to the aircraft by a lanyard.

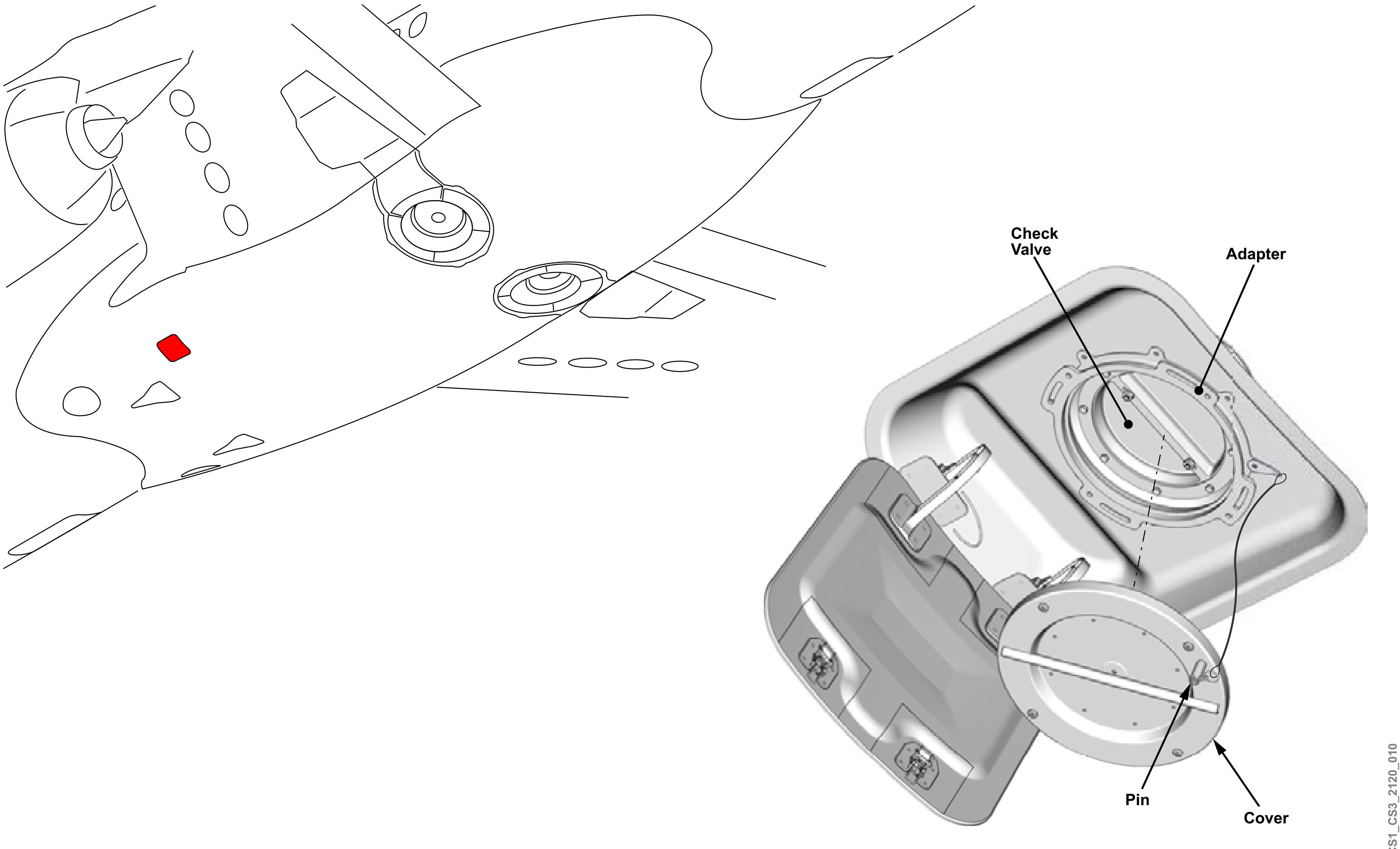


Figure 55: Low-Pressure Ground Connection

MIX MANIFOLD

The mix manifold acts as a mixing chamber achieving, the required air temperature for distribution. Two air-conditioning packs feed fresh air into the mix manifold. This fresh air mixes with recirculated air collected from the underfloor area near the outflow valve. Some pre-mixing occurs prior to the mix manifold from the junction of the recirculation ducts and the pack ducts.

The mix manifold has four main outlets, which split the flow for distribution to the forward cabin, aft cabin, flight deck, and gasper air.

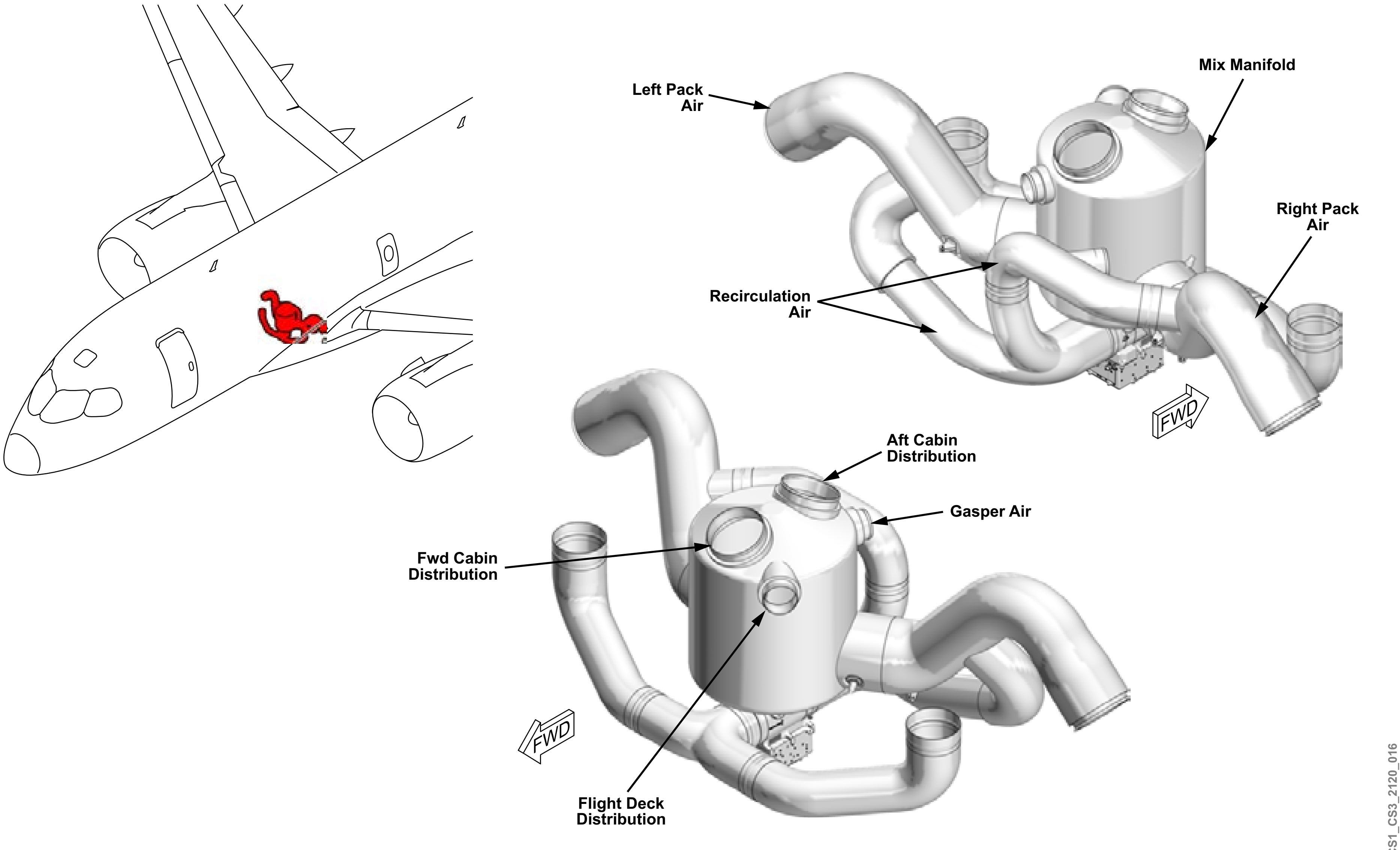


Figure 56: Mix Manifold

21-23 VENTILATION SYSTEM

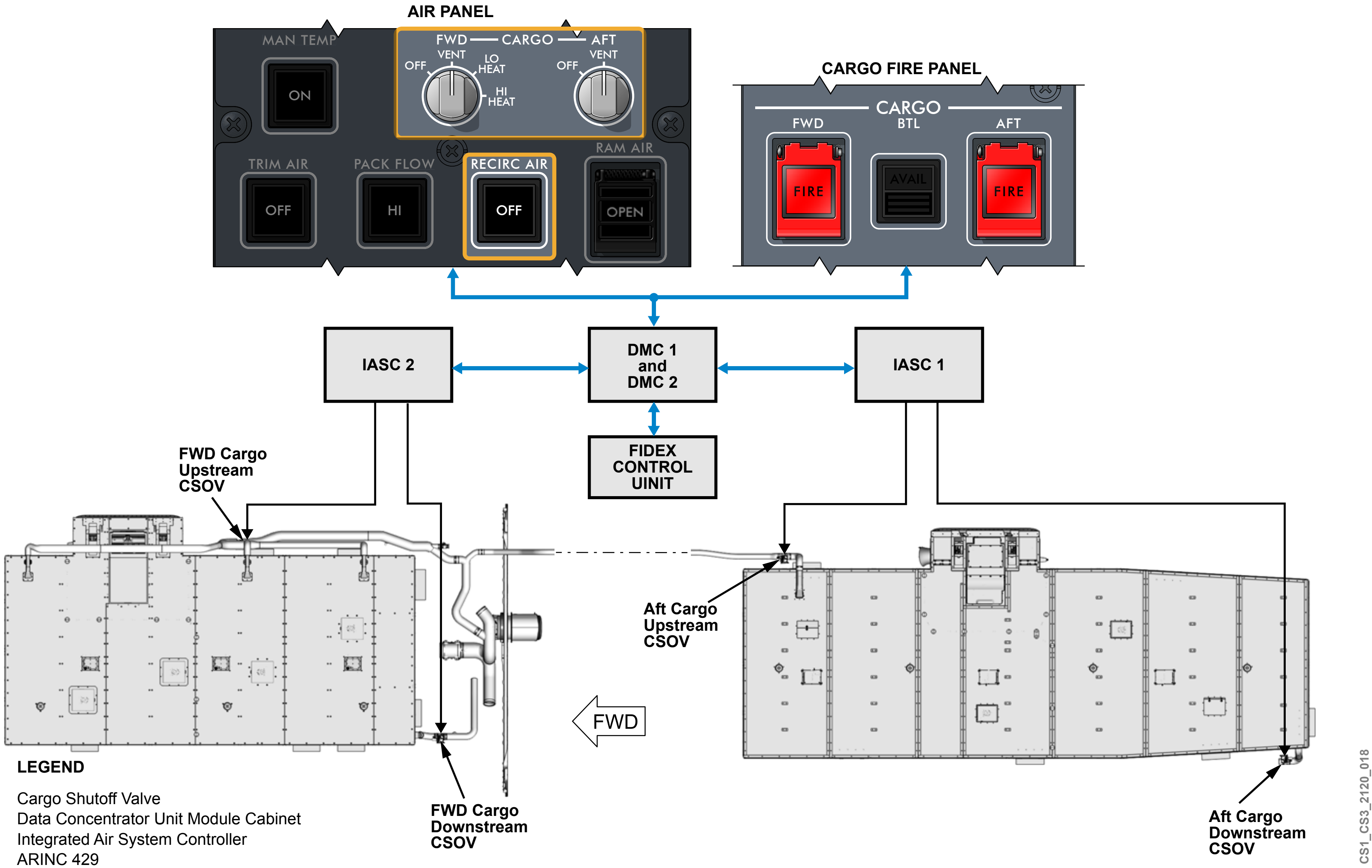
GENERAL DESCRIPTION

Air tapped off the recirculation air ventilates the cargo compartments through a cargo shutoff valve (CSOV). Each cargo compartment has two CSOVs. One is installed in the inlet duct, and the other is installed at the opposite end of the compartment in the exhaust duct. The CSOV is an electrically-actuated butterfly valve. Microswitches report the open and closed positions. The valve remains in its last position if no power is supplied.

The airflow maintains the temperature above 4°C in the forward cargo, and 2°C in the aft cargo. The air is exhausted into the underfloor area surrounding the cargo compartments. The forward cargo compartment CSOV exhausts air close to the outflow valve to avoid recirculating odors when transporting live cargo.

The valves are open for normal operation. When the recirculation fan is selected OFF, the integrated air system controllers (IASCs) close the CSOVs. If the forward cargo heat is selected, the forward CSOVs remain open, and air from the air conditioning units mix with cargo trim air.

The CSOVs isolate the cargo compartment when smoke is detected and reported by the fire detection and extinguishing (FIDEX) control unit. When the CARGO FIRE PBA is pressed, the IASCs close the CSOVs for the selected cargo compartment.



LEGEND

CSOV	Cargo Shutoff Valve
DMC	Data Concentrator Unit Module Cabinet
IASC	Integrated Air System Controller
↔	ARINC 429

Figure 57: Ventilation System Component Location

COMPONENT LOCATION

The following component is installed in the ventilation system:

- Cargo shutoff valves (CSOVs)

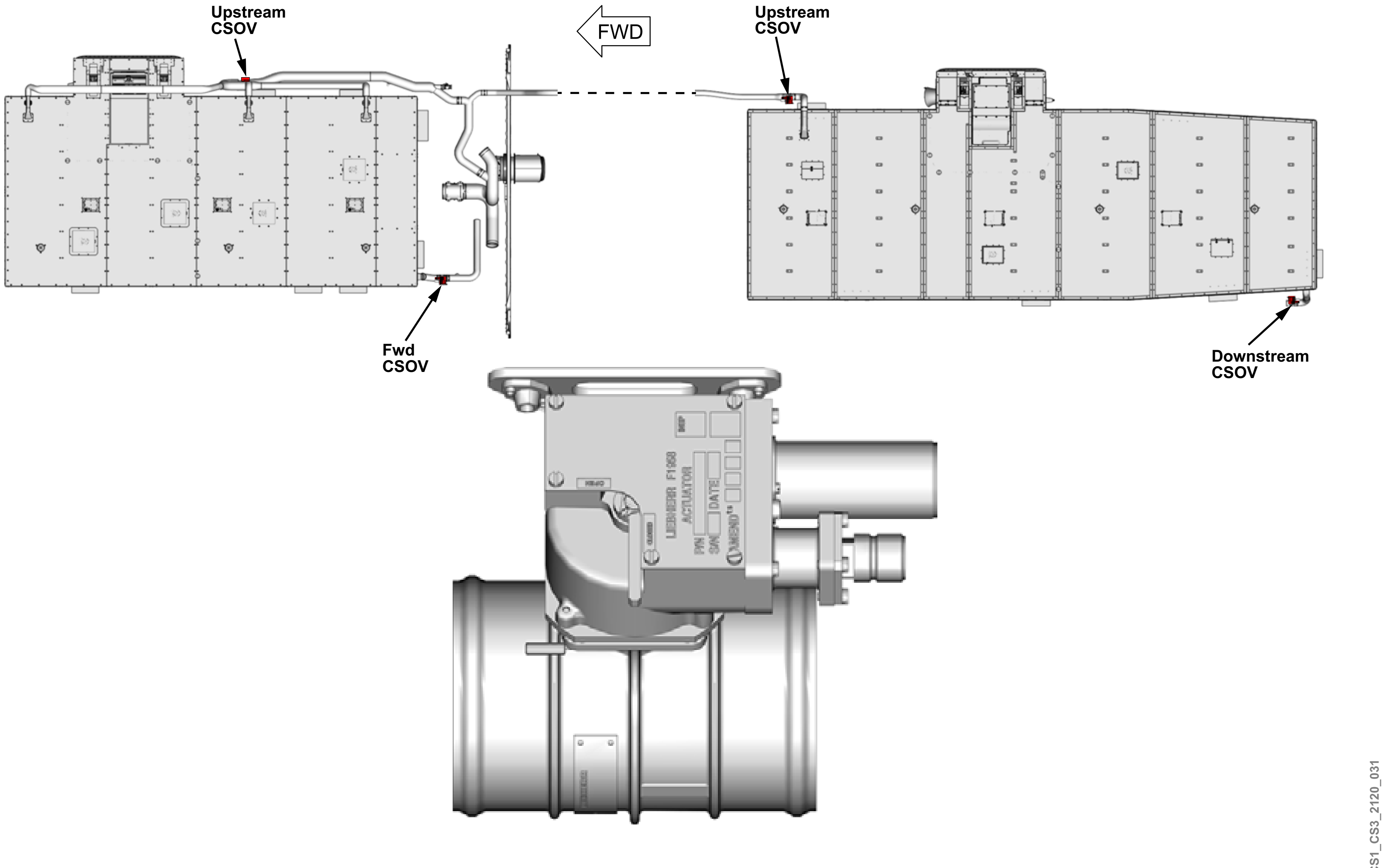


Figure 58: Cargo Shutoff Valves Component Location

CONTROLS AND INDICATIONS

AIR PANEL

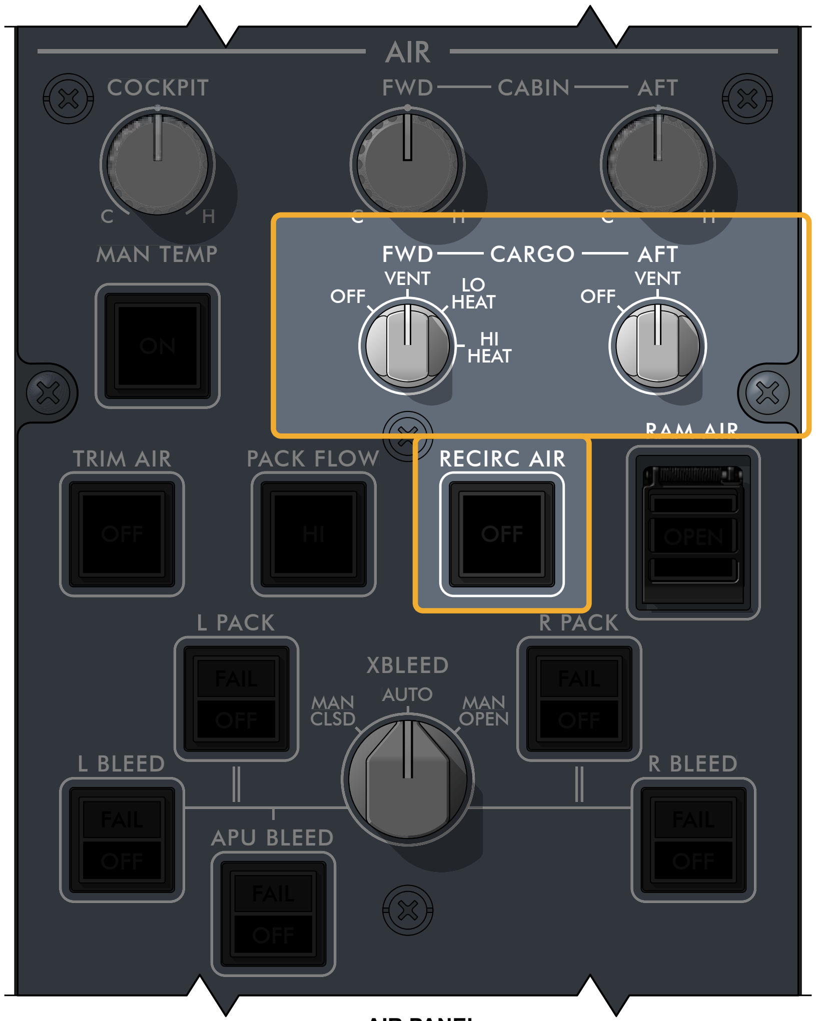
The forward and aft cargo compartment ventilation is controlled from the AIR panel.

The forward CSOVs open anytime the FWD CARGO switch is selected to VENT, LO HEAT, or HI HEAT.

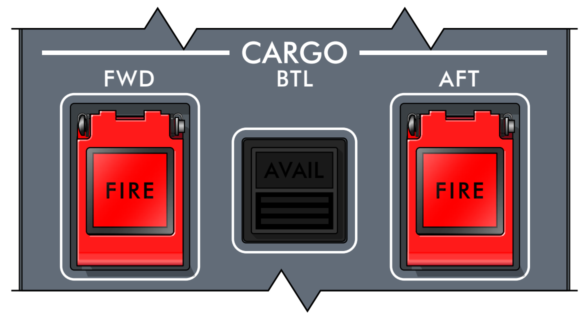
The aft CSOVs open when the AFT CARGO switch is selected to VENT.

CARGO PANEL

If either FIRE PBA is pressed on the CARGO panel, the CSOVs are driven closed by the IASCs.



AIR PANEL



CARGO PANEL

CS1_CS3_2120_033

Figure 59: Ventilation System Controls

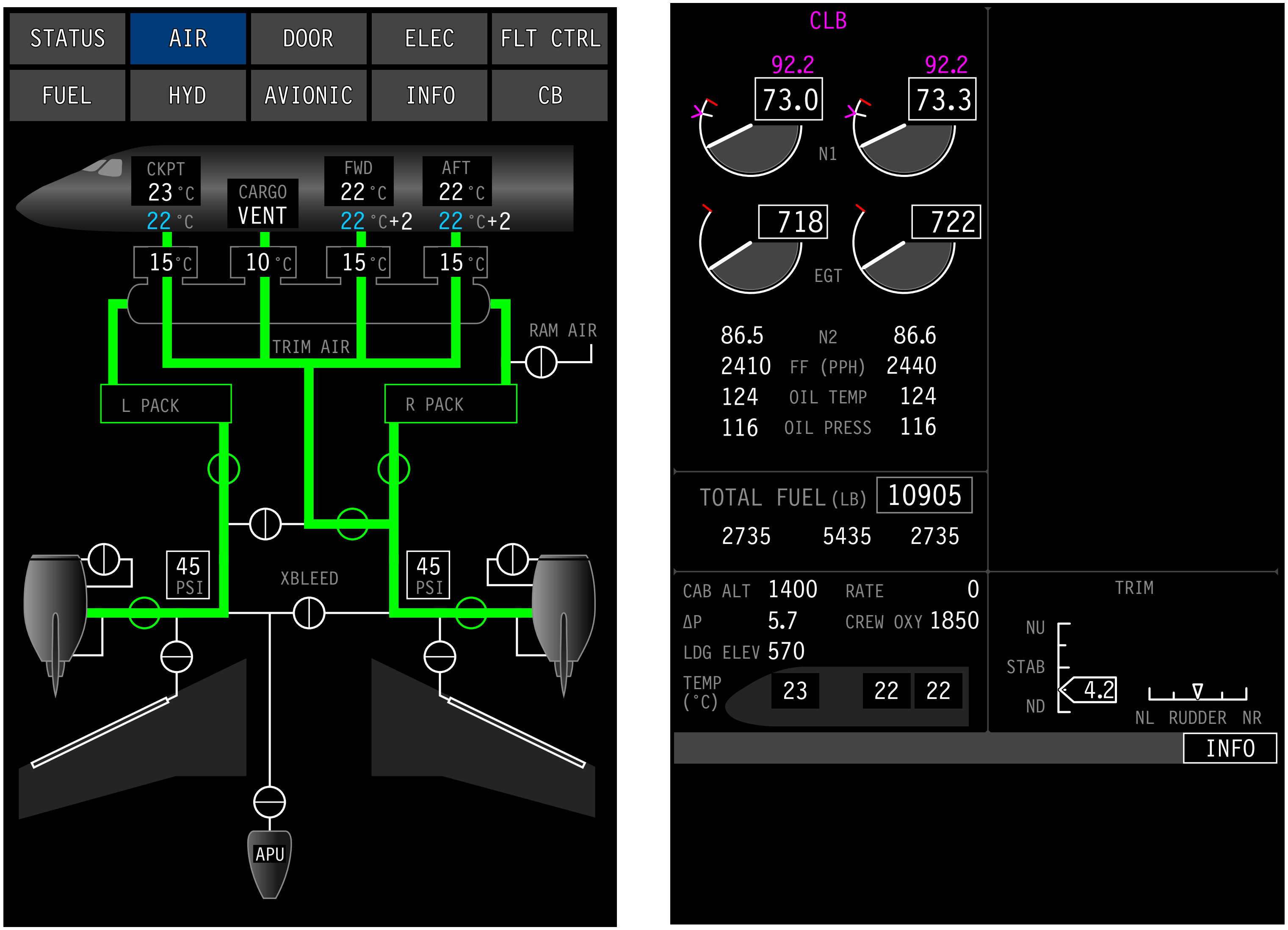
INDICATIONS

Synoptic Page

The FWD CARGO switch selection; OFF, VENT, LO, or HI is displayed on the AIR synoptic page.

EICAS Page

The forward cargo temperature indication on the engine indication and crew alerting system (EICAS) page is grayed out when the FWD CARGO switch is set to OFF or VENT.



SYNOPTIC PAGE – AIR

EICAS

Figure 60: Ventilation System Controls and Indications

DETAILED DESCRIPTION

The cargo shutoff valves are tested at landing by the IASCs, which perform a full open-close cycle to check the motor operation as well as the full close and full open position feedback.

Only the in-control channel driver of each IASC is tested during the test. If a failure is detected, switching to the other channel is performed, and the CSOV cycling is commanded again.

During normal operation, if the CSOV does not respond to the IASC command, an INFO message is displayed

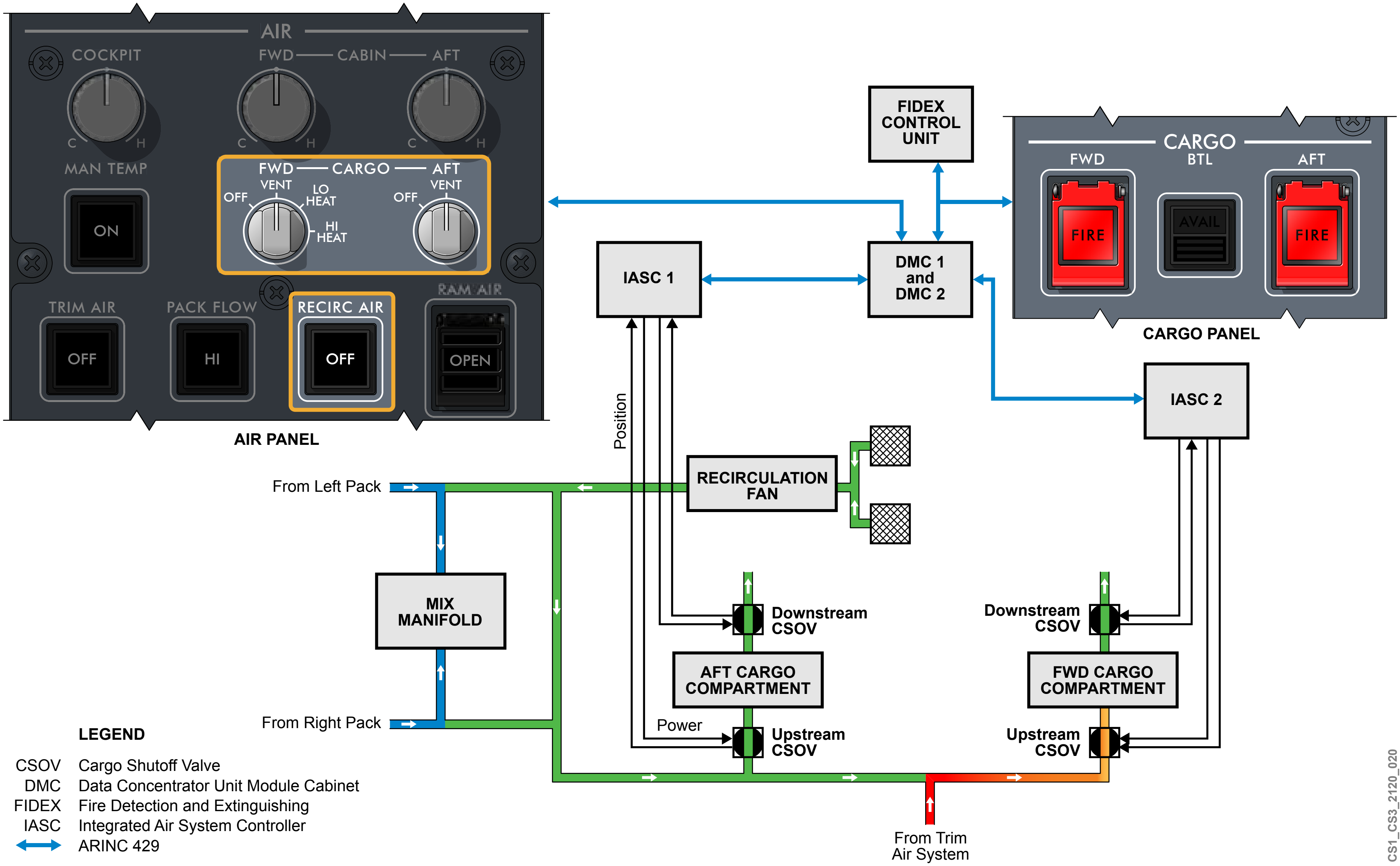


Figure 61: Ventilation System Detailed Description

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages associated with the ventilation system:

CAS MESSAGES

Table 23: STATUS Messages

MESSAGE	LOGIC
FWD CARGO AIR OFF	FWD cargo heating selection selected and confirmed OFF.
AFT CARGO AIR OFF	AFT CARGO selected and confirmed OFF.

Table 24: INFO Messages

MESSAGE	LOGIC
21 AIR SYSTEM FAULT - FWD CARGO SOV INOP	Upstream of downstream forward cargo SOV failed in position or in unknown position.
21 AIR SYSTEM FAULT - AFT CARGO SOV INOP	Upstream of downstream forward cargo SOV failed in position or in unknown position.

PRACTICAL ASPECTS

The cargo shutoff valve (CSOV) has a manual lever used to secure in the closed position before dispatch if the valve has failed.

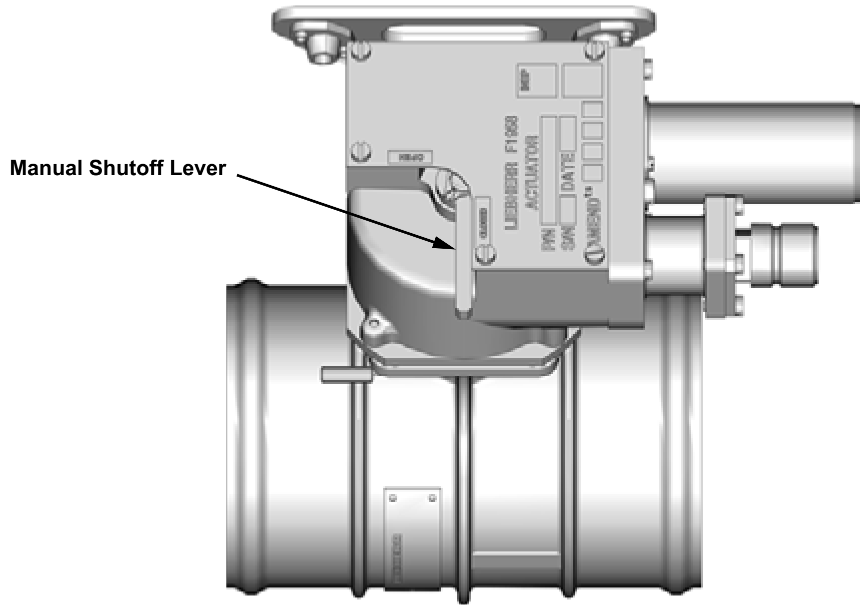


Figure 62: Cargo Shutoff Valve Practical Aspects

21-40 HEATING

FLIGHT CREW FLOOR HEATING

GENERAL DESCRIPTION

The flight crew floor heating system consists of heated mats embedded in each foot rest plate. The heated mats operate as soon as the main DC buses are powered. Thermostats embedded in the mats control the operation of the mats.

COMPONENT LOCATION

- Heated mat

HEATED MAT

The heated mats are installed in the foot rest plates in front of each pilot seat.

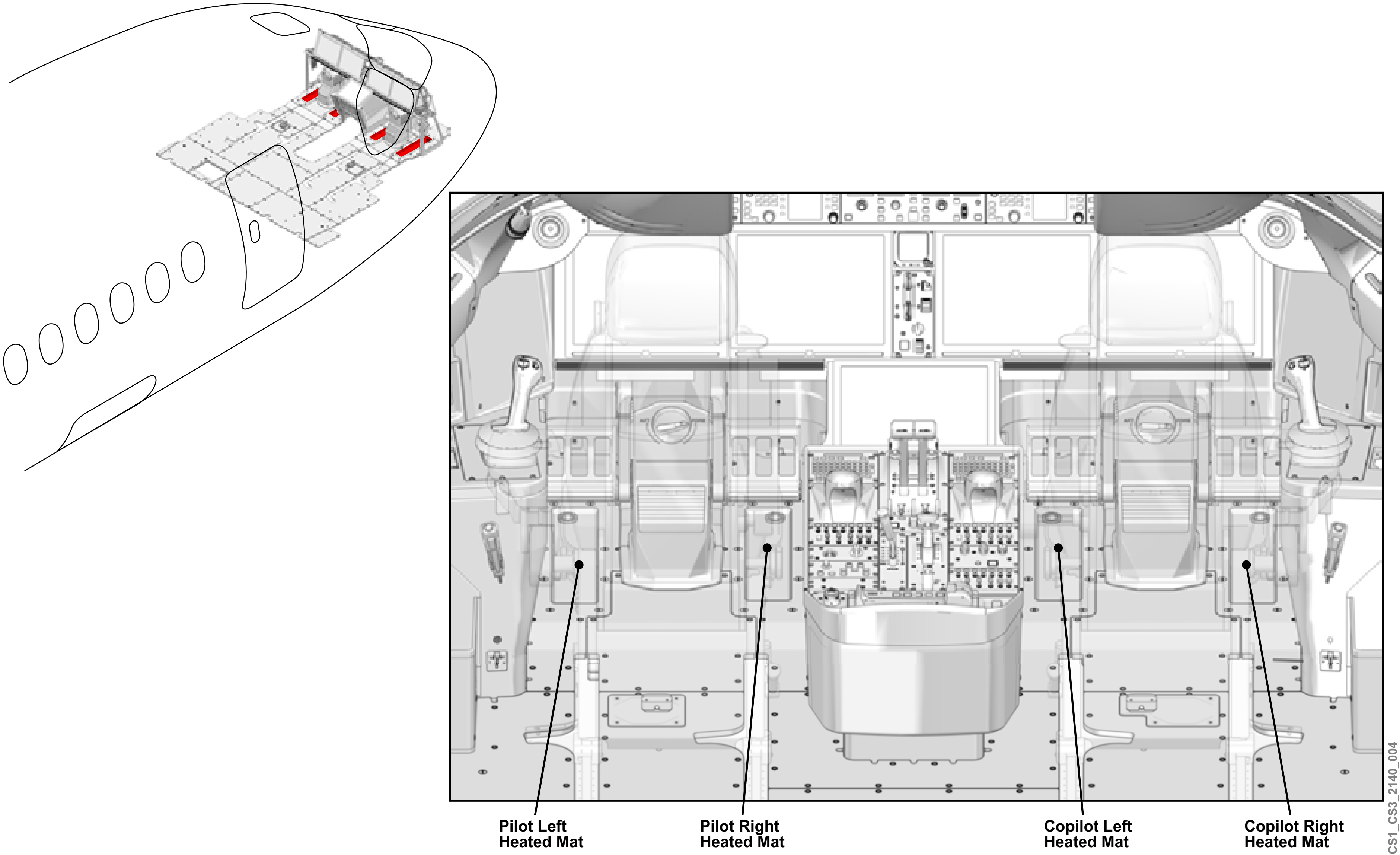


Figure 63: Flight Crew Floor Heating General Description

DETAILED DESCRIPTION

The heated mats are controlled by thermostats that provide power when the temperature is less than 20°C (68°F) and remove power when the temperature exceeds 40°C (104°F).

The control and distribution cabinet 3 (CDC 3) and CDC 4 monitor the current draw. When the current draw exceeds 1 A, the solid-state power controller (SSPC) latches off and reports the overcurrent fault to the onboard maintenance system (OMS).

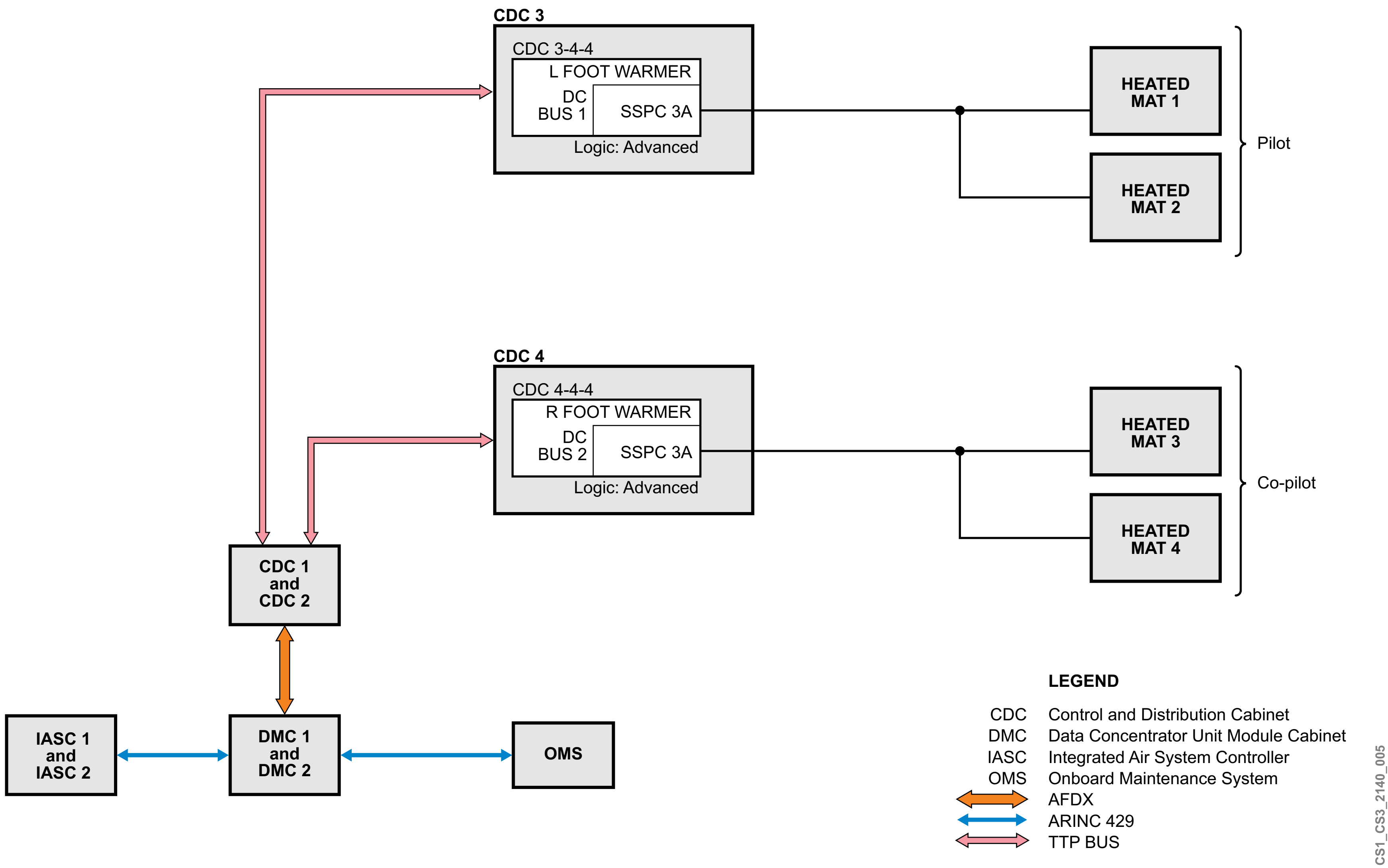


Figure 64: Flight Crew Floor Heating Detailed Description

GALLEY HEATING

GENERAL DESCRIPTION

The forward and aft galley heating consists of a fan and a heater in a recirculation loop that distributes reheated air next to the service and passenger doors. Air is extracted from the galley area by the galley fan, heated by the galley heater and then exhausted near the floor next to the service and main passenger doors.

The respective IASC monitors the supplemental heating system using a dual-element galley heater temperature sensor (GHTS). The sensor limits the duct temperature to 70°C.

Three modes of operation can be selected on a panel installed in the galley area:

- When FAN ON is selected, the fan operates
- When LO HEAT is selected, the fan operates and the heater operates on a two phase power. With a cabin inlet temperature of 25°C (77°F), the system provides an outlet temperature between 30°C (86°F) and 40°C (104°F)
- When HI HEAT is selected, the fan operates and the heater operates on three-phase power. With a cabin inlet temperature of 25°C (77°F), the system provides an outlet temperature between 45°C (113°F) and 55°C (131°F)

A dual-temperature sensor is installed at the heater outlet to provide the duct temperature to the IASCs for system monitoring.

The forward galley heater receives 3-phase 115 VAC from AC BUS 1 and the aft galley heater receives 3-phase power from AC BUS 2.

The forward galley fan is powered by DC BUS 1 and the aft galley fan is powered by DC BUS 2.

Each forward or aft galley heating system is controlled from the flight attendant panels located in the respective forward and aft galley areas.

The system status is sent from IASC to the cabin management system (CMS) for fault indication.

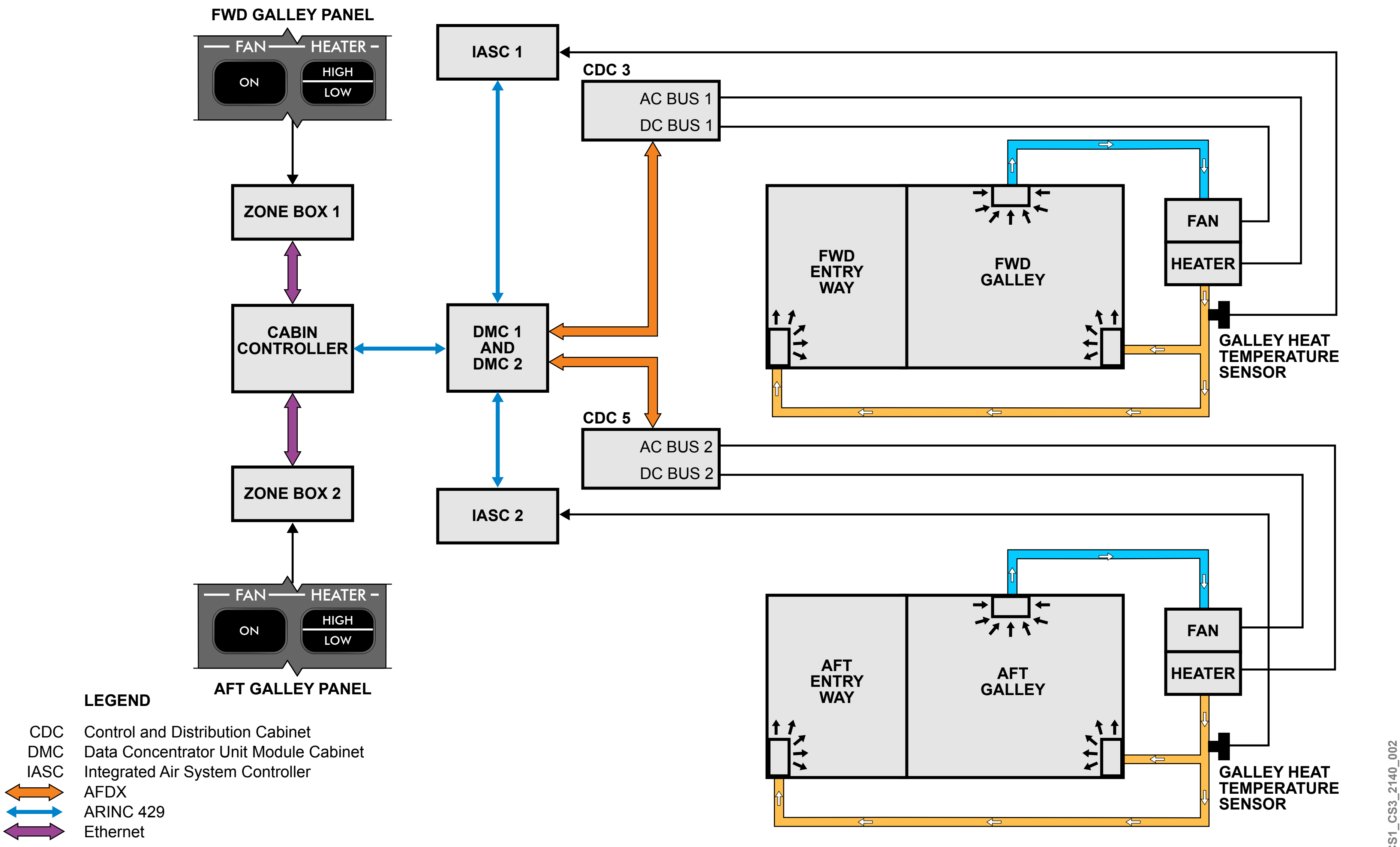


Figure 65: Galley Heating

COMPONENT LOCATION

The galley heating system consists of:

- Galley fan
- Galley heater
- Galley heater temperature sensor

GALLEY FAN

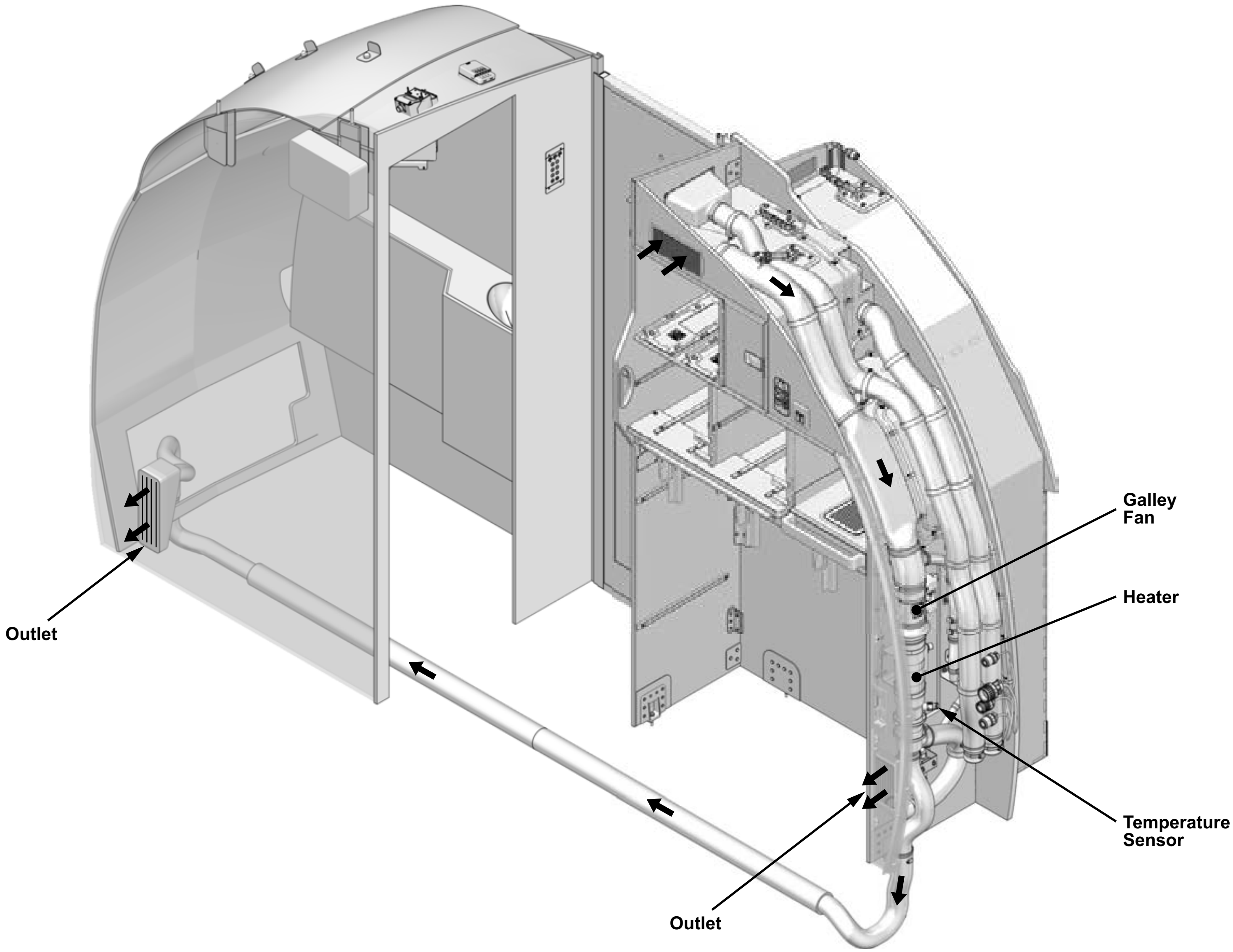
The galley fan is attached to the galley structure.

GALLEY HEATER

The galley heater is attached to the galley structure.

GALLEY HEATER TEMPERATURE SENSOR

The galley temperature sensor is installed in the duct at outlet of the galley heater.



NOTE

Fwd galley shown.
Aft galley similar.

Figure 66: Supplemental Galley Heat Component Location

CONTROLS AND INDICATIONS

GALLEY HEAT CONTROLS

The galley heaters are controlled from the GALLEY panels located in each galley. Each heater has a fan control and a HIGH or LOW heat selection.

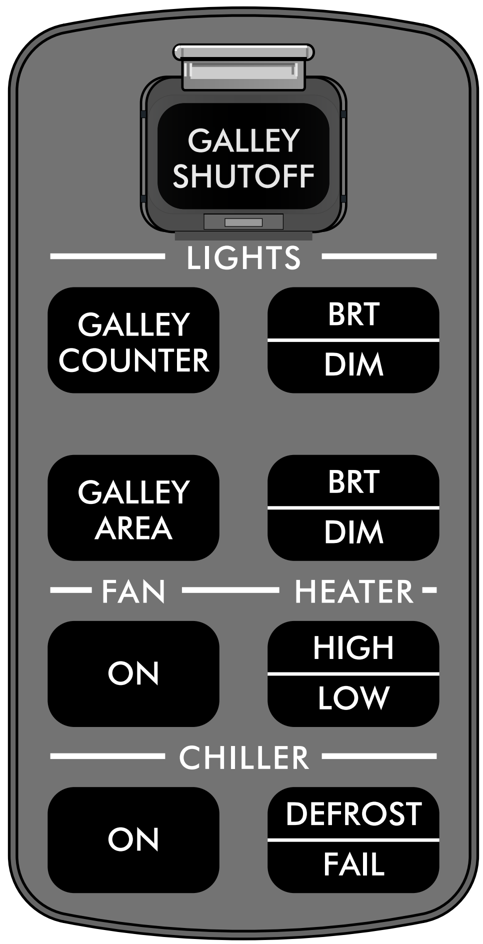


Figure 67: Galley Heater Controls

GALLEY HEAT INDICATIONS

The galley heater faults are displayed on the CMS screen.

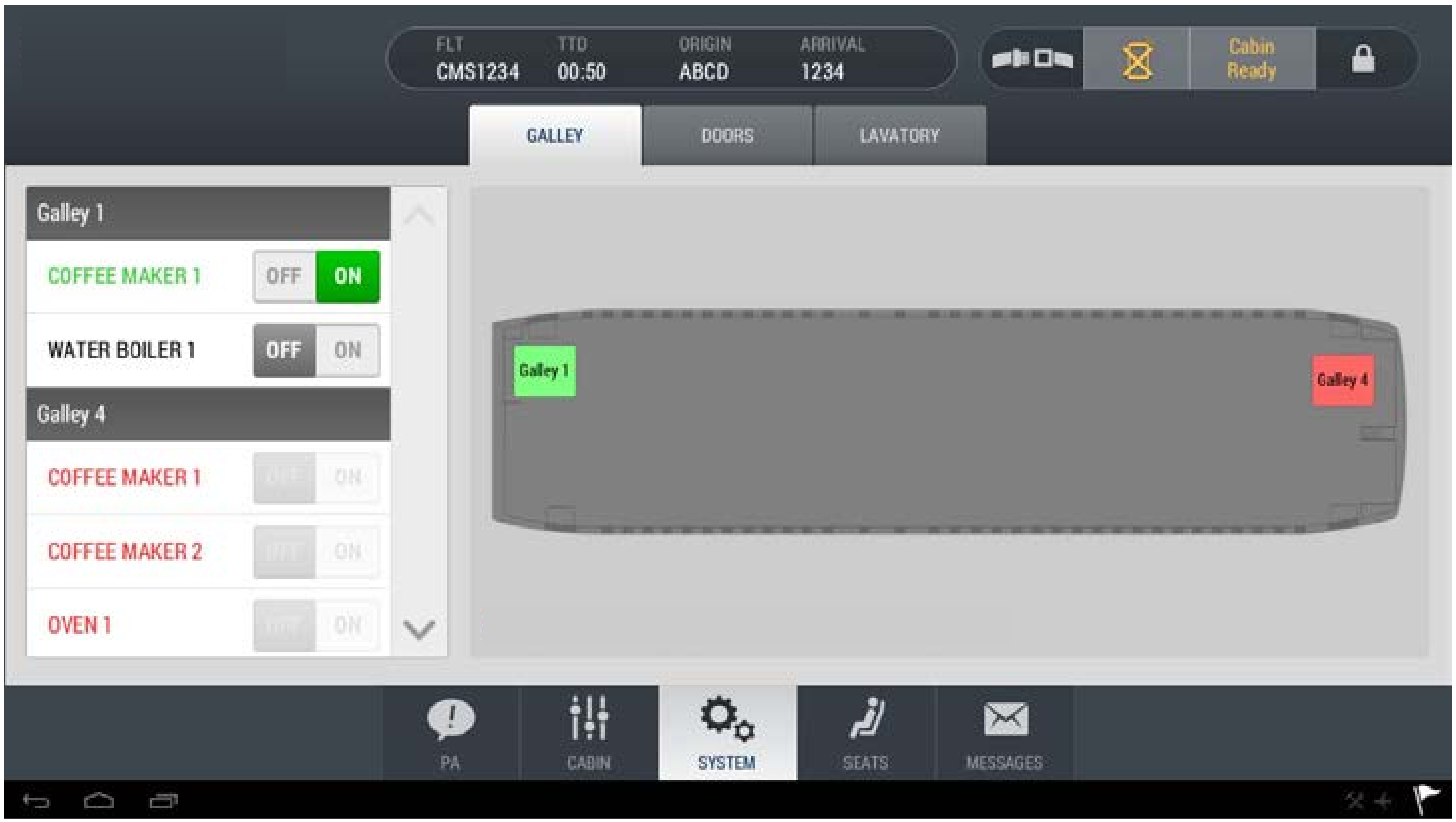


Figure 68: Galley Heater Indications

DETAILED DESCRIPTION

The heater operation is regulated by a built-in thermostat. For overtemperature protection, thermal fuses in-line with the heating elements open at 183°C (362°F).

A heater fault discrete output from the heater assembly is connected directly to the control and distribution cabinet (CDC) that is its power. The galley heater 115 VAC power is removed when a fault occurs. The galley fan remains in operation and only ventilation is provided by the system.

A fan fault discrete output signals overcurrent, overvoltage, underspeed (10,000 rpm) or overheat conditions in the fan assembly, which is connected directly to the CDC that provides power to the galley fan. Both the 28 VDC power to the galley fan and 115 VAC power to the galley heater are removed when a fault occurs.

In both cases, the fault data is sent to the onboard maintenance system (OMS), and a GALLEY HEATING SYSTEM UNAVAILABLE message is displayed on the cabin management system (CMS) crew terminal.

The IASC monitors the galley heater temperature. The galley heater temperature sensor has two elements. One element reports to each of the IASC channels. In case of overheat conditions detected by either element of the galley heater temperature sensor, the heater power is turned off.

The IASC commands the CDC to shut off the galley heating when:

- Galley heater outlet temperature reads above 70°C for more than 1 minute
- The overheat condition is latched until the temperature decreases below 70°C and the GALLEY switch is cycled.

If both elements of the sensor fail, or one element and the opposite IASC channel fail, the heater power is removed. When both elements of the GHTS have failed, the heater is latched off until the GHTS is replaced.

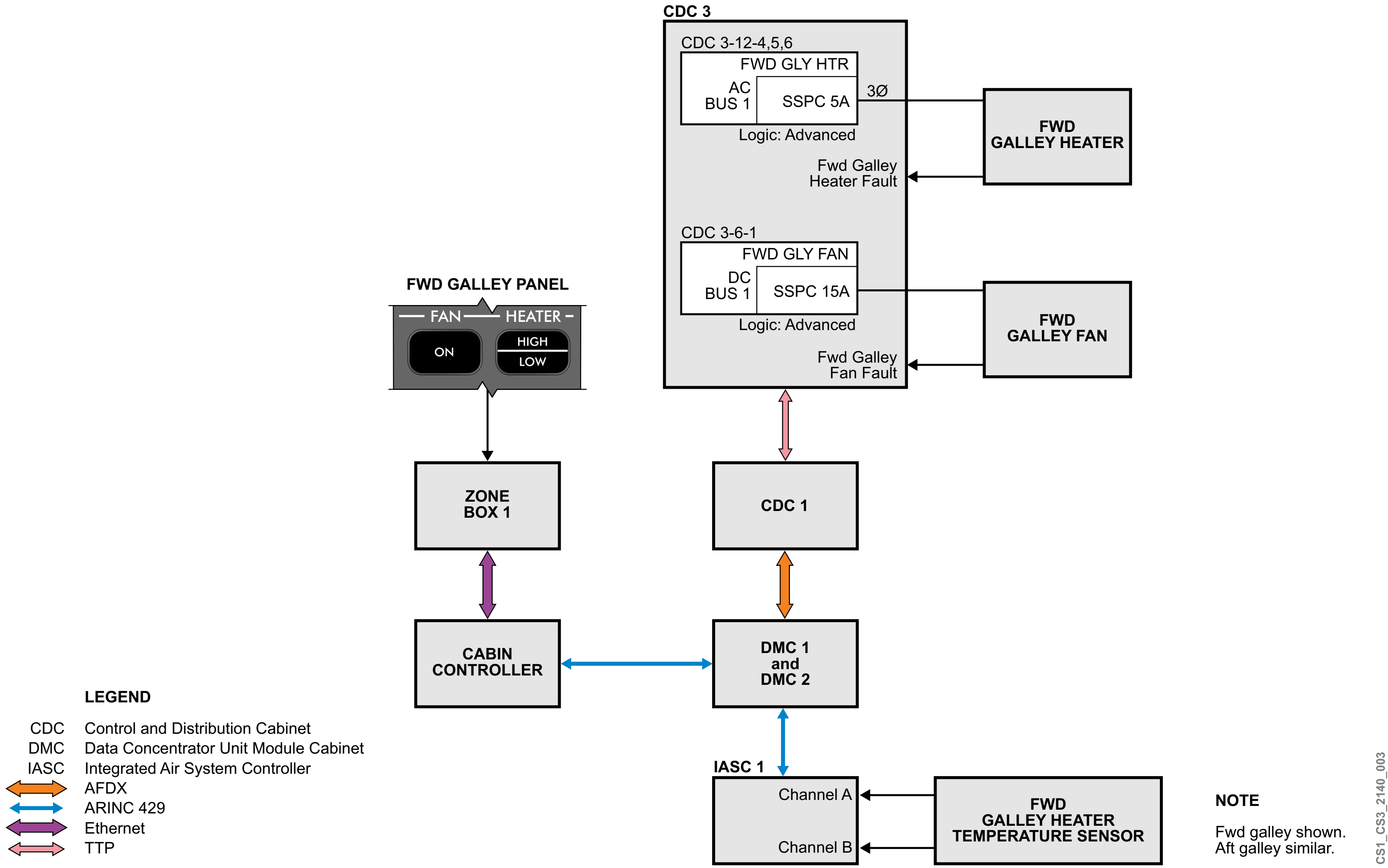


Figure 69: Supplemental Galley Heat Detailed Description

21-26 AVIONICS COOLING AND EXTRACTION SYSTEM

GENERAL DESCRIPTION

The avionics cooling and extraction system consists of three subsystems:

- Air inlet cooling system
- Air extraction system
- Backup exhaust and extraction system

In addition, the wing-to-body fairing (WTBF) is ventilated using the air conditioning ram air system.

The air inlet cooling system draws air from the galleys, and underfloor area, cools it, and supplies it to the forward and mid avionics bay.

The air extraction system draws air from behind the flight deck displays, both forward and mid equipment bays, control and distribution cabinet 5 (CDC 5) and the forward and aft lavatories.

The backup exhaust and extraction system provides an alternate means to extract air from behind the flight deck displays, and both forward and mid equipment bays.

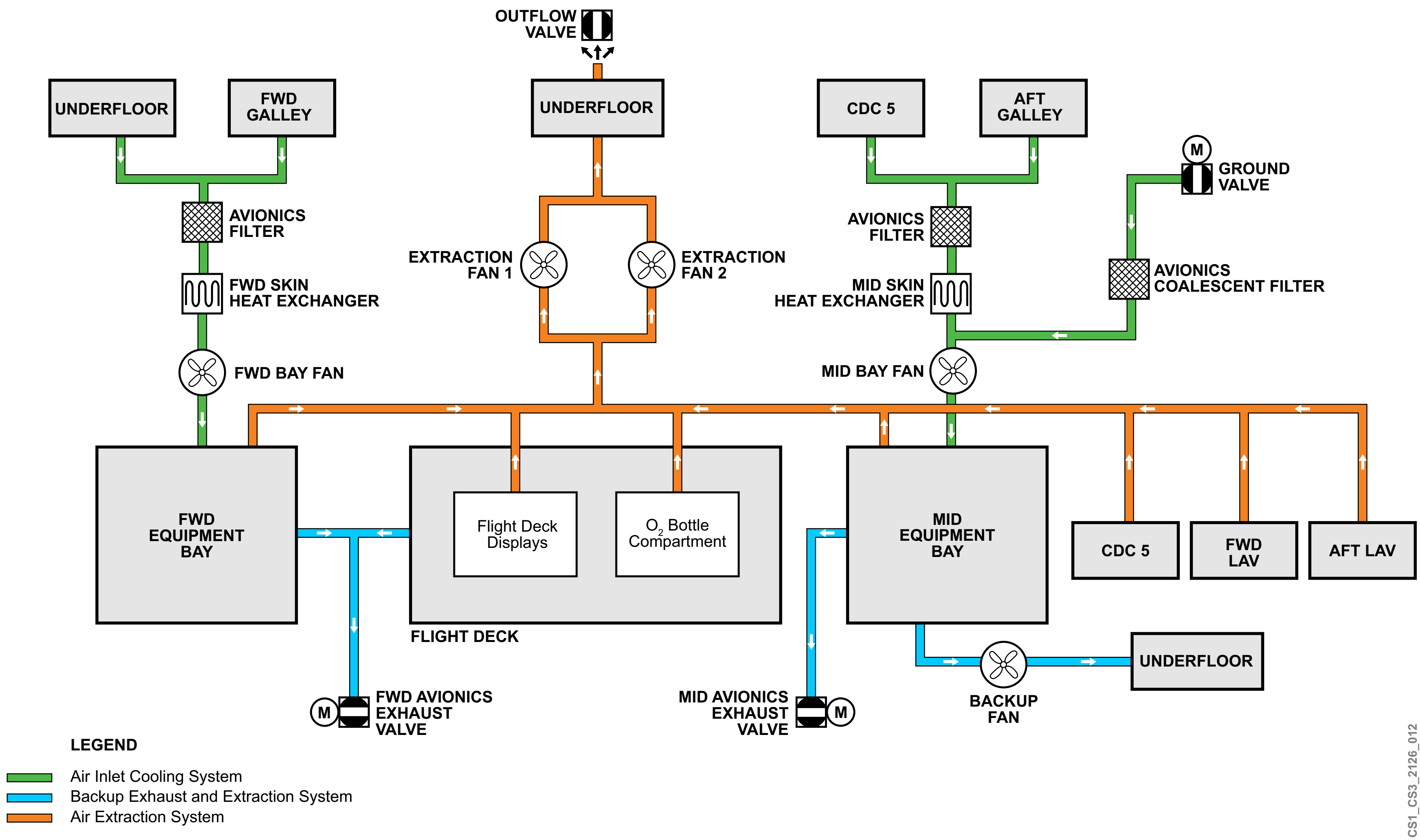


Figure 70: Avionics Cooling and Extraction System General Description

AIR INLET COOLING SYSTEM

GENERAL DESCRIPTION

The air inlet cooling system supplies cooled and filtered air into both forward and mid equipment bays in order to lower their ambient operating temperature.

There is one air inlet cooling system dedicated to each equipment bay. Each system has a fan, skin heat exchanger, and a filter. The fan extracts air from the galley and underfloor areas. The air is cleaned by a filter before passing through the skin heat exchanger. The skin heat exchanger cools the air using the cold aircraft skin. The cooled air is distributed through piccolo tubes in the equipment bay to cool the equipment.

The forward bay fan is powered by DC BUS 2. The mid bay fan is powered by AC BUS 2.

The mid bay air inlet cooling system has a ground valve that provides additional air to the mid equipment bay when the air inlet cooling system operates on the ground. The ground valve is powered by DC BUS 2. On the ground, approximately 70% of the cooling comes from the ground valve, and the remaining 30% comes from the galley and control and distribution cabinet (CDC) 5.

A coalescent filter is installed downstream of a ground valve, located on the aircraft skin. The coalescent filter extracts the water contained in the outside air and then filters the air for the mid equipment bay.

Control of the system is fully automatic. The INLET PBA is normally left in the auto position and only selected OFF when the system has automatically reconfigured to the OFF position. Selecting the INLET PBA OFF turns off the forward and mid bay fans, and closes the ground valve.

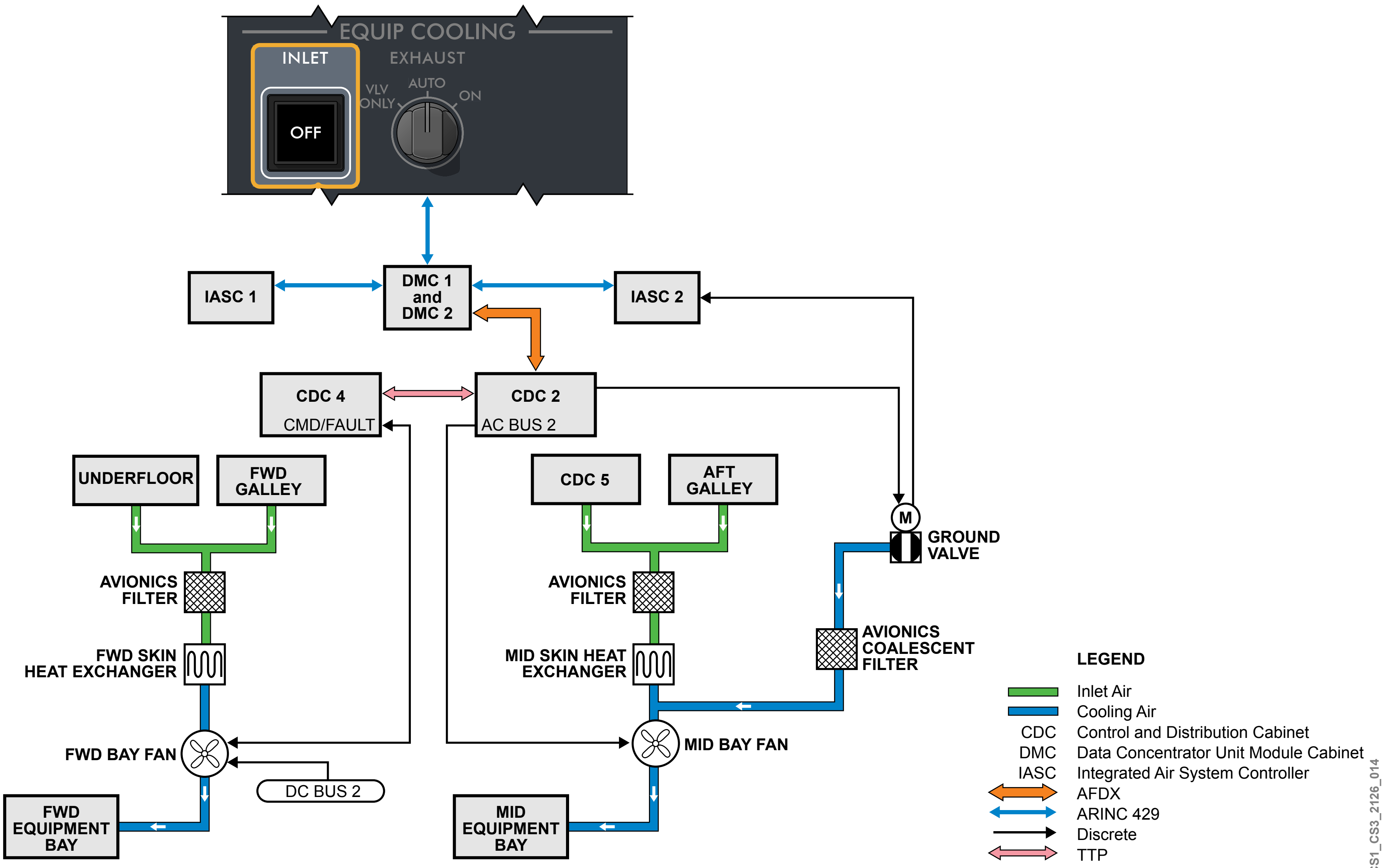


Figure 71: Air Inlet Cooling Schematic

COMPONENT LOCATION

The following are major components of the air inlet cooling system:

- Forward avionics filter
- Forward bay fan
- Forward skin heat exchanger
- Ground valve
- Mid bay fan
- Avionics coalescent filter
- Mid skin heat exchanger
- Mid avionics filter
- Ducting

FORWARD AVIONICS FILTER

The forward avionics filter is located on the right side of the forward equipment bay.

FORWARD BAY FAN

The forward bay fan is located on the right side of the forward equipment bay.

FORWARD SKIN HEAT EXCHANGER

The forward skin heat exchanger is located on the right side of the forward equipment bay.

GROUND VALVE

The ground valve is located in the aft cargo compartment on the right side of the fuselage, forward of the cargo door.

MID BAY FAN

The mid bay fan is located in the aft cargo compartment on the right side of the fuselage, forward of the cargo door.

AVIONICS COALESCENT FILTER

The avionics coalescent filter is located in the aft cargo compartment on the right side of the fuselage, forward of the cargo door.

MID SKIN HEAT EXCHANGER

The aft skin heat exchanger is located in the aft cargo compartment on the right side of the fuselage, aft of the cargo door.

MID AVIONICS FILTER

The mid avionics filter is located in the aft cargo compartment on the right side of the fuselage aft, of the cargo door.

DUCTING

Two separate duct networks provide air to each equipment bay. Each duct runs along the left side of the fuselage. The forward duct connects the forward galleys and underfloor area to the forward equipment bay. The mid duct connects the aft galley and control and distribution cabinet (CDC 5) to the mid equipment bay. The mid bay also has a duct that draws external air when the aircraft is on the ground. The air is distributed in the equipment bays through piccolo tubes.

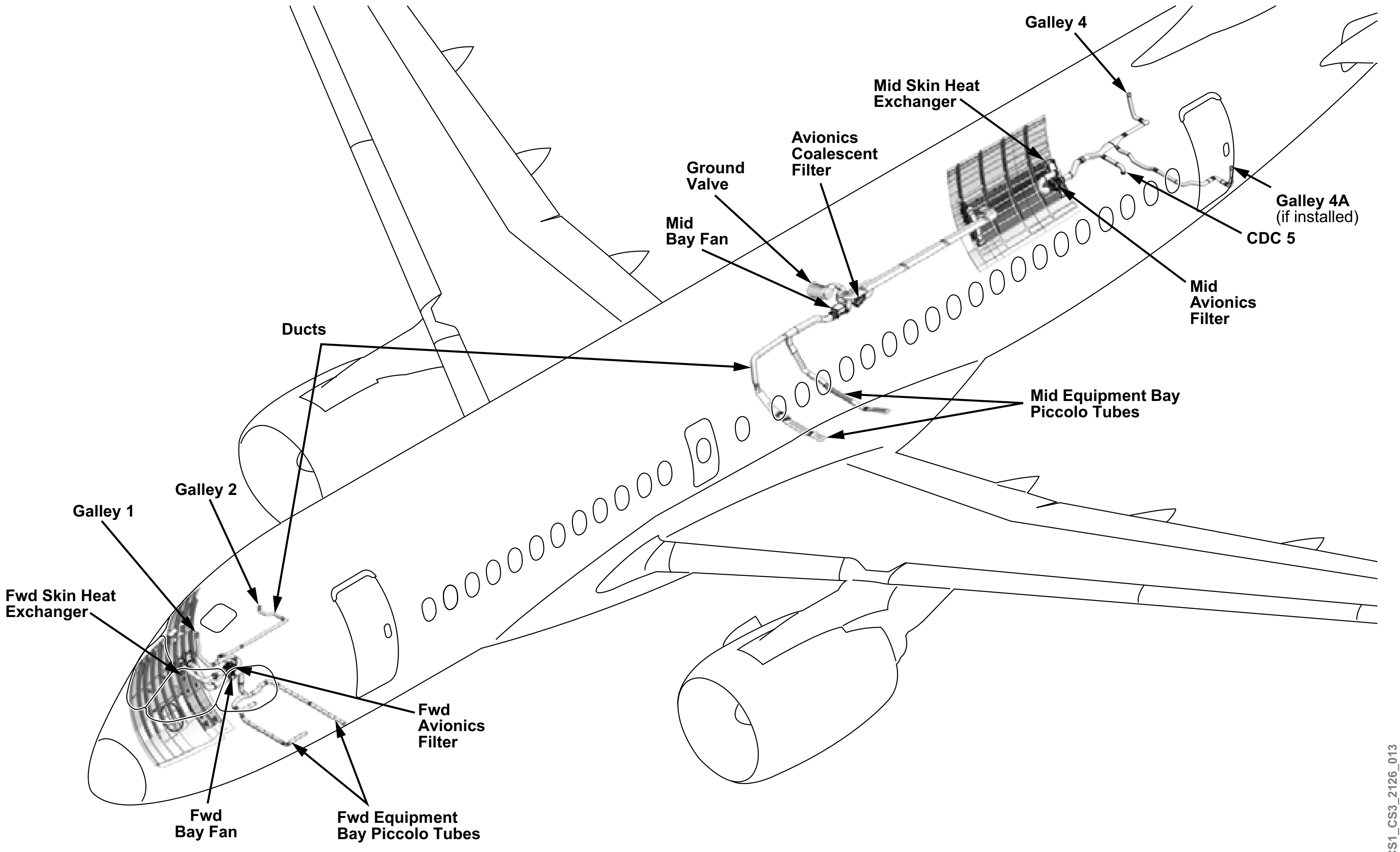


Figure 72: Air Inlet Cooling Component Location

COMPONENT INFORMATION

FORWARD AVIONICS FILTER

The forward avionics filter filters air from the forward galleys and underfloor area. The replaceable filter is installed at the inlet to the forward skin heat exchanger.

FORWARD SKIN HEAT EXCHANGER

The forward skin heat exchanger is installed within the frames of the fuselage structure.

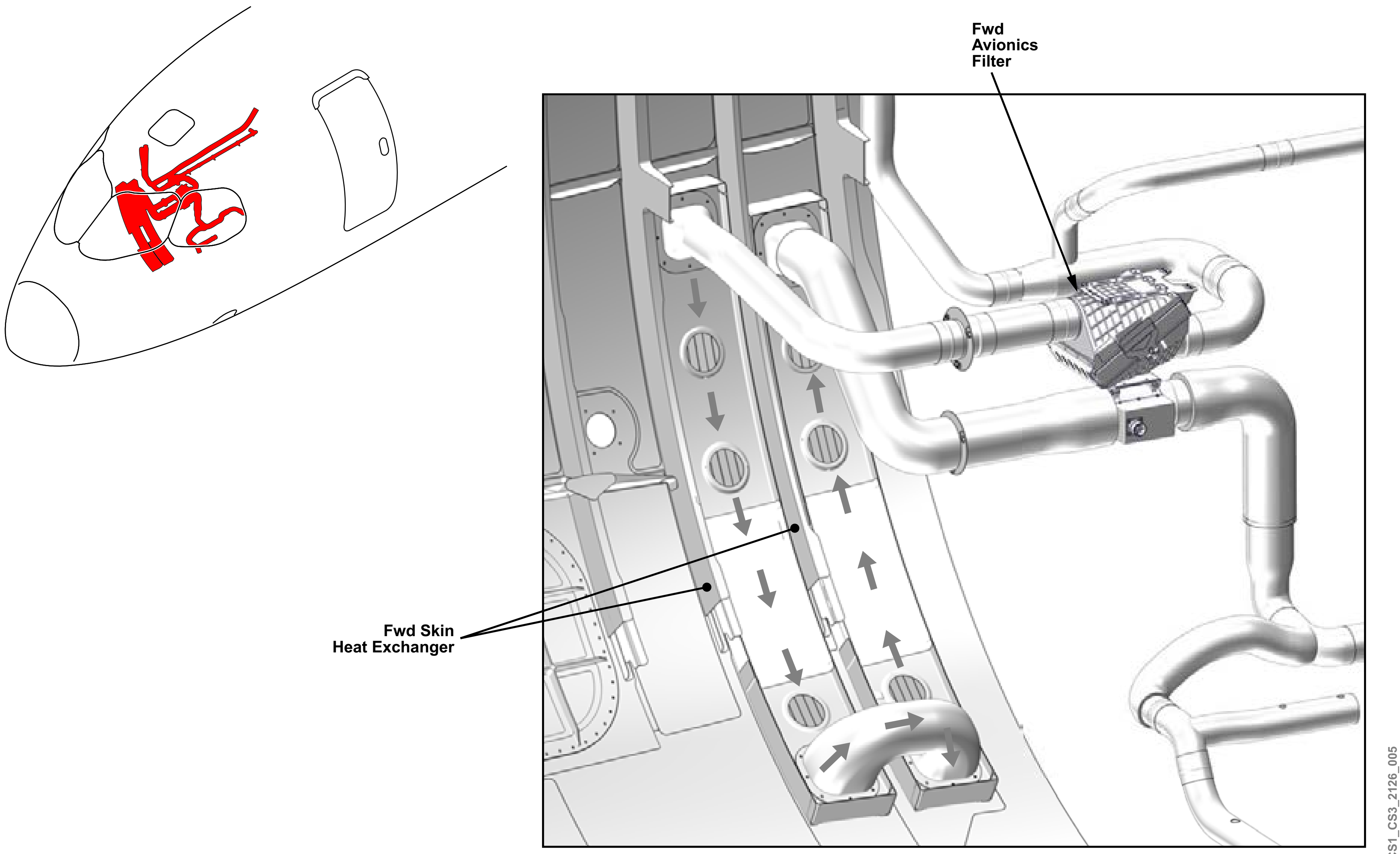


Figure 73: Forward Avionics Filter and Forward Skin Heat Exchanger

GROUND VALVE

The GND VLV is installed on the aircraft skin on the lower right side of the fuselage. The ground valve is driven by a linear electrical actuator powered by 28 VDC. A torque limiter protects the actuator motor and prevents inadvertent opening of the valve in flight. The valve position is monitored by microswitches. The valve has a manual override handle on its external side. This handle allows the valve to be closed from outside of the aircraft.

AVIONICS COALESCENT FILTER

The avionics coalescent filter is located in the duct from the ground valve. The replaceable filter removes contaminants and water from the outside air. A drain line carries the water to the lower fuselage to drain overboard.

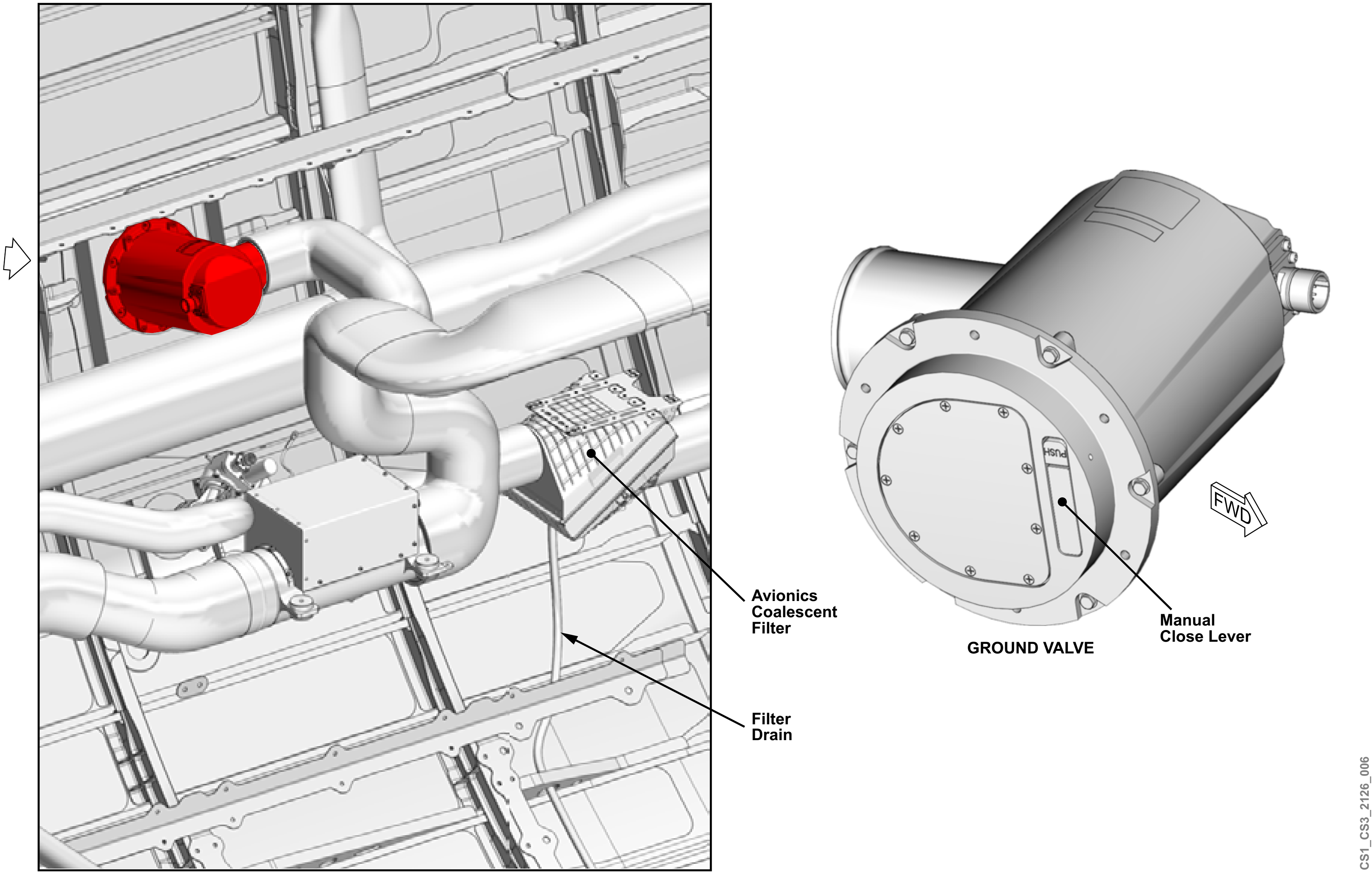


Figure 74: Ground Valve and Avionics Coalescent Filter

MID SKIN HEAT EXCHANGER

The mid skin heat exchanger is installed within the frames of the fuselage structure.

MID AVIONICS FILTER

The mid avionics filter filters air drawn from the aft galley and underfloor area. The replaceable filter is installed at the inlet to the mid skin heat exchanger.

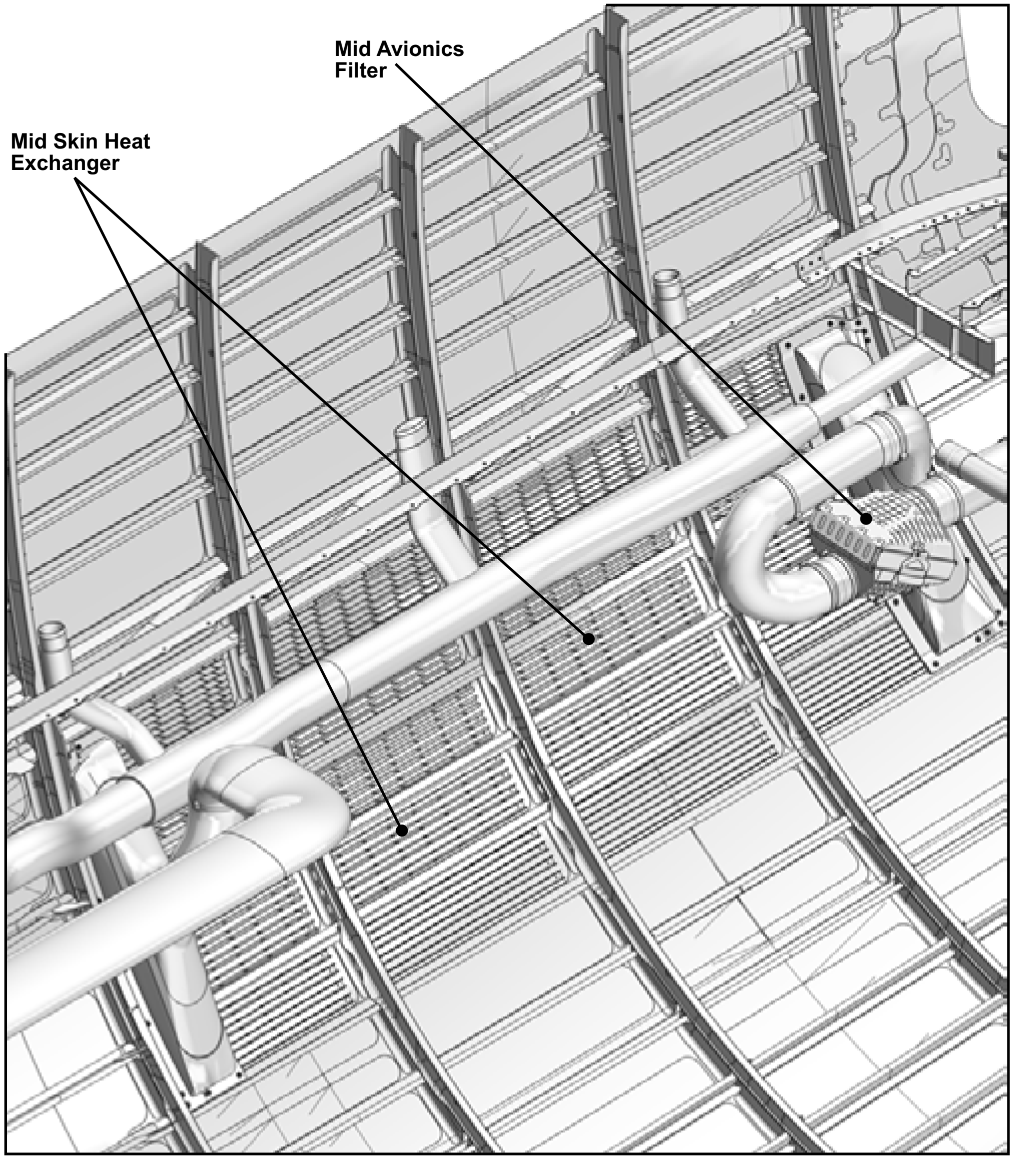
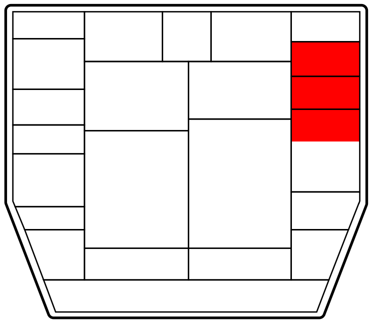


Figure 75: Mid Skin Heat Exchanger and Mid Avionics Filter

CONTROLS AND INDICATIONS

The system is controlled from the EQUIP COOLING panel in the flight deck. In normal operation, the system operation is automatically controlled by the integrated air system controllers (IASCs). The INLET PBA can be selected to turn off the air inlet cooling system.



OVERHEAD PANEL

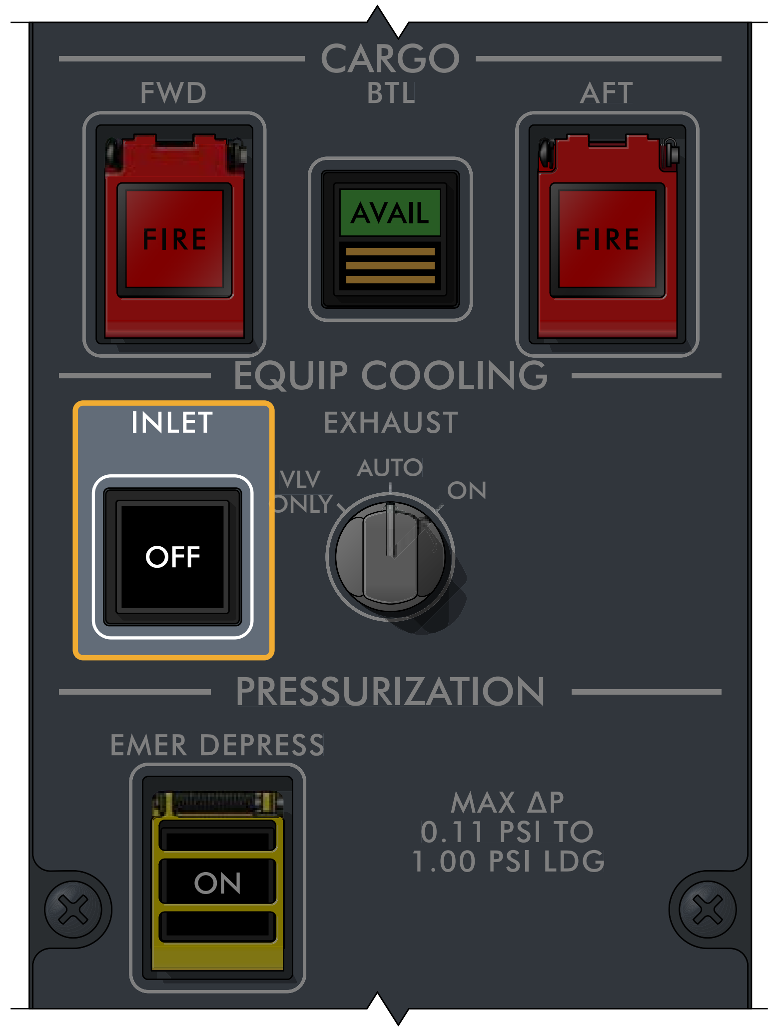


Figure 76: Avionics Cooling and Extraction System Controls

DETAILED DESCRIPTION

The air inlet cooling system operates all the time on ground and in flight. The air inlet cooling system operation is automatically managed by the IASC main channels, which control the mid ground valve and both forward and mid bay inlet fans through logics embedded in the CDCs. The CDCs drive power supplies, discrete and relay signals located in the electrical distribution system according to requests from the IASCs.

In flight, the air inlet cooling system extracts air from the galley, underfloor areas, and CDC 5, cools it through the skin heat exchanger, and then supplies it through a set of distribution lines to cool the avionics bays.

On ground, with an outside air temperature (OAT) ranging between 2°C and 33°C, the IASCs command the ground valve open to allow outside air into the system.

Before takeoff, the ground valve is driven closed once all doors are signaled closed and locked. If the ground valve fails to close, the valve can be manually closed from outside of the aircraft.

After landing, the ground valve is driven open once all doors are not signaled closed. The ground valve is kept closed at landing until the doors open in order to minimize the residual cabin differential pressure, and prevent engine exhaust air going into the filter during the deceleration and taxi phases.

In case of smoke detected by the fire detection and extinguishing system (FIDEX) control unit, the IASCs signal the CDCs to remove power from both forward and mid bay fans to automatically shutoff the air inlet cooling system.

In case of ditching, when the DITCHING PBA is selected on the PRESSURIZATION panel, the ground valve is automatically driven closed.

In case a failure is detected on either forward or mid bay fan or the ground valve, an advisory message EQUIP BAY COOL FAULT is displayed. An INFO message provides guidance for any action to be taken before the next aircraft dispatch.

The air inlet cooling system can be selected OFF by selecting the INLET PBA on the overhead panel. The aircraft can be dispatched without any operating limitation with the system selected OFF.

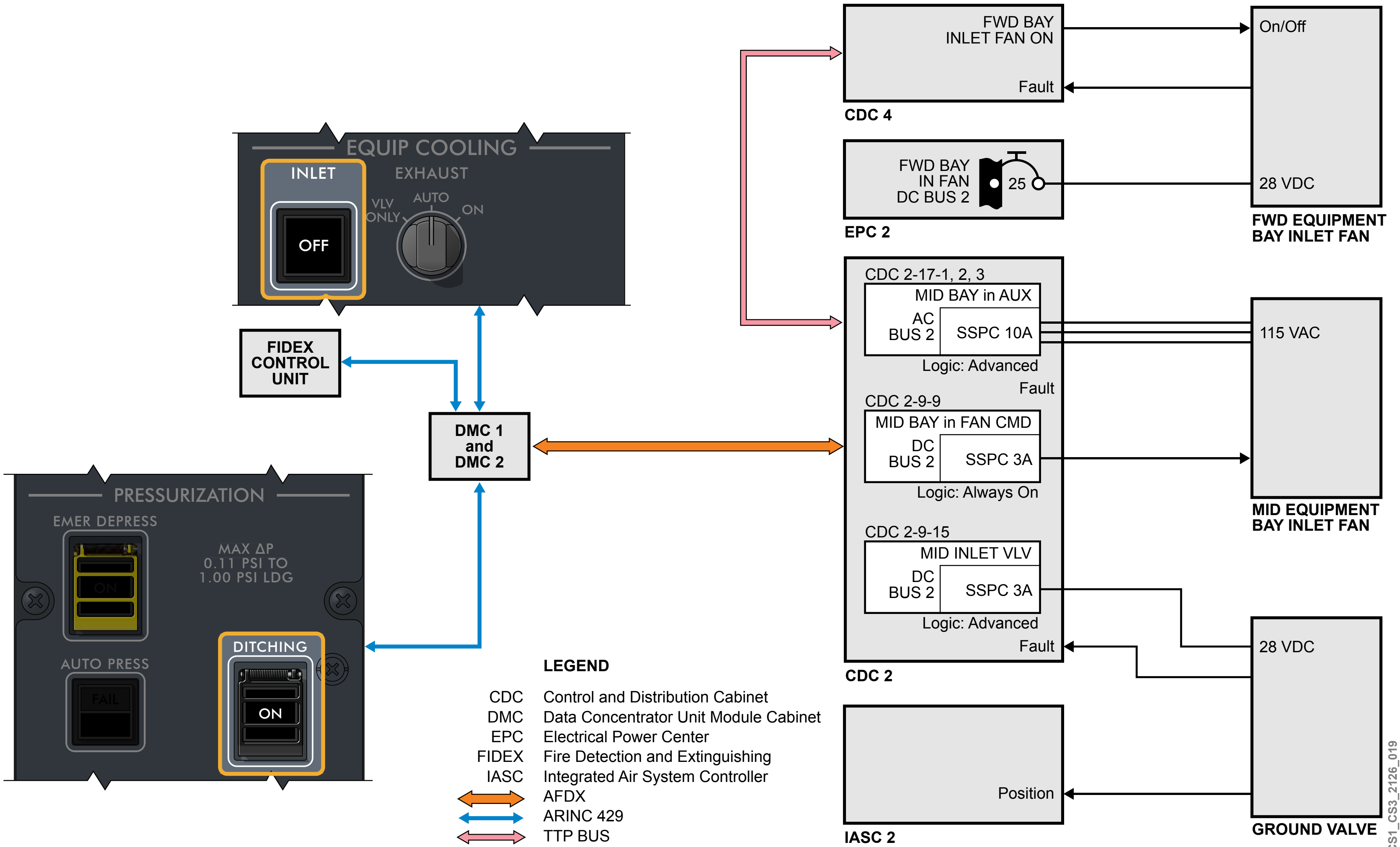


Figure 77: Air Inlet Cooling System

AIR EXTRACTION SYSTEM

GENERAL DESCRIPTION

The extraction system is composed of two identical extraction fans (EFANs) pulling air from the flight deck displays, both forward and mid equipment bays, CDC 5 enclosure, and the forward and aft lavatories. Extracted air is discharged in closed vicinity of the outflow valve, and then dumped overboard. The extracted airflow is kept below 70°C.

The air extraction system operates when the EXHAUST switch on the EQUIP COOLING panel is in the AUTO position. In normal operation, one fan is running, the other is on standby-mode. For reliability and safety reasons, the fans are alternately operated on daily basis.

EFAN 1 is powered by AC BUS 2. The motor controller is powered by DC BUS 2. EFAN 2 is powered by AC BUS 1. The motor controller is powered by DC BUS 1.

On the ground, the EFAN operates at maximum speed to maximize heat extraction, and maintain the equipment bay temperature as low as possible.

In flight, the EFAN operates at a lower speed to maintain extraction flow below the outflow valve (OFV) exhaust flow value.

NOTE

On the ground, powering the aircraft with batteries only must be limited to 5 minutes. After 5 minutes, AC power must be provided for EFAN operation to ensure adequate equipment bay cooling.

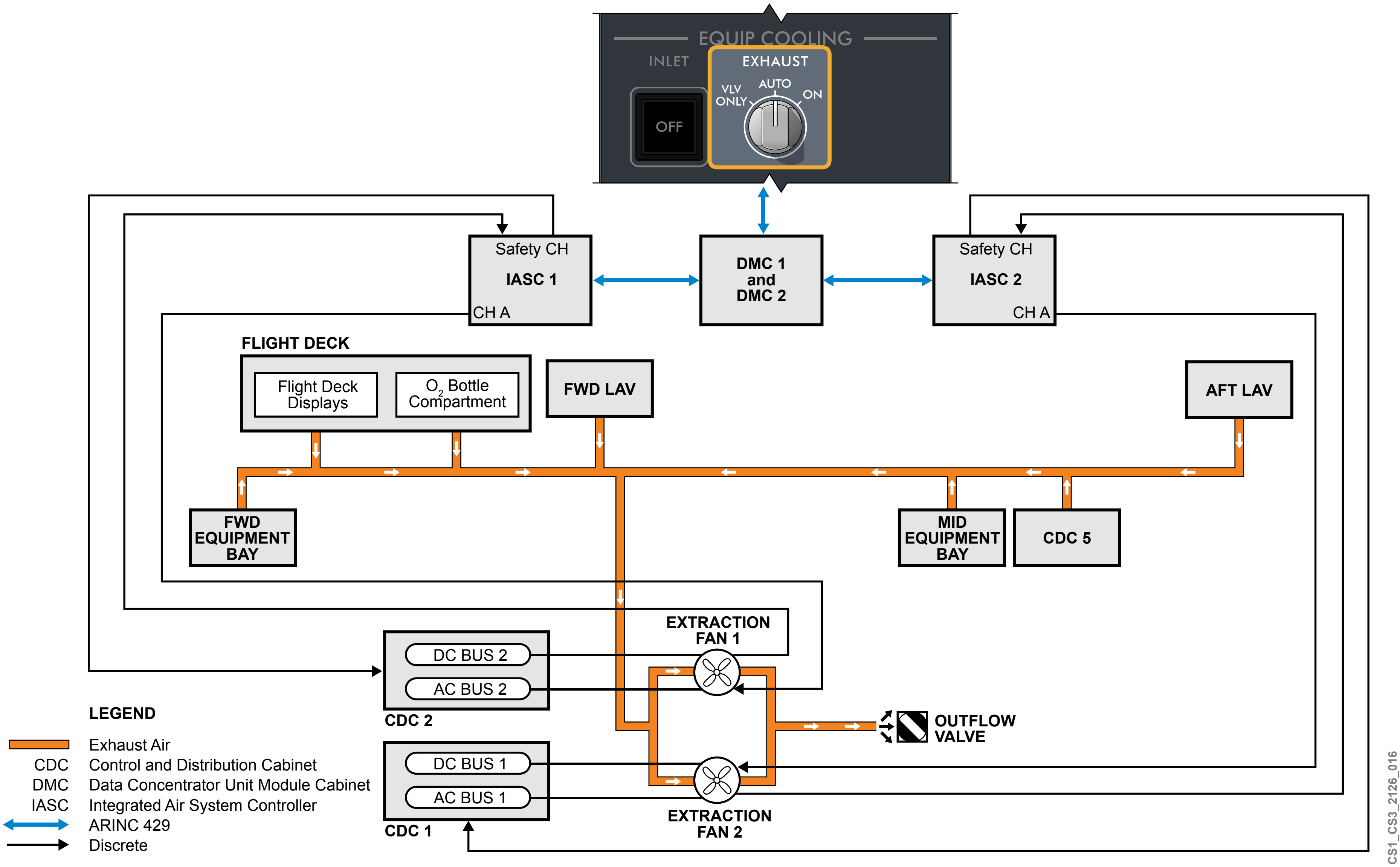


Figure 78: Air Extraction System

COMPONENT LOCATION

The following is a major component of the main air extraction system

- Extraction fans
- Ducting

EXTRACTION FANS

The extraction fans are located in the environmental control system bay.

DUCTING

The extraction ducting runs along both side of the fuselage, connecting the flight deck display lines, oxygen compartment, and lavatories. Piccolo tubes draw air from the forward and mid equipment bays. The ducts join in a common manifold in the environmental equipment bay. The manifold connects all of the ducting to the extraction fans.

The extraction fan exhaust discharges the exhaust air into the environmental bay near the outflow valve.

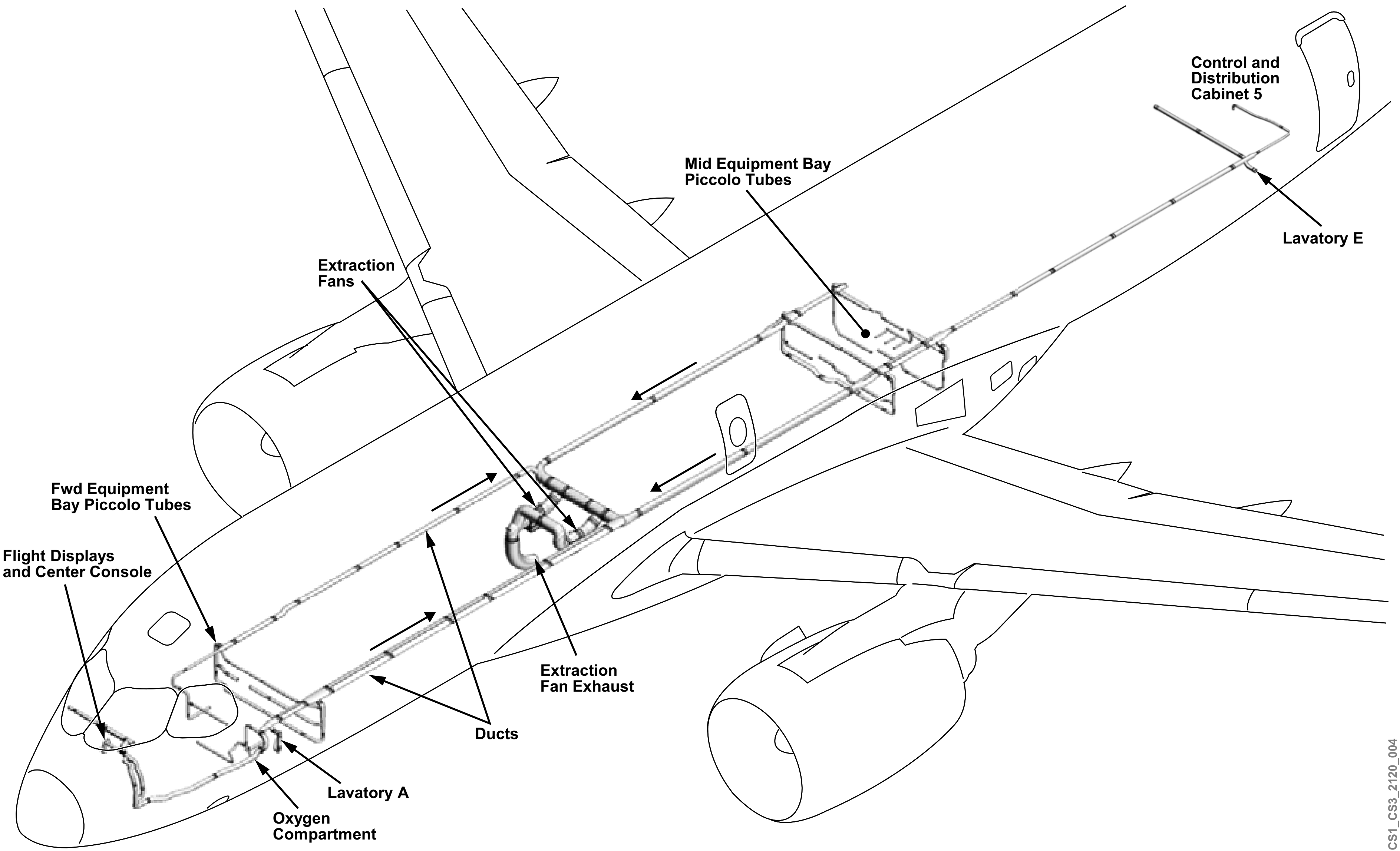


Figure 79: Main Air Extraction System Component Location

DETAILED DESCRIPTION

In normal operation, one extraction fan (EFAN) is running, and the other extraction fan is on standby mode. Both avionics exhaust valves (AEV) are closed and the backup fan is off. The EFAN selection and the system reconfiguration are controlled by the IASC safety channels. Each safety channel controls the power supply of one EFAN by commanding the relay coil that switches the 115 VAC to the EFAN.

Each EFAN is operated on different days. Automatic switching occurs on the ground. The fan selection logic is done by the IASCs safety channel, based on the date provided by the data concentrator unit module cabinet (DMC).

Each extraction fan provides low-speed and internal fault monitoring feedback to the IASCs through a CAN BUS for troubleshooting and monitoring of the EFANs. A fault discrete signal is also sent from the EFAN to the IASC safety channels for monitoring of the EFAN internal failures. Fault detection is done by the fan motor control. Fan speed, overheat, overcurrent, and overvoltage faults are monitored. If one EFAN fails, the system automatically switches to the other EFAN, and an EQUIP BAY COOL FAULT advisory message is displayed.

On the ground, the EFAN operates at full speed in order to maximize heat extraction, and maintain equipment bay ambient temperature as low as possible.

In flight, the EFAN operates at a lower speed to maintain extraction flow below the outflow valve exhaust flow, and to ensure no equipment bay airflow is recirculated into the distribution system.

Dispatch is possible with one EFAN failure, if the backup exhaust system is operative.

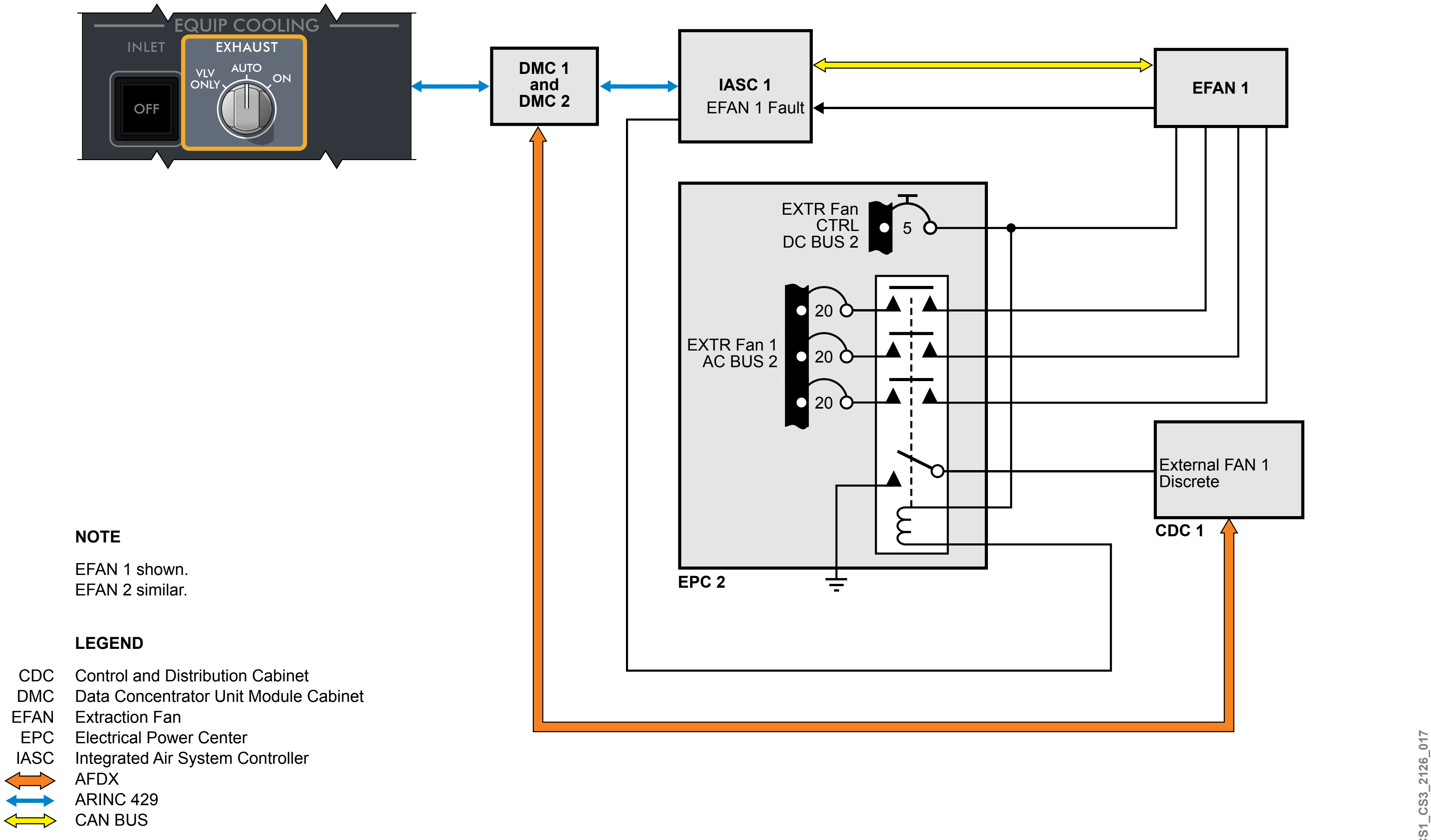


Figure 80: Main Air Extraction System

BACKUP EXHAUST AND EXTRACTION SYSTEM

GENERAL DESCRIPTION

A backup cooling system is provided for the flight deck displays and avionics cooling functions with the addition of two duct networks, extracting air from behind the flight deck displays, and from the forward equipment bay and the mid equipment bay.

Each duct connects to one avionics exhaust valve (AEV) installed on the aircraft skin. The AEV is a shutoff valve, which opens and exhausts the airflow overboard. Each AEV is powered and monitored by both IASCs.

The amount of airflow extracted is mainly dependent on the cabin to ambient pressure differential existing between the inlet and the outlet of a flow restrictor integral to the aircraft skin.

A third backup extraction fan is fitted on top of the mid avionics bay enclosure, and provides air extraction through an opening in the center ceiling of the mid avionics bay. The backup fan is powered by DC ESS BUS 1.

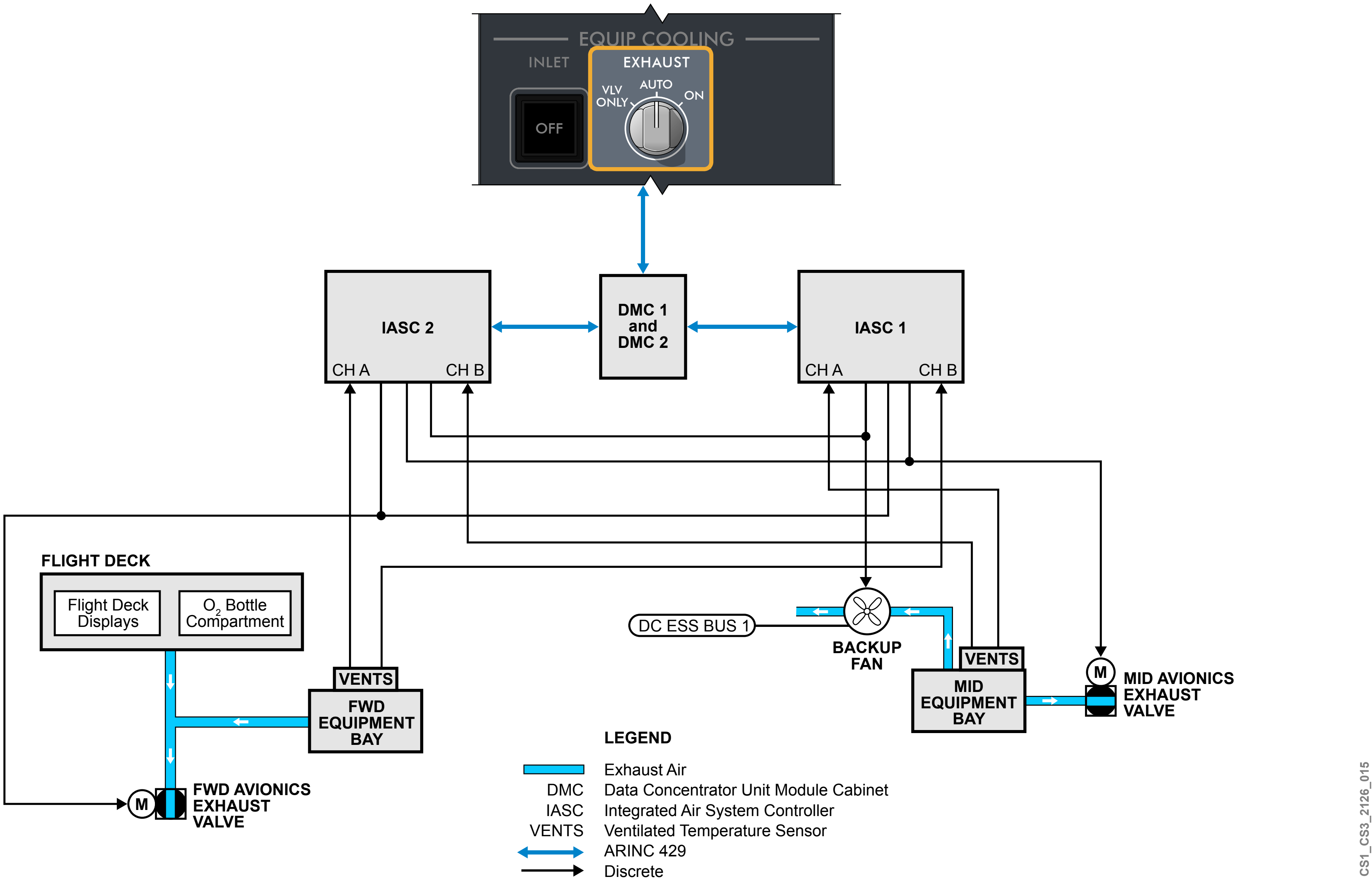


Figure 81: Backup Exhaust and Air Extraction System General Description

COMPONENT LOCATION

The following are major components of the backup exhaust and air extraction system:

- Avionics exhaust valves
- Backup fan
- Ducting
- Avionics bay ventilated temperature sensors (Refer to figure 83)

AVIONICS EXHAUST VALVES

The forward avionics exhaust valve is located in the forward cargo compartment on the right side of the lower fuselage, forward of the cargo door.

The aft avionics exhaust valve is located in the aft cargo compartment on the right side of the lower fuselage, forward of the cargo door.

BACKUP FAN

The backup fan is mounted on the ceiling of the mid equipment bay.

DUCTING

There are two duct networks for the backup exhaust and air extraction system. The forward duct runs along the left side of the aircraft, connecting the flight deck displays, and forward equipment bay to the forward avionics exhaust valve.

The mid equipment bay ducting connects to the mid avionics exhaust valve.

Air from the flight deck displays, forward, and mid equipment bays is drawn through piccolo tubes, and exhausted overboard when the avionics exhaust valves are open.

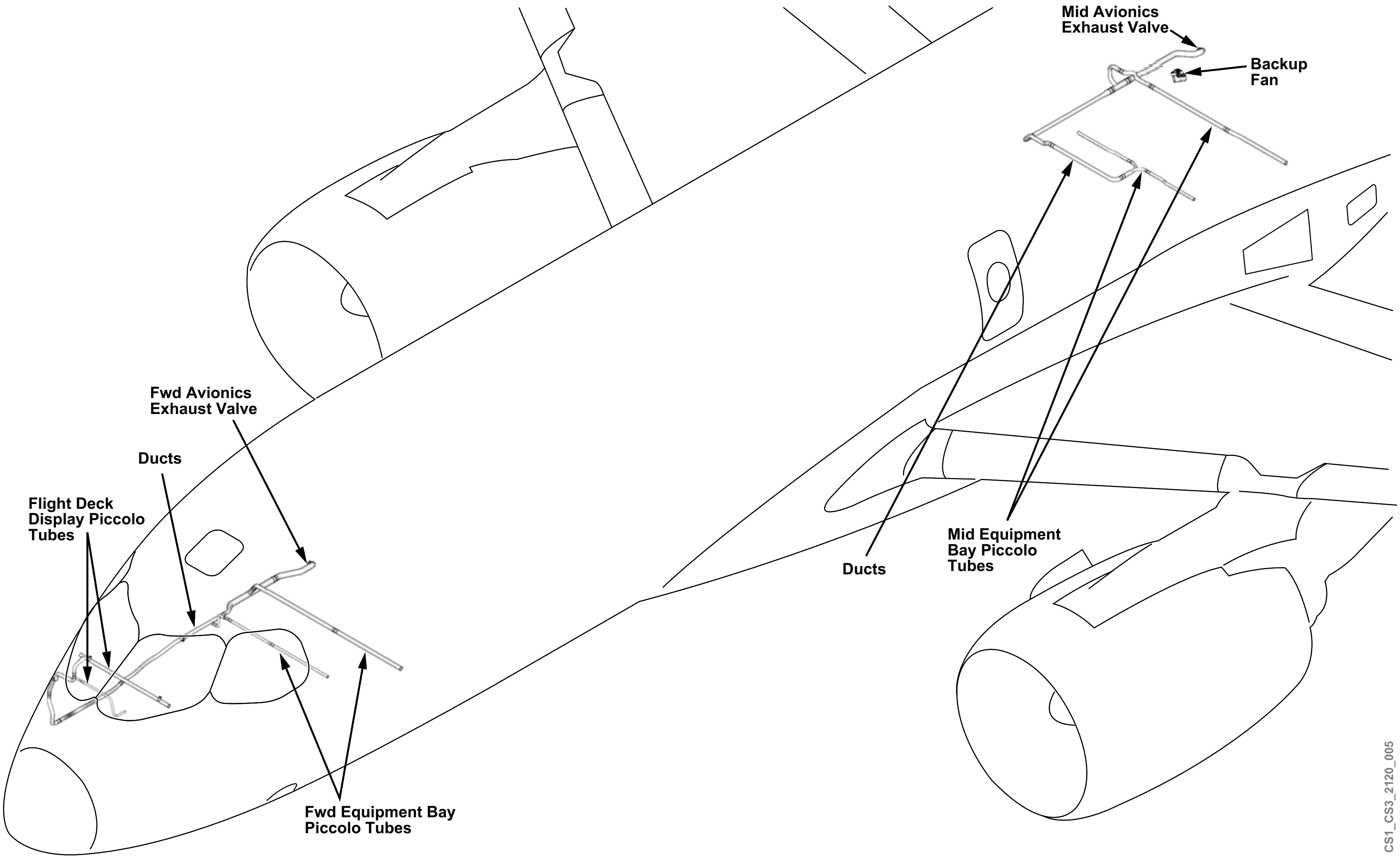
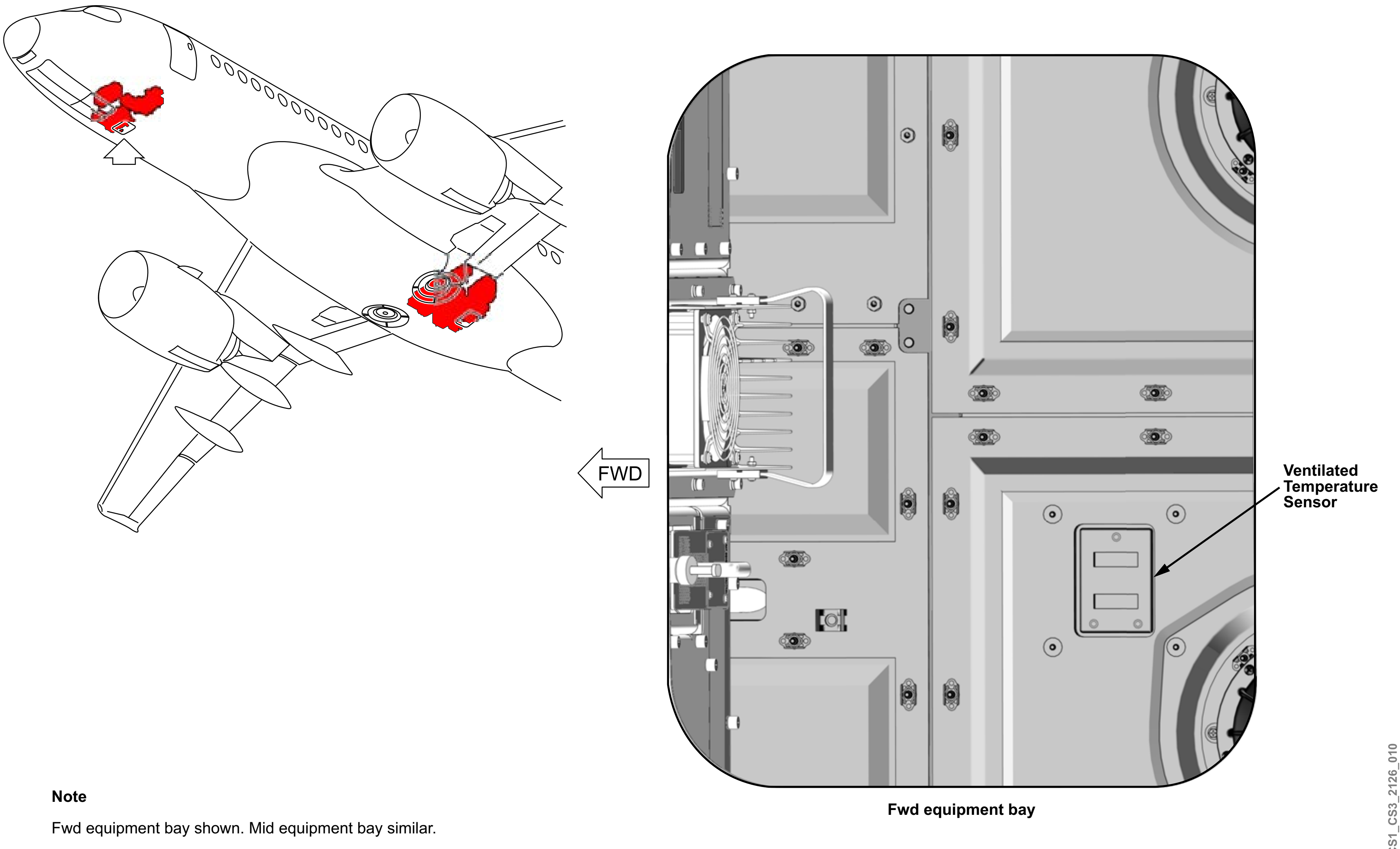


Figure 82: Backup Exhaust and Air Extraction System

AVIONICS BAY VENTILATED TEMPERATURE SENSORS

An avionics ventilated temperature sensor (AV-VENTS) is installed in each of the forward and mid equipment bay ceilings.



Note

Fwd equipment bay shown. Mid equipment bay similar.

Figure 83: Avionics Bay Ventilated Temperature Sensor Location

DETAILED DESCRIPTION

The backup exhaust and extraction systems are in standby during normal operation.

Each bay has an avionics ventilated temperature sensor (AV-VENTS) sensor for monitoring. The AV-VENTS sensor has two temperature elements with one reporting to each IASC. A small fan draws air across the elements.

If the temperature exceeds 67°C (153°F) for more than 30 seconds, an EQUIP BAY OVHT warning message is displayed, and the auxiliary power unit (APU) (multifunction) horn sounds. The message is cleared when the temperature decreases to 62°C (144°F), and the exhaust switch on the EQUIP COOLING panel is moved from the AUTO position.

If both EFANs fail, or both channels of one AV-VENTS fail, the IASCs automatically open both avionics exhaust valves (AEVs), starts the backup fan, and an EICAS EQUIP BAY COOL FAIL caution message is displayed. The EXHAUST switch on the EQUIP COOLING panel is set to ON to match the new system configuration.

In the event of smoke detected in the avionics bay, both AEVs are automatically driven open and the backup fan is kept off by the IASCs if no overheat is detected. The EXHAUST switch is moved to VLV ONLY to match the system configuration. A hardwired contact on the EXHAUST switch provides additional control of the fan.

NOTE

When the EXHAUST switch is set to ON, a discrete hardwired signal from the switch turns on the backup fan. On ground when operating on the batteries only, as well as during approach when the ram air turbine (RAT) generator switches off, the electrical distribution system disables the backup fan in order to protect the batteries.

If the DITCHING PBA is selected on the PRESSURIZATION panel, the AEVs are automatically driven closed to prevent water from entering the aircraft.

Both AEVs and the backup fan are tested 3 minutes after landing to confirm the backup exhaust and extraction systems is operative. In case a failure is detected on either AEV or on the backup fan at landing, an EICAS EQUIP BAY COOL FAULT advisory and relevant INFO messages are displayed.

One or both AEVs can be failed for dispatch. Aircraft dispatch is not possible with the backup fan inoperative.

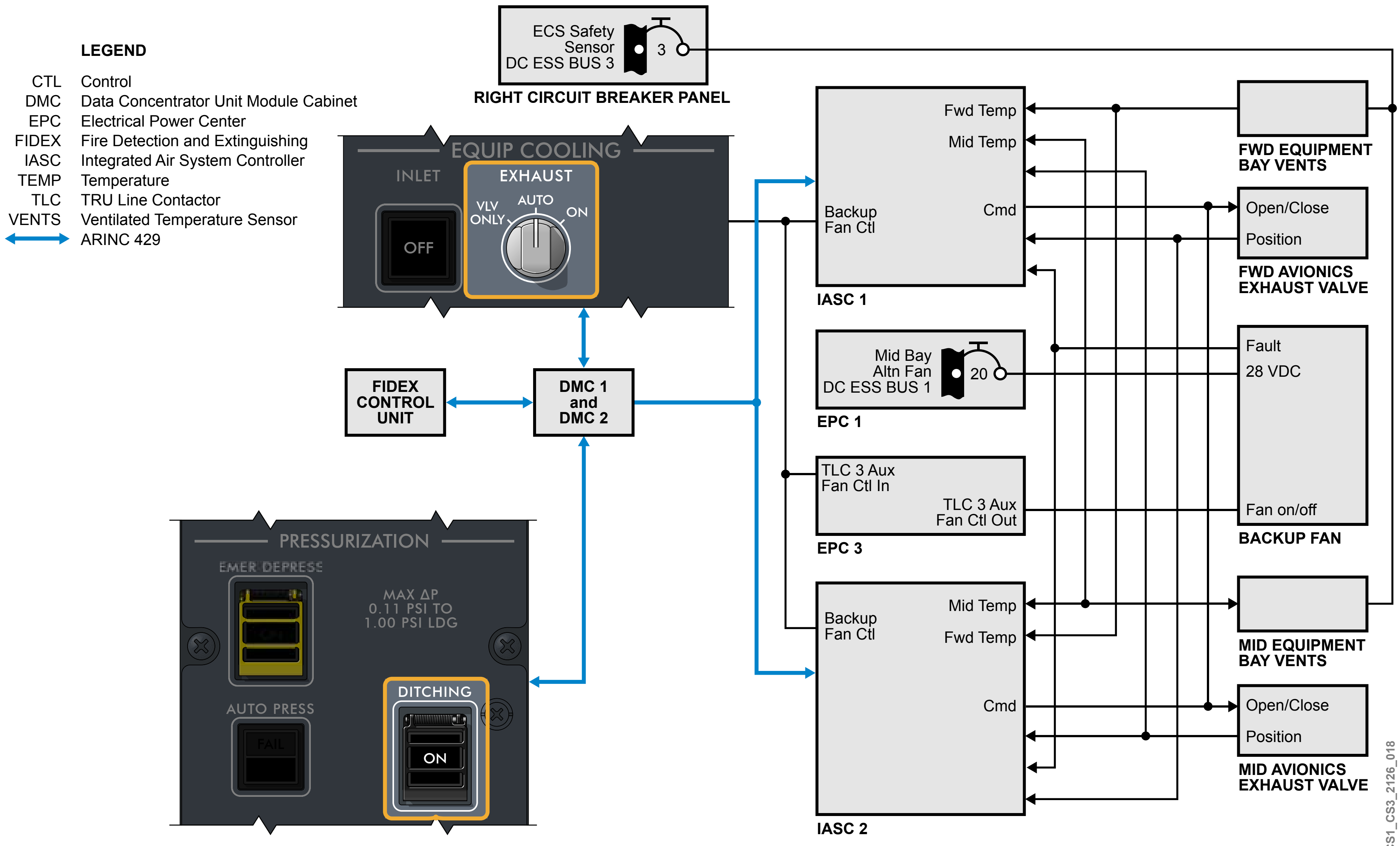


Figure 84: Backup Exhaust and Air Extraction System Schematic

COMPONENT INFORMATION

AVIONICS EXHAUST VALVES

The avionics exhaust valve (AEV) is an electrically actuated butterfly valve powered by a 28 VDC motor. The valve remains in its last position if power is lost.

Microswitches provide position feedback to the IASCs. The AEV has a manual lever which can used to secure the valve in the open position after a failure.

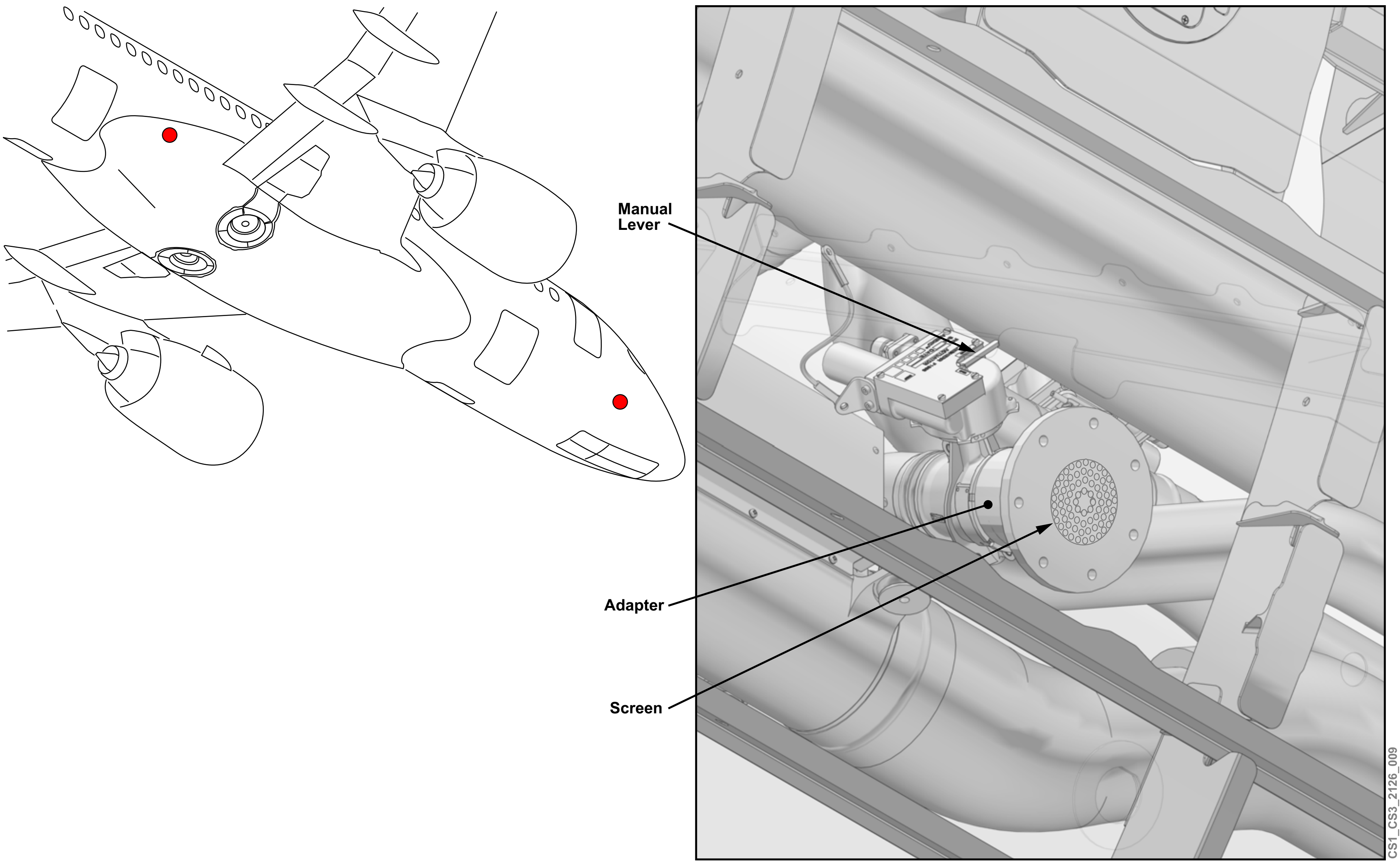


Figure 85: Avionics Exhaust Valve

BACKUP FAN

The backup fan is mounted on the ceiling of the mid equipment bay. The fan runs at a single-speed when powered. A check valve closes off the fan exhaust when the fan is not running.

The backup fan deflector is installed at the backup fan exit to direct smoke away from the floor sills where it could enter the passenger cabin.

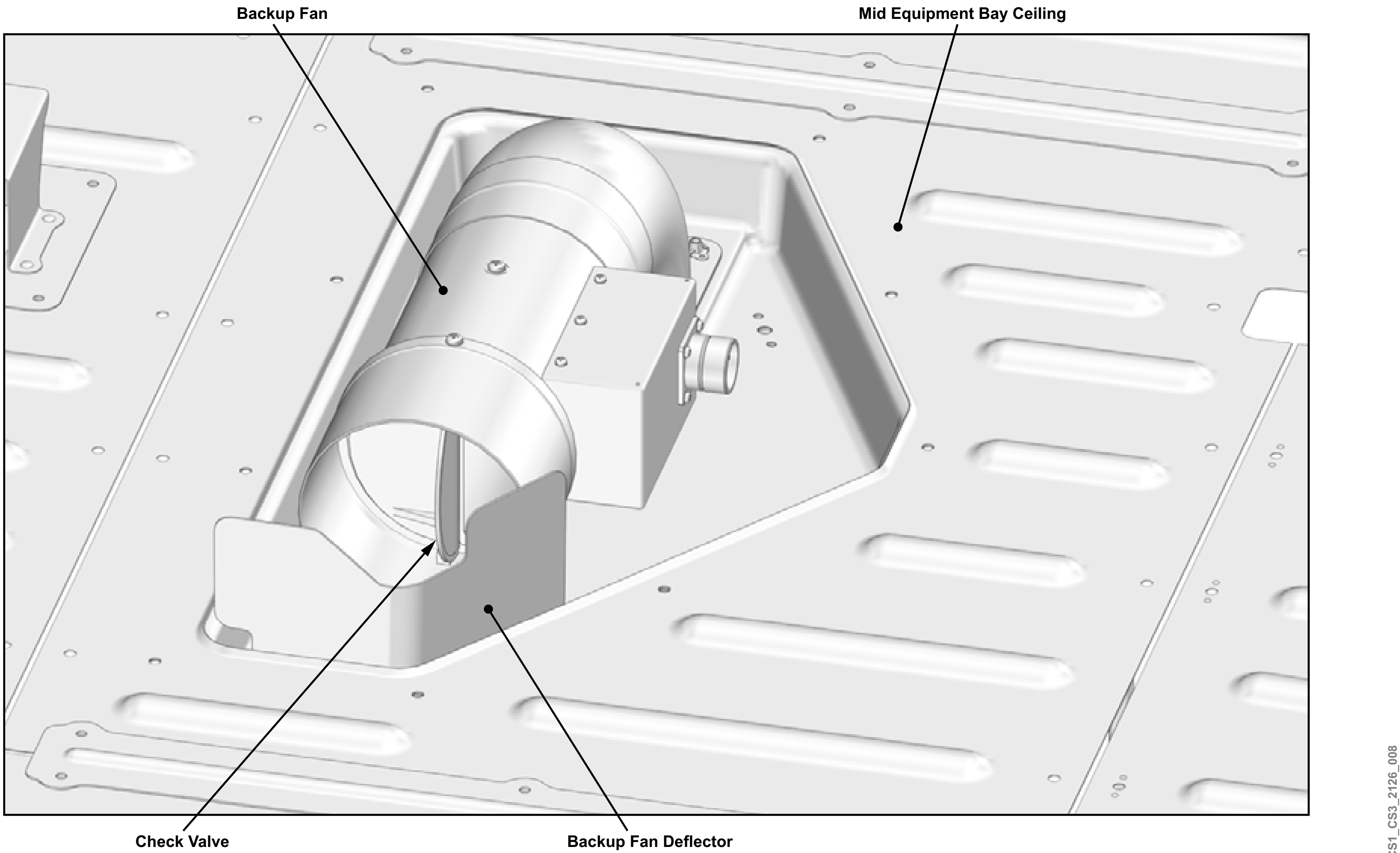


Figure 86: Backup Fan

CONTROLS AND INDICATIONS

The system is controlled from the EQUIP COOLING panel in the flight deck. In normal operation, the EXHAUST rotary switch is in the AUTO position and the system operation is automatically controlled by the IASCs.

If a fault occurs in the main extraction system, the IASC automatically selects the backup exhaust and extraction system. The EXHAUST rotary switch is moved to the ON position to match the system configuration.

If smoke is detected in an avionics bay, the backup extraction fan can be selected off using the VLV ONLY rotary switch position.

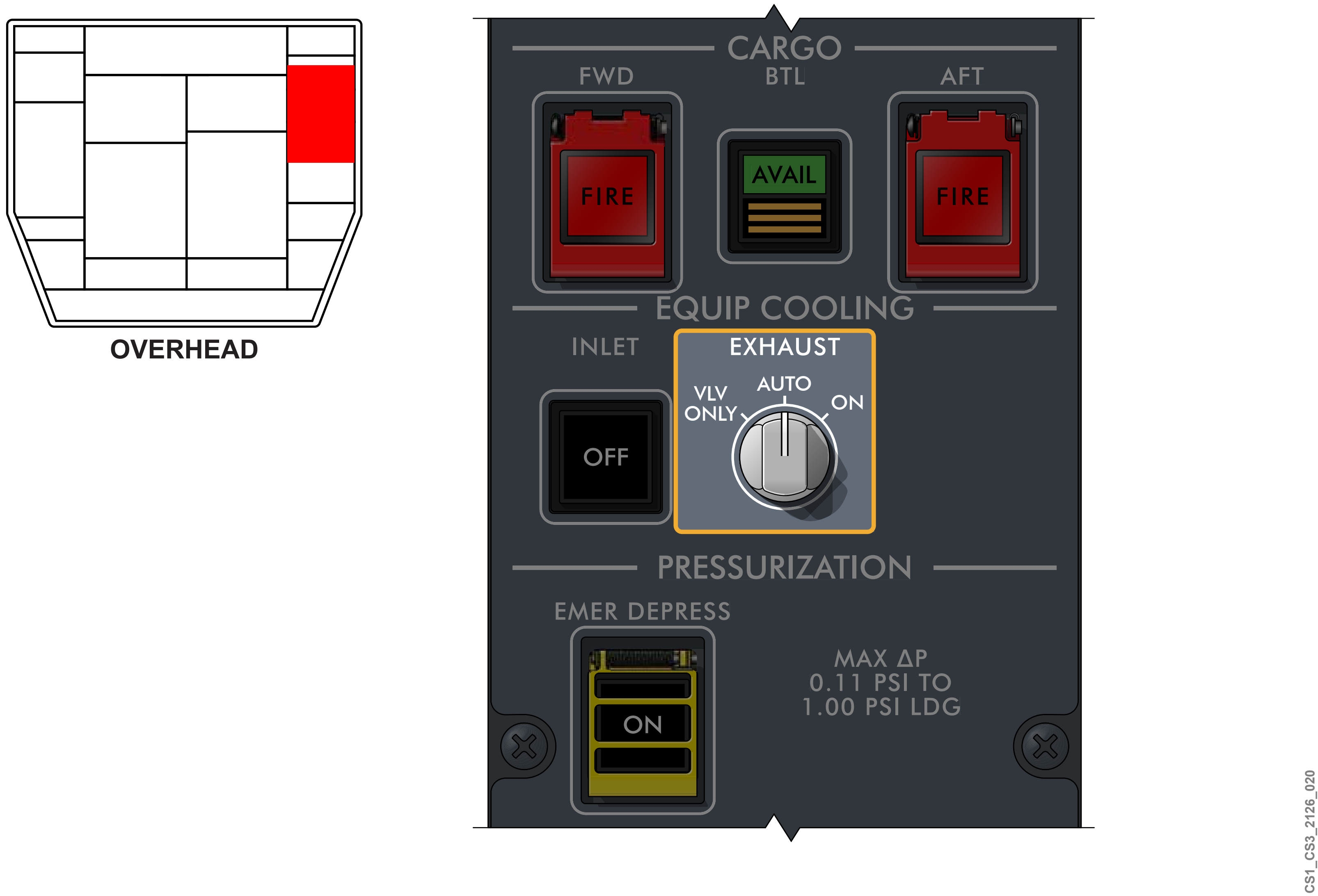


Figure 87: Avionics Cooling and Extraction System - Controls and Indications

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages associated with the avionics cooling and extraction system:

CAS MESSAGES

Table 25: WARNING Message

MESSAGE	LOGIC
EQUIP BAY OVHT	Fwd or mid avionics bay overtemperature above 67° C.

Table 26: CAUTION Message

MESSAGE	LOGIC
EQUIP BAY COOL FAIL	Loss of both extraction FANS confirmed by either IASC control or safety channel. Both FANS are running. Total loss of one zone temperature measurement, with electrical power supplied, and with either IASC safety channel operational.

Table 27: ADVISORY Messages

MESSAGE	LOGIC
EQUIP BAY COOL FAULT	FWD or MID equipment bay cooling fault

Table 28: STATUS Messages

MESSAGE	LOGIC
EXHAUST AIR ON	Exhaust switch selected to ON. Supplemental fan turns on and valves open.
EXHAUST AIR VLV ONLY	EXHAUST selector to VLV ONLY and both AEV commanded OPEN.

Table 28: STATUS Messages

MESSAGE	LOGIC
INLET AIR OFF	INLET PBA selected OFF and both forward and mid inlet fans commanded OFF and the mid ground valve commanded closed.

Table 29: INFO Messages

MESSAGE	LOGIC
21 EQUIP BAY COOL FAULT - EFAN INOP	Extraction FAN 1 or FAN 2 inoperative.
21 EQUIP BAY COOL FAULT - EFAN CAN BUS INOP	Extraction FAN 1 or FAN 2 CAN BUS inoperative.
21 EQUIP BAY COOL FAULT - ALTN FAN INOP	Backup FAN inoperative.
21 EQUIP BAY COOL FAULT - SUPP FAN INOP	Supplementary bay fan inoperative.
21 EQUIP BAY COOL FAULT - IFAN INOP	FWD or MID Inlet FAN inoperative.
21 EQUIP BAY COOL FAULT - AVIO TEMP SNSR REDUND LOSS	One channel of either FWD or MID or SUPP bay ventilated sensor failed.
21 EQUIP BAY COOL FAULT - FWD AVIO EXHAUST VLV INOP	FWD AEV failed in position or unknown position.

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Table 29: INFO Messages

MESSAGE	LOGIC
21 EQUIP BAY COOL FAULT - MID AVIO EXHAUST VLV INOP	MID AEV failed in position or unknown position.
21 EQUIP BAY COOL FAULT - MID GND VLV INOP	MID equipment bay ground valve failed in position.

WING-TO-BODY FAIRING VENTILATION SYSTEM

GENERAL DESCRIPTION

The wing-to-body fairing (WTBF) ventilation system provides ventilation and heat extraction in the forward section of the body fairing to limit the temperature of the WTBF composite structure.

The WTBF ventilation system induces secondary airflow to extract heat radiated by the packs and the high-pressure bleed air ducting.

Each air conditioning unit (ACU) ram air exhaust has a low-pressure ejector and mixer duct to induce ventilation. The ejector and mixer duct are located just downstream the ACU plenum exhaust and ram air regulating valve (RARV).

When the packs are operating, the heat exchanger cooling air is discharged through the RARV into the ram air exhaust. This airflow draws the WTBF compartment air into the ram air exhaust to be discharged overboard.

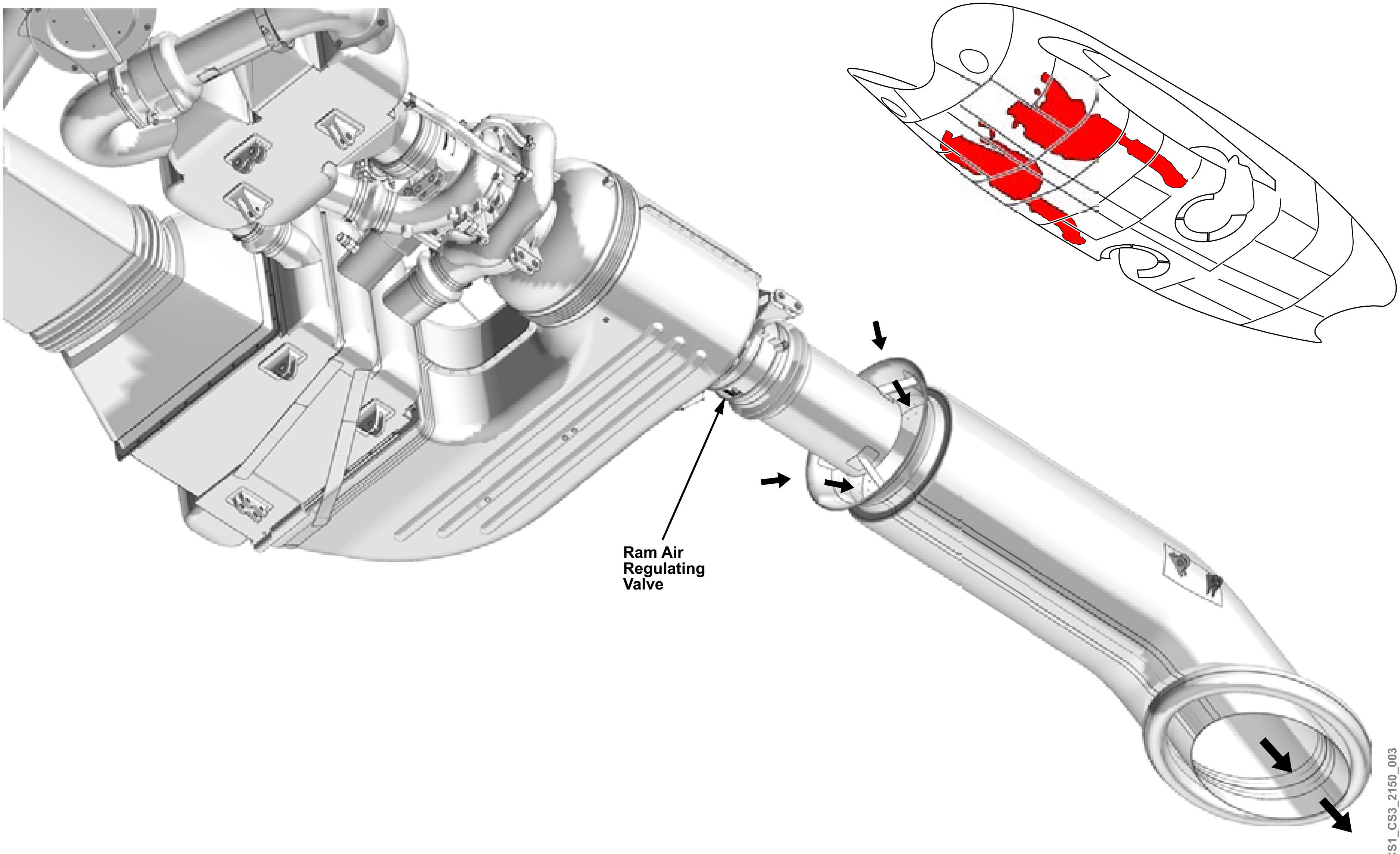


Figure 88: Wing-to-Body Fairing Ventilation System

HIGH-PRESSURE PNEUMATIC DUCT AREA COOLING

Ram air is used to cool the area around the high-pressure pneumatic duct. Air is tapped off the ram air inlet ducts at the heat exchanger inlet. Two supply tubes feed a piccolo tube that direct air towards the bulkhead area to ensure that components installed in that area are not overheated.

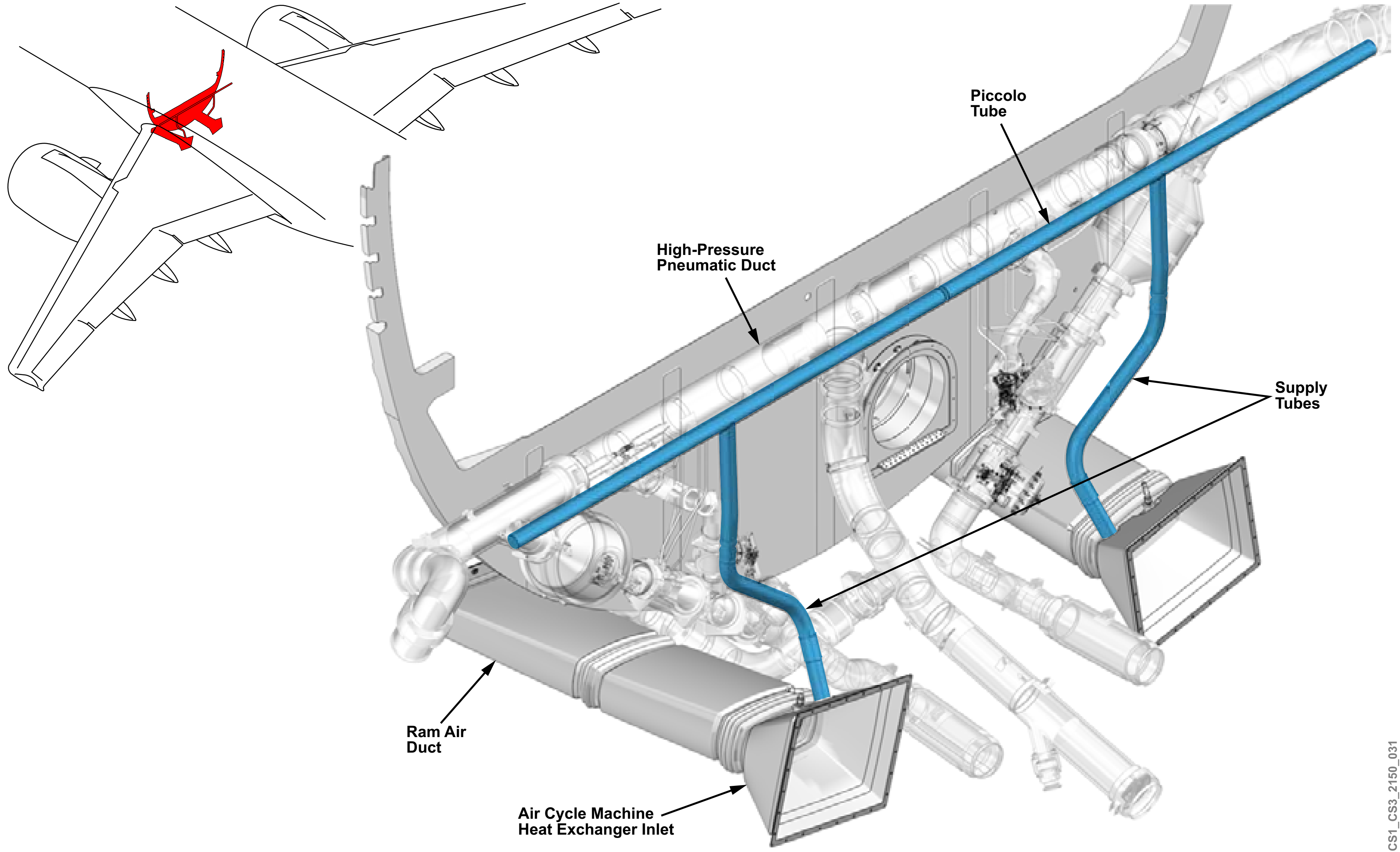


Figure 89: High-Pressure Pneumatic Duct Area Cooling

21-30 CABIN PRESSURE CONTROL SYSTEM

GENERAL DESCRIPTION

The cabin pressure control system (CPCS) automatically controls the pressure inside the cabin to a safe and comfortable level by modulating the flow of pressurized air through the outflow valve (OFV).

The system consists of two integrated air system controllers (IASCs). Each IASC has three channels: channel A, channel B, and a safety channel.

The pressurization system has three independent modes of operation each driving its own actuator on the outflow valve through the CABIN PRESSURE relay. There are two automatic modes controlled by IASC channel A and one manual mode operated from the PRESSURIZATION panel through the IASC safety channel. The manual mode also provides overpressure protection. Auto channel 1 is powered by DC ESS BUS 1, and auto channel 2 is powered by DC ESS BUS 2. The manual channel is powered by the IASC in control.

At takeoff, the outflow valve starts to close to pre-pressurize the cabin. During flight, the OFV is normally slightly open in order to maintain the cabin pressure. After landing, the OFV is driven full open to depressurize the aircraft. On the ground the OFV is normally fully open to equalize the inside pressure to the outside ambient pressure to ensure the doors can be opened safely.

AUTOMATIC MODE

In normal mode, the system operates automatically during all flight phases based on predetermined schedules using IASC channel A. The cabin altitude reference is based on inputs from aircraft systems through the data concentrator unit module cabinets (DMCs). Both IASCs are active in automatic mode, but only one is in control. IASC 1 is in control on odd days, and IASC 2 is in control on even days. If one IASC fails in auto mode, the second IASC automatically takes over.

MANUAL MODE

The manual mode is selected using the AUTO PRESS PBA. IASC channel B controls the outflow valve through the safety channel, and the MAN RATE rotary switch adjusts the cabin rate of change. In this mode, both automatic channels are inactive. Only one manual channel is in control at a time.

DITCHING

The DITCHING function is always available, whether the system is in AUTO mode or in MANUAL mode. When the DITCHING PBA is selected, the OFV is driven open to depressurize the aircraft. The packs are shut OFF automatically, and all valves below the flotation line are closed. Prior to ditching, the OFV is driven close to avoid water intake.

EMERGENCY DEPRESSURIZATION

Emergency depressurization is always available in AUTO or MANUAL mode. When the EMER DEPRESS PBA is selected, the OFV is driven open. The rapid depressurization function has protection against excessive cabin rate of change, and cabin altitude.

SAFETY FUNCTIONS

Two safety valves (SFVs) protect the pressurized fuselage against overpressure conditions. The SFV open if the cabin differential pressure begins to exceed the structural limits. The SFVs also provide negative pressure-relief. A negative pressure-relief valve (NPRV) provides additional protection against negative differential pressure.

Pressure equalization valves (PEV) maintain pressure equilibrium between the pressurized cabin, and the cargo compartment. The large PEV allows the air from cabin into the cargo compartment, while the small PEV allows the airflow out of the cargo compartment into the cabin.

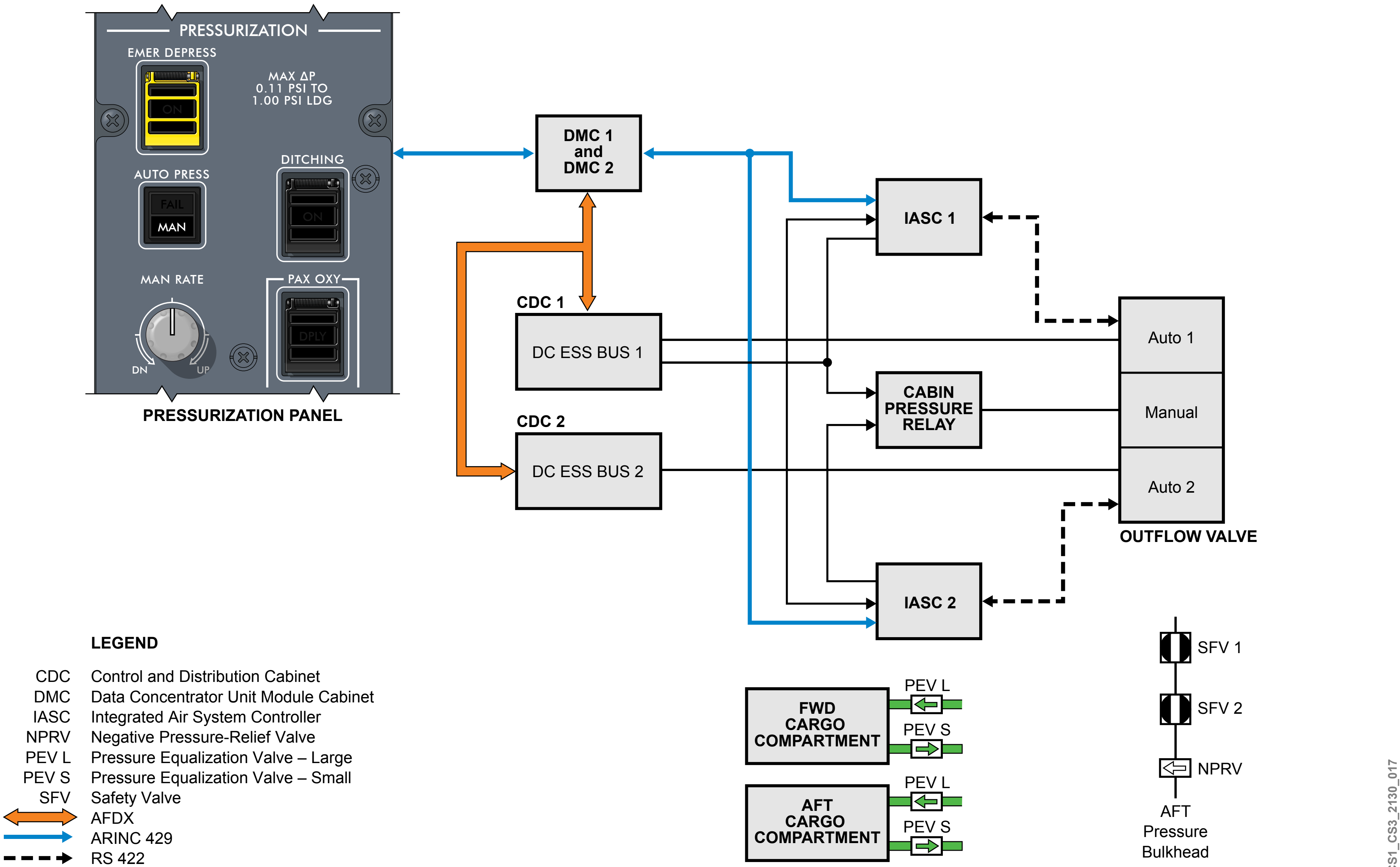


Figure 90: Cabin Pressure Control System

TYPICAL FLIGHT PROFILE

In normal automatic cabin pressure control mode, a typical flight profile includes the following sequences:

- Prepressurization
- Takeoff
- Return to base
- Climb
- Cruise
- Descent
- Depressurization

Prepressurization

Prepressurization starts when the aircraft is on the ground and the thrust levers are advanced to the takeoff range. The aircraft pressurizes to 300 ft below the takeoff field elevation to minimize the pressure bump at takeoff.

Takeoff

When the aircraft leaves the ground, the takeoff sequence is initiated. The takeoff sequence is maintained until the aircraft is more than 6,000 ft above the takeoff field elevation or 10 minutes has elapsed. The main function of the takeoff sequence is to avoid having to reselect the landing altitude in case of a need to return and land at the takeoff field.

Return-to-Base

The return-to-base sequence starts if the aircraft descends at more than 500 fpm for more than 10 seconds during the takeoff sequence. The aircraft depressurizes to 300 ft below the landing field elevation.

Climb

The climb sequence starts 6000 ft above takeoff altitude or 10 minutes after takeoff. The cabin rate of climb is controlled for passenger comfort. The cabin climb profile varies directly with the aircraft climb rate.

Cruise

The cruise sequence starts when the aircraft rate of climb is less than 500 fpm for 10 seconds. The cabin altitude is maintained in accordance with the pressurization schedule. The cabin altitude is controlled to a maximum of 7840 ft at 41000 ft aircraft altitude.

Descent

The descent sequence starts when the aircraft descends at more than 500 fpm for 10 seconds. Cabin altitude descends following the pressurization schedule until the cabin altitude is 300 ft below the landing field elevation. This minimizes the pressure bump at landing.

Depressurization Mode

The depressurization sequence starts when the aircraft lands. The cabin pressurizes to the field elevation at a rate of 300 fpm for 45 seconds, and then is driven fully open to minimize any pressure differential that might prevent the doors from opening safely.

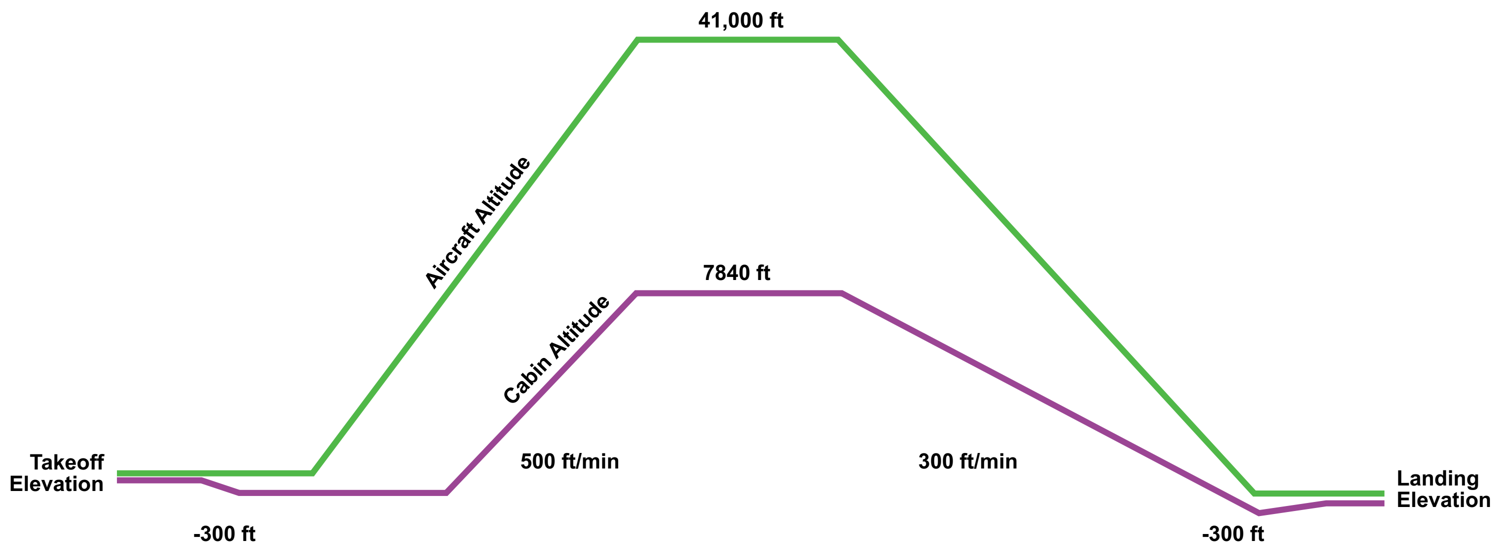


Figure 91: Typical Flight Profile

COMPONENT LOCATION

The following components are part of the cabin pressure control system:

- Outflow valve (OFV)
- Water activated check valve
- Muffler
- Safety valve (SFV)
- Negative pressure-relief valve (NPRV)
- Pressure equalization valves (PEVs)

OUTFLOW VALVE

The outflow valve is located in the environmental control system bay on the bulkhead.

WATER ACTIVATED CHECK VALVE

The water activated check valve is located in the wing-to-body-fairing (WTBF).

MUFFLER

The muffler is located in the WTBF.

SAFETY VALVE

There are two safety valves (SFVs) located on the aft pressure bulkhead.

NEGATIVE PRESSURE-RELIEF VALVE

The negative pressure-relief valve (NPRV) is located on the aft pressure bulkhead.

PRESSURE EQUALIZATION VALVES

The pressure equalization valves (PEVs) are located in the cargo compartment entrance way. The forward cargo compartment PEVs are aft of the cargo door. The aft cargo compartment PEVs are forward of the cargo door.

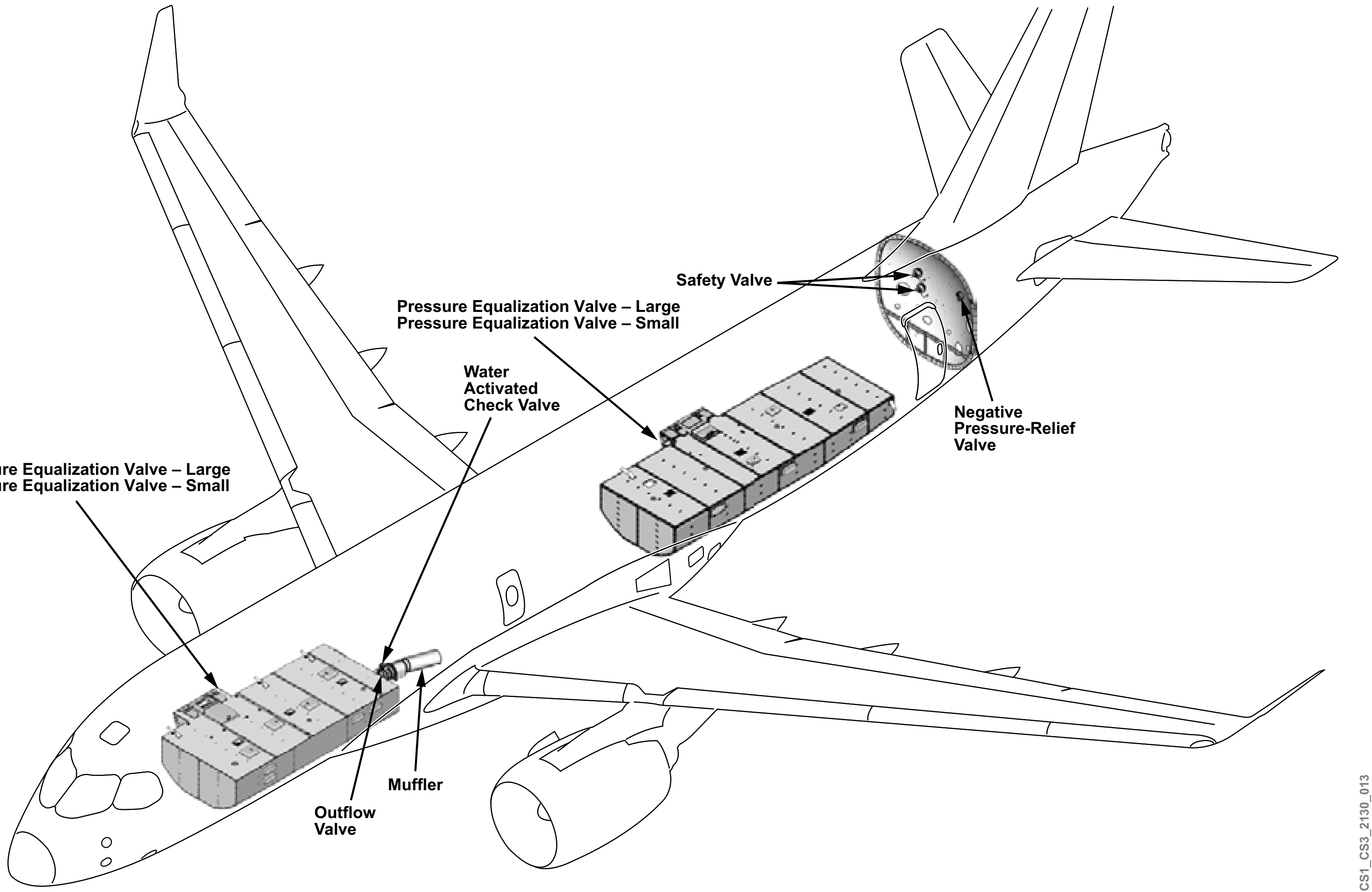


Figure 92: Pressurization Components Location

COMPONENT INFORMATION

OUTFLOW VALVE

The outflow valve modulates the discharge airflow to control the cabin pressure in both AUTO modes and in MANUAL mode. The OFV actuator assembly consists of two identical auto channels and a manual channel.

A travel limiter physically limits the outflow valve opening when deployed. When the aircraft altitude exceeds 17,000 ft, the travel limiter pin extends to limit the outflow valve travel.

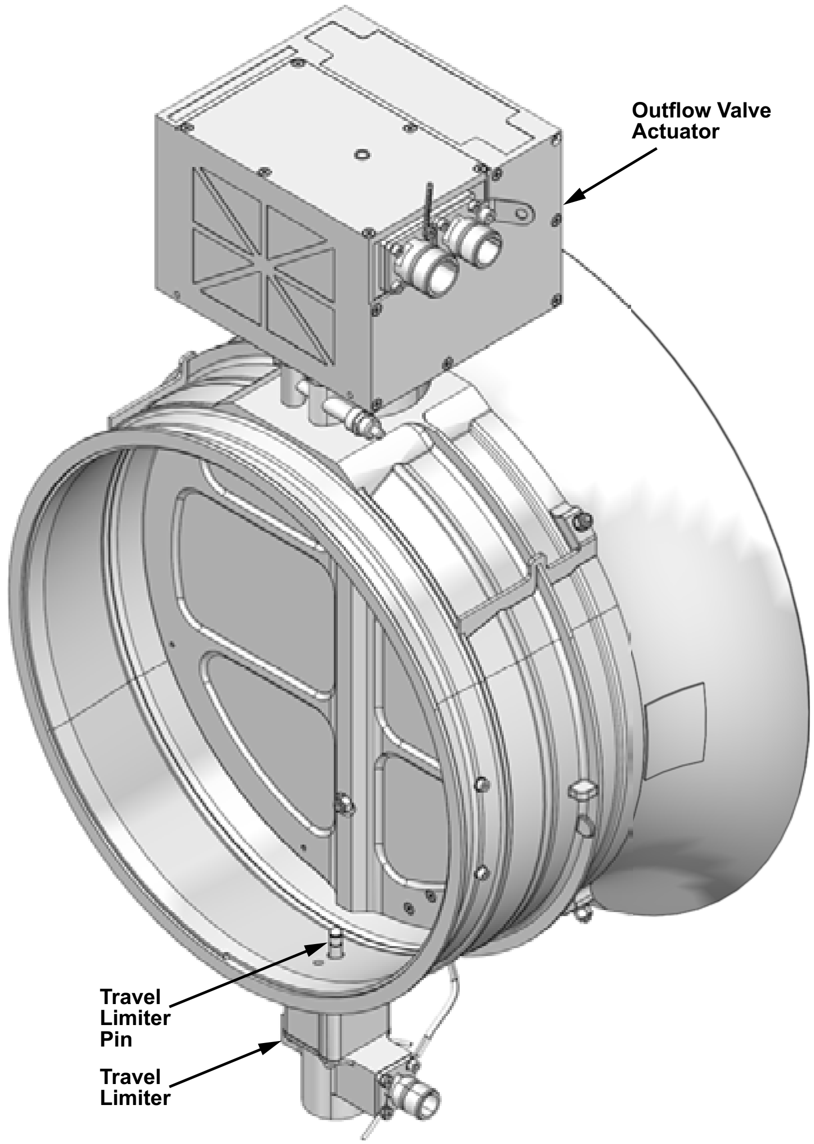


Figure 93: Outflow Valve

WATER ACTIVATED CHECK VALVE

The water activated check valve is a mechanical flapper installed on the environmental control system bay bulkhead. The flapper is supported by a bracket assembly under normal conditions. If the aircraft ditches, the rising water level closes the flapper.

MUFFLER

The optional muffler reduces the outflow valve discharge air noise level.

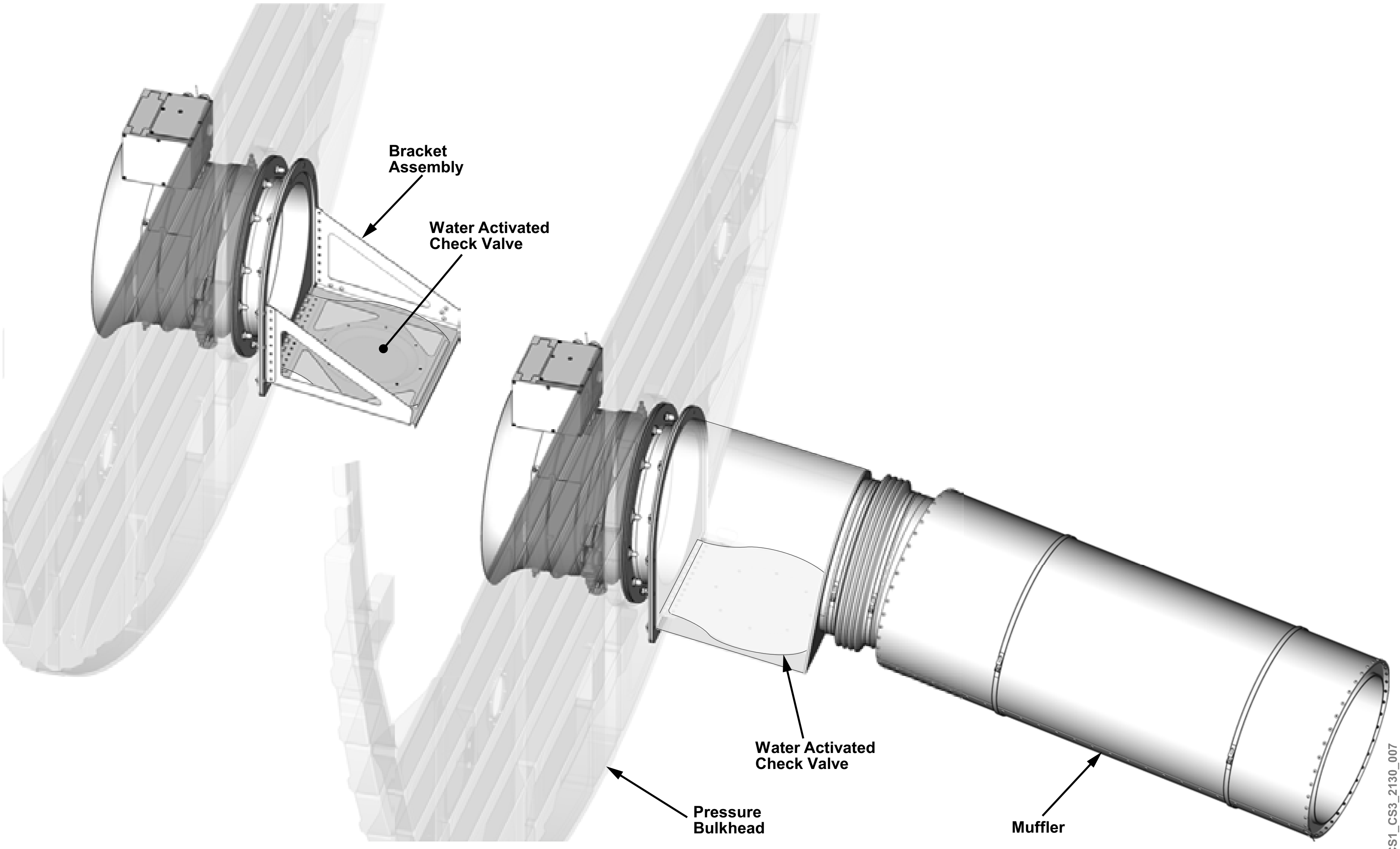


Figure 94: Outflow Valve, Water Activated Check Valve, and Muffler

SAFETY VALVE

Two safety valves protect against any overpressure condition. The safety valves are pneumatically-actuated on the ground or in flight. The SFVs provide protection for both overpressure relief, and negative pressure-relief.

NEGATIVE PRESSURE-RELIEF VALVE

The negative pressure-relief valve (NPRV) is a mechanical check valve with spring-loaded flappers. The NPRV is acted on by ambient pressure on one side, and cabin pressure on the other side to sense the differential pressure.

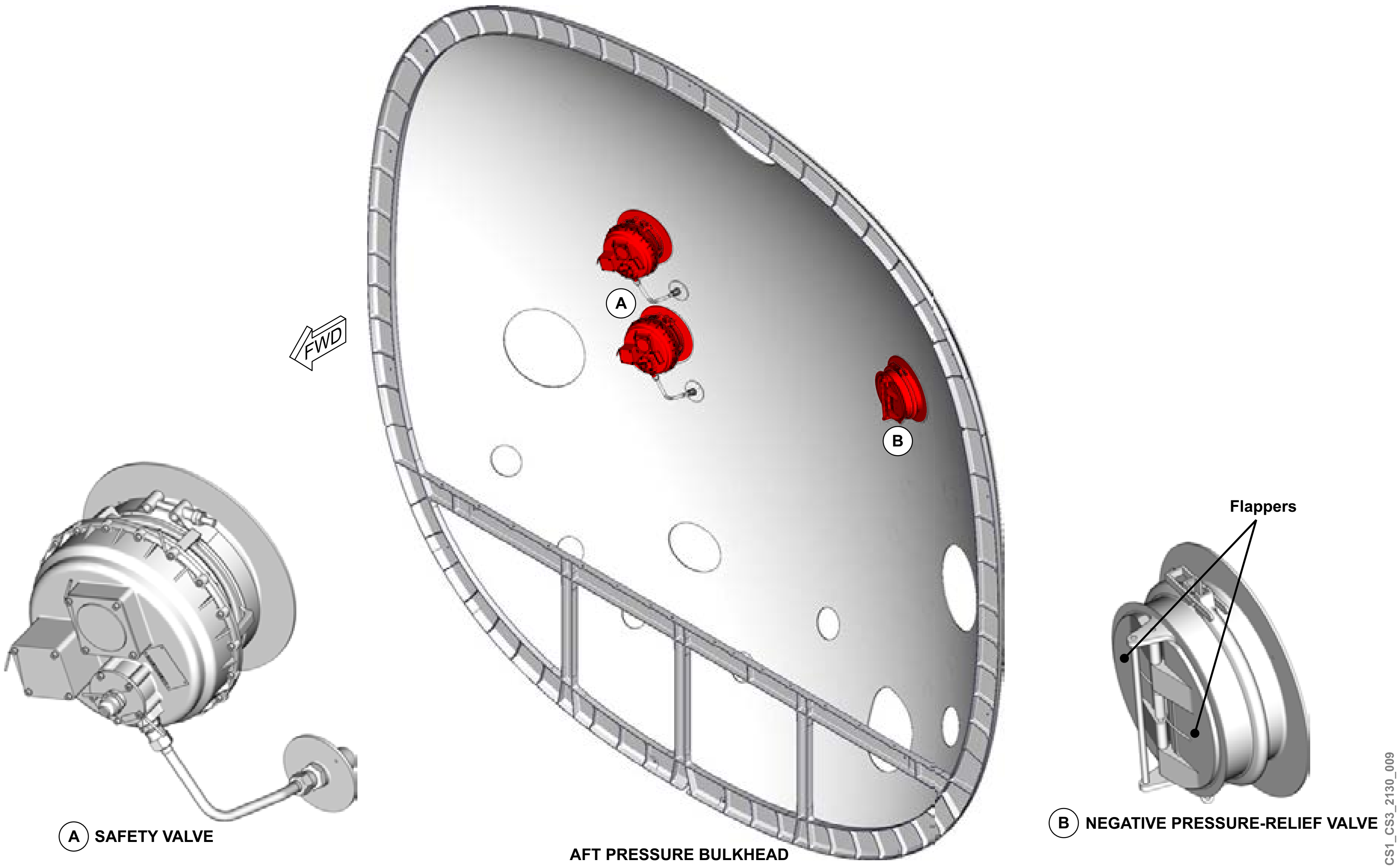


Figure 95: Safety Valves and Negative Pressure-Relief Valves

PRESSURE EQUALIZATION VALVES

The system maintains pressure equalization between the cabin and forward and aft cargo using four pressure equalization valves (PEVs). The PEV is a fully mechanical device that prevents excessive pressure between the cargo compartment and cabin underfloor in both directions (inflow and outflow). Two PEVs are installed on each cargo compartment entryway.

Each cargo compartment has one large cargo PEV, and one small PEV. The large PEV allows the air from the cabin into the cargo compartment, and the small PEV allows the airflow out of the cargo compartment into the cabin.

The large PEV prevents the cabin from pressurizing when the cargo doors are not safely closed, and prevents the cargo blowout panels from opening.

The small PEV allows cargo compartment pressure to equalize pressure with cabin pressure during fast climb.

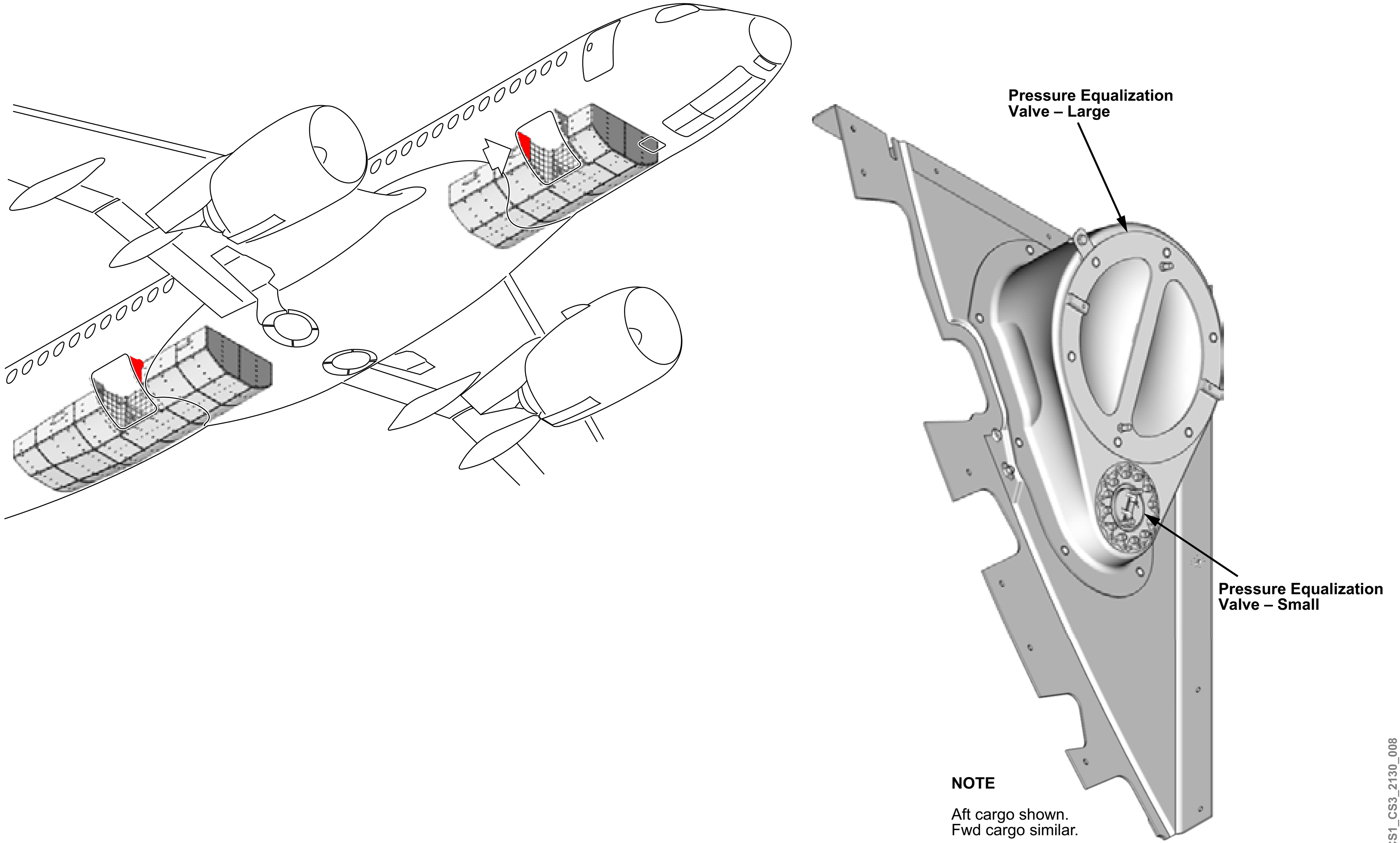


Figure 96: Pressure Equalization Valves

DETAILED COMPONENT INFORMATION

INTEGRATED AIR SYSTEM CONTROLLER

The system consists of two identical integrated air system controllers (IASCs). Each IASC incorporates channel A and channel B and a safety channel.

A cabin pressure sensor is fitted to each channel A and channel B of the IASC so that the cabin pressure control system receives four independent cabin pressure signals. The safety channel receives cabin pressure information from the channel B sensor.

The safety channel limits cabin altitude and cabin rate of change, and sends cabin altitude and cabin pressure rate of change to the data concentrator unit module cabinet (DMC) by an ARINC 429 BUS.

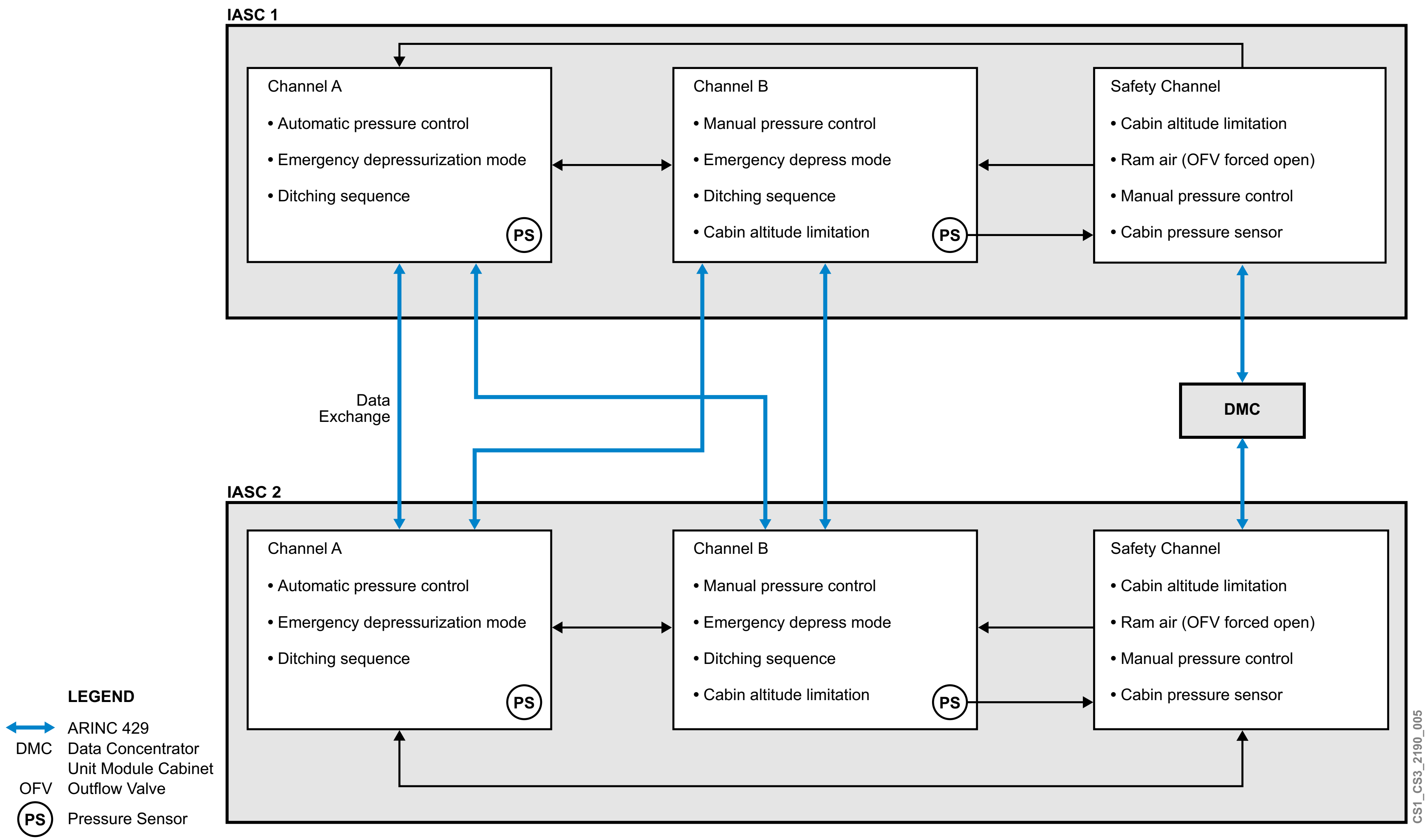


Figure 97: Integrated Air System Controller

CONTROLS AND INDICATIONS

PRESSURIZATION PANEL

The PRESSURIZATION panel has controls for:

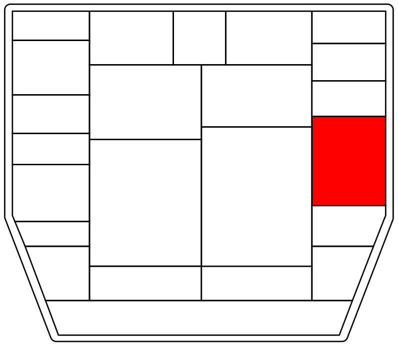
- Automatic cabin pressure control
- Manual cabin pressure control
- Rapid cabin pressure-relief (EMER DEPRESS)
- Ditching

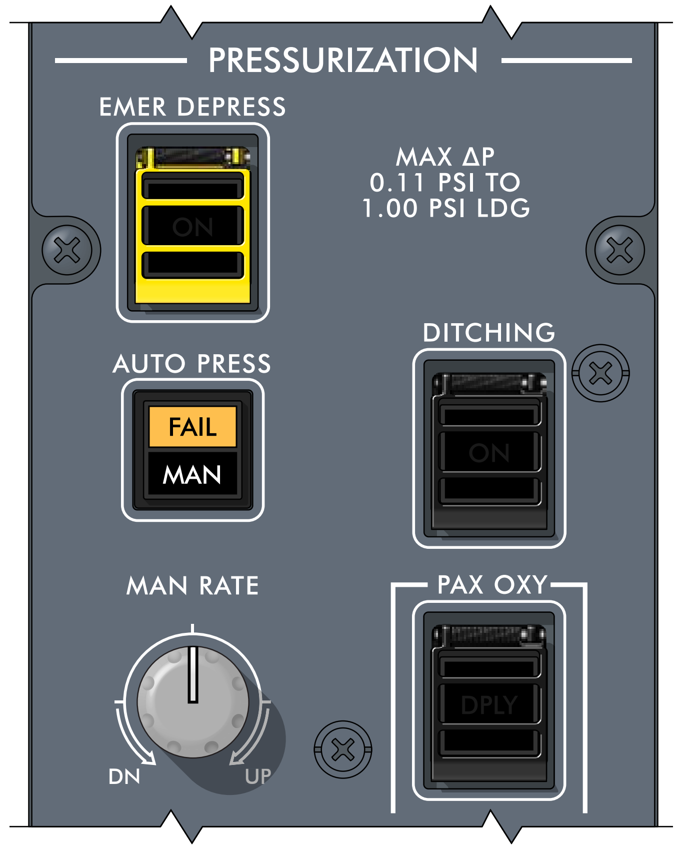
The system is normally in AUTO mode for cabin pressurization. If the cabin pressure AUTO fails, the amber FAIL light illuminates. Selecting the AUTO PRESS PBA to MAN, switches the cabin pressure control mode to MANUAL.

In manual mode, the MAN RATE knob adjusts the cabin altitude. Counterclockwise selection decreases the cabin altitude at a maximum rate of -2,500 ft/min. Clockwise selection increases the cabin altitude at a maximum rate of +2,500 ft/min. There are detents at the ±1,000 ft positions on the switch.

Selecting the guarded EMER DEPRESS PBA to ON, opens the outflow valve. The cabin altitude and rate limitations functions (15,000 ft) remain active. The white OPEN legend indicates the outflow valve is being forced open.

The DITCHING PBA closes all of the valves mounted on the aircraft located below the flotation line, in preparation for a ditching. Selecting the guarded DITCHING PBA opens the outflow valve when altitude is below 15,000 ft. All valves are closed electrically. Prior to ditching, the outflow valve is driven closed.





PRESSURIZATION PANEL

CS1_CS3_2130_004

Figure 98: Pressurization Panel

FMS PAGE

The landing elevation is selected via the flight management system (FMS) through an automatic mode or two backup manual modes. When the flight plan is programmed in the FMS ROUTE tab, the landing elevation of arrival airport data is displayed and also sent via an ARINC 429 BUS to the IASCs.

The landing elevation is also set using:

- Control tuning panel (CTP)
- AVIONIC CTP page

Using the CTP, the LDG ELEV left line select key switches between FMS and MAN, and the LDG ELEV right line select key changes the MAN elevation.

The CTP input can be overridden using the avionics virtual CTP FMS/MAN LDG ELEV selection to set the landing elevation manually.

In either manual mode, the landing elevation is sent to the IASCs via an ARINC 429 BUS.

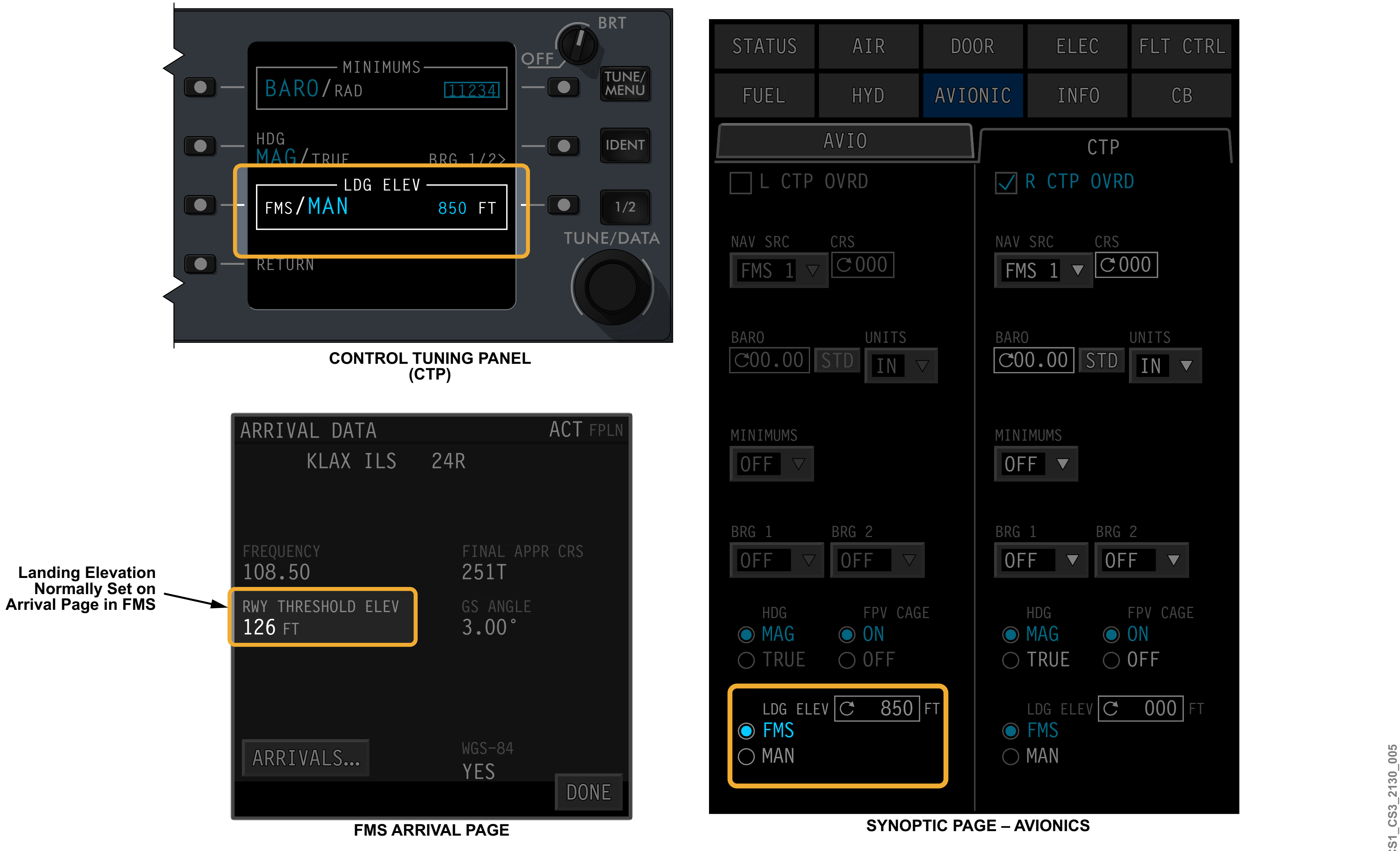


Figure 99: FMS, Control Tuning Panel, and AVIONIC CTP Controls

INDICATIONS

The cabin pressure control system information is displayed on the engine indication and crew alerting system (EICAS) page or the AIR synoptic page.

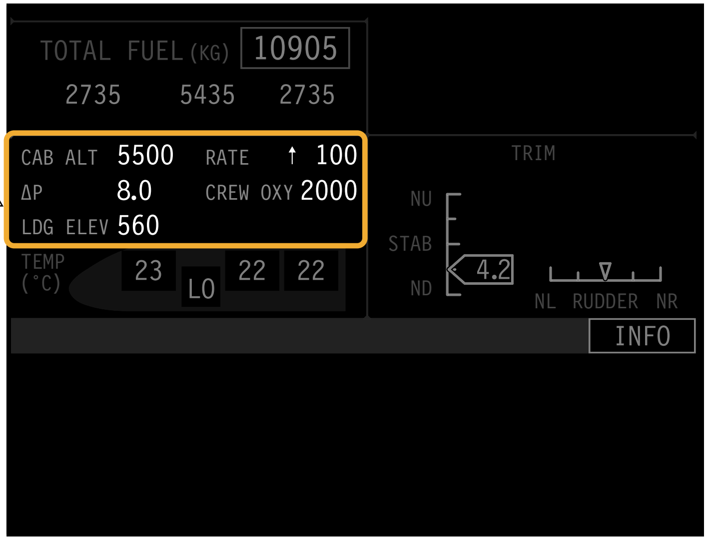
The CAB ALT value is the highest recorded value by any of the six IASC channels.

If the landing elevation is set manually, MAN appears next to the LDG ELEV value.

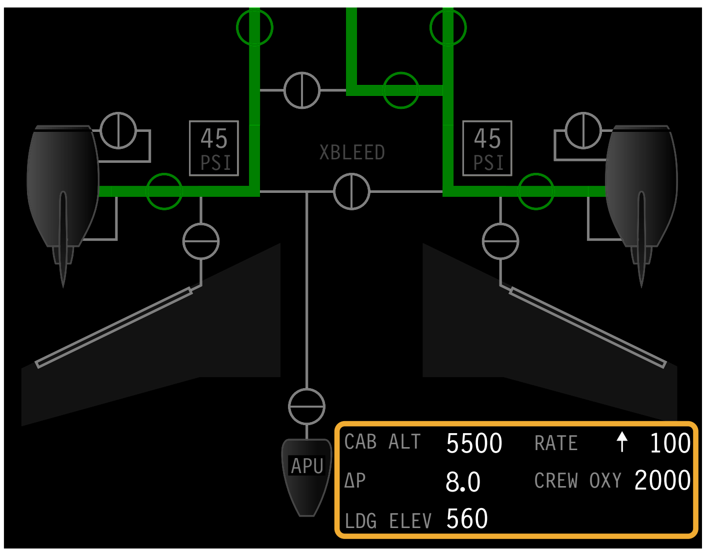
An arrow is displayed beside the RATE to indicate the direction of the cabin rate of climb.

- If the cabin rate of climb increases, an upward white arrow appears
- If the cabin rate of climb decreases, a downward white arrow appears.

1



EICAS PAGE



2

SYNOPTIC PAGE – AIR

CAB ALT	
Value	**Description**
XXXXX (red)	Cabin altitude above warning threshold
XXXX or XXXXX (yellow)	Cabin altitude above caution threshold
XXXXX	Cabin altitude within normal range
------	Invalid value

RATE	
Value	**Description**
↑XXXXX (cyan)	Cabin altitude manually selected or pressurization system in manual mode
↑XXXXX	Cabin altitude actual value
-----	Invalid value

ΔP	
Value	**Description**
X.X (red)	Delta P above warning threshold
XX.X	Delta P within normal range
--.-	Invalid value

3

LDG ELEV	
Value	**Description**
560	LDG ELEV selected from FMS
560 MAN	LDG ELEV selected through CTP or AVIONIC synoptic page
-----	Invalid value

NOTE

1. Air block not shown in EICAS compressed mode.
2. Air block only shown on SYNOPTIC page when EICAS in compressed mode.
3. The cabin rate arrow and digital readout in cyan for 3 seconds when manually turns selected.

CS1_CS3_2130_006

Figure 100: EICAS and AIR Synoptic Page Indications

DETAILED DESCRIPTION

AUTO/MANUAL SELECTION

The system is normally in AUTO mode for cabin pressurization. If the IASC controlling the automatic pressurization fails, the standby channel automatically takes over. If both IASC automatic channels fail, the AUTO PRESS PBA indicates FAIL, and an AUTO PRESS FAIL caution message is displayed.

When the AUTO PRESS PBA is selected to MAN, the cabin pressure control mode switches to manual. The AUTO PRESS PBA MAN indication illuminates white and a CABIN PRESS MAN status message is displayed.

The AUTO PRESS PBA can also be used to select active channels if required. When in automatic mode, pressing the AUTO PRESS PBA selects the manual channel of the IASC. A second press of the pushbutton annunciator (PBA) selects the other IASC automatic channel. A third press of the PBA selects the other IASC manual channel.

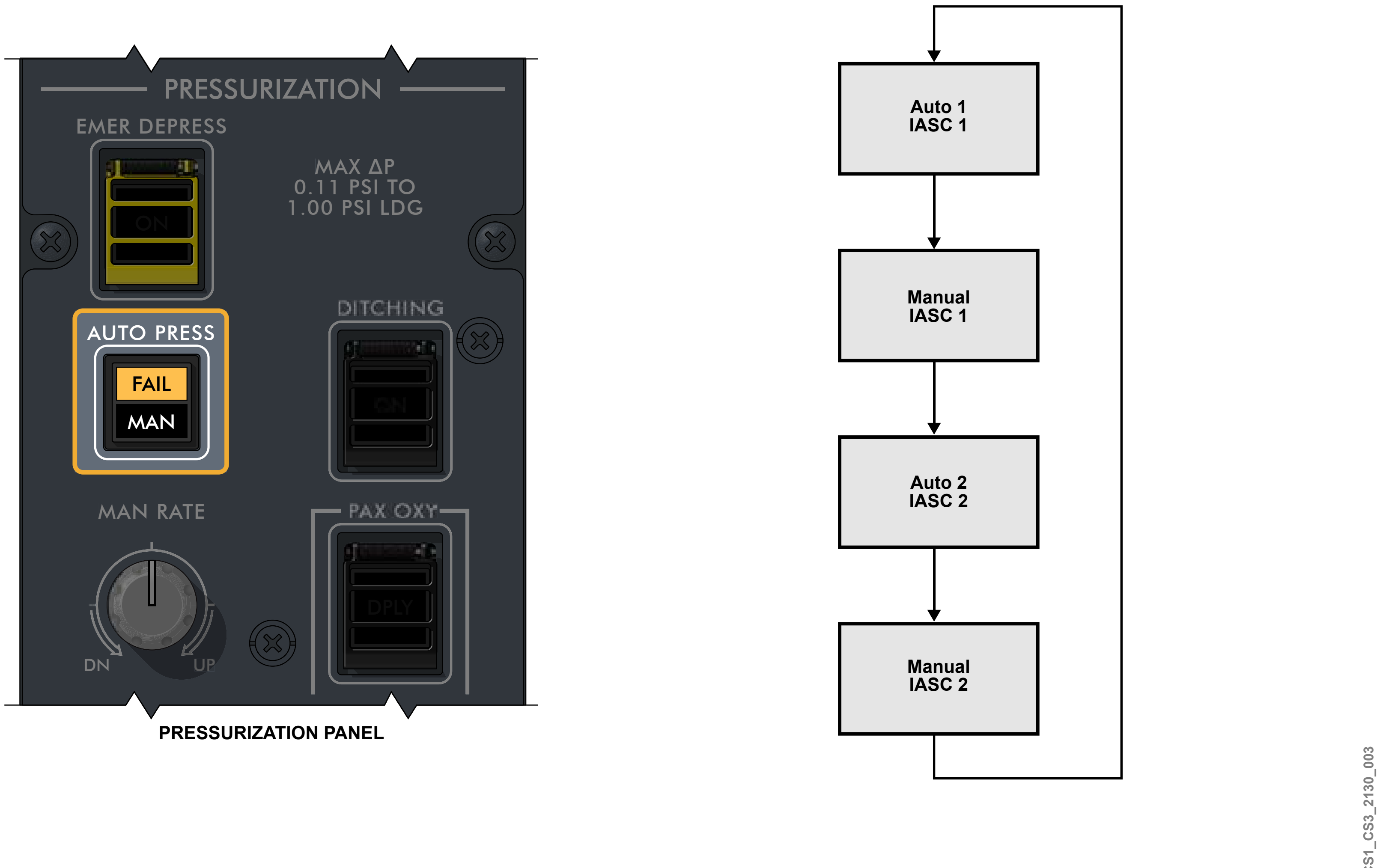


Figure 101: Cabin Pressure Control System Selection

INTERFACE

The outflow valve (OFV) actuator assembly consists of two identical auto channels and a manual channel. The auto channels of the OFV communicate with their corresponding integrated air system controller (IASC) channel A via an RS 422 BUS.

The manual channel is hardwired to the IASCs through the CABIN PRESSURE relay, located in electrical power center 2 (EPC 2). The CABIN PRESSURE relay allows both IASCs to manually command the OFV. Only one controller at a time is connected to the MANUAL channel of the OFV. The outflow valve commands are provided by the IASC in control. The outflow valve manual channel provides position feedback to each IASC.

When the aircraft is on ground and the doors are open, the IASCs perform a MANUAL channel test as follows:

- The valve is moved from the fully open position to the fully closed position in 5 seconds
- The valve is moved back to the fully open position in 5 seconds
- If the valve position feedback does not match the commanded valve position, the MANUAL channel is reported as failed

A travel limiter device physically limits the outflow valve opening when extended. The travel limiter extends based on a preset cabin differential pressure, either in automatic or manual modes, when the aircraft flies over 17,000 ft. The travel limiter is powered by IASC 1. The travel limiter provides position feedback to IASC 1.

The active IASC channel A software continuously monitors for any overpressure condition. If an overpressure condition is detected, the active IASC drives the OFV open.

A force open command signal is generated for an emergency depressurization, or when the emergency ram air valve is opened. The force open command removes power from the auto channels, and drives the OFV open using the manual channel.

The aircraft cannot be dispatched if the travel limiter pin is failed closed. If the pin fails in the extended position, the aircraft can be dispatched but is limited to a maximum altitude of 25,000 ft.

Cabin Altitude Limitation

The cabin altitude limitation system limits cabin altitude to a maximum of 15 000 ft. When the altitude limit is reached, a signal is sent through the CABIN PRESSURE relay to CDC 1 and CDC 2. The CDCs remove power from the auto channels of the OFV. The outflow valve is driven closed using the manual channel of the IASC that was in control.ever

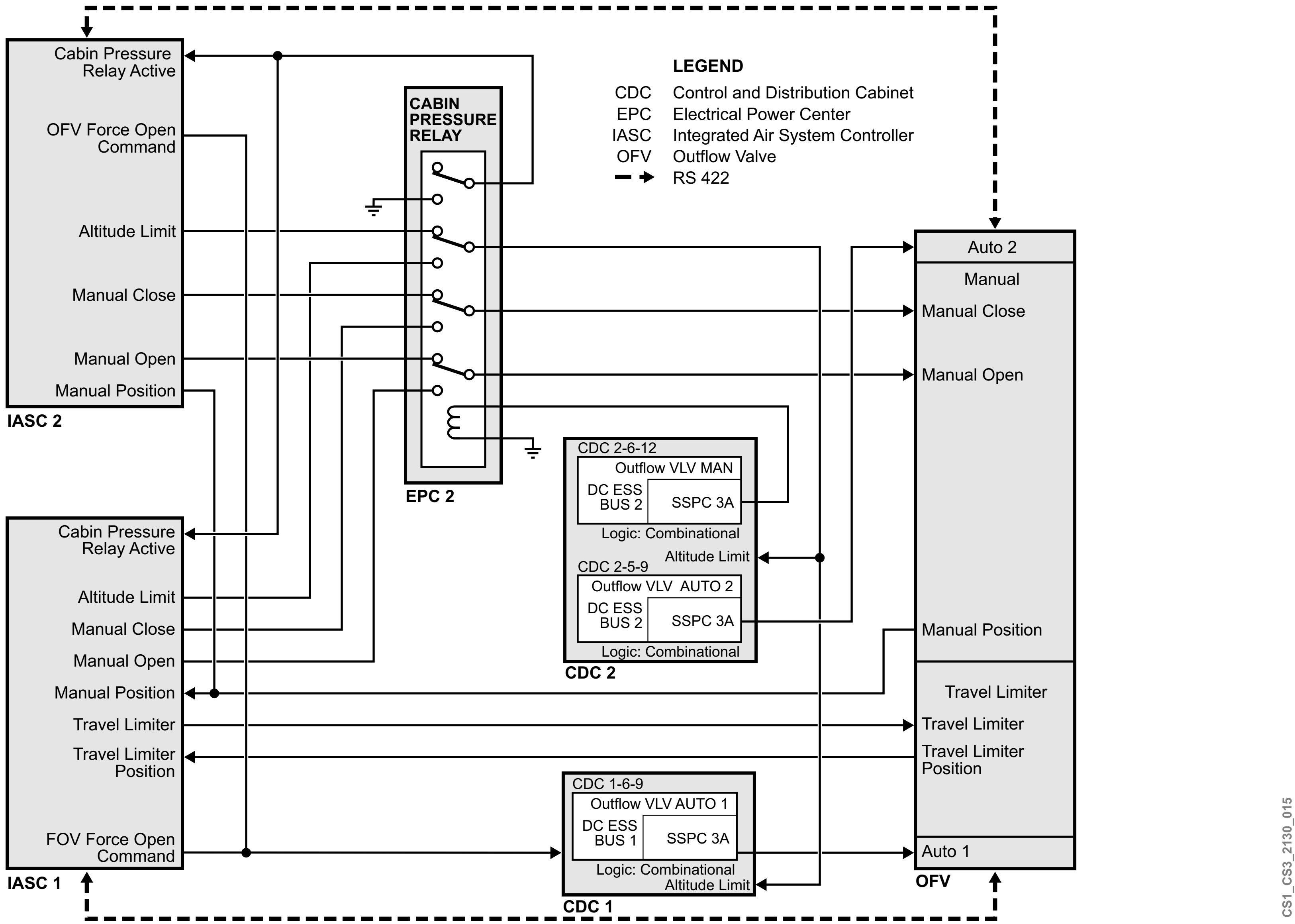


Figure 102: Cabin Pressure Control System Detailed Description

CABIN ALTITUDE LIMITATION ANNUNCIATION

The baro-corrected landing elevation (L/EBARO) has three modes of operation for the cabin altitude limitation. One is for normal operation, and the other modes are for high altitude airport operations (HAAO).They are defined as follows:

- Non-HAAO conditions defined for L/EBARO between -2000 ft and 8000 ft above sea level.
- HAAO-1 conditions defined for L/EBARO between 8000 ft and 14,000 ft above sea level
- HAAO-2 conditions are defined for L/EBARO between 14,000 ft and 14,500 ft above sea level

The cabin altitude caution and warning EICAS messages for non-HAAO operations, are set at:

- CAB ALT CAUTION 8500 ft
- CAB ALT WARNING: 9865 ft

CABIN ALTITUDE CAUTION LOGIC

The normal CAB ALT CAUTION threshold is 8500 ft.

If the landing altitude is above 8000 ft:

- When aircraft starts the descent and the aircraft altitude is below 37,000 ft, the CABIN ALTITUDE caution starts to increase as a function of aircraft altitude to a maximum of landing altitude + 500 ft and limited to 14,500 ft in HAAO-1 OR 14,843 ft in HAAO-2

The caution threshold increase rate is proportional to the A/C descent rate, aircraft altitude and L/EBARO.

If the takeoff is above 8000 ft:

- When aircraft starts its climb and if aircraft altitude is above takeoff altitude (TOA) + 5000 ft or 10 minutes after takeoff, the CABIN ALTITUDE caution starts to decrease as a function of aircraft altitude to it nominal cruise set point of 8500 ft

CABIN ALTITUDE WARNING LOGIC

The normal CAB ALT warning message threshold is 9865 ft.

If the landing altitude is above 8000 ft:

- When aircraft starts the descent, the CABIN ALTITUDE warning starts to increase as a function of cab altitude caution threshold + 1365 ft and limited to 14,500 ft in HAAO-1 or 14,843 ft in HAAO-2.

If the takeoff is above 8,000 ft:

- When aircraft starts its climb and if aircraft altitude is above TOA + 5000 ft or 10 minutes after takeoff, the cabin altitude warning starts to decrease as a function of aircraft altitude to it nominal cruise set point of 9865 ft

In either landing or take off at high altitude, a CABIN ALT LEVEL HIGH advisory message is displayed to indicate that the cabin altitude warning threshold has been set above 9865 ft.

The system limits cabin altitude to a maximum of 14843 ft in non-HAAO or HAAO-1 and 15200 ft in HAAO-2 in the event of a system malfunction.

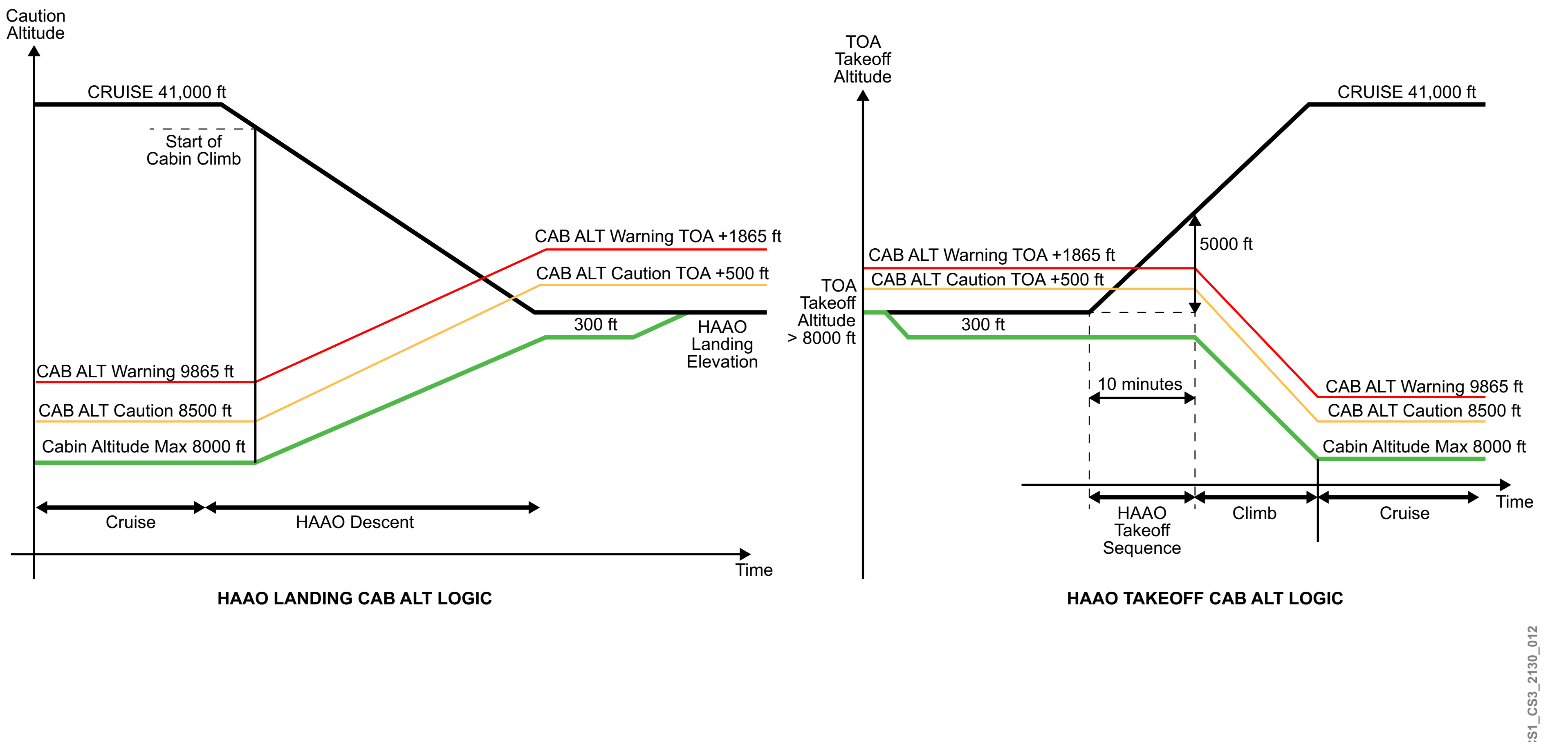


Figure 103: Cabin Altitude Limitations Annunciation

EMERGENCY DEPRESSURIZATION

The cabin pressure can be reduced rapidly towards ambient pressure in an emergency. This function is known as emergency depressurization. Emergency depressurization is always available, whether the system is in AUTO mode, or in MANUAL mode.

While in AUTO mode at the time of the emergency depressurization selection, a cabin rate of change limitation of 3500 ft/min is applied. In the MANUAL mode, the cabin rate of change limitation of 2500 ft/min is applied. The cabin altitude is limited to 14,500 ft.

When the EMER DEPRESS PBA is pressed, the OFV is driven open at a rate that protects against excessive cabin rate of change. The ON legend on the PBA indicates the OFV is being driven open. An EMER DEPRESS ON caution message is displayed on EICAS.

If the system was in automatic mode at the time of emergency depressurization selection, the function is accomplished by the active IASC channel A.

If the system was in manual mode at the time of the emergency depressurization selection, a force open command is supplied through control and distribution cabinet 1 (CDC 1) to the safety channel. The outflow valve is driven open through the manual channel.

The cabin altitude and rate limitations functions remain active. The CAB ALT caution and warning are also raised above the normal set point in order to monitor that the cabin altitude does not exceed 15,000 ft.

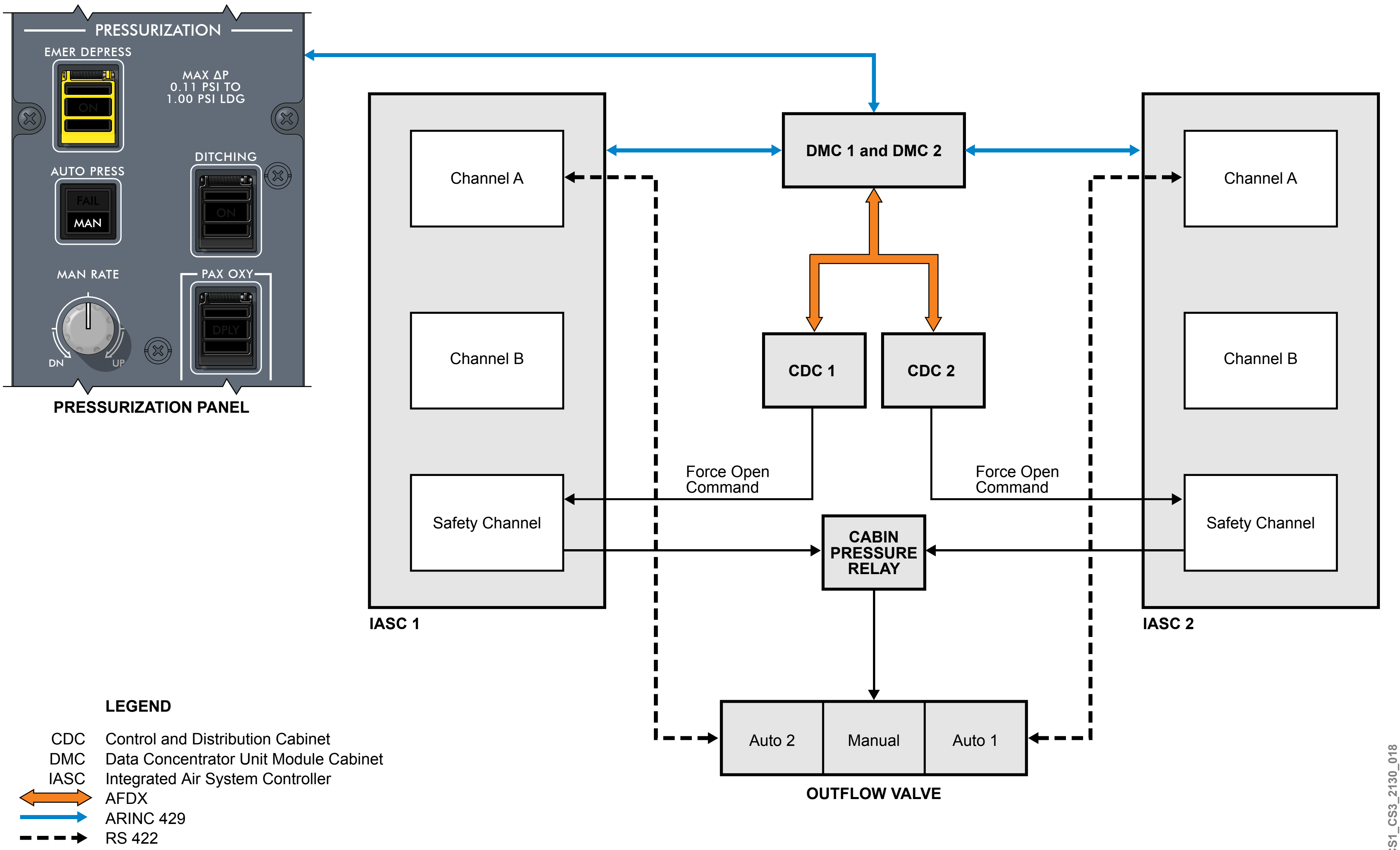


Figure 104: Emergency Depressurization

DITCHING

The ditching mode is used to close all of the valves below the aircraft flotation line. Ditching can be selected in either AUTO or MANUAL pressurization modes. The DITCHING PBA closes all valves in preparation for a water landing. When the aircraft is below 15,000ft, pressing the guarded DITCHING PBA:

- Opens the outflow valve
- Closes the flow control valves
- Trim air pressure-regulating valve
- Trim air shutoff valve

Closes all electrically controlled valves located below the flotation line

- Avionics exhaust valves
- Ground valve

When pressing the DITCHING PBA, it latches and the ON legend illuminates white to indicate the electrically controlled valves have closed.

If the ram air valve is open, the RAM AIR PBA is pressed to close it. A DITCHING MISCONFIG caution message indicates the DITCHING PBA is selected ON, and the ram air valve is open.

When the cabin differential pressure has equalized, all valves are closed, and the airspeed is less than 80 kt, the outflow valve is driven closed.

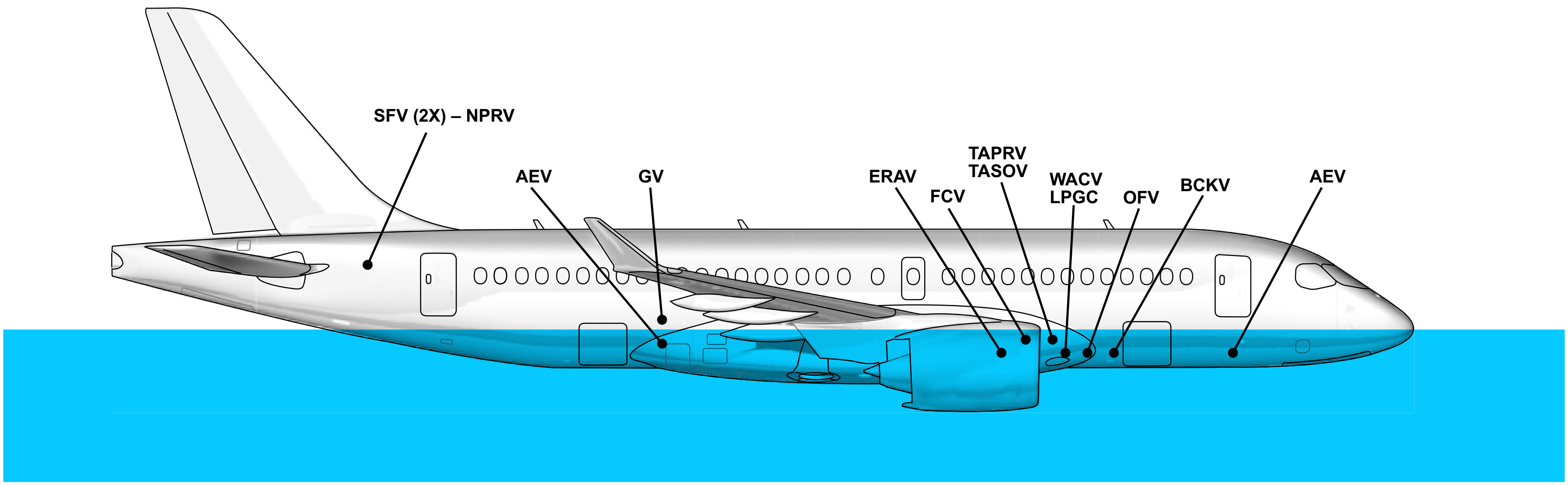
A DITCHING ON status message indicates the DITCHING PBA is pressed and the ditching sequence is complete.

The following valves prevent water intake mechanically

- Low-pressure ground connection
- Bulkhead check valves
- Water activated check valve

The following valves are above the flotation line, but prevent water intake:

- Safety valves
- Negative pressure-relief valve



LEGEND

AEV	Avionics Exhaust Valve
BCKV	Bulkhead Check Valve
ERAV	Emergency Ram Air Valve
FCV	Flow Control Valve
GV	Ground Valve
LPGC	Low-Pressure Ground Connection
NPRV	Negative Pressure-Relief Valve
SFV	Safety Valve
TAPRV	Trim Air Pressure-Regulating Valve
TASOV	Trim Air Shutoff Valve
WACV	Water Activated Check Valve

ELECTRICALLY CLOSED VALVES	MECHANICALLY CLOSED VALVES
Forward AEV	BCKV
OFV	WACV
GV	LPGC
Mid AEV	SFV
FCV, TAPRV, TASOV	NPRV
ERAV	

CS1_CS3_2130_016

Figure 105: Ditching

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages associated with the cabin pressure control system:

CAS MESSAGES

Table 30: WARNING Messages

MESSAGE	LOGIC
CABIN DIFF PRESS	Positive differential pressure is above pneumatic relief setting.
CABIN ALT	Cabin altitude exceeds 10,000 ft under non-HAAO conditions, and up to 15,000 ft under HAAO conditions posted by IASC A channels. Cabin altitude exceeds 15,000 ft under any condition posted by IASC B channels. Safety cabin altitude high alarm from IASC1C or IASC2C.

Table 31: CAUTION Messages

MESSAGE	LOGIC
CABIN ALT	No CABIN ALTITUDE warning message and cabin altitude exceeds 8,500 ft under non-HAAO conditions, and up to 15,000 ft under HAAO conditions. Cabin altitude exceeds 10,000 ft under HAAO conditions for more than 30 minutes.
AUTO PRESS FAIL	Both cabin pressure control AUTO functions are failed.
DITCHING MISCONFIG	DITCHING selected ON and RAM AIR selected OPEN.
LDG ELEV MISCONFIG	High airport altitude landing is selected and HAAO option is not present on aircraft.
EMER DEPRESS ON	EMER DEPRESS selected ON confirmed by either one auto mode or one manual mode.

Table 32: ADVISORY Messages

MESSAGE	LOGIC
CABIN ALT LEVEL HI	High airport altitude landing or takeoff sequence is initiated and the system operates in automatic mode. (increase of caution and warning threshold above normal setting).
PRESSURIZATION FAULT	Loss of redundant or non-critical function for the pressurization system.

Table 33: STATUS Messages

MESSAGE	LOGIC
DITCHING ON	DITCHING selected ON and sequence completed.
CABIN PRESS MAN	MANUAL PRESSURIZATION mode selected.

Table 34: INFO Messages

MESSAGE	LOGIC
21 PRESSURIZATION FAULT - PRIM ALT LIM INOP	Loss of IASC 1 altitude limitation function.
21 PRESSURIZATION FAULT - BACKUP ALT LIM INOP	Loss of IASC 2 altitude limitation function.
21 PRESSURIZATION FAULT - CPCS AUTO MODE REDUND LOSS	One automatic mode failed.

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Table 34: INFO Messages

MESSAGE	LOGIC
21 PRESSURIZATION FAULT - MANUAL MODE INOP	OFV manual mode failed.
21 PRESSURIZATION FAULT - OFV FINGER FAIL OUT	Altitude limiter failed deployed.
21 PRESSURIZATION FAULT - OFV FINGER FAIL IN	Altitude limiter failed in closed position.

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28-00 FUEL SAFETY

FUEL SAFETY HISTORY

Since 1959 there have been 17 fuel tank ignition events, resulting in 542 fatalities, 11 hull losses, and 3 others with substantial damage to the aircraft. Since the 1960's, there have been 5 key accidents involving fuel tank explosions which call into question the fundamental safety strategy applied to fuel systems of large commercial airplanes. Because of this accident, multiple airworthiness directives (ADs) on several aircraft types were issued.

25.981 FUEL TANK IGNITION PREVENTION

This advisory circular (AC) 25.981 fuel tank ignition prevention provides guidance for demonstrating compliance with the certification requirements for prevention of ignition sources within the fuel tanks of transport category airplanes.

The TWA 800 accident brought the realization that some fuel tanks could be flammable for much of their operational time. In their efforts to determine the cause of TWA 800 accident, the FAA concluded that many current airplanes had problems in their ignition prevention approaches. An additional independent layer of protection would be needed to back up the ignition prevention strategy.

SPECIAL FEDERAL AVIATION REGULATION 88

In response to these findings, the FAA issued Special Federal Aviation Regulation (SFAR) No. 88 in June of 2001. SFAR 88 required a safety review of the aircraft fuel tank system to determine that the design meets the requirements of FAR 25.901 and 25.981 (a) and (b).

The safety reviews were very valuable in that they revealed unexpected ignition sources. There was difficulty in identifying all ignition sources and a number of previously unknown failures were found.

A flammability reduction program would be necessary to prevent the operation of transport category airplanes with explosive fuel-air mixtures in the fuel tank. These fuel tanks were called high flammability tanks. The overall goal was to make sure flammability reduction of high flammability tanks became an integral part of aircraft safety.

EUROPEAN AVIATION SAFETY AGENCY REGULATIONS

A similar regulation had been recommended by the European Joint Aviation Authorities (JAA) to the European national aviation authorities in JAA letter 04/00/02/07/03-L024 of February 3, 2003. The review was mandated by European national airworthiness authorities using JAR 25.901(c), and 25.1309.

In August 2005, the European Aviation Safety Agency (EASA) published a policy statement on the process for developing instructions for maintenance and inspection of fuel tank system ignition source prevention that also included the EASA expectations with regard to compliance times of the corrective actions. The EASA policy statement reset the date for compliance of unsafe items to July 1, 2006.

Fuel airworthiness limitations are items from a systems safety analysis that have been shown to have failure modes associated with an unsafe condition as defined in SFAR 88. These are identified in failure conditions for which an unacceptable probability of ignition risk could exist if specific tasks and/or practices are not performed in accordance with the manufacturers requirements.

This EASA airworthiness directive mandated the fuel airworthiness limitation critical design configuration control limitations (CDCCLs) consisting of maintenance and inspection tasks for the type of aircraft, that resulted from the design reviews, JAA recommendations, and the EASA policy statement.

The FAA and EASA have also mandated that operators must train their maintenance and engineering personnel regarding the changes brought about by SFAR 88.

MAJOR ACCIDENTS RELATED TO FUEL TANK EXPLOSIONS			
Aircraft	Location	Year	Cause
Boeing 707	Elkton, Maryland	1963	Lightning
Boeing 747	Madrid, Spain	1976	Lightning
Boeing 737	Manila, Philippines	1990	Unconfirmed
Boeing 747	New York, New York	1996	Center fuel tank explosion. Exact cause undetermined.
Boeing 737	Bangkok, Thailand	2001	Center fuel tank explosion. Possible center fuel pump failure.

TWA 800 B747

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Figure 1: Fuel Safety History

AIRWORTHINESS LIMITATIONS-FUEL SYSTEM LIMITATIONS

Airworthiness limitation items (ALI) are mandatory maintenance items for the fuel system that can include CDCCLs, inspections, or other procedures necessary to make sure the fuel tank flammability exposure does not increase above the certification limits as a result of maintenance actions, repairs, or alterations throughout the operational life of the airplane.

The Airworthiness Limitations (AWL) gives the mandatory scheduled maintenance requirements applicable to the C Series aircraft. The fuel system limitation (FSL) information is found in the AWL section 7.

This section contains task data about the aircraft fuel system maintenance tasks generated by the fuel tank system safety and ignition prevention analysis accomplished in compliance with federal aviation administration (FAA) Special Federal Aviation Regulation (SFAR) No. 88 - fuel tank system fault tolerance evaluation requirements.

FSL tasks specified are mandatory airworthiness limitations and must be incorporated into the operators locally approved maintenance schedule.

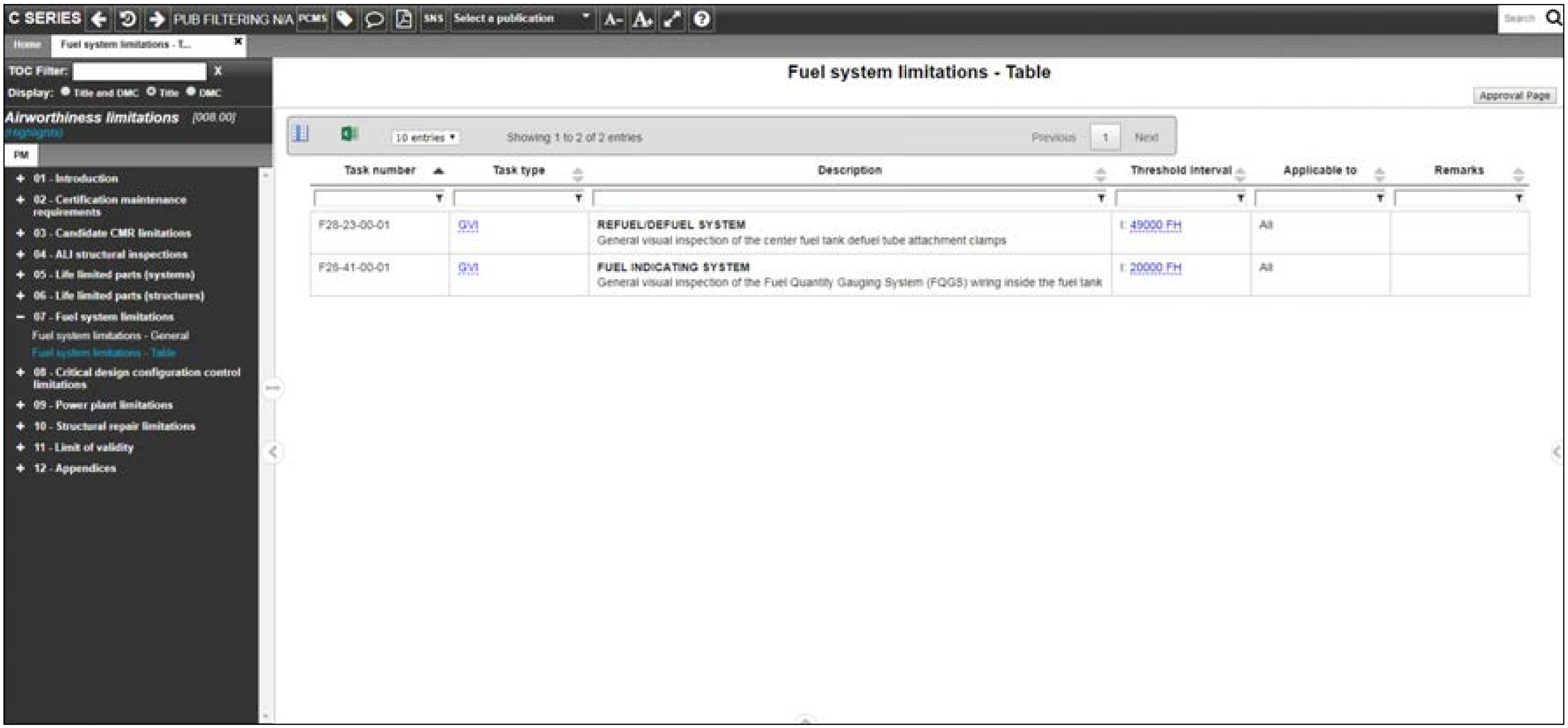

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Figure 2: Airworthiness Limitations-Fuel System Limitations

CRITICAL DESIGN CONFIGURATION CONTROL LIMITATIONS

Critical design configuration control limitations (CDCCLs), a type of airworthiness limitation, include those features of the design that must be present and maintained to achieve the safety level intended by FAR 25.981 for the operational life of the airplane.

The information is found in the airworthiness limitations (AWL) section 7.

This section contains information about the critical design configuration control limitations (CDCCLs) generated by the fuel tank system safety and ignition prevention analysis accomplished in compliance with FAA SFAR 88 - fuel tank system fault tolerance evaluation requirements.

The critical design configuration control limitations (CDCCLs) are established to preserve the critical ignition source prevention features of the fuel system design that are necessary to prevent the occurrence of an unsafe condition identified by the fuel tank safety assessment (FTSA) analysis.

The purpose of the CDCCLs is to provide instructions to aircraft maintenance personnel to retain the critical ignition source prevention features during the time of configuration change that may be caused by alterations, repairs or maintenance actions.

Identification of the CDCCLs design features, parts or components is provided within the applicable task procedures of the instructions for continued airworthiness (ICA), including the Aircraft Maintenance Publication (AMP).

The critical ignition source prevention features of the parts or components identified in the list of CDCCLs shown in Table 2 must be maintained in a configuration identical to the approved type design for the C Series aircraft.

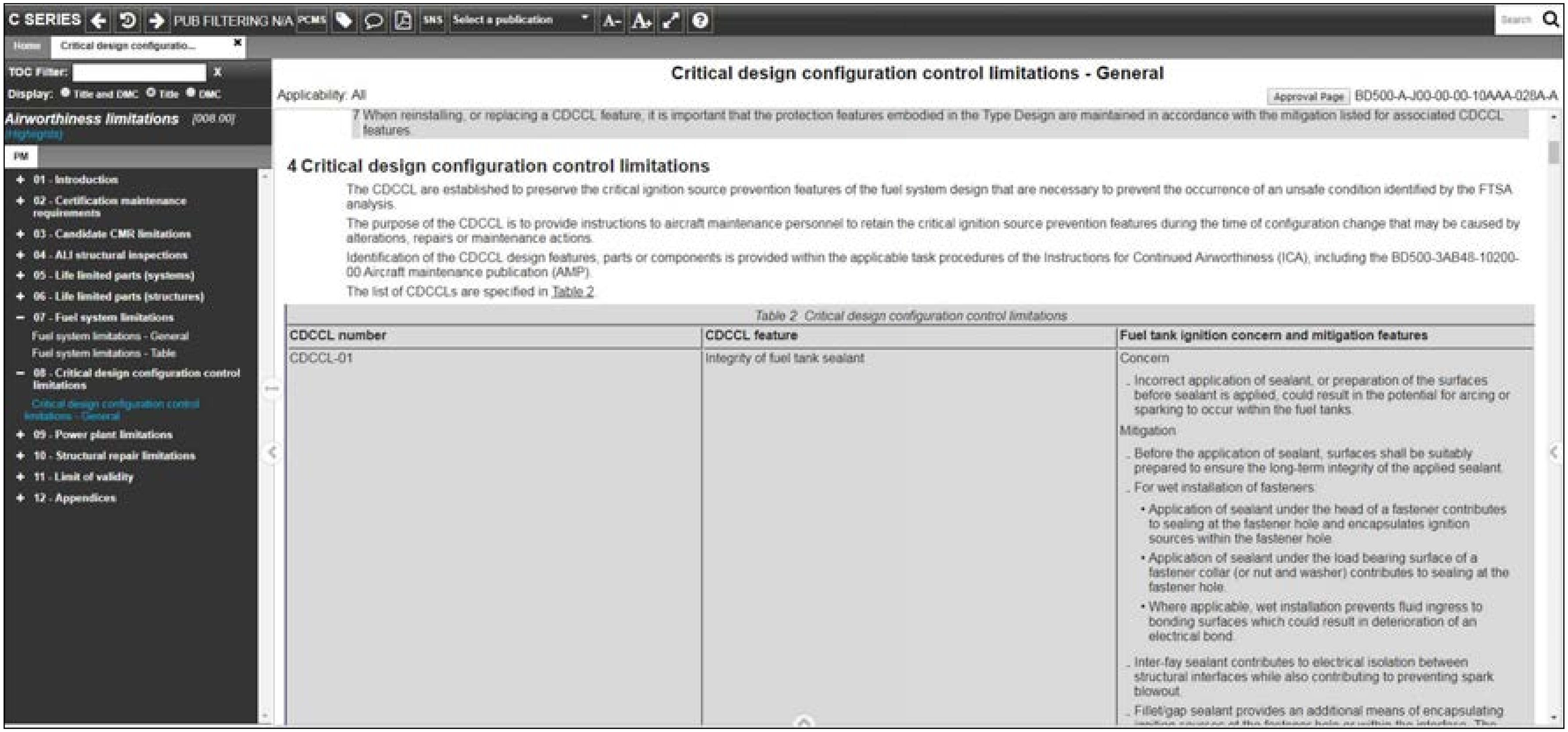

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Figure 3: Critical Design Configuration Control Limitation

CDCCL WARNING

Identification of the CDCCL design features, parts, or components is provided within the applicable task procedures of the instructions for continued airworthiness (ICA), including the Aircraft maintenance Publication (AMP).

The following warning is included when a task can impact CDCCL features.

WARNING

DO NOT DAMAGE OR MODIFY THE FUEL TANK PLUMBING LINES, SELF-BONDING COUPLINGS, CONDUCTIVE FITTINGS, AND METAL-TO-METAL INTERFACE ELECTRICAL BONDING OF THE FUEL TANK COMPONENTS. MAKE SURE THAT PROTECTIVE SEALANT IS RE-APPLIED AND/OR BONDING CHECK IS DONE AS INSTRUCTED. THESE DESIGN FEATURES ARE CLASSIFIED AS CRITICAL DESIGN CONFIGURATION CONTROL LIMITATION (CDCCL) ITEMS. THEY GIVE ELECTROSTATIC AND LIGHTNING PROTECTION. DAMAGE OR MODIFICATION TO THESE ITEMS CAN POSSIBLY CAUSE AN IGNITION SOURCE INTO THE FUEL TANK BECAUSE OF ELECTROSTATICS OR A LIGHTNING STRIKE.

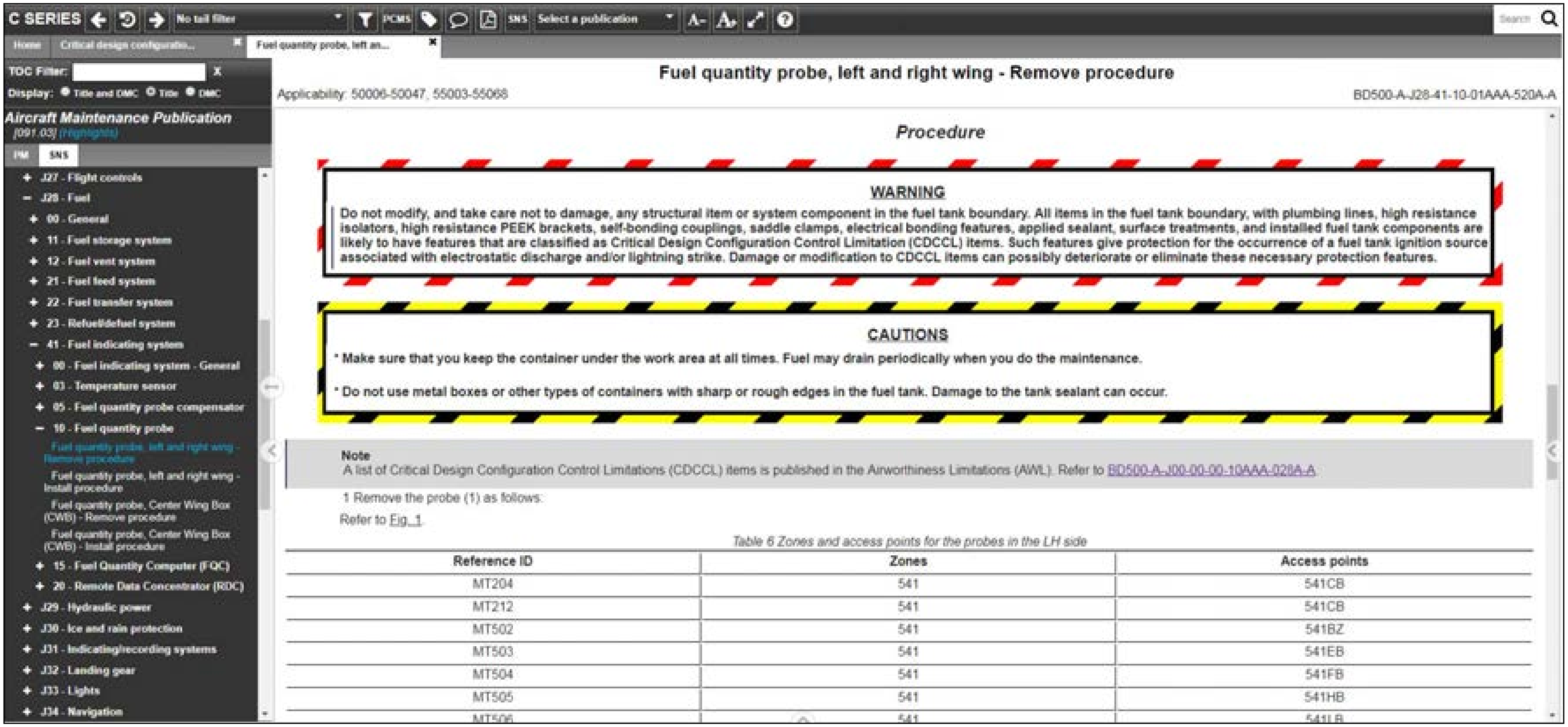

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Figure 4: Critical Design Configuration Control Limitation Warning in Aircraft Maintenance Publication

FUEL TANK ACCESS AND VENTING

FUEL TANK ACCESS

The following is a summary of steps to prepare the fuel tank for access:

1. Disconnect the aircraft batteries.
2. Disconnect external power and put warning placards in the flight deck ‘Do not apply electrical power to the aircraft’.
3. Establish a safety zone around the aircraft and install warning signs ‘Open fuel tank around the aircraft’.
4. After the tanks are defueled, drain the remaining fuel through the water drain valves.
5. Remove the fuel fumes from the fuel tank using the fuel tank venting kit.
6. After 30 minutes, monitor the fuel tank atmosphere using a gas detector.
7. Do not enter the fuel tank until the fuel fume concentration is less than the lower explosion limit and the oxygen level is at least 19.5%.
8. Continue to vent the fuel tank when work is being done inside the fuel tank.
9. When working in the fuel tank, measure the fuel fume concentration in and around the fuel tank every 30 minutes.
10. Only personnel trained to enter a fuel tank are allowed to enter the fuel tank.
11. Remove any residual fuel using cotton cloths or sponges.

FUEL TANK VENTING

The fuel tank venting kit has the necessary equipment to ventilate the fuel tank including air hoses, air movers, antistatic ducts, couplings, and wing adapters. Complete instructions are available with the kit.

The following is a summary of steps to remove the fuel fumes from the fuel tanks using the fuel tank venting kit:

1. Put an air compressor more than 30 m (100 ft) from the aircraft.
2. Connect the air compressor to the same ground point to which the aircraft is connected.
3. Install the inlet wing adapter at the outboard access panel opening.
4. Connect air hose from the air compressor to the inlet wing adapter.
5. Install the outlet wing adapter at the inboard access panel opening.
6. Connect the exhaust hose to the outlet wing adapter and to a ground point. Put the exhaust hose outlet so it discharges outside of the hangar.

WARNING

1. **MAKE SURE NO PERSON ENTERS THE FUEL TANK IF THE FUME CONCENTRATION IS MORE THAN THE LOWER EXPLOSION LIMIT.**
2. **MAKE SURE THAT ALL EQUIPMENT IS CLEAR OF THE EXHAUST DUCT. STATIC ELECTRICITY GENERATED BY AIRFLOW FROM THE EXHAUST DUCT CAN CAUSE AN EXPLOSION.**
3. **OBEY ALL LOCAL REGULATIONS WHEN DISCARDING FUEL.**

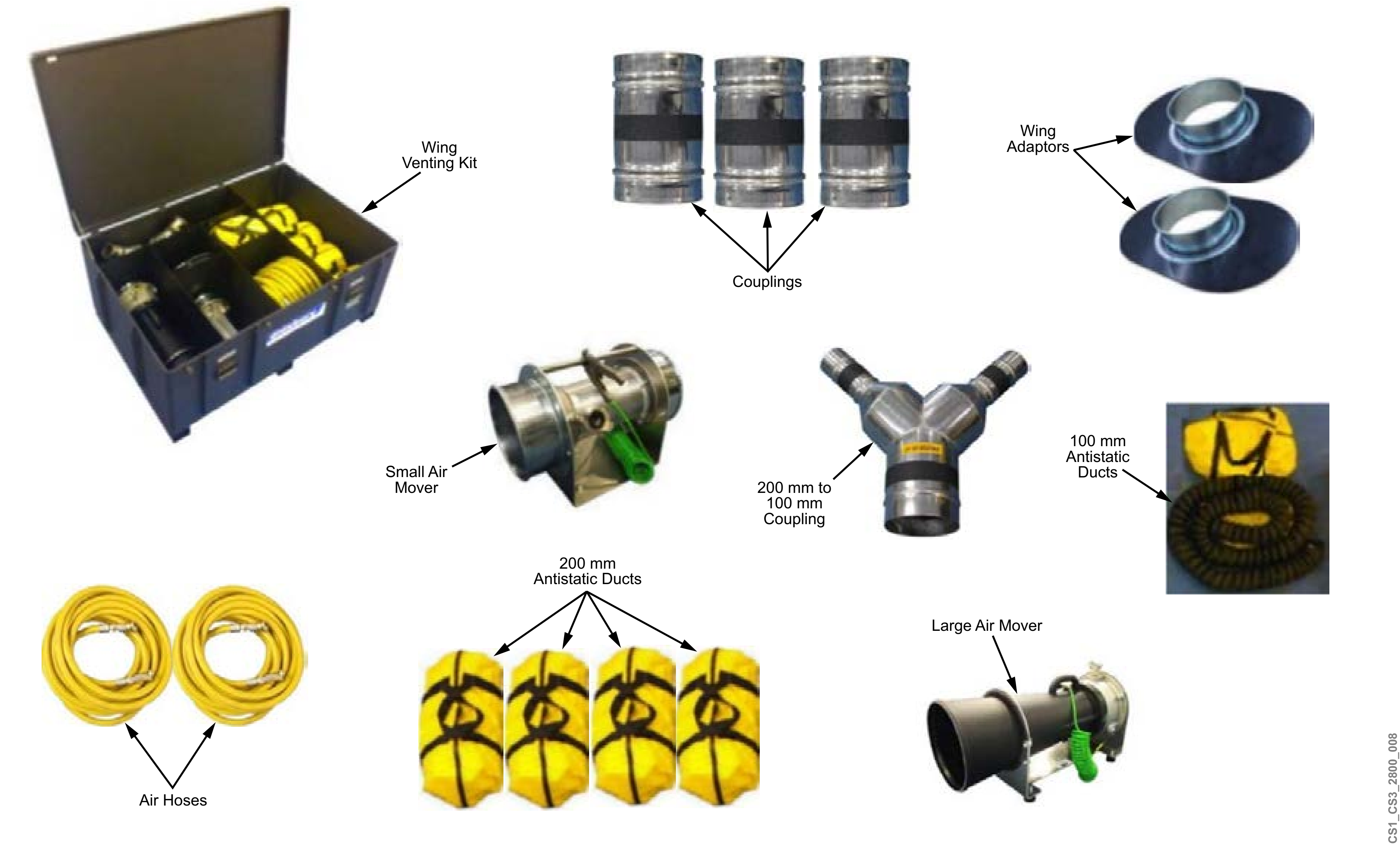


Figure 5: Fuel Tank Access and Venting

FUEL TANK ENTRY

PREPARATION

The following is a summary of steps to prepare for fuel tank entry:

1. Obey all local safety regulations during fuel system maintenance.
2. Firefighting extinguishers must be positioned near the aircraft.
3. Connect the grounding cable to an approved ground and to an aircraft ground point.
4. Ground the maintenance stands and equipment to the same ground point the aircraft is connected to.
5. Do not connect or disconnect electrical equipment to or from energized outlets that are less than 30.5 m (100 ft) from an open fuel tank .
6. Use explosion-proof floodlights and extension cords for external lighting. Use explosion-proof flashlights or explosion-proof lights inside the fuel tank. Explosion-proof lights must be connected to the electrical power before being put it in the fuel tank.

NOTE

Only personnel with fuel tank entry training should enter a fuel tank. The local authorities may require the personnel to have a tank entry permit

ENTRY

Before entering a fuel tank, check that the breathing apparatus kit is in good working condition. The breathing apparatus kit has an instruction manual, pressure regulator and valve, air filter, air hoses, and the necessary accessories to supply air to a full or half face mask.

The following is a summary of steps to enter the fuel tank:

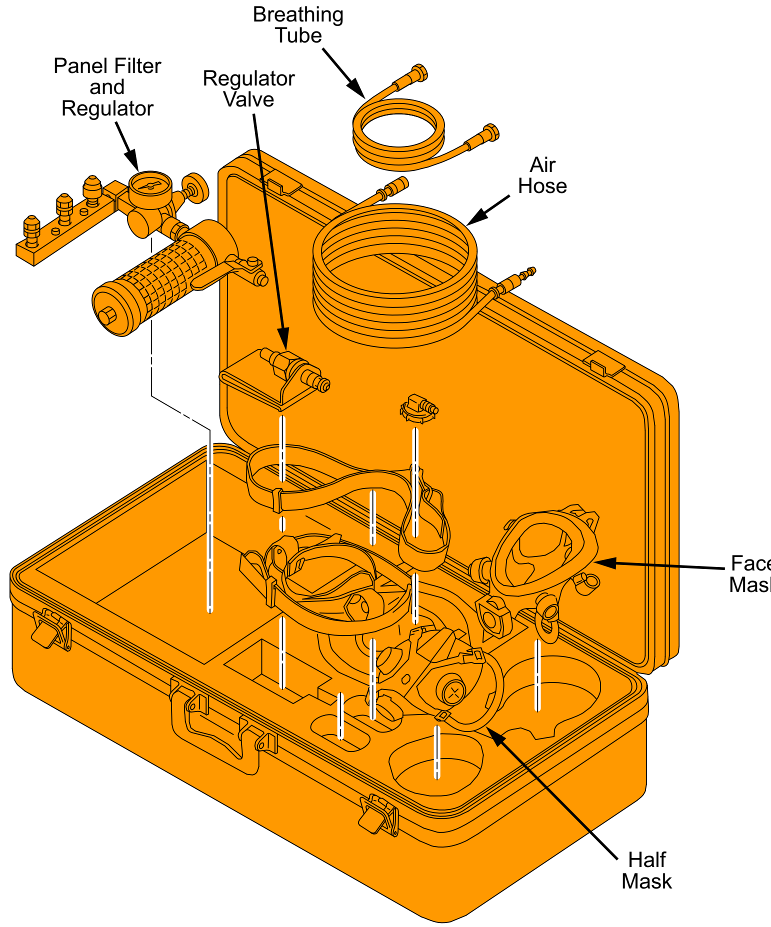
1. Wear clean lint-free cotton coveralls with buttons or zippers that do not cause sparks in the fuel tanks. Do not use wool, silk, nylon, or other synthetic clothing. When entering the fuel tank, use cotton hair and shoe covers as well.
2. When entering a fuel tank, an observer must be stationed near the fuel tank opening. The observer must remain in contact with personnel inside the tank and monitor their condition.
3. Metal boxes or containers with sharp or rough edges can cause damage to the fuel tank. Use containers with rounded corners that do not generate static electricity. Sharp-edged tools must be put it in a container when not in use.
4. Use only sponges or cotton rags to absorb fuel. Other materials can cause a static discharge
5. Leave the fuel tank immediately if the fuel concentration exceeds the lower explosion limit or is below the minimum oxygen concentration level.
6. Keep a list of all the tools, equipment, and materials entering the fuel tank. All tools, equipment, and material must be accounted for before the tank is closed.
7. Make sure the fuel tank is thoroughly inspected for cleanliness before closing the fuel tank.

WARNING

DO NOT USE STEEL WOOL OF ANY TYPE IN THE FUEL TANKS. PIECES OF STEEL WOOL CAN BREAK OFF AND CAUSE DAMAGE OR AN EXPLOSION IF THEY TOUCH ELECTRICAL COMPONENTS.

FIRE EXTINGUISHER

COTTON COVERALLS



BREATHING APPARATUS KIT

EXPLOSION PROOF LIGHTS

COTTON RAGS

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Figure 6: Fuel Tank Entry

28-11 FUEL STORAGE

GENERAL DESCRIPTION

The aircraft fuel is stored in the integral center, right wing, and left wing tanks located within the wing box structure. The tanks are isolated by a common boundary formed by structural ribs, and have independent provisions for fuel gauging and refueling. Baffle ribs are installed in the wing to restrict fuel migration outboard during aircraft maneuvers. The wings and the center wing box are all constructed using carbon fiber reinforced polymer (CFRP), with the exception of the wing ribs, which are metallic.

Each wing tank has a main and a collector tank. The collector tank is partially sealed and is located inboard of the wing tank. A dry bay is located above each engine strut in the wing tanks. The dry bays prevent fuel from spilling onto an engine if a wing tank is ruptured.

Access panels, located on the underside of the wing, provide access for the inspection, repair, and replacement of components inside the wing tanks.

All ribs, including the baffle ribs, incorporate drain passages to prevent the accumulation of water and trapped fuel. Each rib has openings at the upper intersection with the wing inner-skin liner, which facilitates airflow within the tank. The interior of the tanks is chemically treated against corrosion and coated with a biocide compound. All joints and rivet areas are sealed to ensure tank seal integrity.

Flapper check valves allow fuel to gravity feed from the main tanks to the collector tank to ensure that fuel is available at the fuel pumps in all aircraft attitudes. They also prevent the flow of fuel from the collector tank to the main tank during wing down and uncoordinated aircraft maneuvers. The check valve is a rubberized fabric hinged flapper.

One flapper check valve allows fuel to drain from the surge tank back to the main tank.

Two fuel drain valves are installed in each of the fuel tanks at low points to provide water drainage.

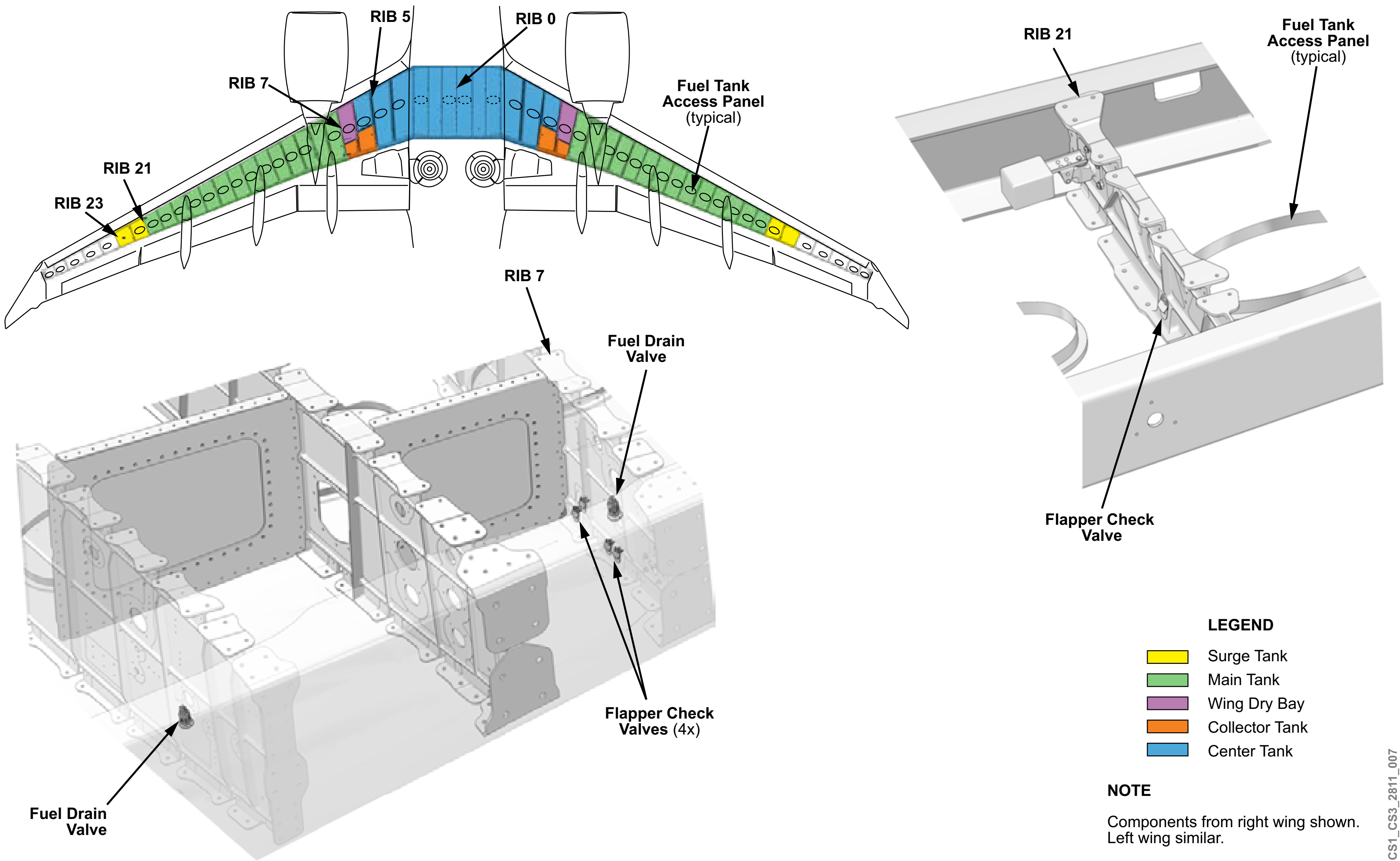


Figure 7: Fuel Storage System

COMPONENT LOCATION

The following components are installed in the fuel storage system:

- Flapper check valves
- Fuel drain valves
- Fuel tank access panels

FLAPPER CHECK VALVES

Four flapper check valves are located on RIB 7, between the main and collector tanks on each wing.

One flapper check valve is located on RIB 21, between the main tank and surge tank on each wing.

FUEL DRAIN VALVES

Fuel drain valves are installed at low points in the center, main, and collector tanks. The fuel drain valves are installed between the RIB 1 and RIB 2 of the left and right side of the center wing box. There is one fuel drain valve between RIB 5 and RIB 6, and another between RIB 7 and RIB 8 on each wing.

FUEL TANK ACCESS PANELS

Fuel tank access panels are installed on the underside of the wings and center tank, between each rib from RIB 0 to RIB 23.

WARNING

BE CAREFUL WHEN YOU REMOVE THE ACCESS PANEL. FUEL CAN BE ON THE INNER SURFACE OF THE PANEL. FUEL CAN CAUSE INJURY TO PERSONS AND/OR DAMAGE TO EQUIPMENT.

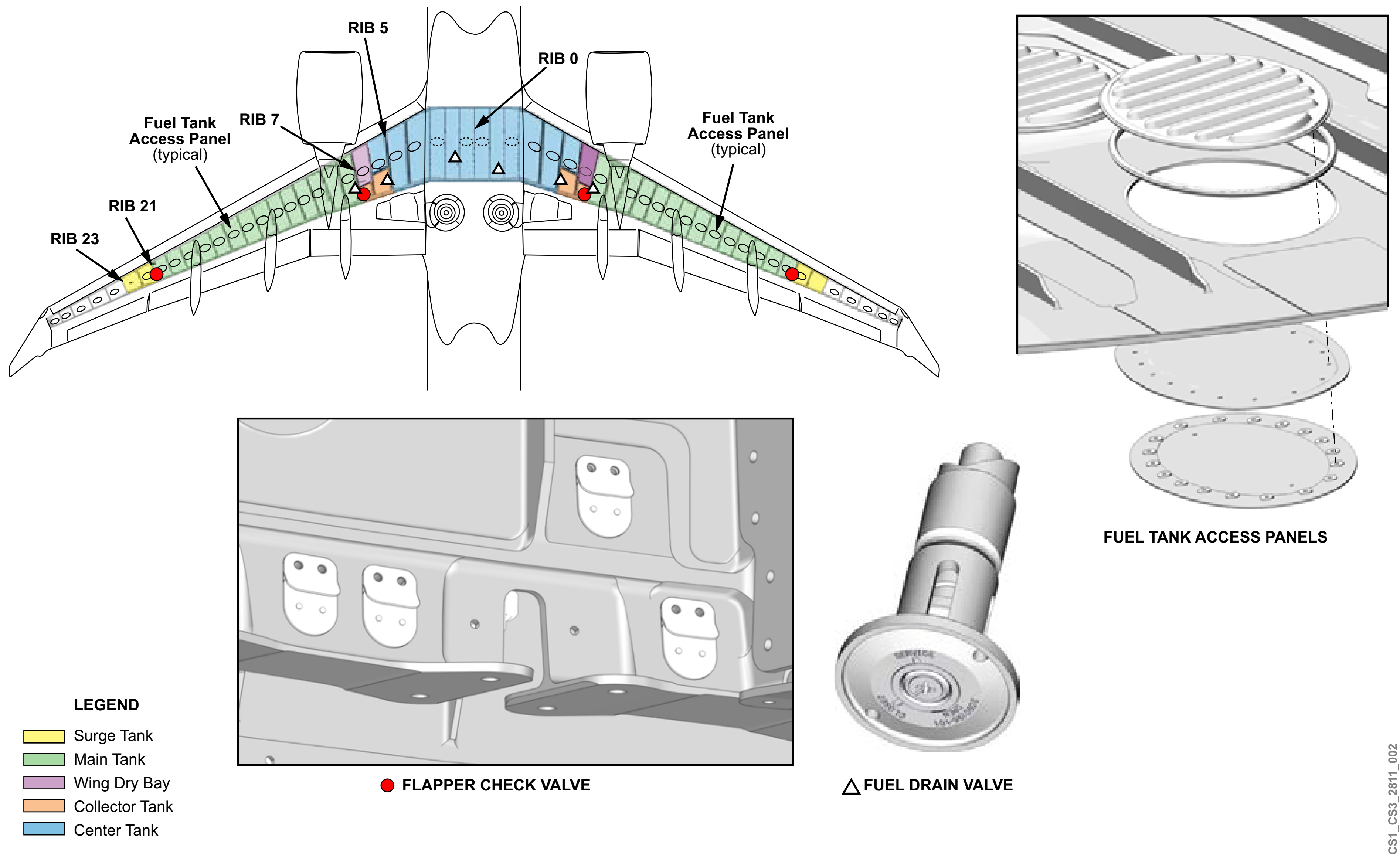


Figure 8: Fuel Storage System Component Location

PRACTICAL ASPECTS

FUEL DRAIN VALVE SERVICING

The fuel drain valves are used for:

- Removing water and contaminants
- Fuel sampling
- Draining the remaining fuel after defueling a tank

The fuel drain valve is a simple poppet operated type device. O-ring seals on the valves may be replaced without removing the valve from the tank. The valve has a push-to-drain operation and can be locked in the open position using a standard screwdriver.

NOTE

The fuel drain valve body is plastic. Use caution when attempting to operate in cold weather.

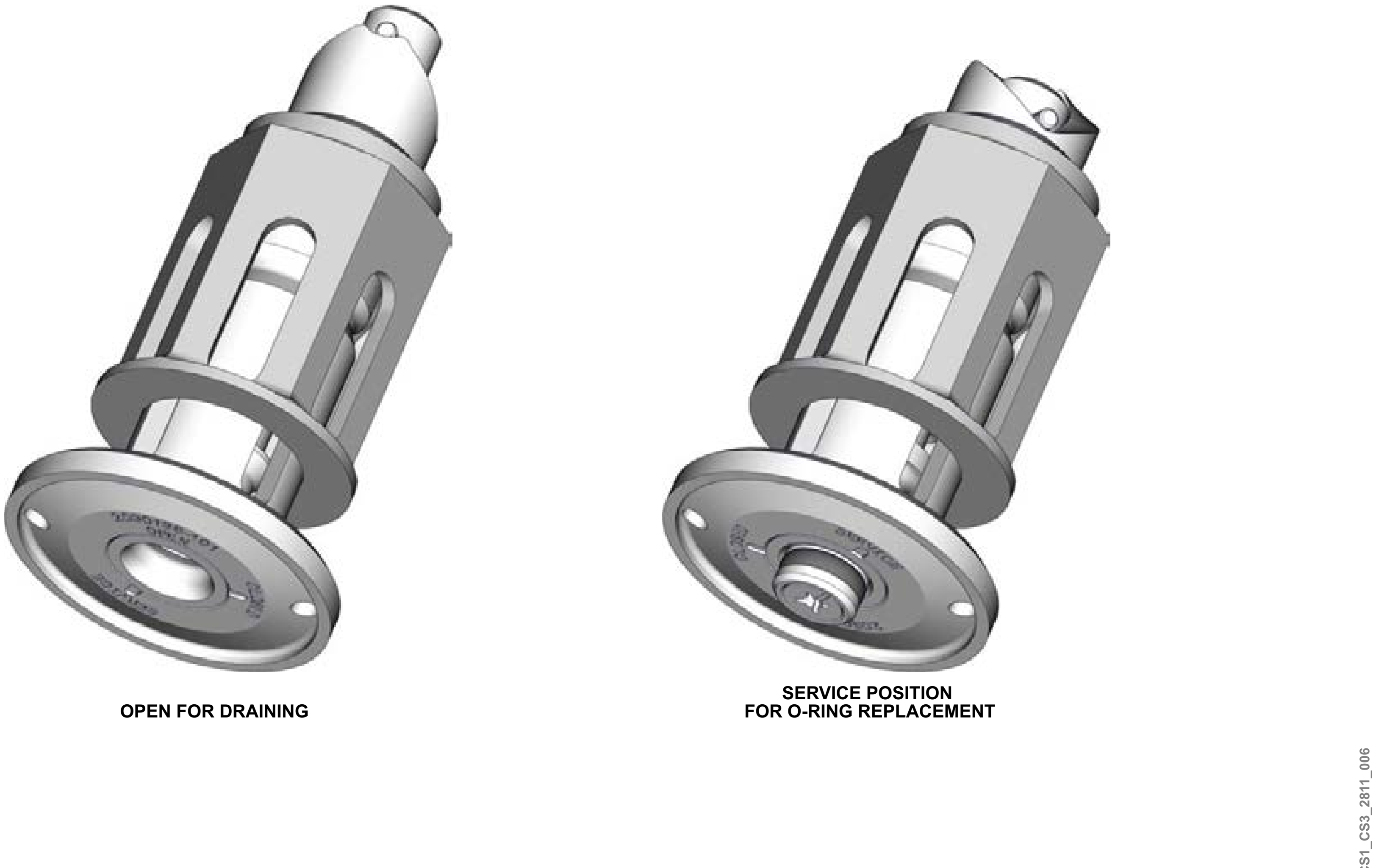


Figure 9: Fuel Drain Valve Servicing

28-12 VENT SYSTEM

GENERAL DESCRIPTION

The fuel vent system maintains the pressure of the fuel tanks near the pressure of the outside atmosphere to prevent damage to the wing structure. The surge tanks are open to outside air through the vent scoops. During flight, the vent scoops maintain positive pressure inside the surge tanks.

Each fuel tank is connected to a wing surge tank through a vent tube. The vent tubes have bifurcated openings at the two ends to provide added protection against blockage. The right wing tank vents through the right surge tank, and the left wing tank and the center tank vent through the left surge tank. The vent tubes keep the pressure of all the fuel tanks near the pressure in the surge tanks.

Float vent valves and drains ensure the vent tubes are not blocked by fuel that might enter the lines. They also drain any fuel that may have accumulated in the vent tubes.

The float drain valves close when the fuel level reaches the vent tubes. The center tank float drain valves also provide additional venting redundancy for the center tank.

Center tank float vent valves are installed in the center tank to ensure venting in case the vent tube openings are submerged in fuel during aircraft maneuvers. The main tank float vent valve remains above the fuel level in any ground condition due to the aircraft wing dihedral. Any fuel entering the surge tank through the float vent valve drains back to the main tank through the surge tank flapper check valve.

The flow of air between the surge tanks and the atmosphere is normally through the NACA inlet scoop and flame arrestor. The inlet scoop creates a small pressure within the surge tank during flight. The inlet scoop also provides a path for fuel to drain overboard should the surge tank overflow with fuel. The flame arrestor ensures that a flame does not enter the fuel tanks through the vent system. In case of a blockage, the flame arrestor has a built-in negative pressure-relief valve to provide backup pressure equalization.

In normal operation, air flows between the main wing tanks and surge tanks through the bifurcated vent openings or through the float vent valves, depending on the aircraft attitude and the level of fuel in the tank. For the center tank vent, air flows through the vent tubes, either by the bifurcated opening or through one of the float vent valves installed in the vent tube.

Main wing tank pressure-relief valves allow pressure relief to the atmosphere. Center tank positive pressure is relieved into the right wing tank and negative pressure is relieved into the right surge tank. Surge tank flapper valves also provide redundant negative pressure relief to the main tanks.

If the flow path is submerged by fuel and the alternate path is blocked, the internal pressure of the tank will cause fuel to be spilled into the surge tank. The fuel either drains back through the flapper valves or spills into the atmosphere through the NACA inlet scoop and flame arrestor. In case of a complete blockage in the normal flow path, the appropriate pressure-relief valve (PRV) will provide backup pressure equalization.

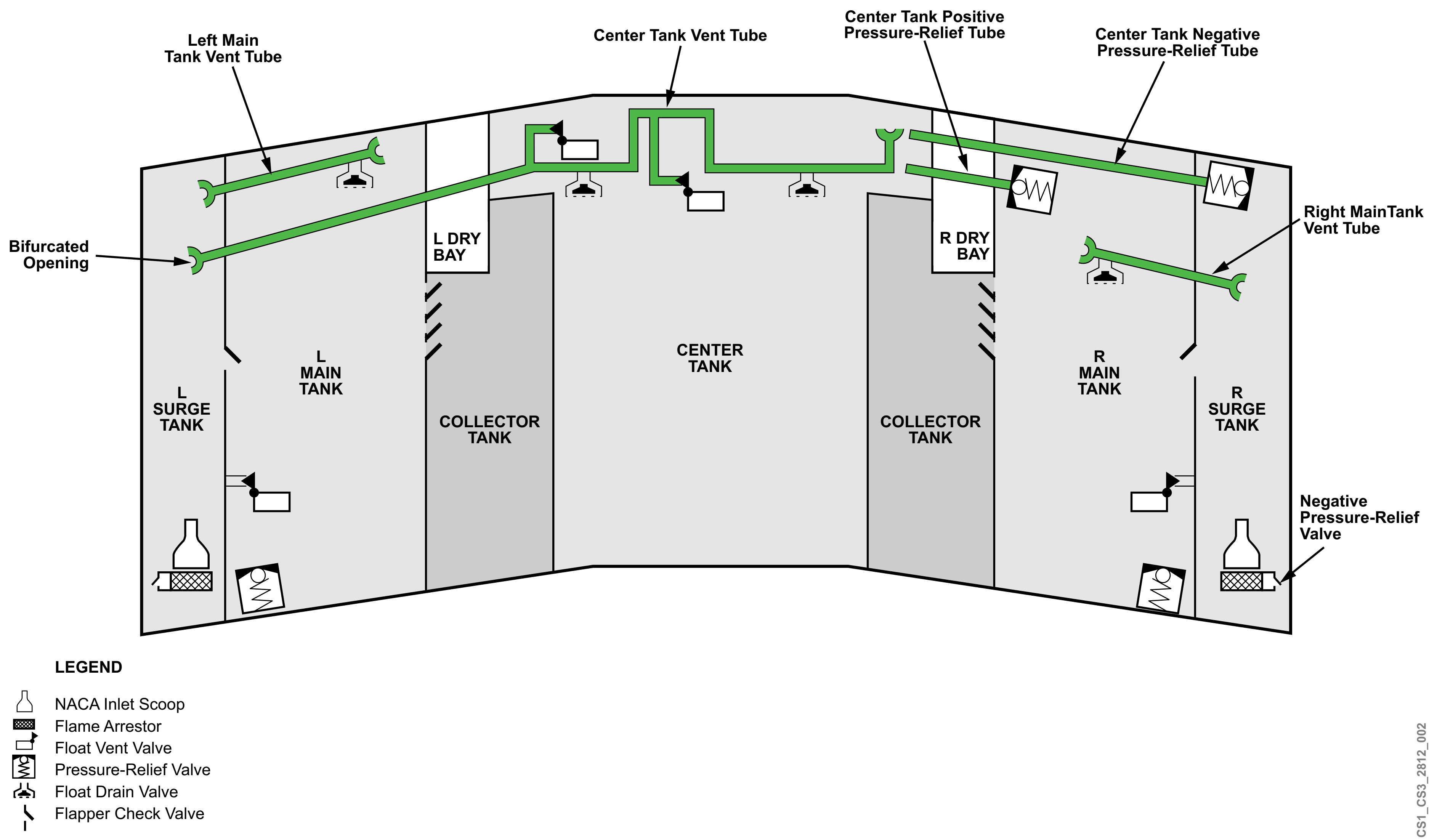


Figure 10: Fuel Vent System

COMPONENT LOCATION

The following components are installed in the fuel vent system:

- Float drain valves
- Center tank float vent valves
- Wing tank float vent valves (refer to figure 12)
- Bifurcated vent openings (refer to figure 12)
- NACA inlet scoop (refer to figure 13)
- Vent relief flame arrestor (refer to figure 13)
- Positive pressure-relief valves (refer to figure 14)
- Center tank negative pressure-relief valve (refer to figure 14)

FLOAT DRAIN VALVES

Float drain valves are installed along the vent tubes in the center and wing tanks at RIB 2 and RIB 9.

CENTER TANK FLOAT VENT VALVES

Two float vent valves are installed in the center tank between RIB 0 and RIB 1 and at RIB 6 of the left wing.

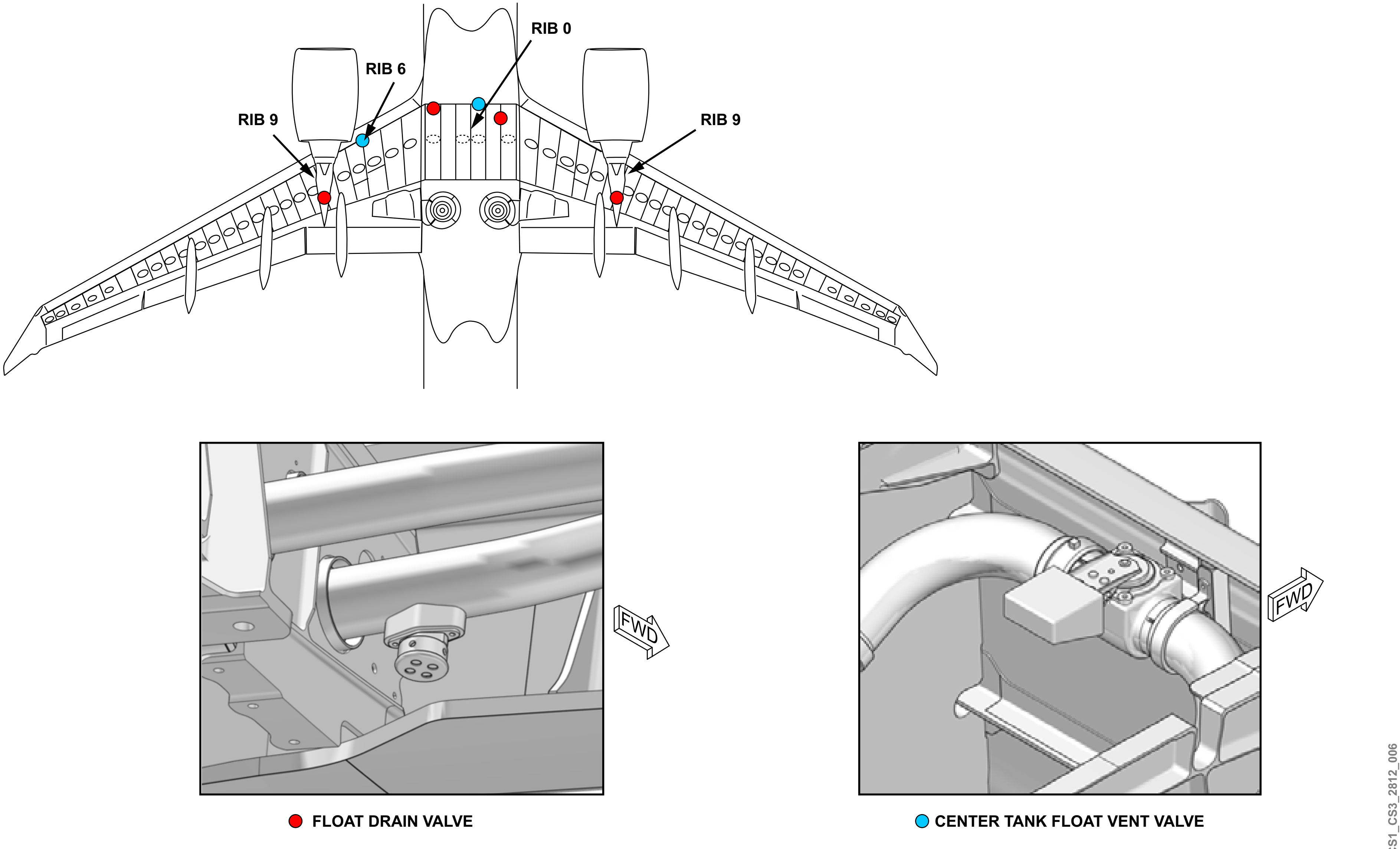


Figure 11: Float Drain Valve and Center Tank Float Vent Valve

WING TANK FLOAT VENT VALVES

Wing tank float vent valves, installed near the top of RIB 21, vent each main tank into the surge tank.

BIFURCATED VENT OPENINGS

The bifurcated vent openings are located in the surge tanks at RIB 22, RIB 8, and at RIB 6R in the right wing only.

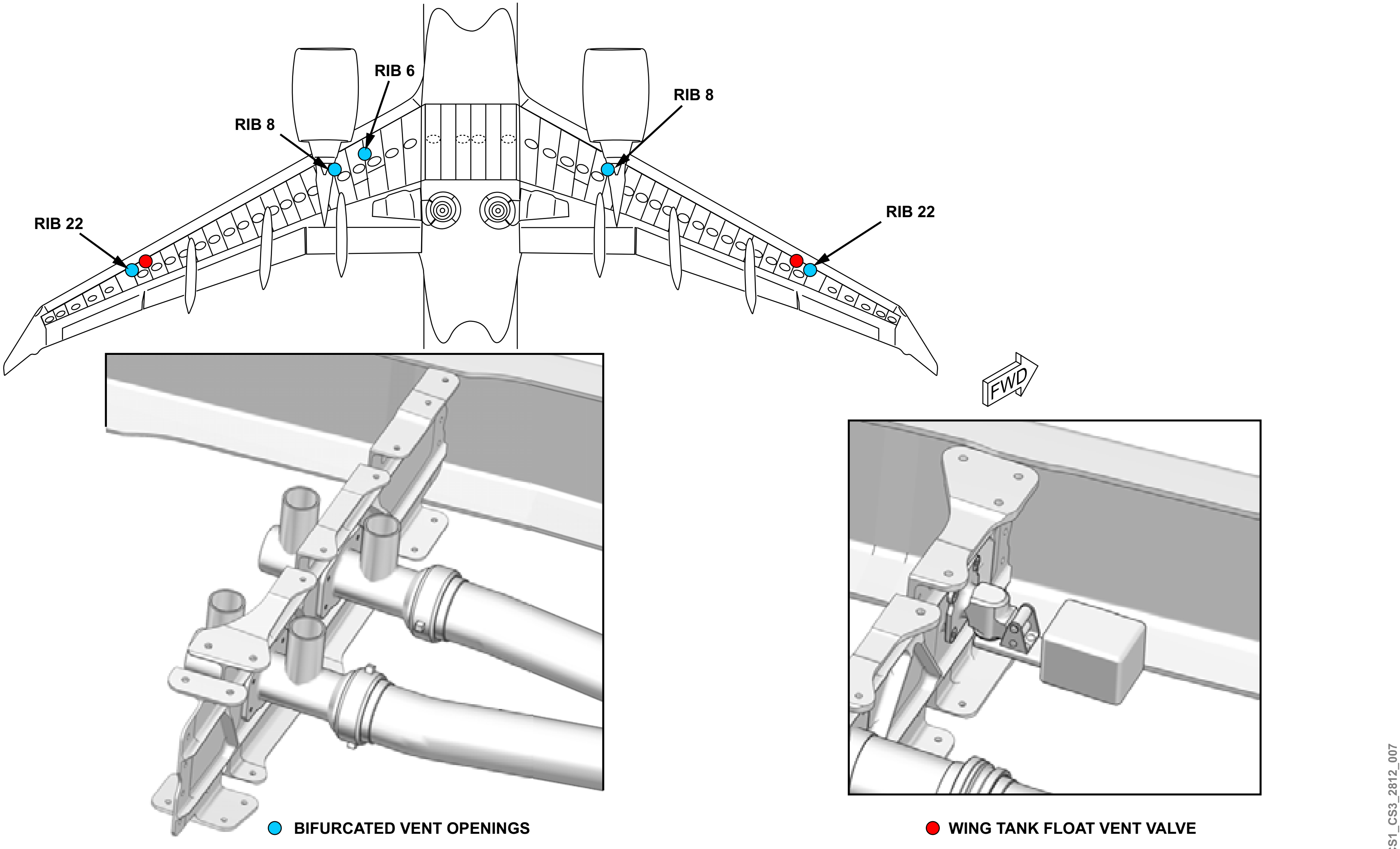


Figure 12: Bifurcated Vent Openings and Wing Tank Float Vent Valve

NACA INLET SCOOP

A NACA inlet scoop is installed in each surge tank outboard of RIB 22.

VENT RELIEF FLAME ARRESTOR

The vent relief flame arrestor is installed in the left and the right wings between RIB 22 and RIB 23.

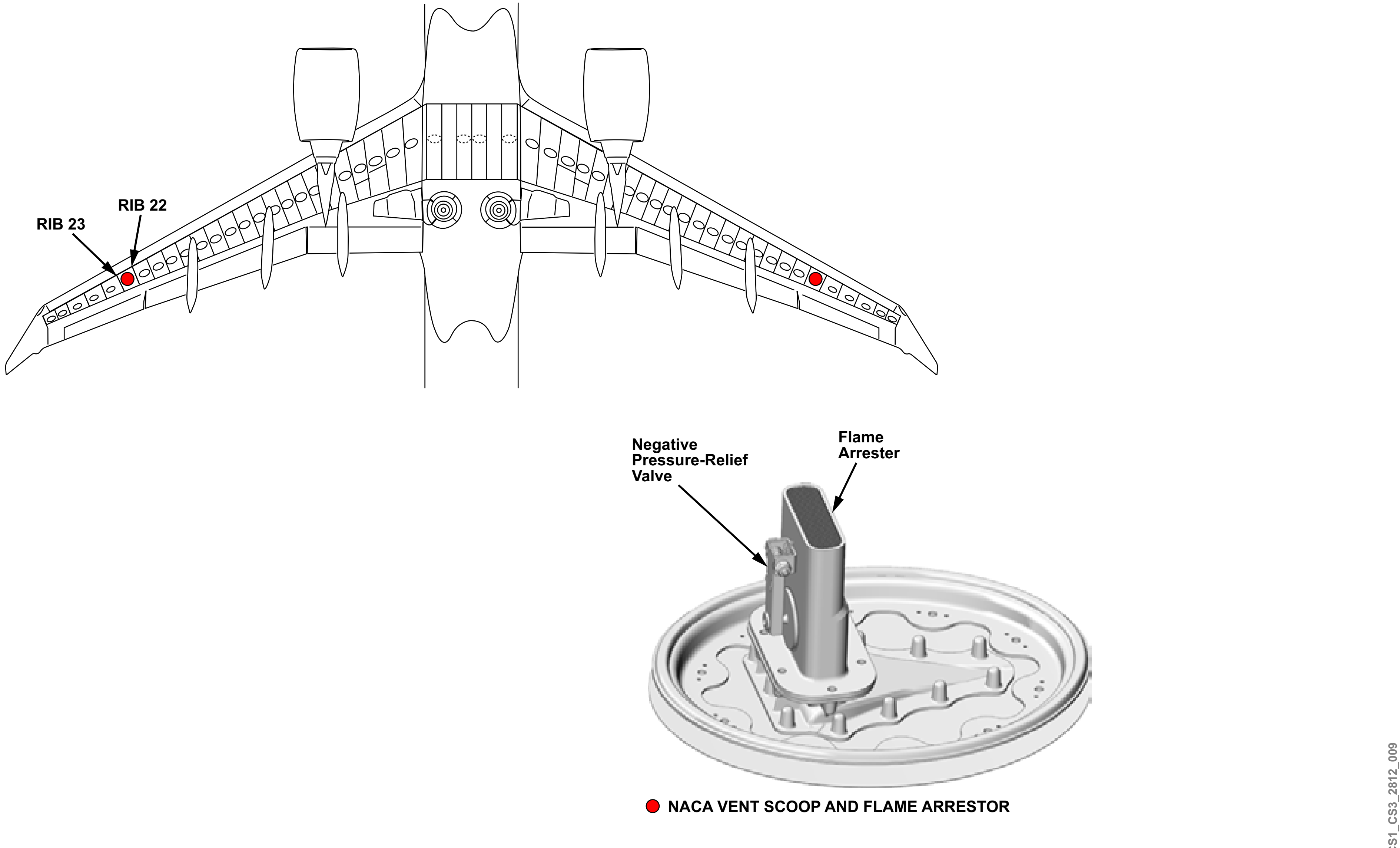


Figure 13: NACA Inlet Scoop and Vent Relief Flame Arrestor

POSITIVE PRESSURE-RELIEF VALVES

Positive pressure-relief valves are installed at the access panel on the left and the right wings between RIB 20 and RIB 21.

CENTER TANK POSITIVE PRESSURE-RELIEF VALVE

The positive pressure-relief valve for the center tank is installed on RIB 7 in the right wing tank.

CENTER TANK NEGATIVE PRESSURE-RELIEF VALVE

The center tank negative pressure-relief valve is installed on the right wing at RIB 21.

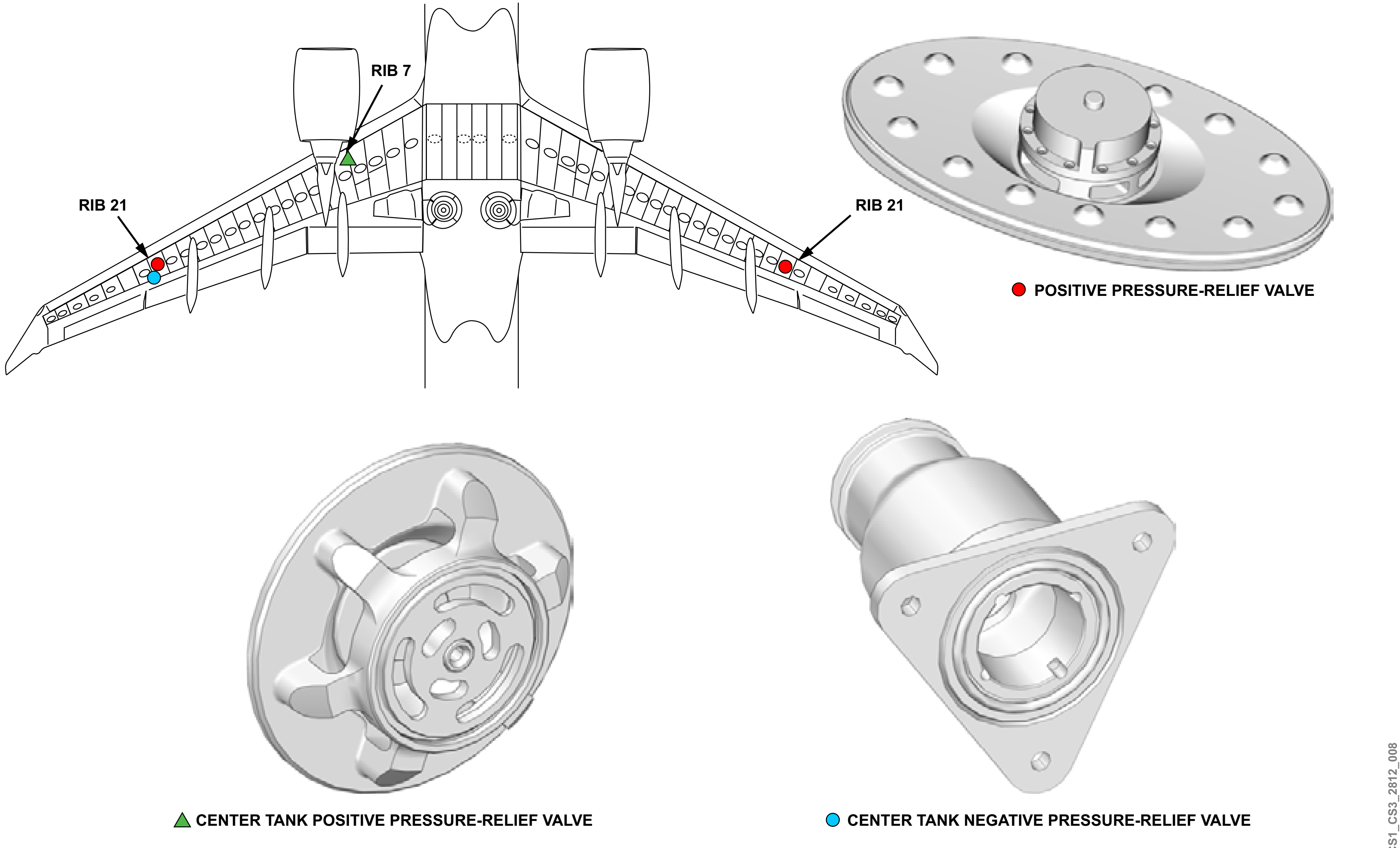


Figure 14: Positive Pressure-Relief Valves, Center Tank Positive, and Center Tank Negative Pressure-Relief Valves

28-23 REFUEL/DEFUEL SYSTEM

GENERAL DESCRIPTION

Aircraft refueling and defueling is carried out using a single-point refuel/defuel adapter, located at the right wing. The refuel adapter connects to the refuel manifold that runs near the rear spar of the fuel tanks. The refuel/defuel tube connects the refuel adapter to all of the tanks via the refuel shutoff valves.

Each tank has one refuel shutoff valve and they connect the refuel/defuel tube to the tanks. The refuel control solenoid valve controls the opening and closing of the refuel shutoff valve. The refuel control solenoid valve is controlled by the fuel quantity computer (FQC). When the solenoid is energized, fuel pressure is directed to open the refuel shutoff valve. When the refuel shutoff valve opens, fuel discharges into the tank through a diffuser.

Refueling and defueling is controlled from the refuel/defuel control panel (RDCP) located on the fuselage at right wing root. Pressure refueling of the aircraft is carried out in manual or automatic mode, through the appropriate AUTO/MANUAL switch selection on the RDCP. The recommended fuel pressure for refueling is 50 psig.

The FQC receives power from DC ESS BUS 1 and DC ESS BUS 2 during normal operation. The refuel shutoff valves receive power from DC ESS BUS 1 during normal operation. When the RDCP door panel is opened, the power for the FQC and the refuel shutoff valves is switched to the DC EMER BUS. The RDCP is powered by the FQC through the door switch. A second contact on the door switch provides a signal that the RDCP door is open.

The aircraft is defueled by connecting the refueling manifold to the engine feed manifold. The defuel/isolation transfer shutoff valve connects the manifolds. The valve is powered and controlled by the FQC. When DEFUEL is selected on the RDCP, the valve opens and DEFUEL is displayed in the PRESEL window.

The refuel adapter must be set to the defuel position before connecting the fuel truck hose. Defueling the aircraft can only be done in MANUAL mode.

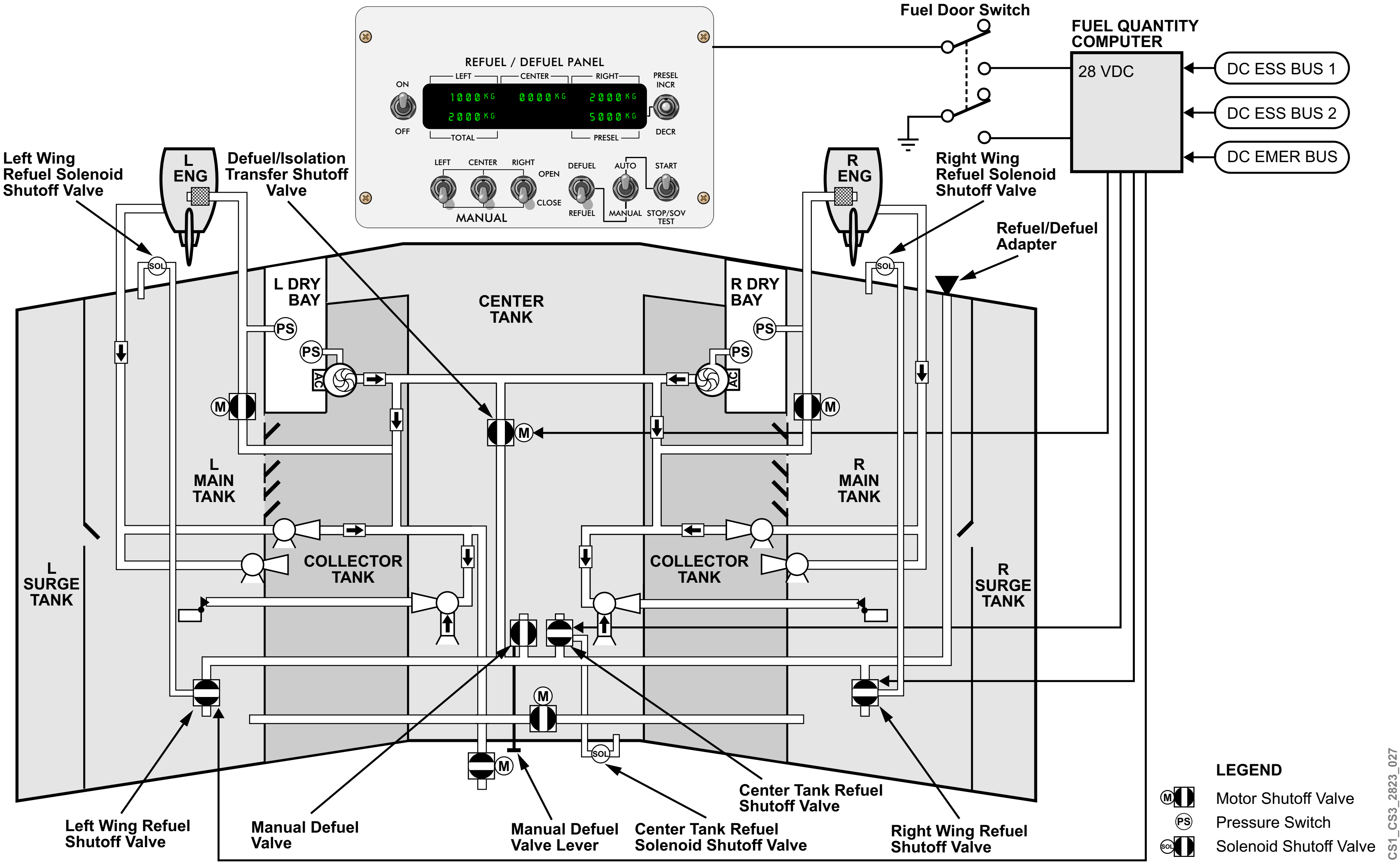


Figure 15: Refuel/Defuel System

REFUELING

Auto Refueling

With the AUTO/MANUAL switch in AUTO, the fuel quantity to be delivered is preselected on the RDCP using the PRESEL INCR/DECR switch setting.

The REFUEL START/STOP/SOV TEST switch has a momentary STOP/TEST SOV, a START position, and a center OFF position. The START position is used for automatic refueling. When refueling is in progress in AUTO MODE, selecting STOP/SOV TEST stops refueling.

The refueling operation starts on selection of the REFUEL/DEFUEL switch to REFUEL, and the START/STOP/SOV TEST switch to the START position. The refueling operation only stops when the preselected fuel quantity is delivered, or the operation is stopped by selecting the START/STOP/SOV TEST switch to the STOP/SOV TEST position.

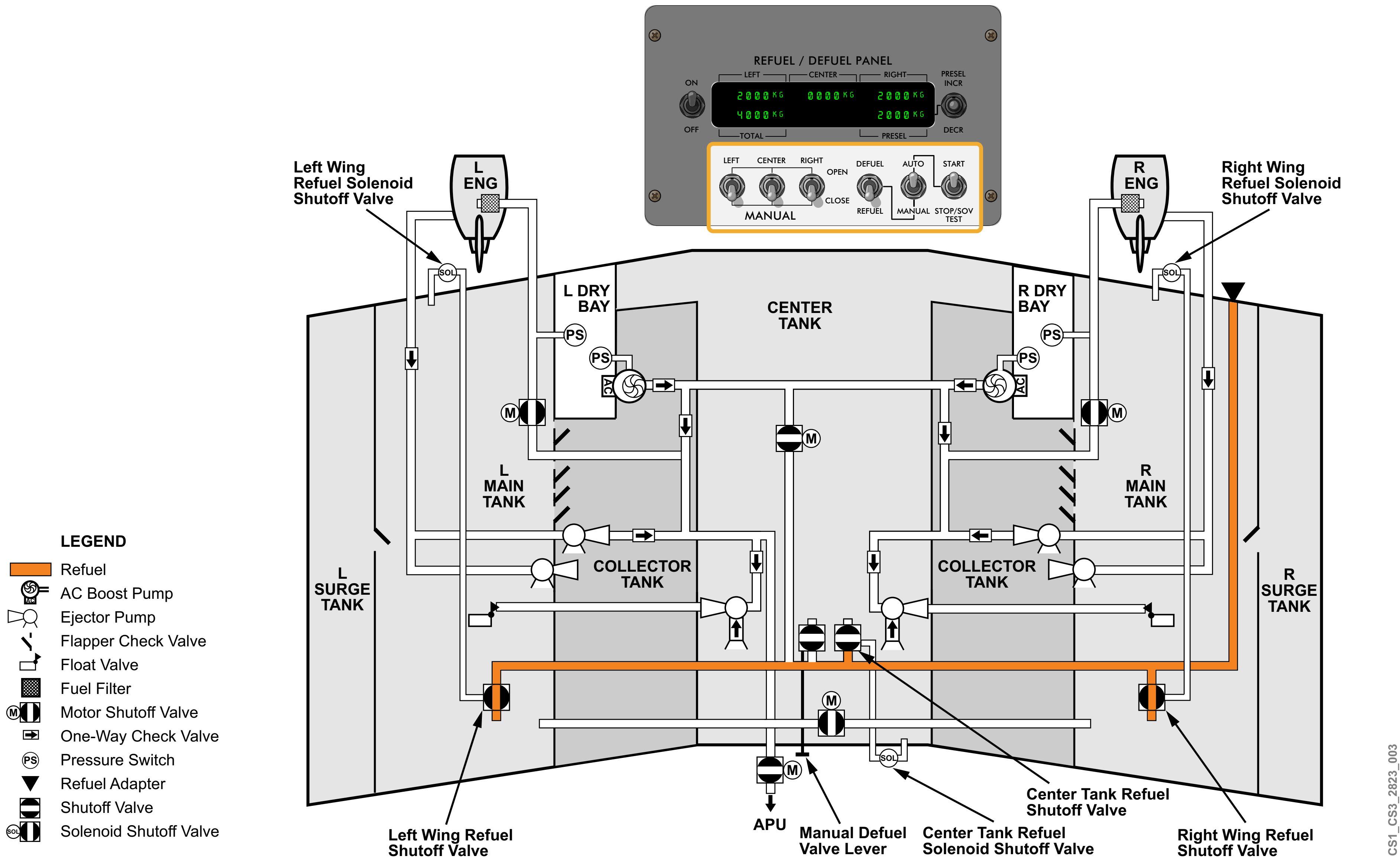


Figure 16: Auto Refueling

Manual Refueling

The manual mode is entered by selecting the MANUAL on the RDCP and the REFUEL/DEFUEL switch to the REFUEL position. MANUAL is displayed in the preselect window.

The refuel SOVs are opened by setting the MANUAL REFUEL VALVE switches to OPEN. The individual tank quantities are monitored on the RDCP. When the desired tank quantity is reached, the respective refuel SOVs are closed manually using the MANUAL REFUEL VALVE switches on the RDCP. In the manual mode, refueling continues until the MANUAL REFUEL VALVE switches are closed, or the fuel high-level condition is reached. At that point, the refuel valves are closed by the FQC.

These switches are hardwired to the FQC to provide a discrete to open or close their respective valve in case the RDCP fails.

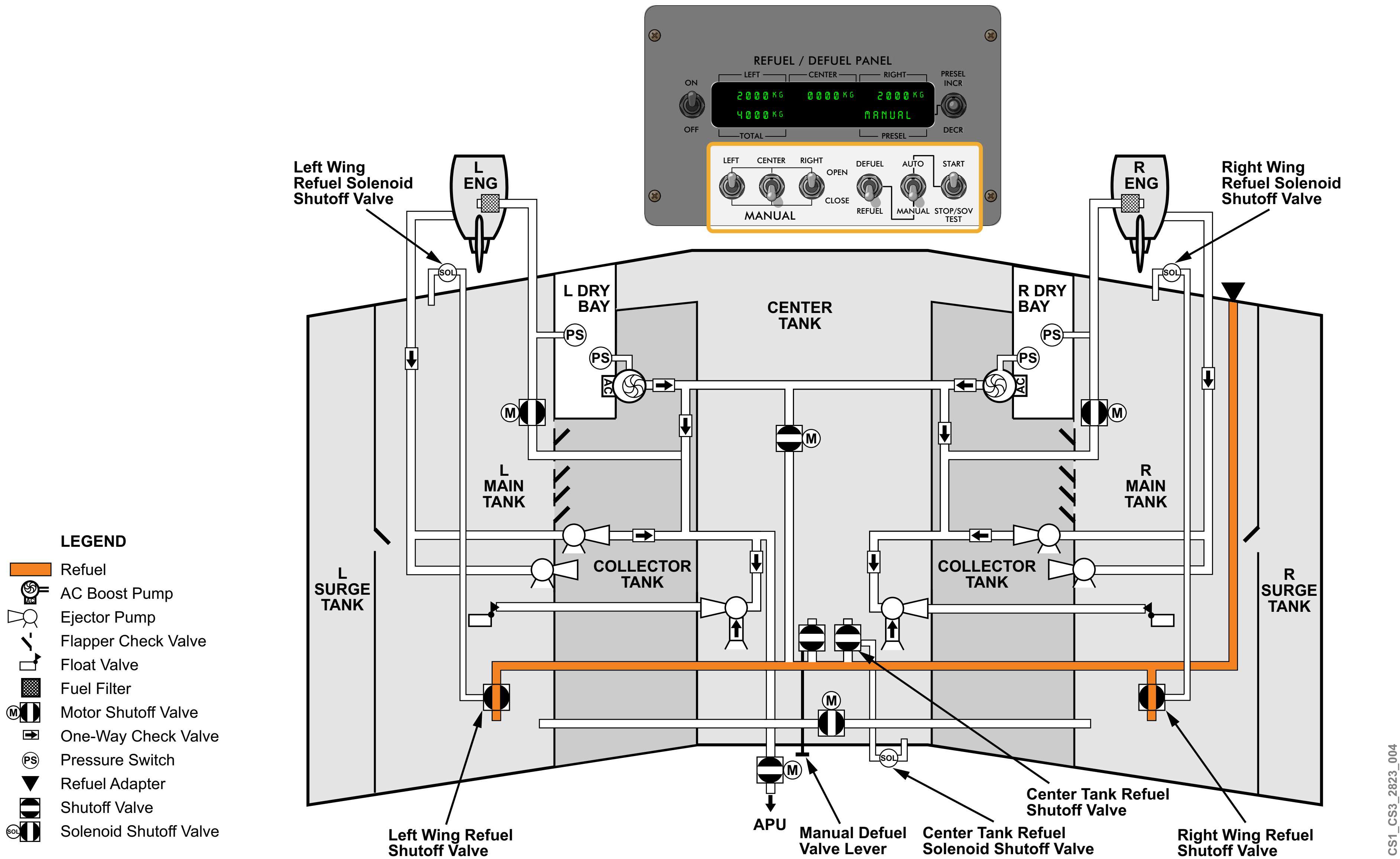


Figure 17: Manual Refueling

DEFUELING

Suction Defueling

Suction defueling is carried out using the fuel truck suction pressure. With the RDCP and refuel adapter configured for defueling, the fuel truck pumps are turned on. The maximum suction pressure allowed is -8 psig.

Fuel from the left and right main tank is suctioned out through the inlet snorkel of the AC boost pumps.

Fuel in the center tank is removed by opening the manual defuel valve on the rear spar of the center tank. Fuel is drawn out of the center tank through the defuel valve using fuel truck suction pressure.

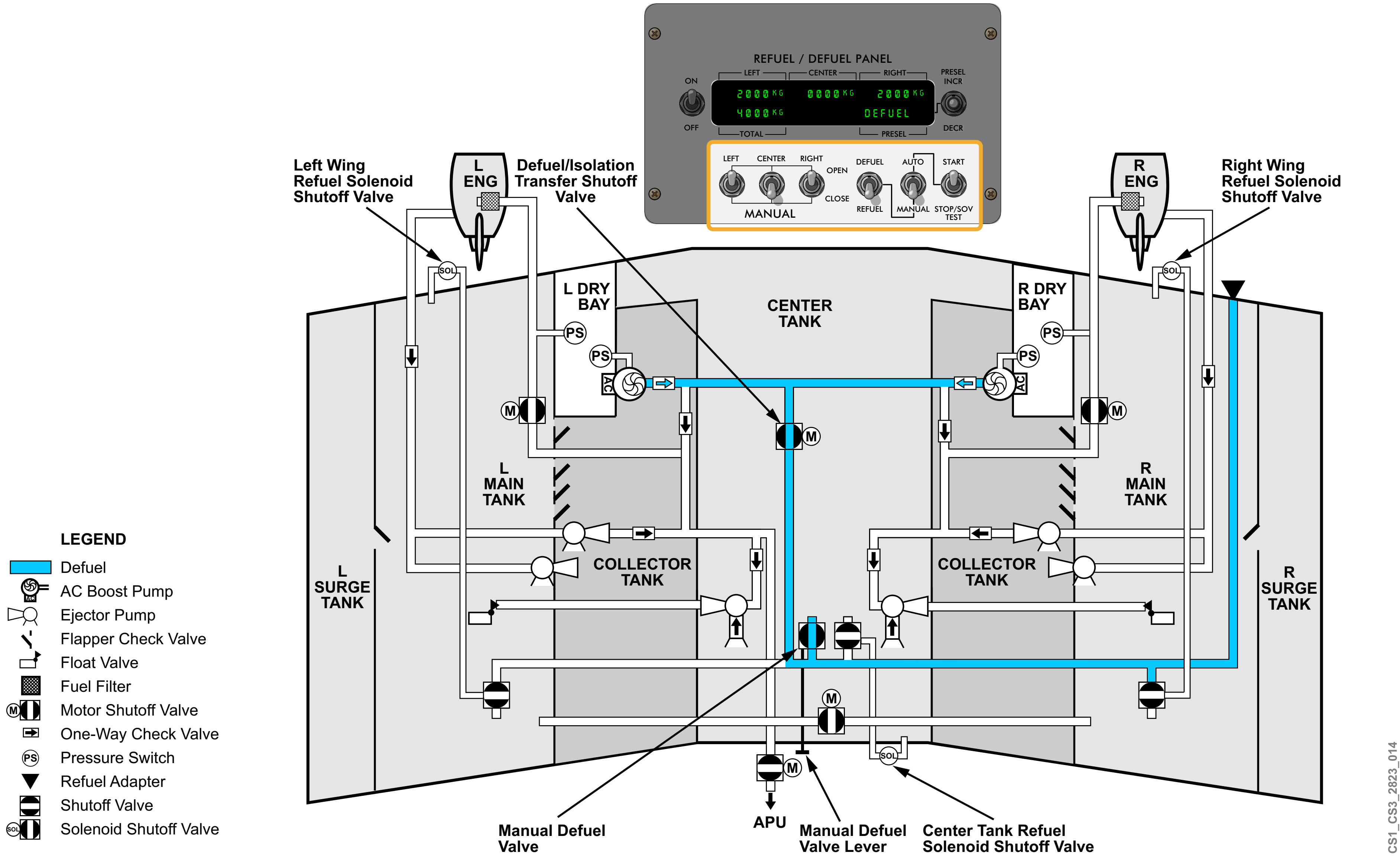


Figure 18: Suction Defueling

Pressure Defueling

Pressure defueling is carried out using the aircraft AC boost pumps. With the RDCP and refuel adapter configured for defueling, the AC boost pumps are turned on manually.

The fuel is pumped out of the main tanks through the engine feed manifold to the refuel manifold. If there is fuel in the center fuel tank, the AC boost pumps provide the motive flow to operate the transfer pumps. The transfer pumps empty the center tank fuel into the main tanks, where it is then pumped out.

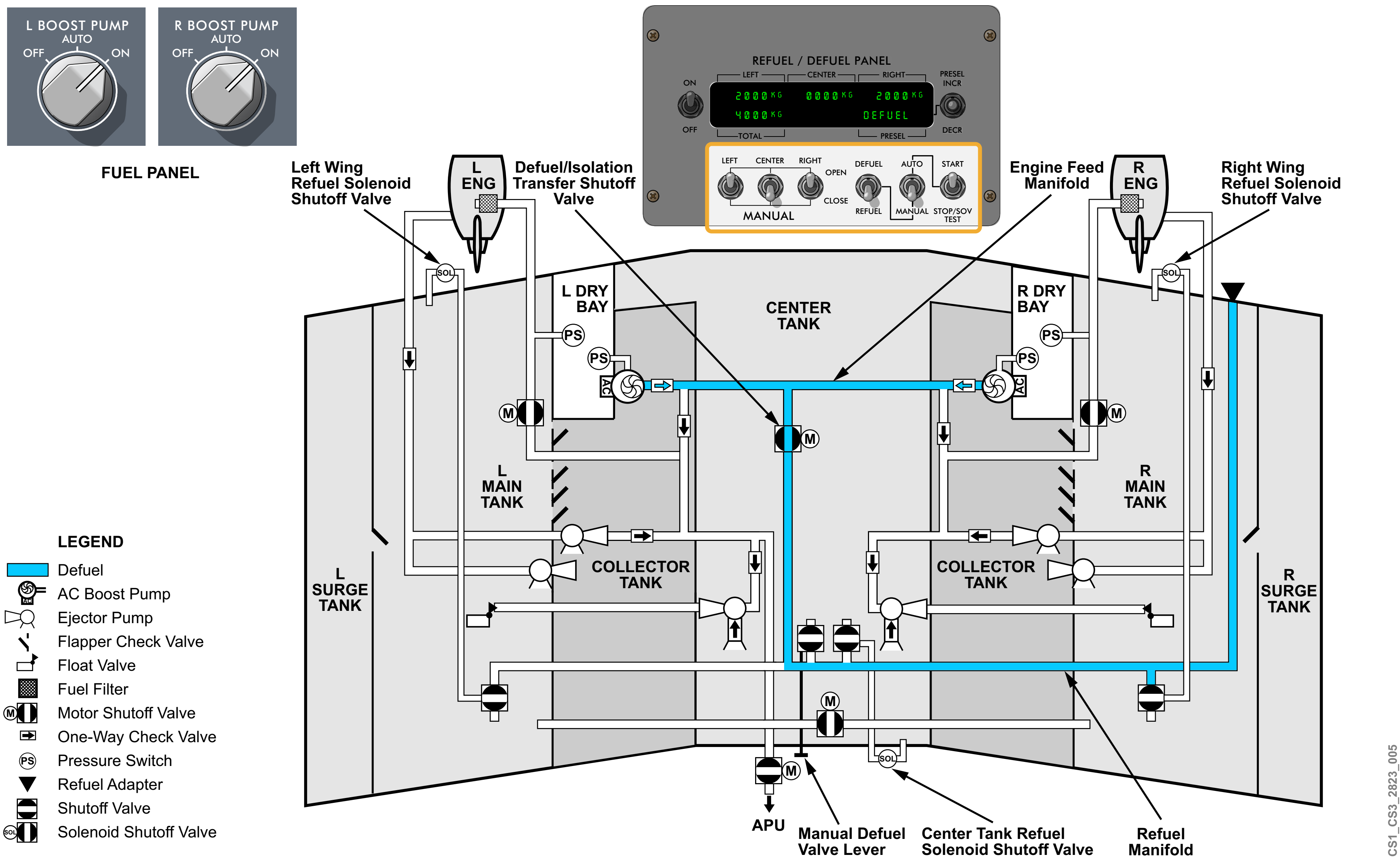


Figure 19: Pressure Defueling

COMPONENT LOCATION

The following components are installed in the pressure refuel/defuel system:

- Refuel adapter
- Refuel/defuel control panel
- Refuel shutoff valve
- Refuel control solenoid valve
- Defuel/isolation transfer shutoff valve and actuator
- Manual defuel valve and lever

REFUEL ADAPTER

The adapter is located on the right wing outboard of the engine at RIB 13R. An optional refuel adapter is available and installed at RIB 13L.

REFUEL/DEFUEL CONTROL PANEL

The refuel/defuel control panel (RDCP) is located on the fuselage near the inboard right wing leading edge.

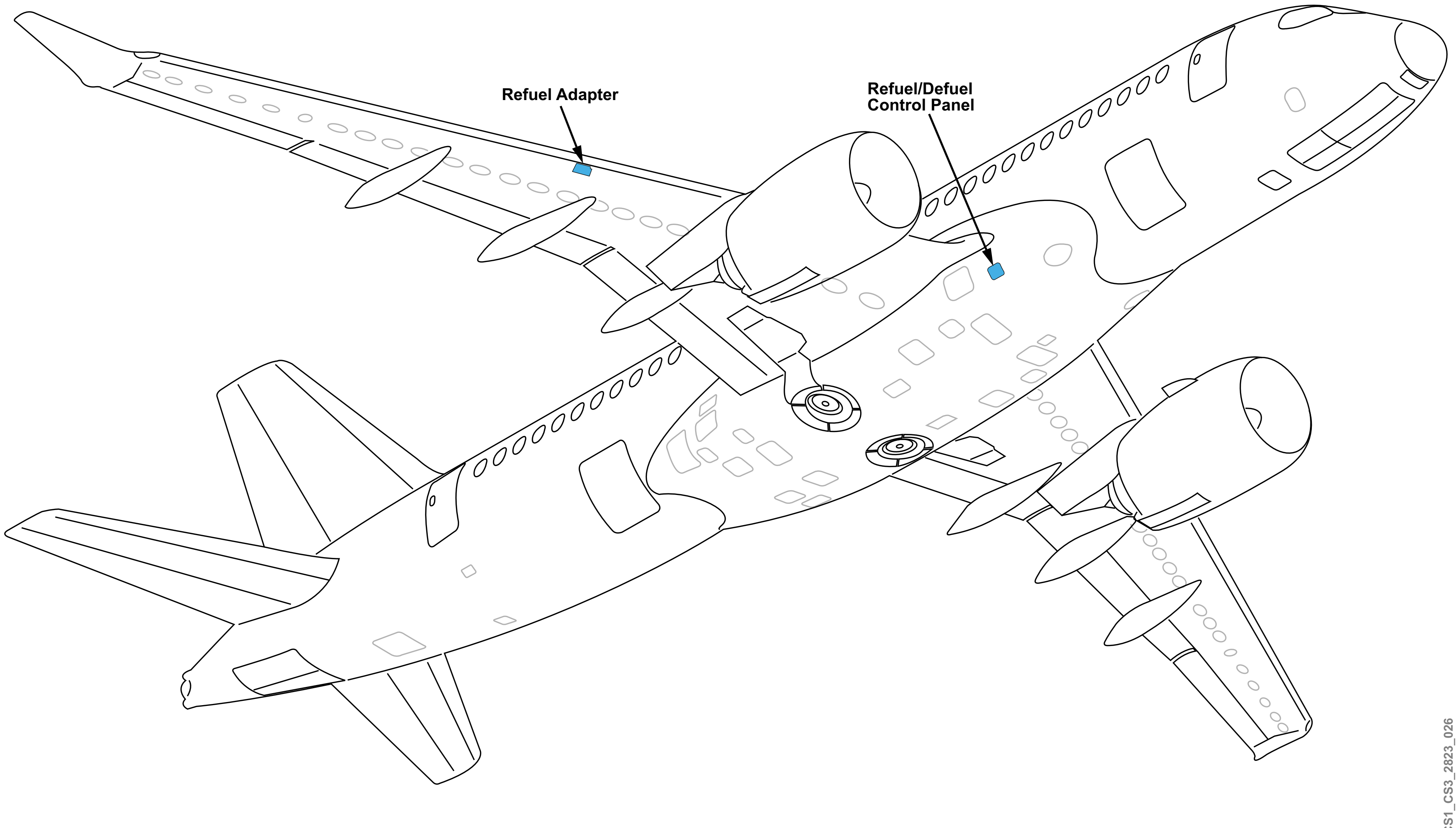


Figure 20: Refuel Adapter and Refuel/Defuel Control Panel Location

REFUEL SHUTOFF VALVE

The refuel shutoff valves for each main tank are mounted on the inboard side of RIB 12 inside the fuel tank. The center tank refuel shutoff valve is mounted on the inboard side of RIB 1L inside the center fuel tank.

REFUEL CONTROL SOLENOID VALVE

The main tank refuel control solenoid valves are mounted on a bracket at RIB 12 on the front spar. The center tank refuel control solenoid valve is mounted on the aft spar at RIB 1L.

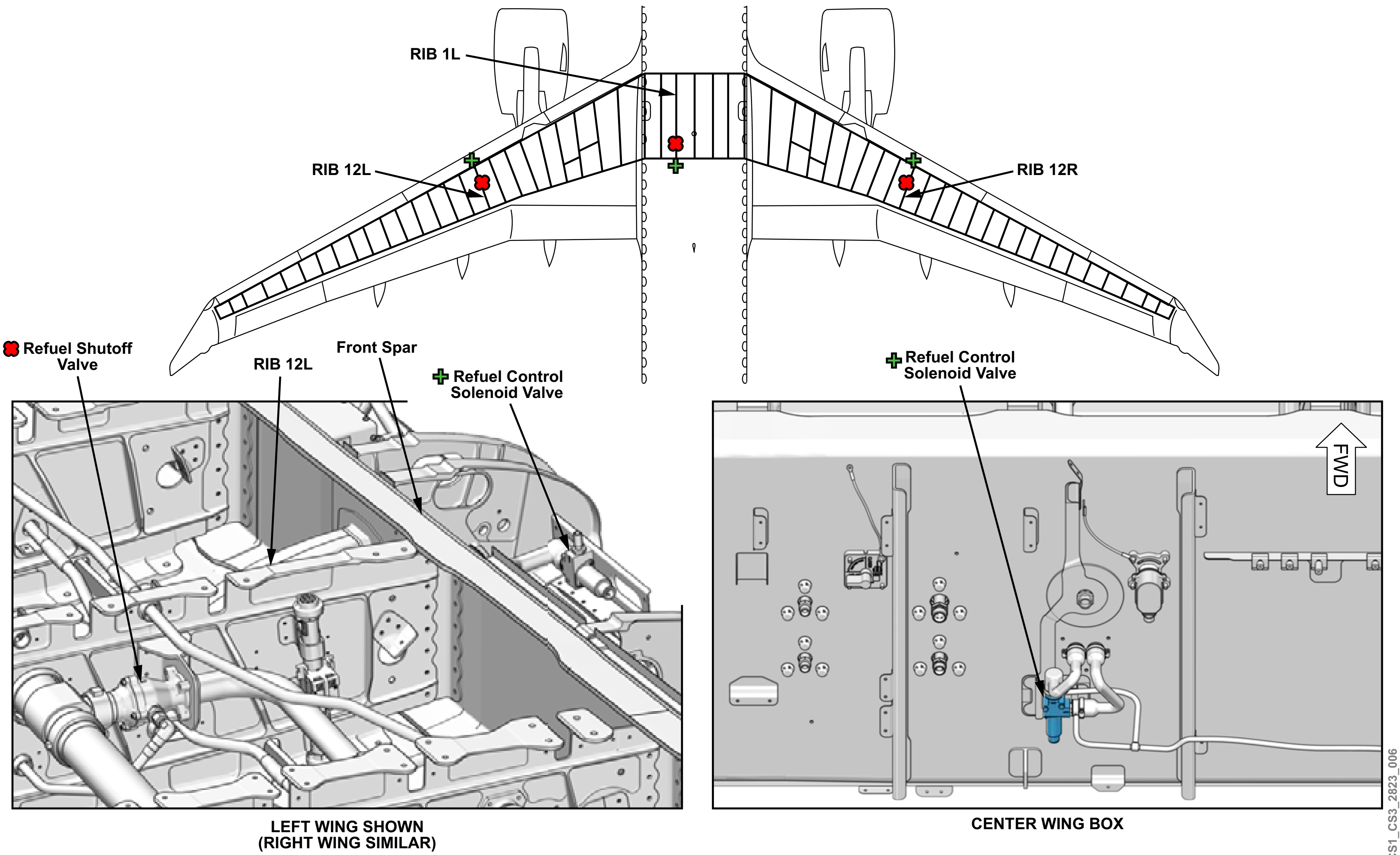


Figure 21: Refuel Control Solenoid Valve and Refuel Shutoff Valve Component Location

DEFUEL/ISOLATION TRANSFER SHUTOFF VALVE AND ACTUATOR

The valve body is mounted inside the center tank between RIB 0 and RIB 1L on the rear spar. The 28 VDC motor actuator is mounted outside of the tank.

MANUAL DEFUEL SHUTOFF VALVE AND LEVER

The center tank manual defuel shutoff valve is mounted inside the center tank on the rear spar between RIB 1L and RIB 2L. A manual lever, mounted on the outside of the tank, is used to operate the valve. The valve lever is pushed in to move it open or closed.

The lever assembly and valve body can be replaced independent of one another.

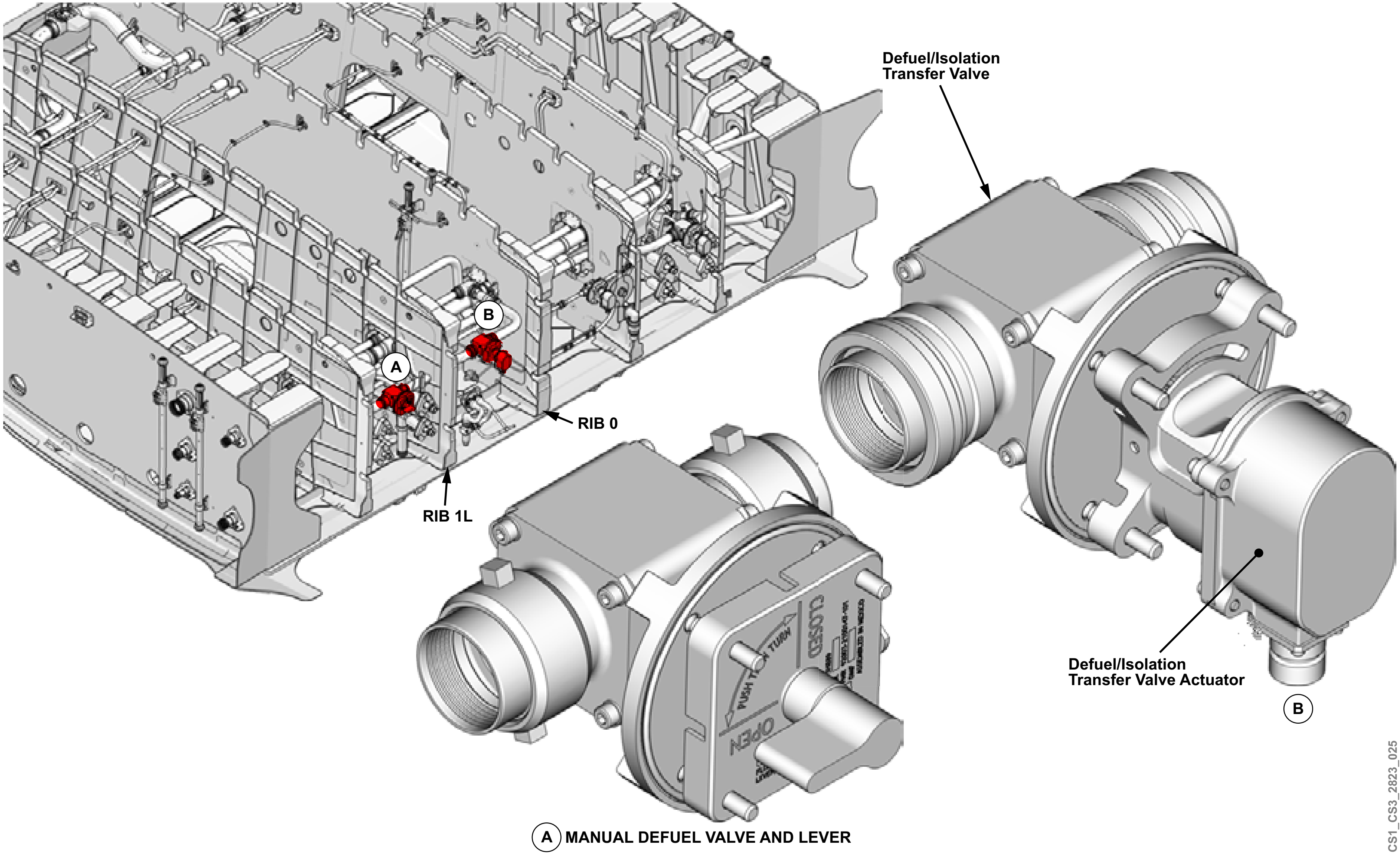


Figure 22: Defuel System Components

COMPONENT INFORMATION

REFUEL ADAPTER

The refuel adapter provides a connection point for single-point defueling/refueling of the aircraft. The refuel adapter has a poppet assembly and a fuel cap attached to the adapter by a cable assembly. During defueling, a locking mechanism is rotated to the DEFUEL position, allowing fuel to flow from the refuel manifold to the truck.

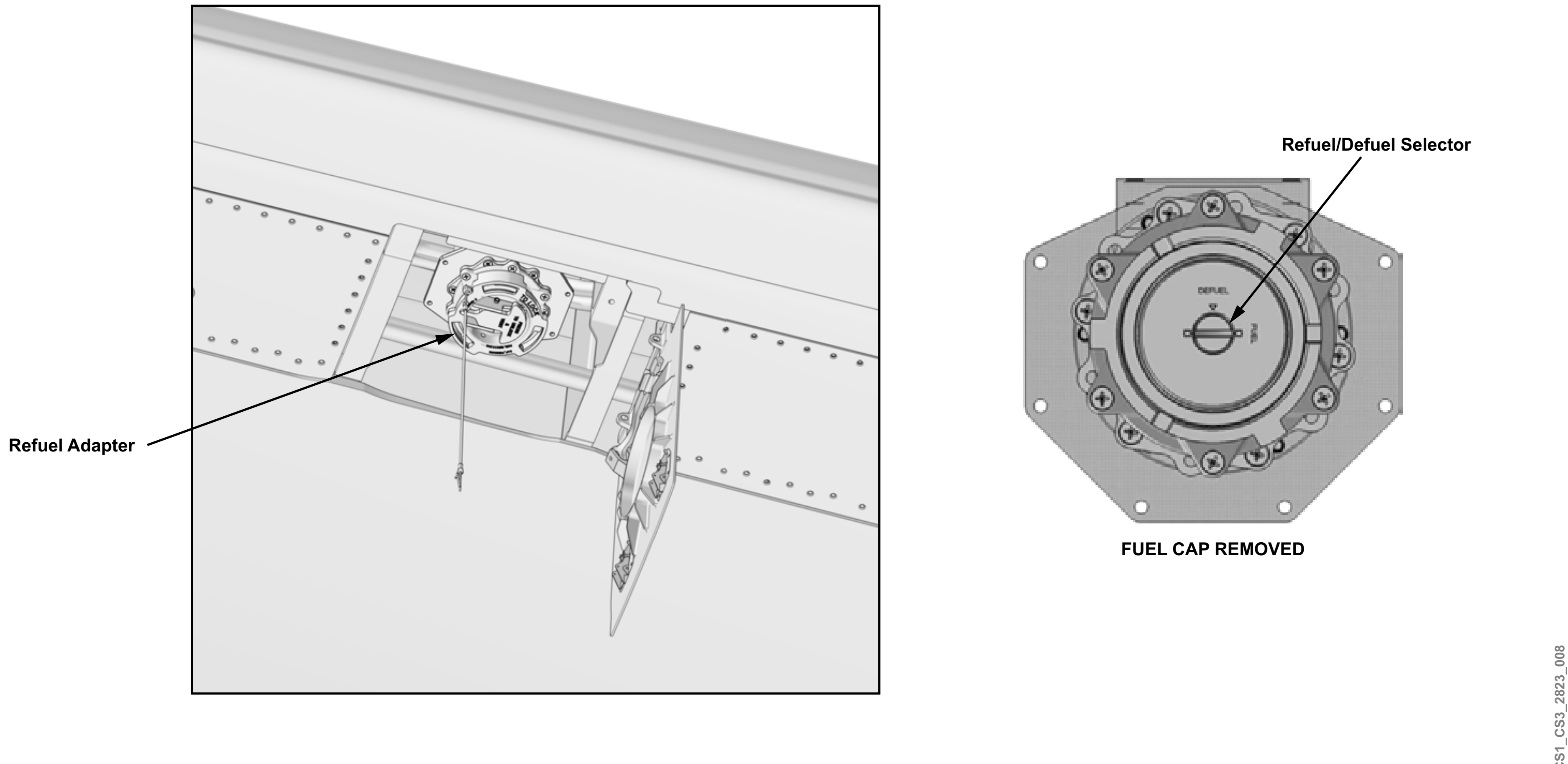


Figure 23: Refuel Adapter

REFUEL/DEFUEL CONTROL PANEL

The RDCP controls and monitors aircraft refueling and defueling. The RDCP functions are enabled when the aircraft is on the ground and the RDCP refueling door is in the open position. The refuel door switch signals the fuel quantity computer (FQC) that the refuel door is open.

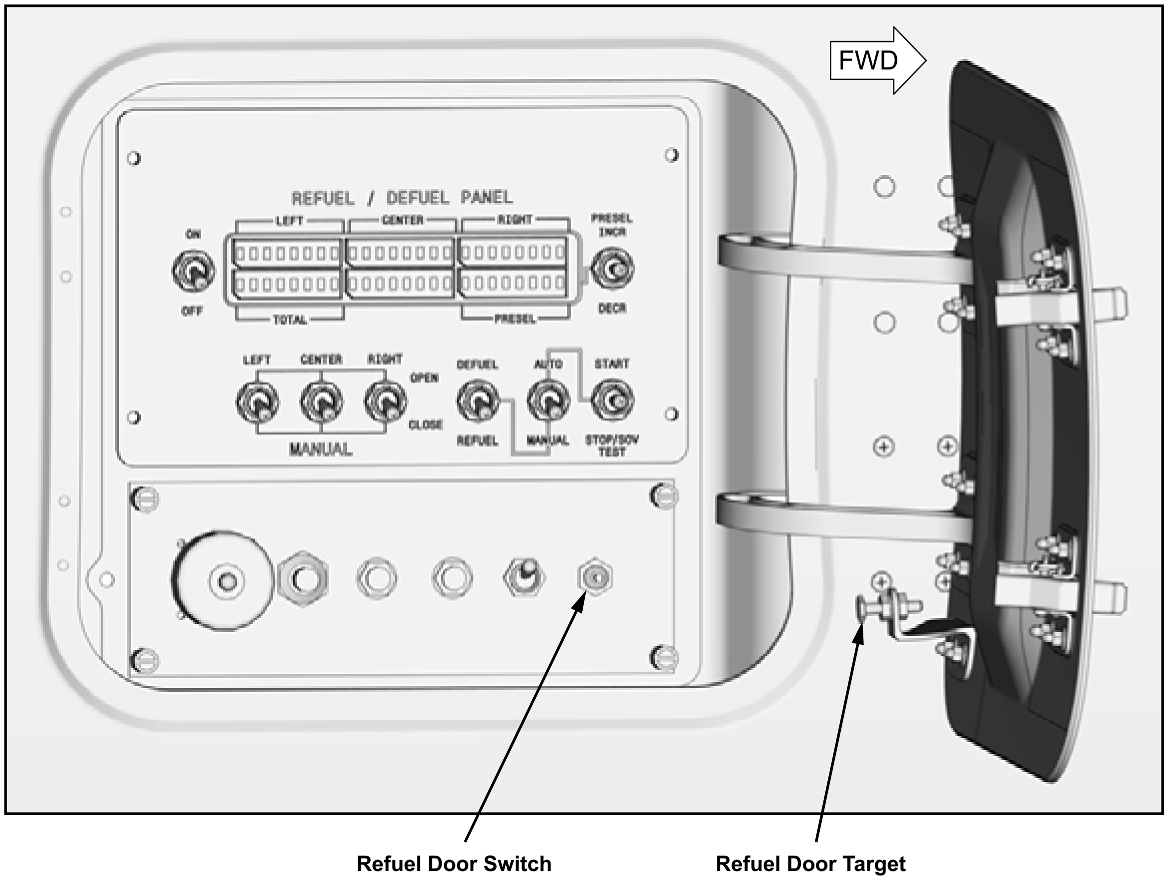


Figure 24: Refuel/Defuel Control Panel

CONTROLS AND INDICATIONS

REFUEL/DEFUEL CONTROL PANEL

The refuel/defuel control panel (RDCP) controls and monitors the aircraft refueling and defueling operations.

An indicator displays the left, center, and right fuel tank quantities, the total fuel quantity, and the load preselect quantity. The panel has switches to control power, automatic, or manual modes, and refuel and defuel modes.

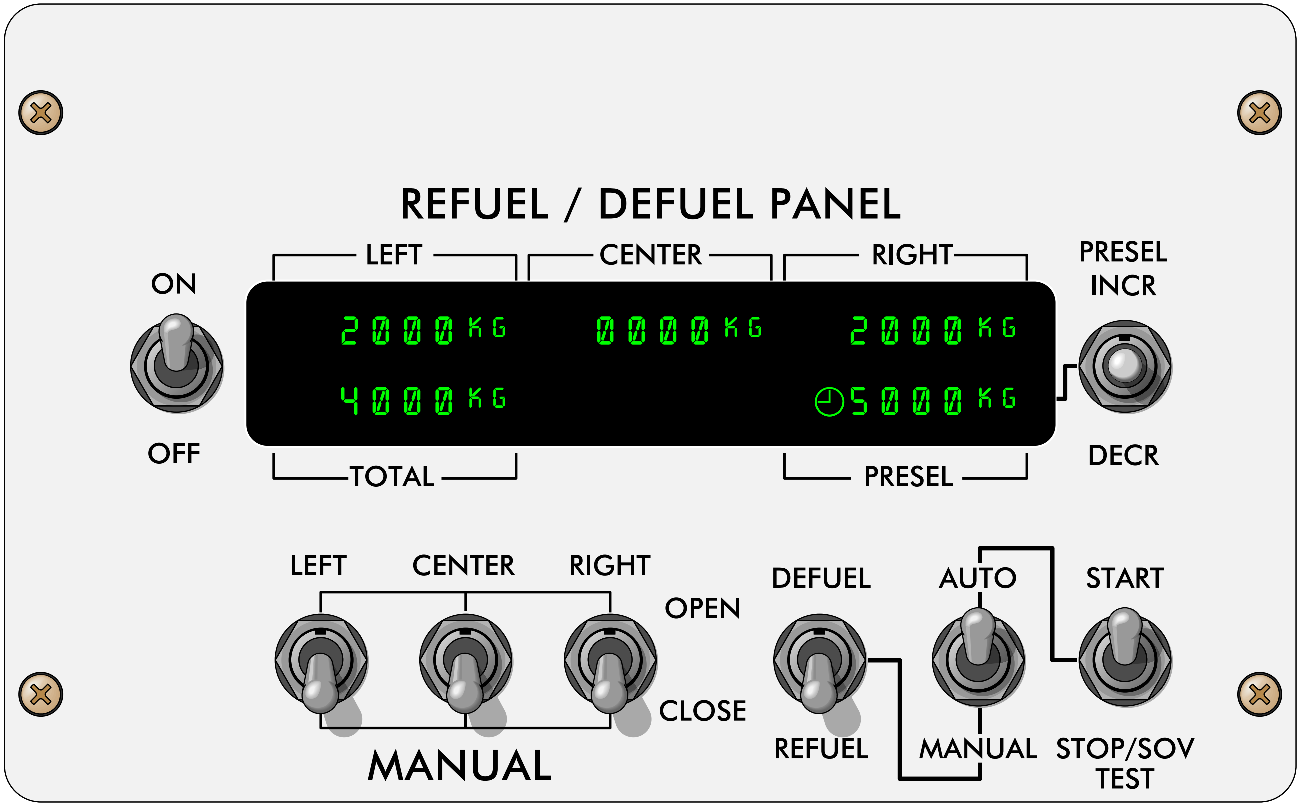


Figure 25: Refuel/Defuel Control Panel

RDCP Messages

Messages related to refueling and defueling display on the RDCP.

INHIBIT

When refueling the aircraft in manual mode or defueling the aircraft, an INHIBIT message is displayed whenever the RCDP is powered in manual mode or when switched to manual mode and any of the MANUAL LEFT/CENTER/RIGHT switches are in the OPEN position. A CLOSE ALL VALVES message is also displayed on the panel. The message clears when all of the MANUAL LEFT/CENTER/RIGHT switches are closed.

FULL

A FULL message indicates the level in the respective tank is between 97% and 98% of total capacity. The message remains as long as the condition exists.

HIGH LVL

A HIGH LVL message indicates the level in the respective tank is greater than 98% of total capacity. The message remains as long as the condition exists.

IMBAL

During refueling, an IMBAL message is displayed, flashing alternately with the quantity information when there is an imbalance between the left and right tank fuel quantities. A fuel quantity difference of 364 kg (800 lb) or greater between the wing fuel tanks is considered an imbalance. An IMBAL message is displayed for the fuel tank containing the lesser of the two imbalance quantities.

RCDP CPU FAULT

A fault has occurred in the RDCP CPU. The displays remain blank and the RDCP does not respond to any switch inputs.

RS-422 COMMUNICATION ERROR

A fault has occurred in the communications bus and the RDCP is inoperative. The displays remain dashed and the PRESEL portion of the window COMM ERR is displayed. The RDCP does not respond to any switch inputs.

RCDP SWITCH FAULT

The RDCP detects a switch fault (a momentary switch is held in the same position for more than 3 minutes). The message SWCH FLT will be displayed on the preselect window. The fuel tank quantities will continue to be displayed on the RDCP.

RDCP FAULT

For other RDCP faults, the message RDCP FLT is displayed on the preselect window. If possible, the RDCP continues to display the fuel tank quantities. The quantities displayed must be verified with information from the flight deck.

FQC FAULT

An FQC fault that affects the refueling process is detected, the message FQC FLT is displayed on the preselect window. If possible, the RDCP continues to display the quantities.

AUTOMATIC REFUELING STOPPED

An automatic refuel has been stopped before completion (START/STOP switch toggled to the STOP position during an automatic refuel), the STOPPED message is displayed in the preselect window for 5 seconds.

Figure 26: Refuel/Defuel Control Panel Messages

OPTIONAL VIRTUAL REFUEL PANEL

The virtual refuel panel is displayed on the FUEL synoptic page. The virtual refuel panel is displayed when the aircraft is on ground only with no engines running. To refuel the aircraft using the virtual refuel panel, the refuel door must be open, the RDCP powered, and the REFUEL switch set to AUTO.

This panel allows the crew to enter the total fuel quantity in the preselect data entry field. Pressing ENTER sends the data to the FQC. If the preselect value entered exceeds the tank capacity, maximum allowable fuel quantity is assumed by the FQC. If refueling is in progress, the preselect fuel quantity can be modified. If the preselect value is less than the current fuel load, refueling stops at the current fuel quantity level. When the virtual refuel panel is in control of the preselect quantity, the RDCP ignores inputs on the RDCP PRESEL INCR/DECR switch.

The refuel SOVs cannot be opened from the flight deck. The START switch on the RDCP must be selected to open the aircraft refuel SOVs. The virtual refuel panel has priority over the RDCP commands. If no value or 0 is entered in the preselect field of the virtual refuel panel, the preselect fuel quantity can be entered in the RDCP. Cycling the RDCP power for 10 seconds overrides the flight deck preselect field.

The source of the preselect fuel quantity used by the FQC is displayed on the virtual refuel panel and on the RDCP. The left, center, right, and total fuel quantities are still displayed on the RDCP in their respective windows. The PRESEL window displays the message COCKPIT alternating with the preselect quantity that is entered in the flight deck refuel panel. The COCKPIT message indicates that the entry has been made in the flight deck and that refueling may be started.

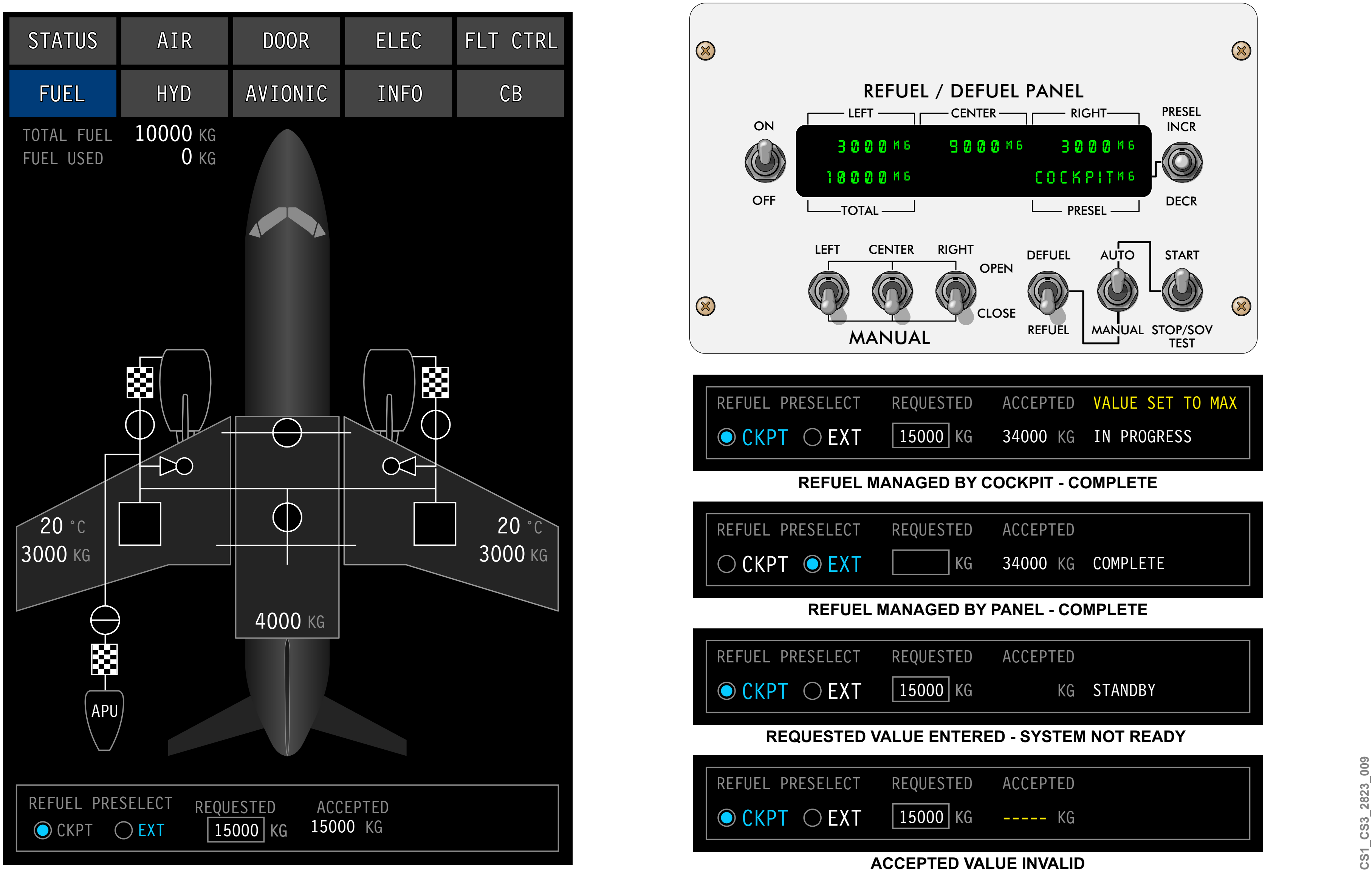


Figure 27: Virtual Refuel Panel

OPERATION

AUTO REFUEL OPERATION

1. Open the REFUEL panel door.
2. Set the ON/OFF panel switch to the ON position.
3. Set the AUTO/MANUAL switch to the AUTO position.
4. Hold the PRESEL INCR/DECR switch to the INCR position to increase the desired refuel quantity displayed in the PRESEL window, or hold the switch to the DECR position to decrease the desired refuel quantity displayed in the PRESEL window.
5. Perform the refuel valve SOV TEST. The STOP/SOV TEST position is used to test the SOV operation at the start of refueling. The refuel valves are tested by applying fuel pressure and then selecting the SOV TEST position. An SOV TEST IN PROGRESS message appears in the display windows, alternating every 2 seconds with the fuel quantity display. The fuel valves close and an SOV TEST PASSED message is displayed on the RDCP indicating a successful test. The message is removed when any RDCP switch is selected.
6. Set the START/STOP/SOV TEST switch to the START position. The PRESEL window then displays a clock icon and the preselected fuel quantity message.
7. The refuel flow to the aircraft stops automatically when the preselected quantity is achieved. In the PRESEL window COMPLETE is displayed, alternating every 2 seconds with the preselected fuel quantity.
8. Set the ON/OFF panel switch to the OFF position and close the REFUEL panel door.

NOTE

1. If a fuel imbalance greater than 364 kg (800 lb) occurs between the wing fuel tanks, refueling must be stopped.
2. When all tanks are refueled simultaneously; the time to completely refuel the center tank is approximately 30 minutes and 20 minutes to completely refuel the wing tanks.
3. If the refueling operation is stopped by selecting the START/STOP/SOV TEST to STOP/SOV TEST, a STOPPED message is displayed for 5 seconds.

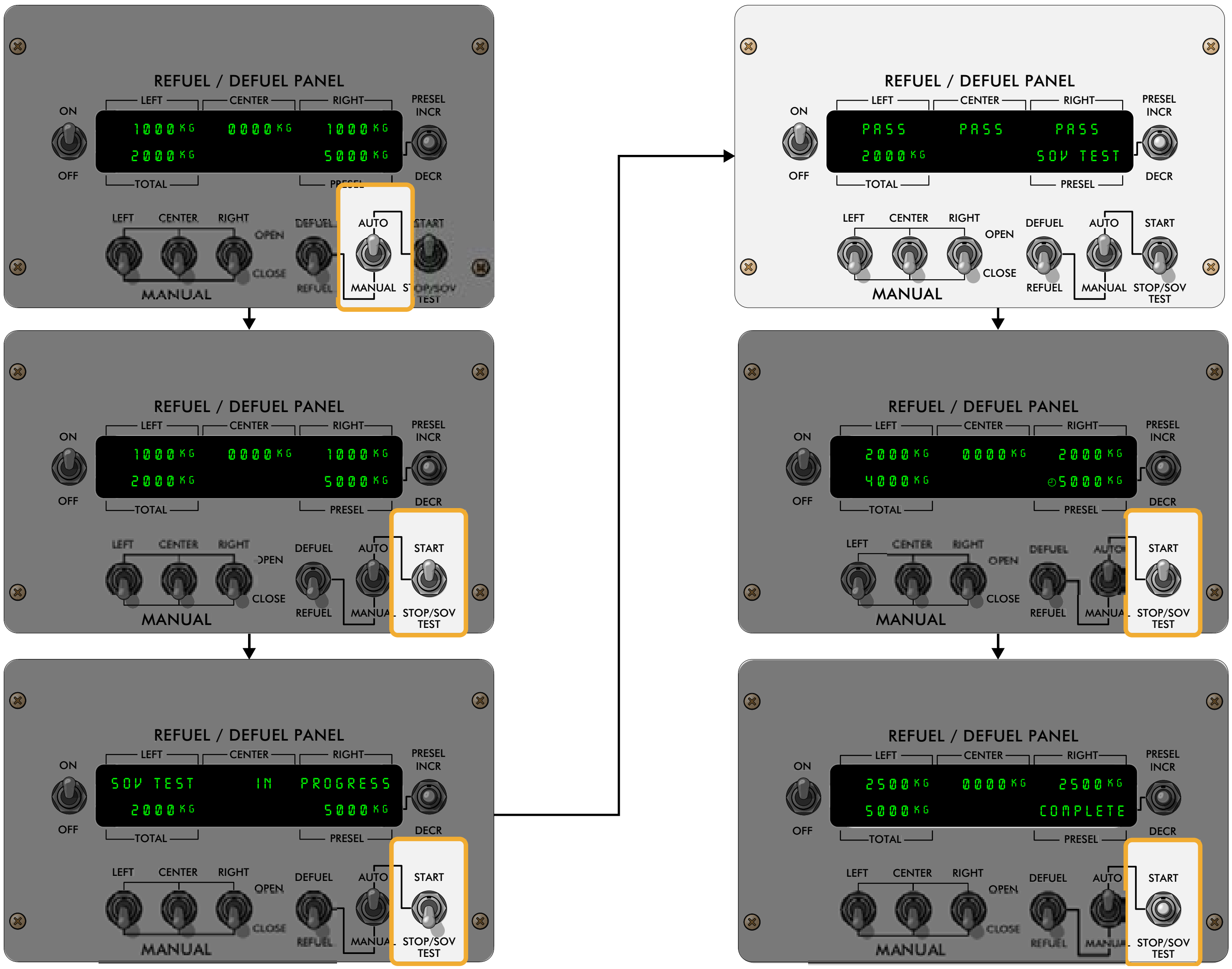


Figure 28: Automatic Refuel Operation

MANUAL REFUEL OPERATION

1. Open the REFUEL panel door.
2. Set the ON/OFF panel switch to the ON position.
3. Set the MANUAL LEFT/CENTER/RIGHT switches to the CLOSE position.
4. Set the REFUEL/DEFUEL switch to REFUEL.
5. Set the AUTO/MANUAL switch to the MANUAL position.
6. Set the MANUAL LEFT/CENTER/RIGHT switches to the OPEN position for the tanks that are to be filled.
7. Set the MANUAL LEFT/CENTER/RIGHT switches to the CLOSE position when the required fuel level is reached.
8. Set the ON/OFF panel switch to the OFF position.

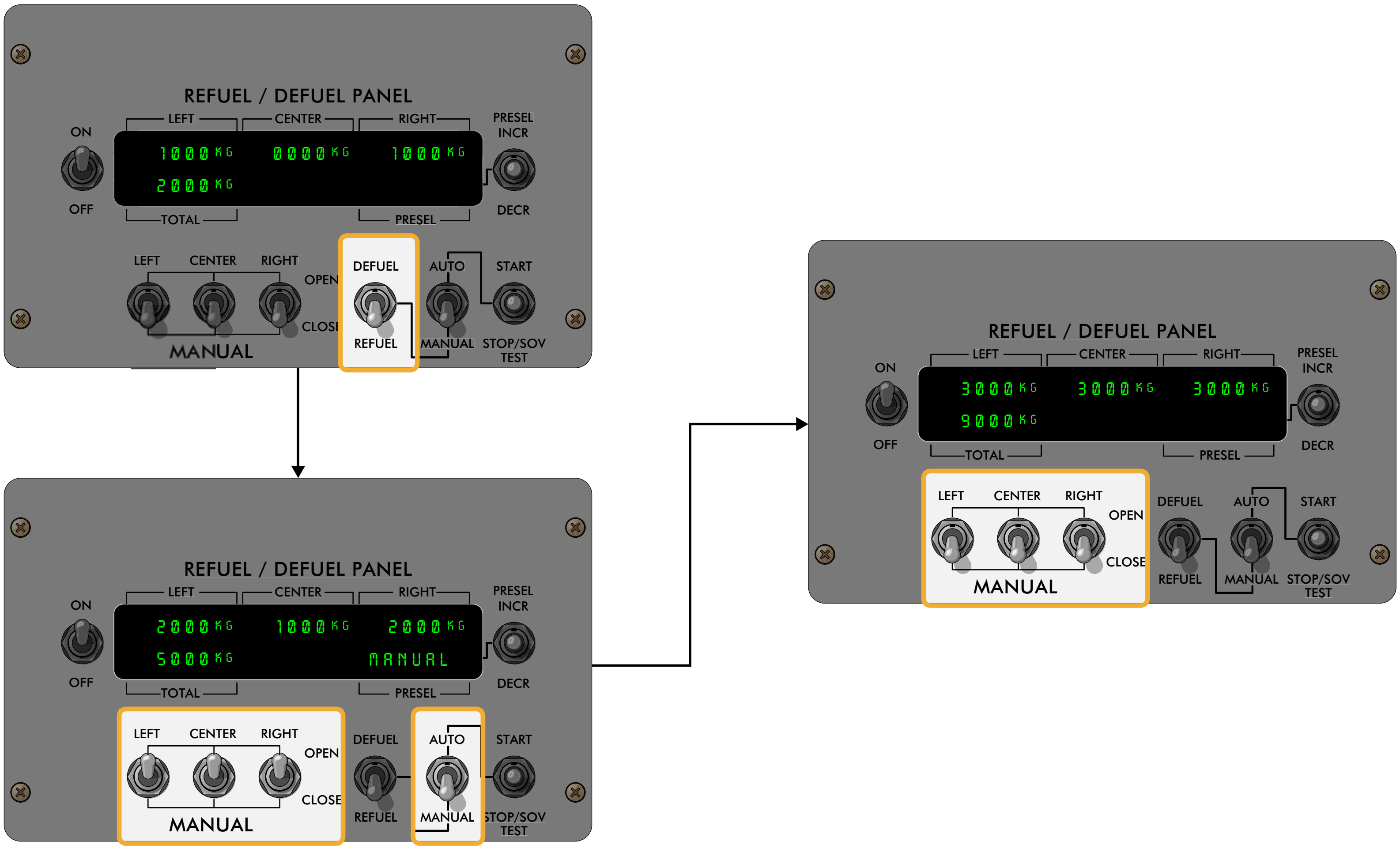
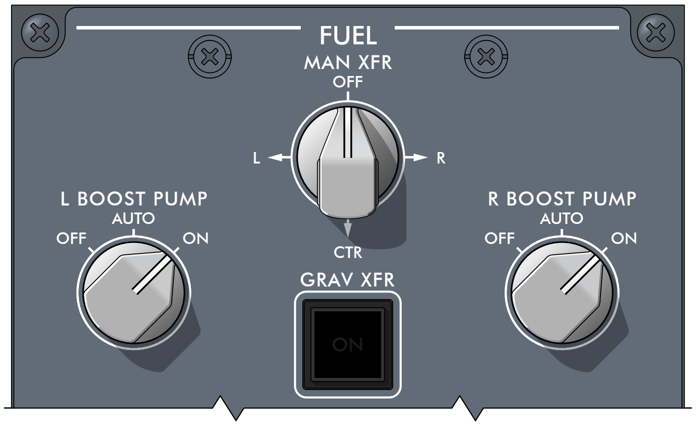


Figure 29: Manual Refuel Operation

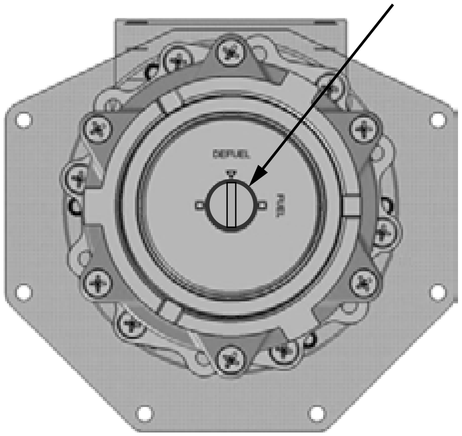
PRESSURE DEFUEL OPERATION

1. Connect external AC power to the aircraft.
2. Open the REFUEL panel door.
3. Set the ON/OFF switch to the ON position.
4. Set all MANUAL LEFT/CENTER/RIGHT switches to the CLOSE position.
5. Set the MANUAL REFUEL/DEFUEL switch to DEFUEL. Verify that the defuel message is displayed on the RDCP window.
6. On the pressure refuel adapter, set the REFUEL/DEFUEL selector to DEFUEL.
7. Connect the refuel hose nozzle to the refuel adapter.
8. Set all MANUAL LEFT/CENTER/RIGHT switches to the OPEN position.
9. Set the GRAV XFR PBA to the CLOSED position.
10. Set the BOOST PUMP switches to the ON position for the fuel tank(s) to be defueled.
11. When an AC boost pump low-pressure indication is displayed on the FUEL synoptic page, set the BOOST PUMP switch to the OFF position.
12. Set all MANUAL LEFT/CENTER/RIGHT switches to the CLOSE position.
13. Set the ON/OFF switch to the OFF position.
14. Disconnect the refuel hose nozzle from the refuel adapter and turn the pressure refuel adapter REFUEL/DEFUEL selector to DEFUEL.



FUEL PANEL

Refuel/Defuel Selector



(FUEL CAP REMOVED)

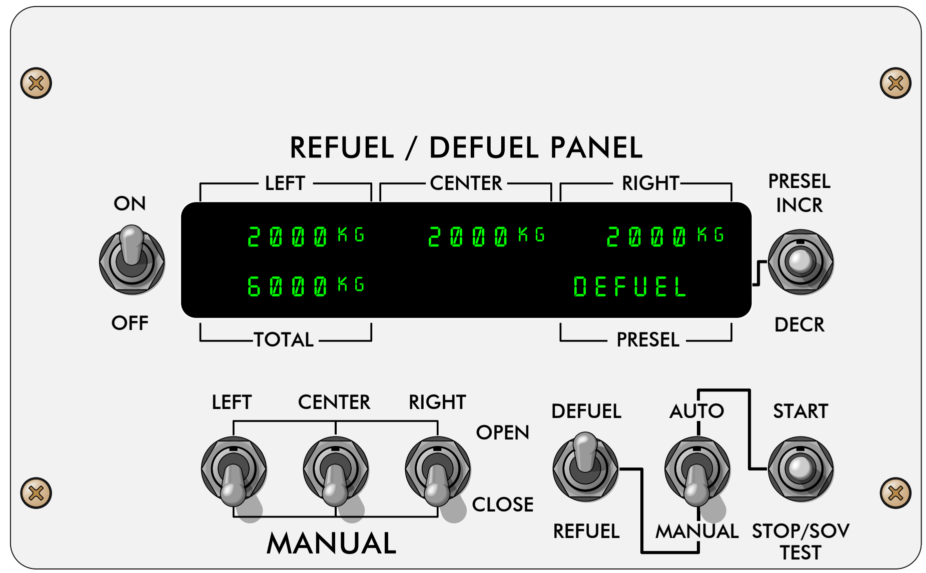
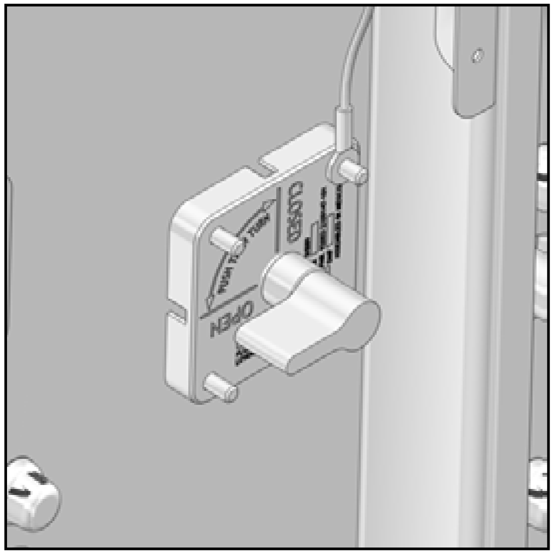


Figure 30: Pressure Defuel Operation

SUCTION DEFUEL OPERATION

1. Open the REFUEL panel door.
2. Set the ON/OFF switch to the OFF position.
3. Set each of the MANUAL LEFT/CENTER/RIGHT switches to the CLOSE position.
4. If the center fuel tank is to be defueled, the MANUAL DEFUEL valve is opened by pushing and turning the MANUAL DEFUEL LEVER, located on the rear spar of the center fuel tank.
5. On the pressure refuel adapter, set the REFUEL /DEFUEL selector to DEFUEL.
6. Set the REFUEL/DEFUEL switch to the DEFUEL position.
7. Set the POWER SWITCH to the ON position. The PRESEL display window shows DEFUEL.
8. Set the AUTO/MANUAL switch to the MANUAL position.
9. Attach the hose to the refuel adapter. Start the fuel truck pump, which must be configured in the suction mode.
10. Set all MANUAL LEFT/CENTER/RIGHT switches to the OPEN position.
11. Fuel is suctioned out of the wing tanks.
12. Monitor the LEFT/CENTER/RIGHT display window and stop the fuel truck pump when desired quantity is reached.
13. Manually close center tank MANUAL DEFUEL valve when the desired quantity in the center wing tank is reached.
14. Set the MANUAL LEFT/RIGHT switch(es) to the CLOSE position to stop defueling in all tanks.
15. Set the ON/OFF switch to the OFF position.
16. Disconnect the refuel hose nozzle from the refuel adapter and turn the pressure refuel adapter REFUEL/DEFUEL selector to DEFUEL.



MANUAL DEFUEL VALVE

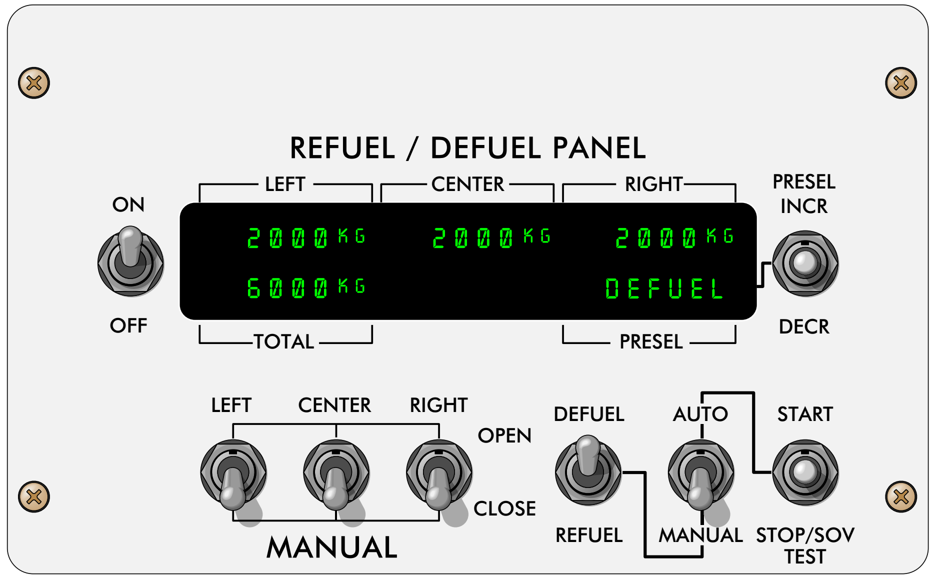


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Figure 31: Suction Defuel Operation

DETAILED DESCRIPTION

REFUEL POWER

The fuel system has two power modes for normal and battery only refuel/defuel operation. In normal operation, the refuel panel door is closed and the system is powered by DC ESS BUS 1 and DC ESS BUS 2. When the aircraft is refueled, the refuel door is open. The refuel door switch signals the FQC that the door is opened. When the RDCP power switch is selected on, the FQC receives a 28 VDC signal indicating the refuel panel is turned on. The refuel door switch also energizes relays in CDC 1 and CDC 2 to switch the fuel power from the DC ESS BUSES to the DC EMER BUSES.

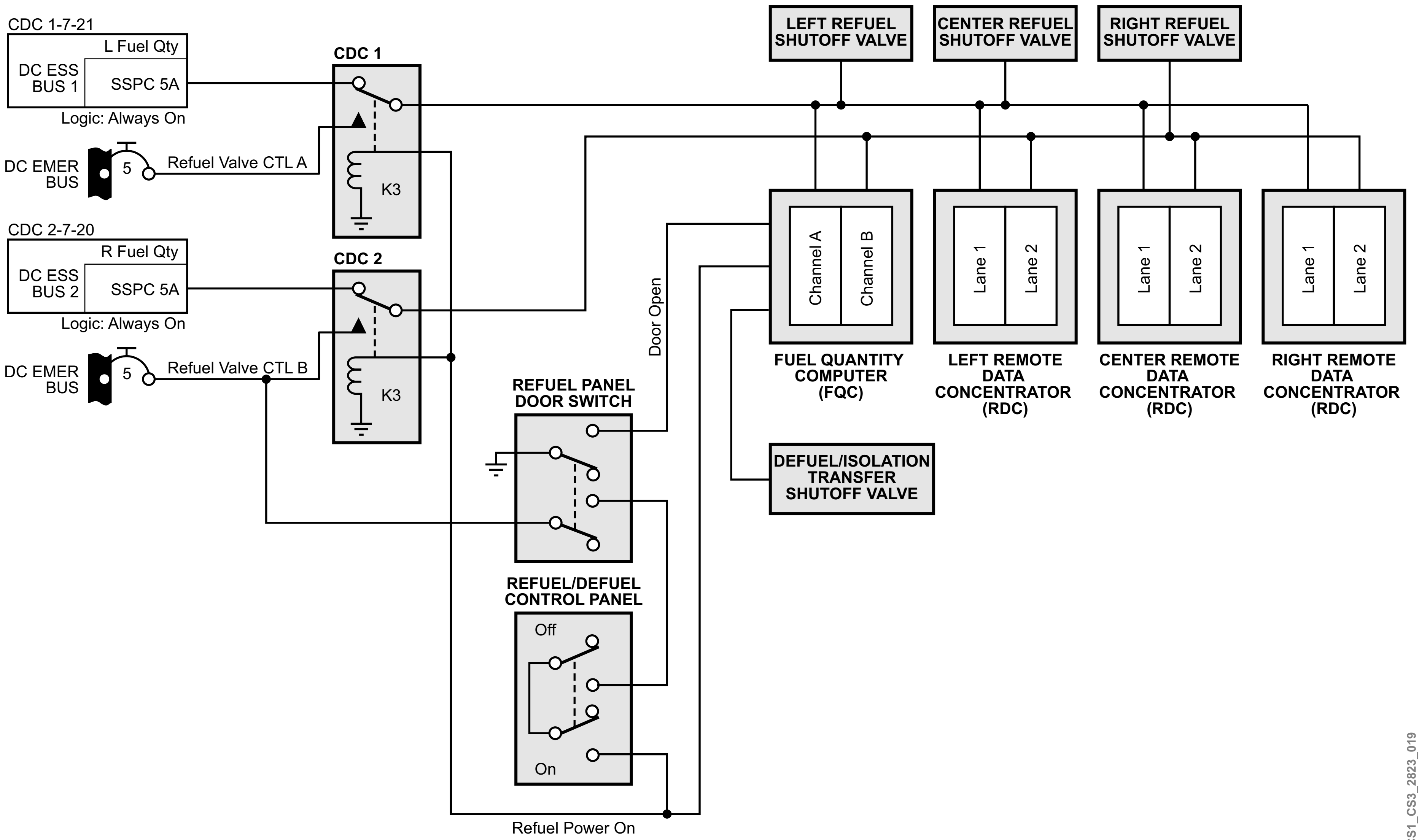


Figure 32: Refuel Power (Normal Operation - Door Closed)

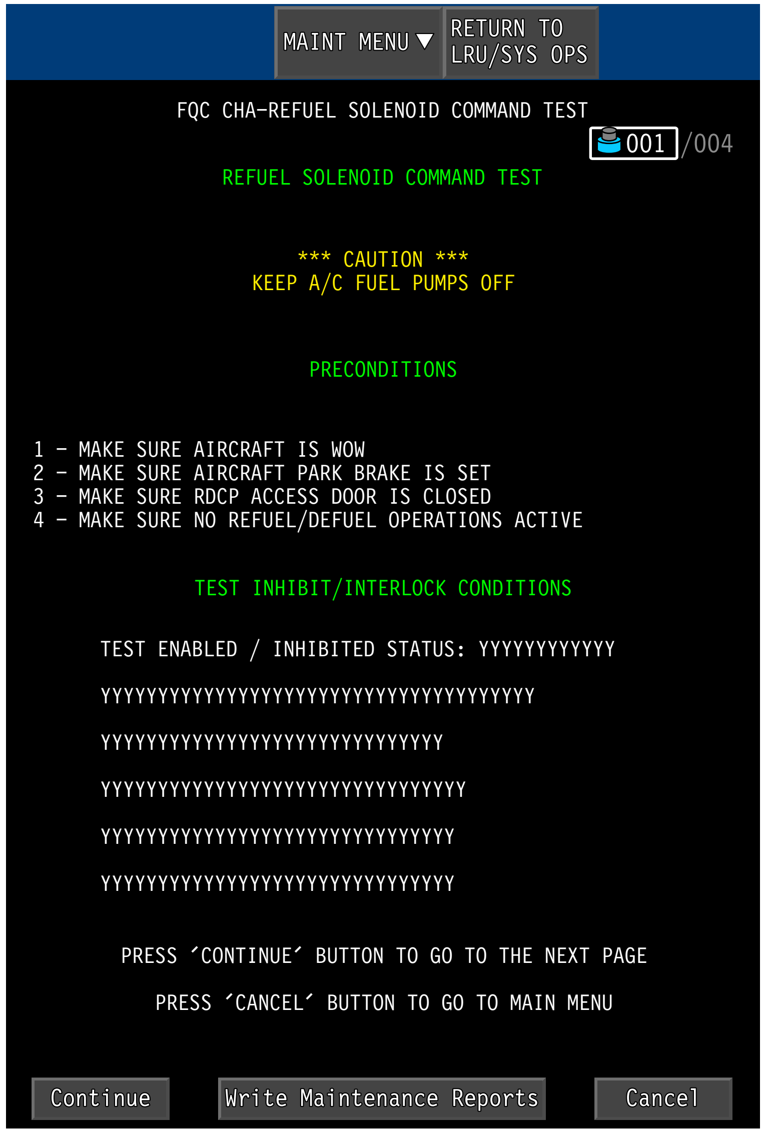
PRACTICAL ASPECTS

ONBOARD MAINTENANCE SYSTEM

The following refuel/defuel components are tested through the onboard maintenance system (OMS).

Refuel Solenoid Command Test

When the OMS commands a refuel solenoid command test, the selected FQC channel energizes each refuel solenoid and verifies that the solenoid is energized. The FQC channel then de-energizes each refuel solenoid and verifies that the solenoid is de-energized. This test checks the electrical command of the refuel solenoids, but does not verify fuel flow.

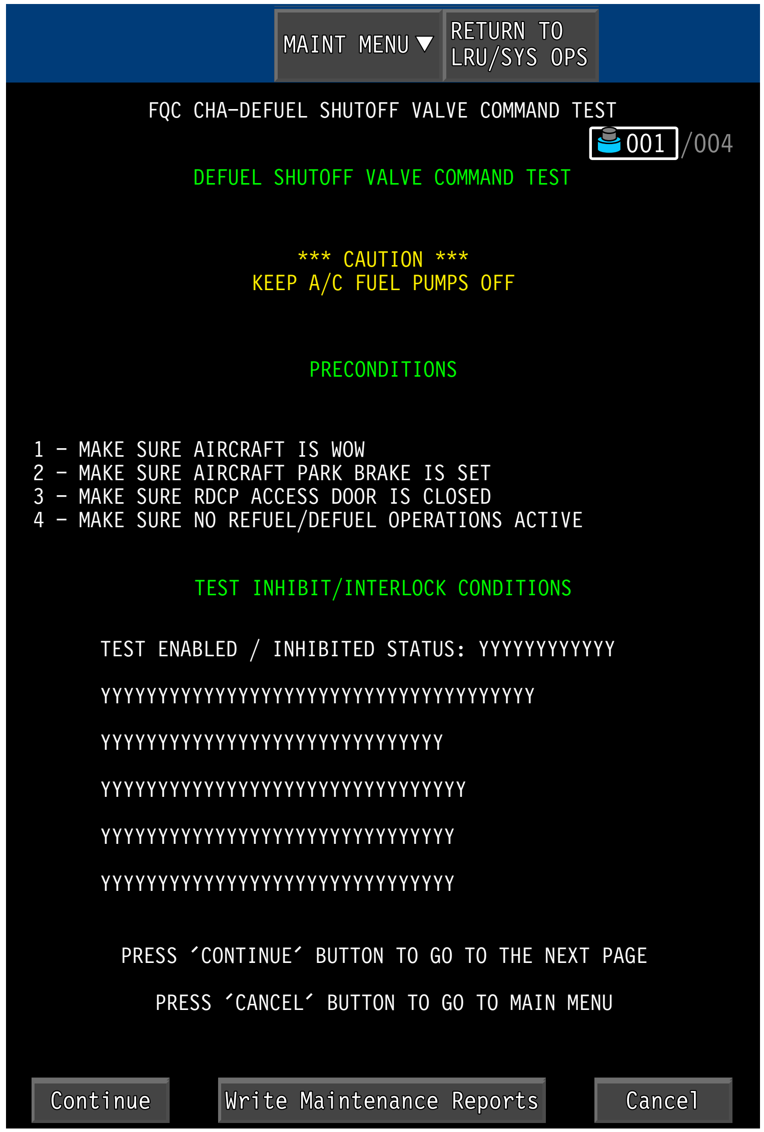


CS1_CS3_2823_016

Figure 33: Refuel Solenoid Command Test

Defuel Shutoff Valve Command Test

When the OMS commands a defuel shutoff valve command test, the selected FQC channel commands the DEFUEL SOV open, and verifies that the SOV position feedback has reported that the valve has opened. The FQC closes the defuel SOV and verifies that the SOV position feedback reports that the valve has closed. This test checks the electrical command of the defuel shutoff valve, but does not verify fuel flow.



CS1_CS3_2823_021

Figure 34: Defuel Shutoff Valve Command Test

28-21 ENGINE FEED SYSTEM

GENERAL DESCRIPTION

The feed system consists of one engine feed ejector pump and one AC boost pump in the collector tank. Each engine has its own feed system. The two engine feeds are connected to allow either AC boost pump to feed both engines.

Check valves in the engine feed lines prevent a loss of fuel pressure when an ejector feed pump or boost pump fails.

Normally-open engine feed shutoff valves in each engine fuel feed line stop the flow of fuel in the event of an engine fire.

Each engine feed system is monitored by two pressure switches. Low fuel pressure at the engine is detected by a pressure switch in the engine feed line. Each AC boost pump has a pressure switch at the pump outlet to detect low-pressure.

In the event of a total failure of the engine feed pumps, the engine can suction feed through the AC boost pump inlet up to 25,000 ft. The engine-driven pump (EDP) suctions fuel through the engine feed ejector induced port and the AC boost pump inlet snorkel.

Motive flow fuel generated by the EDP powers the onside engine feed and scavenge ejector pumps. A check valve in the motive flow line isolates the engine from the fuel tank. The engine-feed ejector pump is the main source of fuel for engine operation.

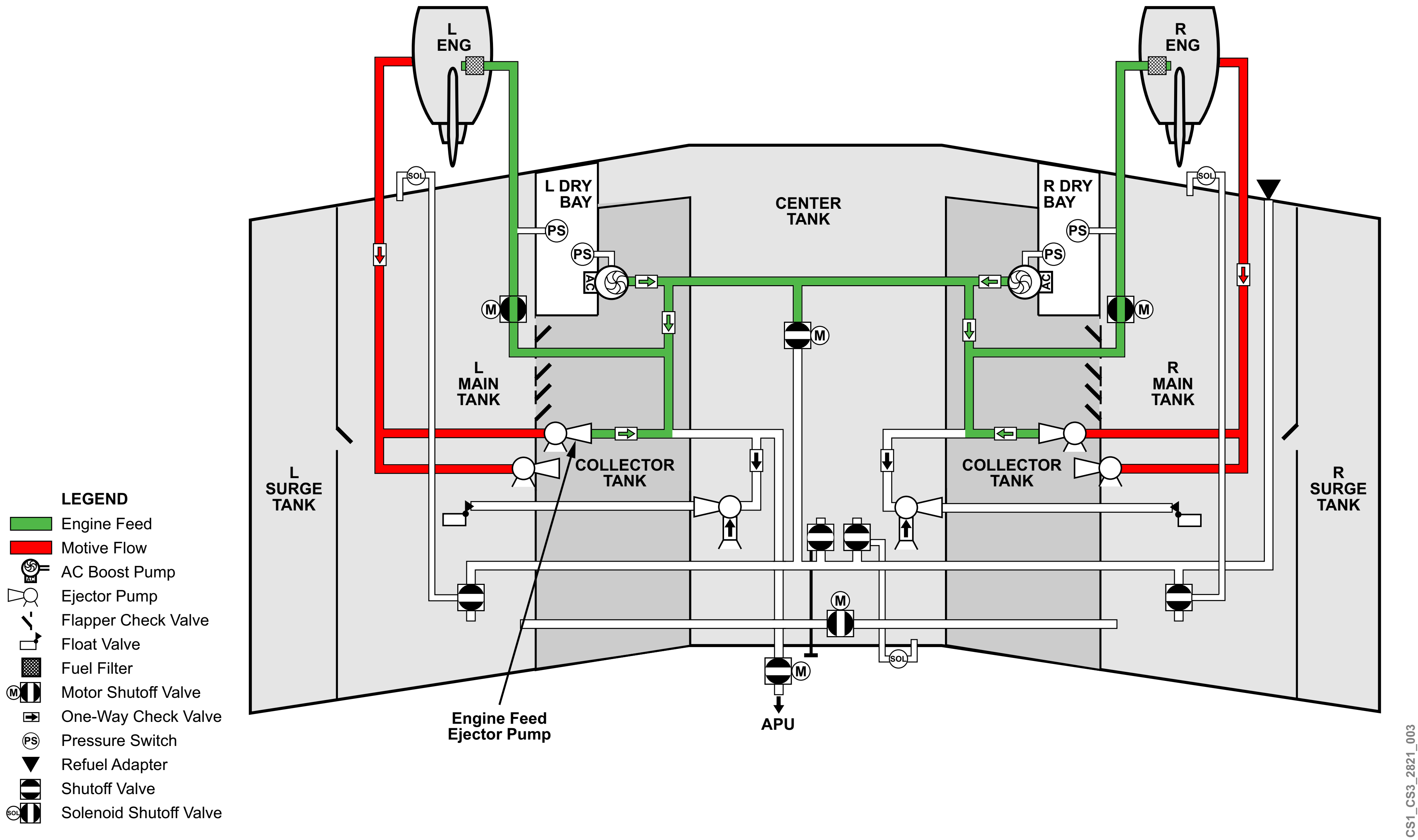


Figure 35: Engine Feed System

COMPONENT LOCATION

The following components are installed in the engine fuel feed system:

- Engine feed ejector pump
- AC boost pump
- Engine feed shutoff valve and actuator
- Engine feed pressure switch

ENGINE FEED EJECTOR PUMP

The engine feed ejector pump is located in the collector tank outboard of RIB 5.

AC BOOST PUMP

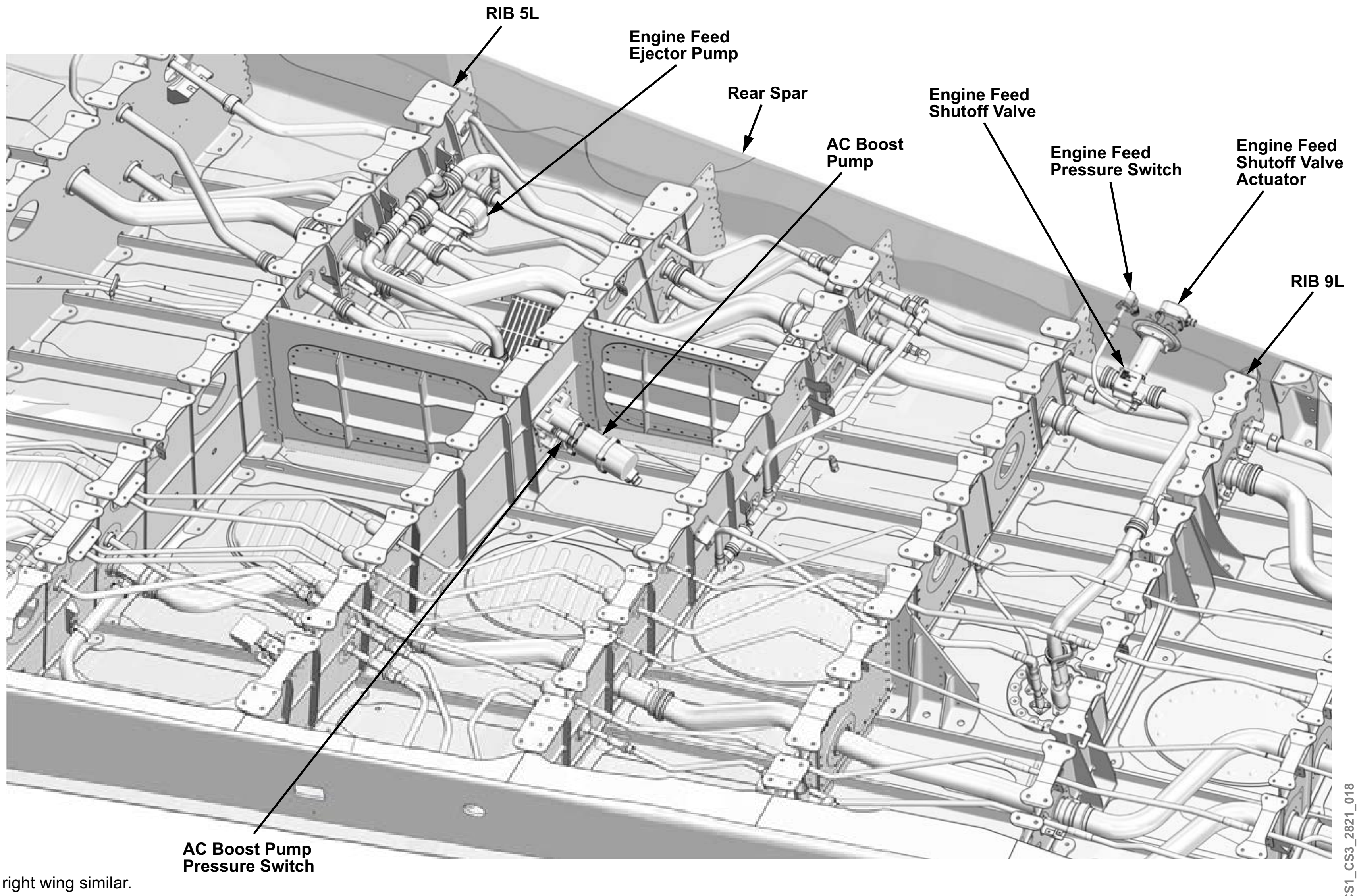
An AC boost pump is mounted on RIB 6 in each collector tank.

ENGINE FEED SHUTOFF VALVE AND ACTUATOR

The engine feed shutoff valve and actuator are installed between RIB 8 and RIB 9.

ENGINE FEED PRESSURE SWITCH

The engine feed pressure switch is mounted on the rear spar, near RIB 8.



NOTE

Left wing shown, right wing similar.

Figure 36: Engine Feed Component Location

COMPONENT INFORMATION

AC BOOST PUMP

An AC boost pump is mounted on RIB 6 in each collector tank. The AC boost pump consists of a cannister and cartridge elements.

The cannister, which contains an impeller, and suction and outlet ports, is located inside the collector tank. The suction port is connected to a pickup fitted with an inlet strainer. The pump outlet is connected to the engine feed lines.

The cartridge element is mounted in the dry bay. The cartridge element has a 115 VAC, variable frequency, 3-phase motor and pressure switch. The motor drives the impeller and the pressure switch monitors the pump discharge pressure.

The cannister has a built-in check valve that allows the cartridge to be removed without draining the fuel tank.

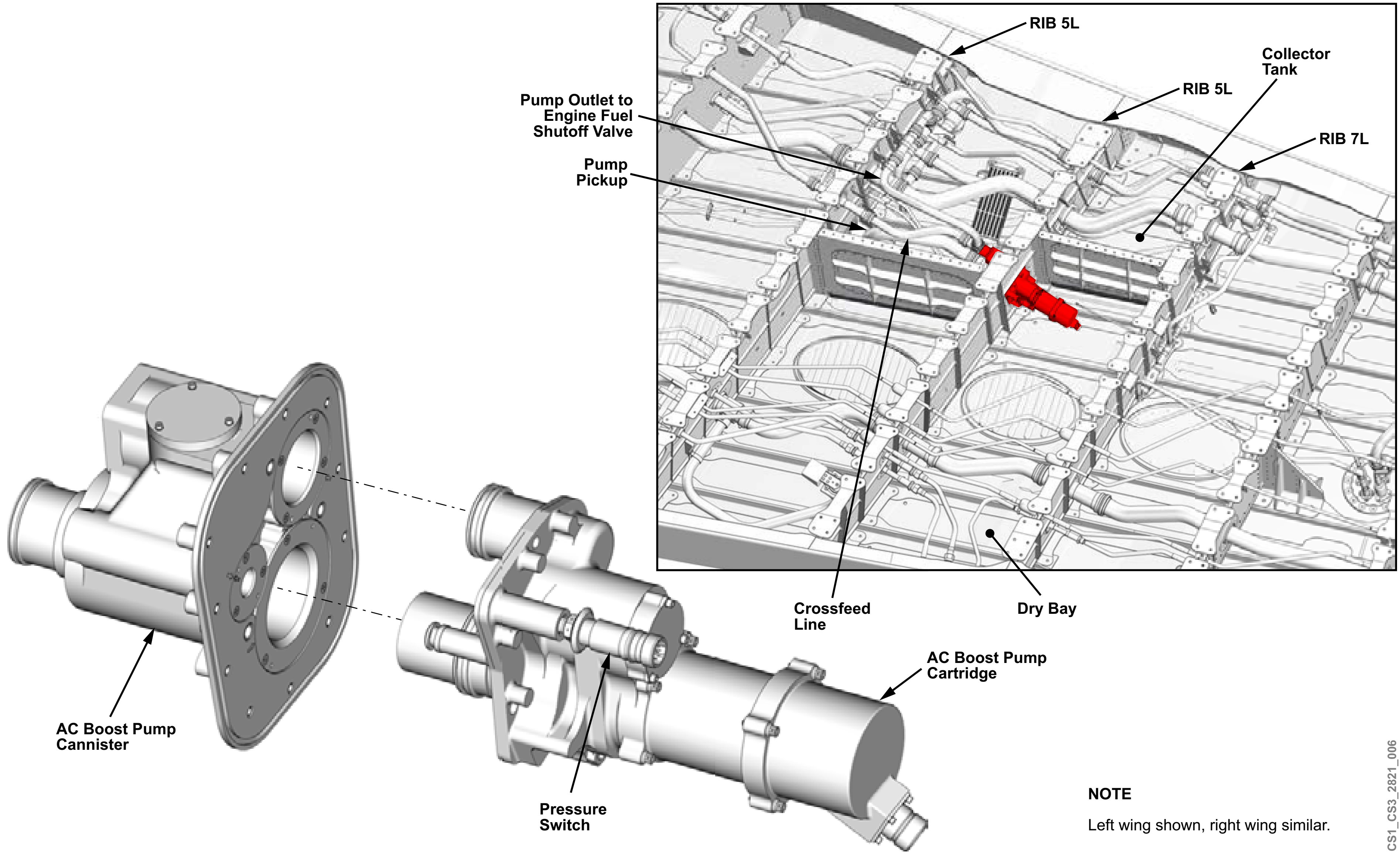


Figure 37: AC Boost Pump

CONTROLS AND INDICATIONS

ENGINE FIRE PBA

The L or R ENG FIRE PBA closes the engine fuel shutoff valve.

FUEL PANEL

The AC L or R BOOST PUMP rotary switches are on the FUEL panel. The FUEL panel is located on the overhead panel.

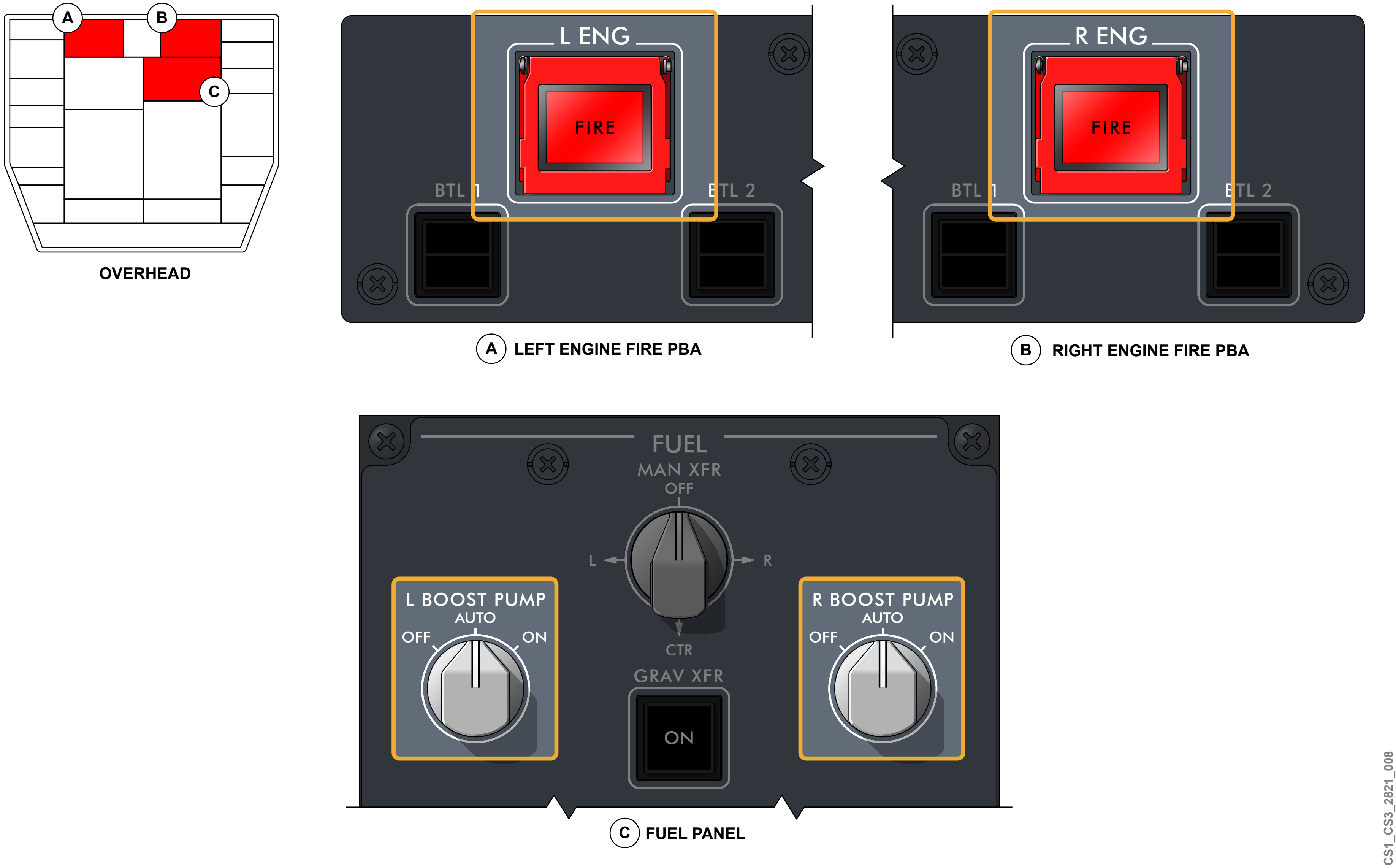
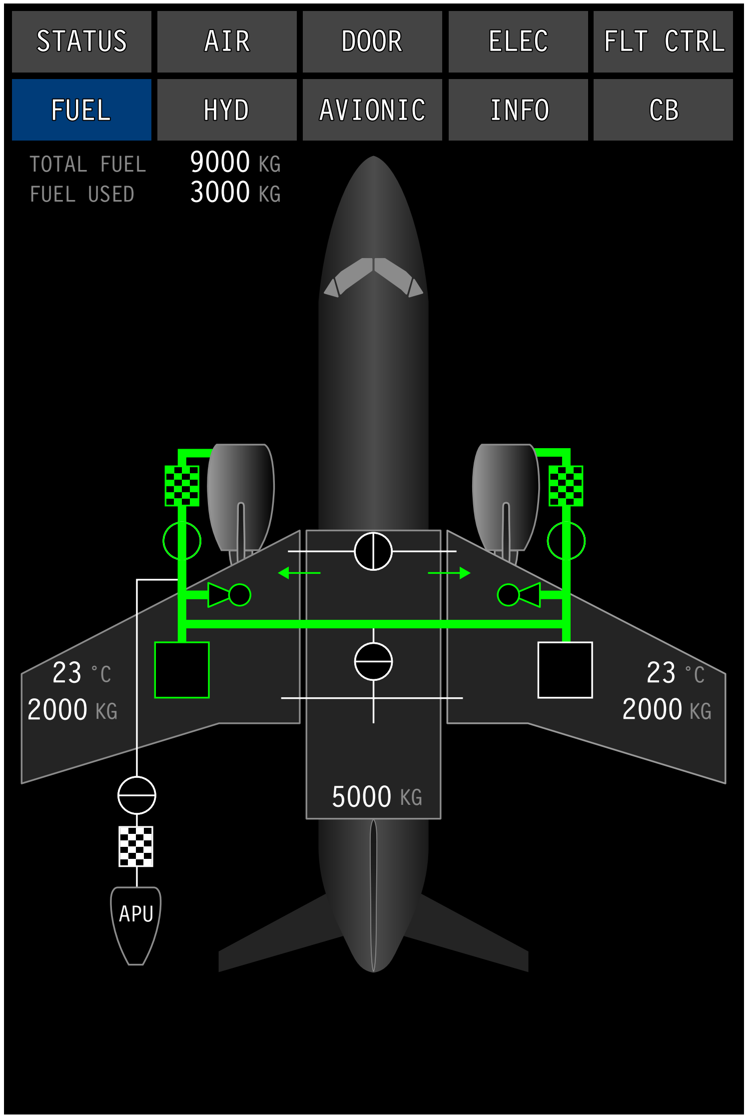


Figure 38: Engine Fire PBAs and Engine Fuel Panel

SYNOPTIC DISPLAY

The engine fuel feed can be monitored on the FUEL synoptic page. The filter shown on the display is the main engine fuel filter and is covered in ATA 73.



SYNOPTIC PAGE - FUEL

VALVE POSITION NORMAL OPERATION	
Symbol	**Condition**
	Open
	Closed
	In Transition
	Invalid

VALVE POSITION FAIL OPERATION	
Symbol	**Condition**
	Open
	Closed
	In Transition
	Invalid

PUMPS	
Symbol	**Condition**
	Active
	Off
	Failed
	Invalid

ENGINE FILTER	
Symbol	**Condition**
	Engine ON - Normal
	Engine OFF - Normal
	Engine ON or OFF - Degraded
	Invalid
	Engine ON - Bypass

CS1_CS3_2821_009

Figure 39: Fuel Synoptic Page

DETAILED DESCRIPTION

LEFT ENGINE FEED

The left AC boost pump can be turned on automatically or manually using the L BOOST PUMP switch on the fuel panel. The ON or OFF positions provide manual control. When the switch is in the AUTO position, the AC boost pump is controlled by the logic processed in the control and distribution cabinet (CDC).

The left AC boost pump is turned on automatically when:

- APU operation on the ground
- Fuel imbalance requiring transfer to the right wing
- Left engine start in flight
- Low-fuel pressure at either engine

The pump turns off automatically once the engine is running and the ejector pump is supplying pressure.

The left AC boost pump is powered through the essential bus contactor located in the electronic power center (EPC) 3. If AC electrical power is lost, the ram air turbine (RAT) powers the AC ESS BUS, allowing the AC boost pump to operate.

If low-fuel pressure is detected by the engine fuel pressure switch, both AC boost pumps are automatically turned on. The dual-contact switch provides logic to the CDC. It also provides status information to the data concentrator unit module cabinet (DMC).

The AC boost pump pressure switch monitors the pump output and reports to the DMC.

The L ENG FIRE PBA closes the engine fuel shutoff valve (FSOV) when pressed.

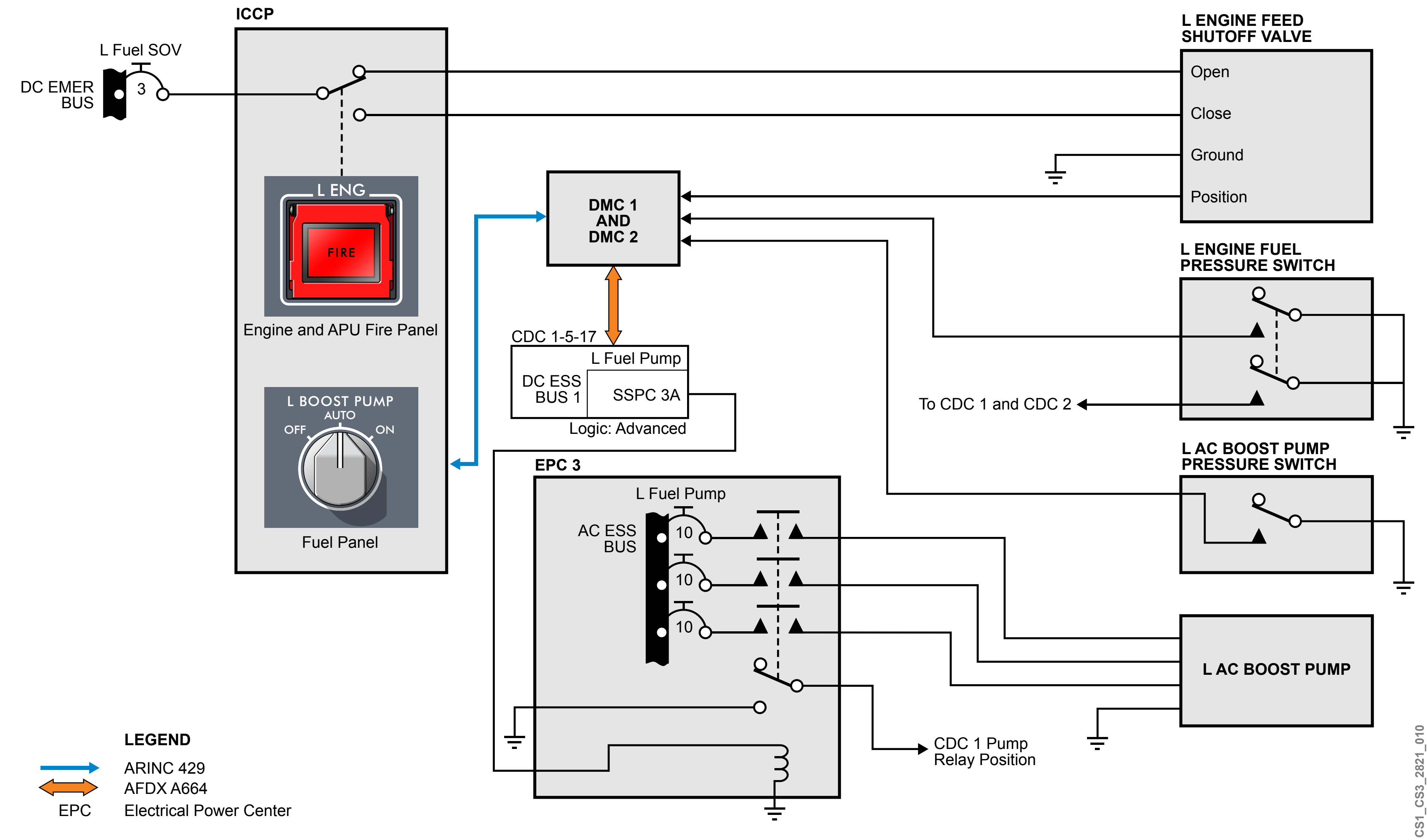


Figure 40: Left Engine Feed Detailed Description

RIGHT ENGINE FEED

The right AC boost pump can be turned on automatically or manually using the R BOOST PUMP switch on the fuel panel. The ON or OFF positions provide manual control. When the switch is in the AUTO position, the AC boost pump is controlled by the logic processed in the CDC.

The right AC boost pump is turned on automatically when:

- Fuel imbalance requiring transfer to the left wing
- Right engine start in flight
- Low-fuel pressure at either engine

The pump turns off automatically once the engine is running and the ejector pump is supplying pressure.

The right AC boost pump is powered by AC BUS 2.

If low-fuel pressure is detected by the engine fuel pressure switch, both AC boost pumps are automatically turned on. The dual-contact switch provides logic to the CDC. It also provides status information to the DMC.

The AC boost pump pressure switch monitors the pump output and reports to the DMC.

The R ENG FIRE PBA closes the engine FSOV when pressed.

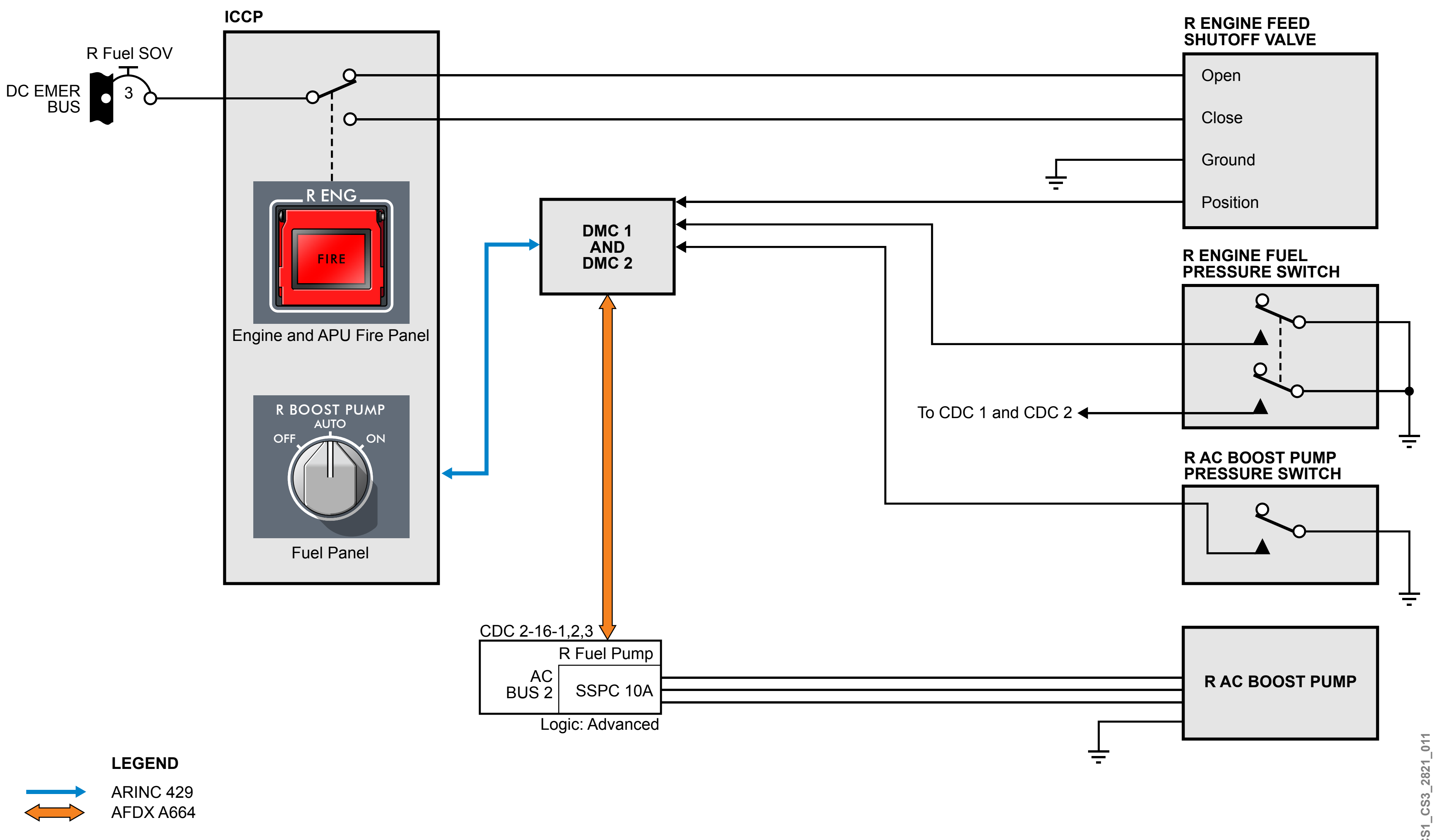


Figure 41: Right Engine Feed Detailed Description

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the engine feed system.

CAS MESSAGES

Table 1: CAUTION Messages

MESSAGE	LOGIC
L ENG FUEL SOV FAIL	Left fuel SOV failed to close/open properly.
R ENG FUEL SOV FAIL	Right fuel SOV failed to close/open properly.
L ENG FUEL LO PRESS	Left engine feed pressure is low.
R ENG FUEL LO PRESS	Right engine feed pressure is low.

Table 2: ADVISORY Messages

MESSAGE	LOGIC
L BOOST PUMP FAIL	L boost pump failed to indicate high-pressure when turned on manually or automatically.
R BOOST PUMP FAIL	R boost pump failed to indicate high-pressure when turned on manually or automatically.
L FUEL EJECTOR FAIL	Left engine motive flow is faulty.
R FUEL EJECTOR FAIL	Right engine motive flow is faulty.
L ENG FUEL SOV CLSD	Left FUEL SOV is closed after the left engine fire switch is pressed.
R ENG FUEL SOV CLSD	Right FUEL SOV is closed after the right engine fire switch is pressed.

Table 3: STATUS Messages

MESSAGE	LOGIC
L BOOST PUMP ON	L boost pump is turned on via the control panel.
R BOOST PUMP ON	R boost pump is turned on via the control panel.
L BOOST PUMP OFF	L boost pump is turned off via the control panel.
R BOOST PUMP OFF	R boost pump is turned off via the control panel.

Table 4: INFO Messages

MESSAGE	LOGIC
28 FUEL FAULT - ENG INLET PRESS SW INOP	Engine inlet pressure switch fault is detected. This may be triggered when the engine pressure switch is not high when the fuel pump is on and providing high pressure. This may be triggered when there is a disagreement between the state of the fuel engine inlet pressure switch state as monitored by the CDCs and as monitored by the DMCs.
28 FUEL FAULT - BOOST PUMP PRESS SW FAIL HI	Left and/or right boost pump pressure switch failed at high-pressure.

28-21 APU FEED SYSTEM

GENERAL DESCRIPTION

The auxiliary power unit (APU) feed shutoff valve (SOV) isolates the APU from the fuel system. The APU feed shutoff valve opens when the APU switch is in the RUN or START position. The APU feed SOV closes when:

- Commanded OFF from the flight deck
- APU protective shutdown occurs
- APU fire pushbutton annunciator (PBA) is pressed

The APU is fed from the left hand engine feed line via an APU feed SOV mounted aft of the center fuel tank. The APU feed SOV is operated by a 28 VDC actuator. The APU feed SOV isolates the APU from the fuel system after APU shutdown or when commanded OFF from the flight deck.

During APU ground operation, the left AC boost pump runs to provide APU feed. The arrangement of the isolation check valves between the main ejector pumps means that only the left ejector pump provides fuel to the APU while either AC pump can feed the APU.

If the APU fuel burn causes an imbalance, the left and right AC pumps alternate automatically, depending on what side needs to have fuel consumed to correct imbalance.

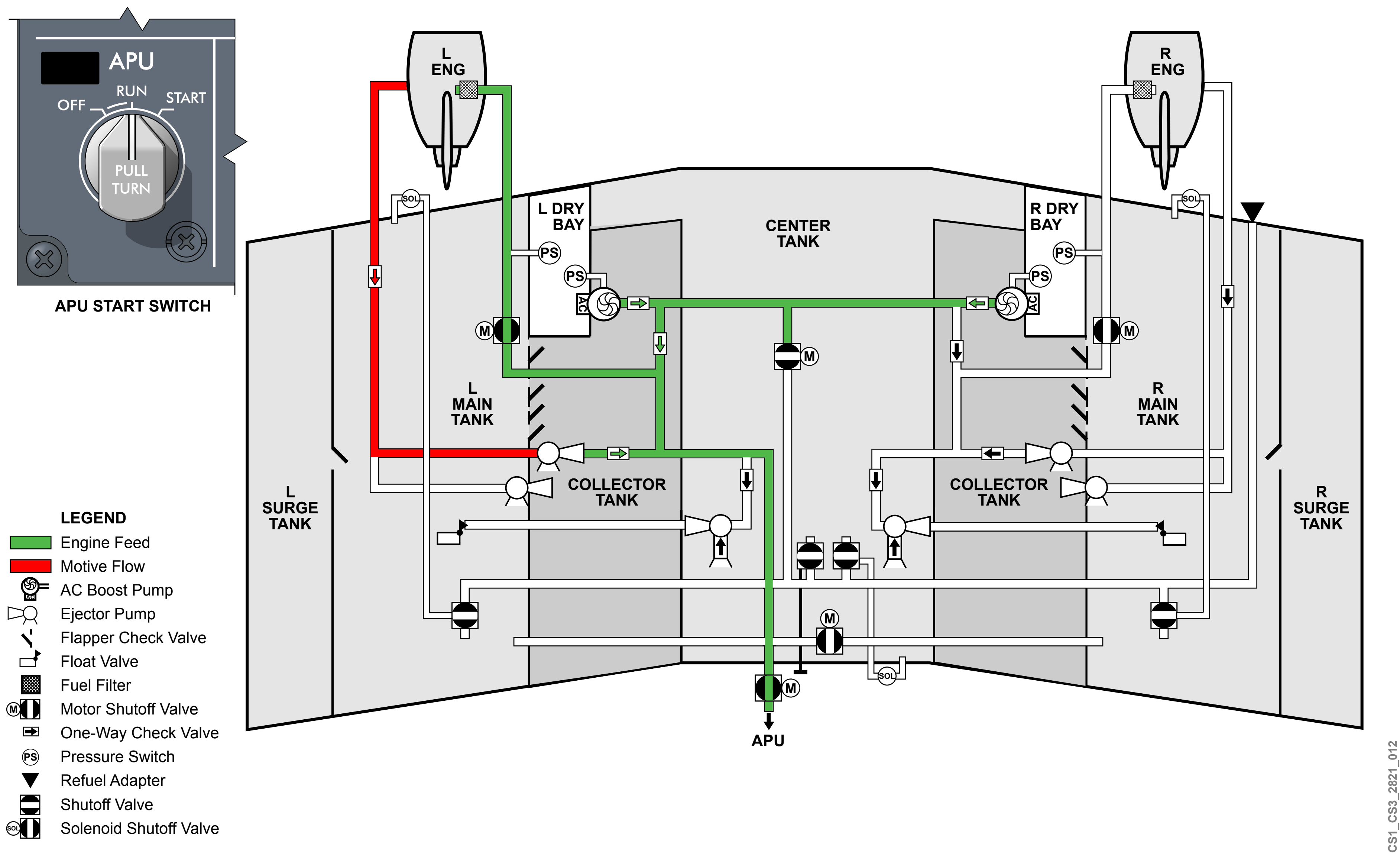


Figure 42: APU Fuel Feed System

COMPONENT LOCATION

The following components are installed in the APU fuel feed system:

- APU feed SOV and actuator
- APU fuel line shroud (refer to figure 44)

APU FEED SHUTOFF VALVE AND ACTUATOR

The actuator is mounted outside of the center fuel tank between RIB 0 and RIB 1R, and can be replaced without draining the fuel tank. The valve is located inside the fuel tank.

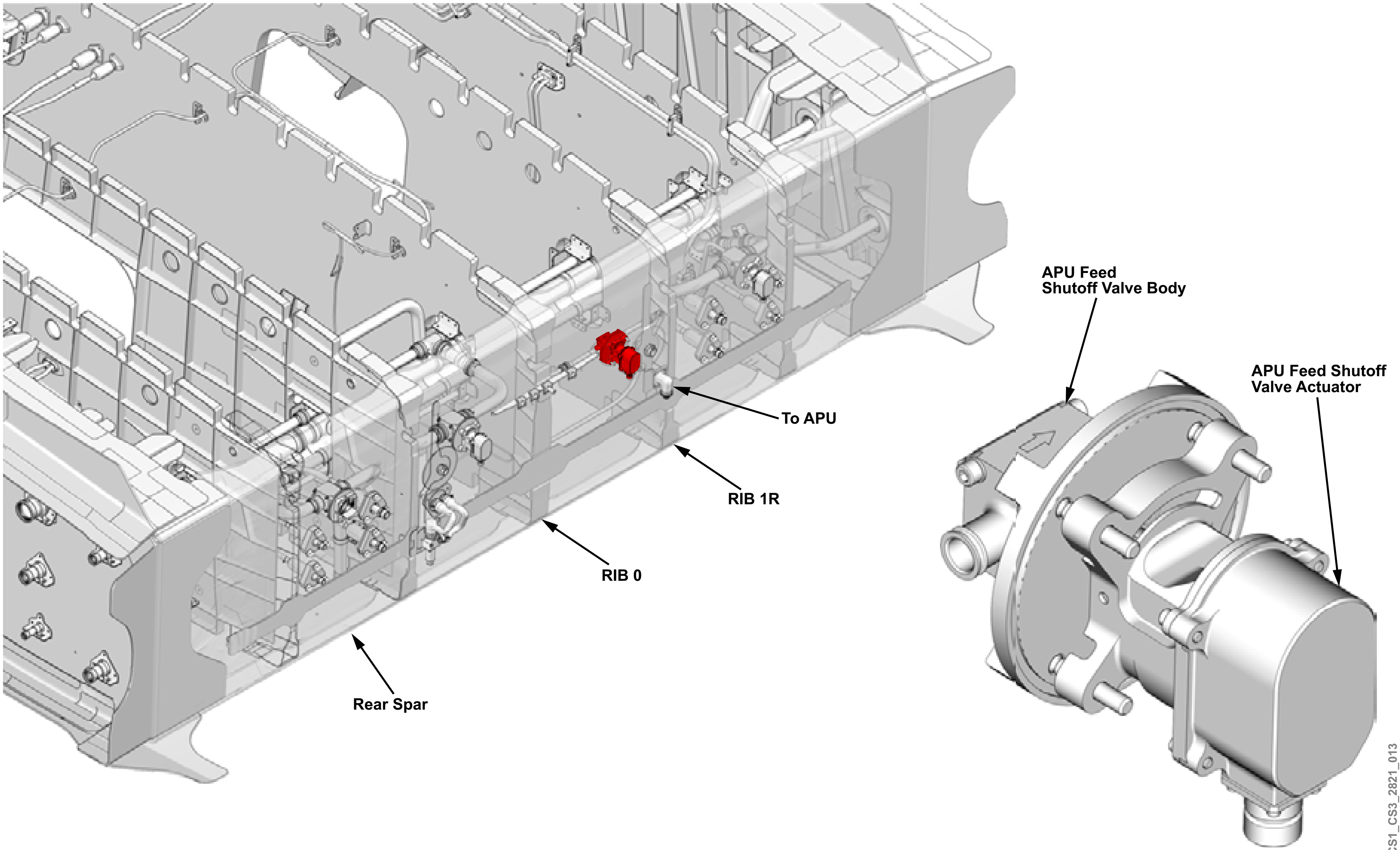


Figure 43: APU Feed Shutoff Valve and Actuator

APU FEED LINE, SHROUD, AND DRAIN MASTS

The APU feed line runs through the fuselage from the center tank rear spar to the APU. The line is shrouded to collect any fuel that might leak from the line. Any fuel leak drains into the shroud and out one of two drain masts, located on the aft fuselage and the aft right side of the wing-to-body fairing (WTBF). The center tank refuel shutoff solenoid valve also drains into the WTBF drain mast.

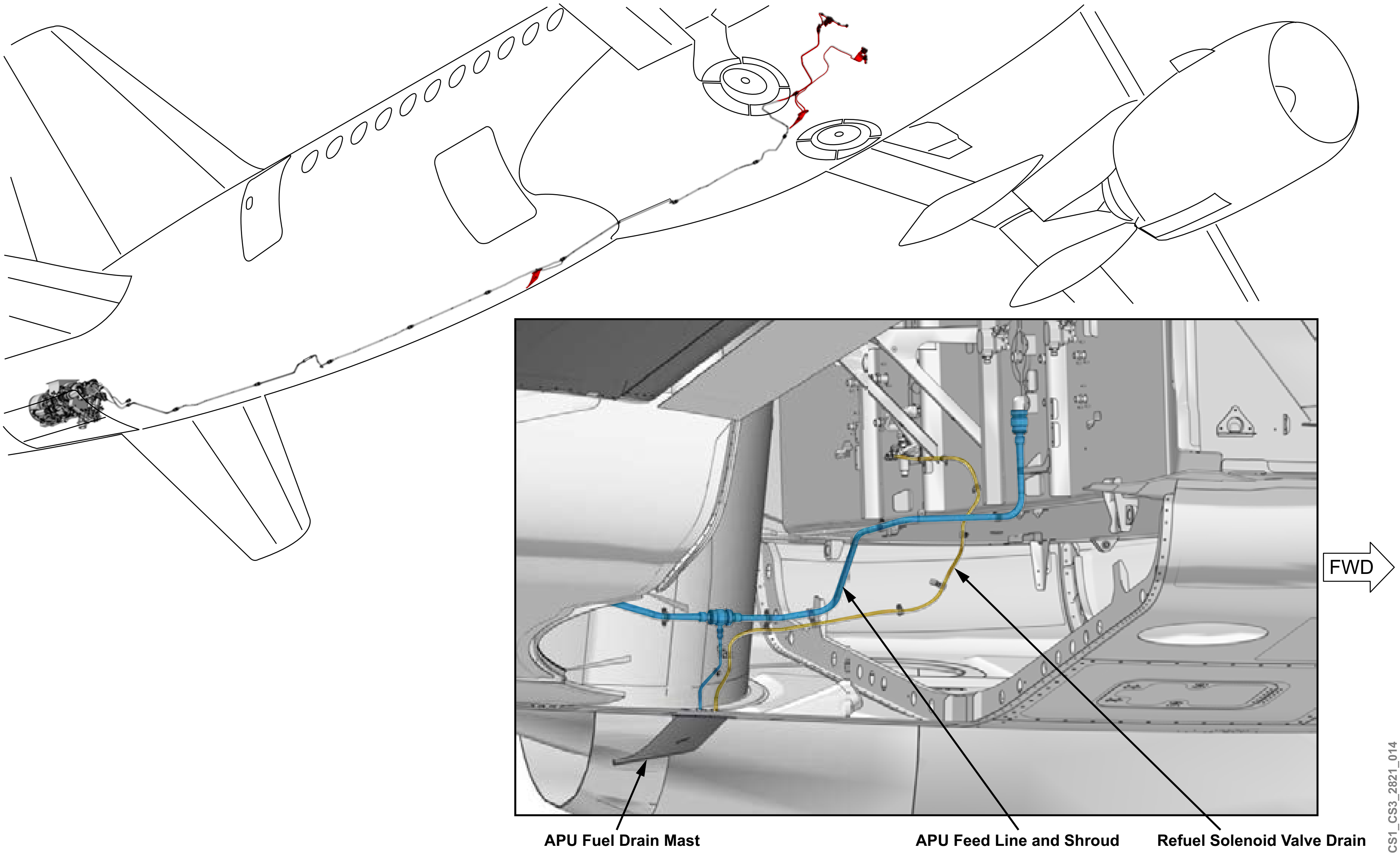


Figure 44: APU Fuel Line, Shroud and Drain Masts

CONTROLS AND INDICATIONS

APU START SWITCH

The APU feed SOV is controlled from the APU start switch located on the APU panel.

APU FIRE PBA

The APU FIRE PBA is located on the ENGINE and APU FIRE panel.

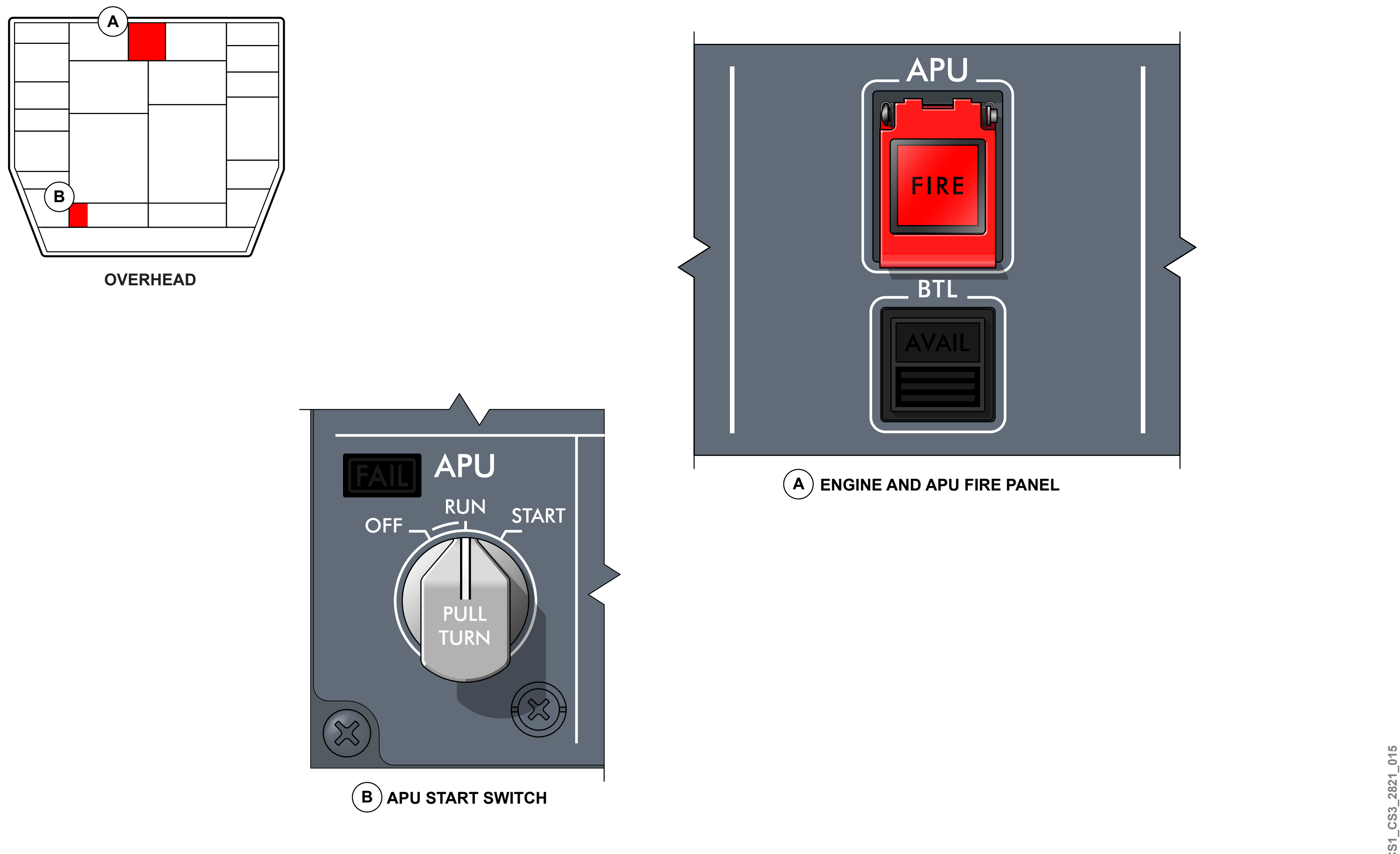


Figure 45: APU Start Switch and Fire PBA

DETAILED DESCRIPTION

The APU electronic control unit (ECU) controls the APU feed SOV through the relaxed contacts of the APU feed SOV relay located in the control and distribution cabinet 1 (CDC 1). When APU switch is in the OFF position, the APU feed SOV closes.

In the event of an APU fire, the APU feed SOV closes. The APU feed SOV relay energizes when the fire detection and extinguishing (FIDEX) control unit detects an APU fire or when the APU FIRE PBA is pressed.

Microswitches in the APU feed shutoff valve provide valve position indication on the engine indication and crew alerting system (EICAS).

The APU feed SOV can be tested through the onboard maintenance system (OMS). The APU feed SOV test is listed under ATA 49 APU tests.

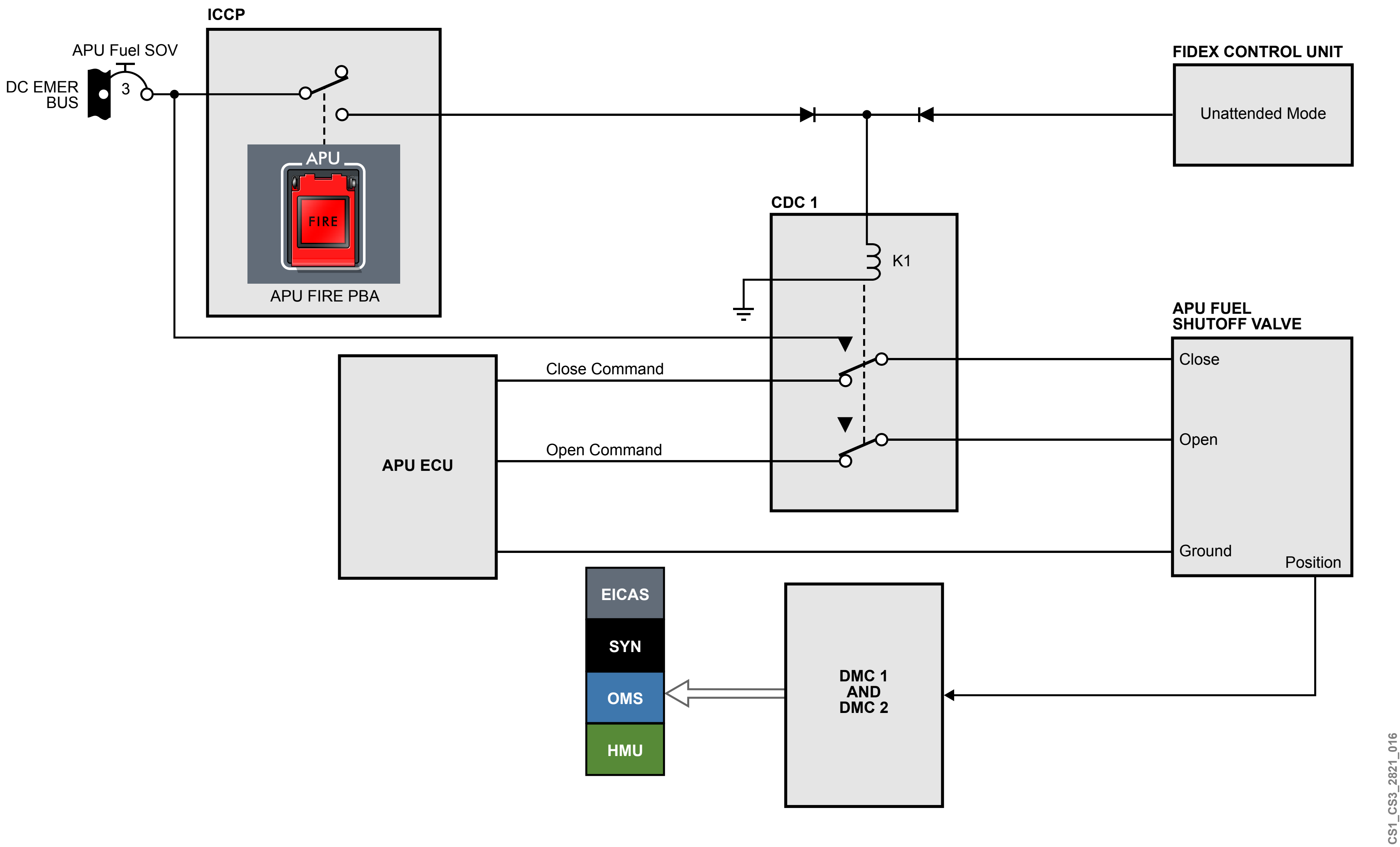


Figure 46: APU Feed Shutoff Valve Detailed Description

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) messages for the APU feed system.

CAS MESSAGES

Table 5: CAUTION Message

MESSAGE	LOGIC
APU FUEL SOV FAIL	APU fuel SOV failed to close/open properly.

Table 6: ADVISORY Message

MESSAGE	LOGIC
APU FUEL SOV CLSD	APU FUEL SOV is closed after the APU fire switch is pressed.

28-22 FUEL TRANSFER SYSTEM

GENERAL DESCRIPTION

The fuel transfer system has the following subsystems:

- Scavenge system
- Automatic transfer
- Center tank to main tank fuel transfer
- Manual transfer
  - Main tank to main tank fuel transfer
  - Main tank to center tank fuel transfer
  - Gravity transfer

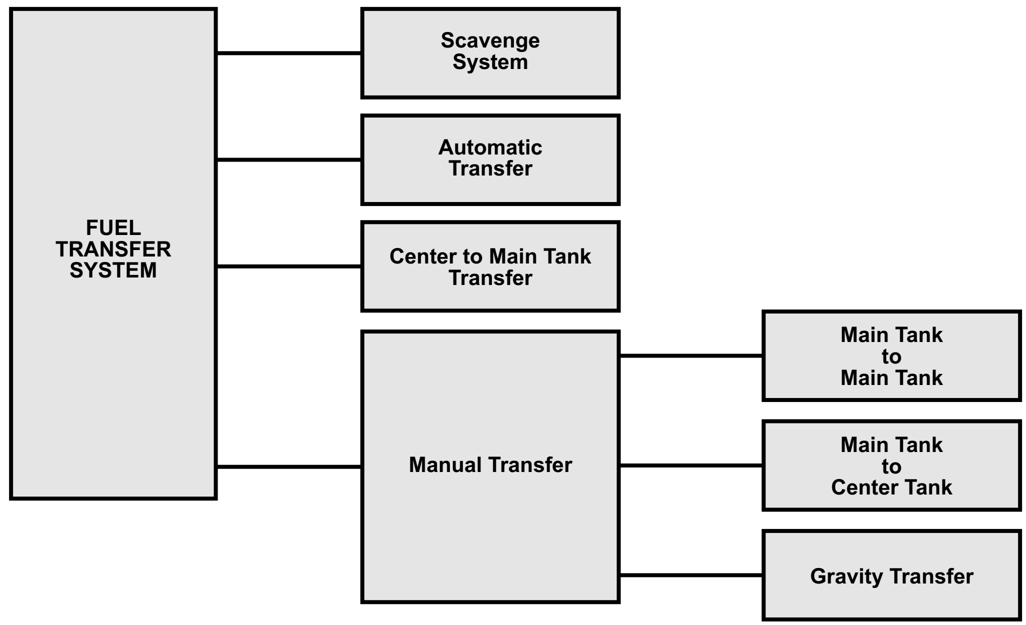


Figure 47: Fuel Transfer System

SCAVENGE SYSTEM

Scavenge pumps minimize the amount of unusable fuel in the main tanks. The engine-driven pump (EDP) provides motive flow to scavenge ejector pumps.

The pumps are located in the sump of each main tank and continuously feed the collector tanks.

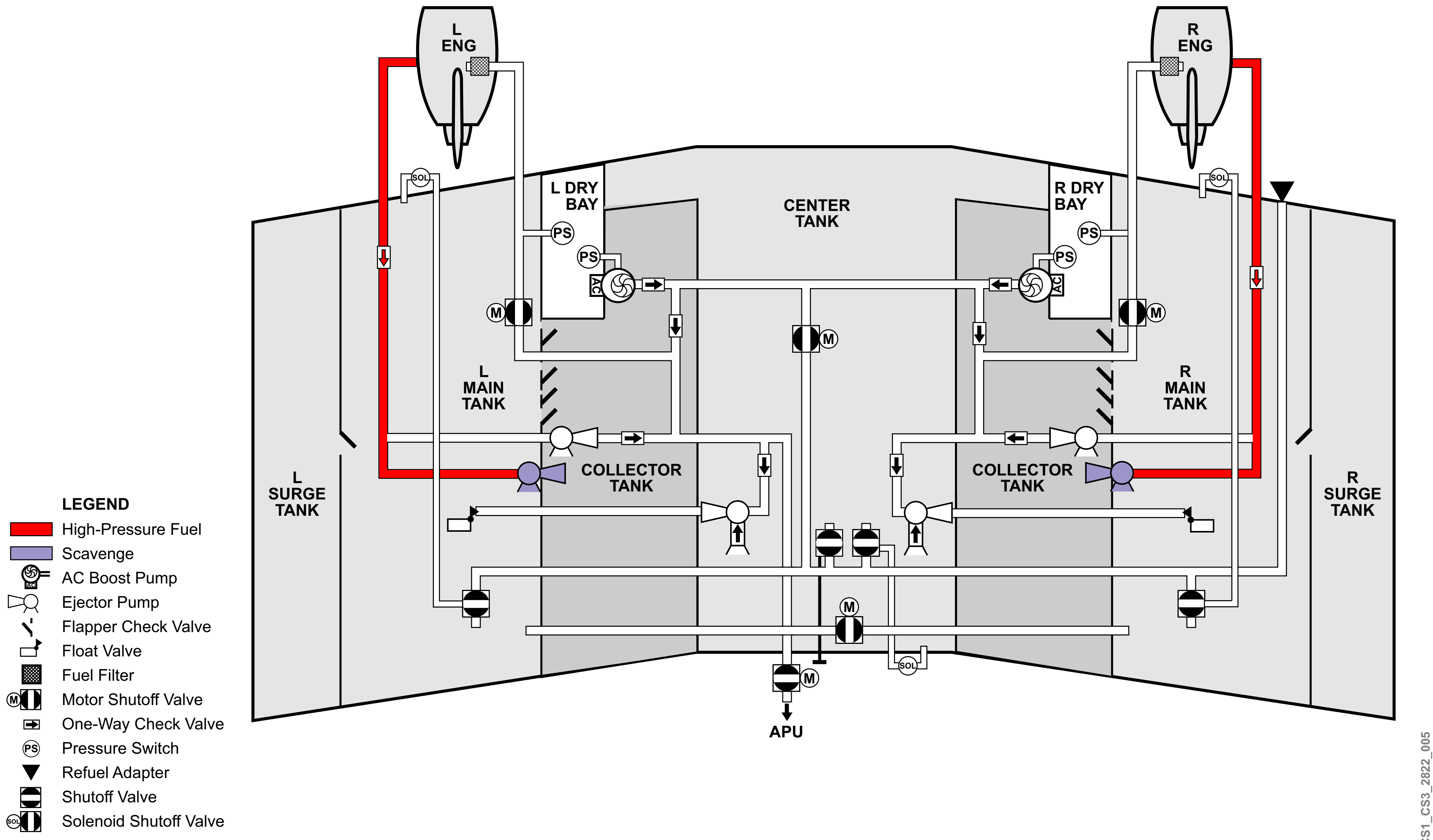


Figure 48: Scavenge System

AUTOMATIC TRANSFER

Automatic transfer corrects fuel imbalances when the MAN XFR switch on the FUEL panel is in the OFF position. Automatic transfer is accomplished through automatic control of the AC boost pumps.

If the AC boost pumps are set to AUTO and the fuel quantity computer senses an imbalance of 182 kg (400 lb) between the left and right main tanks, the AC boost pump of the heavier tank turns on automatically. Both engines receive fuel directly from the heavier tank. When the tanks are within 45 kg (100 lb) of each other, the AC boost pump is switched off.

If the fuel transfer does not correct the imbalance and the imbalance reaches 364 kg (800 lb), a FUEL IMBALANCE caution message is displayed.

The FUEL synoptic page displays the operating boost pump in green, and a white crossfeed arrow indicates the direction of the crossfeed.

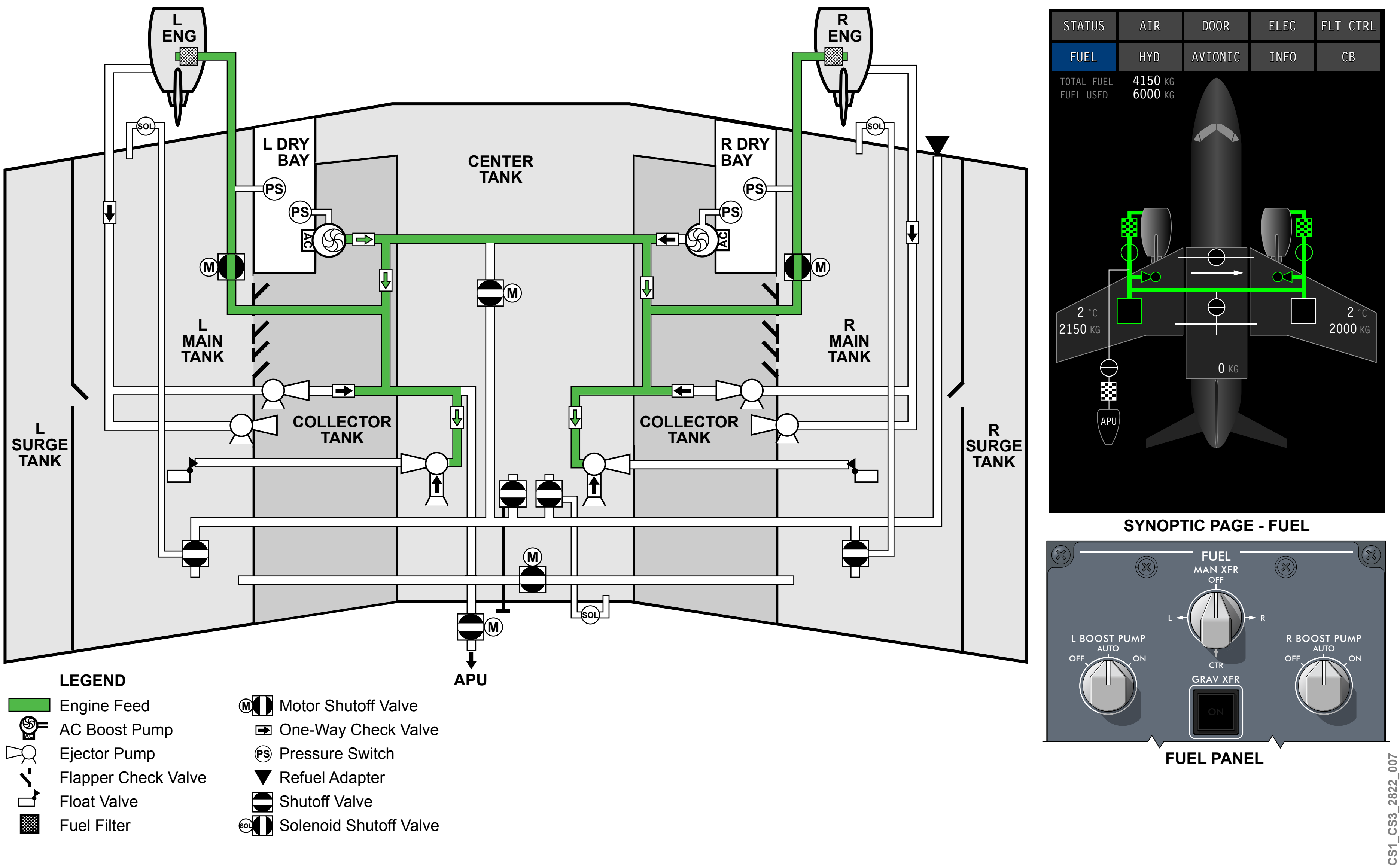


Figure 49: Automatic Fuel Transfer

CENTER TANK TO MAIN TANK FUEL TRANSFER

Fuel transfer from the center tank to each main wing tank, is controlled by its corresponding transfer float valve. The transfer float valve opens when the fuel level of the main tank drops below 86% of its capacity.

Fuel from the center tank is automatically transferred to the main tanks by transfer ejector pumps. The pump pickup is in the center fuel tank sump. Fuel flow only occurs if the fuel quantity in the main fuel tank is low enough that the associated transfer float valve opens to allow the fuel transfer. The motive flow for the pumps is provided by the engine feed ejector pump or the AC boost pump if it is running.

When the main tank level increases above 86%, the transfer float valve closes. A check valve, in the suction port of each transfer ejector prevents fuel transfer back to the center tank. This transfer operation is fully automatic and continuously replenishes the main tanks until the center tank is empty. The automatic transfer of fuel from the center tank to the wing tanks ensures that the center tank fuel is always emptied first.

The center tank fuel transfer is indicated on the synoptic page by green arrows pointing outward from the center tank.

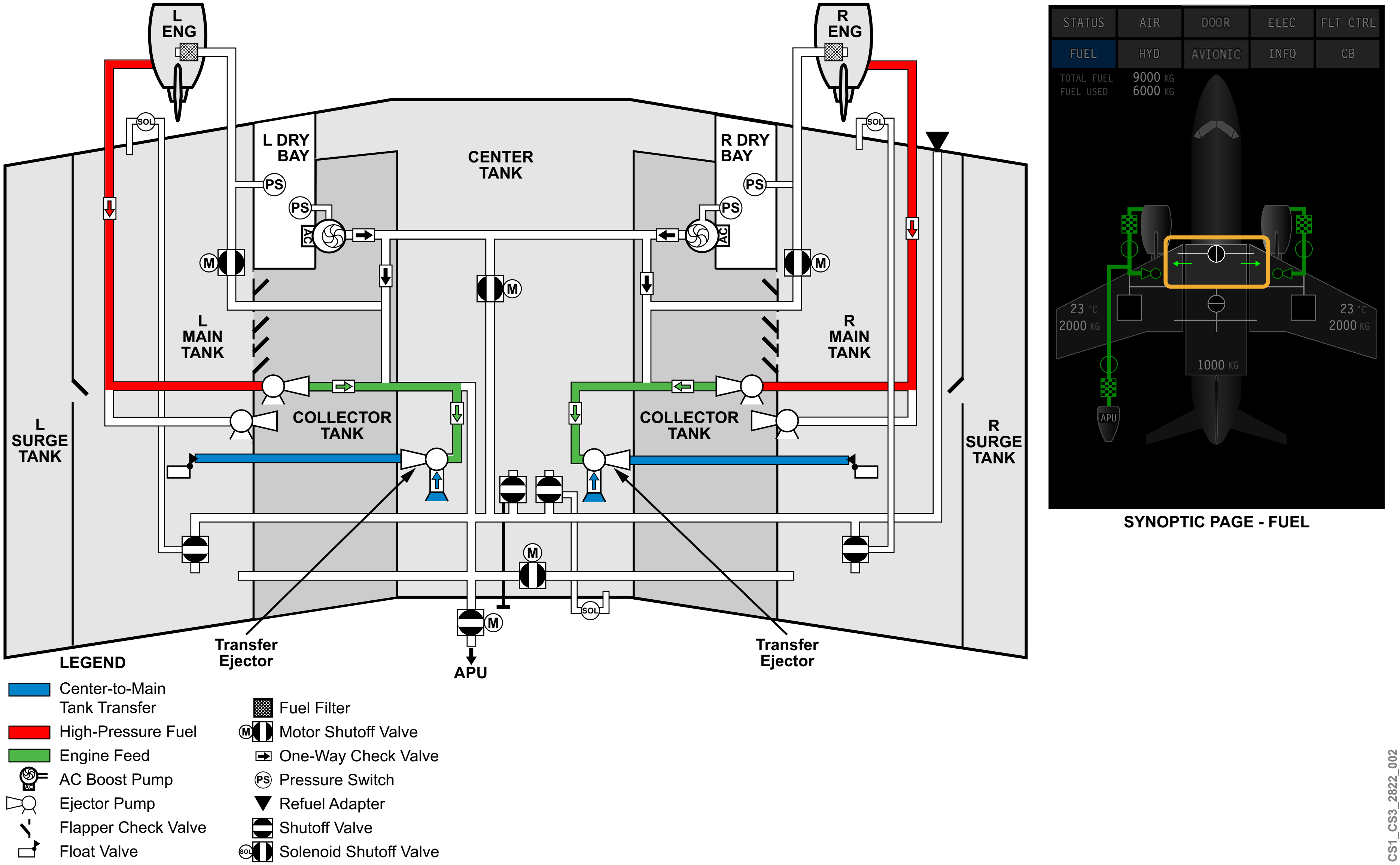


Figure 50: Center Tank to Main Tank Fuel Transfer

MANUAL FUEL TRANSFER

Fuel can also be manually transferred from wing tank to opposite wing tank or to center tank through appropriate selection of fuel switches on the integrated cockpit control panel (ICCP):

- Main tank to main tank, using FUEL MAN XFER switch
- Main tank to the center tank, using FUEL MAN XFER switch and L BOOST PUMP or R BOOST PUMP switch
- Main to main tank by gravity, using GRAV XFR PBA

Manual fuel transfer is accomplished by connecting the engine feed line to the refuel manifold using the defuel/isolation transfer valve and the refuel shutoff valves. The defuel/isolation transfer valve and refuel shutoff valves are controlled by the fuel quantity computer (FQC).

Main Tank to Main Tank Fuel Transfer

Selection of the FUEL MAN XFER switch to L or R opens the defuel/isolation transfer SOV and left or right main tank refuel SOV. When the L or R BOOST PUMP switches are in AUTO, the AC boost pump in the tank the fuel is transferred from turns on.

When 182 kg (400 lb) has been transferred, an EICAS FUEL MAN XFR COMPLETE advisory message is displayed.

The defuel/isolation transfer SOV and the refuel SOV close and the left or right boost pump turns off.

CAUTION

Do not operate the L(R) AC boost pump if the fuel quantity in the associated tank is less than 363 kg (800 lb). The boost pump can be damaged.

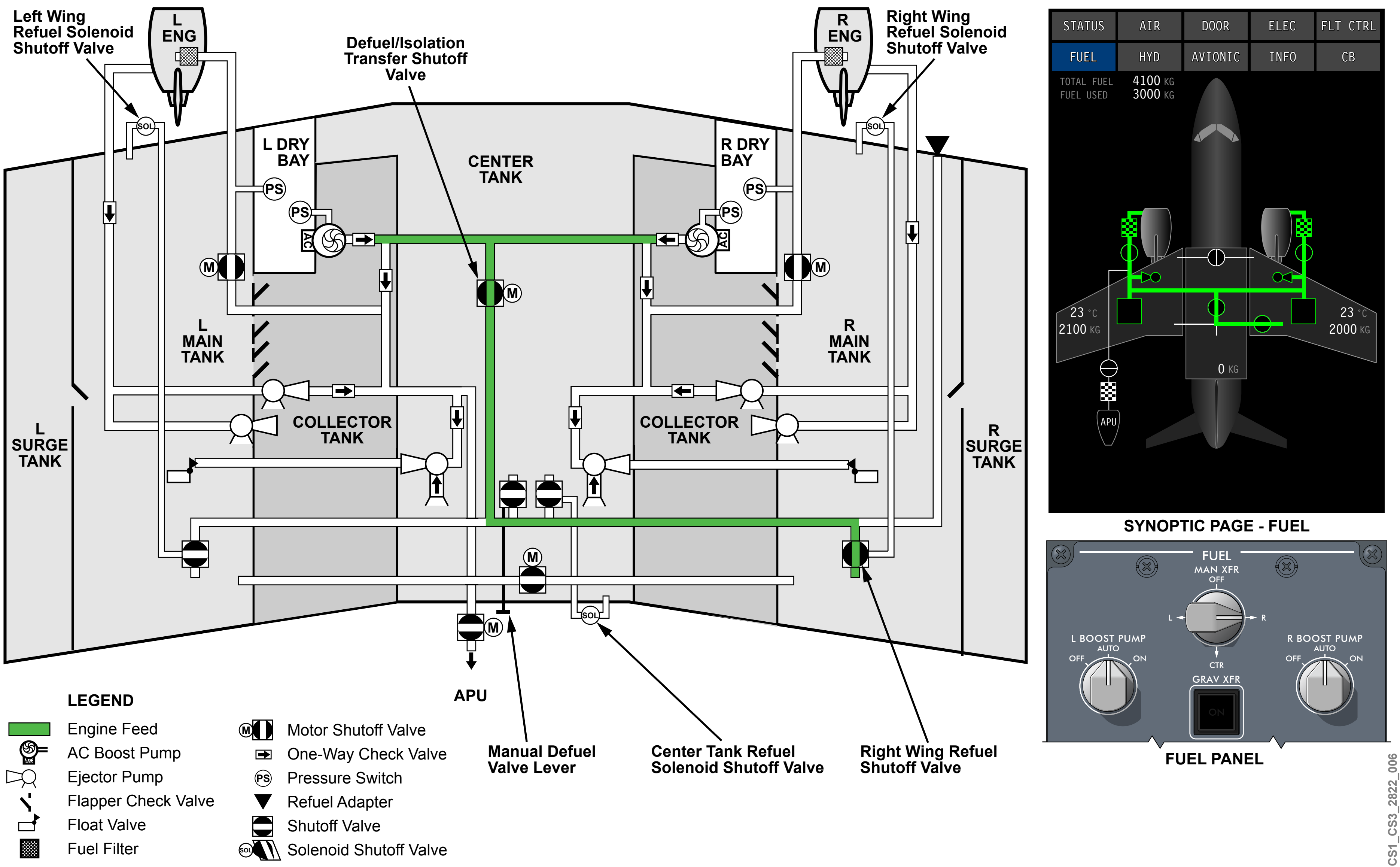


Figure 51: Main Tank to Main Tank Fuel Transfer

Main Tank to Center Tank Fuel Transfer

When the MAN XFR switch is set to CTR, the defuel/isolation transfer SOV and the center tank refuel SOV open. The left or right boost pump operation is not automatic and must be selected ON to enable fuel transfer. A FUEL XFR CTR READY status message is displayed if the pumps are not turned on.

When 364 kg (800 lb) has been transferred, an EICAS FUEL MAN XFR COMPLETE advisory message is displayed.

The defuel/isolation transfer SOV and the center tank refuel SOV close. The AC boost pump switch must be returned to the AUTO or OFF position, otherwise the pump continues to run.

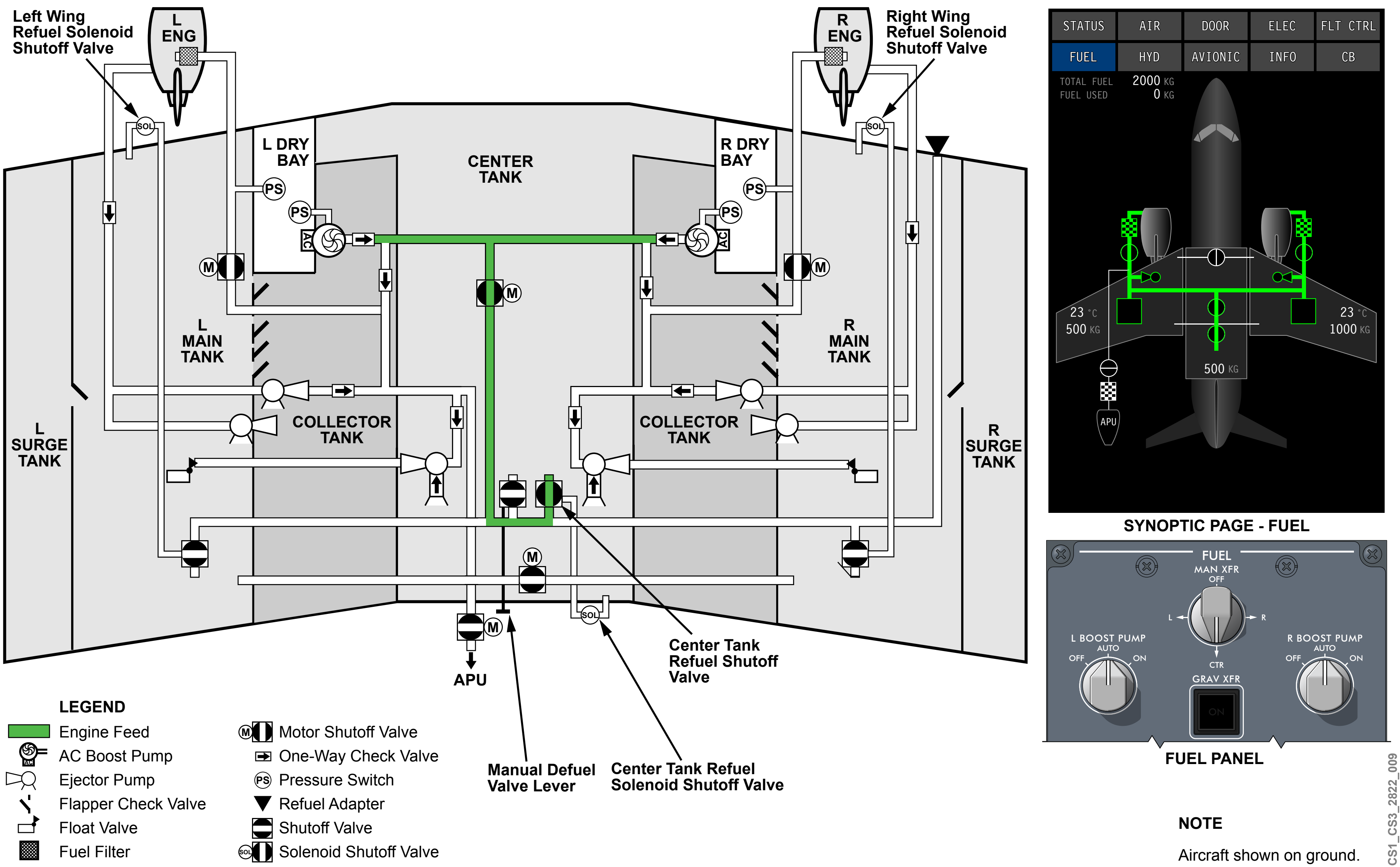


Figure 52: Main Tank to Center Tank Fuel Transfer

Gravity Transfer

The GRAV XFR PBA transfers fuel between the main tanks to correct the imbalances between the wing tanks. When the GRAV XFR PBA is pressed, it opens the gravity crossflow SOV. The gravity crossflow SOV is normally closed to provide isolation between both wing tanks. The GRAV XFR PBA is hardwired to the gravity crossflow shutoff valve.

When the SOV is open, the fuel level in the main tanks is allowed to equalize through gravity.

The gravity crossflow is displayed on the FUEL synoptic page.

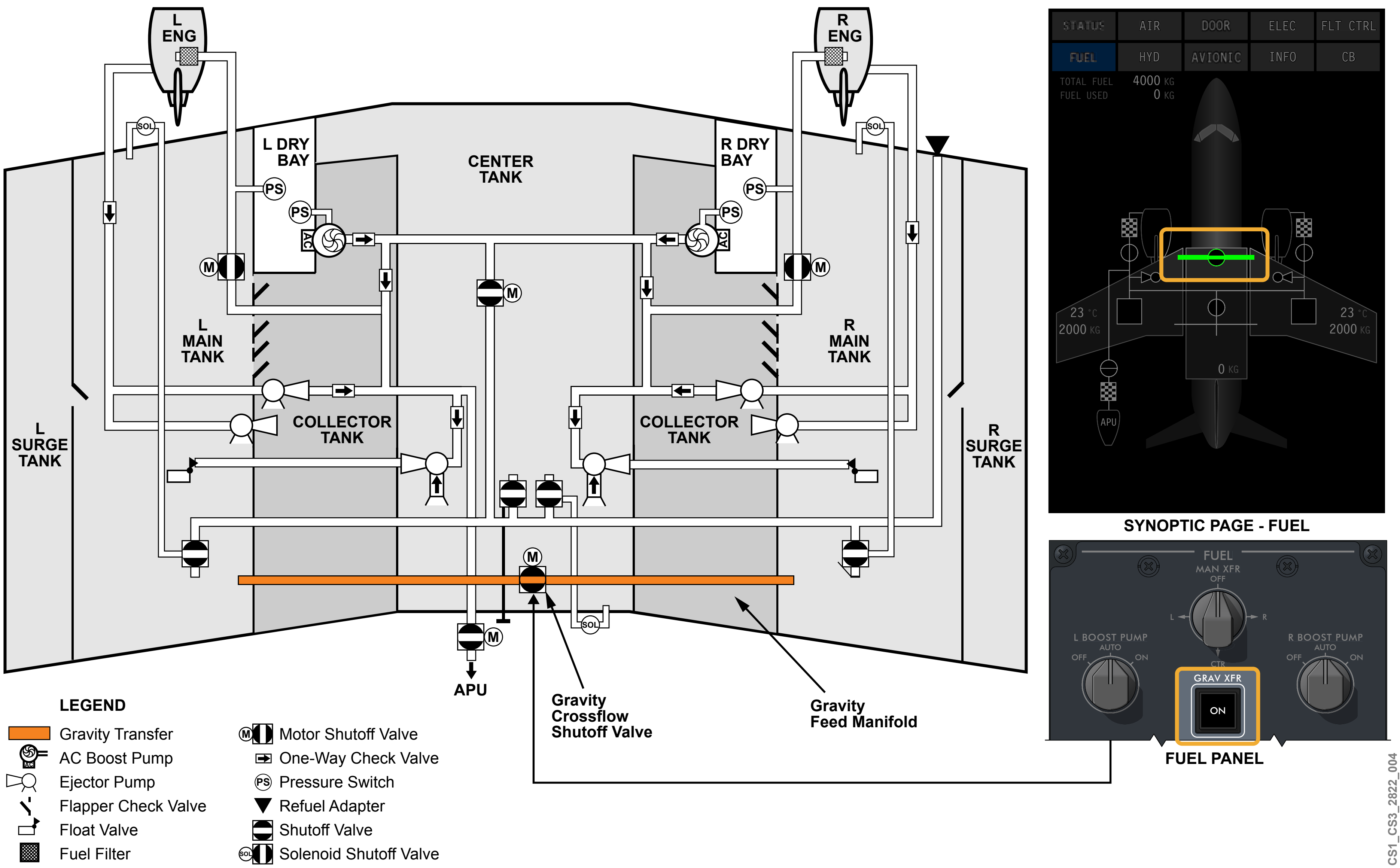


Figure 53: Gravity Fuel Transfer

COMPONENT LOCATION

The scavenge system consists of the following component:

- Scavenge ejector pump

The center tank to main tank transfer system consists of the following components:

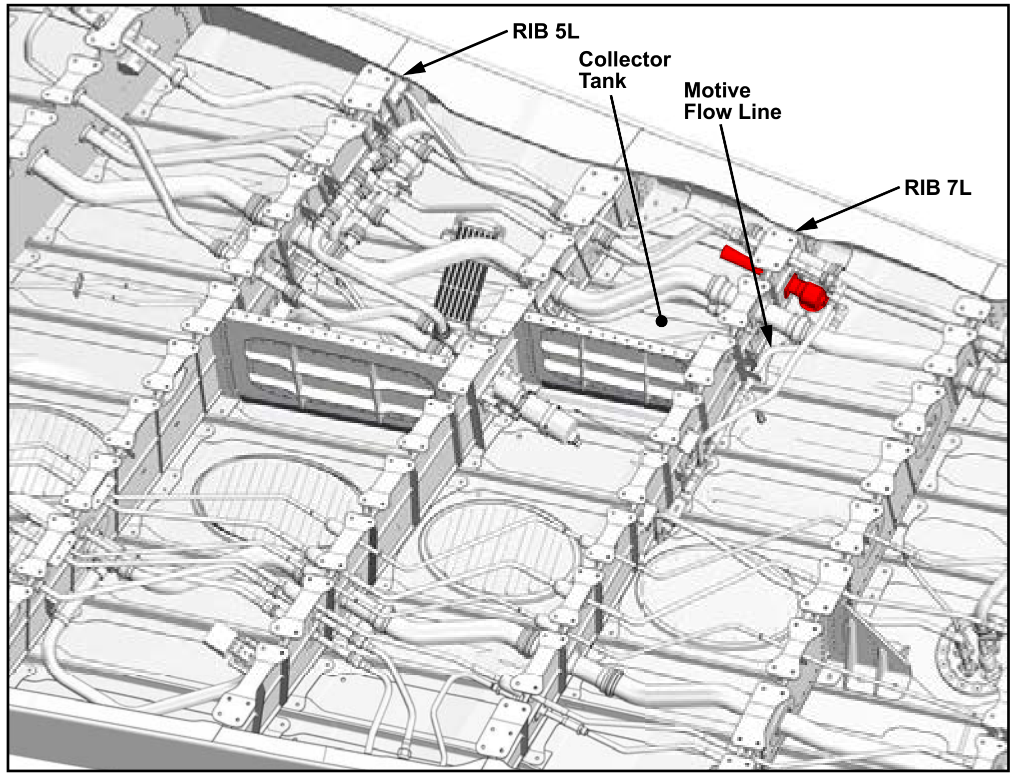
- Transfer float valve
- Transfer ejector pump

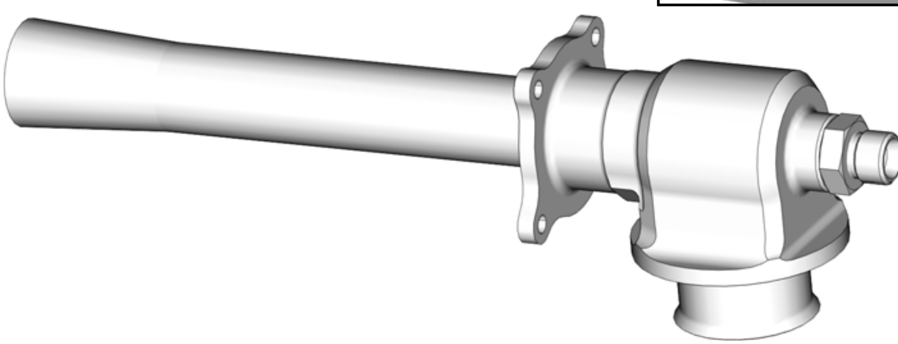
The gravity fuel transfer system consists of the following component:

- Gravity crossflow shutoff valve

SCAVENGE EJECTOR PUMP

The scavenge ejector pumps are mounted on RIB 7 of each wing tank.





SCAVENGE EJECTOR PUMP

NOTE

Left wing shown, right wing similar.

CS1_CS3_2821_004

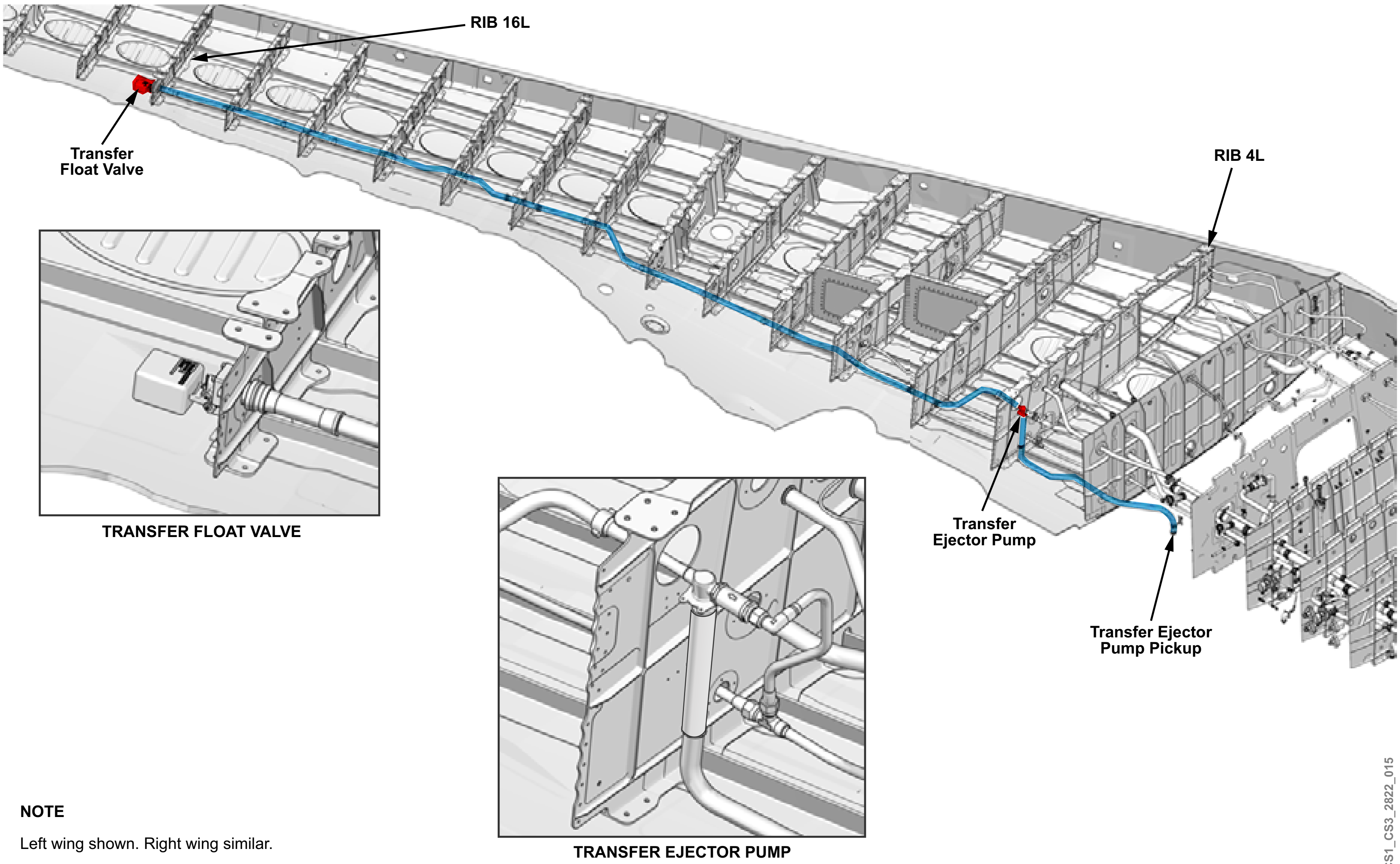
Figure 54: Scavenge Ejector Pump

TRANSFER FLOAT VALVE

This valve is attached to the center tank transfer discharge line at RIB 16.

TRANSFER EJECTOR PUMP

Each of the two transfer systems contains an ejector pump installed at RIB 4 of the center fuel tank.



NOTE

Left wing shown. Right wing similar.

Figure 55: Center Tank to Main Tank Fuel Transfer Ejector Pump and Float Valve

GRAVITY CROSSFLOW SHUTOFF VALVE

The gravity crossflow shutoff valve is located inboard of RIB 2R on the rear spar.

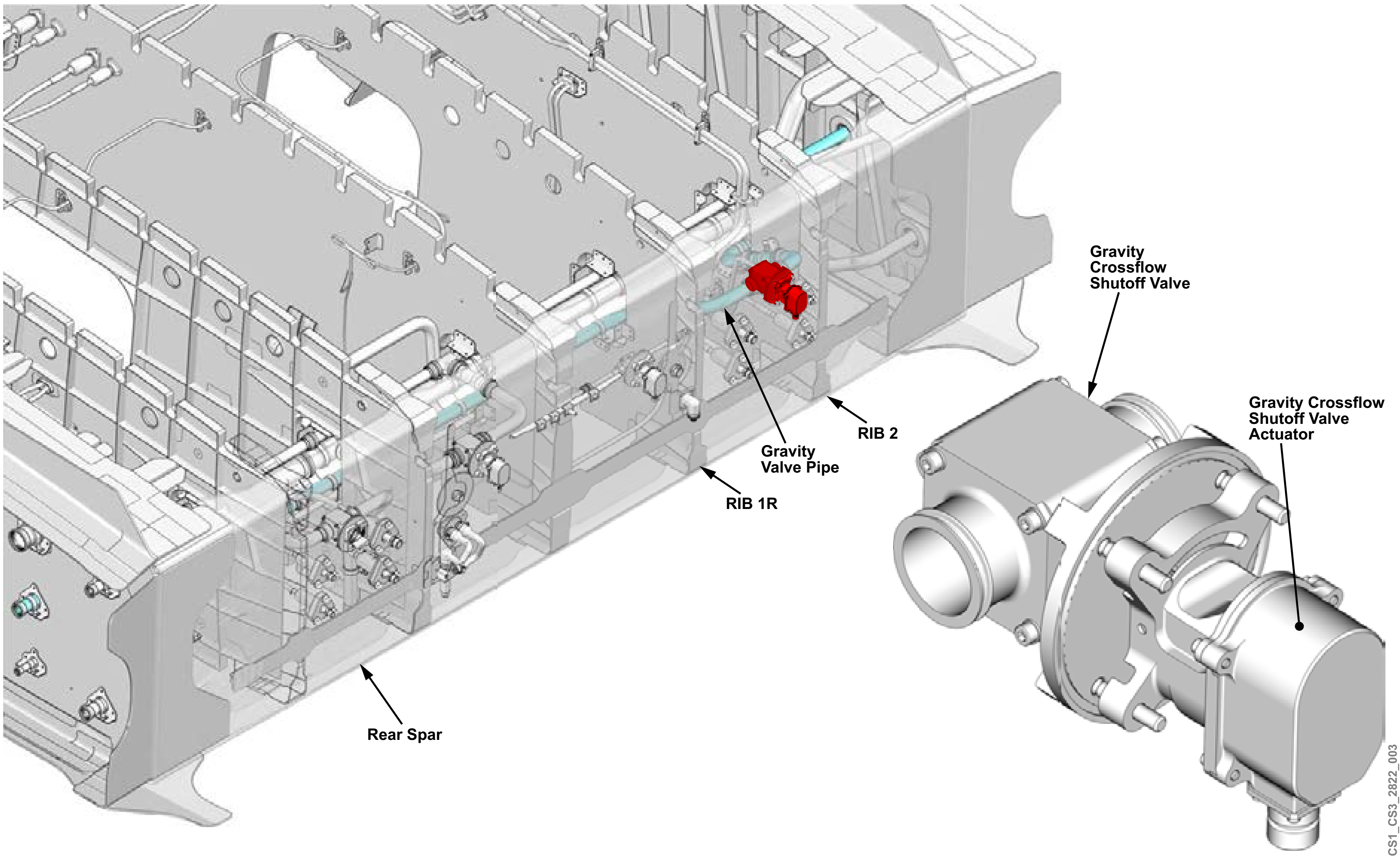


Figure 56: Gravity Crossflow Shutoff Valve

CONTROLS AND INDICATIONS

Fuel transfer is controlled from the FUEL panel on the right inboard integrated cockpit control panel (ICCP).

The MAN XFR switch is used to manually transfer fuel between the tanks. The MAN XFR switch is a four-position rotary switch located on the FUEL panel. The normal position of the switch is OFF. In the OFF position, automatic imbalance correction is enabled.

The GRAV XFR PBA opens the gravity crossflow SOV. The gravity crossflow SOV is hardwired to the valve.

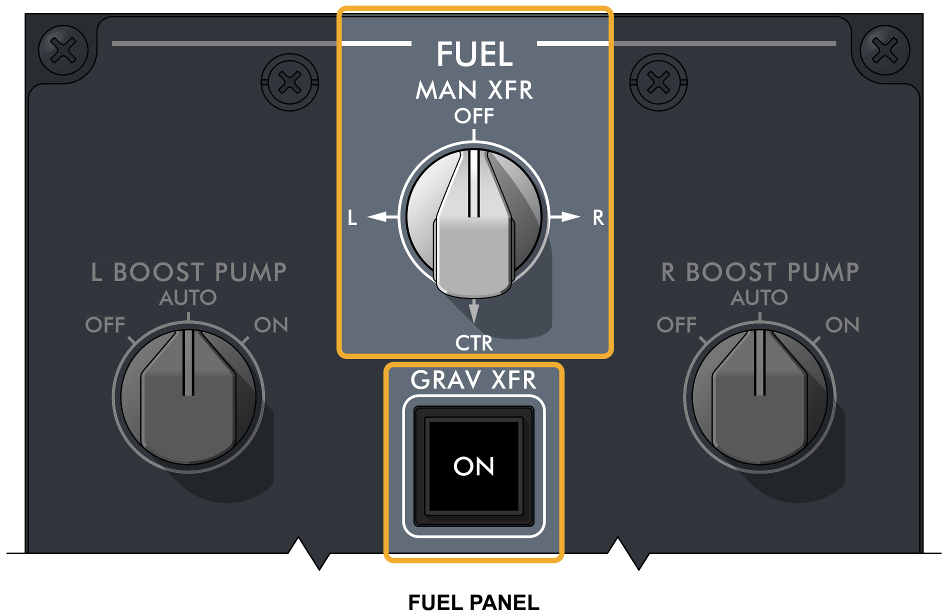
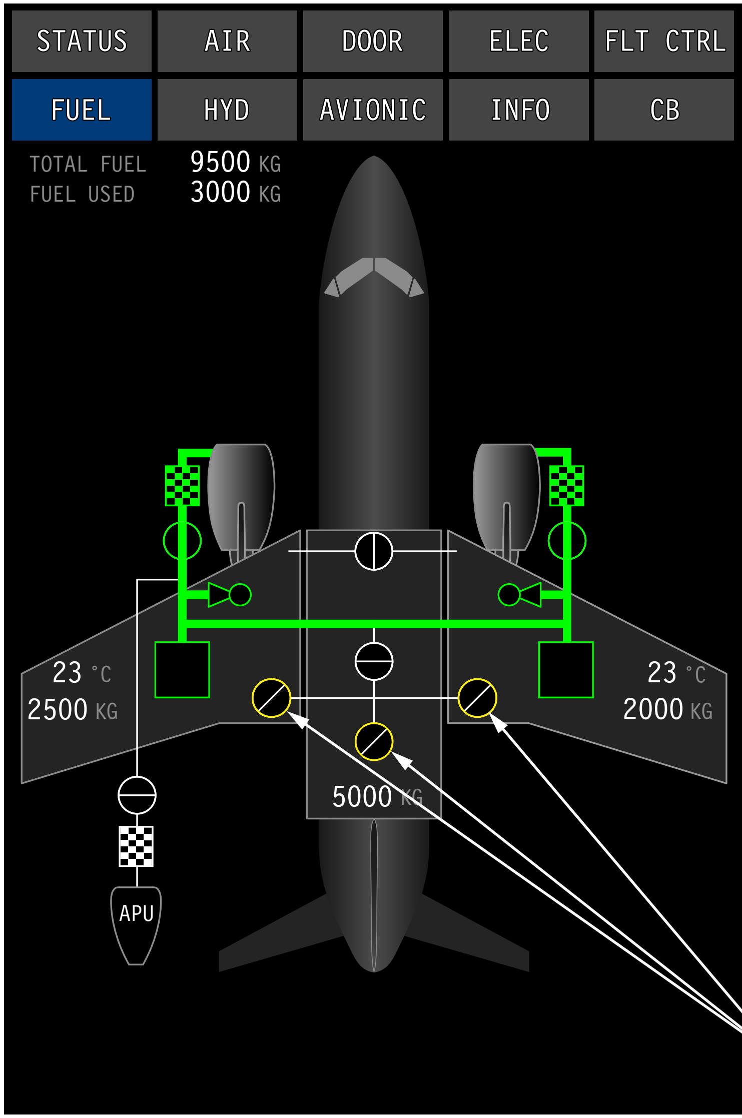


Figure 57: Fuel Transfer Controls

SYNOPTIC PAGE

Fuel transfer operations are shown on the FUEL synoptic page and the EICAS page. The arrows show the direction of transfer during an automatic transfer.

The refuel valves are only shown in amber if they fail during a manual transfer operation.



Refuel Shutoff Valves

SYNOPTIC PAGE - FUEL

VALVE POSITION NORMAL OPERATION	
Symbol	Condition
	Open
	Closed
	In Transition
	Invalid

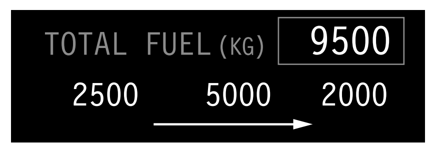
VALVE POSITION FAIL OPERATION	
Symbol	Condition
	Open
	Closed
	In Transition
	Invalid

PUMPS	
Symbol	Condition
	Active
	Off
	Failed
	Invalid

CENTER TO WING XFERS	
Symbol	Condition
	Active
	Failed/Low
	No Transfer
	Invalid

FLOW LINES	
Symbol	Condition
	Fuel Flow
	No Fuel Flow
	Invalid Fuel Flow

TRANSFER FUNCTION ACTIVE	
Symbol	Condition
	To R Wing
	To L Wing



EICAS PAGE

CS1_CS3_2822_008

Figure 58: Fuel Synoptic Page

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) messages for the fuel transfer system.

CAS MESSAGES

Table 7: CAUTION Messages

MESSAGE	LOGIC
FUEL IMBALANCE	Fuel imbalance between left and right tank greater than 364 kg (800 lb).
FUEL CTR XFR FAIL	Fuel transfer out of center tank has failed, center tank fuel may become unusable.
FUEL MAN XFR FAIL	Manual transfer failed to perform the commanded operation or to stop the transfer.

Table 8: ADVISORY Messages

MESSAGE	LOGIC
FUEL MAN XFR COMPLETE	Wing-to-wing target fuel transfer of 182 kg (400 lb) or wing-to-center target fuel transfer of 364 kg (800 lb) has completed.
FUEL GRAV XFR FAIL	FUEL GRAV SOV failed to open or close.
FUEL CTR XFR FAULT	Fuel transfer from center to the left or right tank has failed.

Table 9: STATUS Messages

MESSAGE	LOGIC
FUEL MAN XFR TO L	Manual XFR TO L WING is selected on the fuel control panel.
FUEL MAN XFR TO R	Manual XFER TO R WING is selected on the fuel control panel.
FUEL MAN XFR TO CTR	MAN XFR to the center tank is selected and either L or R boost pump has been turned on.
FUEL GRAV XFR ON	GRAVITY SOV has been commanded open via the overhead panel.
FUEL XFR CTR READY	MAN XFR to the center tank is selected but no pump has been selected.

28-40 FUEL INDICATING SYSTEM

GENERAL DESCRIPTION

The fuel indicating system includes a fuel quantity computer (FQC), remote data concentrators (RDC), fuel probes, and temperature sensors.

The purpose of the fuel probes is to measure the quantity of fuel in the tank. The probes are located strategically throughout the fuel tank to provide the total quantity of fuel in the tank. Compensator probes provide fuel dielectric value used to compensate the fuel quantity for changes in fuel temperature.

Low-level fuel probes provide a warning when approximately 30 minutes fuel remain. The high-level fuel probes prevent an overfill of the fuel tanks.

Temperature sensors in each main tank provide the fuel temperature input to the fuel indicating system. The main fuel tank temperatures are displayed on the FUEL synoptic page.

The fuel probes and compensators in each tank are organized in two arrays called lane 1 and 2. Each array is connected to a processor in the RDC.

Each RDC has two independent and identical processors that convert the fuel probe data into digital format. The RDCs send the tank data digitally to the FQC.

The FQC has dual independent and identical processors for processing the data received from the RDC units. The FQC performs all of the computations associated with the fuel system. The FQC provides fuel quantity and temperature information to the engine indication and crew alerting system (EICAS).

The FQC and RDC software is field-loadable.

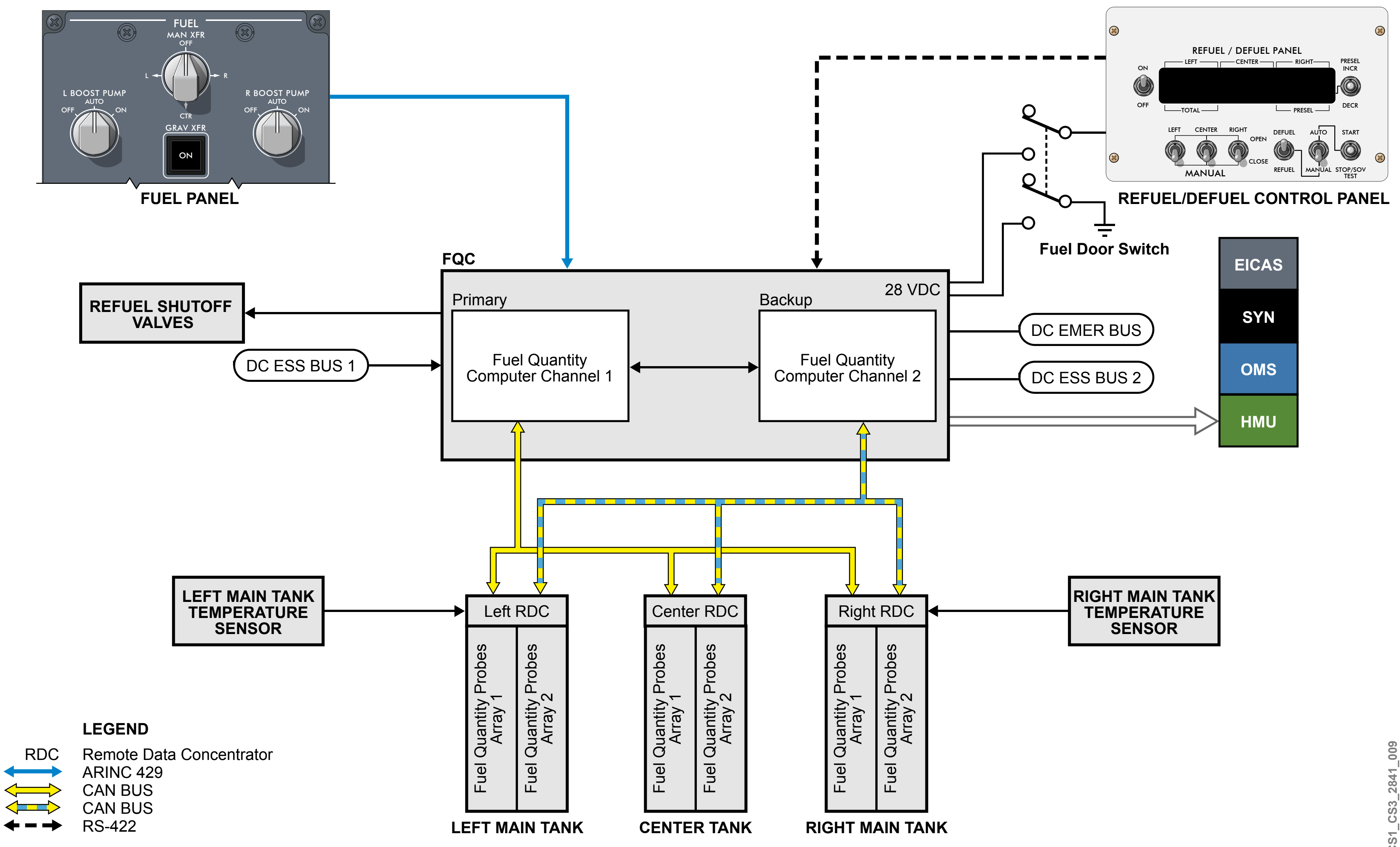


Figure 59: Fuel Quantity Indication

COMPONENT LOCATION

The fuel quantity indication system components consist of:

- Fuel probes
- Temperature sensor
- Remote data concentrator (refer to figure 61)
- Fuel quantity computer (refer to figure 62)

FUEL PROBES

There are 58 fuel probes installed in the aircraft fuel tanks.

TEMPERATURE SENSORS

There is one temperature sensor installed in each of the aircraft main fuel tanks. The temperature sensor is mounted in a housing on RIB 7.

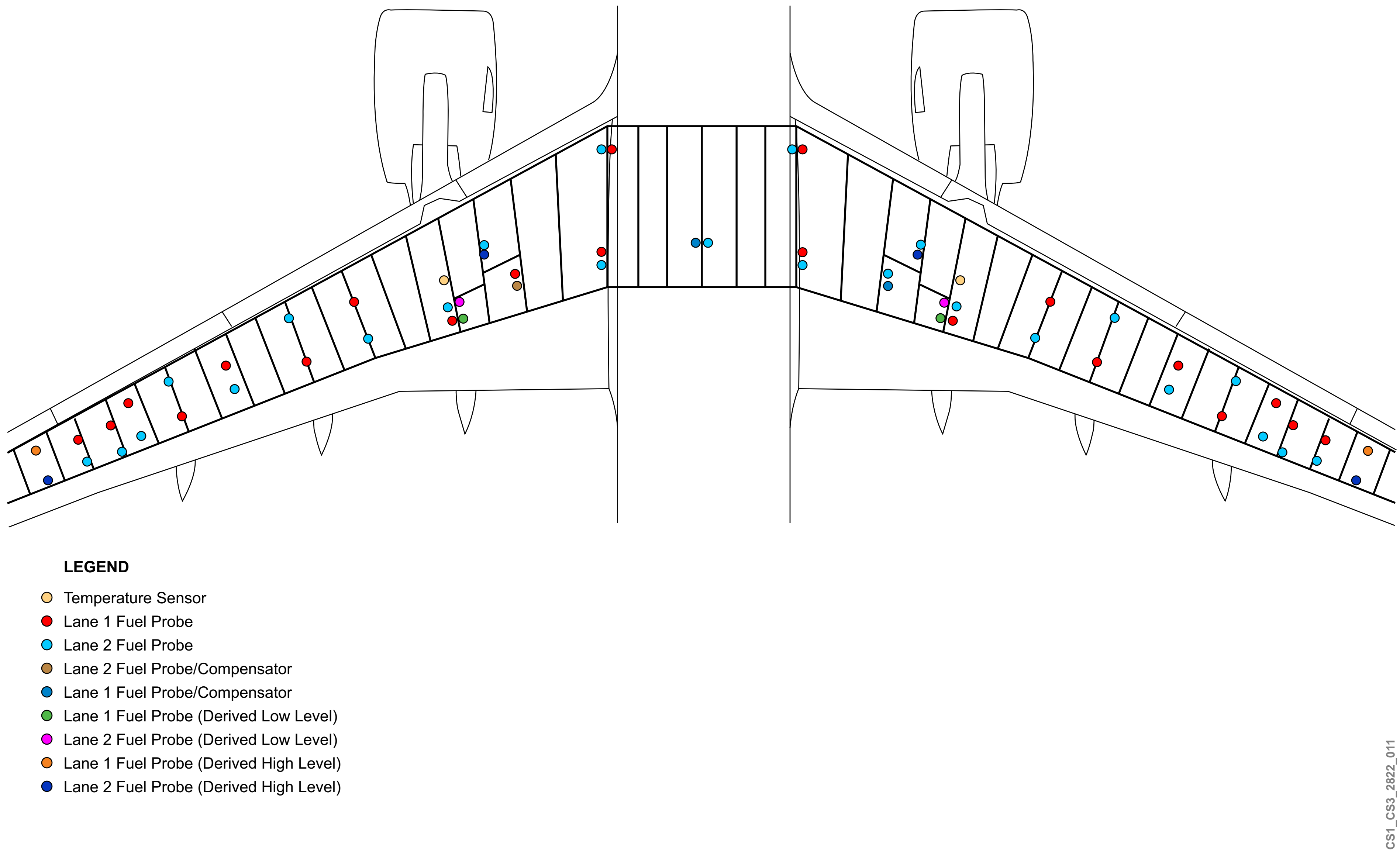
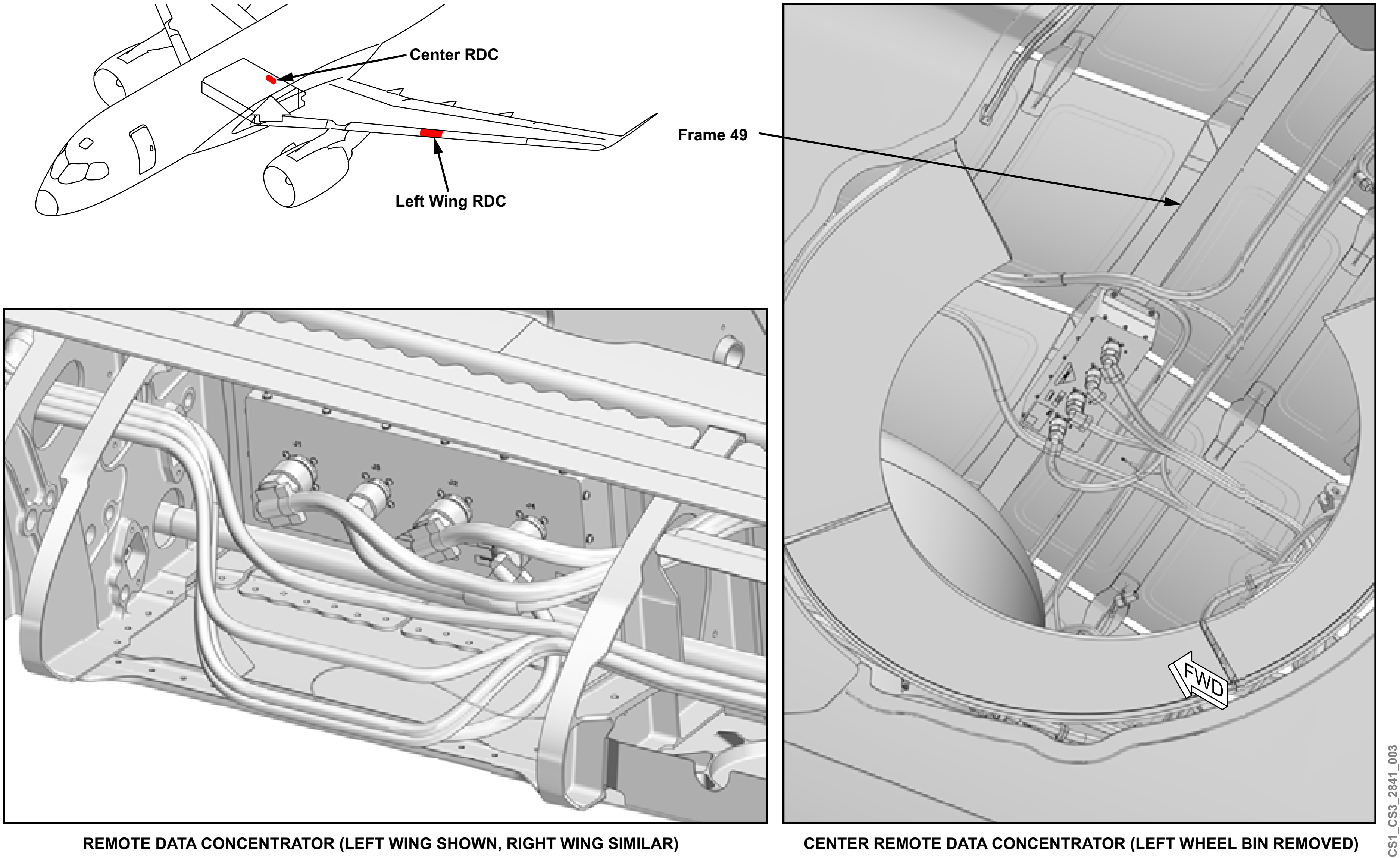


Figure 60: Fuel Quantity Probes and Temperature Sensors

REMOTE DATA CONCENTRATOR

The wing tank remote data concentrators (RDCs) are located on the front spar outboard of the engines. The center RDC is mounted in the wing-to-body fairing (WTBF) on FRAME 49.



REMOTE DATA CONCENTRATOR (LEFT WING SHOWN, RIGHT WING SIMILAR)

CENTER REMOTE DATA CONCENTRATOR (LEFT WHEEL BIN REMOVED)

Figure 61: Remote Data Concentrator

FUEL QUANTITY COMPUTER

The fuel quantity computer (FQC) is located in the mid equipment bay.

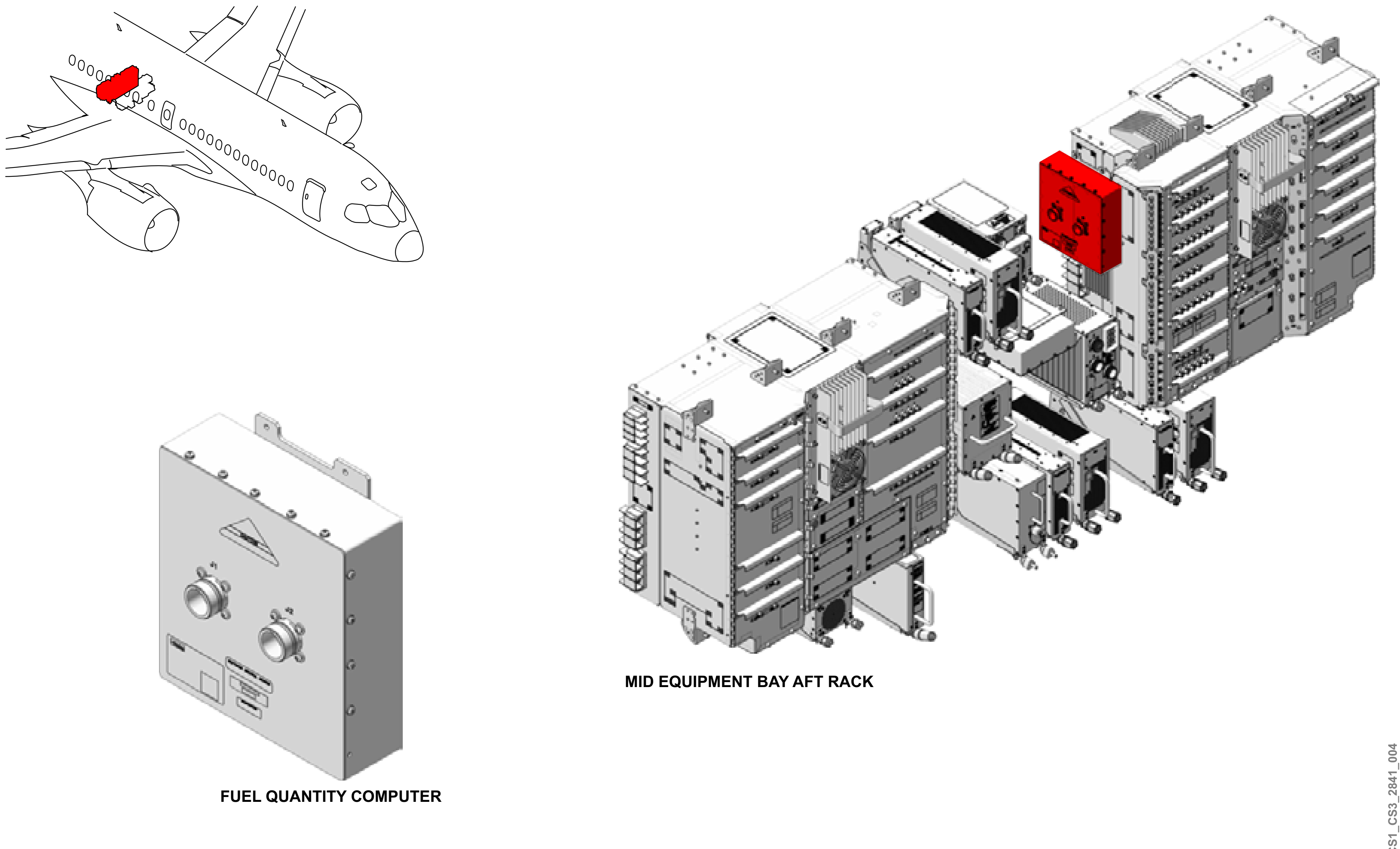


Figure 62: Fuel Quantity Computer

FUEL PROBES

The tank units are two element concentric metal tube assemblies open at each end to allow fuel in. The probes are mounted on brackets attached to ribs and located throughout the tank. The length of each tank unit is determined by the height of the fuel tank at its particular location. In addition to providing quantity information, some fuel quantity probes perform other functions, as follows:

- **High-level probes** - There are two capacitance type fuel high-level probes installed in each main fuel tank and four in the center tank. The high-level probes are mounted outboard of RIB 20 in the main tanks, and at RIB 6 on the left and right sides of the center tank
- **Low-level probes** - There are four capacitance type low-level fuel probes in each wing tank. Two fuel probes are located in each main tank outboard of RIB 7, and two are in each collector tank on the inboard side of RIB 7
- **Compensator probes** - The compensator is a multiple concentric metal tube assembly mounted near the bottom of each main tank. In this position, the compensator is always covered with fuel until each tank almost empty. Any change in capacitance is the result of change of dielectric constant of the fuel. The compensator probes are located at RIB 5 of the main tanks and RIB 0 of the center tank

TEMPERATURE SENSOR

The temperature sensor attaches to the housing with a bayonet fitting. The sensor is accessed from the dry bay.

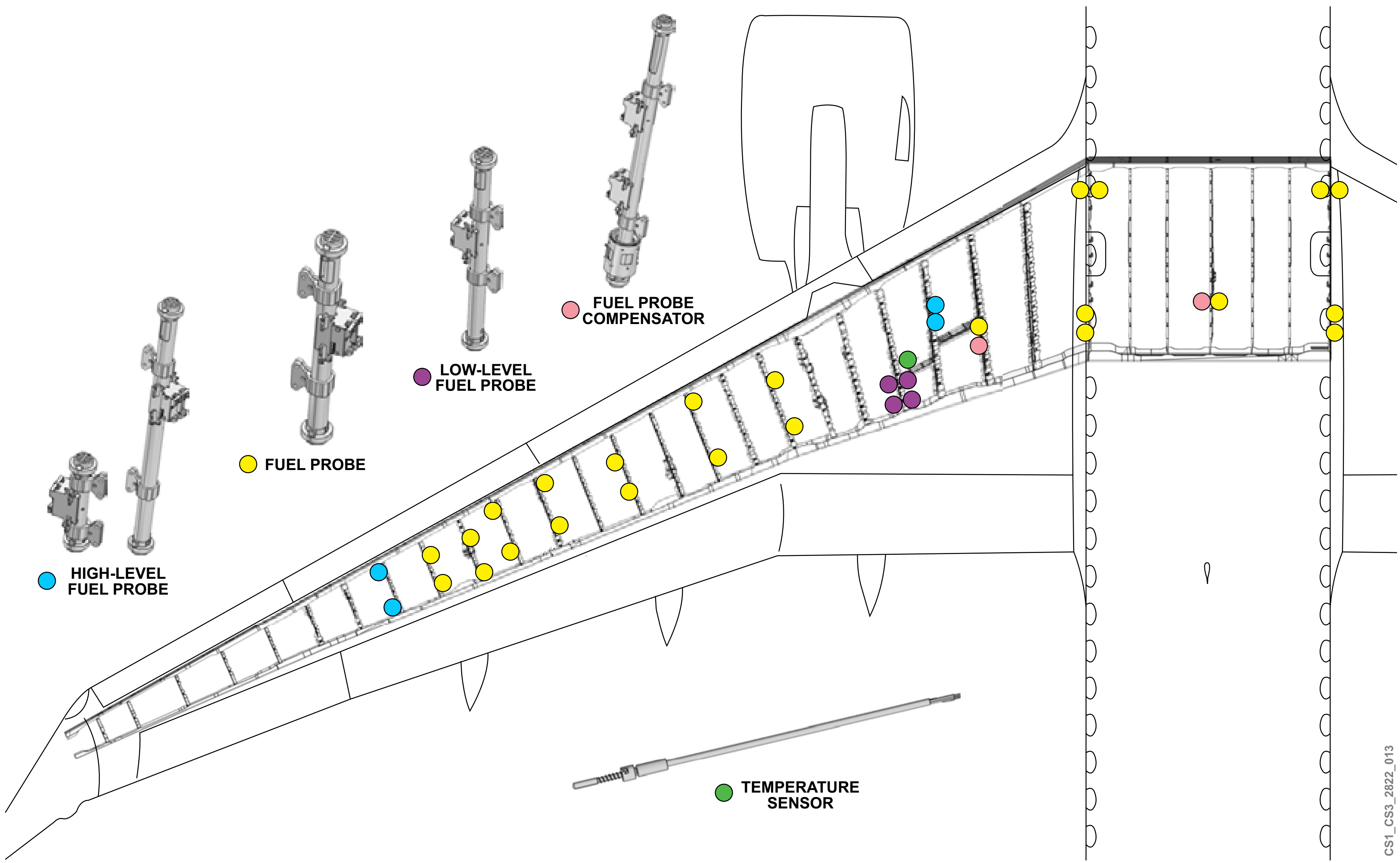


Figure 63: Fuel Probes and Temperature Sensor

DETAILED COMPONENT INFORMATION

FUEL QUANTITY COMPUTER INTERFACE

The fuel quantity computer interfaces with other aircraft systems through the data concentrator unit module cabinet (DMC). The FQC provides fuel quantity and fuel temperature information to the DMC via ARINC 429 BUSES.

The FQC provides:

- Defuel and refuel valve control
- Fuel quantity and temperature related CAS messages
- High and low fuel level warning
- Fuel imbalance warning
- Automatic wing fuel transfer
- Communication with the DMC
- Onboard maintenance system communication
- Communication with the RDCs

Most fuel system electrical components interface directly with the electrical system. The control signals are derived from the direct discrete signals, such as the fire handle, or indirect signals via an ARINC 429 BUS.

The operational commands from these systems control the operation of the fuel system pumps and valves. The operating status of the pumps and valves are fed to the DMC and the electrical system.

The fuel system components interface with the DMC to provide information to the engine indication and crew alerting system (EICAS), FUEL synoptic page display, and the onboard maintenance system (OMS).

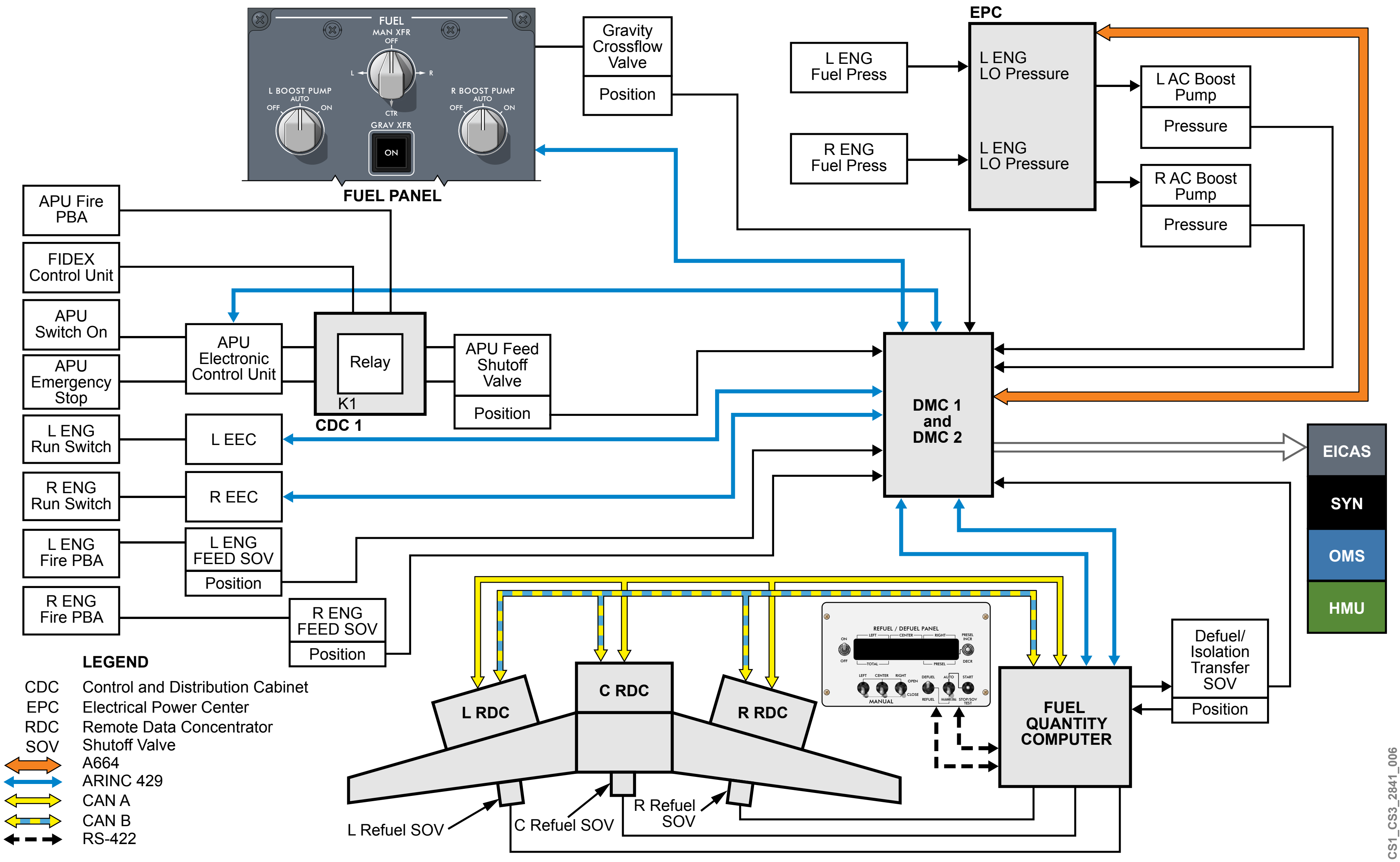
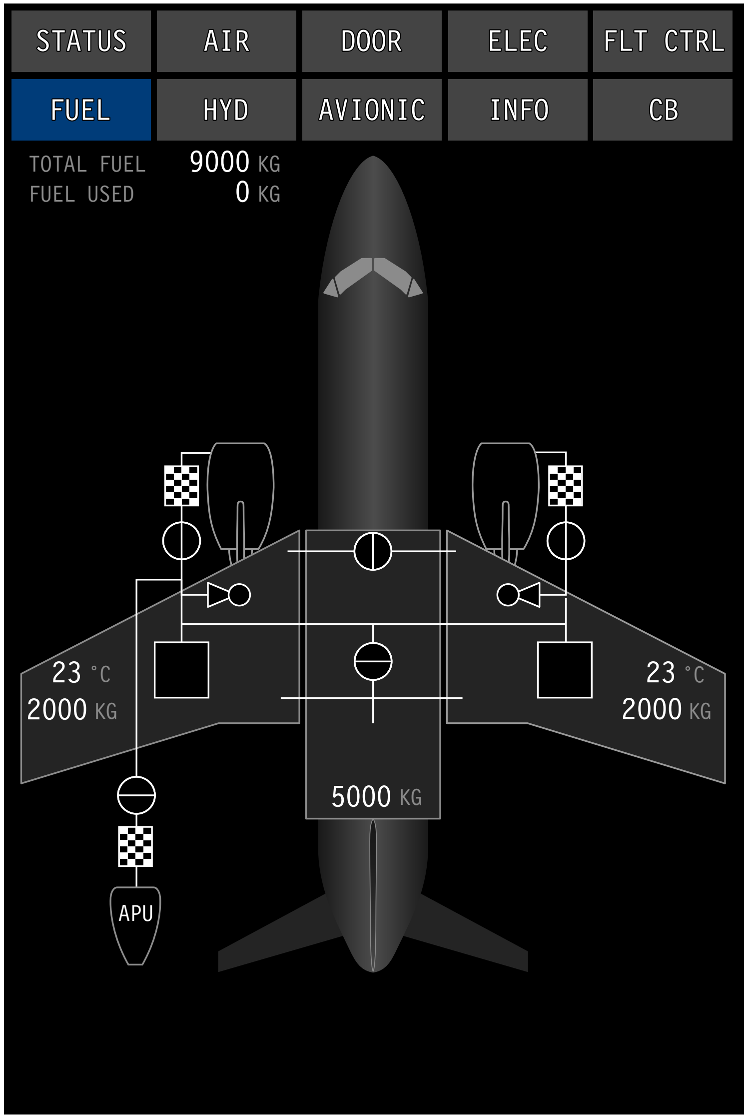


Figure 64: Fuel Quantity Interface

CONTROLS AND INDICATIONS

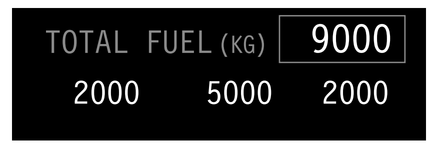
The fuel quantity indication is displayed on the FUEL synoptic page. The fuel quantity, fuel used, and temperature information are also displayed on the EICAS page.



SYNOPTIC PAGE - FUEL

QUANTITY	
Symbol	**Condition**
3000 KG	Normal
1600 KG	Low/Overfilled/ Imbalance/Trapped
-- KG	Invalid

TEMPERATURE	
Symbol	**Condition**
22 °C	Normal
-44 °C	Low or High
-- °C	Invalid



EICAS PAGE

CS1_CS3_2841_002

Figure 65: Fuel Controls and Indications

DETAILED DESCRIPTION

The fuel quantity system consists of arrays of AC capacitance probe units, located in the left wing, right wing and center fuel tanks. Each fuel tank uses an RDC to isolate the tank sensors from the dual-channel FQC. The RDCs provide information digitally to the FQC via two CAN BUSES. The FQC interfaces with a refuel/defuel control panel using an RS-422 BUS.

Each RDC has two lanes for redundancy. The RDC lanes 1 and 2 have independent and identical processors that receive analog data from the fuel probes and sensors. An interprocessor connects the lanes within the RDC. This ensures that all the data from all the tank probes is available in either lane.

Each lane is broken down into groups of fuel probes. An excitation generator in the RDC provides a separate excitation signal to each group. By grouping the fuel probes, the total energy in each circuit is very low. This reduces ignition source hazards.

One fuel probe compensator is located in each tank. Temperature sensors provide the fuel temperature of the left and right wing tank fuel.

To ensure availability of data to FQC from either wing compensator probe or temperature sensor in case of a CAN BUS failure, the temperature sensor in left tank is connected to RDC lane 2 while the temperature sensor in the right tank is connected to RDC lane 1. The compensator in the left tank is connected to RDC lane 2 and the compensator in the right tank is connected to RDC lane 1.

The FQC has two identical and independent channels for processing the fuel quantity data. Each channel receives data digitally from the RDCs. The FQC uses the fuel quantity information to control the refuel/defuel operations and automatic fuel balancing.

The FQC channels communicate with each other over an internal bus. The internal bus allows sharing of any input data from the other channel in case of input failures. Each FQC channel sends data to the L and R DMCs over ARINC 429 BUSES. In normal operation, the DMCs use the data from FQC channel 1. Channel 2 data is used when channel 1 fails.

The FQC continuously monitors the fuel quantities in each tank and provides annunciation of potentially hazardous conditions. The low-level fuel probes in each wing tank are located at a level to provide an output at 30 minutes fuel remaining, which is approximately 443.2 kg (975 lb). The FQC provides two independent low-level indications to the DMC over ARINC 429 BUSES and discrete lines. In the event of an ARINC 429 BUS failure, the discrete signals are used for display on the EICAS for L or R FUEL LO QTY messages.

The high-level fuel probes provide a signal when the fuel level exceeds 98% of the tank volume. During refueling, a high fuel level condition shuts down the refuel system and annunciates a TANK HIGH LEVEL on the refuel/defuel control panel (RDCP).

The flight crew has no indication of a leak other than information derived from fuel quantity data. A FUEL LEAK SUSPECT caution message provides indication of a possible fuel leak if any of the following conditions are present:

- The fuel system suspects a wing tank overfill condition is present
- The electronic engine control detects excessive fuel consumption by comparing both engines
- The FMS calculated fuel used and indicated fuel onboard is different

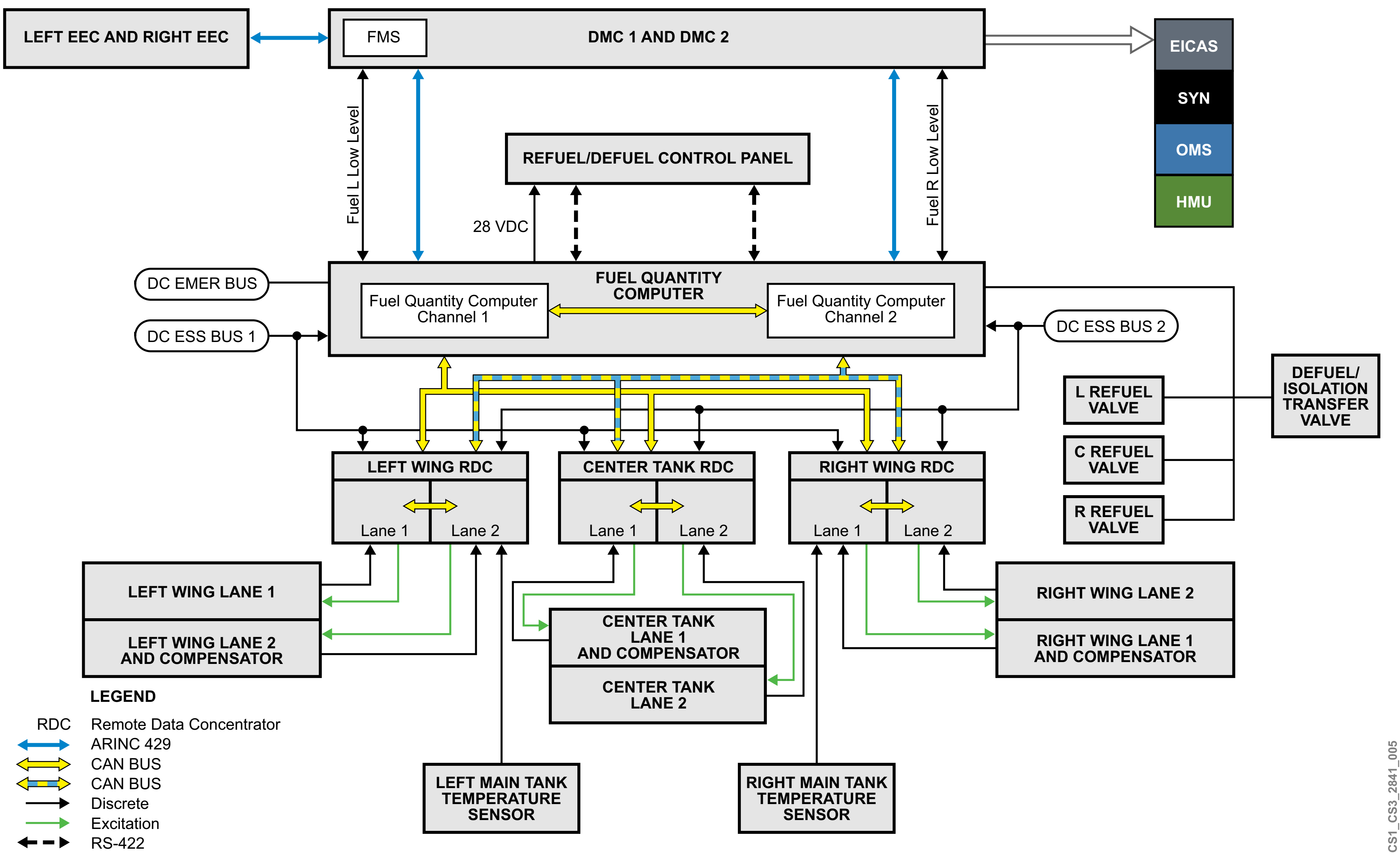


Figure 66: Fuel Quantity System

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the fuel quantity indication system.

CAS MESSAGES

Table 10: CAUTION Messages

MESSAGE	LOGIC
L FUEL LO QTY	Low-level detected in left wing tank.
R FUEL LO QTY	Low-level detected in right wing tank.
FUEL TANK LO TEMP	Fuel temp is near Jet A fuel freezing point (-37 °C).
FUEL TANK HI TEMP	Fuel temp above the operating limit.
FUEL COLLECTOR LO LVL	Fuel in the collector is low - suspect the transfer ejector is inoperative for the left or right tank.
FUEL LEAK SUSPECT	This may be caused by fuel leaking overboard due to: • A wing tank fuel overfill condition from the center transfer float valve stuck open • A manual transfer refuel solenoid valve failed to close Or this message may be displayed when: • The EEC suspects a leak at the engine based upon comparison between the L and R engine fuel consumptions • The FMS detects a suspected fuel leak triggered by a difference in fuel onboard compared to the FMS calculated fuel consumption while the engines are running

Table 11: ADVISORY Messages

MESSAGE	LOGIC
FUEL COMPUTER FAIL	Fuel quantity computer has failed or loss of both ARINC 429 channels from FQC to DMC.
FUEL FAULT	Loss of redundant or non-critical function for the fuel system.

Table 12: INFO Messages

MESSAGE	LOGIC
28 FUEL FAULT - COMPUTER REDUND LOSS	Fuel computer channel fault is detected.
28 FUEL FAULT - L WING RDC INOP	Loss of fuel quantity in the left tank.
28 FUEL FAULT - L WING RDC REDUND LOSS	Fuel left wing remote data concentrator fault is detected.
28 FUEL FAULT - R WING RDC INOP	Loss of fuel quantity in the right tank.
28 FUEL FAULT - R WING RDC REDUND LOSS	Fuel right wing remote data concentrator fault is detected.
28 FUEL FAULT - CTR WING RDC INOP	Loss of fuel quantity in the center tank.
28 FUEL FAULT - CTR WING RDC REDUND LOSS	Fuel center tank remote data concentrator fault is detected.
28 FUEL FAULT - FUEL GAUGING SNSR INOP	Fuel gauging sensor (probe/compensator) fault is detected.
28 FUEL FAULT - FUELING DOOR OPEN	Fuel door is not CLSD prior to flight.
28 FUEL FAULT - FUEL KG-LB MISCOMPARE	Fuel system refuel panel and cockpit primary page use different units (kg/lb).

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Table 12: INFO Messages

MESSAGE	LOGIC
28 FUEL FAULT - DEFUEL/XFR SOV INOP	DEFUEL/XFER SOV has failed when the actual reported SOV position is different from the commanded position, including conflicting reported position.
28 FUEL FAULT - CONFIG STRAPPING INOP	Fuel gauging computer or fuel remote data concentrator has strapping fault(s).
28 FUEL FAULT - SOFTWARE LOAD INOP	Fuel gauging computer software load fault is detected.
28 FUEL FAULT - FUEL TEMP SNSR INOP	One fuel temperature sensor has failed.
28 FUEL FAULT - GAUGING SNSR SHORT CIRCUIT	Gauging sensor short circuit of fuel probe or wiring.
28 FUEL FAULT- BOOST PUMP PRESS SW FAIL HI	Left and/or right boost pump pressure switch is failed to high-pressure.

PRACTICAL ASPECTS

ONBOARD MAINTENANCE SYSTEM

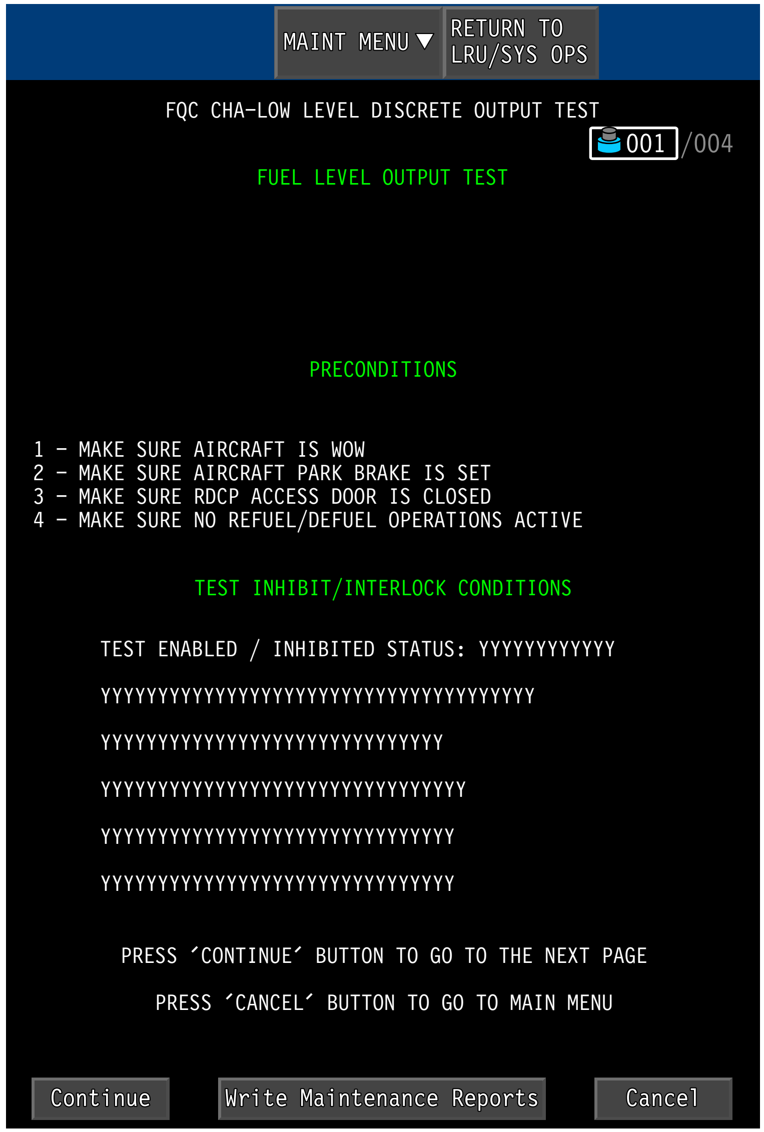
Each channel of the FQC is tested using the OMS. The FQC enters the OMS test mode when the aircraft is weight-on-wheels (WOW) and the parking brake is set.

The OMS provides a system parameter page that displays the real time reported capacitance and temperature sensor data as well as status and other fuel system related faults from the ARINC 429 data.

Low-Level Discrete Output Test

When the OMS commands a low-level discrete output test, the FQC channel sets the FUEL L LOW LEVEL and then the FUEL R LOW LEVEL discrete outputs true for 3 seconds.

The FQC also compares the DMC reported feedback of the FUEL L LOW LEVEL and FUEL R LOW LEVEL discrete outputs to the FQC commanded state. This test checks that the low-level discrete outputs operate properly, and verifies proper wiring and communication with the DMCs. The FQC reports the pass/fail status of this test to the OMS.



CS1_CS3_2841_001

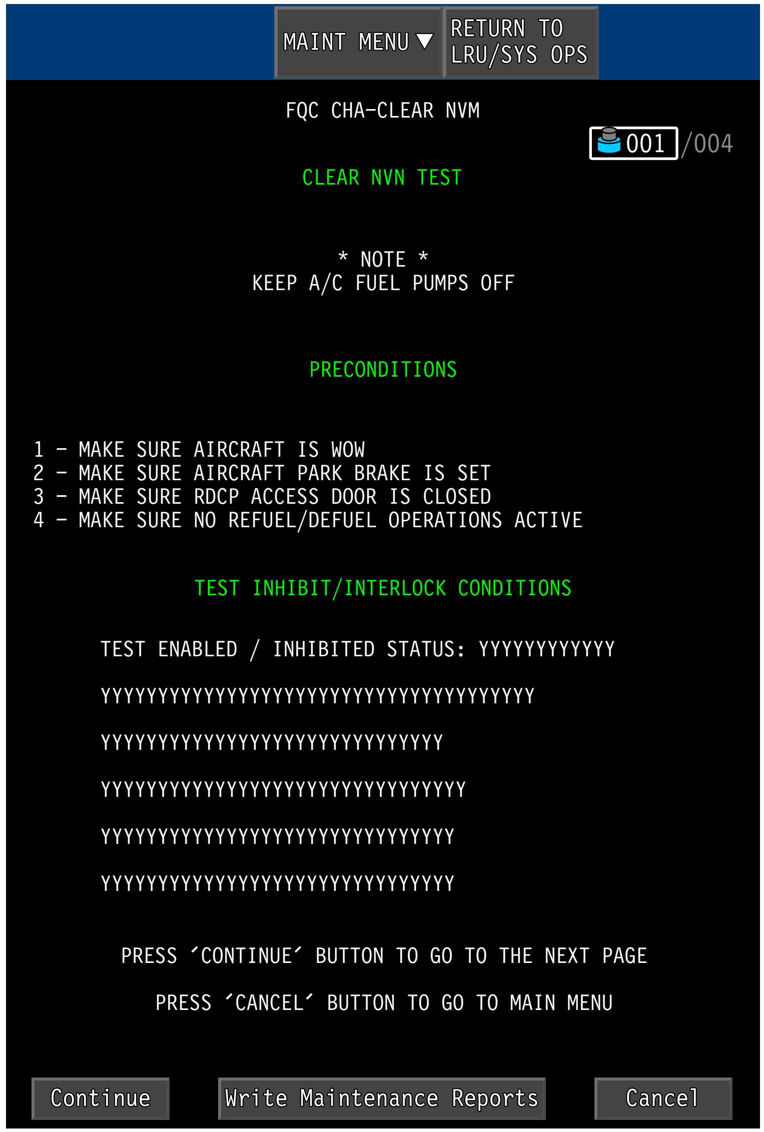
Figure 67: Low-Level Discrete Output Test

Additional Functions

The fuel quantity computer provides the ability to clear non-volatile memory (NVM) faults and report the configuration from the OMS test screen selection.

These initiated functions include:

- The CLEAR NVM function clears faults stored in the FQC NVM
- The REPORT CONFIGURATION function reports the configuration data of the fuel quantity system. The FQC, RDCs, and RDCP report their configuration data to the OMS



CS1_CS3_2823_020

Figure 68: Clear NVM Function

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ATA 30 - Ice and Rain Protection

BD-500-1A10
BD-500-1A11

Table of Contents

List of Figures

ICE AND RAIN PROTECTION - CHAPTER BREAKDOWN

30-81 ICE DETECTION SYSTEM

GENERAL DESCRIPTION

The detection system detects the supercooled liquid water conditions present in the clouds that can cause ice formation on the aircraft.

The ice detectors provide independent measurements of icing conditions.

The system is comprised of two magnetostrictive ice detectors. The ice detector is a one-piece, vibrating sensing element (probe) type device. When ice accumulates on the probe, the drop in frequency triggers the ice detector to generate discrete signals (ice). In addition, heaters inside the probe and strut automatically come on to melt the ice, then turn off. With the heaters off, ice is again allowed to build up. The cycle of accumulation and melting is repeated until ice is no longer present. The discrete signals are switched off 2 minutes later.

The detectors operate using 115 VAC electrical power and are always powered when the aircraft is powered. The left ice detector is powered by the AC BUS 1 and the right is powered by the AC BUS 2.

When ice is detected by either detector, discrete signals are automatically sent via the data concentrator unit module cabinet (DMC) to the wing anti-ice (WAIS) system and the cowl anti-ice system (CAIS) to provide anti-icing. It also signals the three primary flight control computers (PFCCs) to modify the stall warning protection.

As a backup the ice detection signal is also sent to the WAIS through the control and distribution cabinets (CDCs).

The ice signals and ice detector status information are sent to the engine indication and crew alerting system (EICAS) and the onboard maintenance system (OMS).

CAUTION

As a safety precaution do not touch the probes with bare hands, as they could be hot and inflict burns.

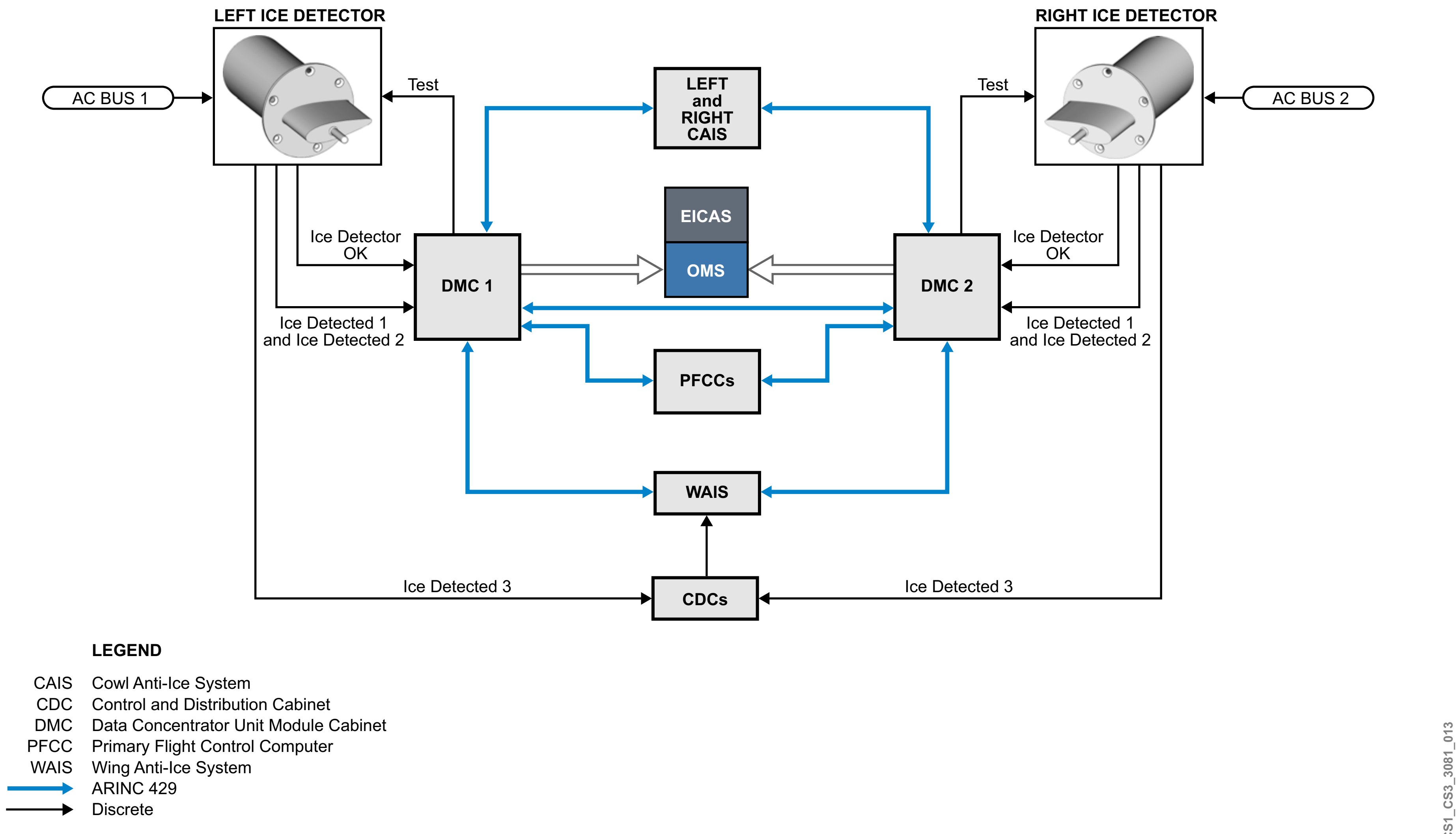


Figure 1: Ice Detection System

COMPONENT LOCATION

There are two identical ice detectors installed in the system.

ICE DETECTOR

The ice detectors are located on each side of the forward fuselage, one is on the left, and the other on the right.

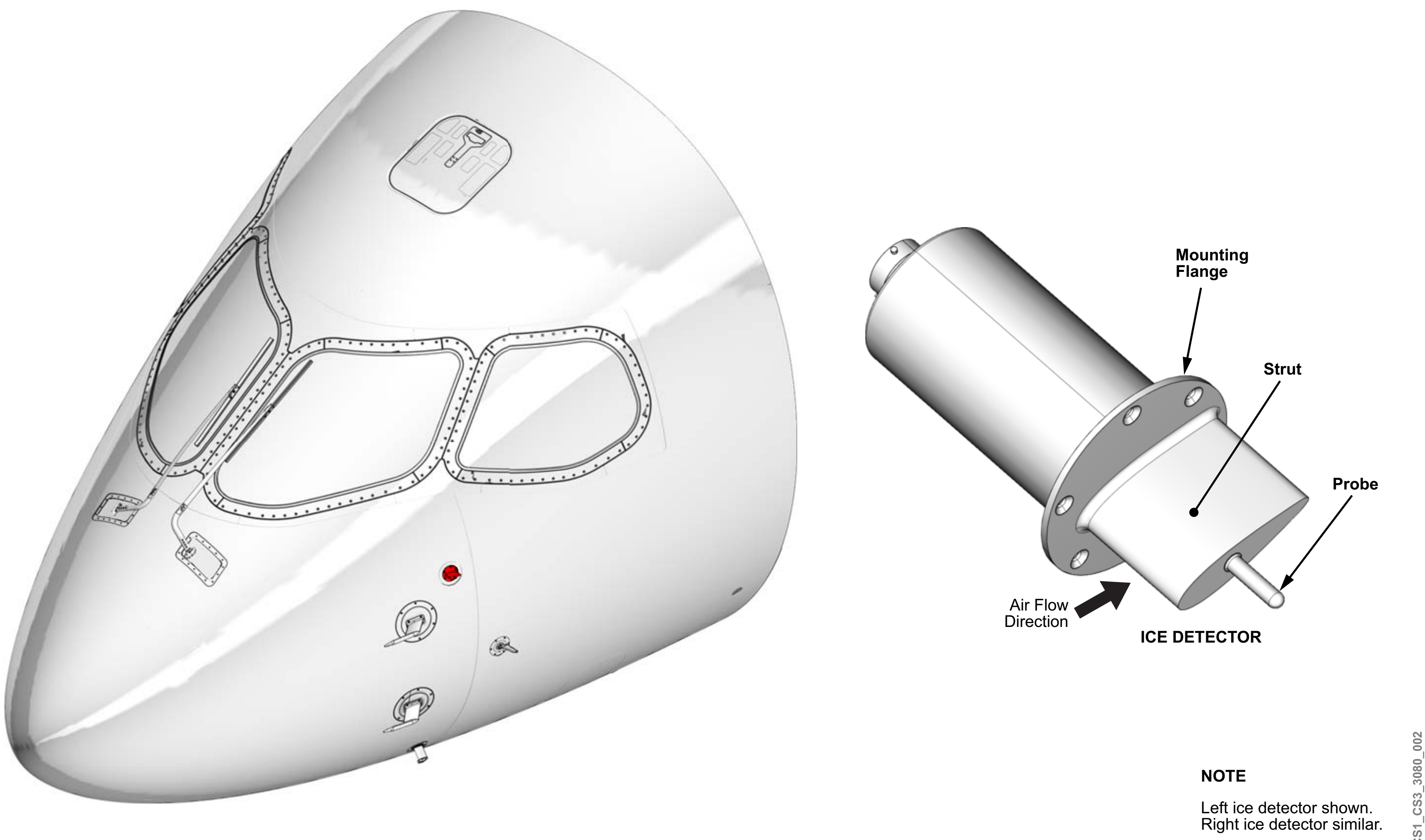


Figure 2: Ice Detector

DETAILED DESCRIPTION

The ice detector is a one-piece, vibrating sensing element (probe) type device. An internal electrical oscillator circuit drives the sensing element to vibrate at its mechanical resonance (tuned to 40,000 Hz) using the principles of magnetostriction. The shift in the measured frequency, which corresponds to an ice thickness of approximately 0.51 mm (0.02 in.) triggers the ICE DETECTED discrete. When ice is detected, the strut and probe are de-iced and melt ice accumulation. The aerodynamic airflow then sheds the ice and provides the probe cooling for continued sensing.

When it is powered up and operational, the ice detector provides an ICE DETECTOR OK signal. When icing is detected, each detector provides three signals: ICE DETECTED 1, ICE DETECTED 2, and ICE DETECTED 3.

The ICE DETECTED 1 and ICE DETECTED 2 signals are sent to the DMCs and distributed to the left and right electronic engine controls (EECs) for cowl anti-ice (CAI) and to the three primary flight computers (PFCCs). The DMC also sends the ice detection signals to the integrated air system controllers (IASCs) to enable wing anti-ice system (WAIS) activation.

ICE DETECTED 3 signal is sent through the control and distribution cabinets (CDCs) for the WAIS.

The ICE DETECTED and the ICE DETECTOR OK signals are used to generate caution and advisory EICAS messages. The ice detection system is also monitored by the OMS.

An ICE caution message is displayed when ice is detected and the wing anti-ice system (WAIS) or cowl anti-ice system (CAIS) are inoperative. An ICE advisory message indicates normal operation of wing and cowl anti-ice systems in icing conditions. When the ICE advisory message is removed the crew can select the anti-ice systems off.

If an ice detector fails, the associated ICE DET FAIL caution message is displayed.

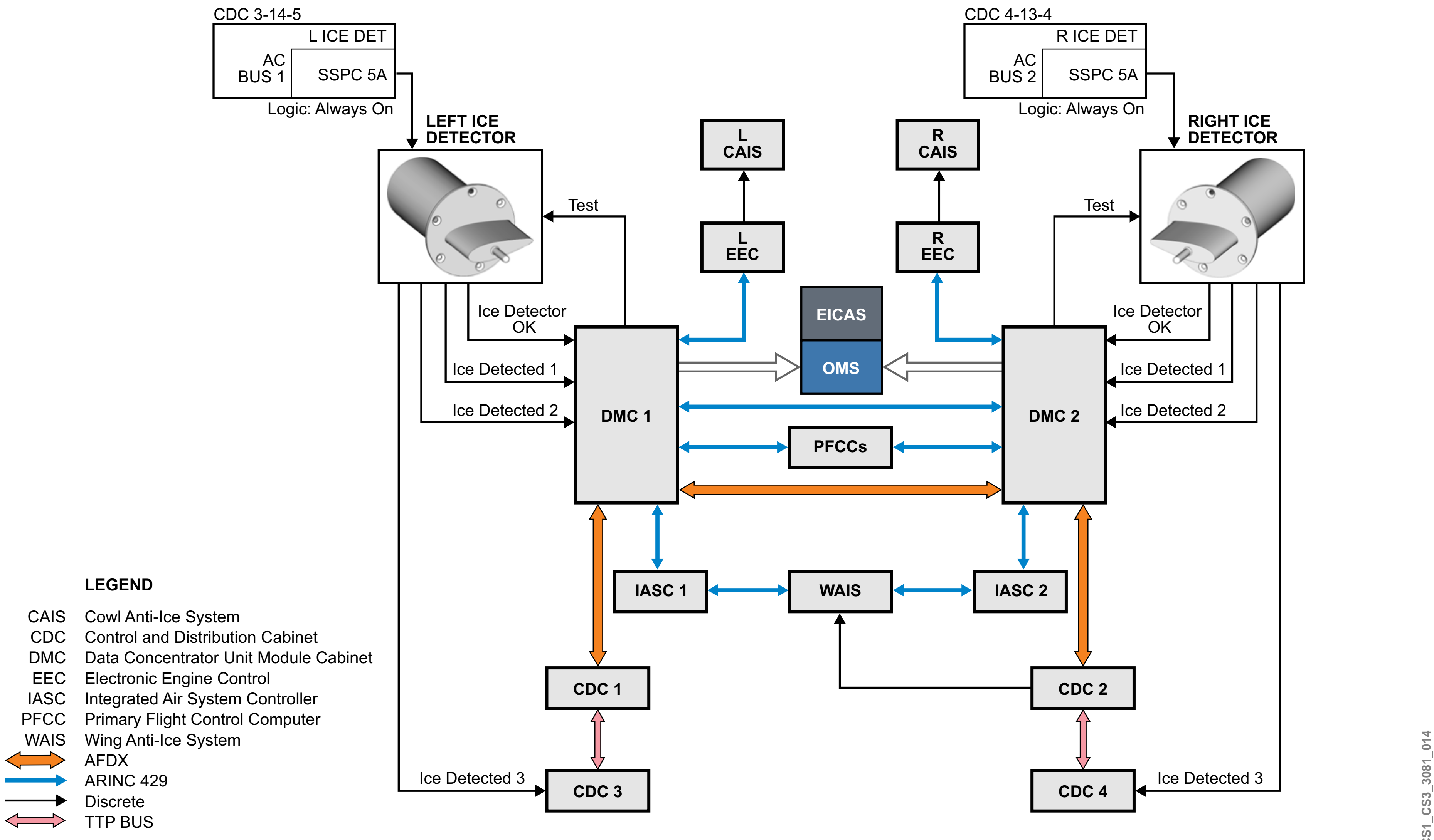


Figure 3: Ice Detection System Schematic

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) messages associated with the ice detection system.

CAS MESSAGES

Table 1: CAUTION Messages

MESSAGE	LOGIC
L ICE DET FAIL	Left ice detector has failed.
R ICE DET FAIL	Right ice detector has failed.
ICE	Ice detected and wing anti-ice or cowl anti-ice are failed or is off.

Table 2: ADVISORY Messages

MESSAGE	LOGIC
ICE	Ice detected but ice detection, wing anti-ice and cowl anti-ice are working properly.

INITIATED BUILT-IN TEST

Once powered, each ice detector initiates its own power-up built-in test (PBIT) lasting about 7 seconds. The detector produces an ICE DETECTOR OK signal after internal testing is successful and remains active, provided it continues to function normally.

An initiated built-in test (IBIT) function performs a diagnostic check of both ice detectors, including communication with interconnecting systems. The IBIT is initiated from the EICAS AVIONICS page on the AVIO tab.

When IBIT is initiated:

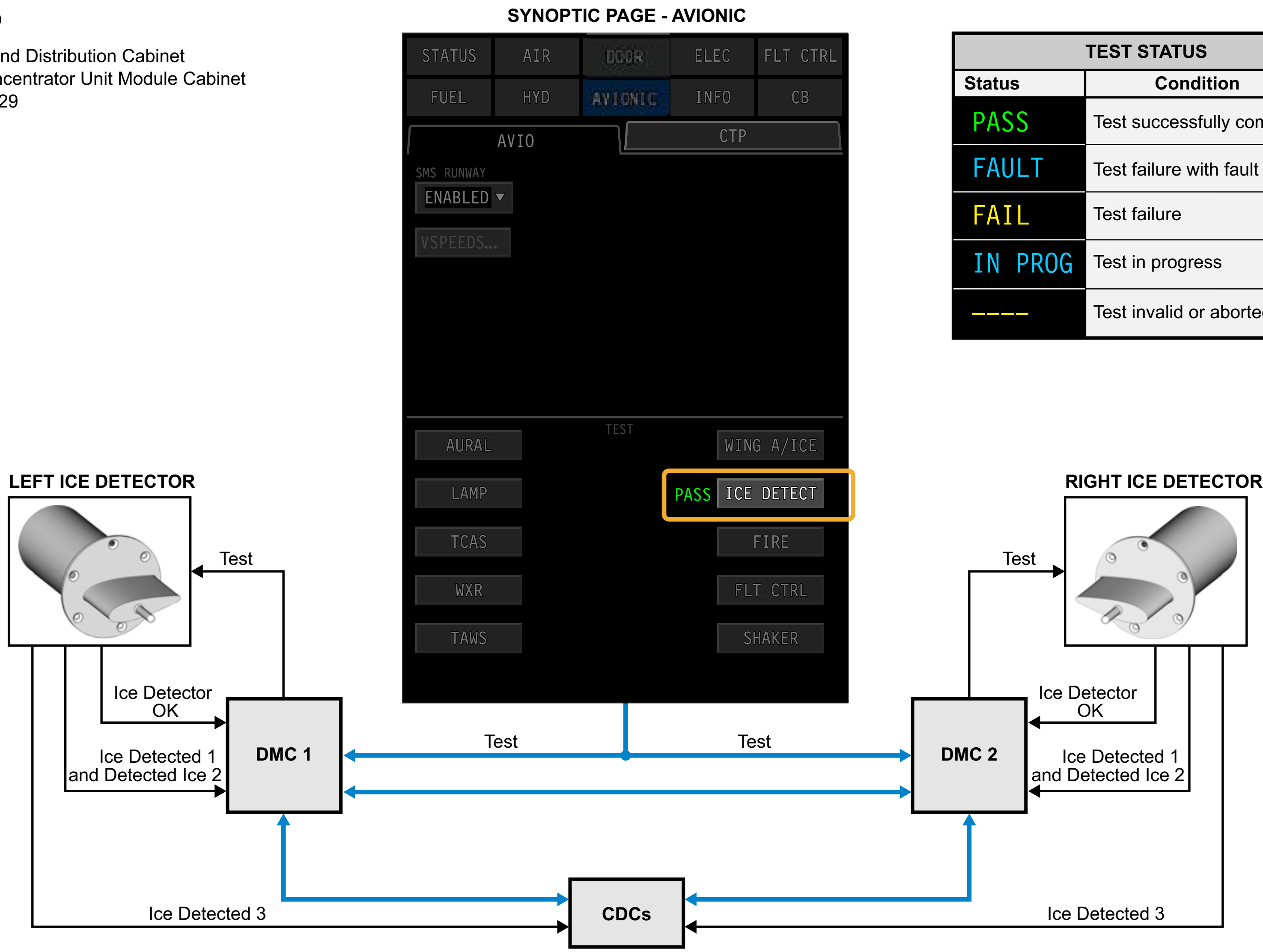
- A message IN PROG is provided
- The ice detectors perform their internal tests
- The data concentrator unit module cabinets (DMCs) monitor ice detection signals, ICE DETECTED 1, and ICE DETECTED 2, from the ice detectors
- The DMCs monitor ice detection signal, ICE DETECTED 3, from the control and distribution cabinets (CDCs)

When the ice detectors internal test is complete, the detector produces an ICE DETECTOR OK signal if internal testing is successful. One of two messages, PASS or FAIL, is displayed on the AVIONICS synoptic page on the AVIO tab and more data is available in the onboard maintenance system (OMS) The PASS or FAIL message is triggered by the value of ICE DETECTOR OK signal.

At any time, if an ice detector has an internal fault, the ice detector turns off and a L (R) ICE DET FAIL caution message is displayed.

LEGEND

CDC Control and Distribution Cabinet
DMC Data Concentrator Unit Module Cabinet
(blue arrow) ARINC 429
(black arrow) Discrete



TEST STATUS	
Status	**Condition**
PASS	Test successfully completed
FAULT	Test failure with fault message
FAIL	Test failure
IN PROG	Test in progress
----	Test invalid or aborted

Figure 4: Ice Detection System Test

30-12 WING ANTI-ICE SYSTEM

GENERAL DESCRIPTION

The wing is protected from icing conditions by the wing anti-ice system (WAIS), which prevents the formation of ice on the leading edge of wing slats no. 2, no. 3, and no. 4 by supplying the hot bleed air from the engines. Ice protection is not provided for the slats that are located inboard of the engines.

The wing anti-ice valves (WAIVs) regulate the pressure to 27 psi in the associated wing anti-ice ducting. Flow is controlled by venturis. After heating the inside of the slat, the bleed air is expelled overboard underneath the wing.

The WAIS includes two independent anti-icing systems for the left and right wings. Each wing system is connected to the other by a normally closed cross-bleed valve (CBV). The engine bleed air system supplies the hot air at required pressure and temperature to the distribution ducts installed in the slats. If one engine or bleed air supply fails, the bleed air system opens the CBV to supply air to the opposite wing.

The integrated air system controllers (IASCs) send a wing anti-ice (WAI) enable signal to the data concentrator unit module cabinets (DMCs) to activate or deactivate the wing anti-ice system (WAIS), based on the ice detection system and WING switch position.

The WAIS is monitored and controlled by the IASCs. With the ANTI-ICE WING switch in the AUTO position, the IASCs automatically activate the WAIS on receiving ice signals from one or both ice detectors.

The WAIS is monitored and controlled by the IASCs. In AUTO mode, the WAIS is inhibited on the ground and during takeoff. In flight, the IASCs automatically activate the WAIS on receiving ice signals from one or both ice detectors.

When ANTI-ICE WING switch is set to ON, the WAIS is activated regardless of icing conditions if the airspeed is above 60 kt on the ground or anytime in flight.

The WAIS can be tested on the ground if the switch is in the ON position and the airspeed is below 60 kt. The test lasts 20 seconds.

Each wing is continuously monitored for temperature and pressure by both IASCs. The wing anti-ice temperature sensor (WAITS) provides monitoring of the temperature of the bleed air supplied to the wing ducting. The pressure sensor measures system pressure from the outboard end of slat no. 4. Pressure is monitored to detect excessive high pressure that might lead to ducting failures, as well as low pressure which may indicate inadequate hot air for effective anti-icing or to detect a duct burst or leak.

In case a leak is detected in the ducting network, the IASC shuts down the WAIS automatically.

IASC 1 is powered by DC ESS BUS 1 and DC ESS BUS 2. IASC 2 is powered by DC ESS BUS 2 and DC ESS BUS 3. The WAIVs are powered by DC ESS BUS 2 provided by the control and distribution cabinets (CDCs). The pressure sensors are powered from DC ESS BUS 3.

NOTE

IASC 1 is powered by DC ESS BUS 1 and DC ESS BUS 2.
IASC 2 is powered by DC ESS BUS 2 and DC ESS BUS 3.

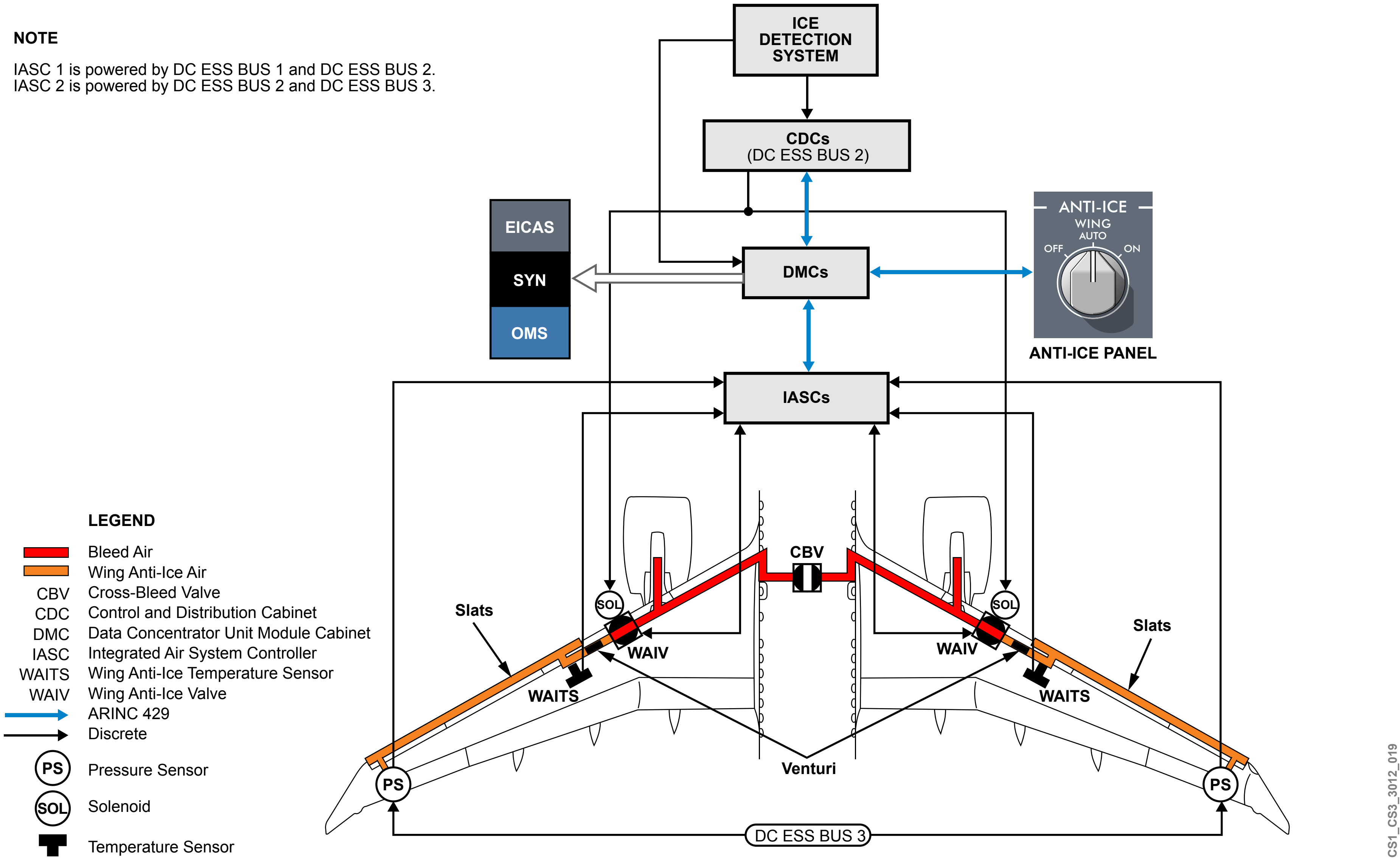


Figure 5: Wing Anti-Ice System

WING ANTI-ICE DUCTING

The wing anti-ice ducting consists of a venturi duct and three piccolo tubes connected by interslat hoses. The air exits the piccolo tubes and circulates around the front bay inner chamber of the slat to be expelled overboard through slots and exhaust holes in the slat rear bay.

A telescopic duct connects the wing anti-ice system air duct to the piccolo tubes to allow for the slats extension and retraction.

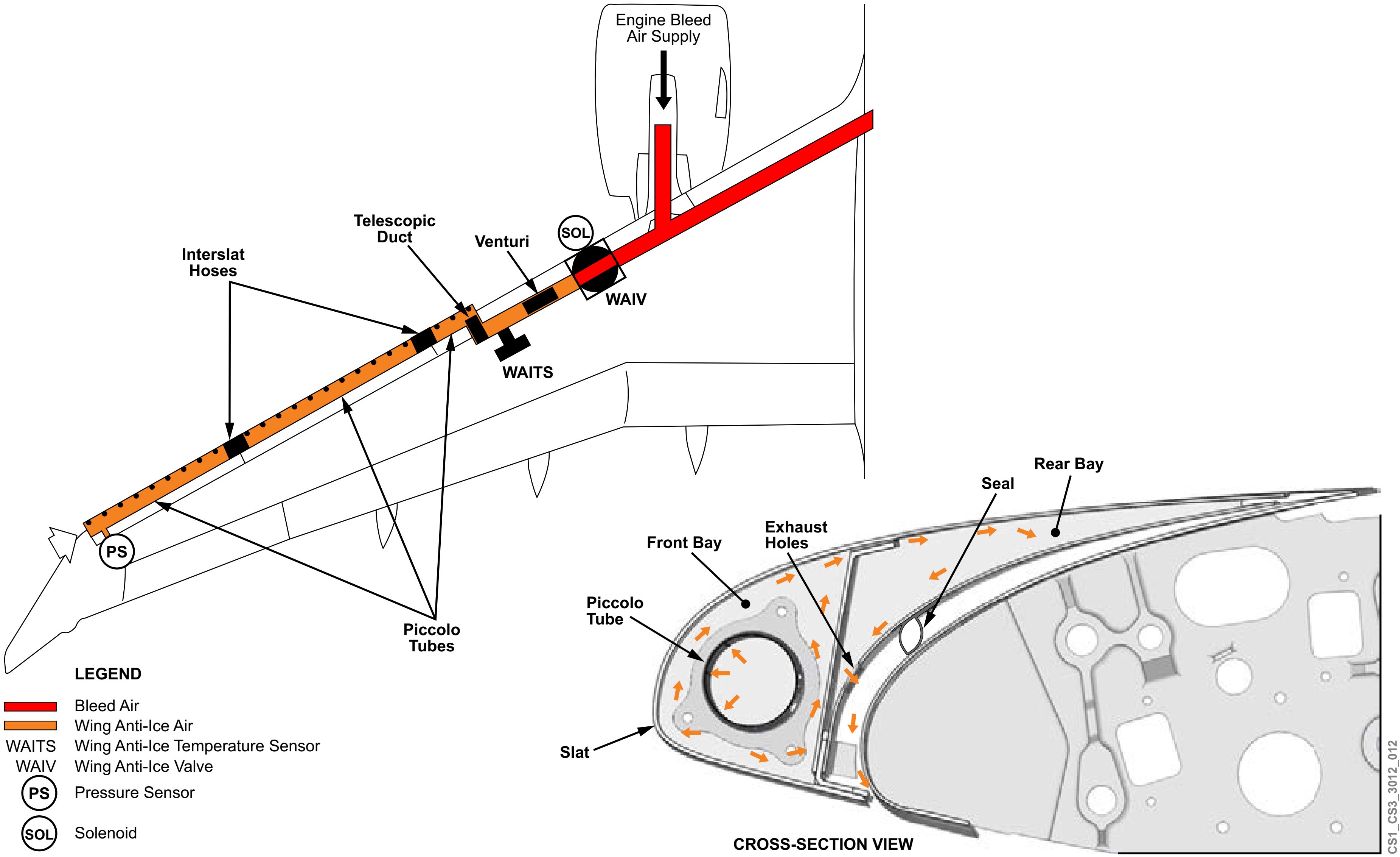


Figure 6: Wing Anti-Ice Air Supply

COMPONENT LOCATION

The wing anti-ice system (WAIS) consists of the following components:

- Wing anti-ice valves (WAIVs)
- Venturis
- Telescopic ducts
- Piccolo tubes
- Wing anti-ice temperature sensors (WAITSs)
- Pressure sensors (PSs)
- Integrated air system controllers (IASCs) (Refer to figure 8).

WING ANTI-ICE VALVES

Each wing anti-ice valve (WAIV) is located in the wing leading edge, outboard of the engine pylon, forward of the front spar.

VENTURIS

A venturi is located immediately downstream and outboard of the WAIV, in the wing leading edge cavity aft of slat no. 2.

TELESCOPIC DUCTS

Each telescopic duct is located in the wing leading edge cavity aft of slat no. 2.

PICCOLO TUBES

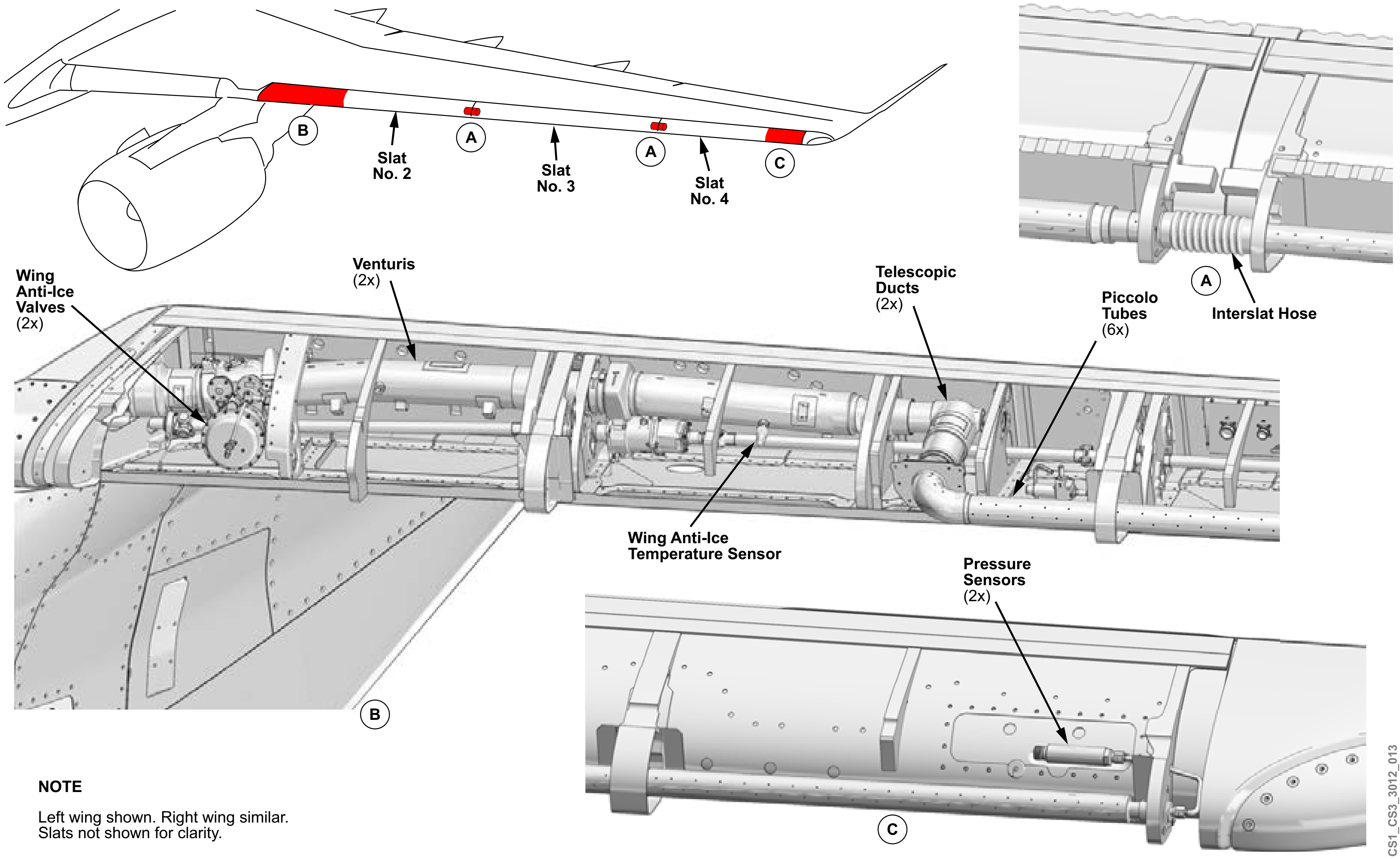
On each wing, slat no. 2, no. 3, and no. 4 each contain a piccolo tube. The piccolo tubes are joined by interslat hoses.

WING ANTI-ICE TEMPERATURE SENSORS

Each wing anti-ice temperature sensor (WAITS) is installed downstream of the venturi duct, before the telescopic duct.

PRESSURE SENSORS

Each pressure sensor is connected by a tube, to the outboard end of the piccolo tube assembly in slat no. 4.



NOTE

Left wing shown. Right wing similar.
Slats not shown for clarity.

Figure 7: Wing Anti-Ice System Leading Edge Components

INTEGRATED AIR SYSTEM CONTROLLERS

There are two integrated air system controllers (IASCs) located in the mid equipment bay.

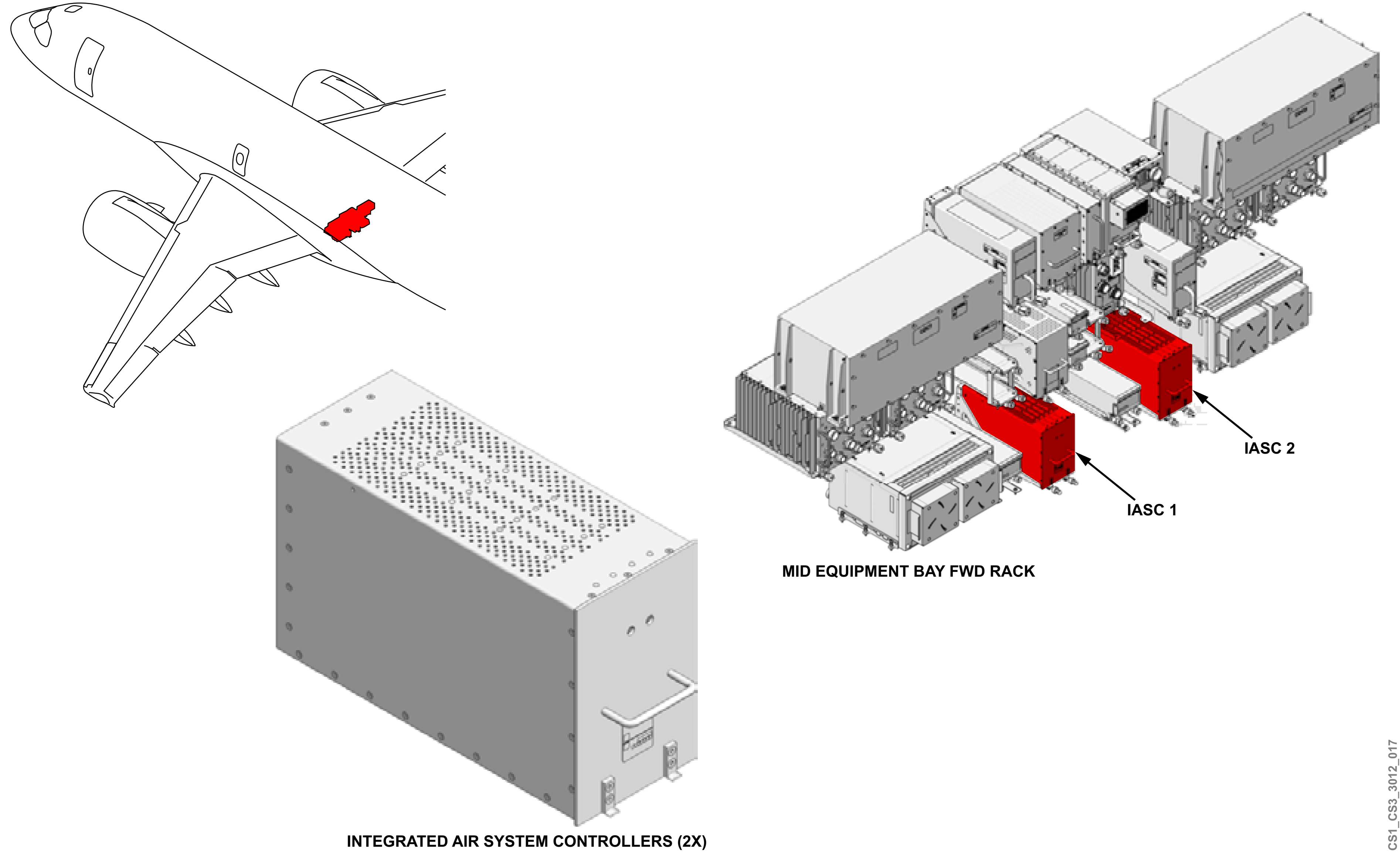


Figure 8: Integrated Air System Controllers

CONTROLS AND INDICATIONS

ANTI-ICE CONTROL PANEL

The wing anti-ice system (WAIS) has a three-position rotary switch located on the overhead panel of the flight deck. The switch is normally left in the AUTO position.

Figure 9: Wing Anti-Ice Controls

WING ANTI-ICE INDICATIONS

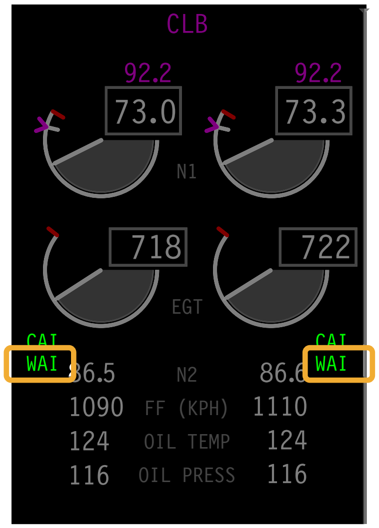
The wing anti-ice system (WAIS) indications and messages are displayed on the EICAS page and the AIR synoptic page.

EICAS Page

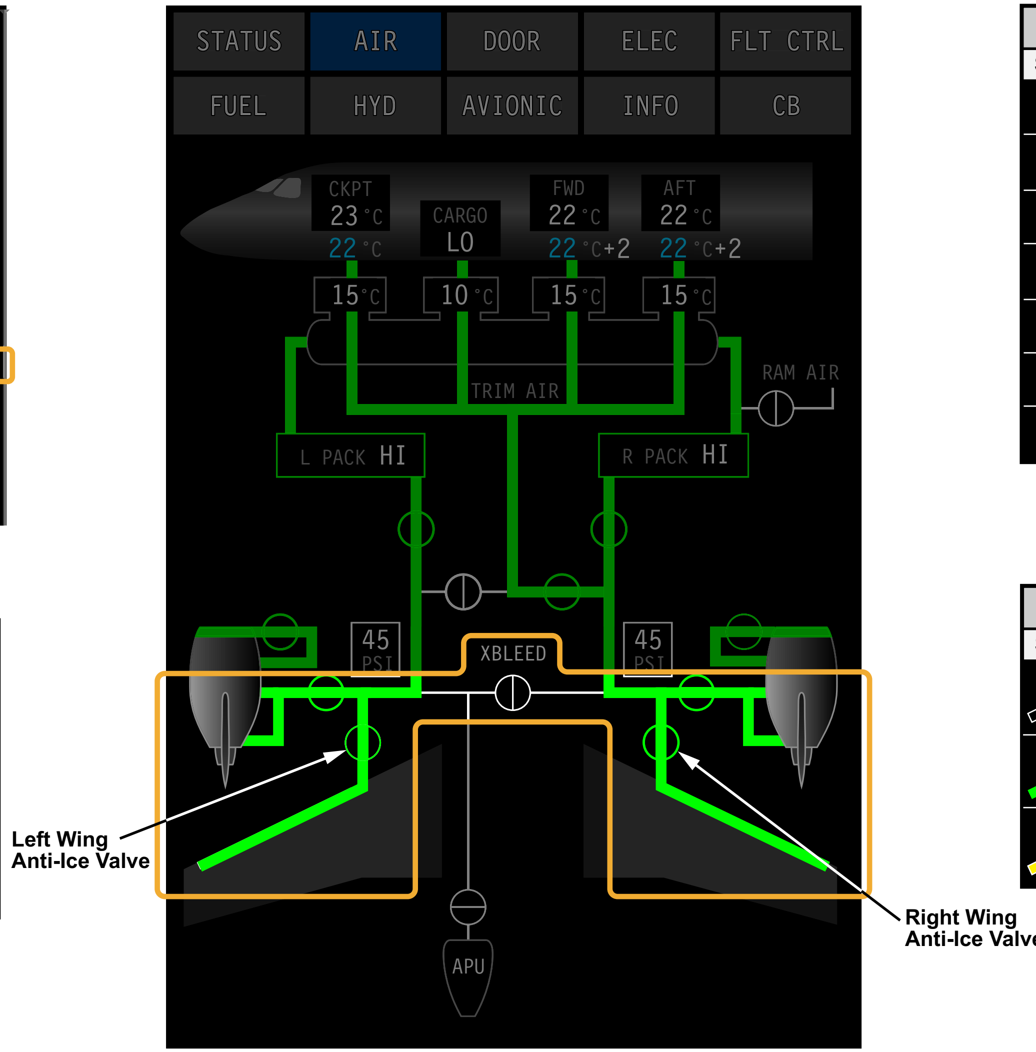
The wing anti-ice (WAI) icon is displayed on EICAS when the system is not OFF. The WAI icon is displayed in green when the system is activated, displayed in white when inhibited, or amber when a failure is reported.

AIR Synoptic Page

The status of the wing anti-ice valves (WAIVs) and the wing leading edge are displayed on the AIR synoptic page.



EICAS



SYNOPTIC PAGE – AIR

WING ANTI-ICE SYSTEM	
Symbol	**Condition**
WAI	Wing anti-ice activated
WAI	Wing anti-ice inhibited
WAI	Wing anti-ice caution failure or overheat
WAI	WAI failed in warning condition

WING ANTI-ICE VALVE	
Symbol	**Condition**
	Open with flow
	Closed
	Open with no flow (normal)
	Open with no flow (fail)
	Open with flow (fail)
	Invalid
	Closed (fail)

WING ANTI-ICE STATE	
Symbol	**Condition**
	Off
	Normal
	Abnormal

CS1_CS3_3012_001

Figure 10: Wing Anti-Ice Indications

DETAILED DESCRIPTION

INTEGRATED AIR SYSTEM CONTROLLERS

The integrated air system controllers (IASCs) control and monitor wing anti-ice system (WAIS) operation using position sensing of the wing ant-ice valve (WAIV) (full closed), wing anti-ice (WAI) temperature and pressure.

Each IASC has three channels: channel A, channel B, and a safety channel. Channel A is not used by the WAIS.

The WAIS uses the IASC to:

- Inhibit or enable activation
- Monitor WAIS pressure and temperature
- Monitor WAIS component health

When the aircraft is powered, the active IASC is determined from odd dates for IASC 1 and even dates for IASC 2. Channel B of the active IASC for the day is assigned to WAIS activation.

The safety channels of both IASCs, in conjunction with channel B of the standby IASC, monitor WAIS temperature and pressure and provide signals to the EICAS, through the data concentrator unit module cabinets (DMCs).

The standby IASC also monitors WAIS component health by performing a continuous built-in test (CBIT). Data from CBIT is reported to the onboard maintenance system (OMS) through the DMCs

The operation of the WAIS is shown schematically on the AIR synoptic page. Synoptic page data comes from the safety channel through the DMCs.

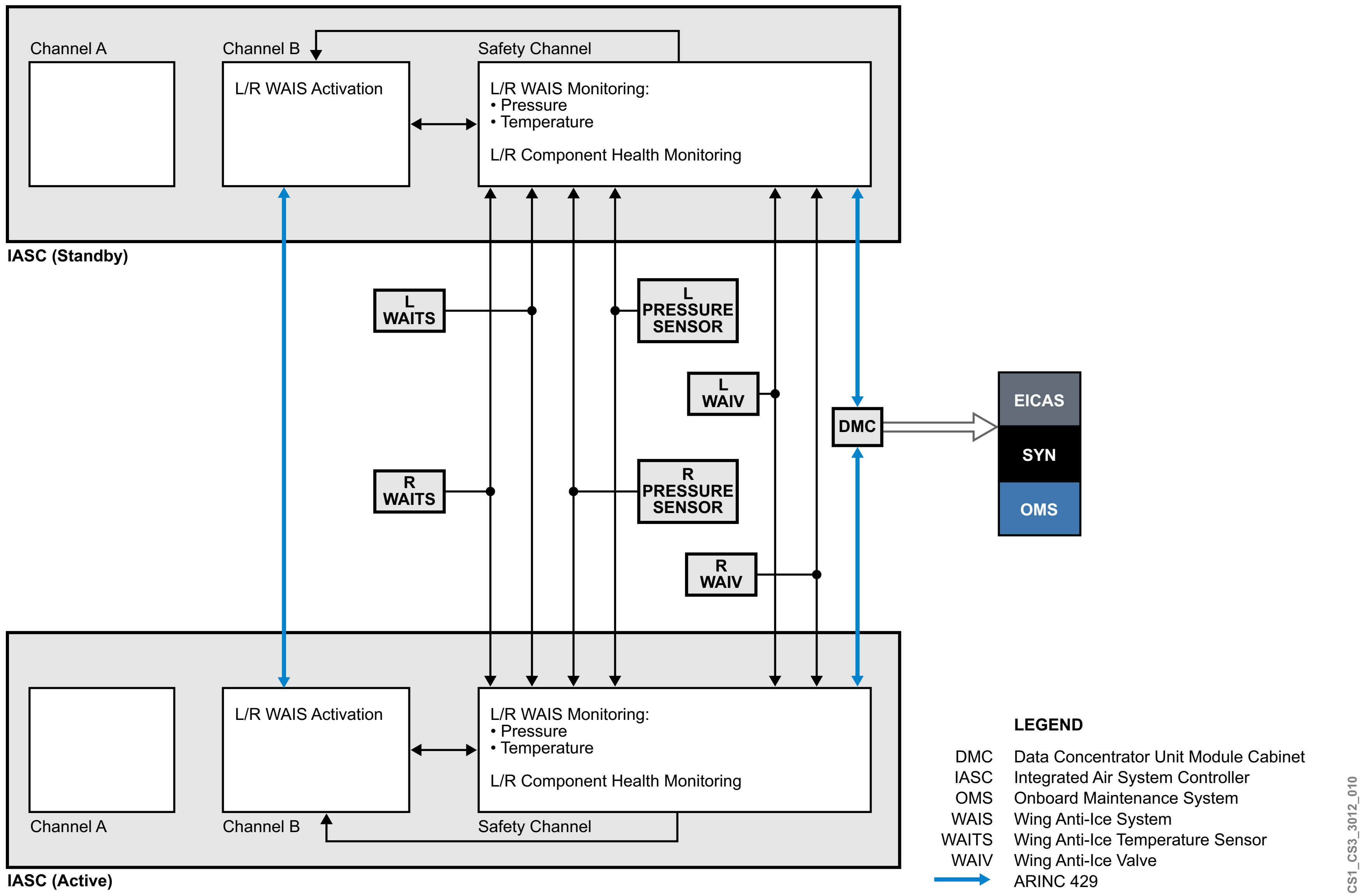


Figure 11: Integrated Air System Controller

WING ANTI-ICE OPERATION

The wing anti-ice system (WAIS) uses precooled engine bleed air, sprayed by the piccolo tubes onto the internal surface of the wing leading edge slats to prevent ice buildup.

The bleed air system controls engine bleed air supply pressure and temperature to the WAIS. The supply bleed air temperature target at the bleed temperature sensor (BTS) is nominally 227°C (440°F) in flight. On the ground, during initiated built-in test (IBIT), the temperature target is 200°C (392°F).

The WAIV controls the pressure of bleed air to the wing leading edge slats. Wing anti-ice temperature sensors (WAITSs) are used by the integrated air system controller (IASC) to monitor the bleed air temperature, downstream of the venturis.

Downstream of the wing anti-ice valve (WAIV), the hot bleed air passes through the venturi and telescopic duct to reach the slat no. 2 piccolo tube, then to slat no. 3 and no. 4 through interslat hoses. The area between the slats is not heated.

The purpose of the venturi is to choke the air flow in case of a burst or crack in the downstream ducting.

The dual-channel WAITS monitors temperature downstream of the WAIV. If a high-temperature condition is detected that exceeds 254°C (489°F) for more than 30 seconds, a WING ANTI-ICE OVHT caution message is displayed on EICAS for the associated wing.

If a low-temperature condition of less than 200°C (392°F) is detected by the WAITS for more than 60 seconds, a WING ANTI-ICE LO HEAT caution message is displayed on EICAS for the associated wing. The actual low-temperature threshold is a function of aircraft altitude and pressure reported by the pressure sensor.

The pressure sensor detects pressure at the end of the piccolo tubes of slats no. 4. A low-pressure condition of less than 14 psi for more than 15 seconds generates a WING ANTI-ICE FAIL warning message on EICAS for the associated wing. If the low-pressure condition persists and results in a low-temperature condition on the leading edge, the WING ANTI-ICE LO HEAT caution message replaces the previous warning message.

If an overpressure condition of more than 21 psi for more than 15 seconds (indicative of a WAIV failed open condition) is detected by the pressure sensor (PS), a WING ANTI-ICE FAIL warning message is displayed on EICAS for the associated wing. This message also appears if the WAIV if not closed when on the ground.

If the WAIV position does not indicate full closed with the WAIS off, a WING ANTI-ICE FAIL ON caution message is displayed on EICAS for the associated wing.

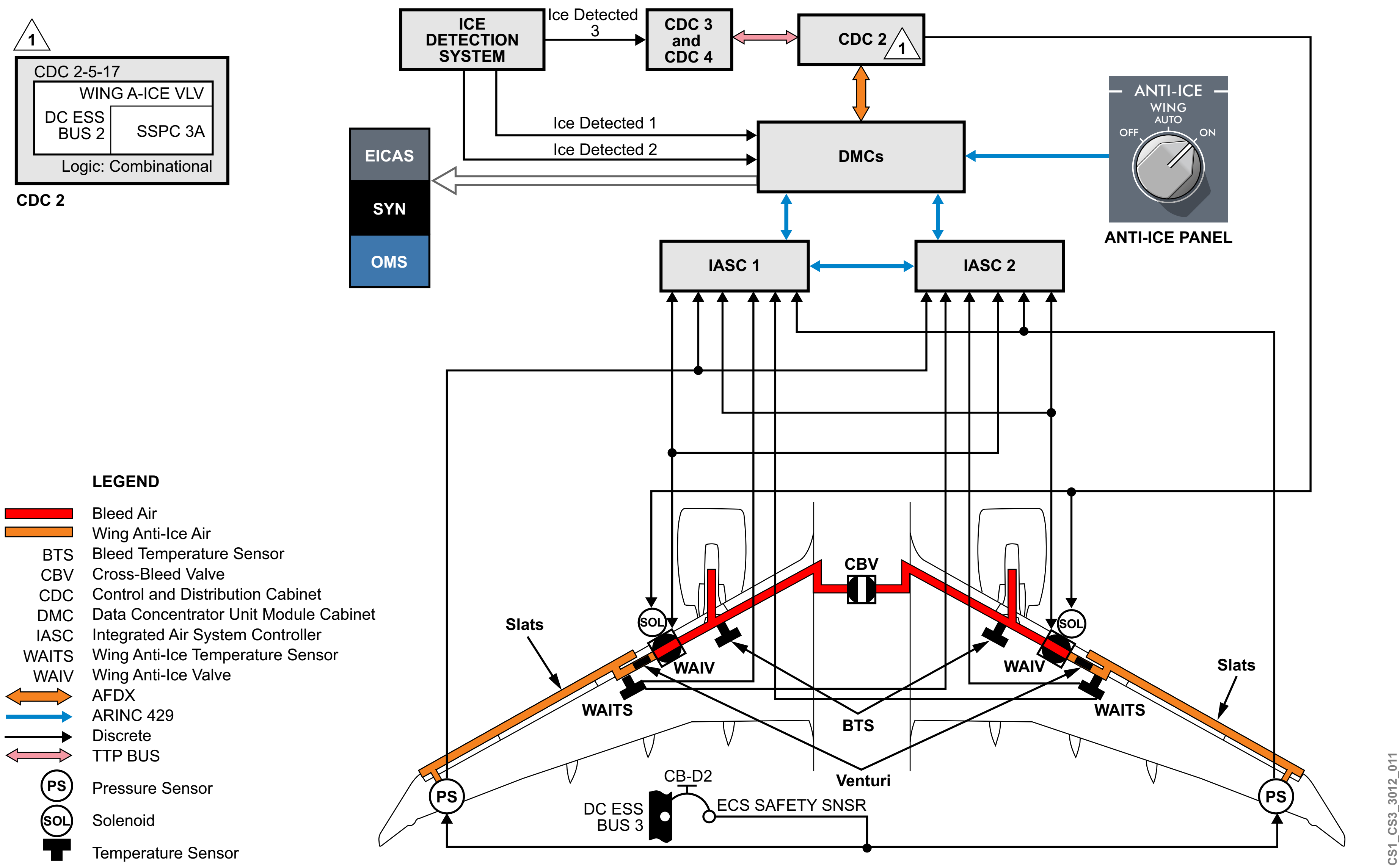


Figure 12: Wing Anti-Ice System Schematic

WING ANTI-ICE ACTIVATION CONTROL

The WAIVs are commanded to simultaneously open when they receive 28 VDC from control and distribution cabinet 2 (CDC 2) and controlled by advanced logic.

The combinational logic within CDC 2 receives input from the following sources:

- An enable/inhibit signal from the IASCs via the DMCs
- The WING anti-ice switch
- Ice detection signals from the ice detection system

With the WAIS switch in the ON position, the system turns ON:

- On ground:
    - Airspeed < 60 kt and WAIS test selected
    - Airspeed > 60 kt
- In flight:
    - Anytime the switch is in ON position

With WAIS switch in AUTO position, and ice is detected, the system turns ON:

- On ground:
    - Never
- In flight, in either of the following conditions:
    - Altitude > departure airport altitude + 457 m (1,500 ft)
    - Takeoff and 2 minutes after airspeed > 60 kt

The WAIS is automatically deactivated when any of the following occurs:

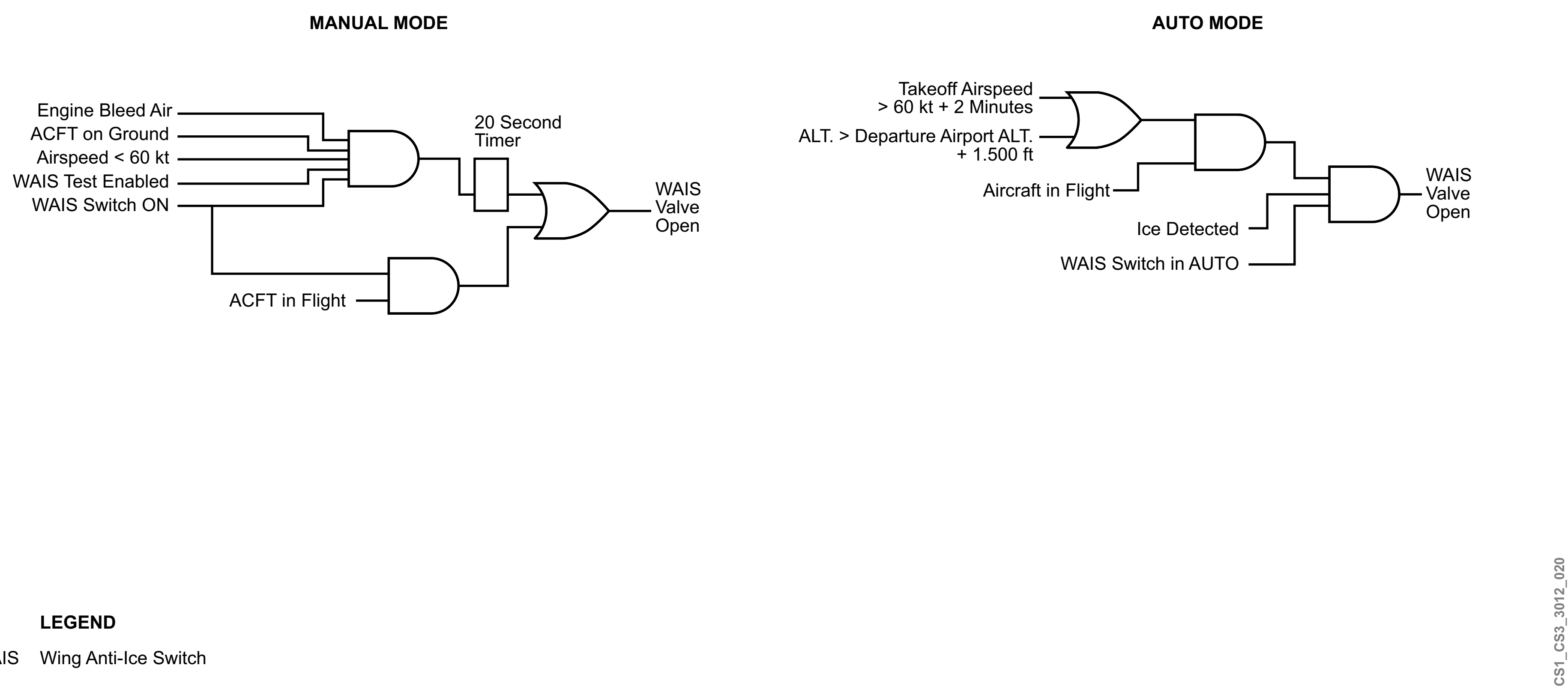
- No engine bleed is available (one engine minimum required)
- Main engine start (MES)
- Bleed air leak detected by bleed air leak and overheat detection system (BALODS)
- Wing anti-ice (WAI) switch selected OFF

The WAIVs are closed 2 minutes after the ICE DETECT signal is no longer present.

When the WAIS rotary switch is set to ON or AUTO and neither engine is running, a WING A/ICE MISCONFIG caution message is displayed on EICAS.

Any loss of redundant or non-critical function results in a WING A/ICE FAULT advisory message.

When in flight, the WAIS can be safely operated below 12°C (53°F) total air temperature (TAT) or airspeed above 60 kt. A WING A/ICE ON caution message is provided by the system when the system is ON and the TAT is above 15°C (59°F).



LEGEND

WAIS Wing Anti-Ice Switch

Figure 13: Wing Anti-Ice Activation Advanced Logic

MONITORING AND TESTS

SYSTEM TEST

A initiated built-in test (IBIT) is available on the AVIONICS synoptic page on the AVIO tab. The test can be carried out on ground, with the bleed air system pressurized by the engines. IBIT is limited to one cycle prior to takeoff to protect the leading edges from damage caused by excessive heat cycles. The airplane flight manual (AFM) limits the test to once per day. In case of test failure, the AFM limits the time between tests to a minimum of 30 minutes.

The WING anti-ice switch must be set to the ON position for the test. If the auxiliary power unit (APU) bleed air is being used, the IASCs automatically switch to engine bleed. The icon is grayed out if conditions are not correct for the IBIT.

When the WING A/ICE icon is pressed, the IASCs open the WAIVs and monitor the pressures obtained. The test period is limited to 20 seconds, after which the WAIVs close.

The initiated built-in test (IBIT) results are displayed on the synoptic page as either PASS, FAIL, or FAULT.

IBIT does not activate if a failure is detected and not rectified, indicated by the FAIL message. After failure rectification, the test can be re-initiated by resetting the IASCs. A wait time of 30 minutes is required to ensure cooling of the wing slats before repeating the test.

A display of ----- indicates the system had communication problems during the test.

Additionally, the IASCs conduct a power-up built-in test (PBIT) when power is first available on the aircraft. This check determines the validity of the two pressure sensor signals.

TEST STATUS	
Status	**Condition**
PASS	Test successfully completed
FAULT	Test failure with fault message
FAIL	Test failure
IN PROG	Test in progress
-----	Test invalid or aborted

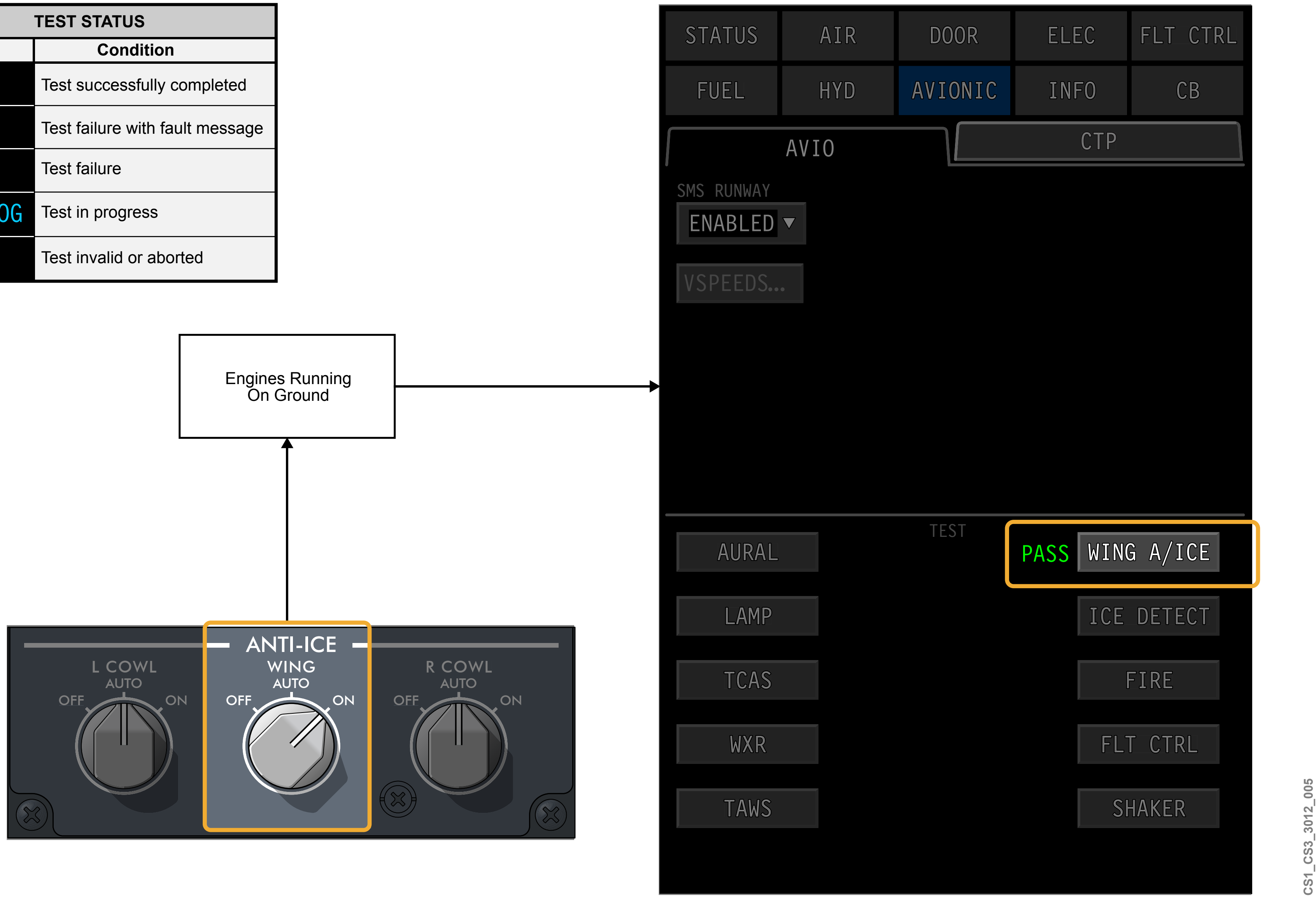


Figure 14: Wing Anti-Ice System Test

CAS MESSAGES

The following page provides the crew alerting system (CAS) and INFO messages associated with the wing anti-ice system (WAIS).

Table 3: WARNING Messages

MESSAGE	LOGIC
L WING A/ICE FAIL	L WAI not closed in any condition on ground out of test sequence, overpressure or low-pressure.
R WING A/ICE FAIL	R WAI not closed in any condition on ground out of test sequence, overpressure or low-pressure.

Table 4: CAUTION Messages

MESSAGE	LOGIC
L WING A/ICE OVHT	Left wing bleed overtemperature detected.
R WING A/ICE OVHT	Right wing bleed overtemperature detected.
L WING A/ICE LO HEAT	Low combination of pressure and temperature for L WAI (also function of altitude).
R WING A/ICE LO HEAT	Low combination of pressure and temperature for R WAI (also function of altitude).
WING A/ICE FAIL	Left or right WAI function failed.
WING A/ICE ON	Wing anti-ice selected ON and TAT above 15°C (59° F).
WING A/ICE MISCONFIG	WAI selected ON or WAI AUTO and ice detected with both engine bleeds OFF.

Table 5: ADVISORY Messages

MESSAGE	LOGIC
WING A/ICE FAULT	Loss of redundant or non-critical function for the wing anti-ice system or WAIV high leakage. Refer to INFO messages.

Table 6: STATUS Messages

MESSAGE	LOGIC
WING A/ICE ON	Wing anti-ice selected ON.
WING A/ICE OFF	Wing anti-ice selected OFF.

Table 7: INFO Messages

MESSAGE	LOGIC
30 WING A/ICE FAULT-WING A/ICE TEMP SNSR REDUND LOSS	Single failure of one WAITS channel.
30 L WING A/ICE LO HEAT - L WING A/ICE TEMP SNSR INOP	L WAITS drift low.
30 L WING A/ICE LO HEAT - CTRL TEMP INOP	L bleed temperature control too low for WAI.
30 L WING A/ICE LO HEAT - L HPV FAIL CLSD	L HPV failed closed leading to WAI low temperature.
30 L WING A/ICE OVHT - L WING A/ICE TEMP SNSR INOP	L WAITS drift high.
30 L WING A/ICE OVHT - CTRL TEMP INOP	L bleed temperature control too high for WAI.
30 R WING A/ICE LO HEAT - R WING A/ICE TEMP SNSR INOP	R WAITS drift low.
30 R WING A/ICE LO HEAT - CTRL TEMP INOP	R bleed temperature control too low for WAI.
30 R WING A/ICE LO HEAT - R HPV FAIL CLSD	R HPV failed closed leading to WAI low temperature.
30 R WING A/ICE OVHT - R WING A/ICE TEMP SNSR INOP	R WAITS drift high.

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Table 7: INFO Messages

MESSAGE	LOGIC
30 R WING A/ICE LO HEAT - CTRL TEMP INOP	R bleed temperature control too low for WAI.
30 WING A/ICE FAULT - WING A/ICE AUTO MODE INOP	Loss of ICE DETECTED signal at IASC input.
30 WING A/ICE FAULT - L WING A/ICE VLV POS SW INOP	Erroneous L WAIV switch acquisition on one channel.
30 WING A/ICE FAULT - R WING A/ICE VLV POS SW INOP	Erroneous R WAIV switch acquisition on one channel.
30 WING A/ICE FAIL - L WING PRESS FAIL	Left wing anti-ice overpressure or low-pressure.
30 WING A/ICE FAIL - R WING PRESS FAIL	Right wing anti-ice overpressure or low-pressure.
30 WING A/ICE FAULT - L WING A/ICE VLV LEAK	Left WAIV internal leakage.
30 R WING A/ICE FAIL - R WING A/ICE VLV LEAK	Right WAIV internal leakage.
30 L WING A/ICE FAIL - L WING A/ICE VLV FAIL OPEN	Left wing anti-ice failed open.
30 R WING A/ICE FAIL - R WING A/ICE VLV FAIL OPEN	Right wing anti-ice failed open.
30 WING A/ICE FAULT - L WING A/ICE PRESS SNSR INOP	Left outboard pressure sensor out of range or drifted.
30 R WING A/ICE FAIL - R WING A/ICE VLV LEAK	Right outboard pressure sensor out of range or drifted.

PRACTICAL ASPECTS

WING ANTI-ICE VALVE

For dispatch purposes, the wing anti-ice valve (WAIV) is equipped with a manual override provision to rotate the valve to the closed position. A deactivation screw locks the valve in the closed position.

When the deactivation screw is removed and used to secure the valve closed, air is bled from the pneumatic actuator line to prevent the valve from operating.

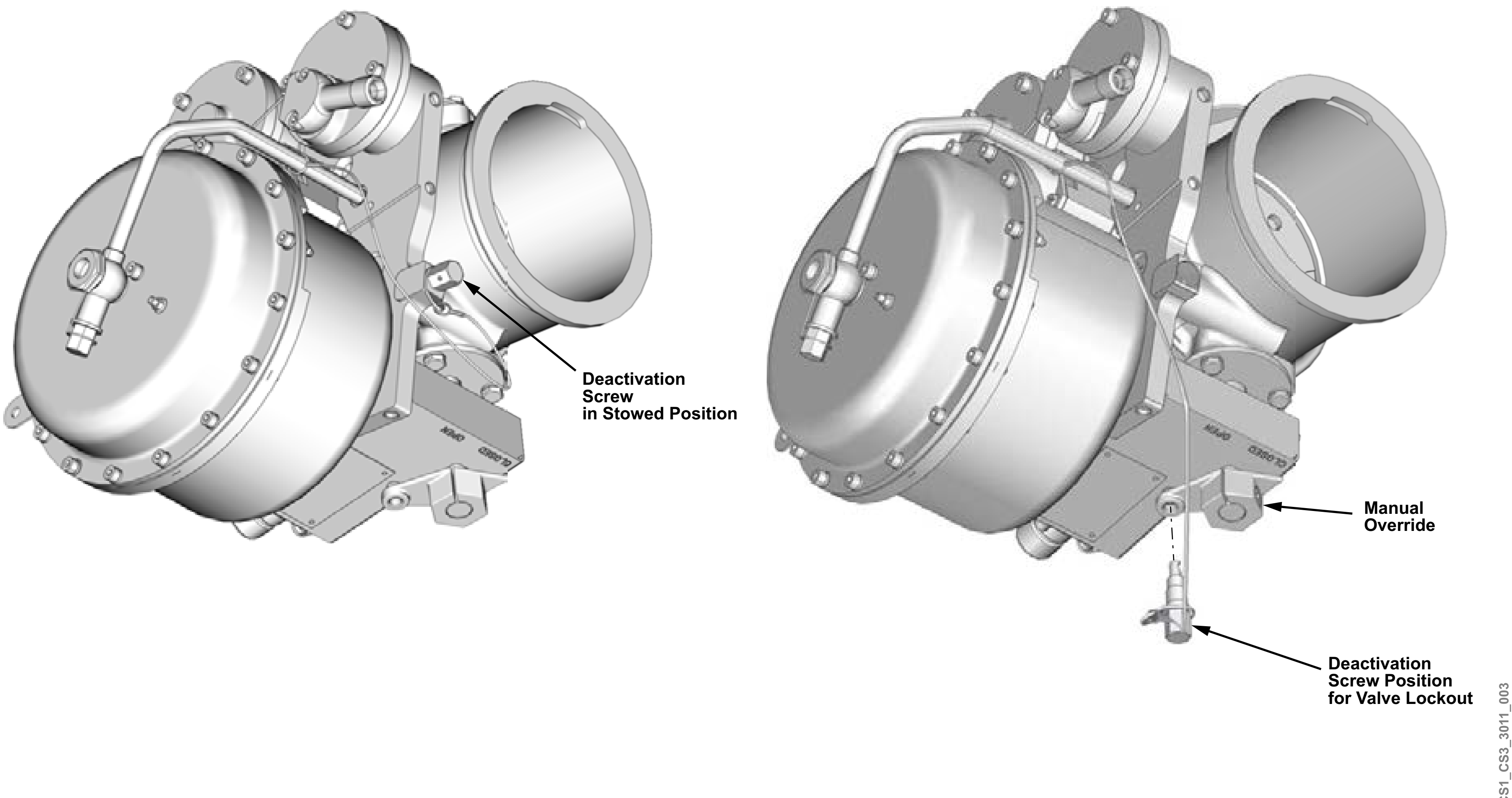


Figure 15: Wing Anti-Ice Valve

30-22 COWL ANTI-ICE SYSTEM

GENERAL DESCRIPTION

The cowl anti-ice system (CAIS) uses engine bleed air from the 6th stage compressor. The air is supplied to the engine inlet cowl, heating the leading edge and preventing ice accumulation.

The cowl anti-ice valves (CAIVs) provide a pressure regulating and shutoff function. The two valves are similar in design, and provide redundancy in case of the failure of a single valve, however, the CAIVs are not interchangeable and have different part numbers.

Each CAIV is a spring-loaded, failsafe open, poppet-type valve. Closure of the valve requires electrical power applied to the solenoid assembly and bleed air pressure. The valves are open when the solenoid assembly is electrically de-energized and/or there is a loss of bleed air pressure.

Each engine CAIS operates independently. The COWL rotary switches located on the ANTI-ICE panel, activates the CAIS when set to the AUTO position, or when selected to the ON position in manual mode.

Primary power for both electronic engine controls (EECs) is from their respective engine-driven permanent magnet generator (PMG) supplying 3-phase 34 VAC. Each EEC channel automatically switches power source to aircraft power source should the PMG fail. The left EEC is powered by DC ESS BUS 3 for channel A, and DC ESS BUS 1 for channel B. The right EEC is powered by DC ESS BUS 3 for channel A, and DC ESS BUS 2 for channel B.

The operational logic for the CAIS is contained in the EEC, operating on two channels, A and B. Further details on the EEC can be found in ATA 73 Engine Fuel and Control.

The EEC activates the system based on switch settings from the ANTI-ICE panel and the ice detection system signals. The CAIS operates when the EEC commands a pair of CAIVs to open.

A dual-element fan cowl temperature sensor in the fan cowl sends fan case area temperature to the EEC. This sensor is used to detect a temperature rise due to a possible CAIS duct rupture.

A dual-channel pressure sensor sends CAIS duct pressure to the EEC for monitoring CAIS operation.

When the engine reaches ground idle after starting, the EEC commands one of the CAIVs to close and monitors downstream bleed pressure, confirming its ON/OFF capability. Each CAIV is checked alternately every other flight.

System messages are sent by the EEC to the engine indication and crew alerting system (EICAS), and the onboard maintenance system (OMS) through the data concentrator unit module cabinets (DMCs). System operation is displayed schematically on the AIR synoptic page.

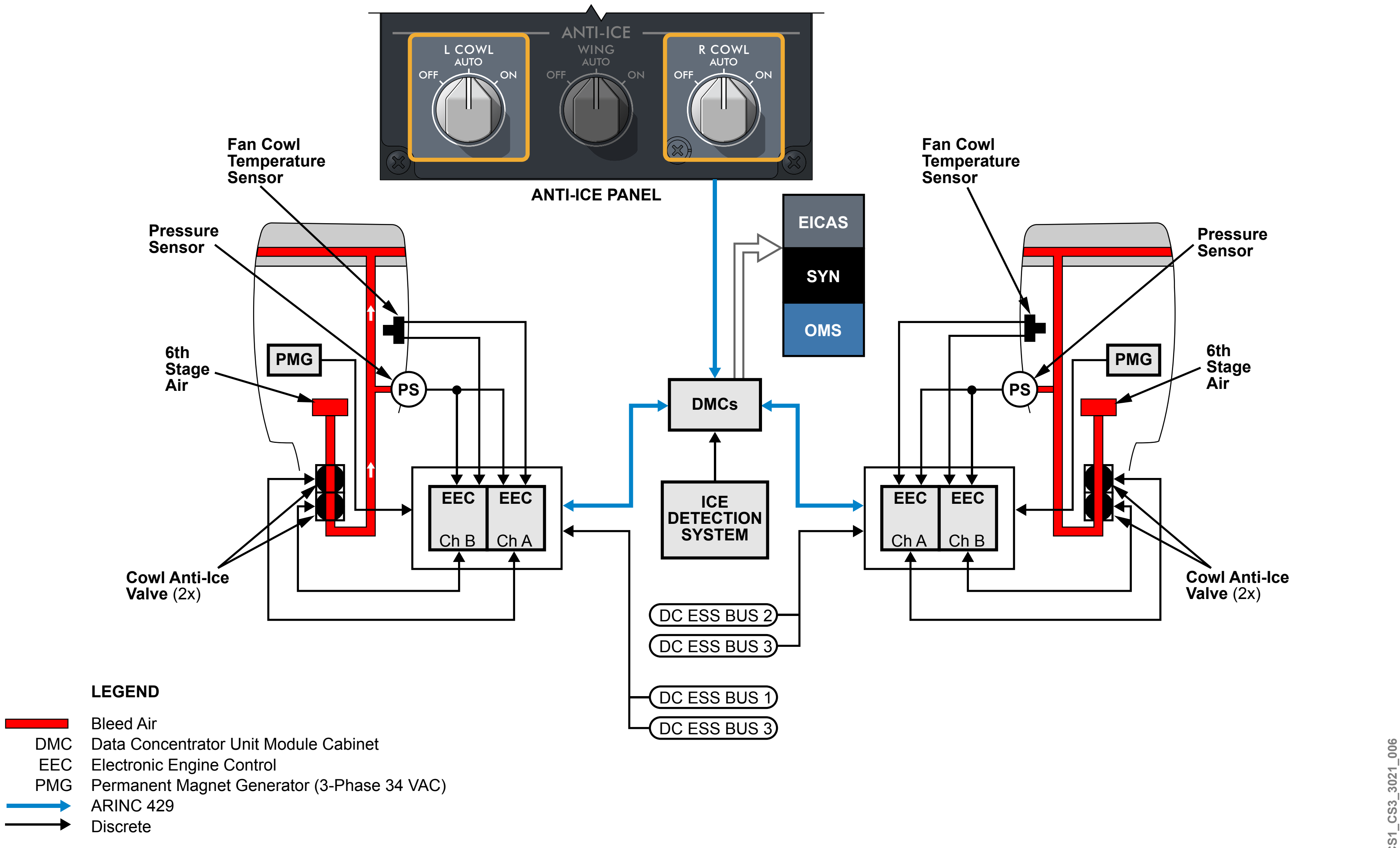


Figure 16: Cowl Anti-Ice System

COMPONENT LOCATION

The cowl anti-ice system (CAIS) consists of left and right subsystems each of which contain:

- Cowl anti-ice valves (CAIVs)
- Pressure sensor
- Fan cowl temperature sensor
- Swirl nozzle

COWL ANTI-ICE VALVES

The two cowl anti-ice valves (CAIVs) are mounted one above the other at the 6 o'clock position under the engine compressor case.

PRESSURE SENSOR

The pressure sensor is located on the engine fan case aft section, at the 5 o'clock position.

FAN COWL TEMPERATURE SENSOR

The fan cowl temperature sensor is located on the right side of the engine fan case at the 4:30 o'clock position.

SWIRL NOZZLE

The swirl nozzle duct is located within the engine inlet cowl.

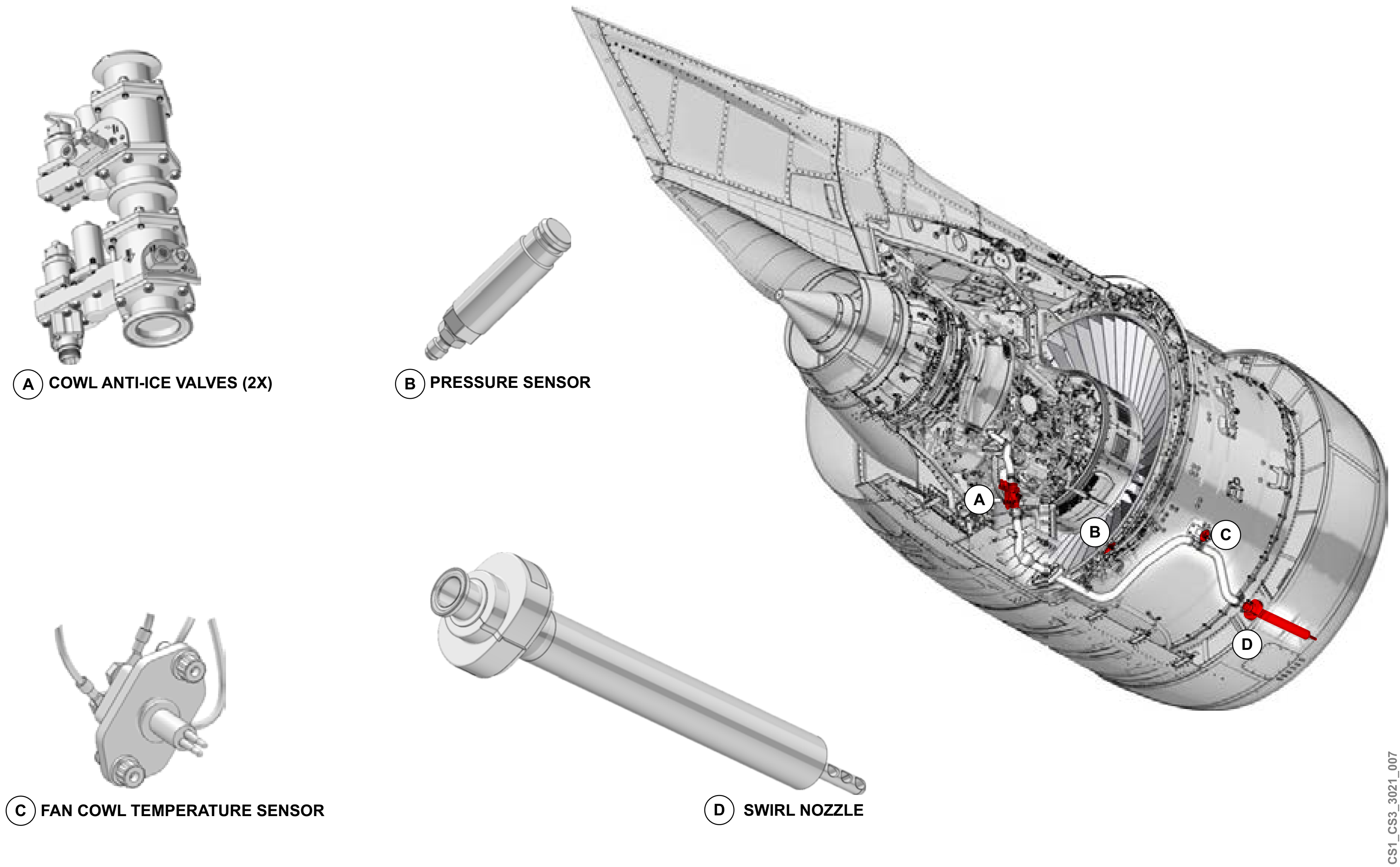


Figure 17: Cowl Anti-Ice System Component Location

DETAILED COMPONENT INFORMATION

SWIRL NOZZLE

The swirl nozzle has an inner wall and an outer wall. Hot air exiting the nozzle swirls within the inlet cowl, heating the surface of the inlet cowl lip to provide anti-icing.

In the event of a rupture to the inner wall, the outer wall contains the hot engine bleed air directing it backwards toward an insulation boot. The insulation boot is designed to allow hot air from a ruptured inner wall to blow back into the engine fan casing area. The rupture is detected by the fan cowl temperature sensors.

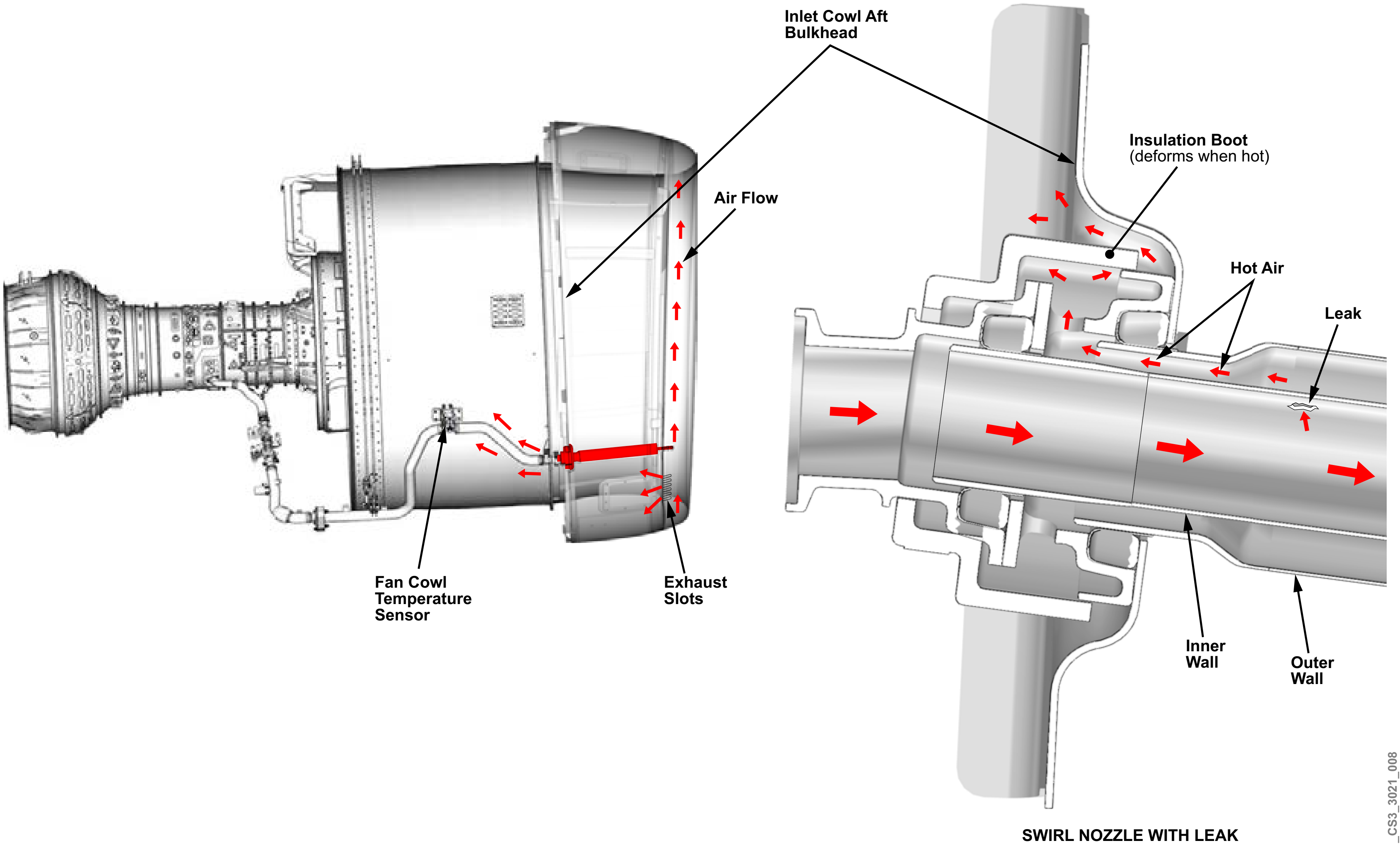


Figure 18: Swirl Nozzle

CONTROLS AND INDICATIONS

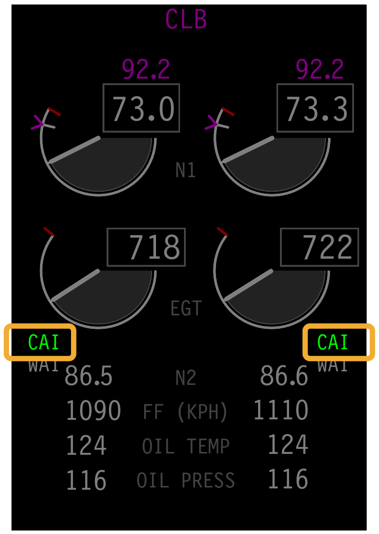
The cowl anti-ice system (CAIS) is controlled by two switches located on the anti-ice panel to provide separate controls for the left, and right systems.

Figure 19: Cowl Anti-Ice System Controls

INDICATIONS

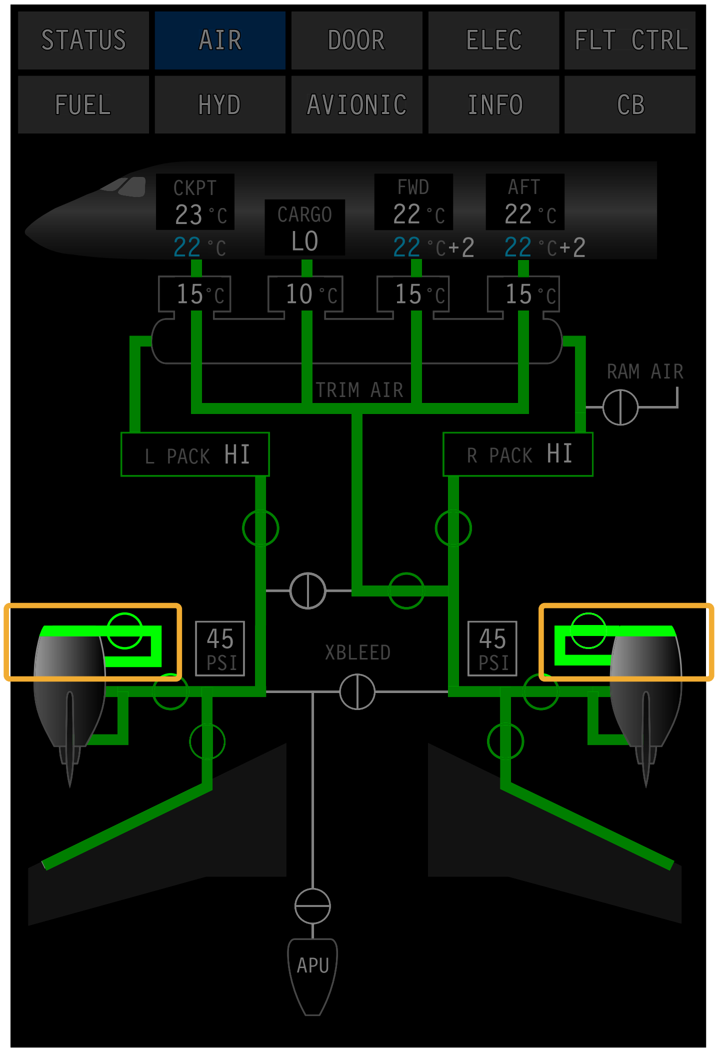
Indication of cowl anti-ice system (CAIS) operation is displayed on EICAS and the AIR synoptic page.

The cowl anti-ice (CAI) icon is displayed on EICAS to advise the crew of the system status. The CAI icon on EICAS is displayed in GREEN when the system is activated, displayed in WHITE when inhibited, or AMBER when a failure is reported.



EICAS

COWL ANTI-ICE ICON	
Symbol	Condition
CAI	Cowl anti-ice activated
CAI	Cowl anti-ice failure reported
CAI	Cowl anti-ice inhibited



SYNOPTIC PAGE – AIR

COWL ANTI-ICE VALVE	
Symbol	Condition
	Open with flow
	Closed
	Open with no flow (normal)
	Open with no flow (fail)
	Open with flow (fail)
	Invalid
	Closed (fail)

COWL ANTI-ICE STATE	
Symbol	Condition
	Off
	Normal
	Abnormal

CS1_CS3_3022_001

Figure 20: Cowl Anti-Ice System Indications

DETAILED DESCRIPTION

The cowl anti-ice system (CAIS) uses bleed air supplied from the 6th stage of the high-pressure compressor (HPC) to heat the engine inlet cowl lip.

When the CAIS is in operation, the cowl anti-ice valves (CAIVs) are commanded open allowing hot air flow to the inlet cowl. The hot bleed air swirls through the inlet cowl and is discharged overboard through the exhaust louvers.

If one of the two CAIVs remains closed, a L (R) COWL A/ICE FAIL caution message on the EICAS appears.

In flight, when the COWL switch is selected to ON, the CAIS is activated regardless of ice detection. If the outside air temperature (OAT) is greater than 15°C (60°F), the COWL ANTI-ICE ON caution message is displayed on EICAS for the associated engine.

On the ground, the CAIS is inhibited when selected to ON if the outside air temperature (OAT) is above 15°C (60°F).

In flight, when the COWL switch is selected to AUTO, the CAIS is activated when ice is detected. The CAIS is turned off when the ice detectors no longer signal ice (2 minutes after leaving icing conditions). If the ice signals are not valid, the electronic engine control (EEC) assumes icing conditions and opens the CAIVs.

On the ground in AUTO mode, CAIS is inhibited with takeoff thrust commanded. The EEC maintains the takeoff inhibit condition until either the aircraft has climbed 457 m (1500 ft) above departure airport elevation, or 2 minutes have elapsed since takeoff inhibit was set.

The EEC provides system data through the data concentrator unit module cabinets (DMCs) for EICAS messages, the AIR synoptic page and for the onboard maintenance system (OMS).

TEMPERATURE AND PRESSURE SENSING

The fan cowl inlet temperature sensor and the cowl anti-ice (CAI) pressure sensor on each engine send signals to EEC for monitoring.

In case of a rupture to the CAIS ducting, the affected temperature sensor sends signals to the EEC. The EEC generates an ENG NACELLE OVHT caution message in the EICAS only if the EEC is not able to close the CAIV. With the CAIV commanded closed due to overheat, the EEC continues to command it closed until the EEC is reset. Unless reset, once the CAIS system is required to provide anti-ice heat in ON or AUTO mode, the EEC generates a COWL A/ICE FAIL caution message on EICAS.

While in operation, if the CAIS pressure falls below 40 psi, the pressure sensor signal causes the EEC to generate a COWL A/ICE FAIL caution message. If CAIS pressure rises above 71 psi, an ENGINE FAULT advisory message is displayed.

Any time an engine is running, and the EEC does not detect valid ICE DETECTION signal, the EEC opens the CAIS valves.

During an engine start, both cowl anti-ice valves (CAIVs) are open until the engine reaches ground idle. Once the engine reaches ground idle, the electronic engine control (EEC) commands one of the CAIVs closed. This enables the EEC to verify system control. Each valve is checked alternately on every start. After the valve is checked, the EEC commands both valves to close.

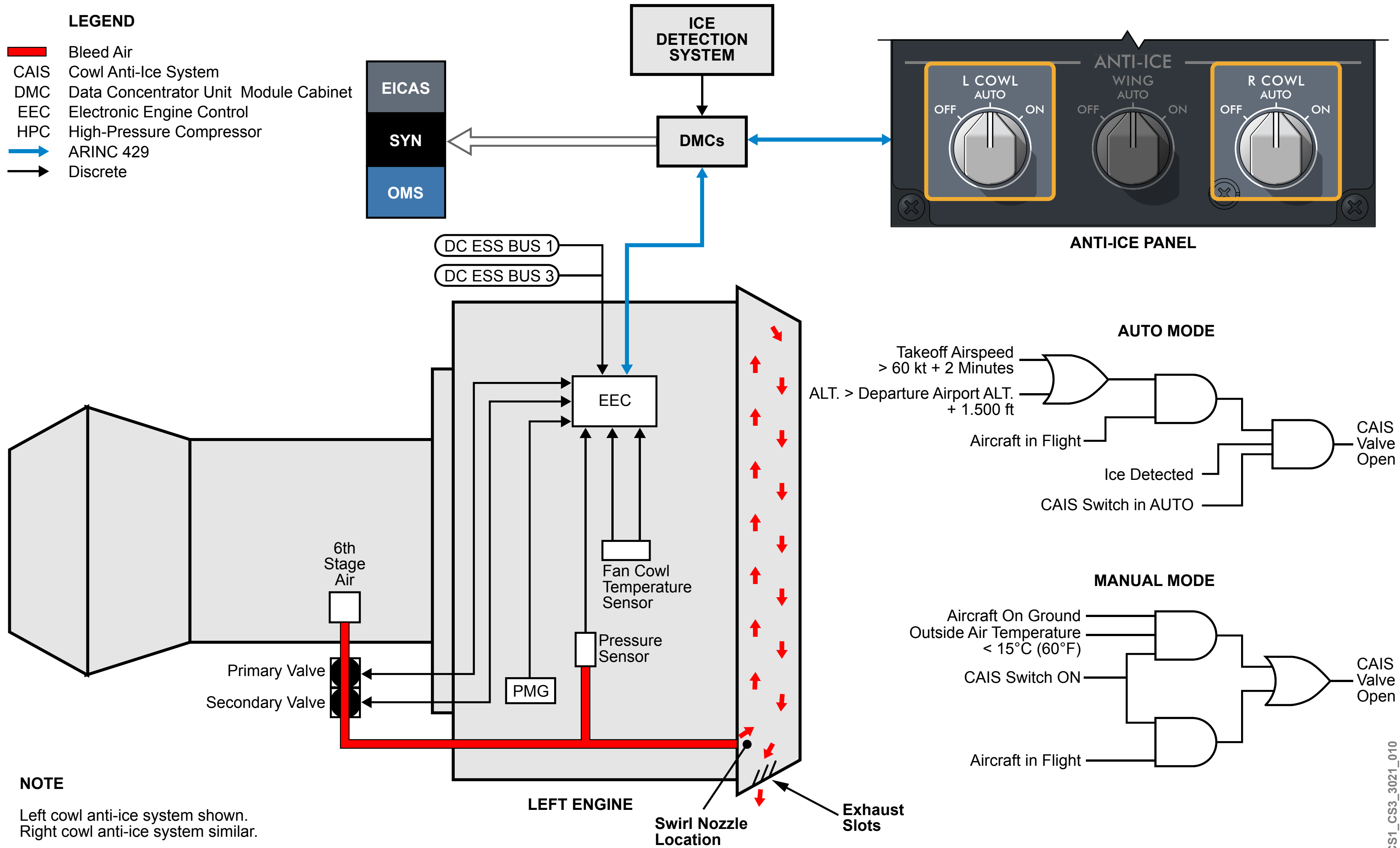


Figure 21: Cowl Anti-Ice System Detailed Description

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) messages associated with the cowl anti-ice system.

CAS MESSAGES

Table 8: CAUTION Messages

MESSAGE	LOGIC
L ENG NACELLE OVHT	Left burst cowl anti-ice duct or L HPC valve failed open or buffer air shutoff valve failed open.
R ENG NACELLE OVHT	Right burst cowl anti-ice duct or R HPC valve failed open or buffer air shutoff valve failed open.
L COWL A/ICE FAIL	Left cowl anti-ice system failed (valves closed).
R COWL A/ICE FAIL	Right cowl anti-ice system failed (valves closed).
L COWL A/ICE FAIL ON	Left cowl anti-ice system failed (valves open).
R COWL A/ICE FAIL ON	Right cowl anti-ice system failed (valves open).
COWL A/ICE ON	L or R COWL anti-ice manually selected ON while the OAT is above 15°C (59°F).

Table 9: ADVISORY Messages

MESSAGE	LOGIC
L ENGINE FAULT	Loss of redundant or non-critical function for the left engine.
R ENGINE FAULT	Right of redundant or non-critical function for right engine.

Table 10: STATUS Messages

MESSAGE	LOGIC
L COWL A/ICE ON	Left cowl anti-ice manually selected ON.
R COWL A/ICE ON	Right cowl anti-ice manually selected ON.
L-R COWL A/ICE ON	Left and right cowl anti-ice manually selected ON.
L COWL A/ICE OFF	Left cowl anti-ice manually selected OFF.
R COWL A/ICE OFF	Right cowl anti-ice manually selected OFF.
L-R COWL A/ICE OFF	Left and right cowl anti-ice manually selected OFF.

PRACTICAL ASPECTS

COWL ANTI-ICE VALVES

A manual override feature is provided on each valve to lock a failed valve in the open position for dispatch, which leaves the remaining valve for control.

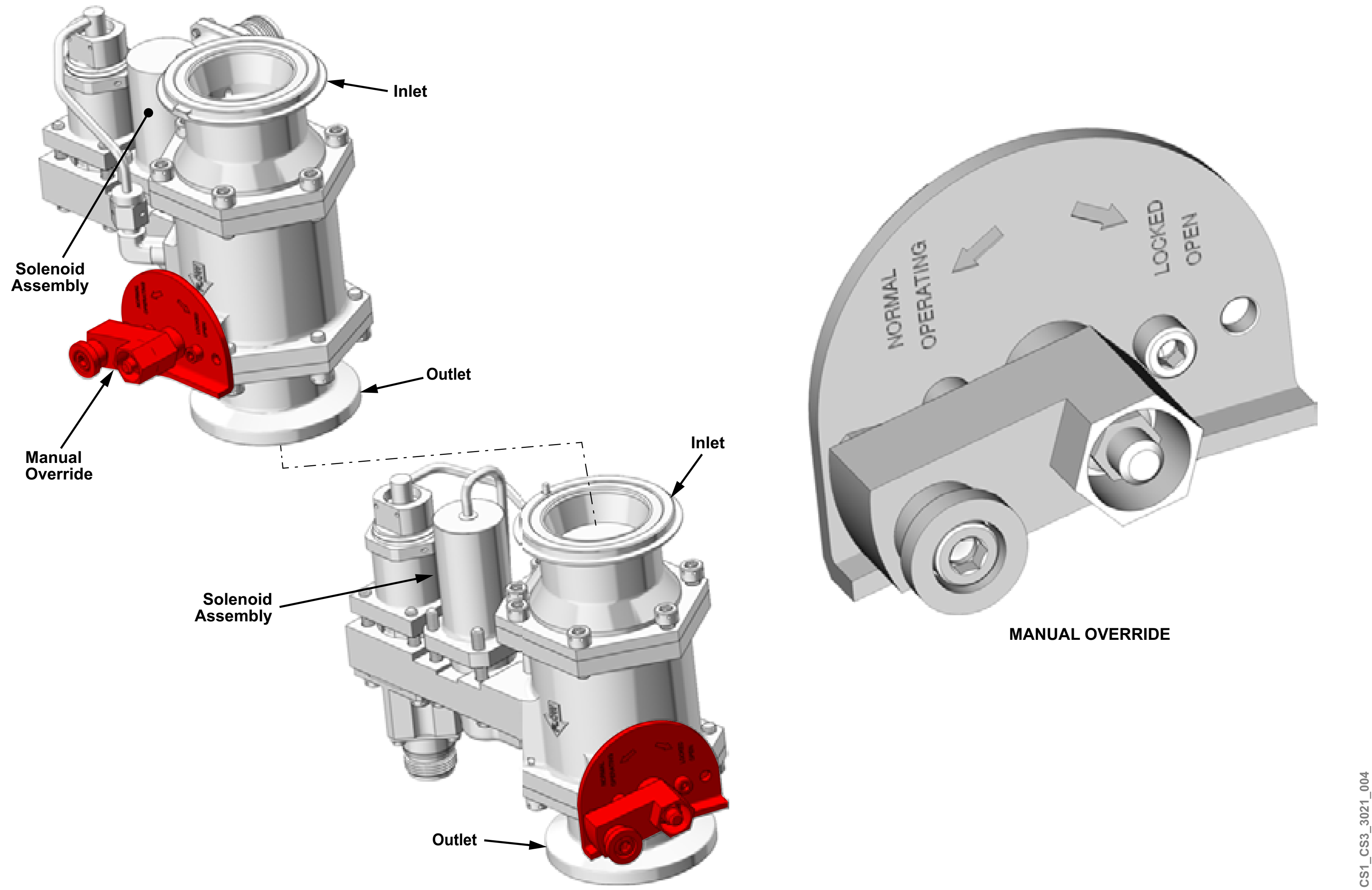


Figure 22: Cowl Anti-Ice Valves

34-10 PROBE HEAT SYSTEM

GENERAL DESCRIPTION

The probe heat system prevents ice accumulation on the:

- Angle-of-attack (AOA) vanes
- Air data system probes (ADSP)
- Total air temperature (TAT) probes

Each AOA vane has a vane heater and a case heater. For the left AOA vane, both heaters are powered by DC BUS 1. For the right AOA vane, both heaters are powered by DC BUS 2. These heaters are on whenever an engine is running or if the aircraft is in the air.

The ADSP and TAT probe heaters operate in two modes: self-regulating and full. The self-regulating mode provides power to these heaters once the engines are running and the aircraft is on the ground under 55 kt. In this mode the probe regulates its own temperature. When the aircraft is in the air the heaters switch to full mode, which provides unregulated heating of the probes.

The ADSP requires AC power for its primary and secondary power sources. Primary AC power is from AC BUS 1, AC BUS 2, and AC ESS BUS. Secondary power for ADSP 1 is AC BUS 2. For ADSP 2, secondary power is from AC BUS 1. The source of secondary power for ADSP 3 and 4 is from independent and dedicated static inverters which use power from the DC ESS BUS 3 to provide AC power.

The primary heating power source for both TAT probes is the AC ESS BUS via their ADSPs. The left TAT probe heater is controlled by ADSP 3. The right TAT probe heater is controlled by ADSP 4. Secondary power is provided directly from the secondary power source from associated ADSPs.

A PROBE HEAT pushbutton annunciator (PBA) powers the ADSP and TAT probe heaters for 2 minutes on the ground provided the auxiliary power unit (APU) is running or external power is connected to the aircraft.

WARNING

AS A SAFETY PRECAUTION, DO NOT TOUCH THE PROBES WITH BARE HANDS, AS THEY COULD BE HOT AND INFLICT BURNS.

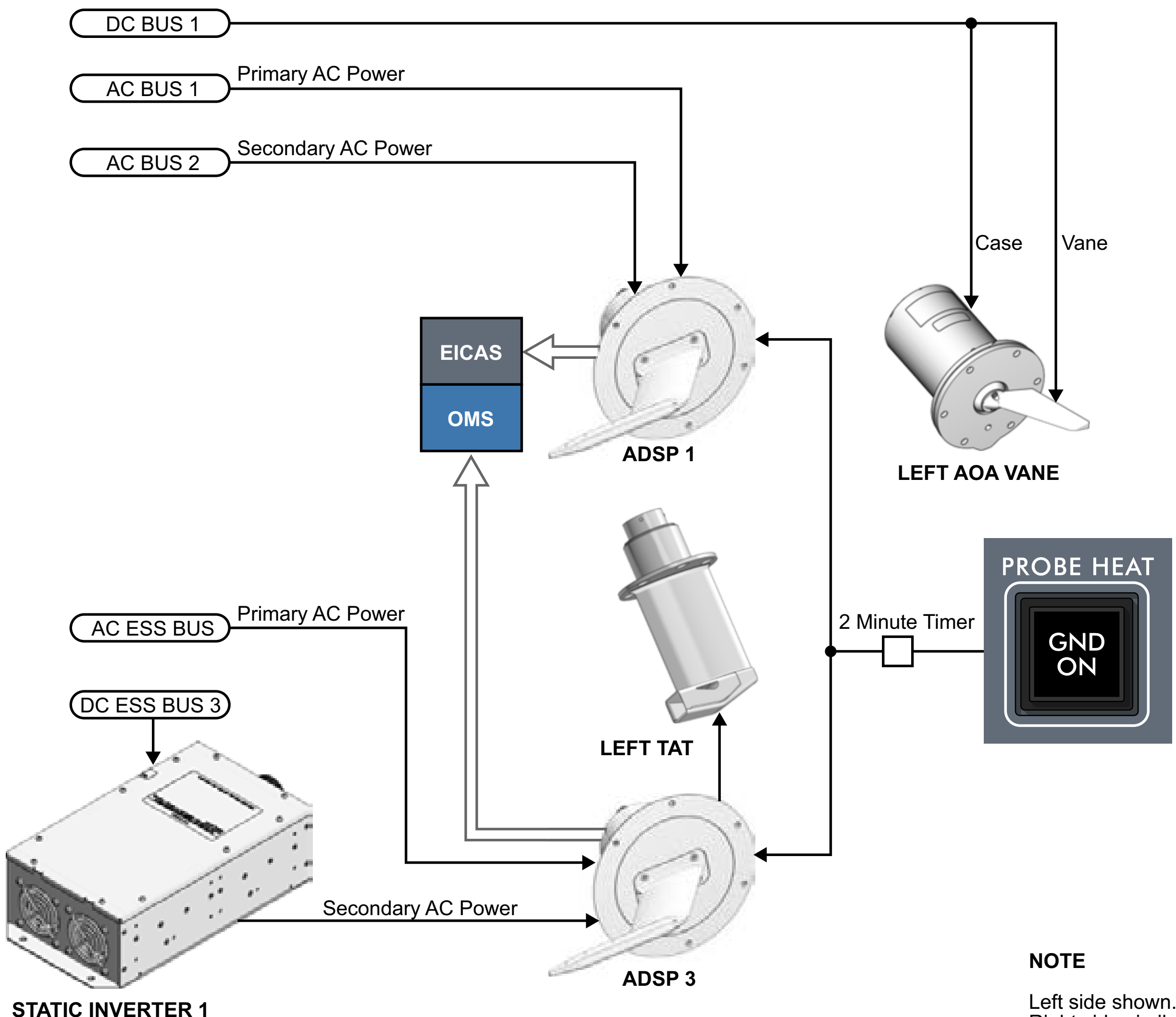
Table 11: Probe Heat Power Sources

PROBE	PRIMARY POWER	SECONDARY POWER
ADSP 1	AC BUS 1	AC BUS 2
ADSP 2	AC BUS 2	AC BUS 1
ADSP 3	AC ESS BUS	DC ESS BUS 3 via static inverter 1
ADSP 4	AC ESS BUS	DC ESS BUS 3 via static inverter 2
TAT L	ADSP 3	-----
TAT R	ADSP 4	-----
AOA L	DC BUS 1	-----
AOA R	DC BUS 2	-----

CS1_CS3_TB30.001

ADSP and TAT heater messages are displayed on the engine indication and crew alerting system (EICAS) and monitored by the onboard maintenance system (OMS).

The AOA vane heaters are not monitored.



LEGEND

ADSP	Air Data System Probe
AOA	Angle-of-Attack
EEC	Electronic Engine Control
TAT	Total Air Temperature

NOTE

Left side shown.
Right side similar.

Figure 23: Probe Heat System

COMPONENT LOCATION

The probe heat system consists of the following:

- Angle-of-attack (AOA) vanes
- Air data system probes (ADSPs)
- Total air temperature (TAT) probes
- Static inverters

ANGLE-OF-ATTACK VANE

There are two angle-of-attack (AOA) vanes, one on each side of the forward fuselage.

AIR DATA SYSTEM PROBE

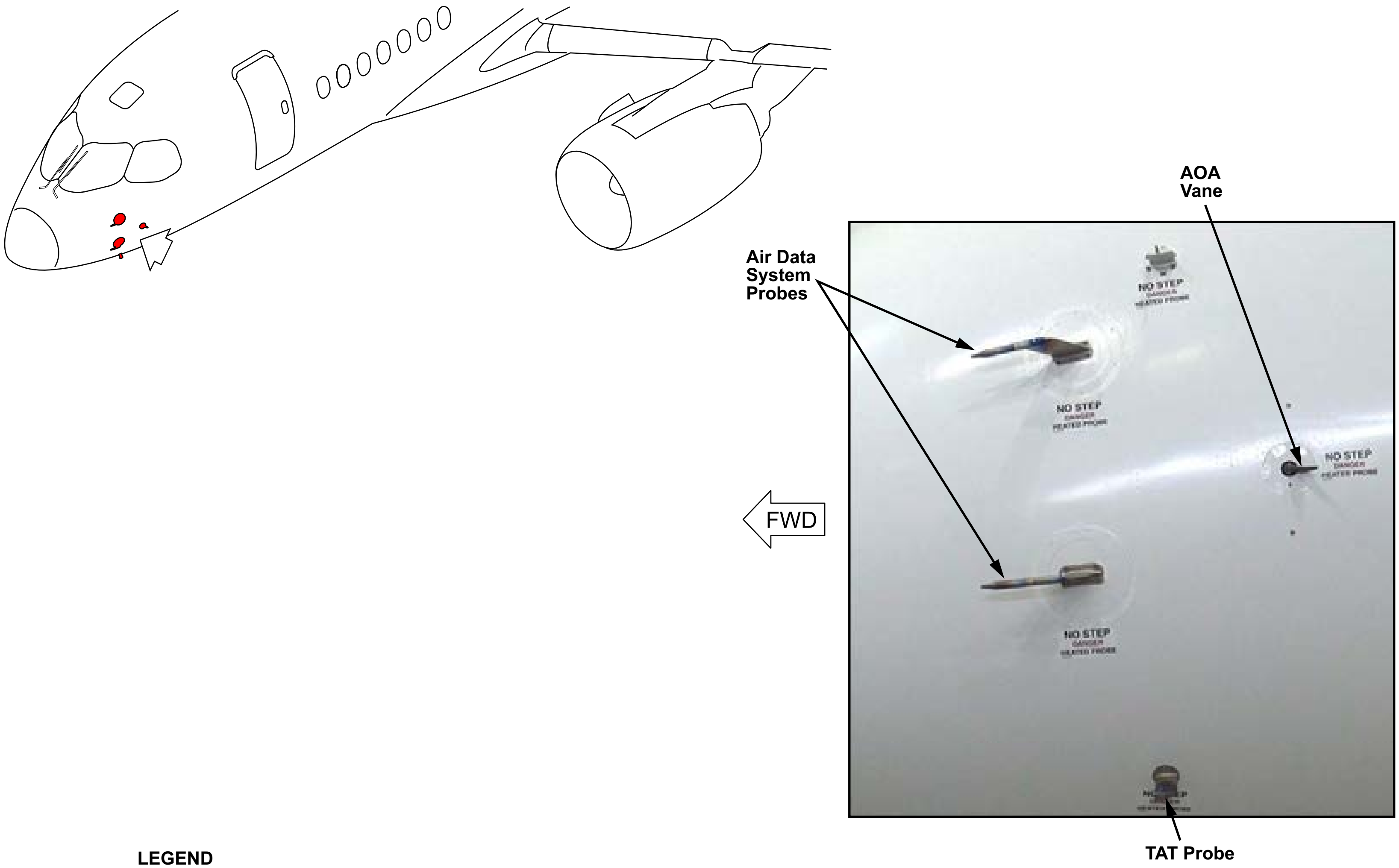
The are four air data system probes (ADSPs), two on each side of the forward fuselage.

TOTAL AIR TEMPERATURE PROBE

There are two total air temperatures (TATs) probes, one on each side of the forward fuselage.

STATIC INVERTER

There are two static inverters, one on each side of the forward equipment bay.



LEGEND

ADSP	Air Data System Probe
AOA	Angle-of-Attack
TAT	Total Air Temperature

NOTE

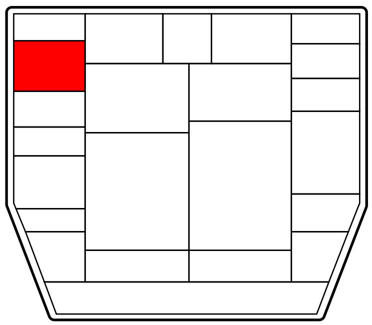
Left side probes shown.
Right side similar.

CS1_CS3_3030_005

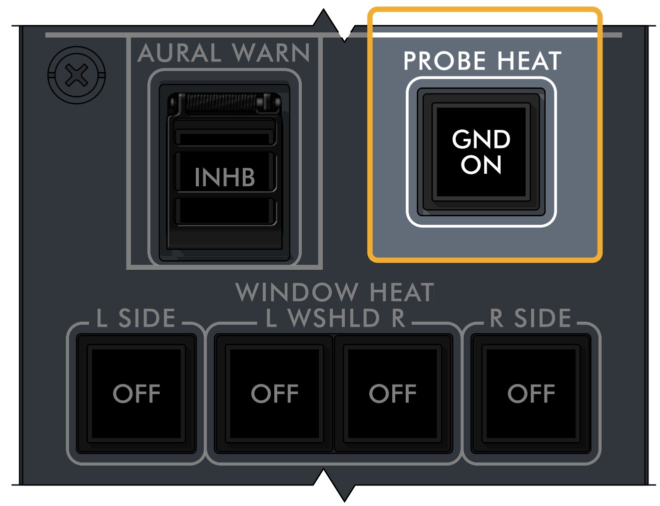
Figure 24: Probe Heat System Components

CONTROLS AND INDICATIONS

A PROBE HEAT pushbutton annunciator (PBA) powers the ADSP and TAT probe heaters for 2 minutes on the ground provided the auxiliary power unit (APU) is running or external power is connected to the aircraft.



OVERHEAD PANEL



CS1_CS3_3030_007

Figure 25: Probe Heat Control

DETAILED DESCRIPTION

AIR DATA SYSTEM PROBE

Each angle-of-attack (AOA) vane has a vane heater and a case heater. For the left AOA vane, both heaters are powered by DC BUS 1. For the right AOA vane, both heaters are powered by DC BUS 2. The control and distribution cabinets (CDCs) provide power to the heaters. The heaters turn on when electrical power is available on the associated bus, and at least one engine is running or the aircraft is in the air.

The air data system probes (ADSPs) use AC power for primary and secondary power sources. Primary AC power is from AC BUS 1, AC BUS 2, or AC ESS BUS. Secondary power for ADSP 1 is AC BUS 2. For ADSP 2, secondary power is from AC BUS 1. The source of secondary power for ADSP 3 and 4 is from static inverters which use 28 VDC power from the DC ESS BUS 3 to provide AC power.

AC power from AC BUS 1 and AC BUS 2 is provided through solid-state power controllers (SSPCs). AC power from the AC ESS BUS is through thermal circuit breakers. The static inverters receive 28 VDC through thermal circuit breakers on electrical power control 3 (EPC 3).

Power for the total air temperature (TAT) heaters is provided through the ADSP, from the AC ESS BUS.

The ADSP heaters and TAT probe heaters are switched off whenever the aircraft is on the ground and both engines are off. Otherwise these heaters operate in one of two modes: self-regulating or full mode.

The self-regulating mode is enabled once the engines are running, the aircraft is on the ground, and airspeed is less than 55 kt. The probe heat is regulated to maintain a lower probe temperature to avoid overheating.

The full mode is enabled when the airspeed is greater than 55 kt or the aircraft is in the air. The ADSP and TAT probe heaters operate in full mode if the controller integral heater fails.

When the PROBE HEAT pushbutton annunciator (PBA) is pressed, the white GND ON light illuminates and powers the probe heaters ON for 2 minutes and automatically turns off. The ADS PROBE HEAT GND ON info message is displayed on EICAS when the PROBE HEAT function is operating.

The PROBE HEAT function is inhibited on ground unless at least one engine is running and AC power is feeding the aircraft.

An ADSP PROBE HEAT FAIL caution message is displayed on EICAS if the associated probe heater is detected failed. The ADSP ISI PROBE HEAT caution message is displayed if both ADSP 3 and ADSP 4 heaters fail.

The ADSPs provide data on ADSP and TAT heater operation to EICAS and OMS through the data concentrator unit module cabinets (DMCs).

WARNING

ENSURE PROBE HEAT IS DEACTIVATED BEFORE JACKING THE AIRCRAFT.

PUTTING THE AIRCRAFT INTO THE AIR MODE AUTOMATICALLY ACTIVATES THE ADSP, TAT AND AOA PROBE HEATERS. ENSURE COVERS ARE REMOVED. HOT PROBES CAN CAUSE INJURY.

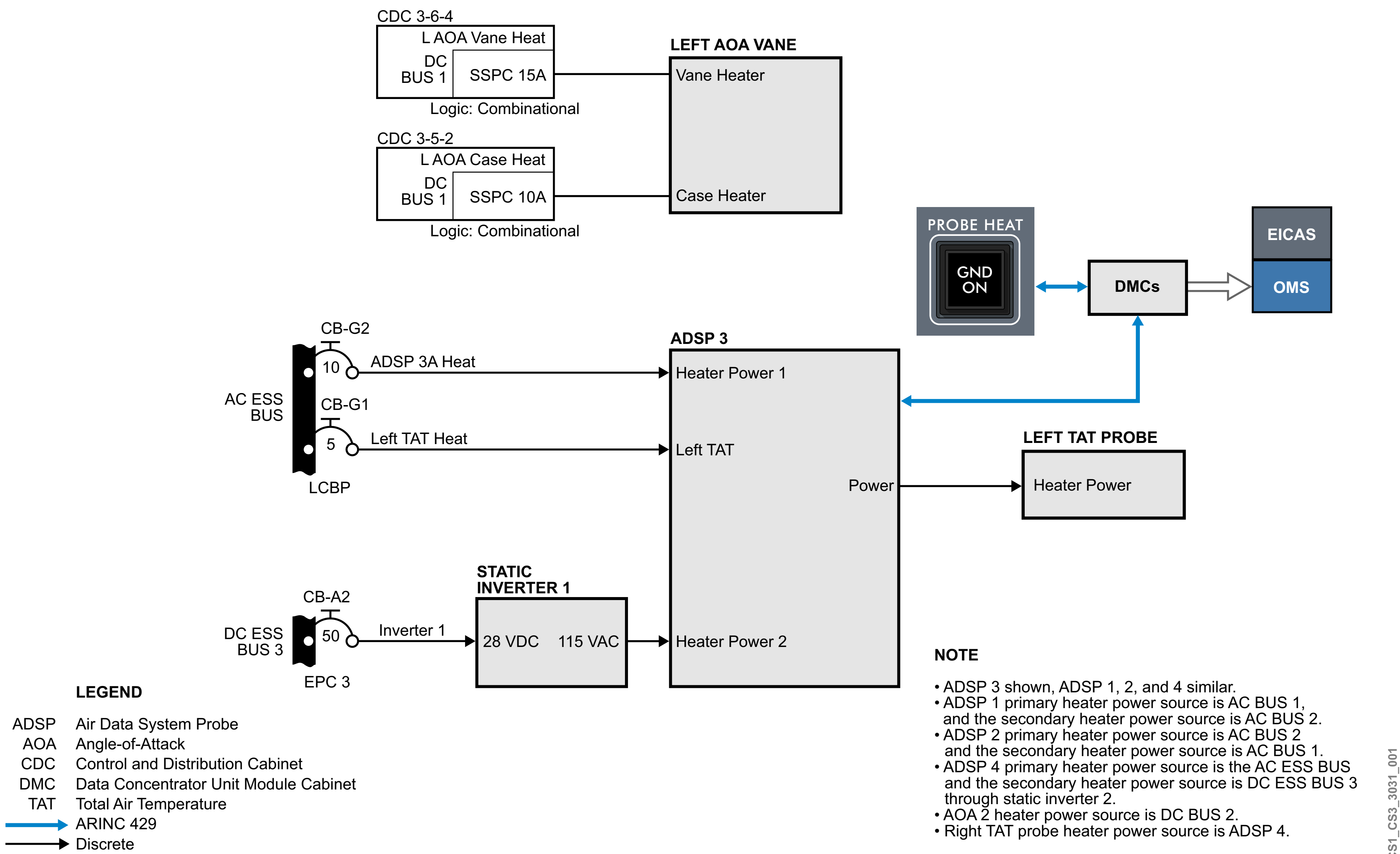


Figure 26: Probe Heat Electrical Interface Detailed Description

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages associated with the probe heat system.

CAS MESSAGES

Table 12: CAUTION Messages

MESSAGE	LOGIC
ADS 1 PROBE HEAT FAIL	ADS 1 probe not heated per expectations.
ADS 2 PROBE HEAT FAIL	ADS 2 probe not heated per expectations.
ADS 3 PROBE HEAT FAIL	ADS 3 probe not heated per expectations.
ADS 4 PROBE HEAT FAIL	ADS 4 probe not heated per expectations.
ADS ISI FAIL	Both ADS channels of ISI failed (channel 3 and 4).
ADS SAME SOURCE	Combination of two displays using same source of air data (cover loss of channels 1-2, 1-3, 2-3, 1-4, 2-4).
ADS 1 SLIPCOMP FAIL	Loss of all sideslip compensation capability (cross and diagonal) in ADS.
ADS 2 SLIPCOMP FAIL	Loss of all sideslip compensation capability (cross and diagonal) in ADS.
ADS 3 SLIPCOMP FAIL	Loss of all sideslip compensation capability (cross and diagonal) in ADS.
ADS 4 SLIPCOMP FAIL	Loss of all sideslip compensation capability (cross and diagonal) in ADS.
ADS ISI SLIPCOMP FAIL	Loss of all sideslip compensation capability (cross and diagonal) in ADS 3 and 4.
ADS 3 FAIL	Loss of ADS 3 channel.
DUAL ADS FAIL	2 ADS sources are failed.

Table 13: ADVISORY Messages

MESSAGE	LOGIC
ADS FAULT	Loss of redundant or non-critical function for the air data system.
ADS 1 DEGRADED	SSEC correction lost and based on default input value(s) for ADS 1 - includes loss of AOA offset.
ADS 2 DEGRADED	SSEC correction lost and based on default input value(s) for ADS 2- includes loss of AOA offset.
ADS 3 DEGRADED	SSEC correction lost and based on default input value(s) for ADS 3- includes loss of AOA offset.
ADS 4 DEGRADED	SSEC correction lost and based on default input value(s) for ADS 4- includes loss of AOA offset.
ADS 1 FAIL	Loss of ADS 1 channel.
ADS 2 FAIL	Loss of ADS 2 channel.
ADS 4 FAIL	Loss of ADS 3 channel.

Table 14: STATUS Message

MESSAGE	LOGIC
ADS PROBE HEAT GND ON	PROBE HEAT PBA selected and active.

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Table 15: INFO Messages

MESSAGE	LOGIC
34 ADS FAULT - ADS SENSE LINE HEATER 1 INOP	Loss of sense line heater capability. Potential for long-term moisture buildup.
34 ADS FAULT - ADS SENSE LINE HEATER 2 INOP	Loss of sense line heater capability. Potential for long-term moisture buildup.
34 ADS FAULT - ADS SENSE LINE HEATER 3 INOP	Loss of sense line heater capability. Potential for long-term moisture buildup.
34 ADS FAULT - ADS SENSE LINE HEATER 4 INOP	Loss of sense line heater capability. Potential for long-term moisture buildup.
34 ADS FAULT - ADS HEATER 1 REDUND LOSS	Loss of heater control redundancy due to either loss of one controller circuit, or loss of one power input.
34 ADS FAULT - ADS HEATER 2 REDUND LOSS	Loss of heater control redundancy due to either loss of one controller circuit, or loss of one power input.
34 ADS FAULT - ADS HEATER 3 REDUND LOSS	Loss of heater control redundancy due to either loss of one controller circuit, or loss of one power input.
34 ADS FAULT - ADS HEATER 4 REDUND LOSS	Loss of heater control redundancy due to either loss of one controller circuit, or loss of one power input.
34 ADS FAULT - ADS 1 PRIMARY SIDESLIP COMP INOP	Failure/loss of ADSP 2 heater or failure/loss of ASDSP 2 OSP electronics.
34 ADS FAULT - ADS 2 PRIMARY SIDESLIP COMP INOP	Failure/loss of ADSP 1 heater or failure/loss of ASDSP 1 OSP electronics.
34 ADS FAULT - ADS 3 PRIMARY SIDESLIP COMP INOP	Failure/loss of ADSP 4 heater or failure/loss of ASDSP 4 OSP electronics.

Table 15: INFO Messages

MESSAGE	LOGIC
34 ADS FAULT - ADS 4 PRIMARY SIDESLIP COMP INOP	Failure/loss of ADSP 3 heater or failure/loss of ASDSP 3 OSP electronics.
34 ADS FAULT - L TAT HEATER INOP	Fuselage TAT not able to be heated on left side. ADS 1 and 3 revert to onside engine TAT (loss of redundancy only).
34 ADS FAULT -R TAT HEATER INOP	Fuselage TAT not able to be heated on right side. ADS 2 and 4 revert to onside engine TAT (loss of redundancy only).
34 ADS FAULT - ADS 1 TAT ELEMENT INOP	ADS 1 loss of on-side fuselage TAT measurement (automatic reversion to onside engine TAT - loss of redundancy only).
34 ADS FAULT - ADS 2 TAT ELEMENT INOP	ADS 2 loss of on-side fuselage TAT measurement (automatic reversion to onside engine TAT - loss of redundancy only).
34 ADS FAULT - ADS 3 TAT ELEMENT INOP	ADS 3 loss of on-side fuselage TAT measurement (automatic reversion to onside engine TAT - loss of redundancy only).
34 ADS FAULT - ADS 4 TAT ELEMENT INOP	ADS 4 loss of on-side fuselage TAT measurement (automatic reversion to onside engine TAT - loss of redundancy only).
34 ADS FAULT - L AOA VANE INOP	ADS 1 or 3 is reporting a loss or internal failure (HW, SW, strapping changed since power-up) of the LEFT AOA VANE by setting SSM to failure warning or NCD.

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Table 15: INFO Messages

MESSAGE	LOGIC
34 ADS FAULT - R AOA VANE INOP	ADS 2 or 4 is reporting a loss or internal failure (HW, SW, strapping changed since power-up) of the LEFT AOA VANE by setting SSM to failure warning or NCD.
34 ADS FAULT - L AOA VANE HEATER INOP	ADS 1 or 3 is reporting a heating failure of the LEFT AOA VANE. This INFO message is inhibited in case the AOA vane heater should not be heated per expectation.
34 ADS FAULT - R AOA VANE HEATER INOP	ADS 2 or 4 is reporting a heating failure of the LEFT AOA VANE. This INFO message is inhibited in case the AOA vane heater should not be heated per expectation.
34 ADS FAULT - L AOA VANE DEGRADED	L AOA vane local AOA miscompares with ADSP 1 and ADSP 3 local AOA.
34 ADS FAULT - R AOA VANE DEGRADED	L AOA vane local AOA miscompares with ADSP 2 and ADSP 4 local AOA.
34 ADS FAULT - L AOA CASE HEATER INOP	ADS 1 or 3 is reporting that the left AOA case heater current is lower than minimum for operational condition in flight. This INFO message is inhibited in case the AOA case heater should not be heated per expectation.
34 ADS FAULT - R AOA CASE HEATER INOP	ADS 2 or 4 is reporting that the left AOA case heater current is lower than minimum for operational condition in flight. This INFO message is inhibited in case the AOA case heater should not be heated per expectation.

30-41 WINDSHIELD AND SIDE WINDOW HEATING SYSTEM

GENERAL DESCRIPTION

The windshield and side window heating system (WWHS) applies electrical power to transparent films within the windshields and side windows. The windshields are heated to a higher temperature than the side windows to provide anti-icing and defogging. The side windows are heated to provide defogging only.

All the windows are monitored for temperature by the associated windshield ice protection controller (WIPC) by using three temperature sensors embedded within each windshield and side window. During normal operation, one sensor provides temperature control and the other two sensors monitor the performance of the first one. In case of a sensor failure, the WIPC switches to one of the remaining two sensors for control and the last one is used as a monitor.

Each window contains a transparent heating film, which is embedded between glass laminations. The heating film is configured into three zones. The WIPC supplies 3-phase 115 VAC to the heating film. Each zone is powered by single phase AC power.

Electrical power can be turned OFF to each windshield and side window individually using the associated pushbutton annunciator (PBA) on the WINDOW HEAT panel located on the overhead panel.

The left windshield WIPC heater power is supplied from the AC BUS 1 and the control power is supplied from DC BUS 1. The right windshield WIPC heater power is supplied from AC BUS 2 and the control power is supplied from DC BUS 2.

The left side window WIPC heater power is from the AC ESS BUS and the control power is supplied from DC ESS BUS 3. The right side window WIPC heater power is supplied from AC BUS 2 and the control power is supplied from DC BUS 2.

The WIPCs provide data on system operation for the engine indication and crew alerting system (EICAS) messages through the data concentrator unit module cabinets (DMCs).

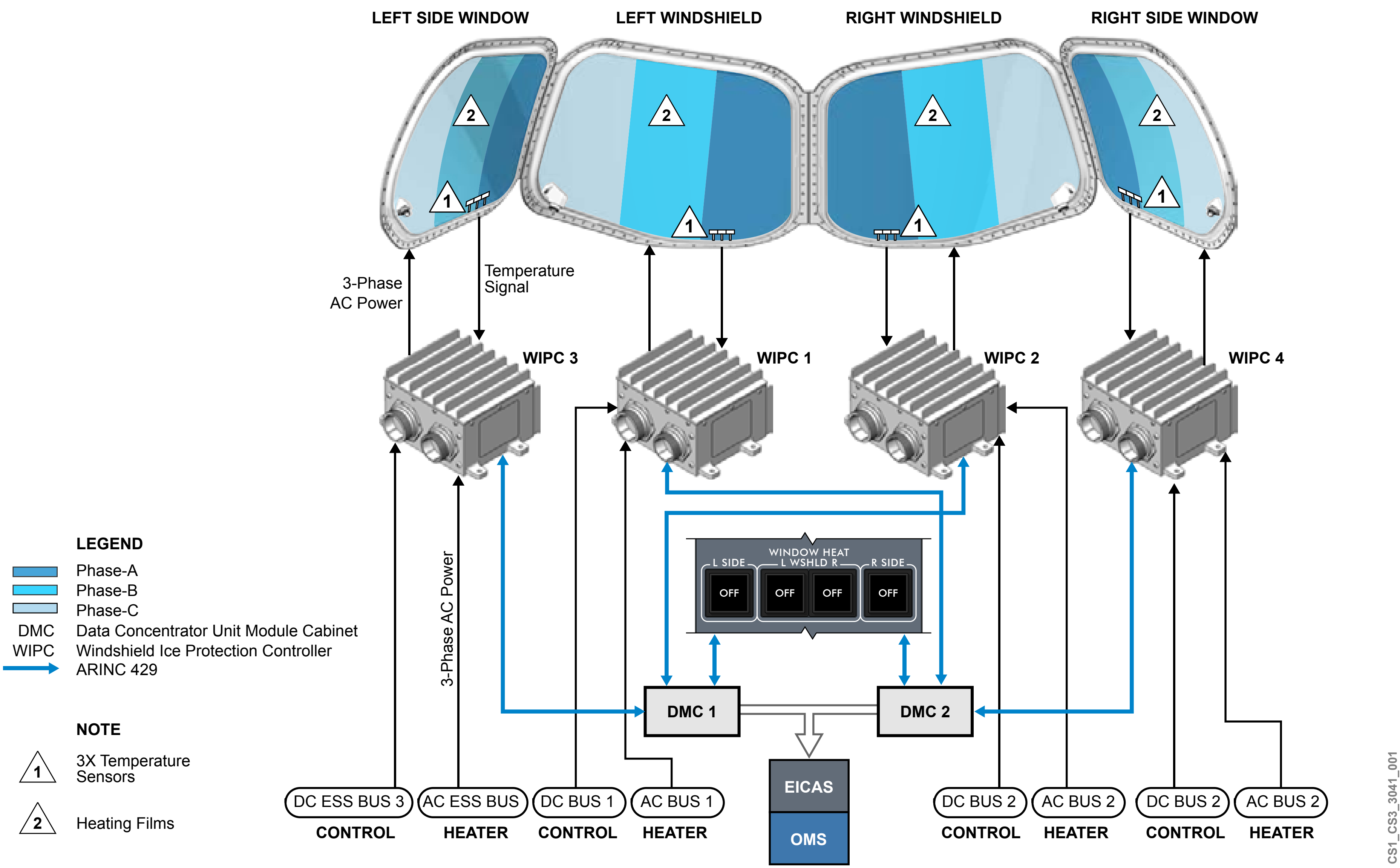


Figure 27: Windshield and Window Heat System

COMPONENT LOCATION

The windshield and side window heating system (WWHS) consists of:

- Left and right heated windshield
- Left and right heated side window
- Windshield ice protection controllers (WIPCs)

WINDSHIELD

There are two electrically-heated windshields located on each side of the forward section of the flight deck.

SIDE WINDOW

There are two electrically-heated side windows, one located on each side of the flight deck.

WINDSHIELD ICE PROTECTION CONTROLLERS

Four windshield and ice protection controllers (WIPCs) are installed in the forward equipment bay. Each window is assigned to a dedicated WIPC.

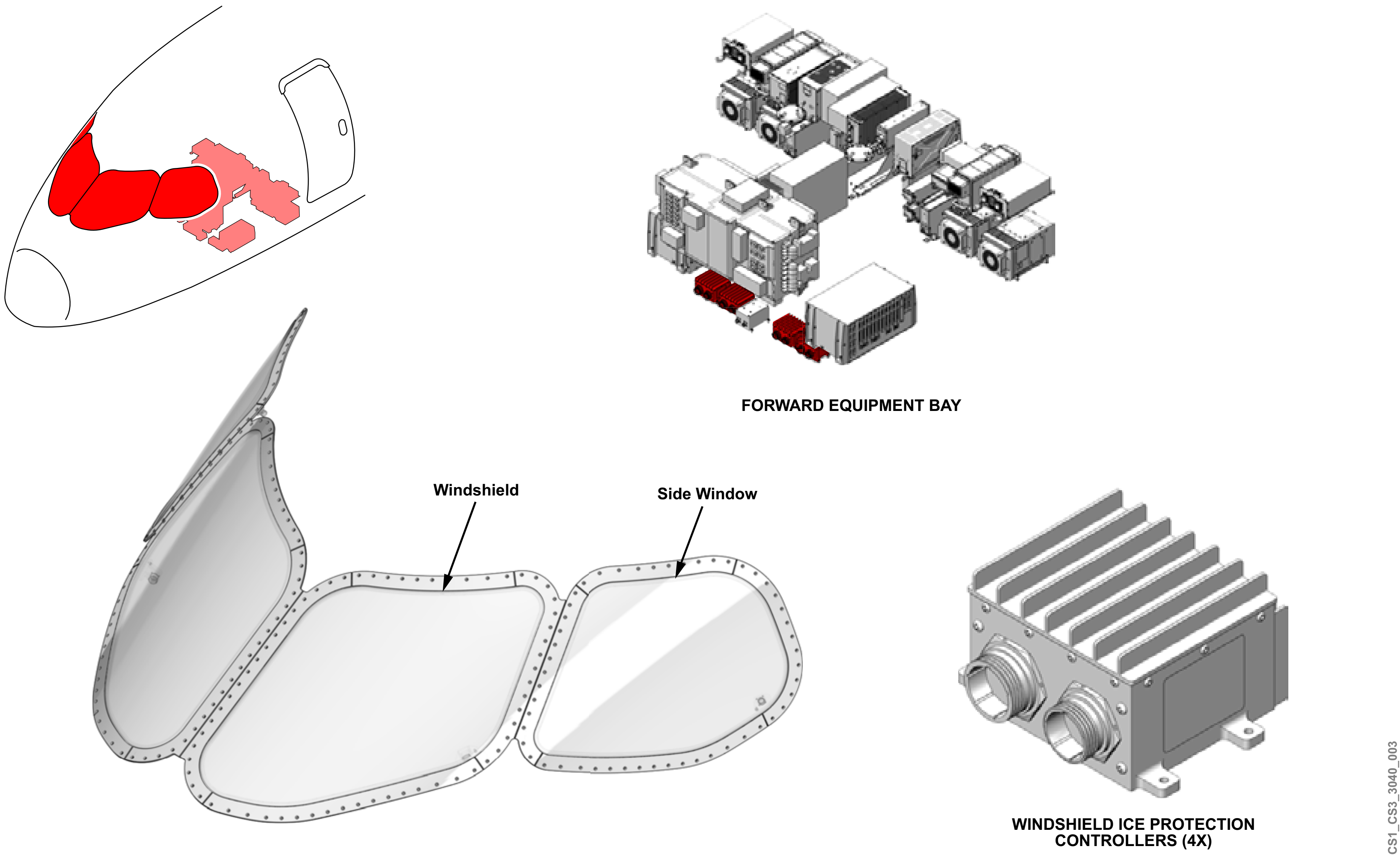


Figure 28: Windshield, Side Window, and Windshield Ice Protection Controllers

CONTROLS AND INDICATIONS

Each window has its own pushbutton annunciator (PBA). The PBAs are located on the WINDOW HEAT panel in the flight deck overhead panel. The PBAs command their own windshield ice protection controller through the data concentrator unit module cabinets (DMCs) to turn off the windshield/window heat.

The OFF legend illuminates on the PBA to confirm that the corresponding window heat function is disabled. Cycling the PBA to OFF then ON again resets the WIPC.

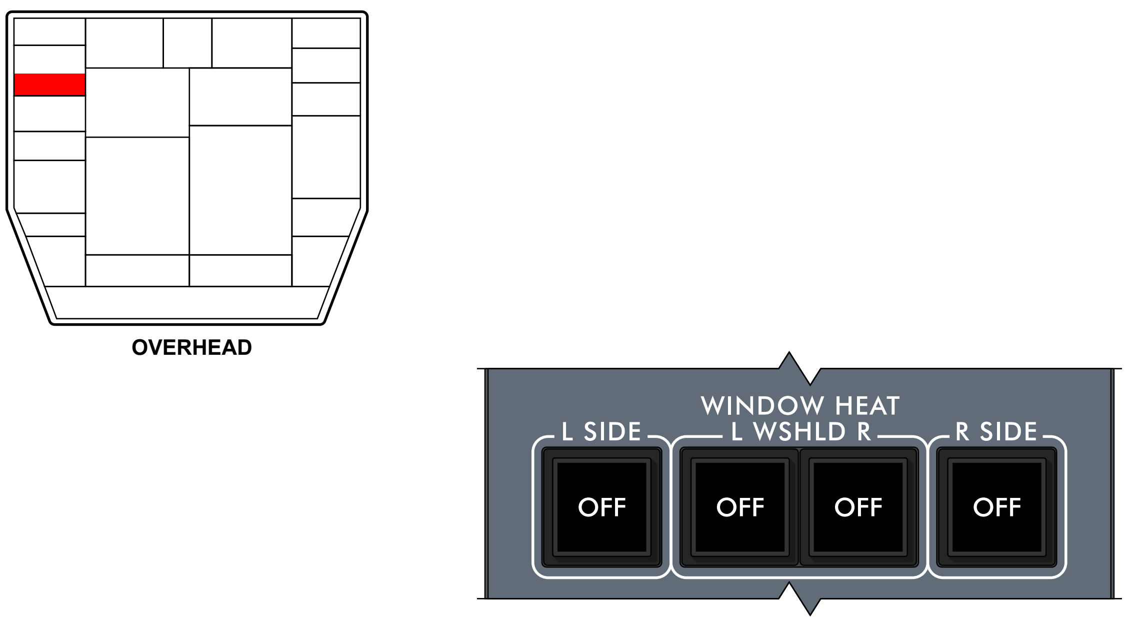


Figure 29: Window Heat Panel

DETAILED DESCRIPTION

The windshield and side window heating system (WWHS) applies electrical power to transparent films within the windows. The WWHS is normally on when the aircraft is powered. Window heating can be individually shut off through selection of the associated PBA located on the overhead panel.

The controlled temperature of the windshield is set between 45.5°C (114°F) and 51°C (124°F). The side window temperature is set between 36.7° C (98°F) and 43.9°C (111°F).

The left side window receives heater power from AC ESS BUS and control power from DC ESS BUS 3. As a result, under emergency conditions where normal AC power is lost, only the left side window can be heated.

The WIPCs receive phase of flight data from the data concentrator unit module cabinet (DMC) to moderate power requirements for regulating window heat. For example, at initial power-up in cold soaked conditions, the WIPC regulates window temperature to a lower value to avoid damage to frozen windshields and windows due to thermal stresses.

In the event of window heat failure, the WIPC reports through the DMCs, the appropriate WINDSHIELD HEAT FAIL or WINDOW HEAT FAIL caution message for display on EICAS. If the signal from the WINDOW HEAT PBAs is lost or ARINC 429 communications is lost with the DMCs, the WIPC turns on and a WHSLD FAIL ON, or a SIDE WDW HT FAIL ON advisory message is displayed for the affected window.

The WIPCs periodically performs built-in tests (BITs) that are capable of detecting electrical power faults, temperature sensor faults, heating film faults, and internal WIPC faults. Test result data is communicated to the onboard maintenance system (OMS) through the DMCs.

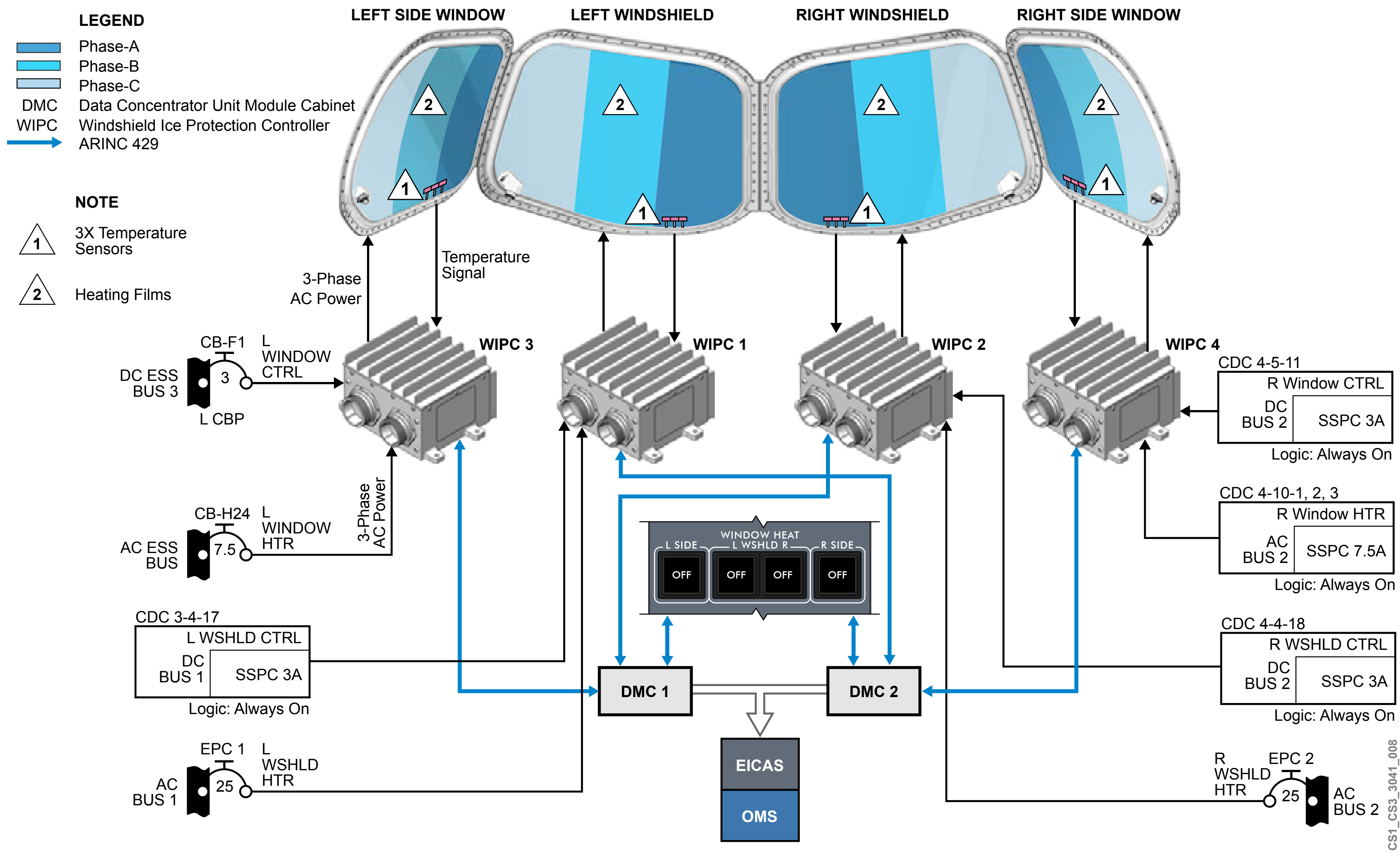


Figure 30: Windshield and Window Heat System Schematic

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) messages associated with the windshield and side window heating system.

CAS MESSAGES

Table 16: CAUTION Messages

MESSAGE	LOGIC
L SIDE WDW HEAT FAIL	Heater for left side window failed.
R SIDE WDW HEAT FAIL	Heater for right side window failed.
L WSHLD HEAT FAIL	Heater for left windshield window failed.
R WSHLD HEAT FAIL	Heater for right windshield window failed.

Table 17: ADVISORY Messages

MESSAGE	LOGIC
L WSHLD HEAT FAIL ON	Heater for left windshield failed operative.
R WSHLD HEAT FAIL ON	Heater for right windshield failed operative.
L SIDE WDW HT FAIL ON	Heater for left side window failed operative.
R SIDE WDW HT FAIL ON	Heater for right side window failed operative.

Table 18: STATUS Messages

MESSAGE	LOGIC
L SIDE WDW HEAT OFF	Heater for left side window selected OFF.
R SIDE WDW HEAT OFF	Heater for right side window selected OFF.
L WSHLD HEAT OFF	Heater for left windshield window selected OFF.
R WSHLD HEAT OFF	Heater for right windshield window selected OFF.

30-42 WINDSHIELD WIPERS SYSTEM

GENERAL DESCRIPTION

Each windshield is provided with a windshield wiper with separate power sources, wiper motors, and associated switches, arms, and wiper blades.

The left wiper motor is powered by AC BUS 1 and the right wiper motor is powered by AC BUS 2.

The wiper motors have electronic controllers and are electrically connected to each other for synchronization. This allows them to move together when both windshield wiper switches are selected to the same speed. The synchronization feature does not prevent the other wiper from operating if one wiper motor fails.

The wiper controller incorporates a thermal protection that removes power to the motor if the motor overheats. After a cool down period, the controller reapplies power to the motor.

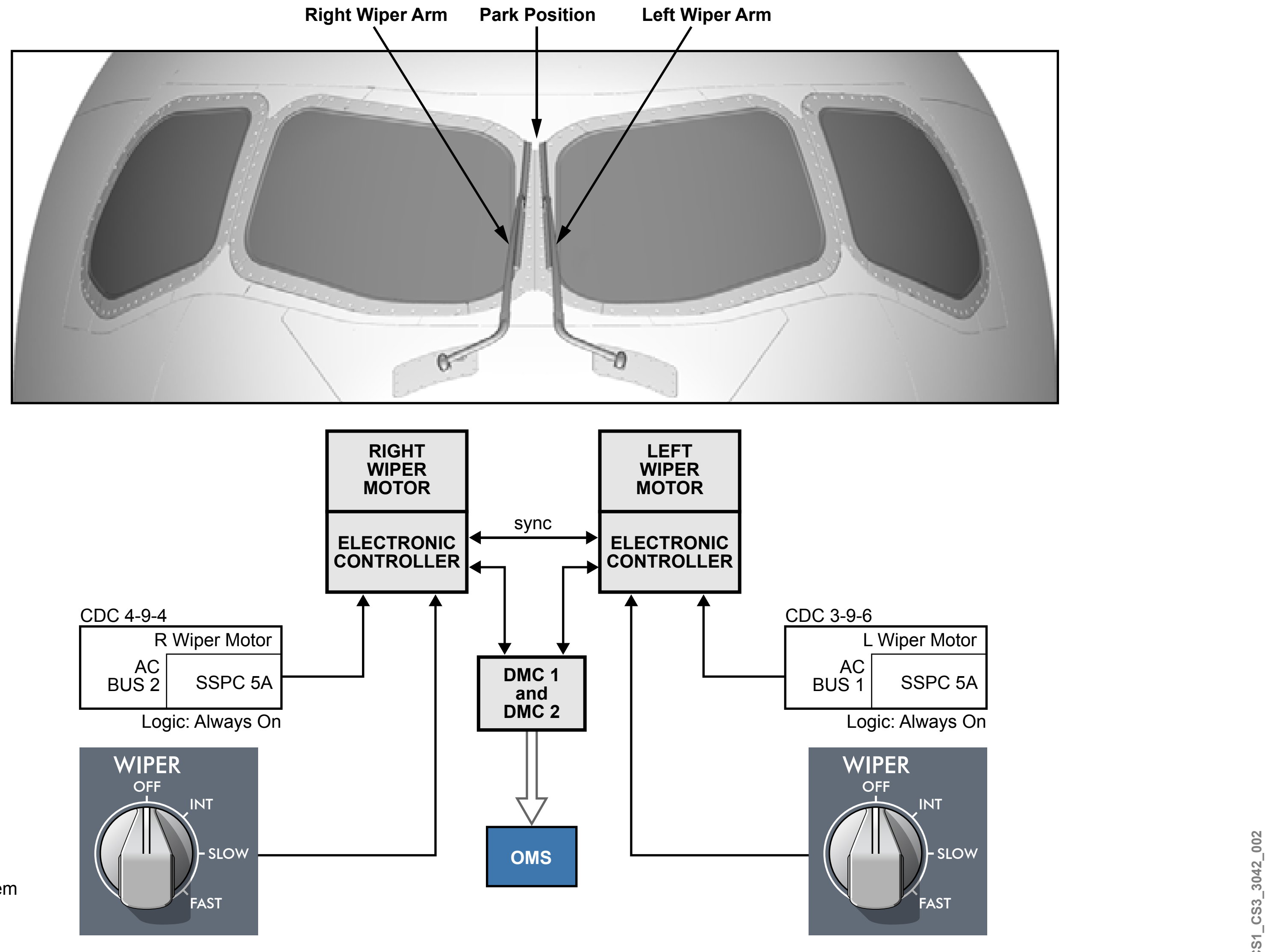
When not in operation, the wipers move to the parked position, resting vertically at the center post of the windshield frame.

If a wiper motor fails, it moves to the park position. The windshield wiper operation can be reset by setting the rotary switch to OFF and back to any of the other positions.

The winshield wiper switch has the following selections:

- OFF: The onside wiper moves to the vertical parked position
- INT: One wiping cycle every 5 seconds
- SLOW: 80 cycles per minute
- FAST: 120 cycles per minute

The wiper motors run continuous built-in tests (CBIT). In the event of failure the electronic controller sends failure information through the data concentrator unit module cabinets (DMCs) to the onboard maintenance system (OMS). There are no crew alerting system (CAS) messages for the windshield wiper system.



LEGEND

DMC Data Concentrator Unit Module Cabinet
OMS Onboard Maintenance System
ARINC 429
Discrete

Figure 31: Windshield Wiper System

COMPONENT LOCATION

The windshield wiper system consists of the following:

- Wiper motors
- Wiper arms
- Wiper blades

WIPER MOTOR

There are two wiper motors. Each motor is located below its windshield.

WIPER ARM

There are two wiper arms. Each wiper arm is attached to its windshield wiper motor.

WIPER BLADE

There are two wiper blades. A wiper blade is attached to each wiper arm.

CAUTION

To avoid damage to the windshield do not operate windshield wipers on dry glass.

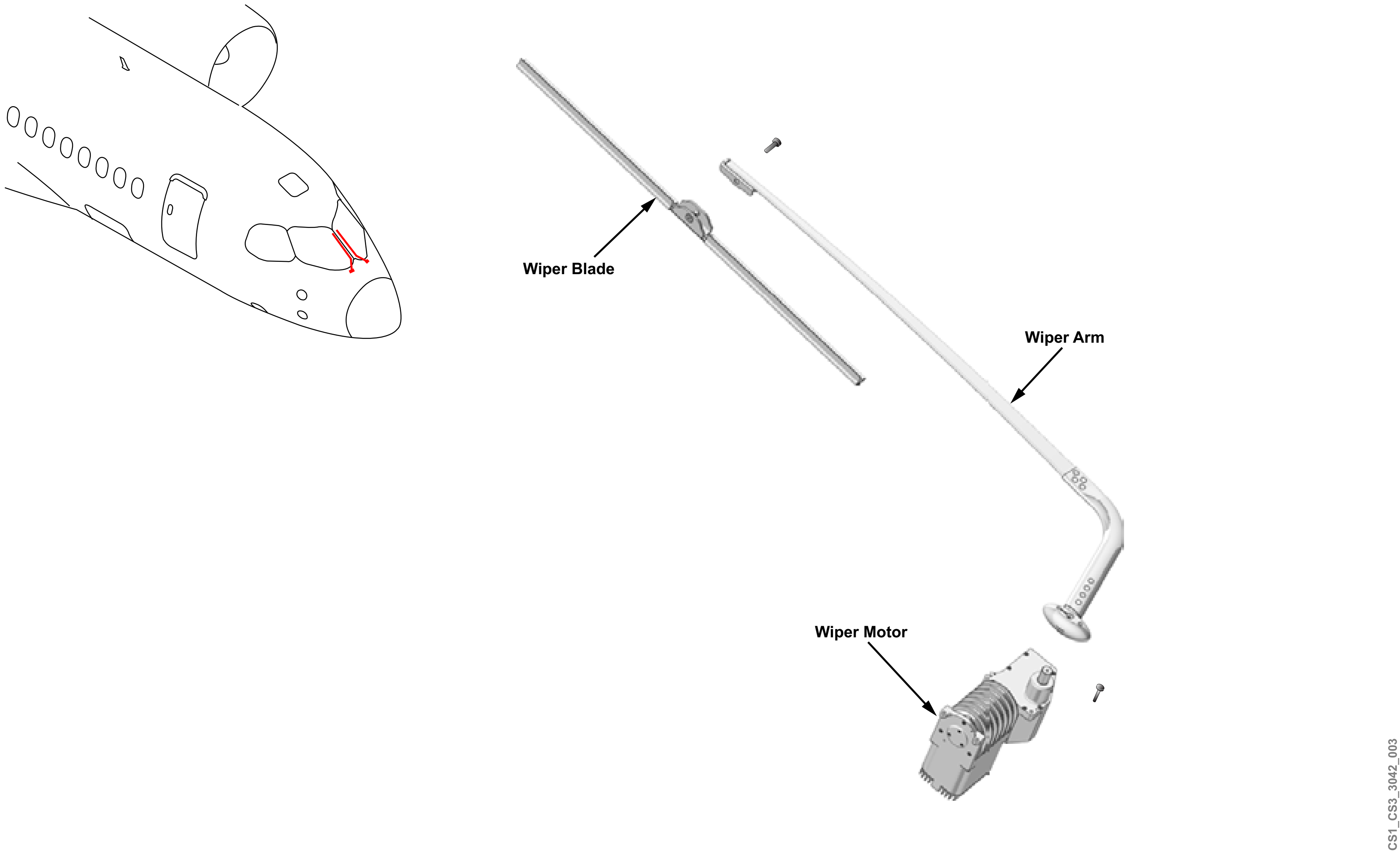


Figure 32: Windshield Wiper Motor, Wiper Arm, and Wiper Blade

CONTROLS AND INDICATIONS

The WIPER selector switches are located on the left and right wiper panel on the overhead panel of the flight deck. The switches allow selection of three speeds: intermittent (INT), SLOW, and FAST.

Figure 33: Windshield Wipers System Controls

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ATA 36 - Pneumatic

BD-500-1A10
BD-500-1A11

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PNEUMATIC - CHAPTER BREAKDOWN

36-10 BLEED AIR SYSTEM

GENERAL DESCRIPTION

The aircraft bleed air system (BAS) receives hot bleed air from the engines, auxiliary power unit (APU), or an external air source. Pressurized air from these sources enters common ducting in the mid fuselage belly before being distributed to the user systems. A cross-bleed valve (CBV) located in the manifold, isolates left and right side bleed air systems. Opening the CBV interconnects the two sides. The integrated air system controllers (IASCs) control and monitor the BAS. Each IASC is powered by two DC ESS BUSES.

LEFT AND/OR RIGHT ENGINE BLEED AIR

The left and right engines are the normal source of bleed air in flight. The BAS uses intermediate pressure bleed air from the 4th stage engine compressor, or high-pressure 8th stage bleed air supplied through the high-pressure valve (HPV). An intermediate pressure check valve (IPCKV) prevents 8th stage air from back flowing into the 4th stage port when the HPV is open.

Switching between 4th and 8th stage engine bleed air is based aircraft configuration and demand as calculated by the electronic engine control (EEC). The IASC uses this information to control the operation of the high-pressure valve (HPV). A bleed monitoring pressure sensor (BMPS) sends pressure information to the IASC for HPV position monitoring.

Downstream of the BMPS, a pressure-regulating and shutoff valve (PRSOV) regulates bleed air pressure according to the bleed air demand requirements to a maximum of 59 psi. The torquemotor controlled PRSOV has a full closed microswitch for valve position reporting to the IASCs. The air conditioning pack inlet pressure sensors (PIPSs), (ATA 21), are used by the IASCs to regulate the pressure downstream of the PRSOV.

A precooler, located downstream of the PRSOV control the engine bleed air temperature using engine fan air. A torquemotor controlled fan air valve (FAV) modulates the amount of engine fan air passing through the precooler in order to maintain the bleed air temperature limits. A bleed temperature sensor (BTS) provides the IASCs with temperature information to control the FAV.The fan air exhausts into the engine core compartment where it exhausts overboard. A precooler ex-haust (PCE) door opens automatically under certain high bleed air demand conditions to increase cool-ing airflow.

Bleed air, downstream of the bleed air supply of both engines, is directed to a common high-pressure duct. A cross-bleed valve (CBV), mounted on the high-pressure duct, connects and isolates bleed air from both engines.

The PRSOV and FAV are commanded closed by the IASC when the associated ENG FIRE PBA is pressed. Each valve has a thermal fuse that cuts its power if the temperature exceeds 250°C.

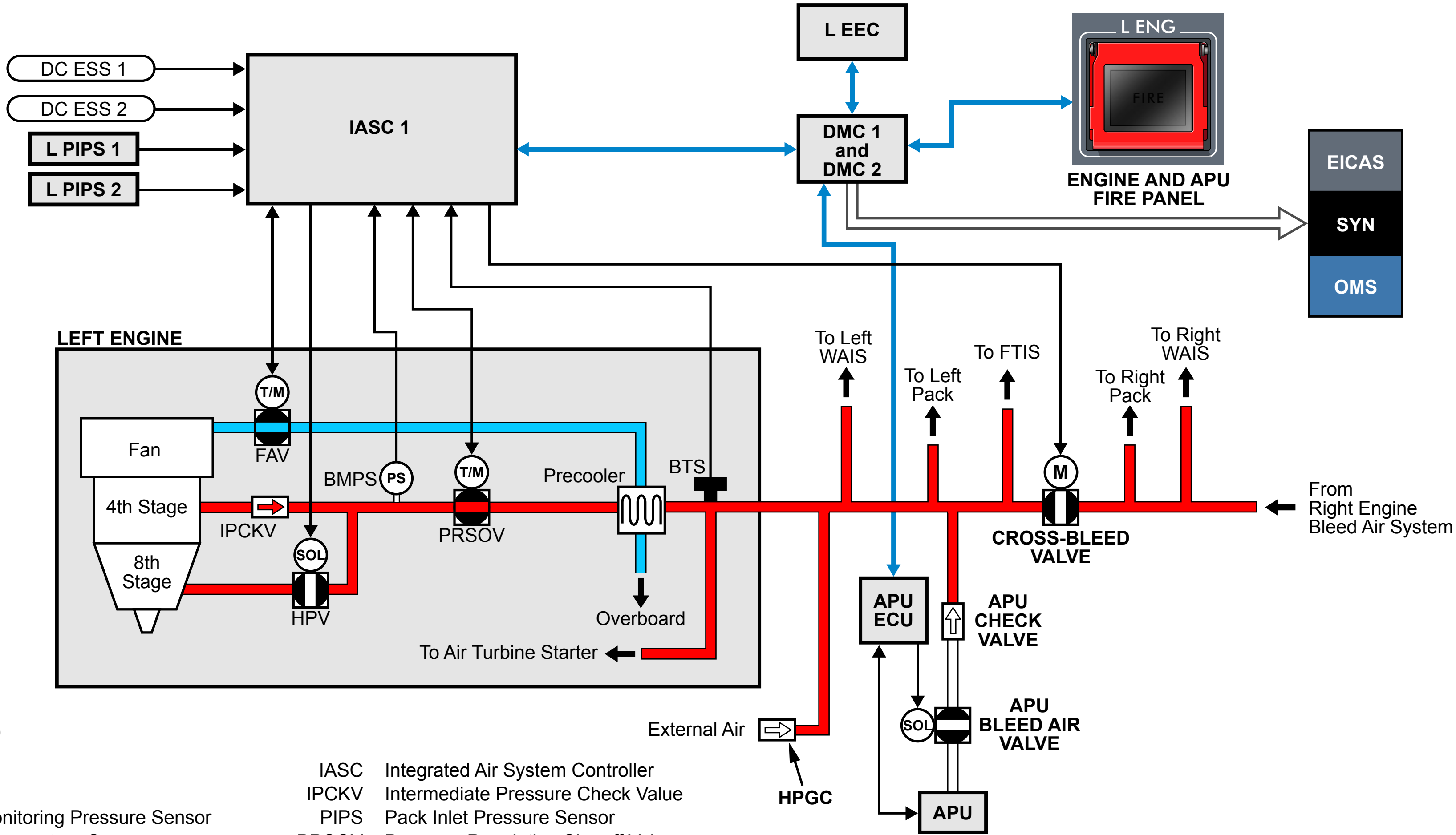
AUXILIARY POWER UNIT BLEED AIR

During normal ground operation, the IASC defaults the bleed air source to the APU. The APU bleed air valve (BAV) is controlled by the APU electronic control unit (ECU), (ATA 49). The IASC provides a load demand signal to the APU ECU. The ECU adjusts the APU load compressor output to meet the BAS demand.

When the engines are started, the APU remains the primary bleed source to optimize engine performance during takeoff. When the APU performance decreases with altitude, the IASC automatically closes the APU BAV and switches the bleed source to the engines. When the APU bleed is selected off, the APU check valve prevents engine bleed air from reaching the APU.

EXTERNAL AIR SOURCE

On the ground, the BAS can be supplied with pressurized air from an external air source, through the high-pressure ground connection (HPGC). The HPGC check valve closes when not in use to prevent bleed air system loss.



LEGEND

	Fan Air	IASC	Integrated Air System Controller
	Bleed Air	IPCKV	Intermediate Pressure Check Value
BMPS	Bleed Monitoring Pressure Sensor	PIPS	Pack Inlet Pressure Sensor
BTS	Bleed Temperature Sensor	PRSOV	Pressure-Regulating Shutoff Valve
DMC	Data Concentrator Unit Module Cabinet	WAIS	Wing Anti-Ice System
ECU	Electronic Control Unit		ARINC 429
EEC	Electronic Engine Control	M	Motor
FAV	Fan Air Valve	SOL	Solenoid
FTIS	Fuel Tank Inerting System	T/M	Torque Motor
HPGC	High Pressure Ground Connection		
HPV	High-Pressure Valve		

NOTE

Left Bleed Air System shown.
Right Bleed Air System similar.

CS1_CS3_3600_013

Figure 1: Bleed Air System Schematic

COMPONENT LOCATION

The main components of the bleed air system include:

- Precooler exhaust (PCEs) doors
- Bleed temperature sensors (BTSs)
- Bleed monitoring pressure sensors (BMPSs)
- Precoolers
- Fan air valve (FAV)
- Intermediate pressure check valve (IPCKV)
- High-pressure valves (HPVs)
- Pressure-regulating shutoff valves (PRSOVs)
- Cross-bleed valve (CBV) (Refer to figure 3)
- High-pressure ground connection (HPGC) (Refer to figure 3)
- APU check valve (Refer to figure 3)
- APU bleed air valve (BAV) (Refer to figure 4)
- Integrated air system controllers (IASCs) (Refer to figure 5)

PRECOOLER EXHAUST DOOR

The precooler exhaust (PCE) door is located on the engine right core cowl inner fixed structure, at the 3 o'clock position, aft of the oil tank access door.

BLEED TEMPERATURE SENSOR

The bleed temperature sensor is located on the high-pressure air duct, downstream of the precooler in the pylon area.

BLEED MONITORING PRESSURE SENSOR

The bleed monitoring pressure sensor (BMPS) is located in the pylon area. It is connected to the HP air duct downstream of the high-pressure shutoff valve, by a sense line.

PRECOOLER

The precooler, also known as the precooler exchanger is located on the engine pylon. The engine needs to be lowered in order to replace the precooler.

FAN AIR VALVE

The fan air valve (FAV) is located upstream of the precooler.

INTERMEDIATE PRESSURE CHECK VALVE

An intermediate pressure check valve (IPCKV) is located on the engine bleed air duct, at the 1 o'clock position.

HIGH-PRESSURE VALVE

The high-pressure valve (HPV) is located on the HP air duct, at the 2 o'clock position. It is also referred to as the high-pressure shutoff valve (HPSOV).

PRESSURE-REGULATING AND SHUTOFF VALVES

The pressure-regulating and shutoff valve (PRSOV) is located on the engine bleed air duct downstream of the HPV at the 1 o'clock position.

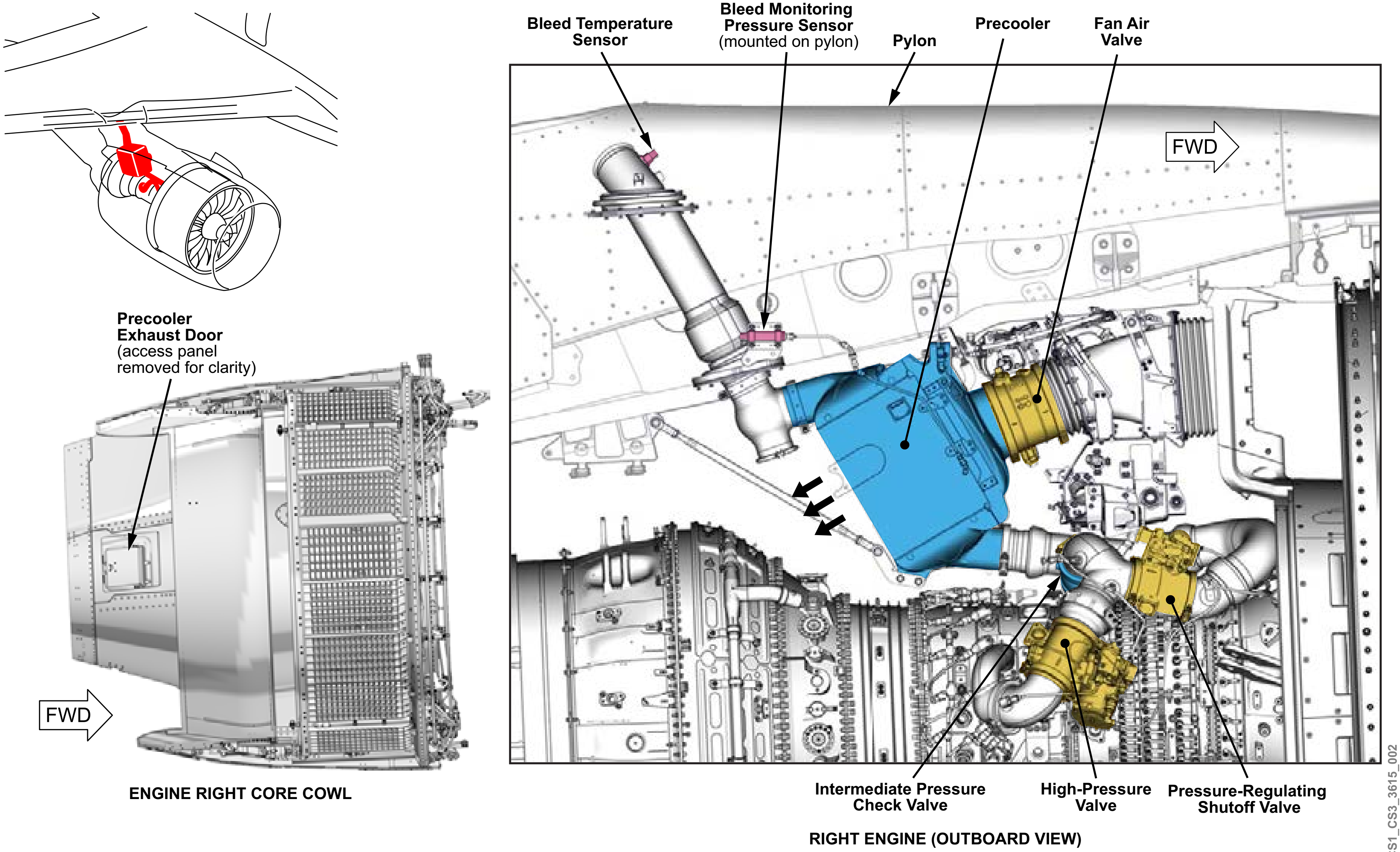


Figure 2: Engine Bleed Air System Components

CROSS-BLEED VALVE

The cross-bleed valve (CBV) is located in the mid forward section of the wing-to-body fairing (WTBF) area.

HIGH-PRESSURE GROUND CONNECTION

The high-pressure ground connection (HPGC) is located in the left forward section off the WTBF area.

AUXILIARY POWER UNIT CHECK VALVE

The auxiliary power unit (APU) check valve is located in the right aft section of the WTBF area.

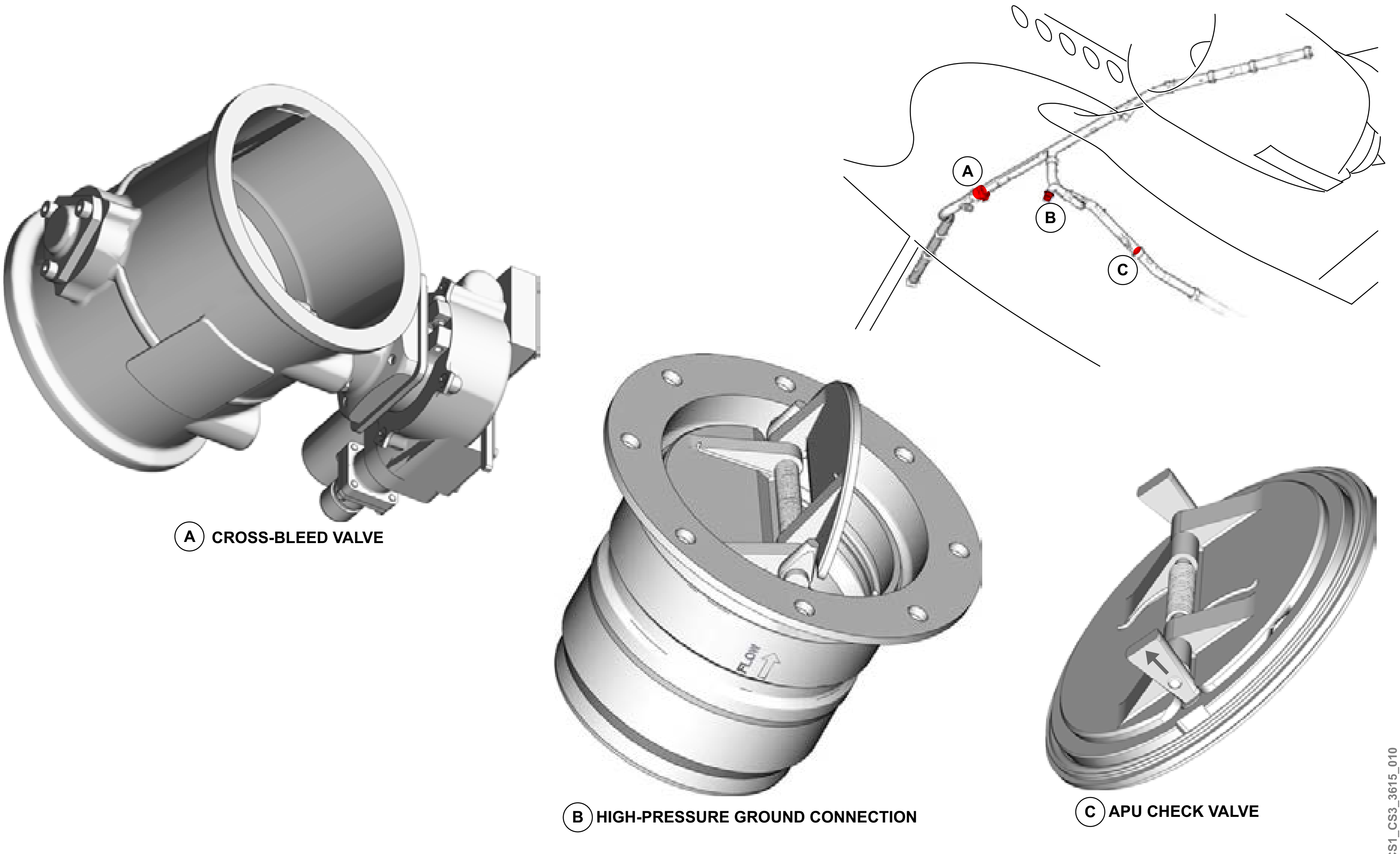


Figure 3: APU Check Valve, Cross-Bleed Valve, and High-Pressure Ground Connection Location

APU BLEED AIR VALVE

The APU bleed air valve (BAV) is located on the APU.

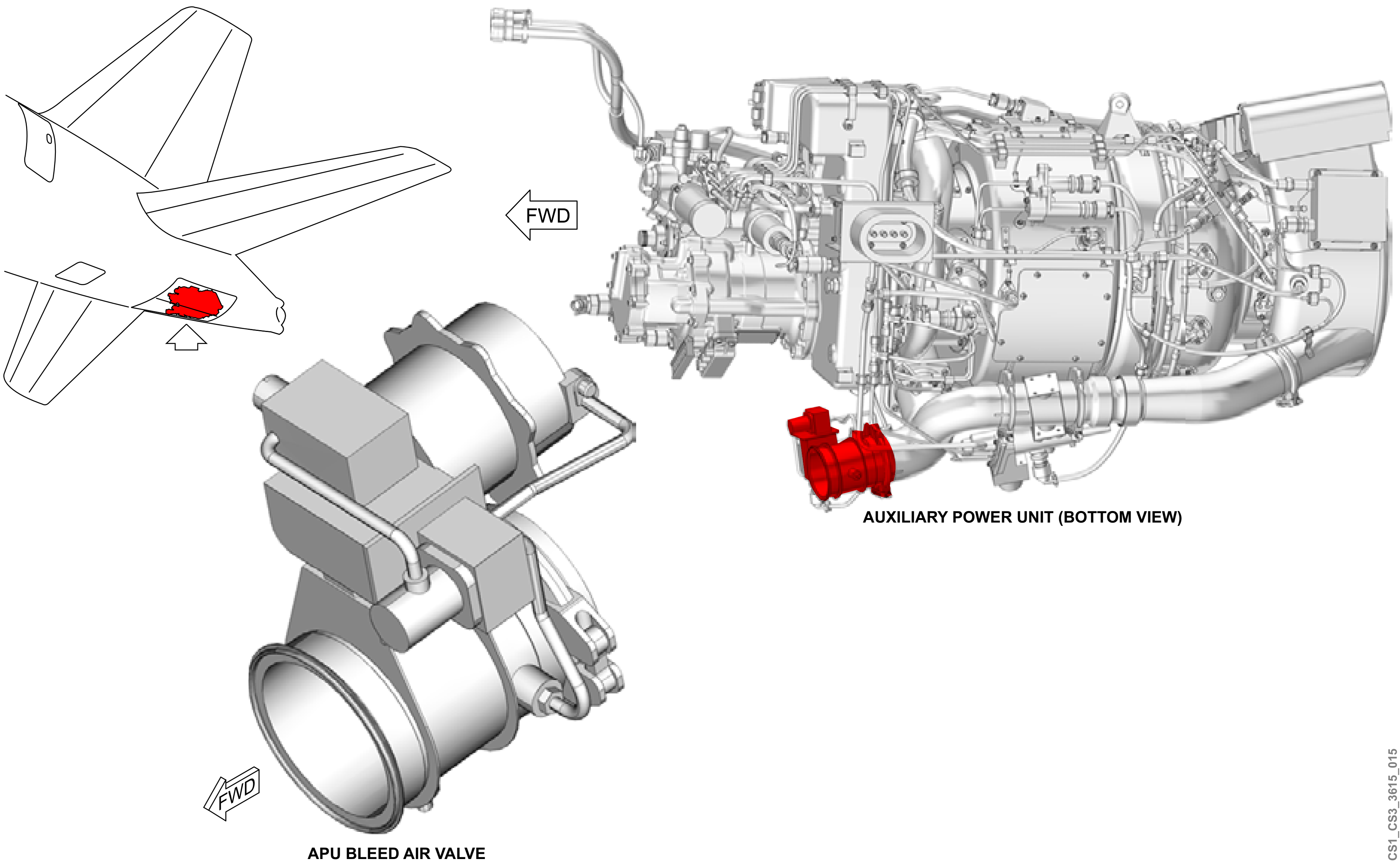
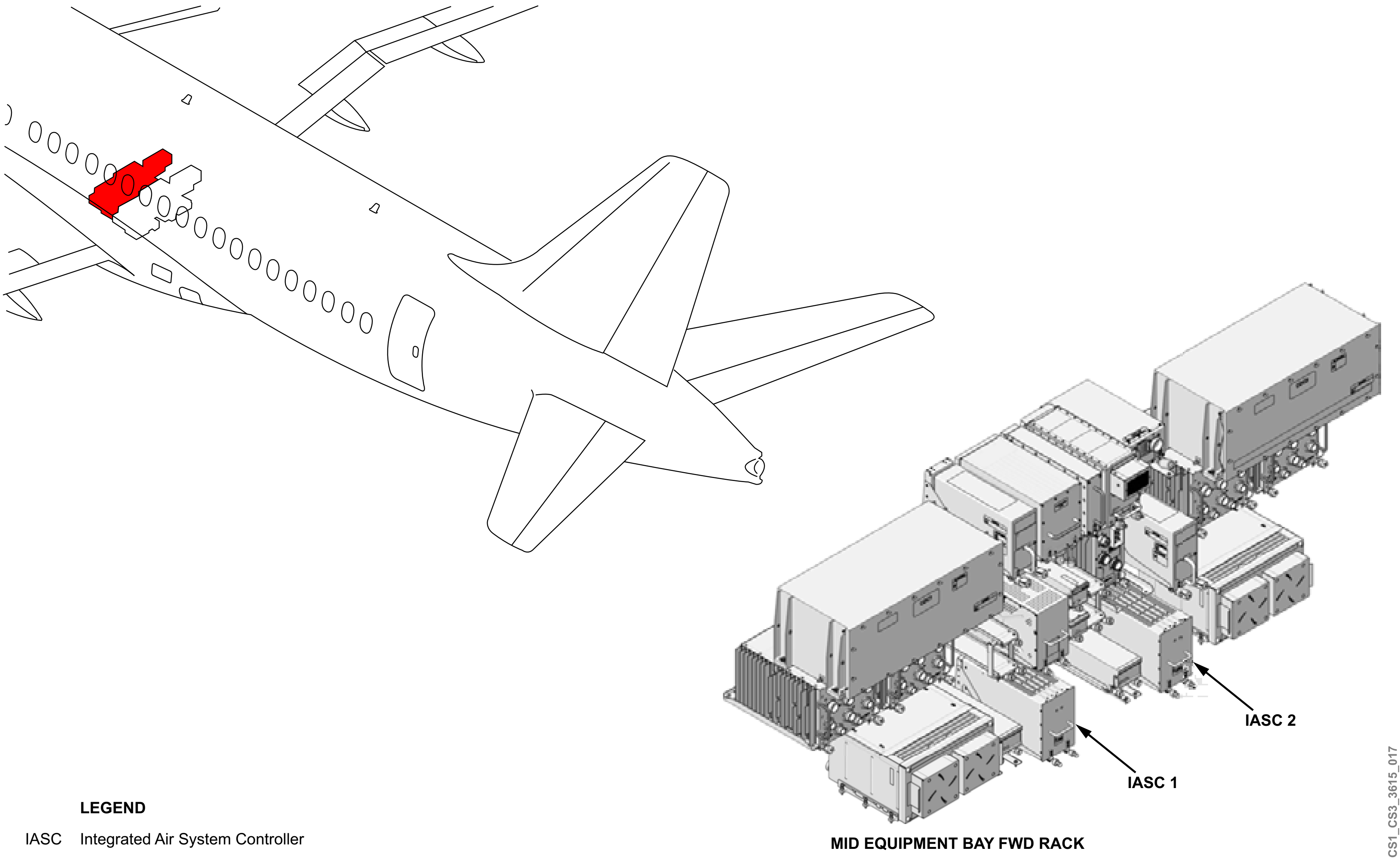


Figure 4: APU Bleed Air Valve

INTEGRATED AIR SYSTEM CONTROLLERS

Two integrated air system controllers (IASCs), IASC 1 and IASC 2 are located in the mid equipment bay forward rack.



LEGEND

IASC Integrated Air System Controller

Figure 5: Integrated Air System Controllers (IASCs)

DETAILED COMPONENT INFORMATION

INTEGRATED AIR SYSTEM CONTROLLER

Each IASC incorporates a channel A and a channel B. Channel A is the primary control channel, and channel B is the standby channel. On the ground the IASC 1 alternates with IASC 2 based on odd and even days. If a failure is detected on the primary control channel, the IASC switches to the healthy channel.

For safety purposes, a third independent and dissimilar channel is integrated and referred to as the safety channel C. This allows system control independence from system failures annunciation.

IASC channel functions are assigned accordingly:

- The left and right bleed pressure and temperature control is performed by the associated IASC primary control channel A or channel B
- The left and right bleed pressure monitoring is performed by the associated IASC primary control channel A and channel B
- The left and right bleed temperature monitoring is performed by the associated IASC channel C, and the opposite IASC channel C. Temperature monitoring is also performed by all channels for auto-shutdown purposes
- The CBV control and monitoring is performed by the left IASC channel B, and the right IASC channel B
- The APU bleed control is performed by the left and right IASC primary control channel A or channel B

An initiated built-in test (IBIT) of the bleed air system (BAS) can be performed from the onboard maintenance system (OMS) for each channel and controller.

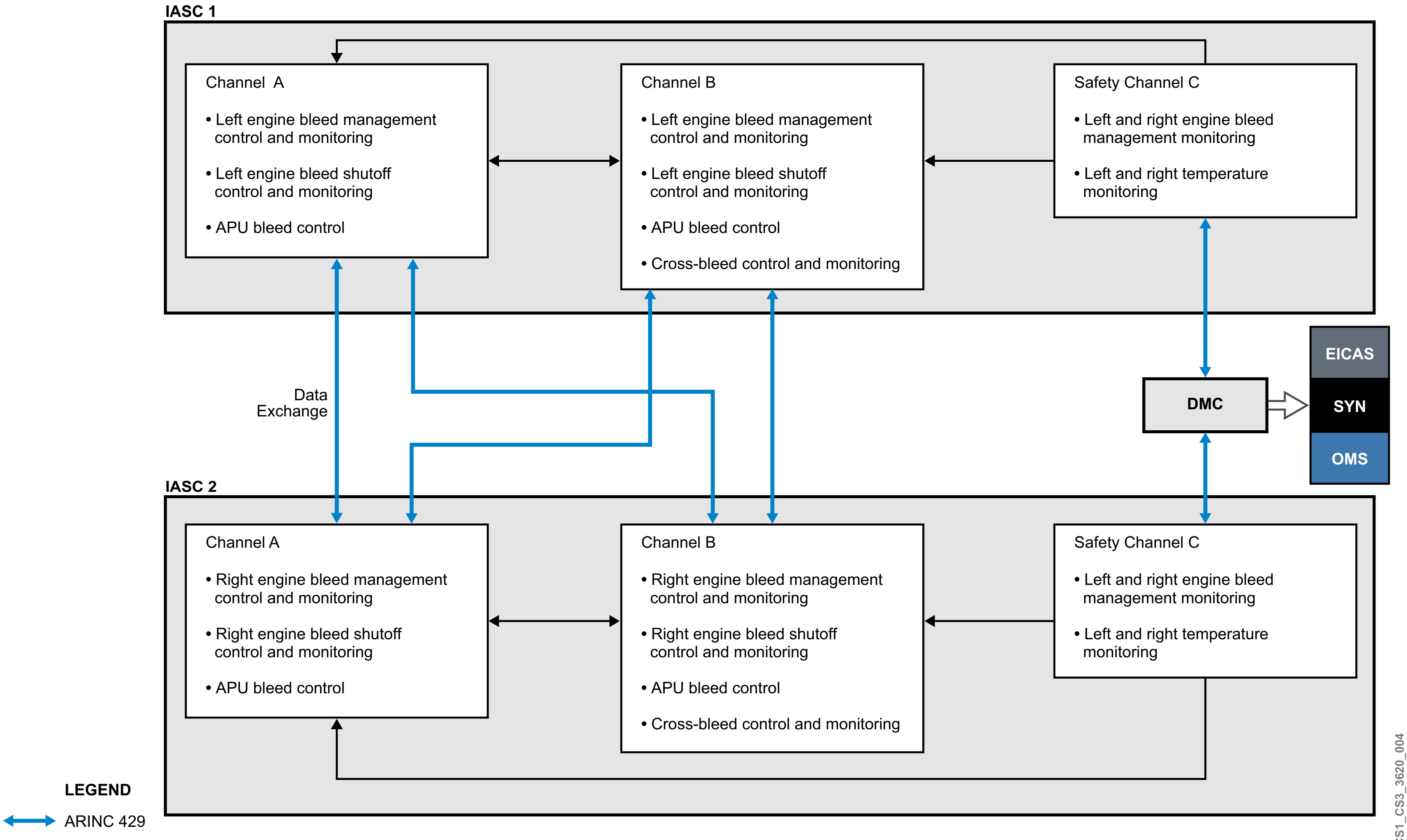


Figure 6: Integrated Air System Controller

ELECTRICAL INTERFACE

Each IASC is electrically powered by two independent DC ESS BUSES:

- IASC 1 is powered by DC ESS BUS 1 and DC ESS BUS 2
- IASC 2 is powered by DC ESS BUS 2 and DC ESS BUS 3

The IASCs power the high-pressure valve (HPV), fan air valve (FAV), cross-bleed valve (CBV), and the pressure-regulating shutoff valve (PRSOV).

The control and distribution cabinet (CDC) provides the signal to enable PRSOV operation through the IASC. The CDC command is provided when the AIR panel BLEED PBA is selected and a minimum of 15 psi bleed air is available.

The PRSOV is monitored by the pack inlet pressure sensors (PIPS). When the PRSOV is open, the PIPS provide pressure information to control the output of the PRSOV. The PIPS also provide monitoring for low-pressure and overpressure conditions. The PRSOV is monitored for the closed position by a microswitch.

The HPV solenoid operated valve is controlled by the IASC. The opening and closing of the HPV is based on intermediate pressure temperature information calculcated by the electronic engine control (EEC) and the bleed monitoring pressure sensors (BMPSs). The BMPS, located downstream of the intermediate pressure port check valve is used to monitor the position of the HPV.

The PIPS and BMPS are powered from the DC ESS BUSES.

The bleed temperature sensors (BTS) provide temperature information to the IASC for monitoring and control of the FAV.

The IASC provides the required pneumatic demand information to the APU electronic control unit (ECU), when the APU is supplying bleed air. The APU load compressor provides the required output through the APU bleed air valve (BAV). The APU BAV is powered by the APU ECU.

When a main engine start signal is received when the aircraft is on the ground: the following valves are closed for the duration of the start plus 30 seconds.

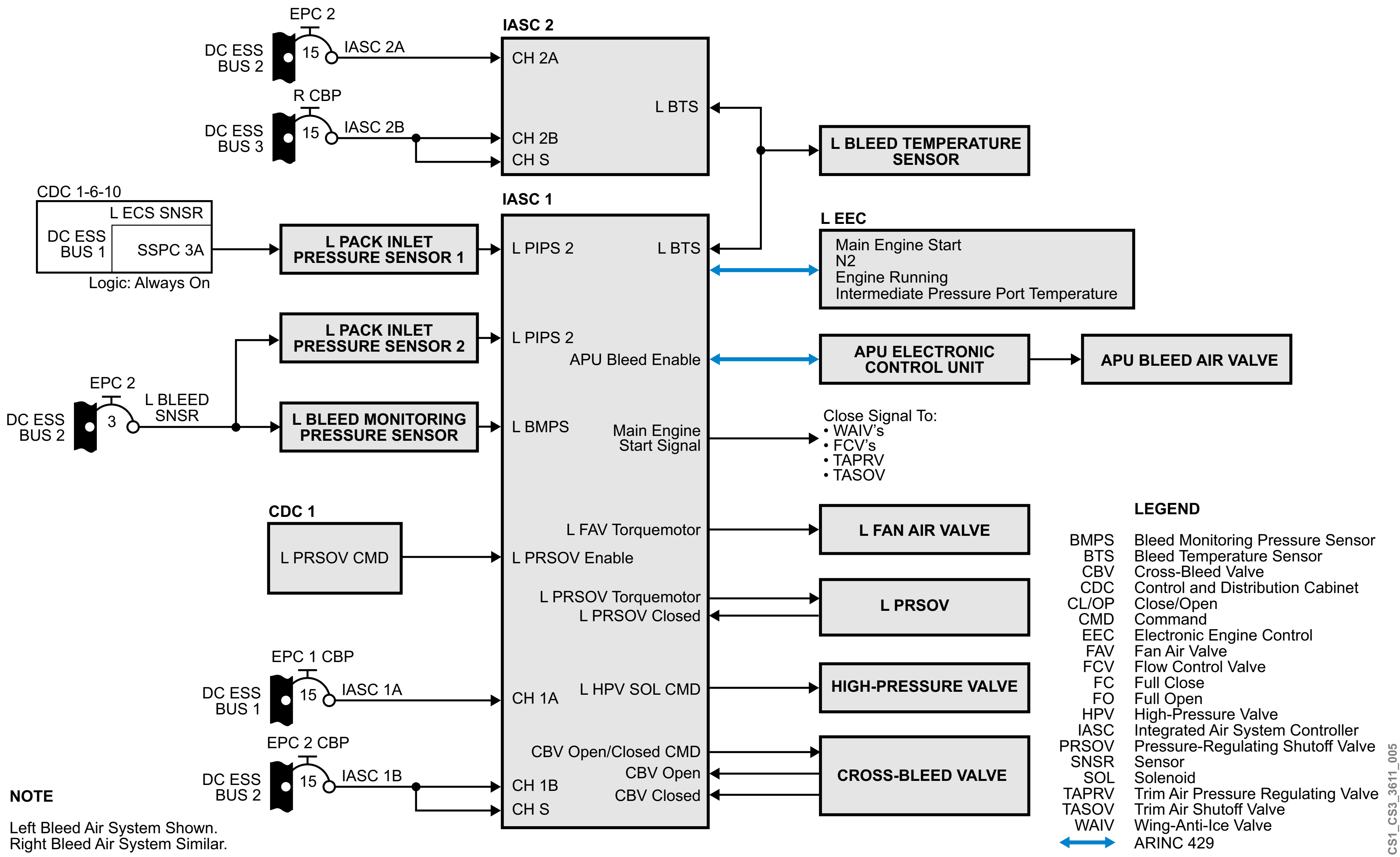
- Wing anti-ice valve (WAIV)
- Left and right flow control valves (FCV)
- Trim air pressure regulating valve (TAPRV)
- Trim air shutoff valve (TASOV)

When the engine start is complete, the valves reopen. In flight, the 30 second delay is not used and the valves open immediately after the engine starts.

The EEC provides the following information:

- Engine running
- Main engine start
- N2
- Intermediate pressure port temperature

The EEC information is used to configure the bleed air system and control system operation.



NOTE

Left Bleed Air System Shown.
Right Bleed Air System Similar.

Figure 7: Electrical Interface

CONTROLS AND INDICATIONS

CONTROLS

The bleed air system (BAS) is controlled from the AIR panel, located on the flight deck overhead panel.

Bleed air valve (BAV) operation is controlled by pushbutton annunciators (PBAs) and a rotary selector.

When the L BLEED PBA, the R BLEED PBA, and the APU BLEED PBA are in the ON position, the operation of the associated BAV is automatically managed by the IASCs. When a PBA is selected OFF, the valve is forced closed and the OFF legend illuminates.

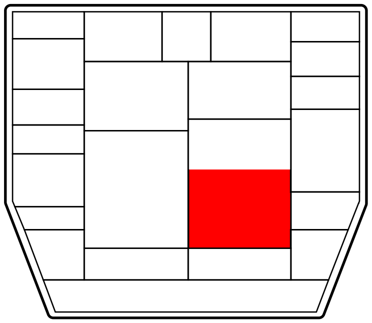
The amber L (R) BLEED PBA FAIL light illuminates for the following conditions:

- Overpressure
- Low pressure
- Loss of both BTS sensing elements
- HPV failed open
- PRSOV failed open
- IASC 1 or IASC 2 failure

The corresponding L (R) BLEED FAIL caution message is displayed on the engine indication and crew alerting system (EICAS) page.

An amber FAIL light illuminates in the APU BLEED PBA in the event of failure to supply APU bleed air or the loss of bleed leak detection capability. The APU BLEED FAIL caution message is displayed on EICAS.

The cross-bleed valve (CBV) is controlled by the rotary XBLEED switch. The CBV is controlled automatically by the integrated air system controller (IASC) when the XBLEED switch is in the AUTO position. The IASC opens the CBV when there is a single source of bleed air supply (one engine, APU or external air). The CBV is closed or open when the XBLEED switch is selected to MAN CLSD or MAN OPEN respectively. An ENG BLEED MISCONFIG caution message is displayed if the CBV is open with both engine bleeds on.



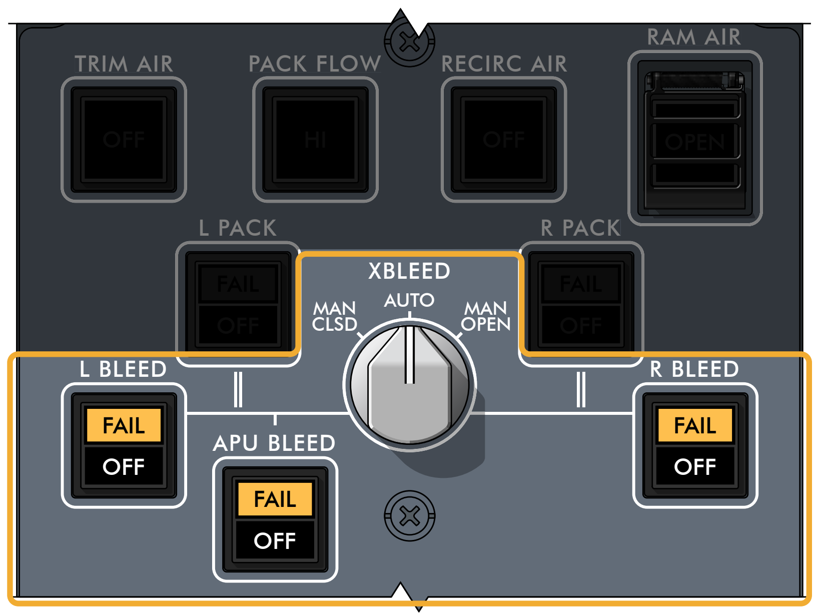


Figure 8: AIR Panel

INDICATIONS

The bleed air system (BAS) operation indication is displayed on the AIR synoptic page.

Bleed air flow lines, actual duct temperature, valve position in normal operation, valve position in fail operation, and pressure display with symbols and color codes.

The EXT AIR symbol is displayed on the AIR synoptic page when the air pressure is detected, and the engines and APU bleed air are not operating.

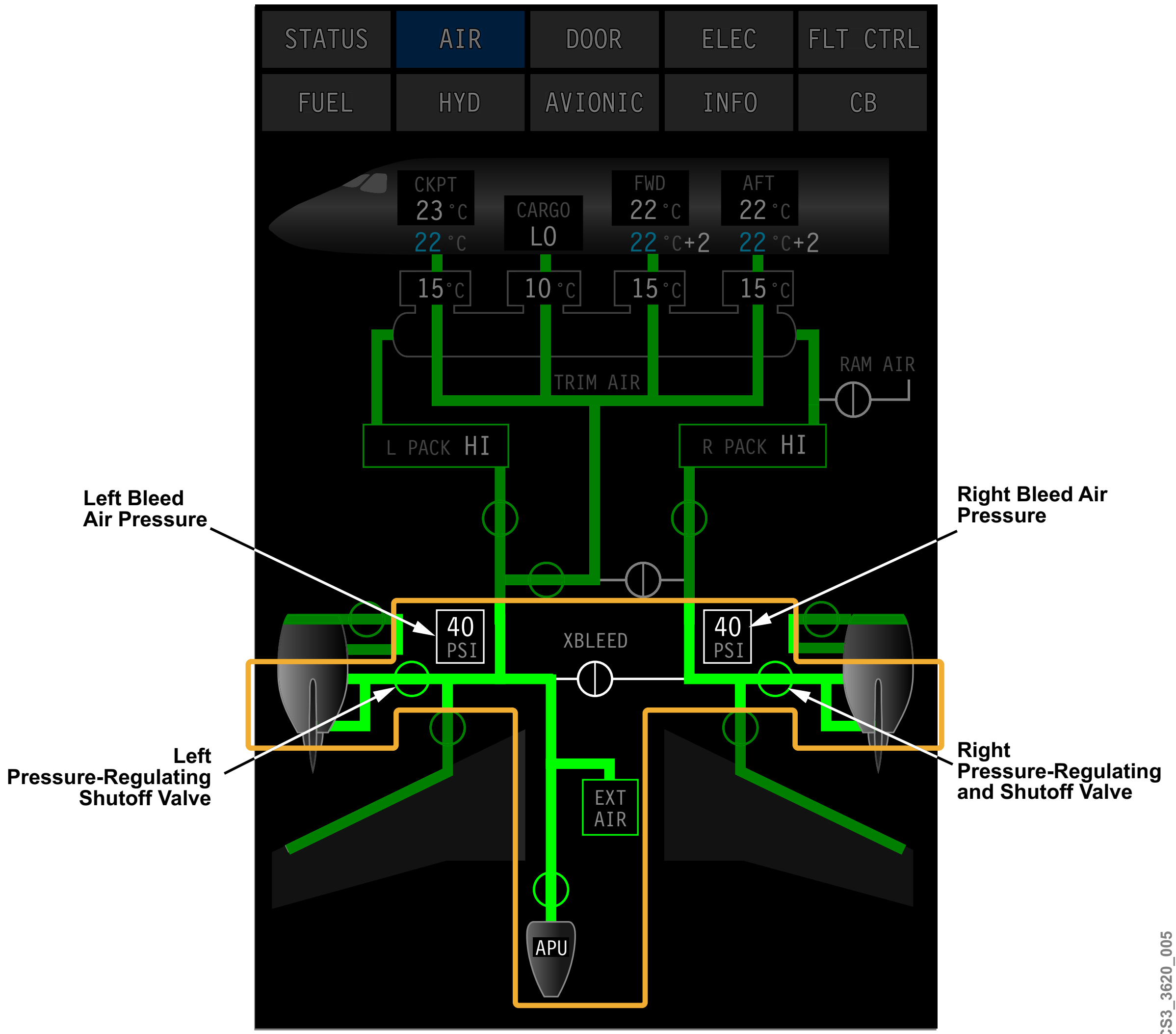
FLOW LINES	
Symbol	**Condition**
	Normal flow.
	No flow.
	Abnormal flow (overheat, out of range pressure,leak).

VALVE POSITION NORMAL OPERATION	
Symbol	**Condition**
	Closed.
	Open with flow.
	Open with no flow.

VALVE POSITION FAIL OPERATION	
Symbol	**Condition**
	Closed.
	Open with flow.
	Open with no flow.
	Invalid.

PRESSURE	
Symbol	**Condition**
5 PSI	Abnormal.
40 PSI	Normal.
-- PSI	Invalid.

DECLUTTERED SYMBOLOGY LOGIC	
Symbol	**Condition**
EXT AIR	Displayed when pressure detected and engine/APU bleeds are off.



SYNOPTIC PAGE – AIR

CS1_CS3_3620_005

Figure 9: Bleed Air System Indications

DETAILED DESCRIPTION

The active integrated air system controller (IASC) automatically performs engine bleed air 4th and 8th bleed port switching, and controls the operation of the pressure-regulating shutoff valve (PRSOV), when the associated BLEED PBA on the AIR panel is selected ON.

ENGINE BLEED AIR 4TH AND 8TH BLEED PORT SWITCHING

The high-pressure (HP)/intermediate-pressure (IP) switching selects the bleed air from the engine IP or HP port based on the system configuration (number of packs, WAIS on) and demand of the user systems. The IASC automatically selects the proper engine bleed port based on pressure requirements and temperature limitations.

Pressure requirements are based on the required bleed flow, user system configuration, and engine thrust settings as calculated by the electronic engine control (EEC).

Temperature limitations are based on the maximum allowable system temperature, and the minimum temperature to provide satisfactory WAIS operation.

The engine bleed port switching is done by electrical control of the HPV.

Pressure

The IASC opens the HPV based on the EEC intermediate pressure port (IPP) calculated pressure information. The EEC calculation is the minimum pressure necessary to provide the required total bleed airflow in the system.

If the actual pressure at the IP port, as indicated by the BMPS goes below a specified threshold, the IASC opens the HPV and the HP port supplies the pressure. The threshold varies based on the system configuration and demand. The HPV downstream pressure is limited to 75 +/- 5 psi.

To switch back to the engine IP port, the IASC uses the IPP information from the EEC. Once the actual pressure at the IP port matches the IPP calculated pressure, the IASC closes the HPV.

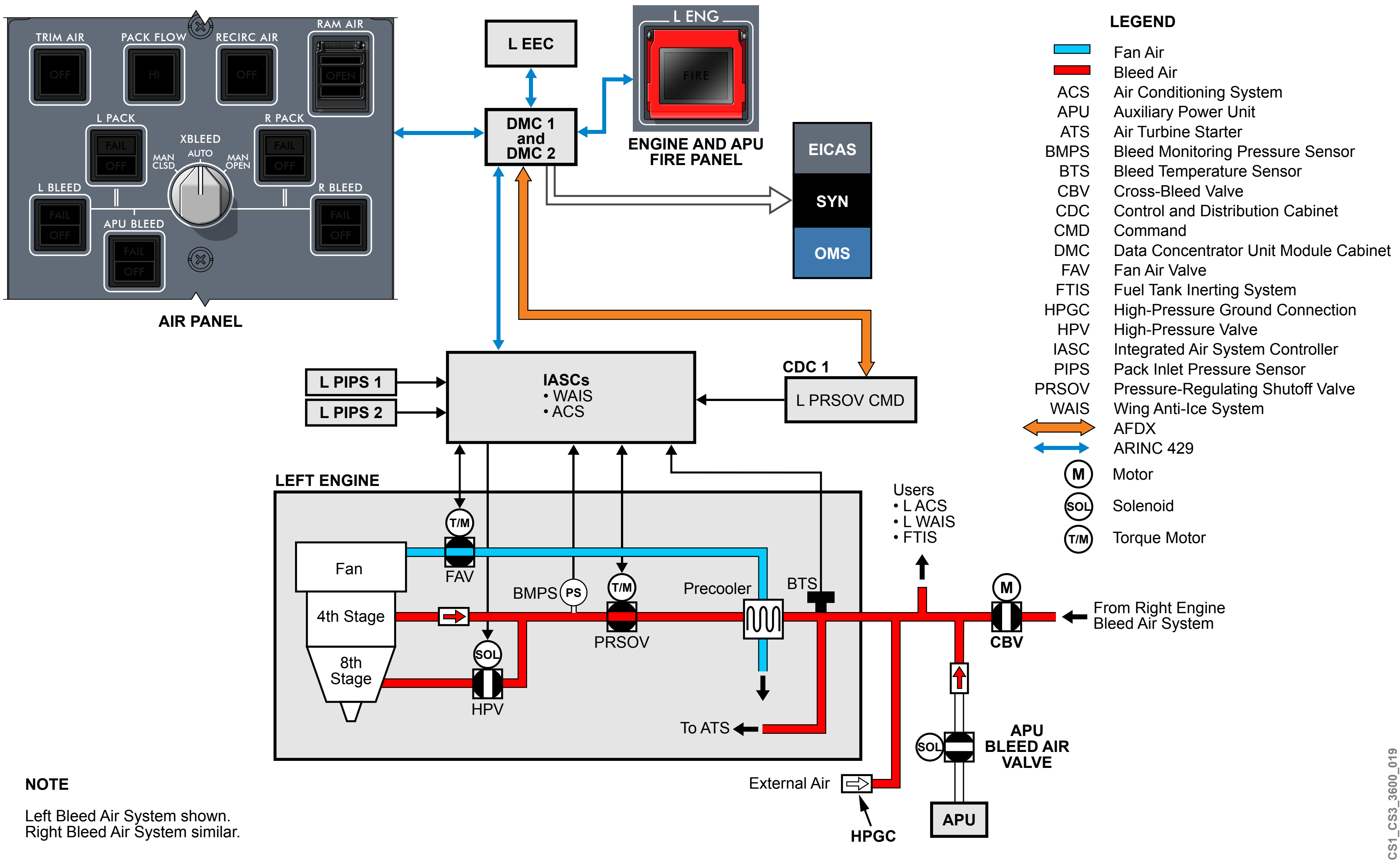
During, descent, and approach or steep approach, the HP/IP threshold is increased to prevent excessive bleed port switching.

Temperature

The bleed port switching logic also considers the IP temperature (IPT) to maintain WAIS performance under single or dual bleed source operation. If the IPT is lower than the WAIS threshold of 230°C (446°F), the IASC commands the HPV open. The HPV closes if the IPT increases above 240°C (464°F),

To protect the bleed air system downstream of the engine ports, the IASC inhibits the IP/HP switching when the HP port temperature (HPT) is above the maximum temperature threshold of 490°C (914°F). The HPV closes if the HPT increases above

If the HPV is open, and the maximum temperature threshold of 500°C (932°F) is reached, the IASC closes the HPV.



NOTE

Left Bleed Air System shown.
Right Bleed Air System similar.

Figure 10: Engine Bleed Port Switching

PRESSURE-REGULATING SHUTOFF VALVE OPERATION

When the BLEED PBA is selected ON, the control and distribution cabinet (CDC) provides an enable signal to the IASC, allowing the PRSOV to open and supply engine bleed air.

Once enabled, PRSOV operation is controlled by the IASC. The IASC uses the pressure signal supplied by the air conditioning pack inlet pressure sensors (PIPSs), to maximize air conditioning performance. The IASC modulates the torque motor of the PRSOV to provide bleed air at a pressure of 41.5 psi. Refer to ATA 21, Environmental Control for additional information on PIPS.

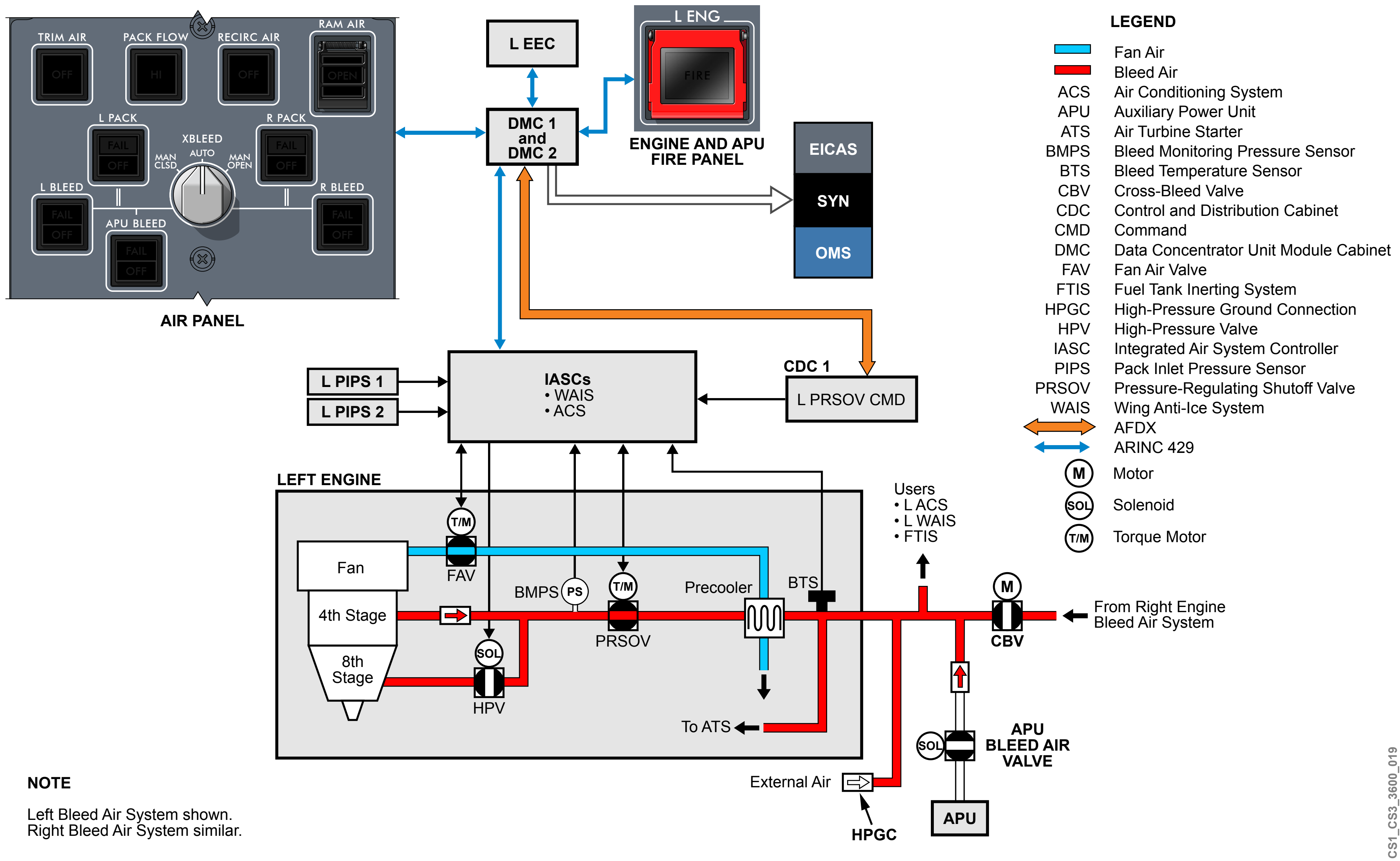
During a cross-bleed main engine start, the IASC controls the operating engine PRSOV to maintain 45 psi for engine starting. The PIPSs on the operating engine side provides the pressure signal to the IASC.

When the EEC signals that the engine is not running, the IASC closes the associated PRSOV and HPV.

The PRSOV is commanded closed by any of the following conditions:

- BLEED PBA is selected OFF
- During a start of the associated engine. The PRSOV remains closed for the duration of the start plus an additional 30 seconds during a ground start. In flight, the 30 second delay is removed.
- No bleed air flow demand (no WAI demand or air conditioning demand), provided the RAM AIR PBA is not selected open. If the RAM AIR PBA is selected open, the PRSOV is prevented from closing in order to provide trim air to the trim air system. Refer to ATA 21, Air Conditioning for more information
- The associated ENG FIRE PBA is selected. This signal is provided through the data concentrator unit module cabinets (DMCs)
- Overpressure of 45 psi (if bleed demand) or 60 psi (if no bleed demand), for more than 15 seconds, as reported by the associated PIPS. This protects the bleed air ducting from high engine bleed pressure
- The BTS detects an overtemperature (refer to Fan Air Valve Operation)
- The bleed air leak and overheat detection system (BALODS) detects a bleed air leak in the downstream ducting

If the PRSOV fails in the open position, the amber FAIL light illuminates in the associated L (R) PBA.



NOTE

Left Bleed Air System shown.
Right Bleed Air System similar.

Figure 11: PRSOV Operation

FAN AIR VALVE OPERATION

The IASC uses feedback from the BTS to control the fan air valve (FAV) in order to maintain a BAS temperature range between 170°C (338°F) and 232°C (450°F). The temperature regulation setpoint is determined by the IASC based on the number of engines operating, the number of air conditioning packs operating, and wing anti-ice system (WAIS) operation. The nominal setpoint with air conditioning running is 200°C (392°F).

With the L (R) BLEED PBA selected on and the associated engine operating, the IASC enables the FAV to open.

When the engine bleed air supply is commanded closed automatically or the L (R) BLEED PBA is selected OFF, and the PRSOV is confirmed closed by the IASC, the FAV closes. Should the PRSOV fail in the open position, the FAV to stay open to allow fan air cooling to the precooler.

The FAV closes simultaneously with the PRSOV when the associated ENG FIRE PBA is selected.

If the BTS detects a temperature greater than 257°C (495°F) for more than 30 seconds, the L (R) BLEED OVHT caution message is displayed on EICAS and the FAIL light on the L (R) BLEED PBA illuminates. The IASC closes the associated PRSOV (and HPV). The associated BLEED PBA on the AIR panel is selected OFF to match the valve position.

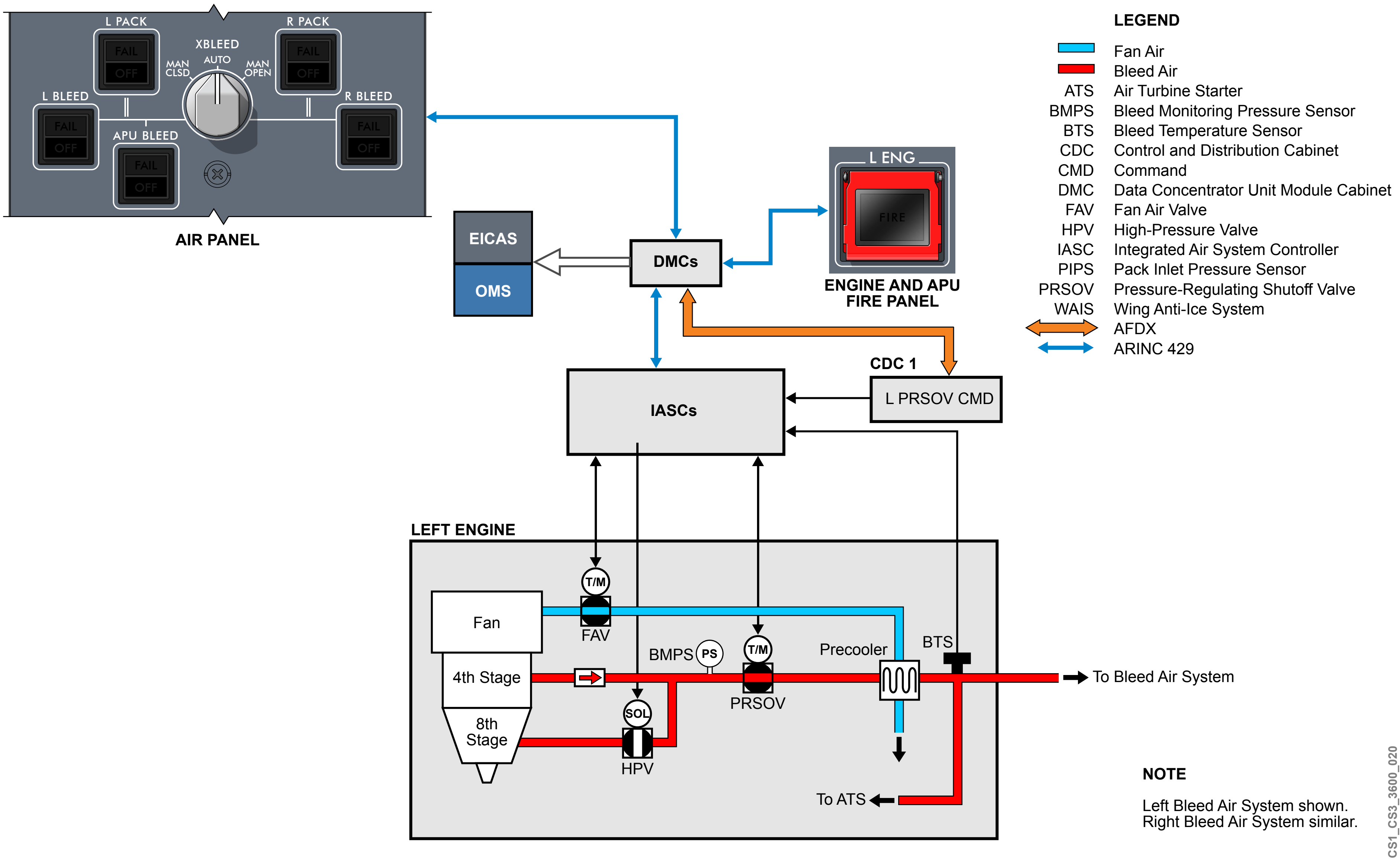


Figure 12: Fan Air Valve Operation

PRECOOLER EXHAUST DOOR

The precooler exhaust (PCE) door is located on the right inner fixed structure (IFS) panel at the 3 o'clock position. The door actuator is controlled by channel A of the electronic engine control (EEC). Operation of the actuator unlatches the exhaust door which is pushed open by two pistons.

The PCE door can be opened electrically by the EEC, or manually by rotating the latches in a counterclockwise direction using a 0.635 cm (0.250 in.) drive tool. The door can only be closed manually.

The PCE door has square drive adaptors to operate the upper and lower pistons, actuator, and striker latch. When closing the PCE door, an actuator reset check hole is used to confirm the actuator is in the correct position before securing the upper and lower pistons.

When the precooler demand is high, the door opens based on logic from the EEC. The door actuator receives a 28 VDC signal command from EEC when all the following conditions are true:

- Single bleed air operation (from IASC)
- Wing anti-ice is on (from IASC)
- Bleed air is being extracted from the engine high-pressure port (from IASC)
- Precooler outlet bleed temperature from the IASCs is above threshold for more than 10 seconds as determined by the EEC. The EEC determines the temperature limit based on altitude
- Aircraft in flight

The L (R) ENG PCE DOOR OPEN advisory message is displayed on EICAS when the precooler exhaust (PCE) door is open.

The EEC monitors the door latch actuator commands and reports failures to the onboard maintenance system (OMS).

The PCE door opening function is also tested from the OMS.

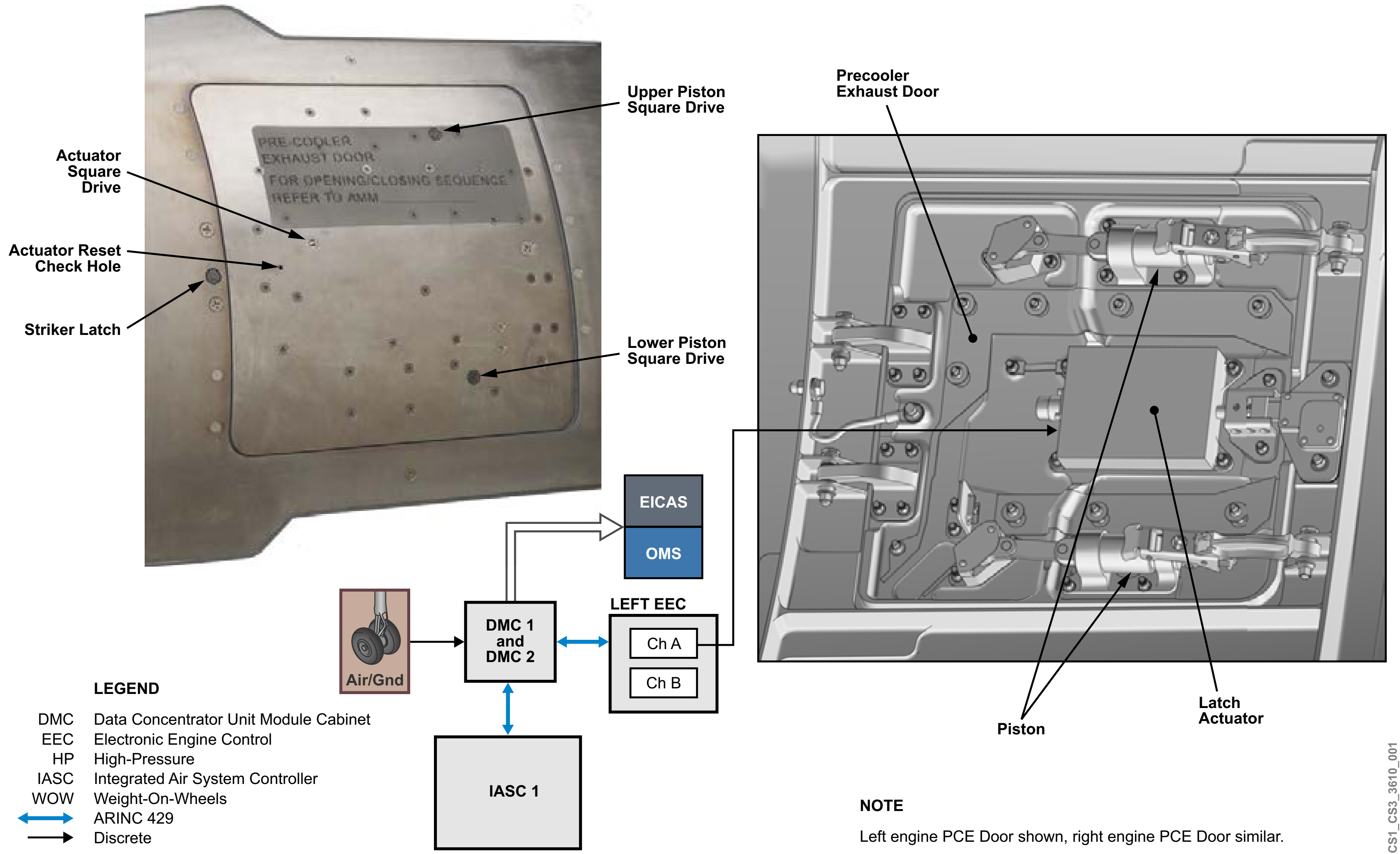


Figure 13: Precooler Exhaust Door Operation

BLEED AIR SOURCE SELECTION

The IASC provides the logic for switching between the available bleed air sources.

When the engines and the APU are not running, or their associated bleed switches are off, the IASC assumes a no bleed state. Pneumatic air from an external air source can then be used to supply air to the BAS. When bleed pressure is detected with no engine or APU source selected, the EXT AIR symbol appears on the AIR synoptic page.

The IASC selects the APU as the default bleed air source, when the APU electronic control unit (ECU) provides a ready-to-bleed signal, and all of the following conditions apply:

- The APU BLEED PBAs is in the on (normal) position
- The WING ANTI-ICE switch is in the AUTO position and the system has not been activated.
- Slats and flaps extended (not fully retracted)
- Aircraft altitude <15,500 ft

In addition, whenever both engines are selected OFF (or not running), as signaled from the EECs, the IASCs switch to the APU as the default bleed air source. When the IASC selects APU as the bleed air source, it signals the APU ECU to open the APU bleed air valve (BAV) and the cross-bleed valve (CBV).

During and after engine start, the APU remains the default bleed air source. It supplies the environmental control system (ECS) on the ground and during takeoff to minimize the bleed demand on the engines.

The IASC automatically switches to engine bleed air, provided either or both engines are running and the associated BLEED PBA is in the AUTO position, and any of the following conditions apply:

- APU BLEED PBA selected OFF or the APU is not ready to bleed
- WING ANTI-ICE switch selected to ON or in AUTO and active
- Slats and flaps fully retracted
- Aircraft altitude is less than 16,000 ft
- Ice is detected above 1,500 ft or 2 minutes after takeoff

When the IASC switches to the engine bleed air source, the APU BAV closes through the APU ECU and both PRSOVs are opened.

During descent, with the APU available, the bleed air system switches from the engines to the APU when the aircraft altitude is less than 15,500 ft, flaps and slats are extended, and wing anti-ice is not required.

The ENG BLEED MISCONFIG caution message is displayed on EICAS if both engine BLEED PBAs are selected OFF, and a low bleed air flow is detected from the APU. This indicates that the APU is unable to support the bleed air users while the APU is the bleed source.

CROSS-BLEED VALVE OPERATION

The cross-bleed valve (CBV) is directly controlled by the IASCs channel B. With the XBLEED switch in the AUTO (normal) position, the IASC automatically controls the opening and closing for the CBV. The crew can force the IASC to command the CBV open or closed by selection of the MAN OPEN or MAN CLSD positions. When the CBV is manually operated, XBLEED MAN OPEN or XBLEED MAN CLSD EICAS status messages are displayed.

In the AUTO position, the IASC commands the CBV open whenever a single bleed source is available (one PRSOV open, APU bleed air source or external air). The CBV is closed when both PRSOVs are open.

The ENG BLEED MISCONFIG caution message is displayed on EICAS if the XBLEED valve is selected to MAN OPEN while both engine BLEED PBAs are in the ON position.

If the IASC detects that the CBV has failed to move (failed in position) when commanded, the XBLEED FAIL caution message is displayed on EICAS. This situation can be detected whether the XBLEED switch is selected to AUTO or any of the MAN positions. Under these conditions the IASC latches the EICAS message until an initiated built-in test (IBIT) is performed.

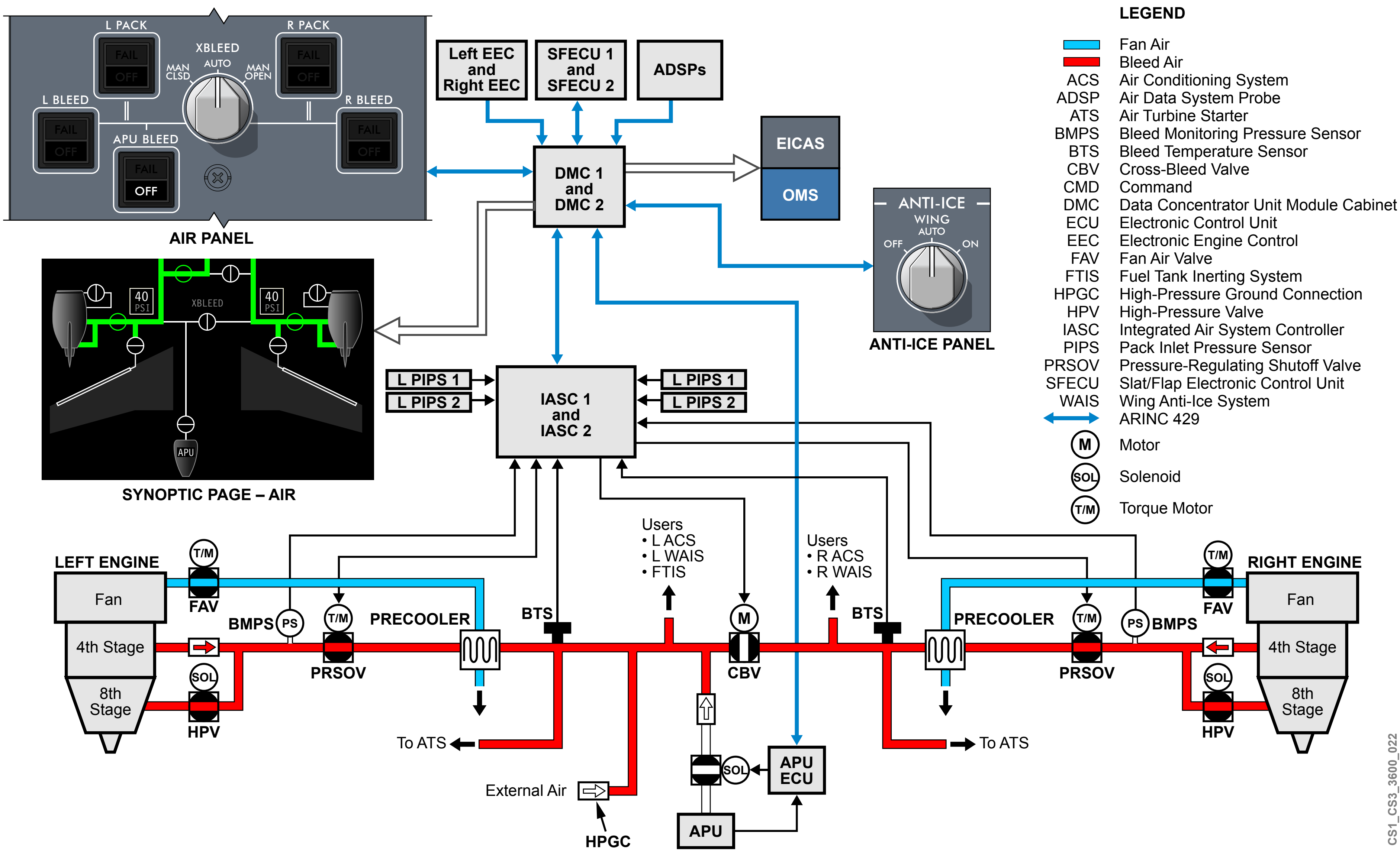


Figure 14: Bleed Air Source Selection and Cross-Bleed Valve Operation

MONITORING AND TESTS

The following page displays the crew alerting system (CAS) and INFO messages associated with the bleed air system.

CAS MESSAGES

Table 1: CAUTION Messages

MESSAGE	LOGIC
L BLEED FAIL	Left engine bleed failure or loss of left bleed leak detection or loss of both IASC 1 channels.
R BLEED FAIL	Right engine bleed failure or loss of right bleed leak detection or loss of both IASC 2 channels.
APU BLEED FAIL	The APU is not able to provide bleed air when requested or loss of bleed leak detection.
L BLEED OVHT	Left engine bleed overtemperature detected.
R BLEED OVHT	Right engine bleed overtemperature detected.
ENGINE BLEED MISCONFIG	XBLEED valve selected open and both engine bleed selected on or low flow on APU and left and right engine BLEED PBAs OFF.
XBLEED FAIL	XBLEED valve failed in position in AUTO or MAN mode.

Table 2: ADVISORY Messages

MESSAGE	LOGIC
AIR SYSTEM FAULT	Loss of redundant or non-critical function for the air system (bleed or trim).
L ENG PCE DOOR OPEN	L PCE door commanded opened (due to precooler overtemperature condition).
R ENG PCE DOOR OPEN	R PCE door commanded opened (due to precooler overtemperature condition).

Table 3: STATUS Messages

MESSAGE	LOGIC
L BLEED OFF	Left engine bleed selected OFF.
R BLEED OFF	Right engine bleed selected OFF.

Table 3: STATUS Messages

MESSAGE	LOGIC
XBLEED MAN CLSD	Crossbleed valve manually selected and confirmed closed.
XBLEED MAN OPEN	Crossbleed valve manually selected and confirmed opened.
APU BLEED OFF	APU bleed selected OFF.

Table 4: INFO Messages

MESSAGE	LOGIC
36 L BLEED FAIL - L HPV FAIL CLSD	Left HPV detected failed closed by either left IASC channel.
36 L BLEED FAIL - L HPV FAIL OPEN	Left HPV detected failed open by either left IASC channel.
36 L BLEED FAIL - L PRESS REG SOV INOP	Left PRSOV causing a low or high-pressure.
36 L BLEED FAIL - L PRESS SOV FAIL OPEN	Left PRSOV detected failed open by either left IASC channel.
36 R BLEED FAIL - R HPV FAIL CLSD	Right HPV detected failed closed by either right IASC channel.
36 R BLEED FAIL - R HPV FAIL OPEN	Right HPV detected failed open by either right IASC channel.
36 R BLEED FAIL - R PRESS REG SOV INOP	Right PRSOV causing a low or high-pressure.
36 R BLEED FAIL - R PRESS REG PRESS SOV HPV FAIL OPEN	Right PRSOV detected failed open by either right IASC channel.
36 L BLEED FAIL - L TEMP SNSR INOP	L BTS failure.
36 R BLEED FAIL - R TEMP SNSR INOP	R BTS failure.

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Table 4: INFO Messages

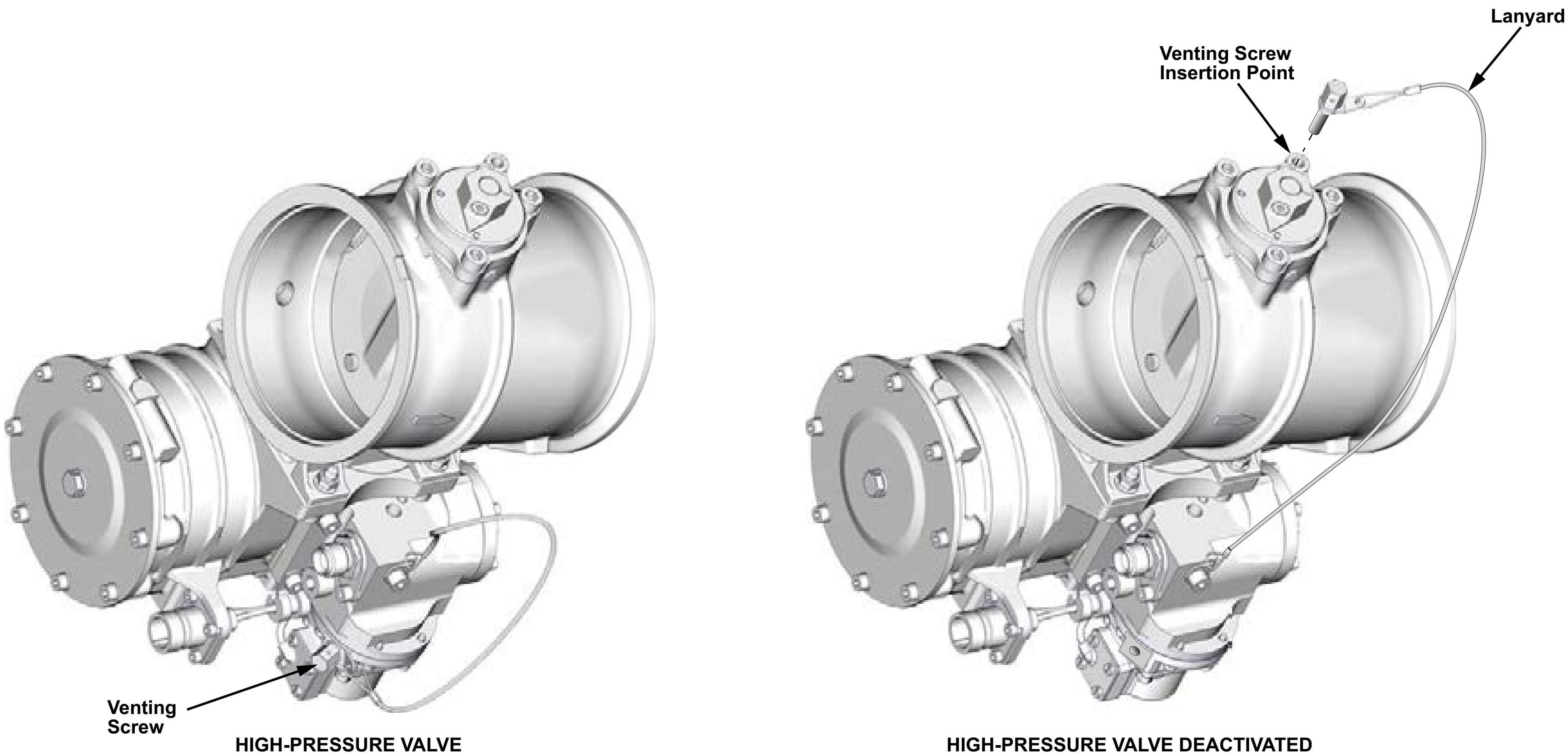
MESSAGE	LOGIC
36 AIR SYSTEM FAULT - L BLEED TEMP SNSR REDUND LOSS	L BTS drift or single channel failure.
36 AIR SYSTEM FAULT - R BLEED TEMP SNSR REDUND LOSS	R BTS drift or single channel failure.
36 AIR SYSTEM FAULT - L BLEED MON PRESS SNSR INOP	Left BMPS failure.
36 AIR SYSTEM FAULT - R BLEED MON PRESS SNSR INOP	Right BMPS failure.

PRACTICAL ASPECTS

HIGH-PRESSURE VALVE, PRESSURE-REGULATING SHUTOFF VALVE, AND FAN AIR VALVE DEACTIVATION

The high-pressure valve (HPV), pressure-regulating shutoff valve (PRSOV), and the fan air valve (FAV) can be secured closed for dispatch by rotating the manual override lever to the CLOSED position.

The venting screw is used to secure the manual override lever in the closed position. Removing the vent screw from its stowed position vents the pneumatic actuator and prevents valve operation.



NOTE

High-pressure valve shown, pressure-regulating shutoff valve and fan air valve similar.

CS1_CS3_3600_015

Figure 15: High-Pressure Valve, Pressure-Regulating Shutoff Valve, and Fan Air Valve Deactivation

CROSS-BLEED VALVE DEACTIVATION

The cross-bleed valve (CBV) can be secured closed by rotating the manual override lever to the closed position and securing the lever with lockwire.

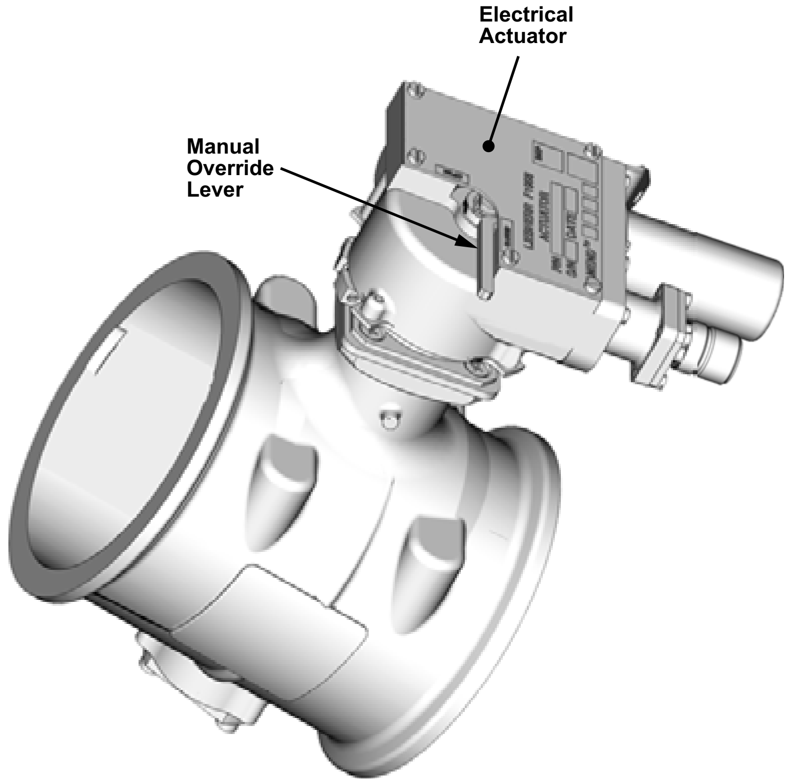


Figure 16: Cross-Bleed Valve Deactivation

PRECOOLER EXHAUST DOOR MANUAL OPENING AND CLOSING

The PCE door can be manually operated using a square drive tool.

Open the PCE Door

1. Rotate the upper piston assembly square drive 90° counterclockwise.
2. Rotate the lower piston assembly square drive 90° counterclockwise.
3. Rotate the striker latch square drive 60° counterclockwise.
4. Manually pull the door to the open position.
5. Rotate the upper piston square drive 90° clockwise.
6. Rotate the lower piston assembly square drive 90° clockwise.
7. Rotate the striker latch square drive 60° clockwise.

Close the PCE Door

1. Rotate the upper piston assembly square drive 90° counter-clockwise.
2. Rotate the lower piston assembly square drive 90° counterclockwise.
3. Rotate the striker latch square drive 60° counterclockwise.
4. Rotate the actuator square drive 10.5° counterclockwise.

NOTE

The square drive will return to initial position.

5. Manually push the pre-cooler exhaust door to the closed position.
6. Hold the pre-cooler exhaust door in the closed position and rotate the striker latch square drive 60° clockwise.
7. Insert a 1/8 in. allen key into the actuator reset check hole. Verify that the allen key can be inserted from 1.524 to 3.048 cm (0.6 to 1.2 in.). If not, repeat the previous steps.
8. Rotate the upper piston assembly square drive 90° clockwise.
9. Rotate the lower piston assembly square drive 90° clockwise.

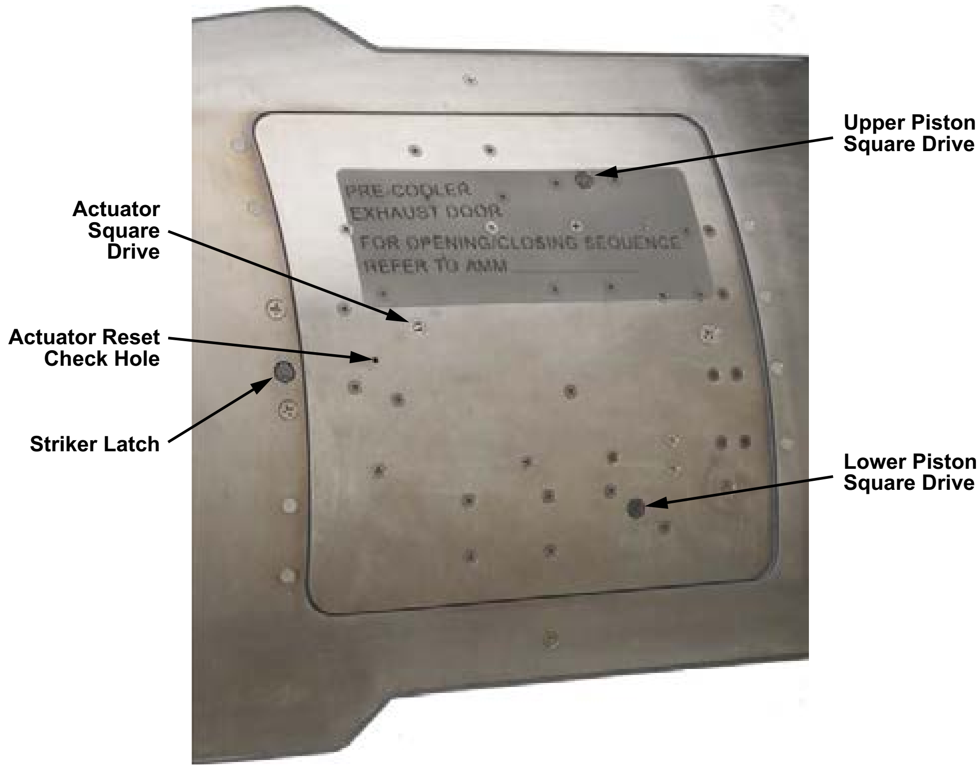
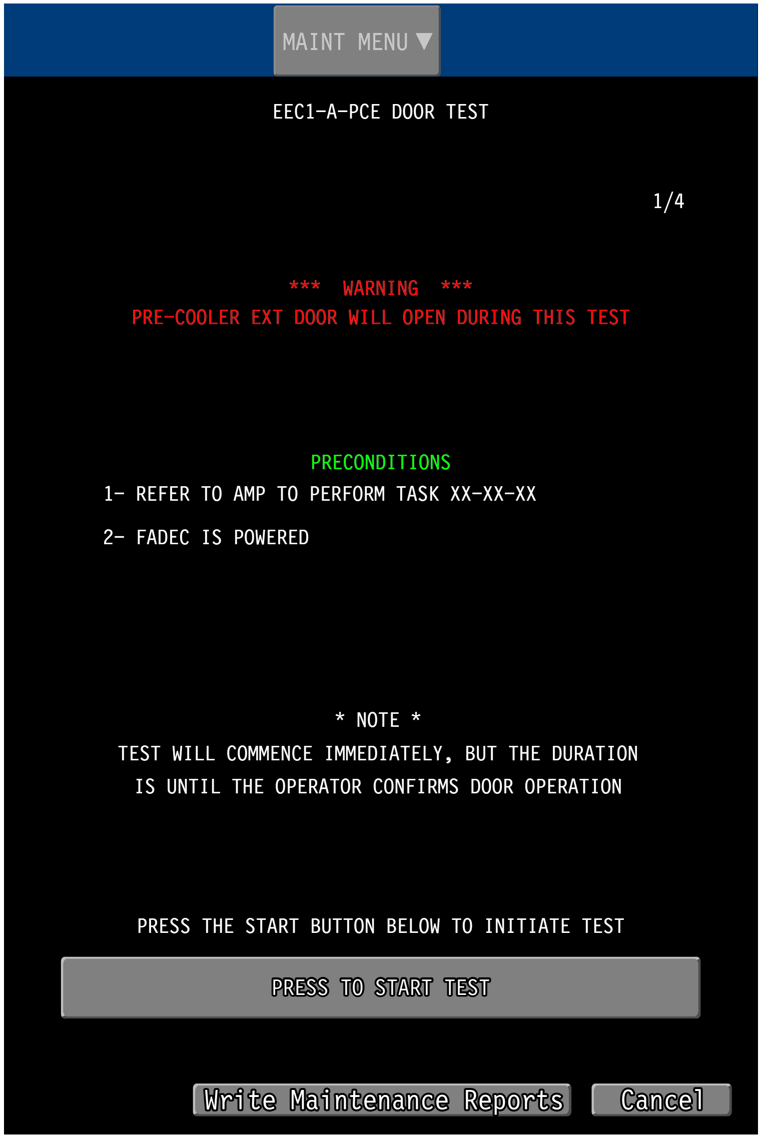


Figure 17: PCE Exhaust Door Manual Opening and Closing

PCE DOOR TEST

The PCE Door Test is used to detect any operational faults of the PCE system, including the electrical interface, solenoid, and door assembly.



CS1_CS3_7321_025

Figure 18: Precooler Exit Door Test

ONBOARD MAINTENANCE SYSTEM TESTS

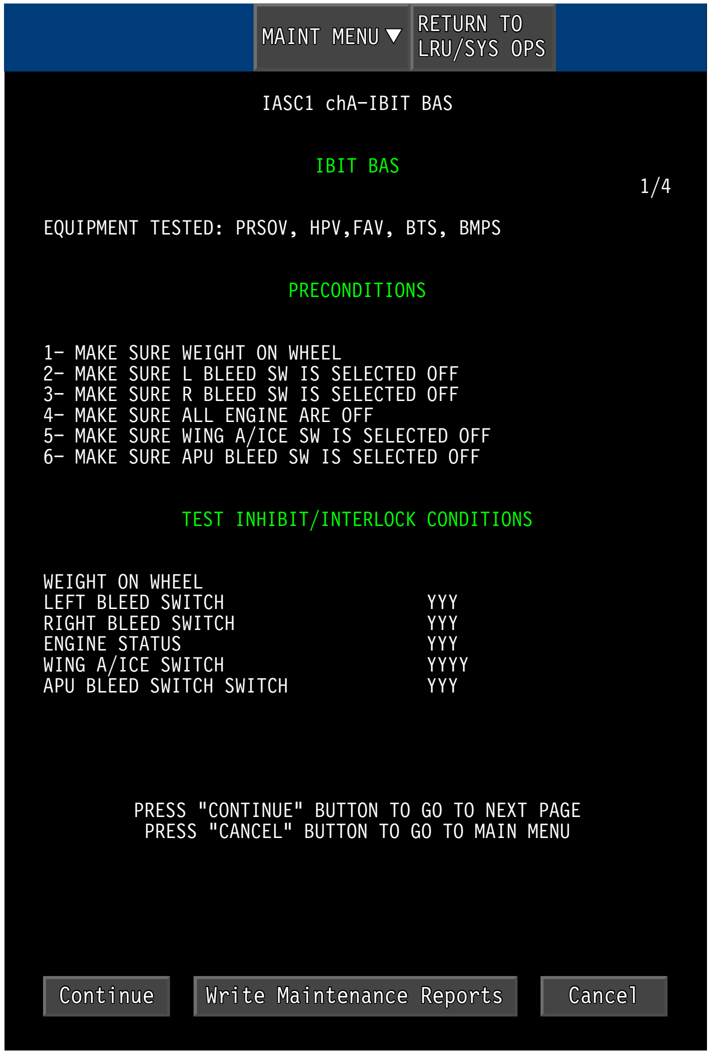
On power-up, each channel of the IASCs performs a power-up built-in test (PBIT) to validate the input and system functions. During system operations each channel performs a continuous built-in test (CBIT) to validate input, active function operations, and proper active output operations.

PBIT and CBIT monitor adjacent and opposite channel faults and failures, and store these in each channel’s memory for maintenance retrieval.

Initiated built-in tests (IBIT) of the PRSOV, HPV, FAV, BTS and BMPS can be conducted on the OMS.

Preconditions and instructions for the tests are provided on the test pages.

The OMS also provides data pages for alarms status, valve position, and sensor status. Refer to ATA 45, Central Maintenance System for additional information.



CS1_CS3_3600_006

Figure 19: Bleed Air System Initiated Built-In Test

36-21 BLEED AIR LEAK AND OVERHEAT DETECTION SYSTEM

GENERAL DESCRIPTION

The purpose of the bleed air leak and overheat detection system (BALODS) is to protect the aircraft from hot bleed air leaks, and subsequent overheat damage. The BALODS is a continuous monitoring system that detects the location of hot air leaks or overheat, and automatically isolates or reconfigures the affected system. The BALODS uses dual overheat detection loops that monitor the high-pressure ducting and air conditioning bays.

Upon detection of overheat in the affected zone, the integrated air system controllers (IASCs) automatically command one or more valves to close within the following systems:

- Bleed air system: Left pressure-regulating shutoff valve (PRSOV), right PRSOV and cross-bleed valve (CBV)
- Wing anti-ice system: Left and right wing anti-ice valves (WAIVs)
- Air conditioning system: Left flow control valve (FCV) and right FCV
- Trim air system (a sub-system of air conditioning): Left trim air shutoff valve (TASOV), and the right trim air pressure-regulating valve (TAPRV)
- Auxiliary power unit (APU) bleed air: APU bleed air valve

Indication of the overheat condition is displayed on the engine indication and crew alerting system (EICAS) messages and on the AIR synoptic page. Failure of bleed leak detection capability is indicated by the EICAS messages and FAIL lights on the AIR panel.

The IASCs are also capable of identifying and recording the location of the leak for use in troubleshooting. This information is available through the onboard maintenance system (OMS).

After an overheat detection and automatic reconfiguration, the flight crew may manually reconfigure the affected system from the AIR control panel.

The BALODS system is active when the integrated air system controllers (IASCs) are powered. The BALODS function of the IASC is controlled by channel B, which is powered by DC ESS BUS 2 for IASC 1 and DC ESS BUS 3 for IASC 2.

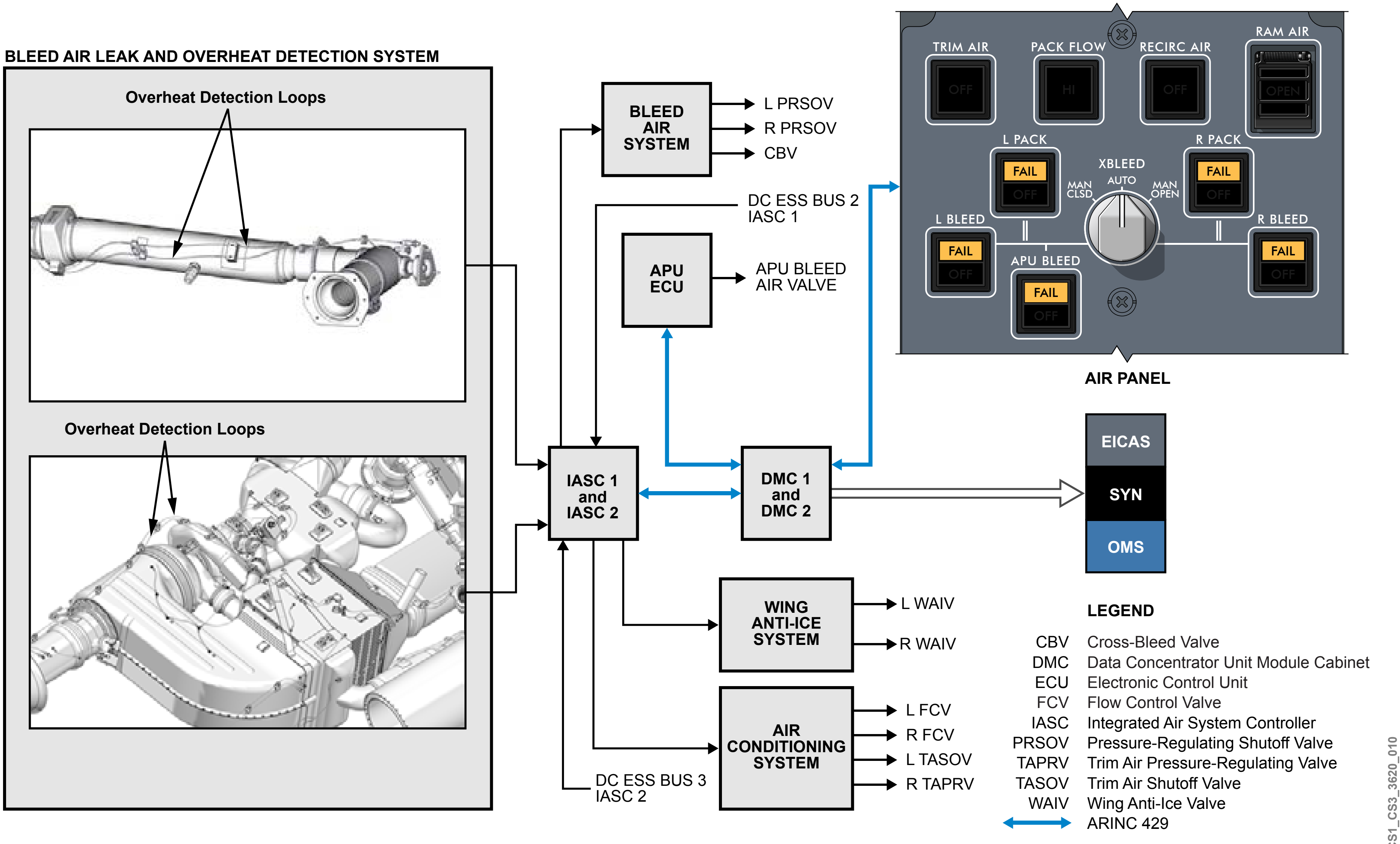


Figure 20: BALODS Operation

BLEED AIR LEAK AND OVERHEAT DETECTION SYSTEM OPERATION

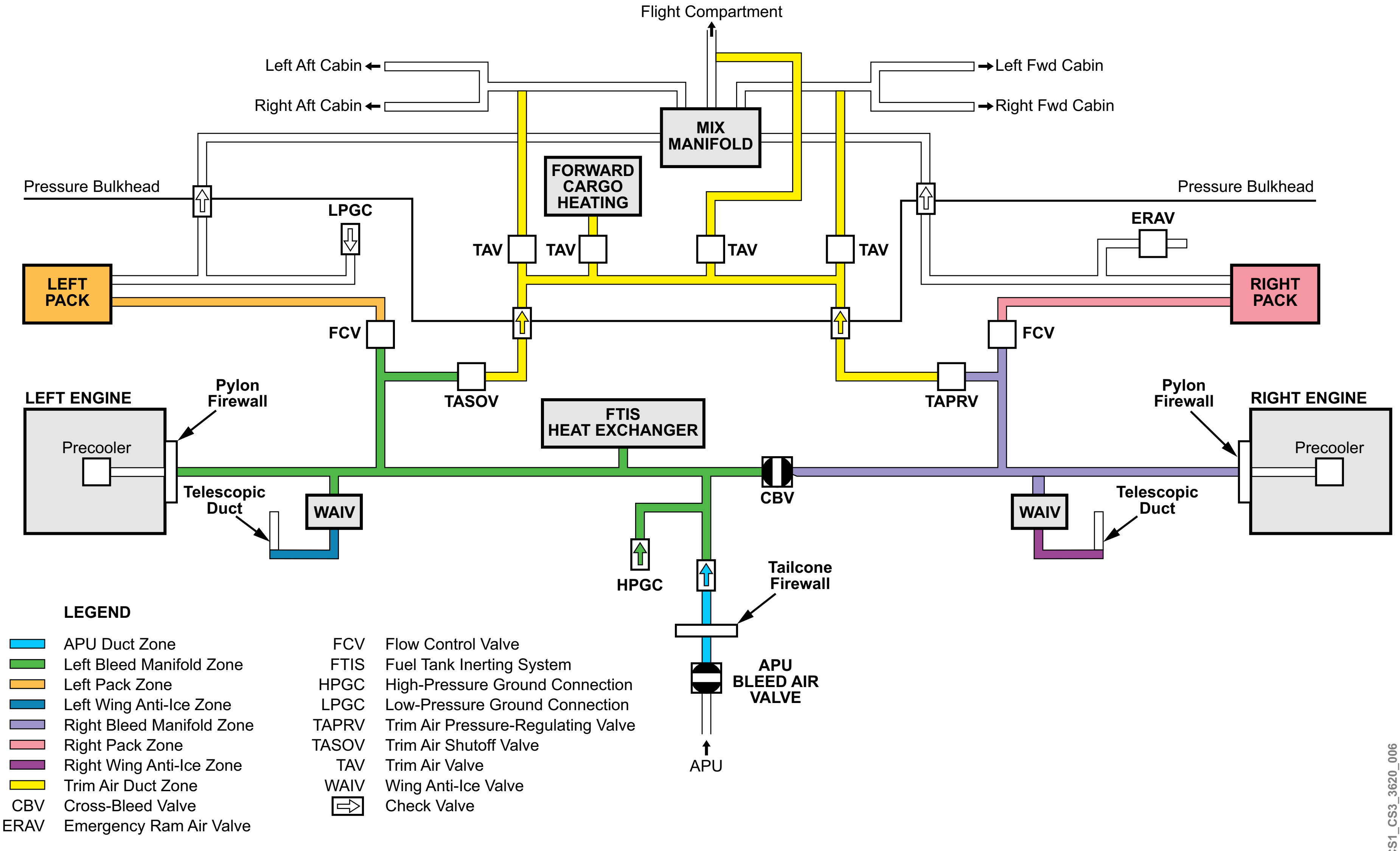
The dual-detection loops are installed along the high-pressure air ducts and associated components, in the following eight zones:

- Left bleed manifold zone: Located downstream of the left engine pylon firewall and along the left inboard wing leading edge to the cross-bleed valve, the fuel tank inerting system (FTIS) heat exchanger, the auxiliary power unit check valve, the high-pressure ground connection, the trim air shutoff valve (TASOV), and on to the left flow control valve (FCV)
- Right bleed manifold zone: Located downstream of the right engine pylon firewall and along the right inboard wing leading edge to the cross-bleed valve (CBV), the trim air pressure-regulating valve (TAPRV), and on to the right FCV
- Left wing anti-ice zone: Located downstream of the left wing anti-ice valve to the left wing telescopic duct. There are no detection loops in the slats
- Right wing anti-ice zone: Located downstream of the right wing anti-ice valve to the right wing telescopic duct. There are no detection loops in the slats.
- Left pack zone: Located downstream of the left FCV to the mix manifold and associated aft cabin ducting
- Right pack zone: Located downstream of the right FCV to the mix manifold and associated forward cabin ducting
- Trim air duct zone: Located downstream of the left TASOV and right TAPRV to the respective ducting of the flight deck, forward cabin, aft cabin, and forward cargo trim air valves. In contrast to the forward and aft cabin, where only the trim air ducting is monitored, the complete air conditioning ducting to the flight deck, is monitored
- APU duct zone: Located downstream of the APU bleed air valve to the APU check valve

Upon detection of overheat in the affected zone, the IASCs automatically perform the following actions within 15 seconds:

- Left bleed manifold zone:
  - If on engine bleed, close the left pressure-regulating shutoff valve (PRSOV) and close the cross-bleed valve (CBV)
  - If on APU bleed, the IASCs sets the APU bleed request, to the APU electronic control unit (ECU) to false (closing the APU bleed air valve) and command the CBV to close
- Right bleed manifold zone:
  - If on engine bleed, close the right PRSOV and close the CBV
  - If on APU bleed, close the CBV
- Left wing anti-ice zone: Close both wing anti-ice valves (WAIVs)
- Right wing anti-ice zone: Close both WAIVs
- Left pack zone: Close the left flow control valve (FCV)
- Right pack zone: Close the right FCV
- Trim air duct zone: Close the left TASOV and the right TAPRV
- APU duct zone: Set the APU bleed request to false. The ECU, upon receiving the signal, closes the APU bleed air valve

The valves are locked out until the appropriate bleed air system (BAS) pushbutton annunciator (PBA) is selected OFF.



LEGEND

	APU Duct Zone
	Left Bleed Manifold Zone
	Left Pack Zone
	Left Wing Anti-Ice Zone
	Right Bleed Manifold Zone
	Right Pack Zone
	Right Wing Anti-Ice Zone
	Trim Air Duct Zone
CBV	Cross-Bleed Valve
ERAV	Emergency Ram Air Valve
FCV	Flow Control Valve
FTIS	Fuel Tank Inerting System
HPGC	High-Pressure Ground Connection
LPGC	Low-Pressure Ground Connection
TAPRV	Trim Air Pressure-Regulating Valve
TASOV	Trim Air Shutoff Valve
TAV	Trim Air Valve
WAIV	Wing Anti-Ice Valve
⇨	Check Valve

Figure 21: BALODS Zones

COMPONENT LOCATION

The principal components of the bleed air leak and overheat detection system (BALODS) are:

- Integrated air system controllers
- Dual-detection loops (Refer to figure 23)
- Manifolds (Refer to figure 23)

INTEGRATED AIR SYSTEM CONTROLLERS

The integrated air system controllers (IASCs) are located in the mid equipment bay forward rack.

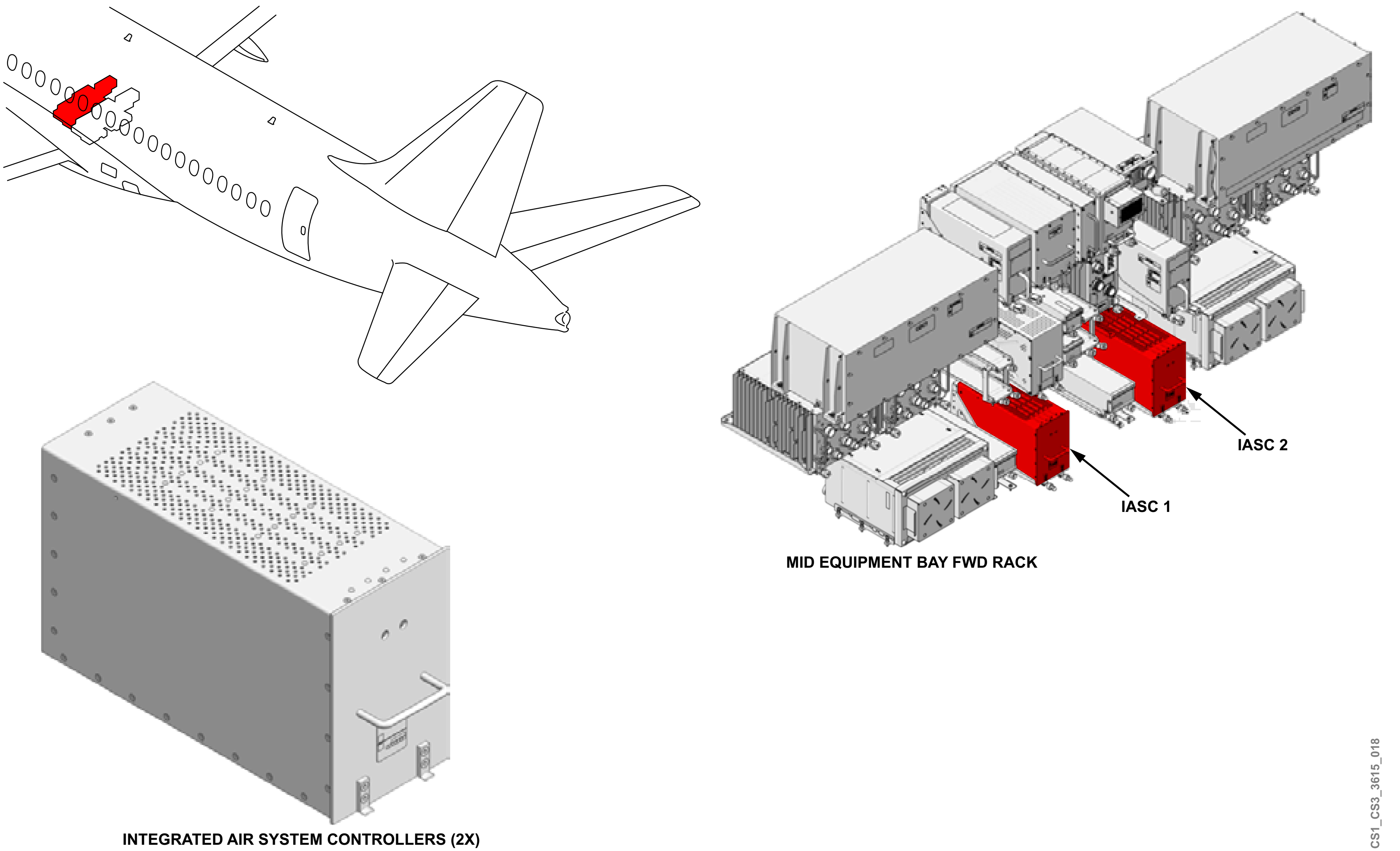


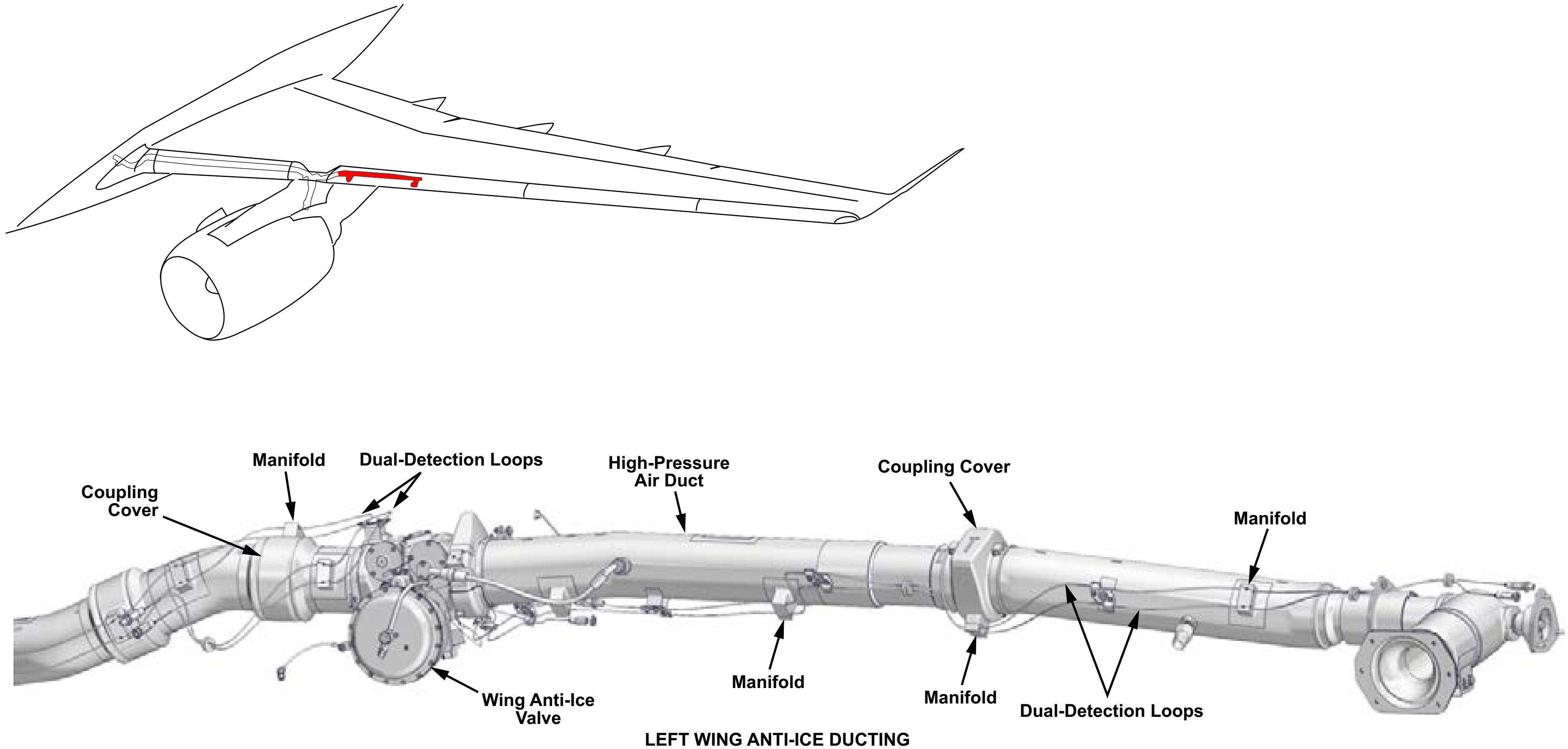
Figure 22: Integrated Air System Controller

DUAL-DETECTION LOOPS

The dual-detection loops are located on high-pressure bleed air ducts and in areas of bleed air system components.

MANIFOLDS

The manifolds are located on the high-pressure air ducts and coupling covers.



NOTE

Dual-detection loops and manifolds are similar in other areas of the aircraft high-pressure air ducts and components.

CS1_CS3_3615_012

Figure 23: Bleed Air Leak and Overheat Detection System Components

DETAILED COMPONENT INFORMATION

DETECTION-LOOPS

Dual-detection loops consist of parallel rows of sensing elements. The sensing elements are connected in series, either directly to the next element or using a wire to interconnect elements where there is little chance of exposure to high temperatures. The loops are identified as loop A and loop B. Both ends of each loop are connected to the integrated air system controller (IASC). The loops function independently to minimize false leak indications and improve dispatch reliability.

The loops are powered by the integrated air system controller (IASC) with 15 VAC. When a loop is exposed to a hot air leak its resistance changes and is detected by the IASC.

MANIFOLDS

Dual-detection loops are installed on manifolds that are located on high-pressure air ducts and coupling covers. The purpose of the manifolds is to direct hot air leaks to the loops when a leak occurs. The loops are also routed over hot bleed air components in some zones to provide overheat damage protection.

A hot bleed air leak in the duct assemblies is contained by the metal (stainless steel) shroud. Bleed air flows through the insulation air gaps and is routed to the manifold which directs the hot air onto the dual-detection loops, triggering an overheat condition.

The manifolds on the coupling covers differ from those on the duct assemblies. A poppet valve (spring-loaded ball) is used to prevent nuisance warnings caused by the allowable natural leakage at the flange interfaces. This arrangement allows a minor buildup of pressure before the poppet valve opens releasing the hot air and heating the loops.

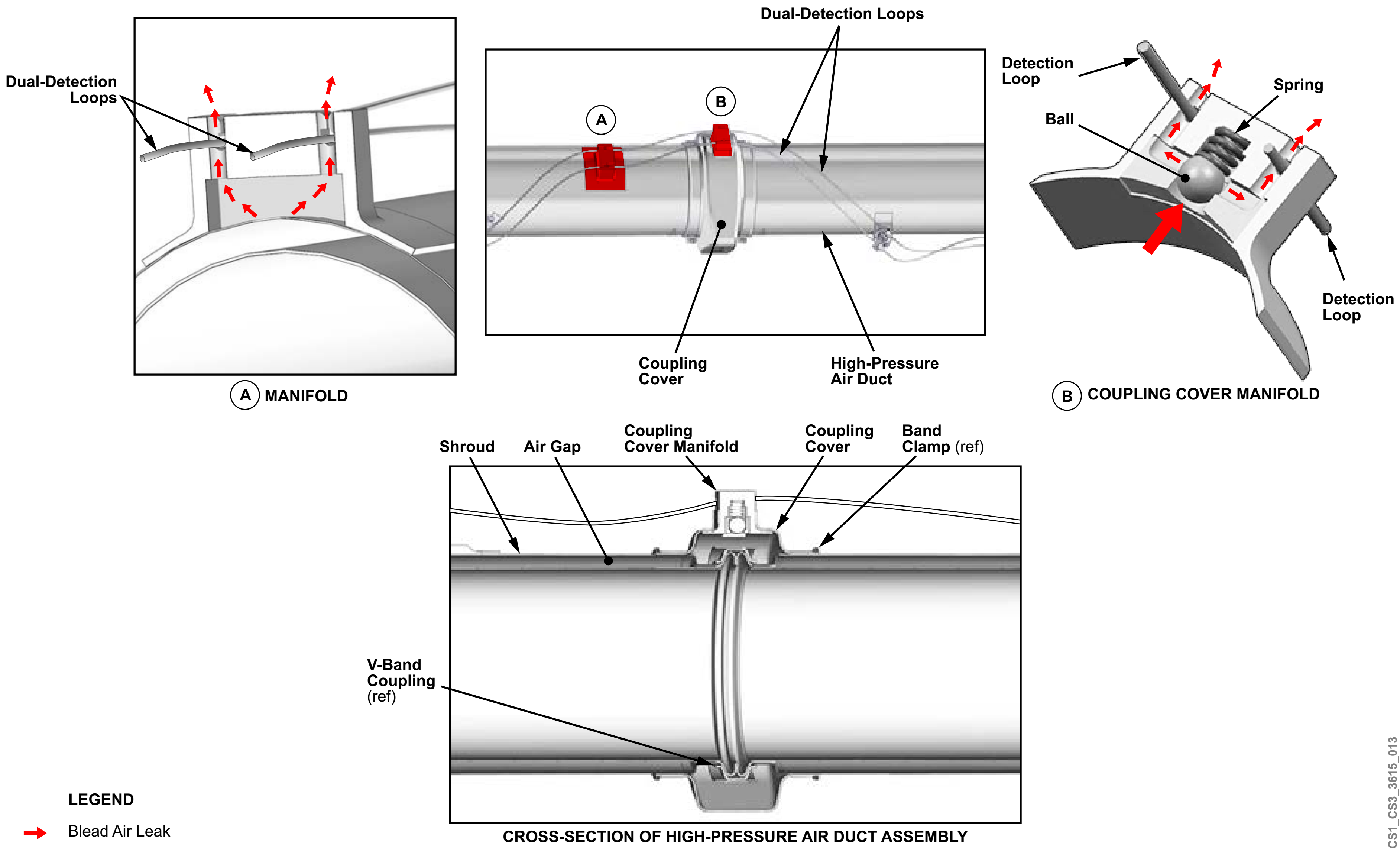


Figure 24: Manifold

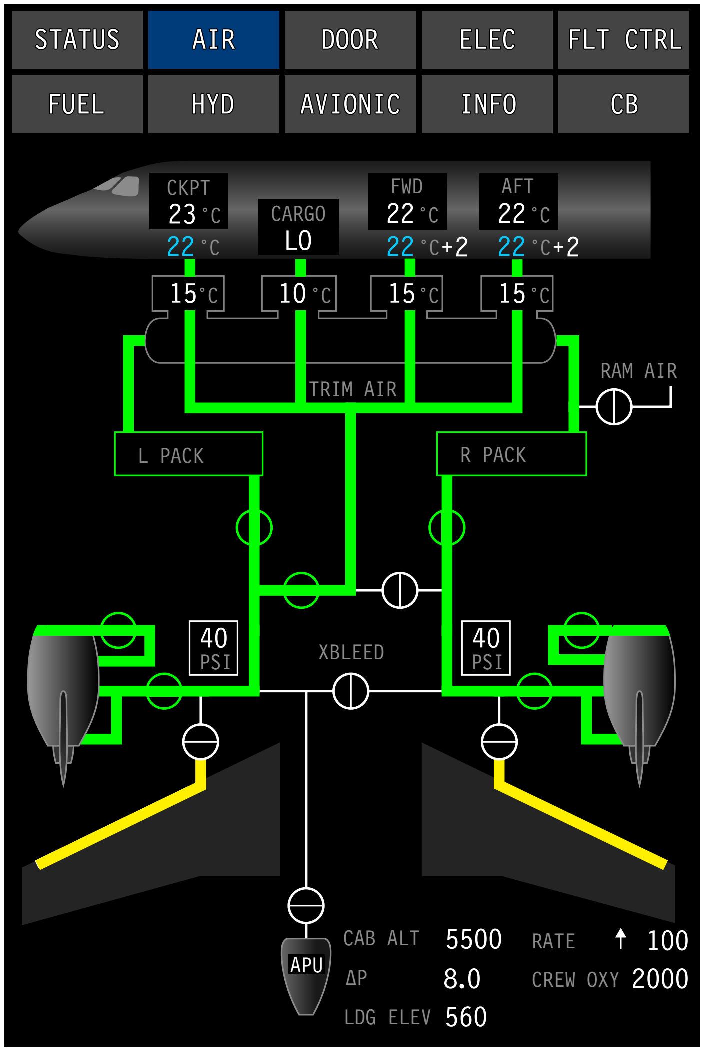
CONTROLS AND INDICATIONS

The detection of a bleed air leak or overheat is displayed on the AIR synoptic page.

Green flow lines indicate the normal presence of bleed air.

When a leak is detected, the affected zone flow line(s) turn amber. A LEAK caution message is displayed on EICAS for the affected zone.

White flow lines indicate that the particular zone has been confirmed isolated by the selection of the corresponding PBA or switch to OFF.



SYNOPTIC PAGE – AIR

FLOW LINES	
Symbol	Condition
	Valve displayed in white and closed position.
	After isolation flow line turns to white no flow.
	Overheat displayed by amber flow line.

CS1_CS3_3620_007

Figure 25: Bleed Air Leak and Overheat Detection System Indications

DETAILED DESCRIPTION

BLEED AIR LEAK AND OVERHEAT DETECTION SYSTEM ACTIVATION

Loop A of each dual-detection loop is connected to IASC 1 channel B, and loop B of each dual-detection loop is connected to IASC 2 channel B.

Loop A and loop B of each detection loop is supplied with 15 VAC, and the IASC monitors the resistance of the current passing through each loop. During an overheat condition, the resistance of loop A and loop B changes. At a predetermined temperature, the IASC signals an overheat condition and automatically closes the valves in the affected system.

Normal Mode Operation

Under normal operation, detection by both of the dual-detection loops trigger the IASC with an overheat condition. The IASC automatically shuts down the affected zone and the appropriate crew alerting system (CAS) message is displayed on EICAS.

Degraded Mode Operation

The BALODS operates in a degraded mode if one of the IASCs channel B is inoperative, or if loop A or loop B of the dual-detection loop fails. Loop failure is determined if one loop is shorted for more that 5 minutes, or one loop provides high-resistance (open) for more than 60 seconds.

The LEAK DET FAULT advisory message is displayed on EICAS, indicating the fault and lost redundancy.

In the degraded mode, overheat detection relies on the remaining single loop to trigger the overheat annunciation for the affected zone.

Failure Mode Operation

In the event of a dual-loop failure in a particular zone, detection capabilities for that zone are lost. The IASC automatically shuts down the affected zone to eliminate the risk of a bleed air leak. A LEAK DET FAIL caution message is displayed on EICAS for the affected zone and the associated FAIL light (L BLEED FAIL, R BLEED FAIL, L PACK FAIL or R PACK FAIL light) illuminates on the AIR panel. The crew are required to confirm the shutdown of the affected zone (s) by selecting the appropriate switches OFF. Selecting the switch to OFF resets the valve closing latch in the IASC.

BALOD Failure

If channel B of both IASCs fail, the entire BALODS is lost and considered failed and all zones are shut down. The LEAK DET FAIL caution message is displayed on EICAS.

In this event the L and R BLEED pushbutton annunciators (PBAs) are selected OFF. If in flight, the aircraft must fly at a lower altitude as air conditioning is no longer available.

Built-in Tests

The IASC performs a power-up built-in test (PBIT), continuous BIT (CBIT), and landing BIT (LBIT). Initiated BIT (IBIT) can also be performed.

PBIT is performed on every cold start on the ground. However, this cannot be done when bleed air is extracted as the system performs a calibration test of the loops and recording the resistance of the detection loops. This resistance value is stored by the IASC for use in detecting the location of bleed air leaks.

CBIT continually checks the loops for shorts and open circuits (continuity). A failure is used to switch to the degraded mode. The failure is indicated by the LEAK DET FAULT advisory message.

While the PBIT does a complete system test (shorts, continuity, etc) and calibration, the L-BIT only checks for loop continuity and internal system 5 minutes after landing. A loop failure is indicated by the LEAK DET FAULT advisory message.

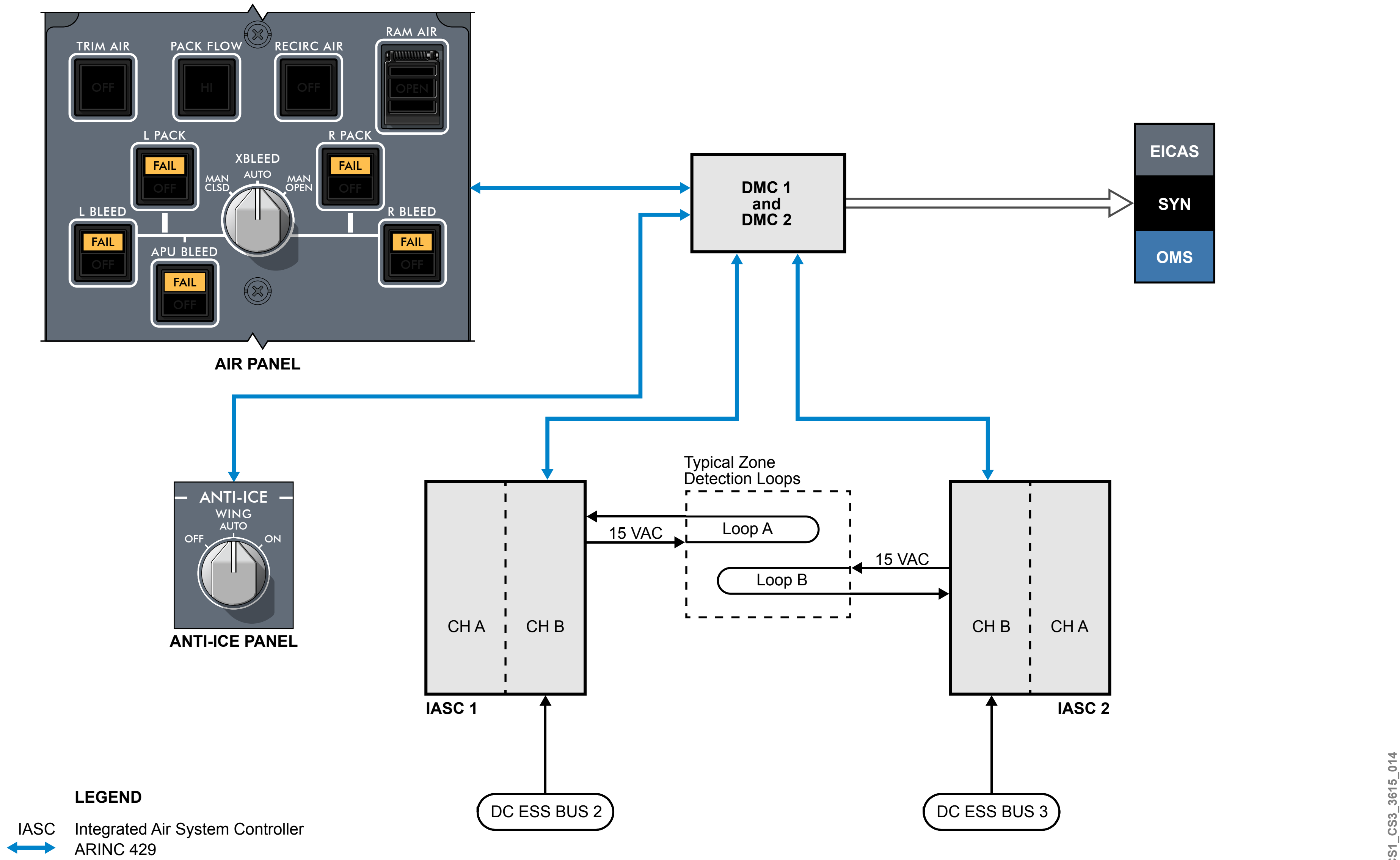


Figure 26: Bleed Air Leak and Overheat Detection System Operation

BALODS LEAK LOCALIZATION

The BALODS performs leak localization using software embedded in the IASCs. Leak location accuracy is within 50.8 cm (20 in.). The operating principle is based on the linear electrical resistance of the sensing elements inner conductor. Upon leak detection, the increasing temperature decreases the resistance between the inner and outer conductor. Software in the IASC processes this change in resistance with total loop resistance without a leak, and uses a ratio of the two resistances to indicate a leak location.

Calibration of the loop resistance is performed automatically on power-up except if power-up occurs during bleed air extraction.

BALODS leak localization by maintenance personnel is obtained through the OMS and compared to a reference table in ATA 05-51-44 of the aircraft maintenance publication (AMP).

NOTE

The bleed air extraction system MUST BE OFF during the LEAK LOCALIZATION test to prevent the radiant heat from the bleed extraction from affecting the resistance value of the loops.

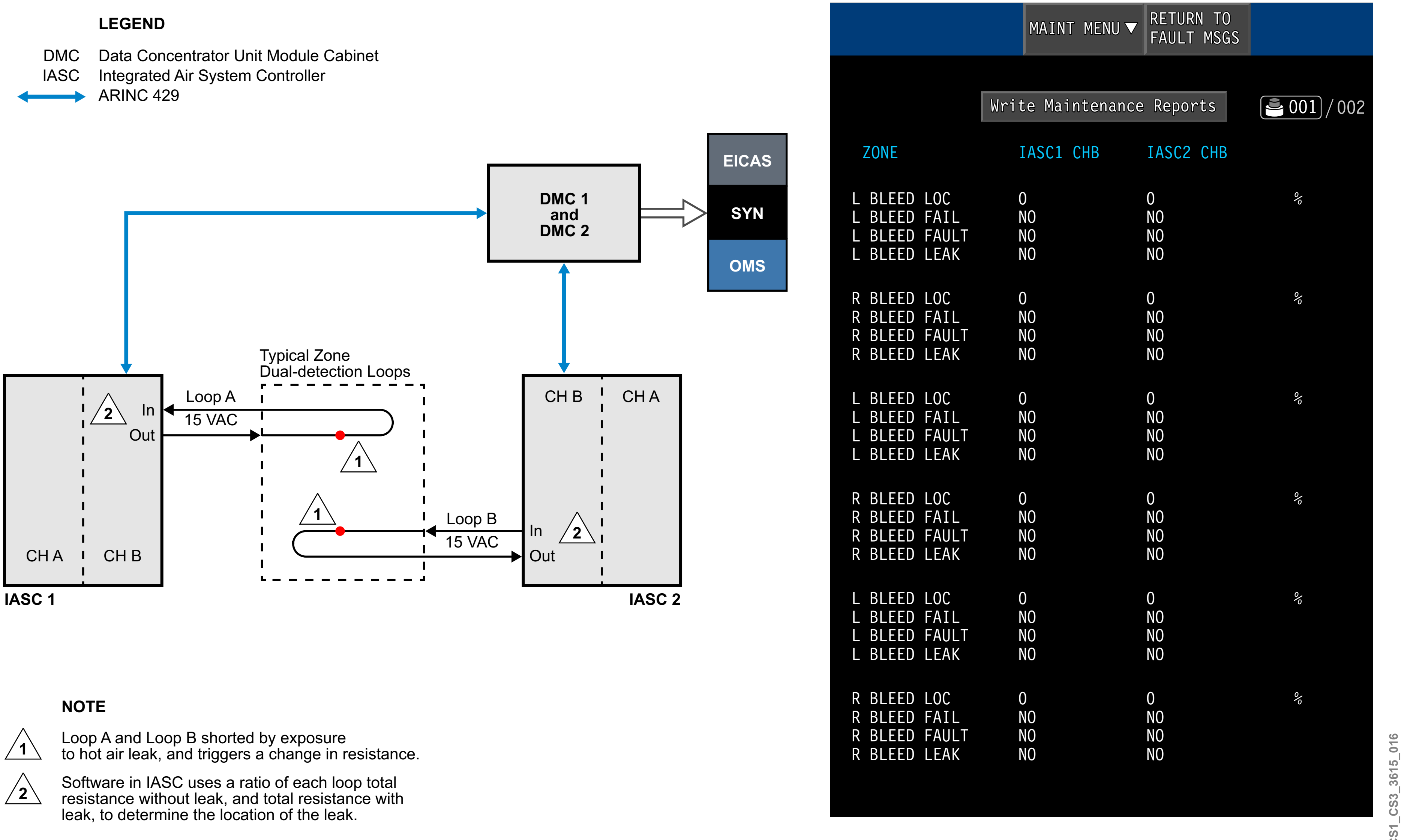


Figure 27: BALODS Leak Localization

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the bleed air leak and overheat detection system.

CAS MESSAGES

Table 5: CAUTION Messages

MESSAGE	LOGIC
L BLEED LEAK	Left bleed leak detected.
R BLEED LEAK	Right bleed leak detected.
APU BLEED LEAK	APU bleed leak detected.
L PACK LEAK	Left pack leak detected.
R PACK LEAK	Right pack leak detected.
TRIM AIR LEAK	Trim air leak detected.
WING A/ICE LEAK	Wing anti-ice leak detected.
LEAK DET FAIL	Entire leak detection system failed and either L BLEED, R BLEED, or APU BLEED PBA selected ON.

Table 6: ADVISORY Message

MESSAGE	LOGIC
LEAK DET FAULT	One leak detection loop failed. Refer to INFO messages.

Table 7: INFO Message

MESSAGE	LOGIC
36 LEAK DET FAULT - LOOP REDUND LOSS	Loss of one detection loop in the wing anti-ice zone.

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ATA 47 - Liquid Nitrogen (Fuel Tank Inerting System)

BD-500-1A10
BD-500-1A11

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LIQUID NITROGEN - CHAPTER BREAKDOWN
(FUEL TANK INERTING SYSTEM)

47-10 FUEL TANK INERTING SYSTEM GENERATION

GENERAL DESCRIPTION

The fuel tank inerting system (FTIS) includes an onboard inert gas generation system (OBIGGS). The OBIGGS uses engine bleed air, auxiliary power unit (APU) air, or air from a ground cart to supply nitrogen-enriched air (NEA) to the fuel tanks. The inert NEA floods the empty space above the fuel, thus reducing the flammability in the fuel tanks. The system is controlled automatically through the inerting control unit (ICU).

The bleed air is routed to the ozone converter, through an inlet isolation valve (IIV), to reduce the ozone concentration. The air supplied by the ozone converter passes through the heat exchanger assembly where it is cooled by the ram air.

Some bleed air bypasses the heat exchanger via the temperature control valve (TCV). The TCV modulates to maintain the heat exchanger outlet temperature within a range that provides optimal OBIGGS performance.

A ground cooling fan provides cooling air flow through the heat exchanger when the aircraft is at low airspeeds. A ground cooling fan check valve closes when the fan is operating. The check valve ensures that air is drawn over the heat exchanger during fan operation.

The cooled air is filtered by the air separation module (ASM) inlet filter. Water vapor from the supply airflow is removed through a coalescer within the filter. The water drains from the filter assembly into the aircraft belly through a drain orifice in the bottom of the filter bowl. The cooled and filtered bleed air is supplied to the air separation module (ASM), which removes the oxygen from the air and creates the NEA.

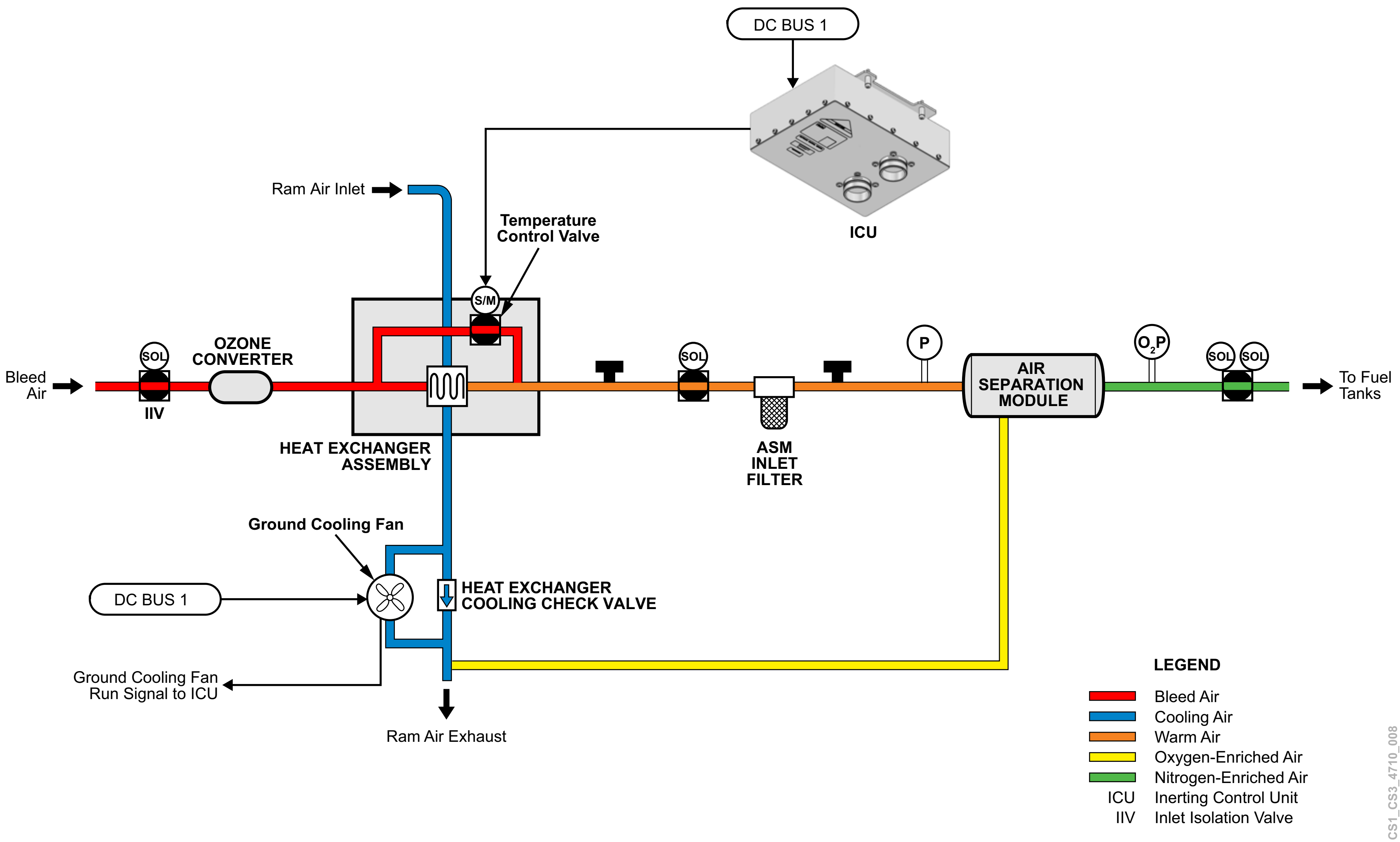


Figure 1: Fuel Tank Inerting System Generation

COMPONENT LOCATION

The following components of the FTIS are installed in the wing-to-body fairing (WTBF):

- Heat exchanger
- Ground cooling fan
- Heat exchanger cooling check valve
- Ozone converter
- Air separation module inlet filter
- Air separation module

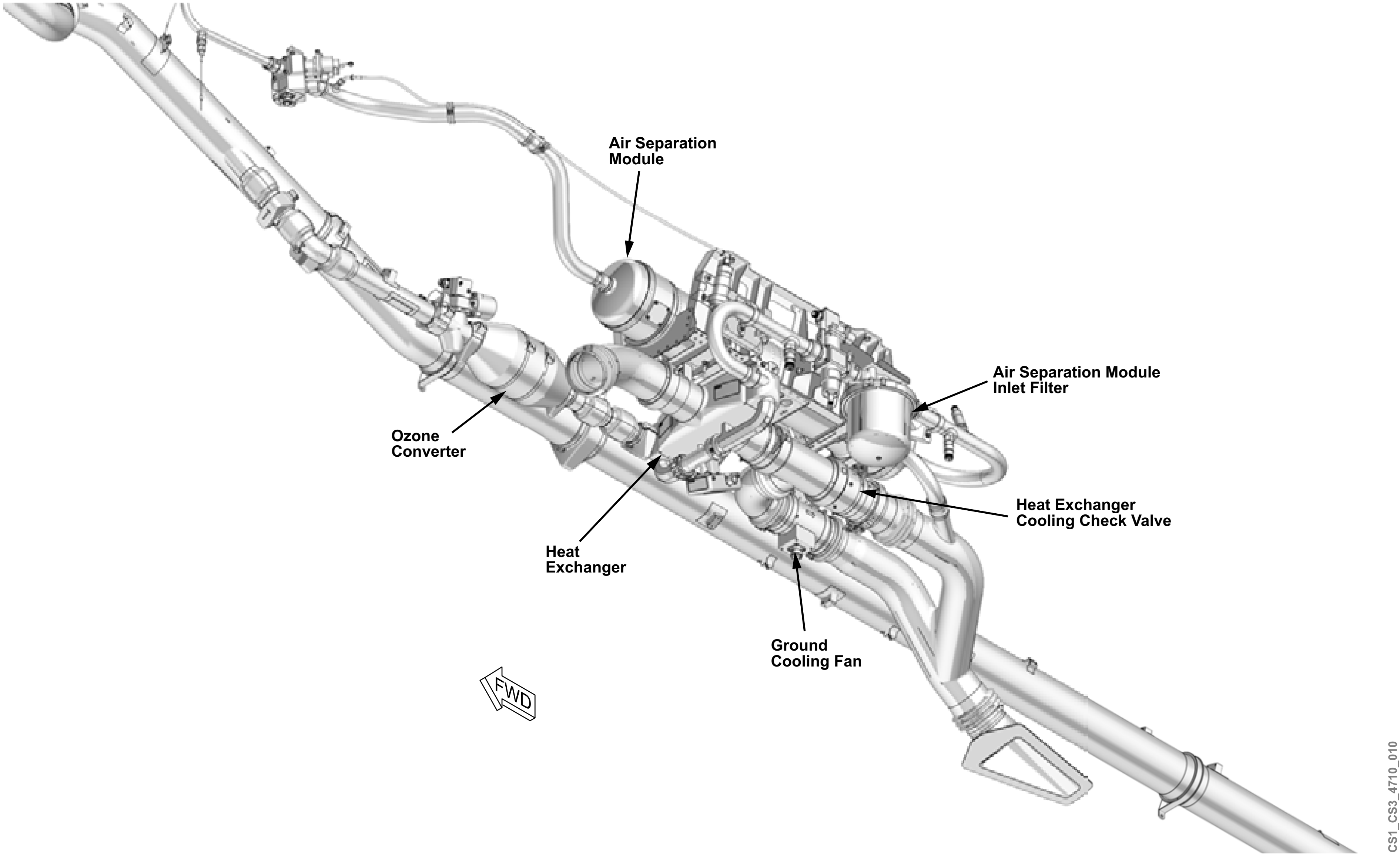


Figure 2: Fuel Tank Inerting System Generation Components

COMPONENT INFORMATION

HEAT EXCHANGER

The heat exchanger consists of a cross flow heat exchanger element, a temperature control valve (TCV), and associated bypass ducting. The heat exchanger regulates the temperature of the air entering the air separation module (ASM), by mixing hot bleed air with air cooled by the heat exchanger.

The TCV is a butterfly valve driven by a stepper motor. The ICU commands the TCV based on temperature signals from the ASM inlet temperature sensor. Limit switches on the valve provide position indication to the ICU. If the stepper motor fails, the valve remains in the last position.

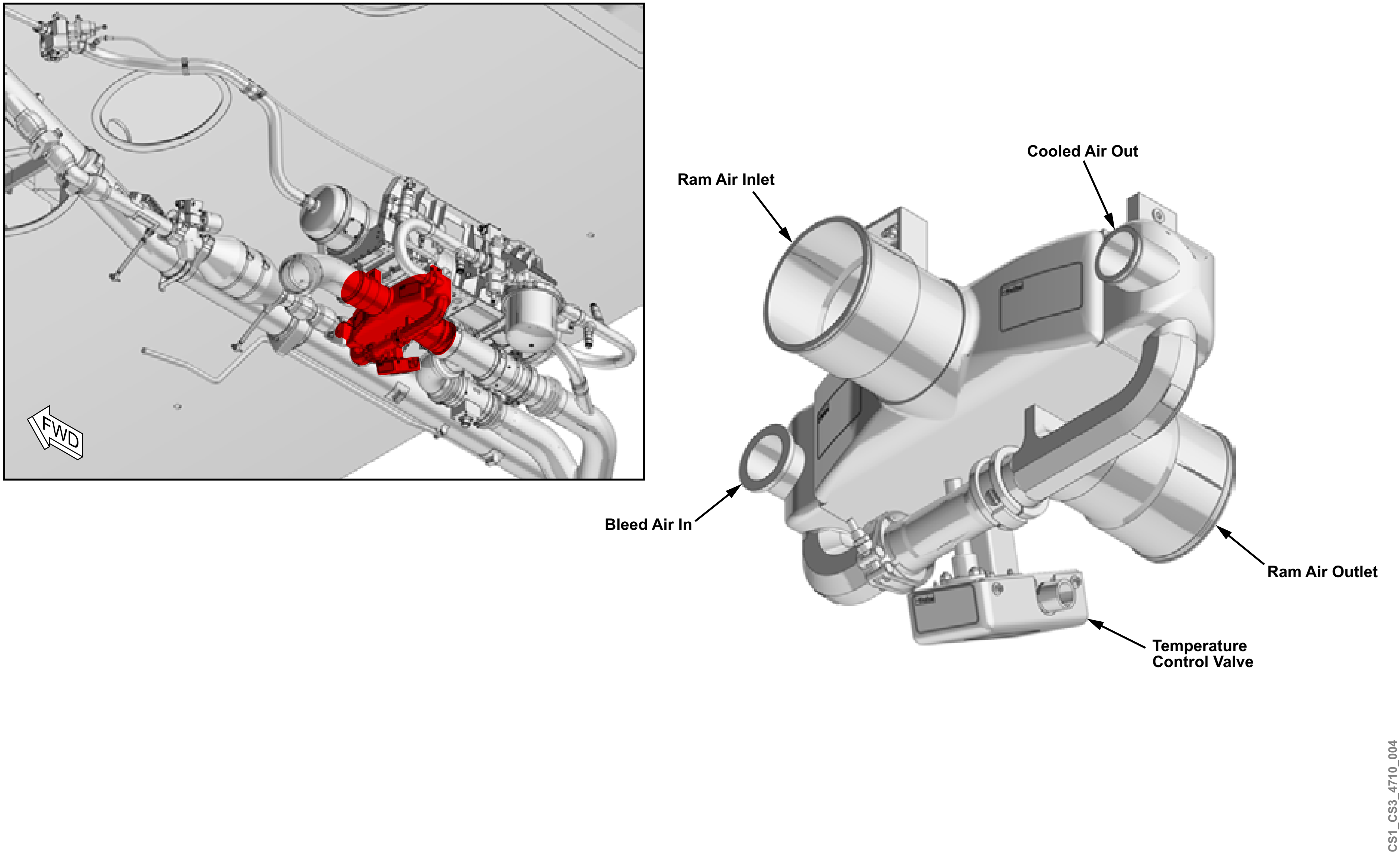


Figure 3: Heat Exchanger Assembly

AIR SEPARATION MODULE

The air separation module (ASM) is constructed of thousands of hollow fiber membranes that run the length of the module. The ASM separates the nitrogen and oxygen components of the bleed air into individual gas streams, using a process known as selective permeation. Air contains 78% nitrogen, 21% oxygen, and 1% other gases. Each gas has a characteristic permeation rate that is a function of its ability to dissolve and diffuse through a membrane. This characteristic allows fast gases, such as oxygen, to be separated from slow gases, such as nitrogen.

Bleed air enters the ASM inlet at a controlled pressure and temperature. The bleed air feeds through hollow fiber membranes. The oxygen permeates through the walls of the membrane and discharges through the oxygen-enriched air (OEA) outlet. The remaining nitrogen-enriched air (NEA) flows down the hollow fiber membranes, to the NEA outlet, and on to the fuel tanks.

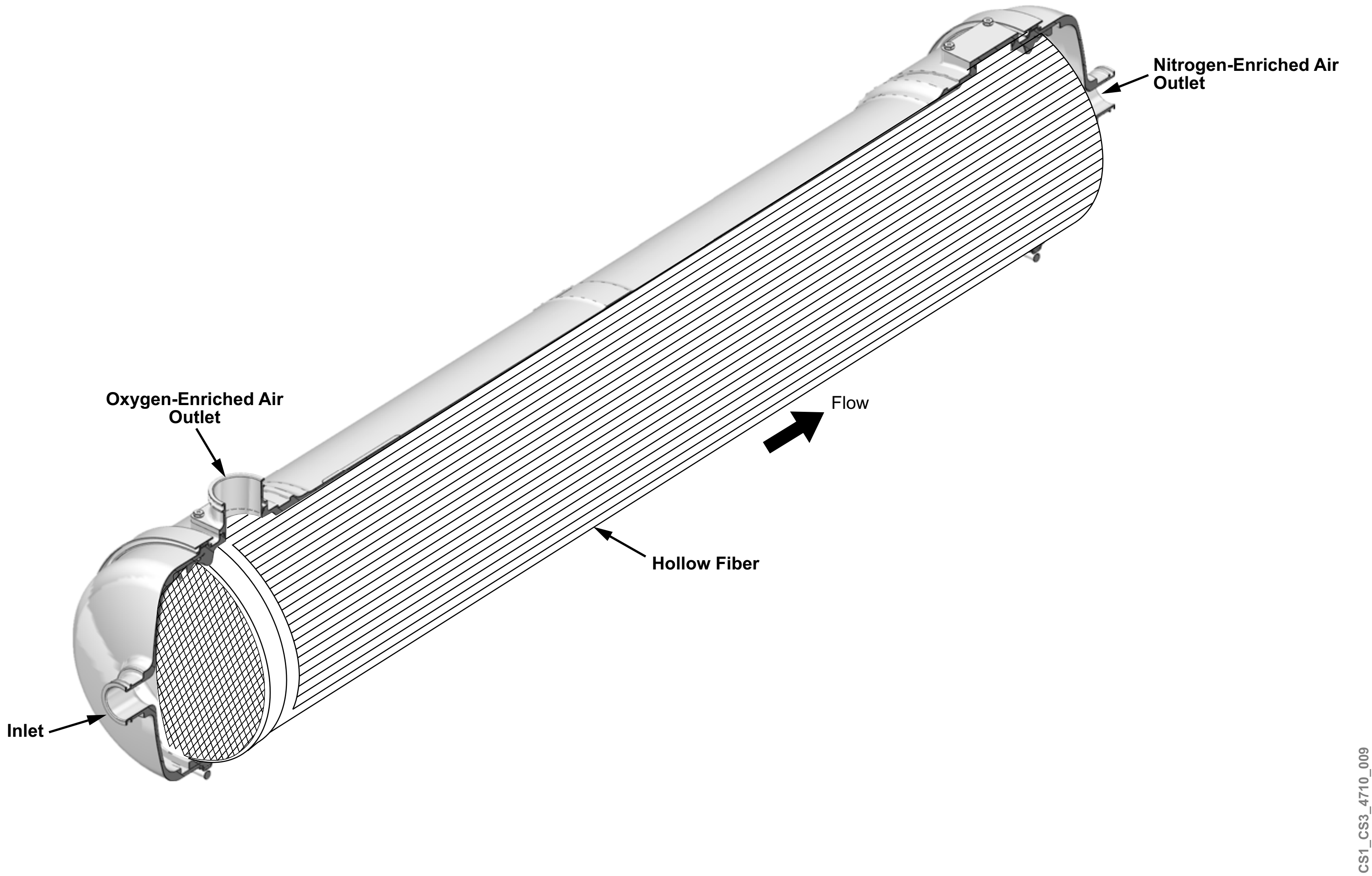


Figure 4: Air Separation Module

DETAILED DESCRIPTION

If the amount of ram cooling air is insufficient, the ground cooling fan (GCF) provides the cooling air flow to the heat exchanger assembly. The ICU uses the aircraft mach number information from the data concentrator unit module cabinet (DMC) to determine when the GCF should be running. The GCF runs anytime the mach number is less than 0.2 Mach. The GCF is powered by a solid-state power controller (SSPC), therefore a turn ON request is sent to the DMCs to activate the SSPC. At startup, the ICU energizes the GCF before opening the inlet isolation valve (IIV). This ensures that the heat exchanger is capable of cooling the initial flow of bleed air to avoid nuisance fuel tank inerting system (FTIS) shutdowns. The GCF sends a run signal to the ICU to indicate that it is operating. The fan speed and overspeed protection is managed by the fan control software. The fan automatically shuts down when overspeed is detected and a fault signal is set.

When the ICU has determined the bleed air supply is within the normal range, the IIV opens to allow bleed air flow into the FTIS. The temperature control valve (TCV) is modulated to maintain the proper air separation module (ASM) inlet temperature of 60°C (140°F). If the heat exchanger outlet temperature exceeds 60°C (185°F), the TCV is driven towards the closed position and is fully closed at 85°C (185°F).

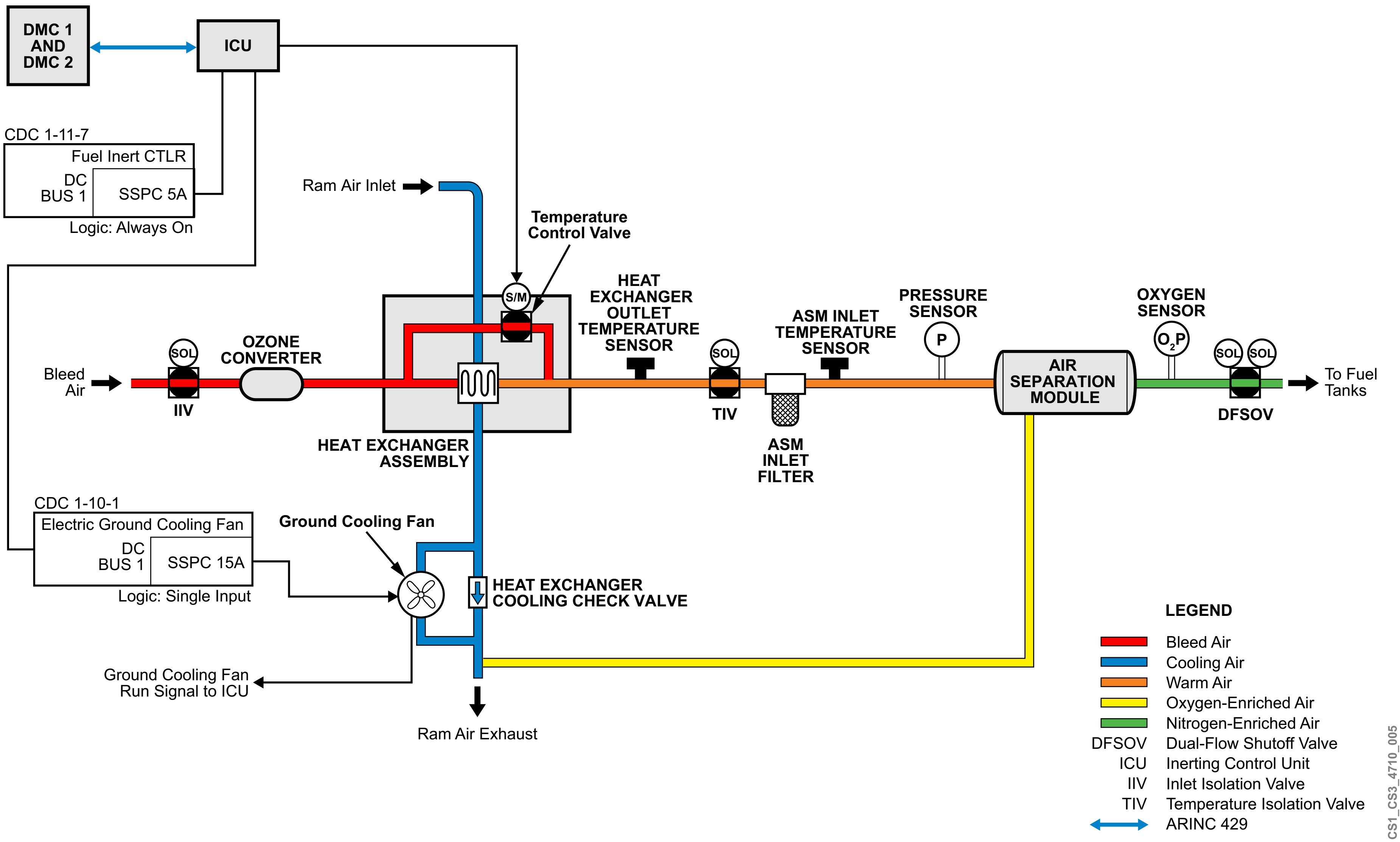


Figure 5: Fuel Tank Inerting System Generation Detailed Description

47-00 FUEL TANK INERTING SYSTEM CONTROL AND INDICATING

GENERAL DESCRIPTION

The inerting control unit (ICU) receives power from DC BUS 1. The ICU powers all of the fuel tank inerting system (FTIS) electrical components, except the ground cooling fan (GCF), which also receives power from DC BUS 1. The ICU software is field loadable.

The FTIS enters the normal operating mode when the aircraft is fully powered, bleed air is available, and the ICU power-up built-in test (BIT) is complete. Operation of the FTIS is fully automatic.

The inlet isolation valve (IIV) is the primary ON/OFF control for the FTIS. The IIV isolates the FTIS from the pneumatic system. The valve moves to the closed position for a loss of power or pneumatic pressure.

The temperature isolation valve (TIV), located at the inlet to the air separation module (ASM), is used to shut off air flow to the distribution system. This prevents hot air from contacting fuel or fuel vapor during an overtemperature condition. When an overtemperature is detected, the valve is commanded closed. The TIV is an electrically-controlled, pneumatically-operated gate valve. A microswitch provides position sensing to the ICU.

The dual-flow shutoff valve (DFSOV) supplies the distribution system with nitrogen-enriched air (NEA) at a flow rate based on altitude and vertical speed. It also prevents fuel and fuel vapor migration to the FTIS when the system is not operating.

The DFSOV outputs three flow rates. The low-flow solenoid is energized on the ground, and in climb and cruise. The medium-flow solenoid is used in approach and slow descent, and the high-flow rate is used in descent. The high-flow is created by energizing both solenoids.

System monitoring is limited to crew alerting system (CAS) messages indicating a fault has occurred and the system is shut down.

Temperature and pressure sensors are used for control, system health monitoring, and provide overtemperature and overpressure shutdown protection.

The heat exchanger outlet temperature sensor monitors the heat exchanger discharge temperature.

The air separation module (ASM) inlet temperature sensor monitors the temperature of the bleed air entering the ASM.

The oxygen sensor monitors the outlet of the ASM. The sampling port is located at the inlet to the dual-flow shutoff valve. If the oxygen level concentration is detected above the preset threshold, a crew alerting system (CAS) INFO message is displayed.

System monitoring is limited to CAS messages that indicate a fault has occurred and the system is degraded or shut down.

The FTIS has five operating modes:

- Normal operating mode - the FTIS provides NEA when bleed air is available
- Health monitoring mode - the ICU performs a pressure and oxygen concentration data check of the NEA once during cruise
- Inhibit mode - power is removed from all FTIS components. The system recovers to the normal operating mode when the inhibit condition no longer exists
- Latched shutdown mode - an overtemperature, overpressure fault, or sensor fault is detected and the system shuts down
- Maintenance mode - provides the capability to test and troubleshoot FTIS faults to the line replaceable unit (LRU) level

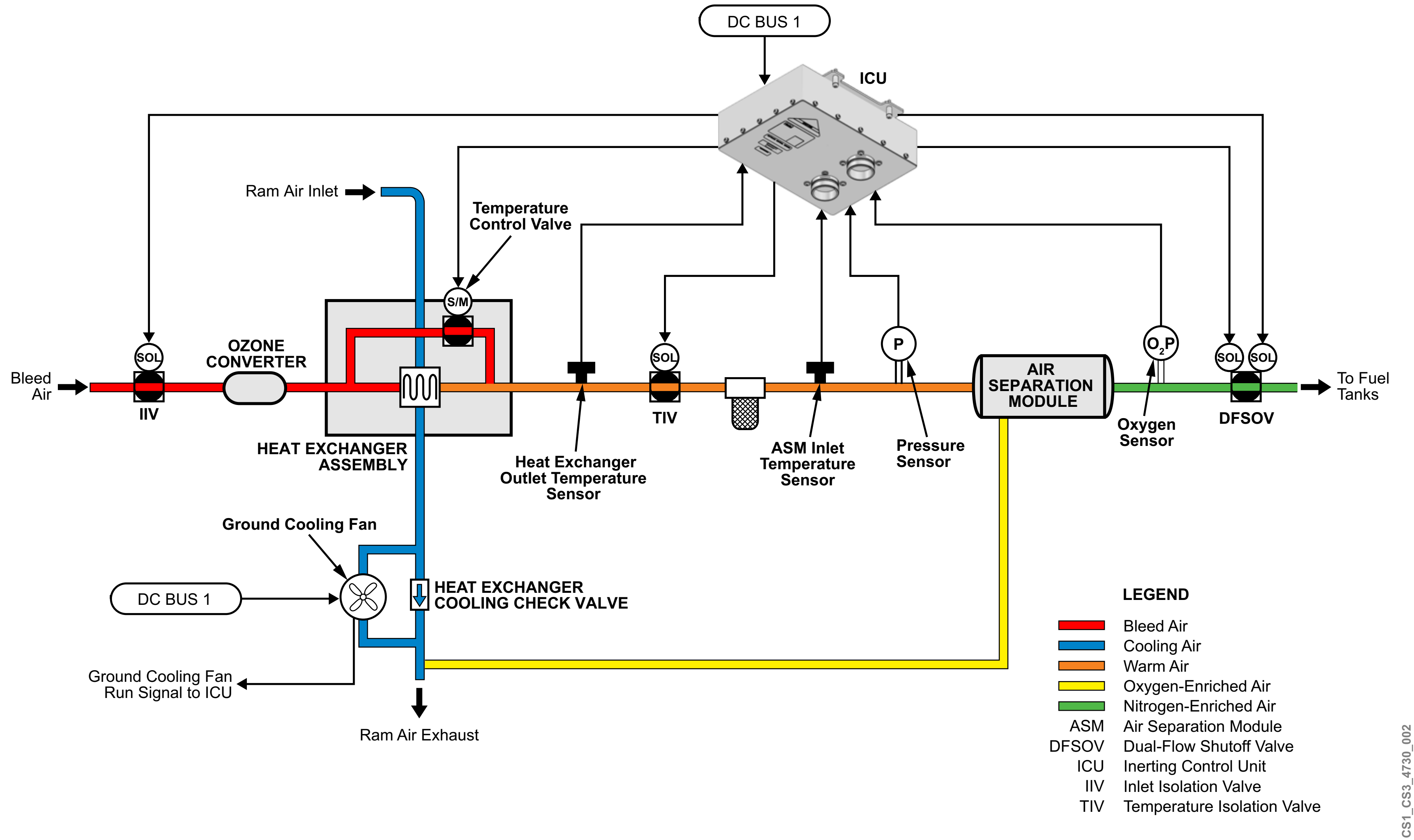


Figure 6: Fuel Tank Inerting System Control and Indicating

COMPONENT LOCATION

The following fuel tank inerting control and indicating system components are located in the wing-to-body fairing (WTBF):

- Inlet isolation valve
- Temperature isolation valve
- Dual-flow shutoff valve
- Heat exchanger outlet temperature sensor
- Pressure sensor
- Air separation module (ASM) inlet temperature sensor
- Oxygen sensor
- Inerting control unit (refer to figure 8)

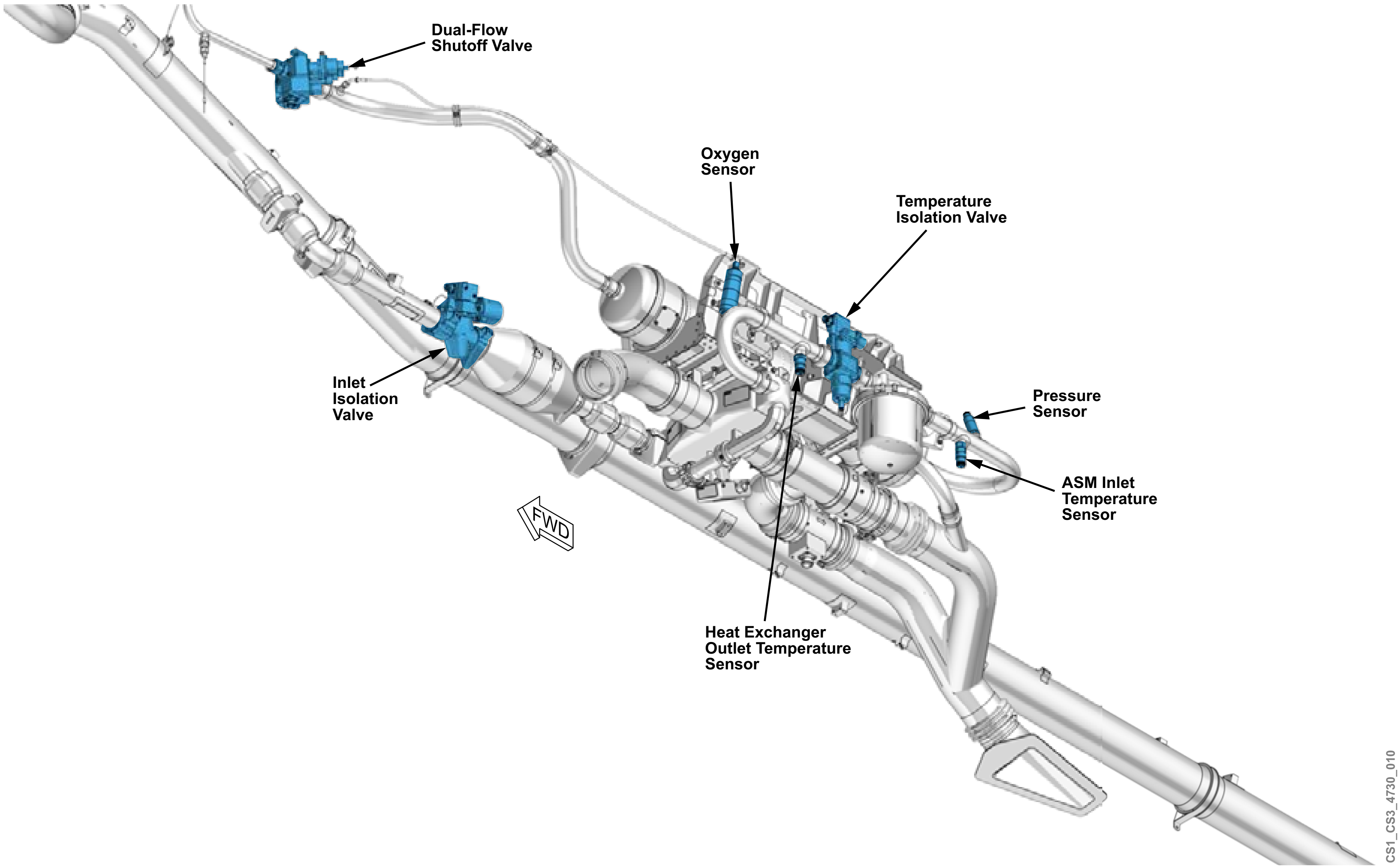


Figure 7: Fuel Tank Inerting System Control Components

INERTING CONTROL UNIT

The inerting control unit (ICU) is located in the mid equipment bay.

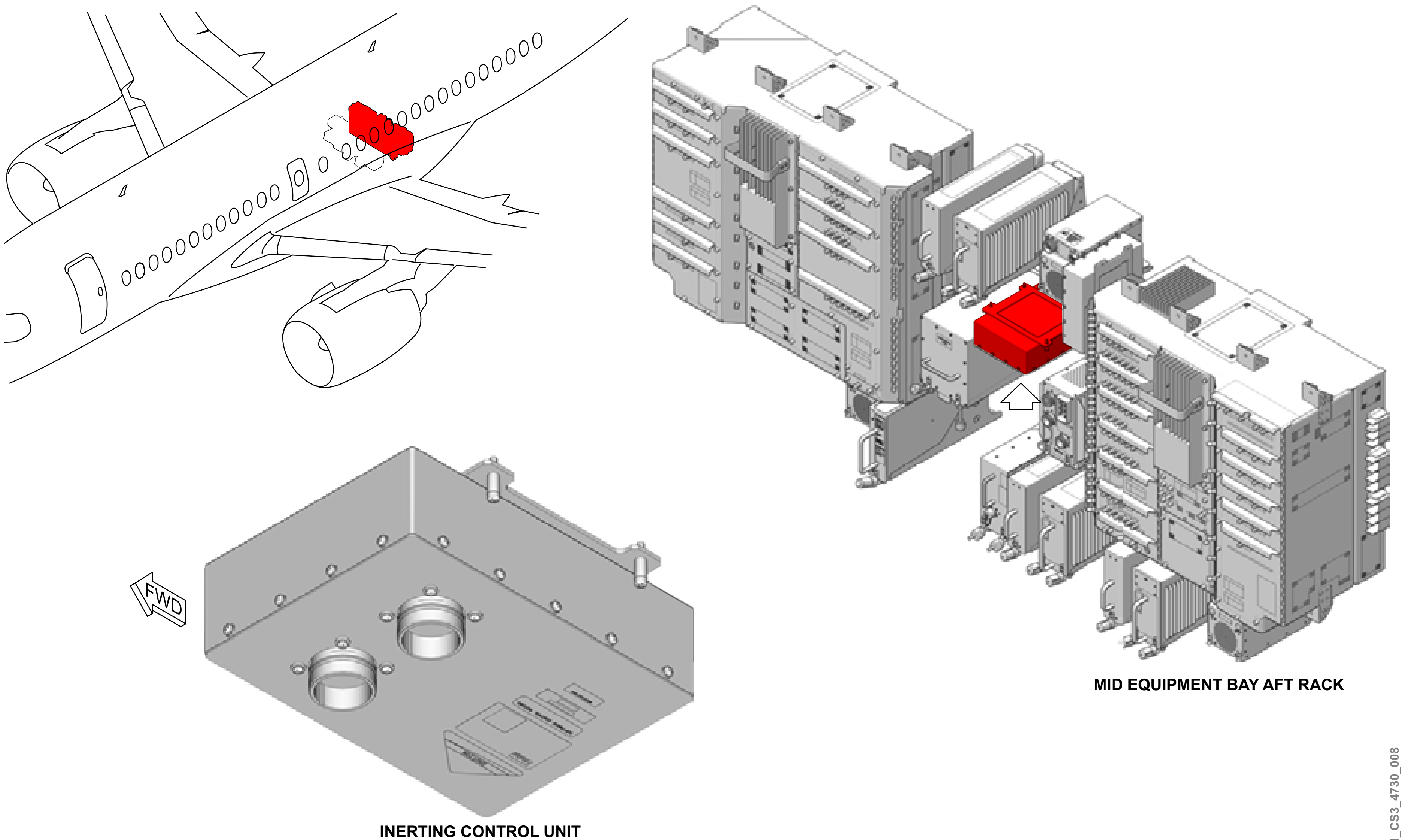


Figure 8: Inerting Control Unit

COMPONENT INFORMATION

INLET ISOLATION VALVE

The inlet isolation valve (IIV) is a solenoid-controlled, pneumatically-operated butterfly valve. The valve has a position indicator that must not be used to move the valve.

TEMPERATURE ISOLATION VALVE

The temperature isolation valve (TIV) is an electrically-controlled, pneumatically-operated gate valve. A microswitch provides position sensing to the ICU. The TIV also has a position indicator.

DUAL-FLOW SHUTOFF VALVE

The DFSOV is a dual-poppet, 28 VDC electrically-controlled, pneumatically-operated valve. The poppets are spring-loaded closed when the solenoids are de-energized. Visual indicators show when the poppets are in the open or closed position. Microswitches provide position information to the ICU when the poppets are closed.

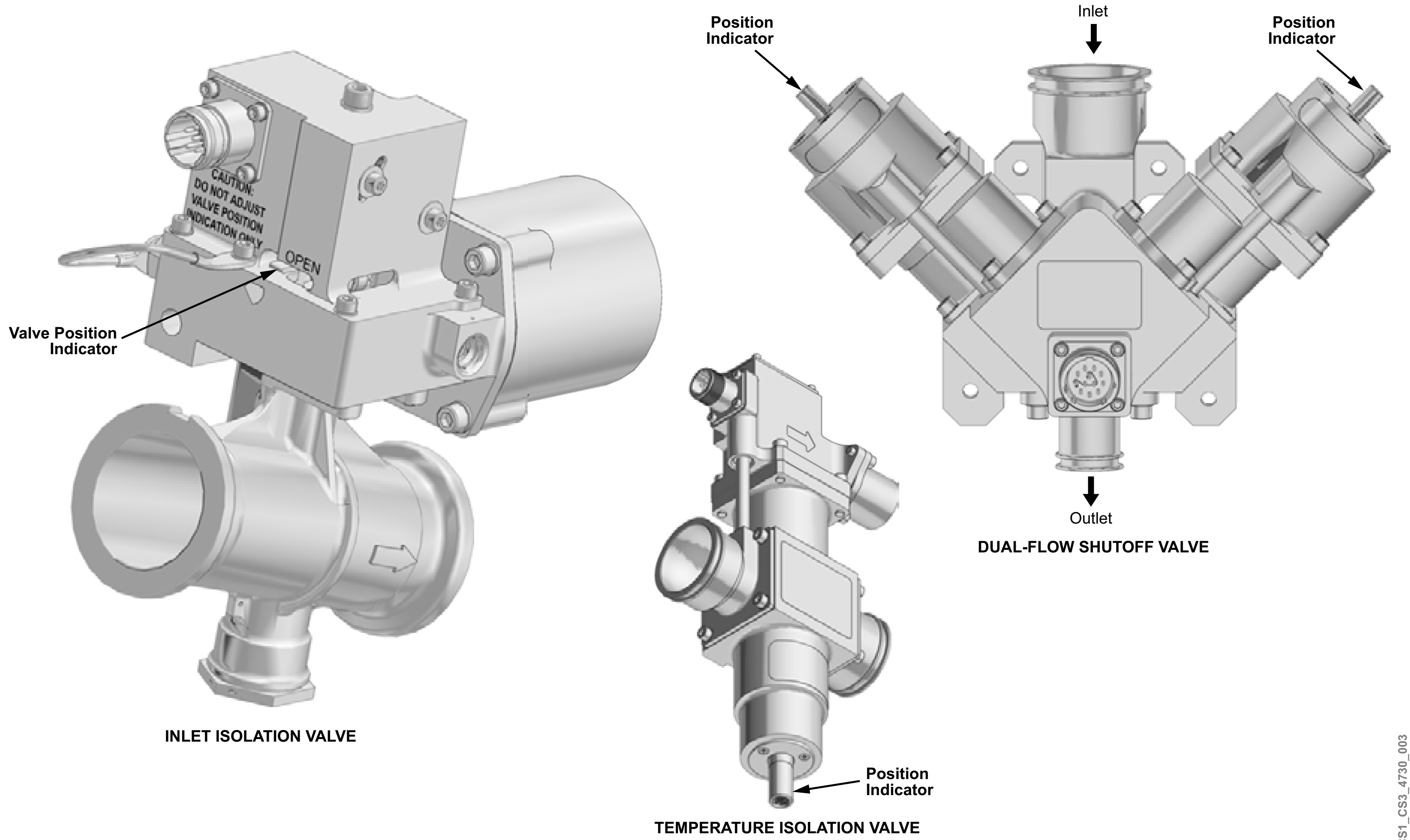


Figure 9: Fuel Tank Inerting System Control Components

DETAILED DESCRIPTION

FUEL TANK INERTING SYSTEM INTERFACE

Communication with the aircraft systems is through the data concentrator unit module cabinet (DMC). The ICU communicates with DMC 1. If DMC 1 becomes unavailable, the ICU will switch to DMC 2. The ICU transmits system and component health information, and receives flight phase and conditions information that is used by the ICU to determine the necessary nitrogen-enriched air (NEA) flow.

The ARINC 429 inputs received via the DMC are from:

- Air data system for environmental data and Mach number
- Fuel quantity computer for refuel status
- Integrated air system controller (IASC) for bleed pressure
- Onboard maintenance system (OMS) for testing

The ICU remains in inhibit mode while performing power-up BIT (PBIT). After completing a successful PBIT, the ICU begins energizing valves in a controlled startup sequence. As each valve is energized, a position sensor check is performed and other safety related parameters are checked.

The ICU digital channel is the primary monitor of FTIS temperature. Two independent temperature monitoring analog backup channels are provided for shutdown of the FTIS in case of a digital channel failure.

The heat exchanger outlet temperature sensor monitors the heat exchanger discharge temperature. The sensor is a dual-element type, One element reports to analog channel 1 of the inerting control unit (ICU), and the other element reports to the digital channel.

The ASM inlet temperature sensor monitors the air temperature entering the ASM. This temperature is used by the inerting control unit (ICU) for temperature control and health monitoring. It is a dual-element type. One element reports to analog channel 2 of the ICU, and the other element reports to the digital channel.

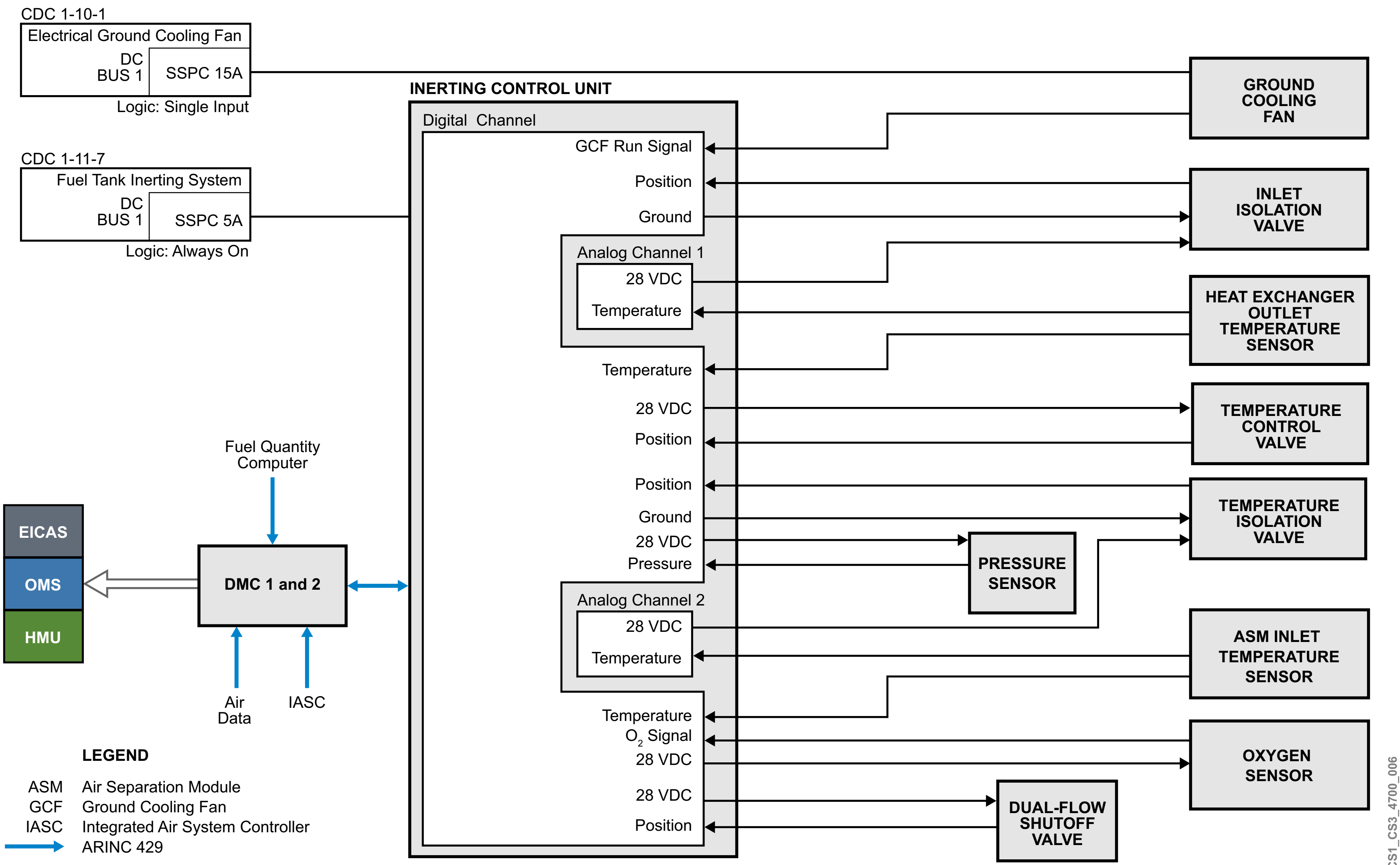
The digital channel controls the ground for the IIV and the TIV. Each analog circuit controls the 28 VDC for its respective valve. Analog channel 1 controls the power to the IIV, and analog channel 2 controls power to the TIV.

In normal operation, the digital channel will remove the grounds from the IIV, TIV, and DFSOV when an overtemperature is sensed. If the digital channel fails, and the heat exchanger outlet temperature sensor senses an overheat, analog channel 1 removes power from the IIV. If the ASM inlet temperature sensor detects an overheat, analog channel 2 removes power from the TIV.

The temperature control valve (TCV) position is based on both the heat exchanger outlet temperature sensor, and the ASM inlet temperature sensor. The ASM inlet sensor is the primary driver of temperature, but the heat exchanger outlet sensor provides a safety function and commands the valve to full-closed at 85°C (185°F).

The pressure sensor at the air separation module (ASM) inlet is used for health monitoring and detection of overpressure conditions that could damage system components.

The oxygen sensor provides an indication of the oxygen concentration and absolute pressure of the nitrogen-enriched air (NEA) delivered by the inerting system. The output from the oxygen sensor is used to monitor the performance of the ASM.



LEGEND

ASM Air Separation Module
GCF Ground Cooling Fan
IASC Integrated Air System Controller
ARINC 429

Figure 10: Fuel Tank Inerting System Interface

INERTING CONTROL UNIT

The inerting control unit (ICU) has a digital channel that is the primary monitor of FTIS temperatures. Two independent temperature monitoring analog channels shut down the FTIS in case of failure of the digital channel. When the digital or analog monitor circuits detect a failure, the controller enters a latched shutdown mode.

The ICU provides automatic control of the FTIS. Primary functions include:

- System startup
- System control
- Temperature control
- Overtemperature protection
- Overpressure protection
- Health monitoring
- System shutdown
- System inhibits

Secondary functions are part of a maintenance mode, which includes activating and monitoring each FTIS component as commanded by the aircraft's maintenance computer.

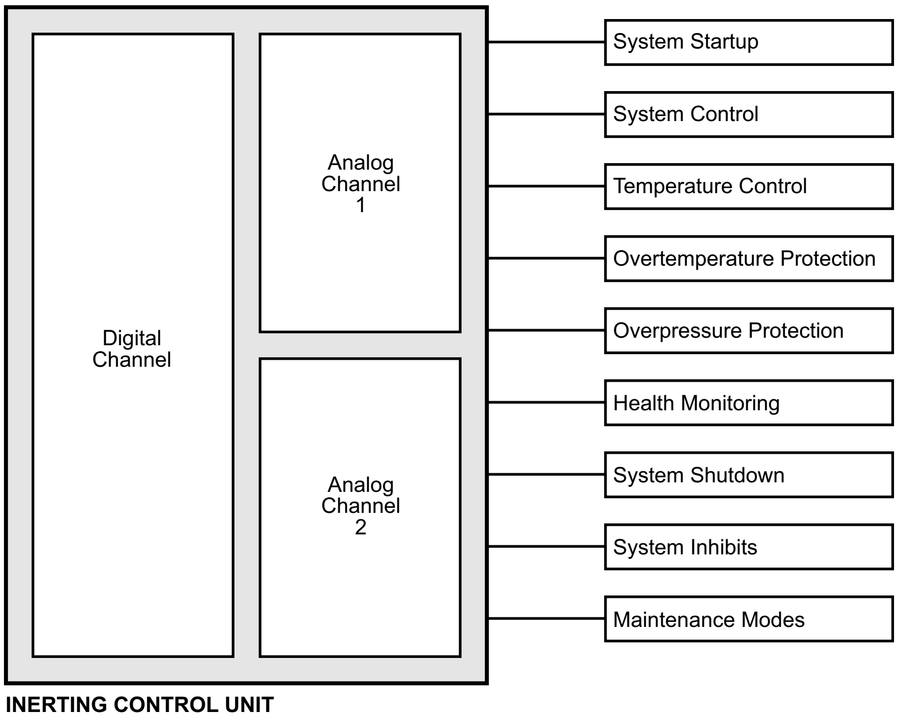


Figure 11: Inerting Control Unit

NORMAL OPERATING MODE

The normal operating mode consists of the following submodes:

- Starting sequence
- Flow control
- Overpressure protection
- Temperature control and overtemperature protection

When initially energized with 28 VDC, the ICU remains in inhibit mode while performing power-up BIT (PBIT). After completing a successful PBIT, the FTIS ICU begins energizing valves in a controlled startup sequence. As each valve is energized, a position sensor check is performed and other safety related parameters are checked.

If the amount of ram cooling air is insufficient, the ground cooling fan (GCF) provides the cooling air flow to the heat exchanger assembly. The GCF runs if the aircraft mach number is less than 0.2 M.

When the ICU is satisfied with the initialization checks, and has determined the bleed air supply is within the normal range, the IIV opens to allow bleed air flow into the FTIS. The temperature control valve (TCV) is modulated to maintain the proper air separation module (ASM) inlet temperature and the FTIS is considered powered-up and operational.

Starting Sequence

The ICU controls the system startup as follows:

1. Perform a power-up BIT (PBIT).
2. Begin temperature monitoring.
3. Command the GCF to run if the aircraft is on ground, and check for GCF feedback.
4. Cycle the TCV open and closed. Confirm valve closed.
5. Open the inlet isolation valve (IIV) and confirm valve position.
6. Check that the heat exchanger outlet temperature sensor does not indicate overtemperature.
7. Open the TIV.
8. Check that ASM inlet temperature sensor does not indicate overtemperature.
9. Open mid flow solenoid of the dual-flow shutoff valve (DFSOV) for 30 second purge and then open low flow solenoid of the DFSOV.
10. If bleed air is not available, enter the inhibit mode.
11. If bleed air is available, enter the operating mode and begin temperature and flow control.

NOTE

The ICU uses a bleed pressure signal from the IASC to determine if there is enough bleed air available to open the IIV and start the fuel tank inerting process.The PBIT test runs even if bleed air is not available. Since the IIV, TIV, and DFSOV do not open if bleed air is not available, they are only monitored for the closed position. The bleed pressure signal is also used to determine if the IIV has failed.

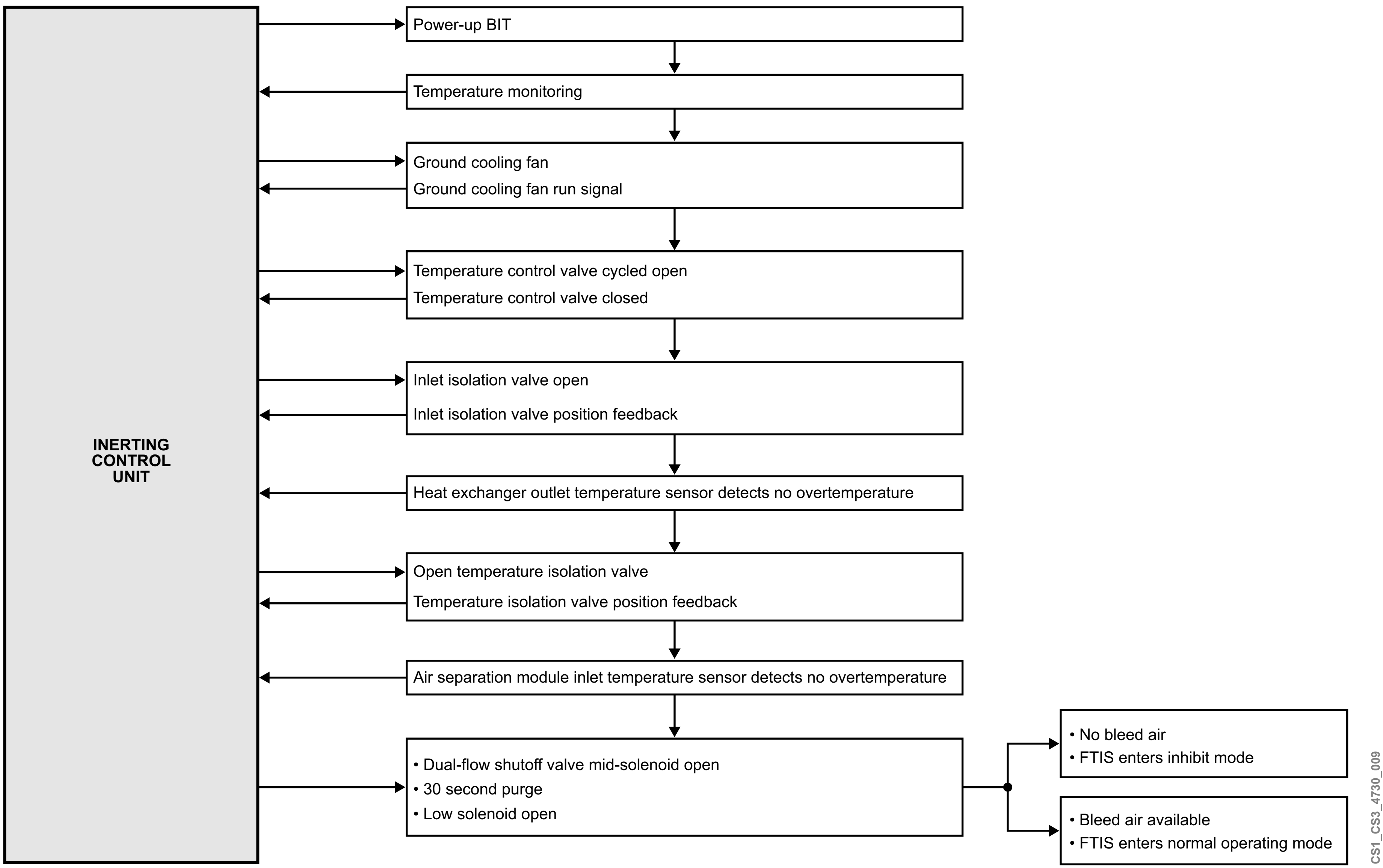


Figure 12: Fuel Tank Inerting System Start Sequence

Flow Control

The ICU energizes the appropriate dual-flow shutoff valve (DFSOV) solenoids based on aircraft altitude and vertical speed.

When low-flow is required, only the low solenoid is energized. For mid-flow, the mid solenoid is energized. High-flow is achieved with both solenoids energized.

During ground, climb, and cruise operations, the fuel tank inerting system (FTIS) operates in low-flow mode. During this time, the NEA reduces the fuel tank oxygen concentration and fills the fuel tank space, storing NEA to minimize the amount required to be generated during descent. As the aircraft climbs, the NEA will be venting overboard in order to maintain a pressure slightly above ambient in the tank.

During descent, a vent inflow condition occurs to maintain ambient pressure. The FTIS enters the high-flow mode to ensure that there is adequate mixing of the NEA and vent air.

Overpressure Protection

The ICU provides overpressure protection of the FTIS components and ducting. If an overpressure condition of 60 psi is detected at the inlet to the air separation module ASM, the FTIS is automatically shut down by closing the inlet isolation valve (IIV) and DFSOV. The temperature inlet valve (TIV) is held open for a short period of time to allow any trapped pressure to bleed off through the filter drain and ASM oxygen-enriched air (OEA) port and then closed.

Overpressure protection of the fuel tank is provided by the fixed orifices in the DFSOV.

Temperature Control and Overtemperature Protection

The ICU maintains the ASM inlet temperature at 60°C (140°F). The ICU uses ASM inlet temperature sensor and HE outlet temperature sensor information to modulate the temperature control valve (TCV).

An overtemperature sensed by the digital channel closes all of the valves. An overtemperature sensed by either of the analog channels results in its respective valve closing.

The controller enters the latched shutdown mode when either the digital or analog monitor circuits detect an overtemperature.

The temperature shutdown thresholds are as follows:

Digital:

- 85°C (185°F) for 2 minutes
- 88°C (190°F)

Analog:

- 90°C (194°F)

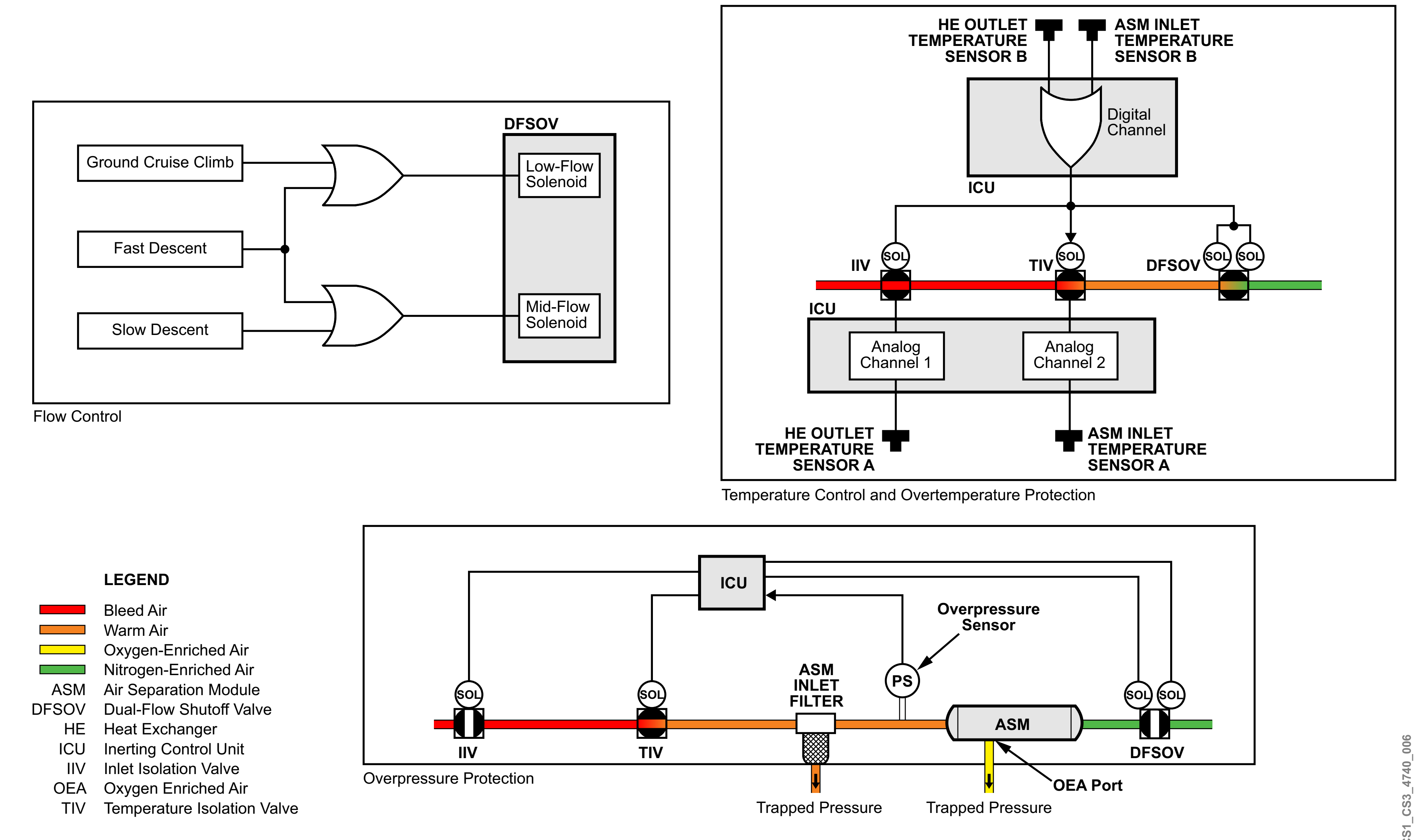


Figure 13: Flow Control, Overtemperature, and Overpressure Protection

HEALTH MONITORING MODE

Although oxygen sampling is done continuously, the FTIS enters the health monitoring mode one time per flight. This occurs during the operating mode when the aircraft is in the cruise flight phase and above 18,000 ft. When the health monitoring interval is initiated, the oxygen sensor will be turned on to measure the NEA oxygen concentration at the ASM outlet for a duration of 1 minute.

A small gas sample, drawn from the outlet of the ASM, is fed to the oxygen sensor. The pressure and oxygen concentration is sent to the ICU, where it is used in conjunction with other FTIS and aircraft parameters to determine system health. An INFO message is displayed on the engine indication and crew alerting system (EICAS) when the oxygen concentration is out of limits over a period of several flights.

INHIBIT MODE

In the inhibit mode, electrical power is removed from the system. System operation is stopped, but returns to the normal operating mode after the inhibit conditions cease.

The FTIS enters the inhibit mode upon receiving a request from aircraft systems. All FTIS components de-energize in the inhibit mode, but return to an operational state when a change occurs in the status of another system. The following conditions cause an inhibit:

- Fuel quantity computer (FQC) in refuel mode
- FQC in defuel mode
- Main engine start
- Inflight air-assisted engine start
- One engine inoperative inflight only
- Single environmental control system bleed with wing anti-ice on
- Steep approach with wing anti-ice on
- Loss of bleed air

The ICU exits inhibit mode and starts normal FTIS operation when the conditions causing the inhibit mode are no longer present for at least 3 seconds.

LATCHED SHUTDOWN MODE

In the latched shutdown mode, all FTIS components de-energize with no automatic recovery. The ICU enters into latched shutdown mode if one of the following events occurs:

- Overtemperature
- Overpressure
- Loss of ARINC 429 BUS data
- Any FTIS sensor defective, disconnected, or out of range
- Valve failure

The FTIS is reset through the onboard maintenance system (OMS). The fault is stored in the ICU non-volatile memory (NVM). The FTIS reactivates after a successful built-in test (BIT) and power-up cycle.

MAINTENANCE MODE

The maintenance mode is accessed through the OMS. To aid in troubleshooting, the FTIS faults occurring during normal operation are recorded in NVM within the ICU. These NVM faults include the values of related parameters at the time of the fault, and are downloaded through the OMS while the aircraft is on the ground.

The initiated BIT provides the capability to test the electrical line replaceable units (LRUs).

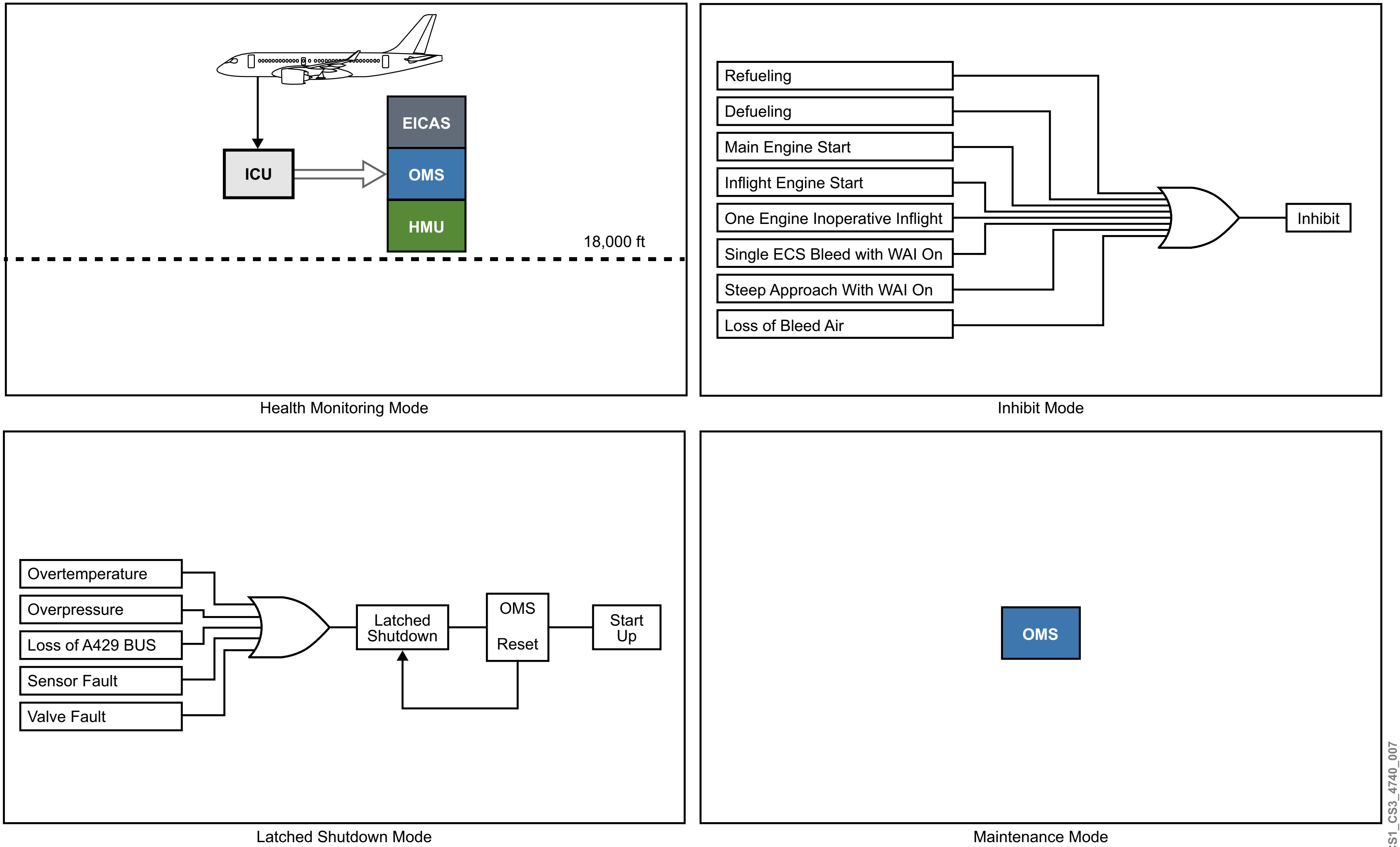


Figure 14: Fuel Tank Inerting System Modes

MONITORING AND TEST

The following page provides the crew alerting system (CAS) and INFO messages for the control and monitoring system.

CAS MESSAGES

Table 1: ADVISORY Message

MESSAGE	LOGIC
FUEL INERTING FAULT	The fuel tank inerting system fault exists. See INFO message.

Table 2: INFO Messages

MESSAGE	LOGIC
47 FUEL INERTING FAULT-FUEL INERTING SHUTDOWN	The fuel tank inerting system is prevented from operating (latched shutdown).
47 FUEL INERTING FAULT-FUEL INERTING DEGRADED	The FTIS oxygen sensor detects abnormally high oxygen percentage in the nitrogen-enriched air stream or oxygen sensor fault.
47 FUEL INERTING FAULT-FUEL INERTING REDUND LOSS	A loss of redundancy in the ICU overtemperature analog backup circuit has occurred, but has not resulted in the FTIS becoming inoperative.
47 FUEL INERTING FAULT-FUEL INERTING INOP	For inability to confirm IIV or TIV position or loss of FTIS communication or controller fault.
47 FUEL INERTING FAULT - INLET ISOL VLV INOP	The fuel tank inerting system is prevented from operating (latched shutdown) until successfully completing a power-up BIT and an IBIT is successfully completed because a fault has been detected with the performance of the FTIS inlet isolation valve.

Table 2: INFO Messages

MESSAGE	LOGIC
47 FUEL INERTING FAULT - TEMP ISOL VLV INOP	The fuel tank inerting system is prevented from operating (latched shutdown) until successfully completing a power-up BIT and an IBIT is successfully completed because a fault has been detected with the performance of the FTIS temperature isolation valve.
47 FUEL INERTING FAULT - DUAL FLOW SOV INOP	The fuel tank inerting system is prevented from operating (latched shutdown) until successfully completing a power-up BIT and an IBIT is successfully completed because a fault has been detected with the performance of the FTIS dual flow shutoff valve.

PRACTICAL ASPECTS

INLET ISOLATION VALVE LOCKOUT (TBC)

The fuel tank inerting system (FTIS) can be deactivated by locking out the IIV. Access to the IIV is through a small panel in the wing-to-body fairing (WTBF).

The valve is locked out by removing the deactivation plug from the stowed position and threading it into the deactivation port. The plug is attached to the valve by a lanyard.

Additional procedures require the FUEL INERT CTLR circuit breaker to be locked out.

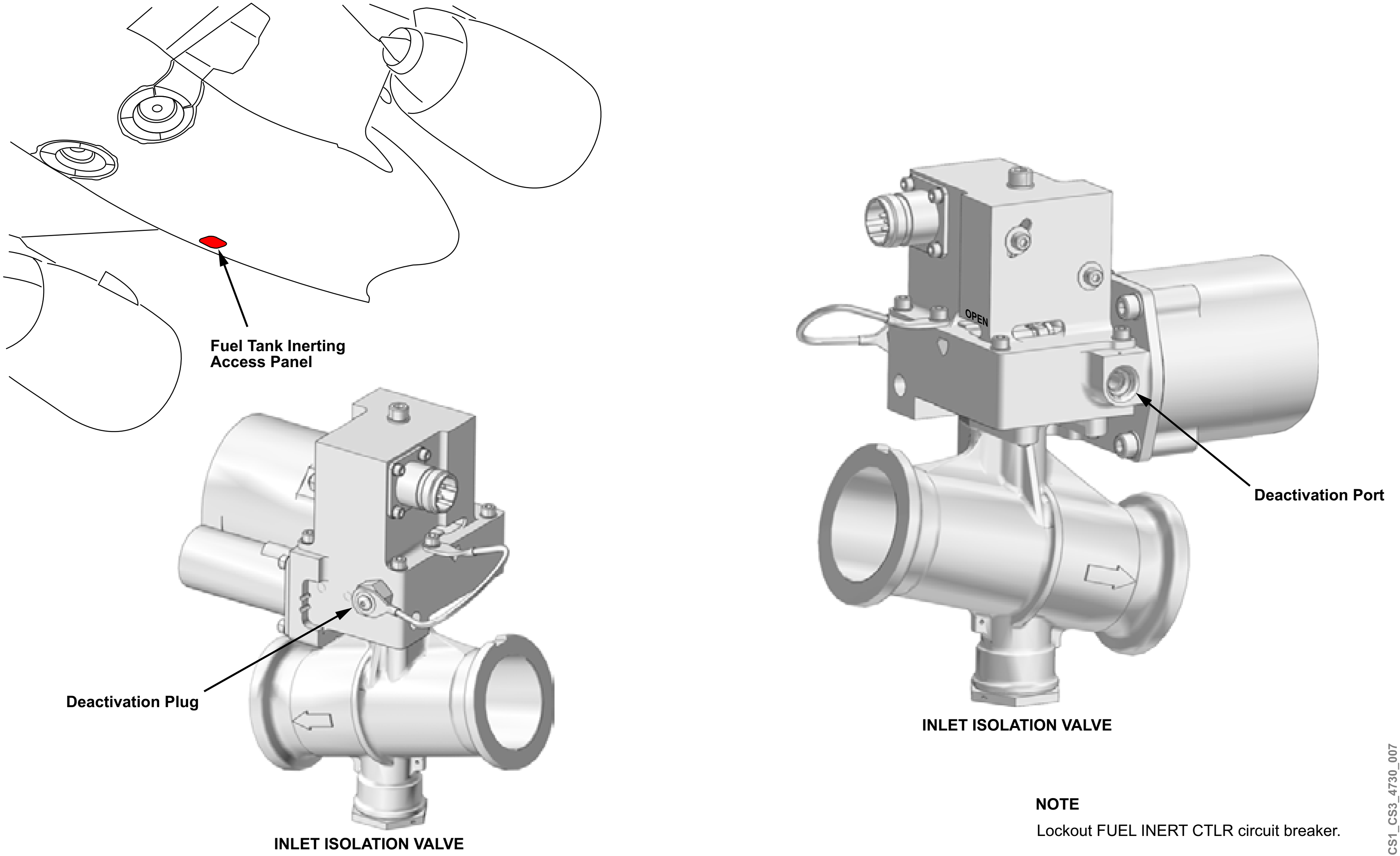


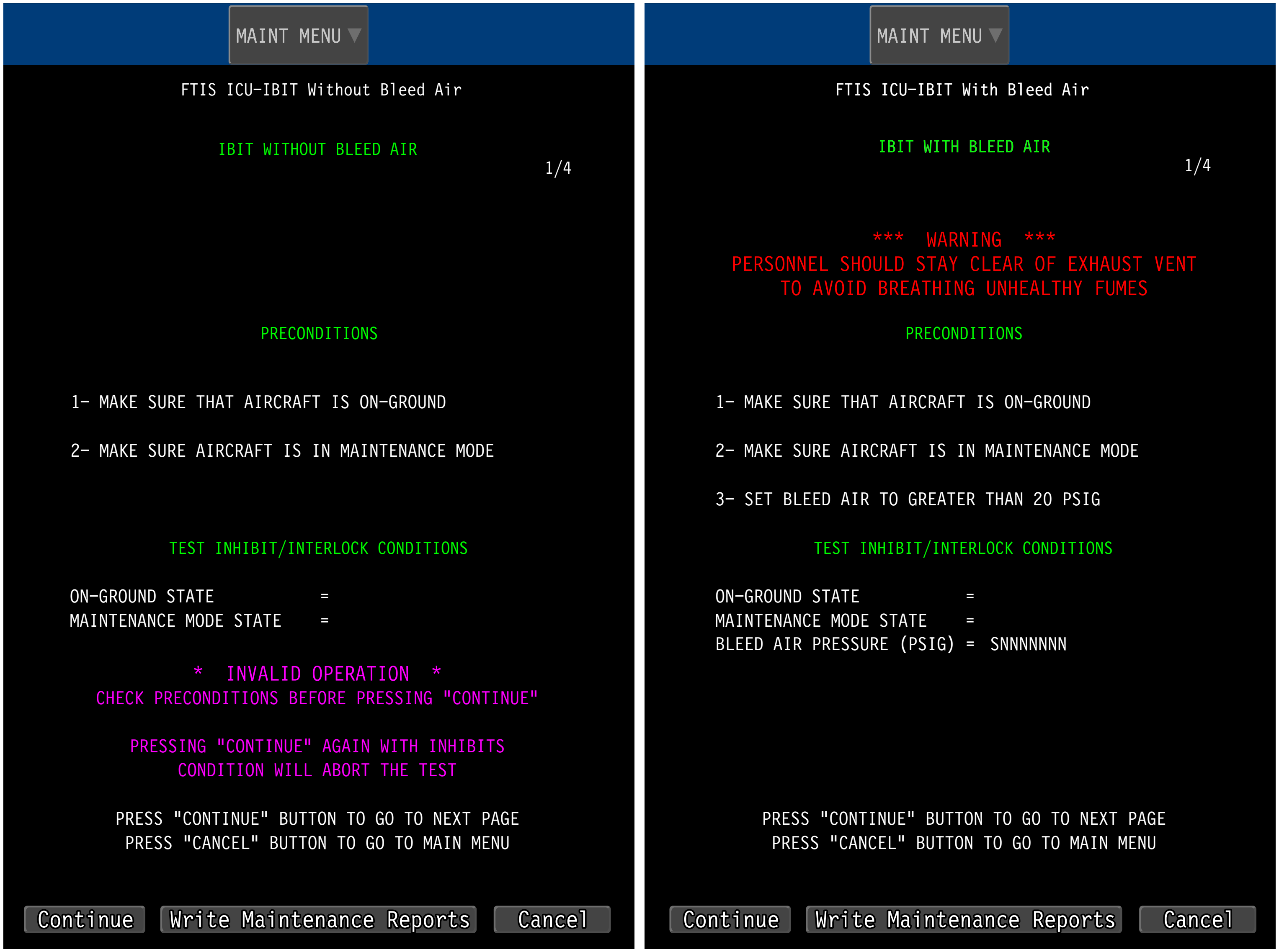
Figure 15: Inlet Isolation Valve Lockout

ONBOARD MAINTENANCE SYSTEM

The onboard maintenance system initiated built-in test (IBIT) can be carried out with or without bleed air.

The IBIT without bleed air checks all electrically actuated LRUs, however, the IIV, TIV, and DFSOV are only checked for the closed position as these valves are pneumatically operated.

To carry out a full test of the IIV, TIV, and DFSOV, the IBIT with bleed air is used.



CS1_CS3_4740_009

Figure 16: Onboard Maintenance System

47-20 FUEL TANK INERTING SYSTEM DISTRIBUTION

GENERAL DESCRIPTION

The nitrogen-enriched air (NEA) distribution system connects the onboard inert gas generation system to the fuel tanks. The dual-flow shutoff valve controls the amount of inert air flow to the fuel tanks. During ground, climb, and cruise operations, the system operates in low-flow mode, generating NEA. During this time, the air space above the fuel is used for NEA storage. This minimizes the amount of NEA required to be generated during descent. During descent, the fuel tank inerting system (FTIS) operates in mid- to high-flow mode, based on the descent rate. When the dual-flow shutoff valve (DFSOV) closes, the valve provides protection against fuel entering the FTIS.

The distribution line from the DFSOV to the fuel tank has two spring-loaded flapper check valves that prevent any fuel that may enter the distribution lines from flowing out of the tank to the FTIS generating system.

The line incorporates a flame arrestor, which prevents fire from reaching the fuel tanks. The flame arrestor consists of a metal canister that contains a grid, specifically designed to eliminate passage of any flame.

A J-trap is located immediately downstream of the distribution check valve inside the fuel tank. The J-trap provides a collection point for any fuel that enters the distribution system. This configuration prevents fuel from coming in contact with the distribution check valve flapper, minimizing the chance of fuel entering the FTIS lines. Any fuel in the J-trap is purged from the line during system startup.

The distribution network consists of tubing arranged in symmetrical left and right paths from the entry point. Flow through each path is controlled by flow balancing orifices located in the dry bay. Each of the orifices in the wing and center tanks have unique part numbers and are not interchangeable. They are sized to ensure that NEA flow is evenly distributed to each tank under all flow conditions. To minimize the chance of fuel entering the lines through sloshing or gravitational effects, the tubing is located at high points.

The NEA discharge outlets are located as far as possible from the fuel vent line openings, to ensure that the tank is adequately flooded before the NEA flows overboard through the vent scoop.

Maintenance port plugs are used to inspect for the presence of fuel in the distribution lines. They also provide access points for conducting pressure tests of the distribution lines and check valves.

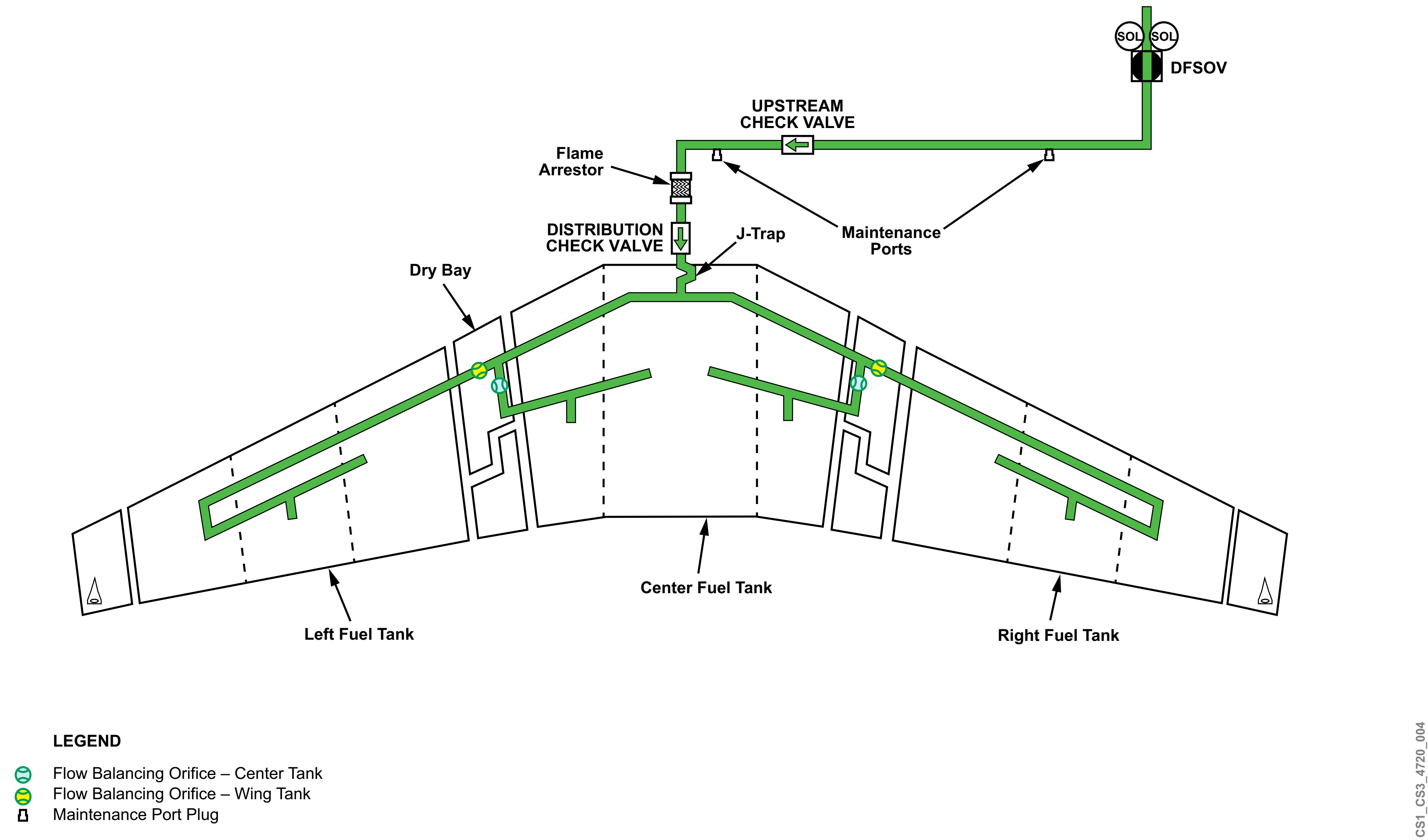


Figure 17: Fuel Tank Inerting System Distribution

COMPONENT LOCATION

The following components are installed in the distribution system:

- Check valves
- Flame arrestor
- Flow balancing orifices
- Maintenance port plugs

CHECK VALVES

The distribution check valve and upstream check valves are located at the front spar outside of the fuel tank near RIB 0.

FLAME ARRESTOR

The flame arrestor is mounted between the distribution and upstream check valves.

FLOW BALANCING ORIFICES

There are two flow balancing orifices in each dry bay outboard of RIB 6. One orifice is for the center tank distribution line and one is for the wing tank distribution line.

MAINTENANCE PORT PLUGS

Two maintenance port plugs are located in the distribution line near the center tank front spar.

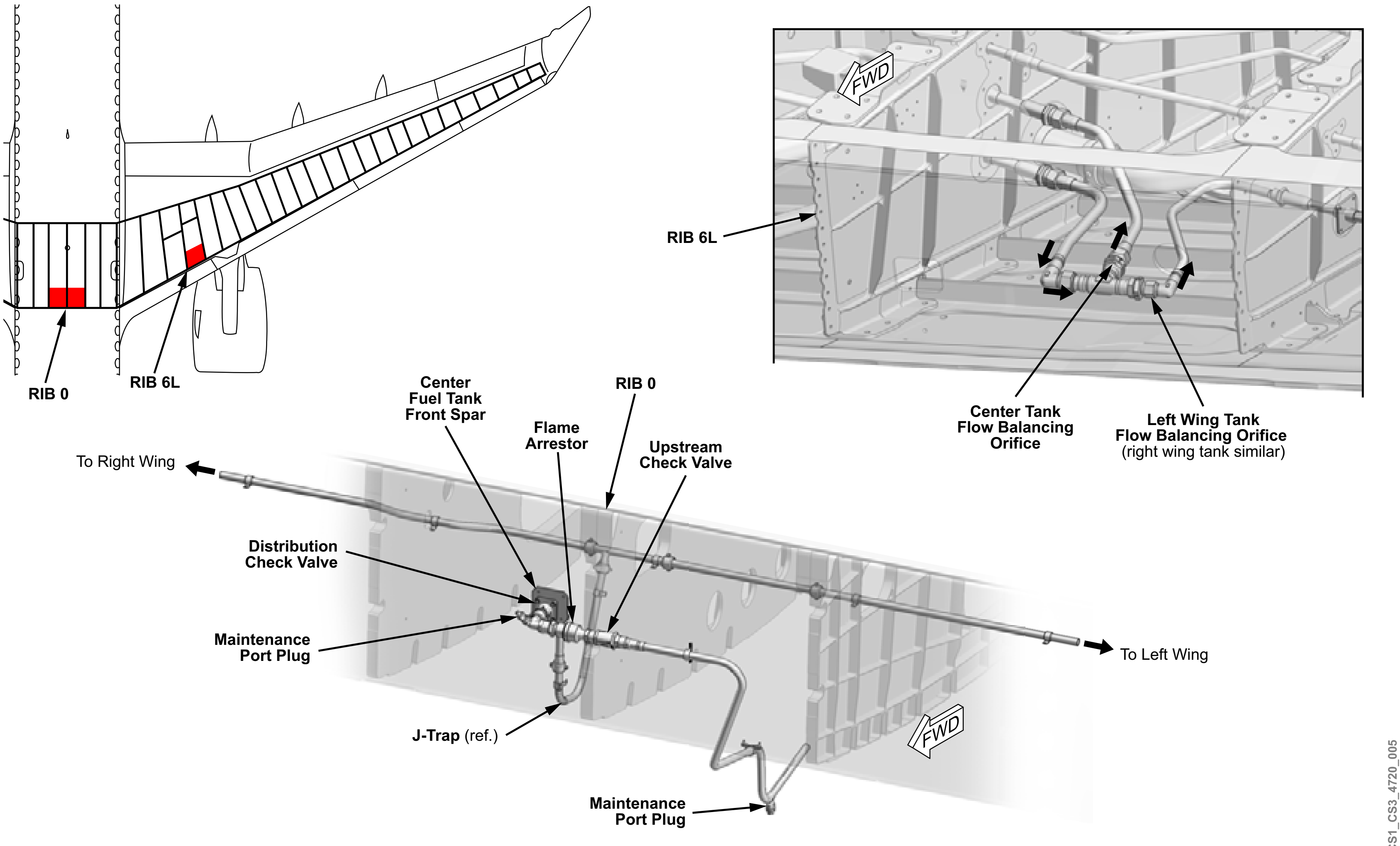


Figure 18: Fuel Tank Inerting System Distribution Components

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ATA 49 - Airborne Auxiliary Power

BD-500-1A10
BD-500-1A11

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AIRBORNE AUXILIARY POWER - CHAPTER BREAKDOWN

49-00 APU CONSTRUCTION

GENERAL DESCRIPTION

The auxiliary power unit (APU) design consists of two modules: the power section and load compressor, and the gearbox with accessories.

LEFT SIDE VIEW

The following major components are installed on the APU left side:

- Fuel control unit (FCU)
- Lube module
- APU drains
- Ignition unit
- Thermostatic pressure-relief valve
- Air/oil cooler (AOC)

The wiring harness provides an interface between the APU and the electronic control unit (ECU). The starter motor and APU generator have their own wiring harnesses to interface with the aircraft.

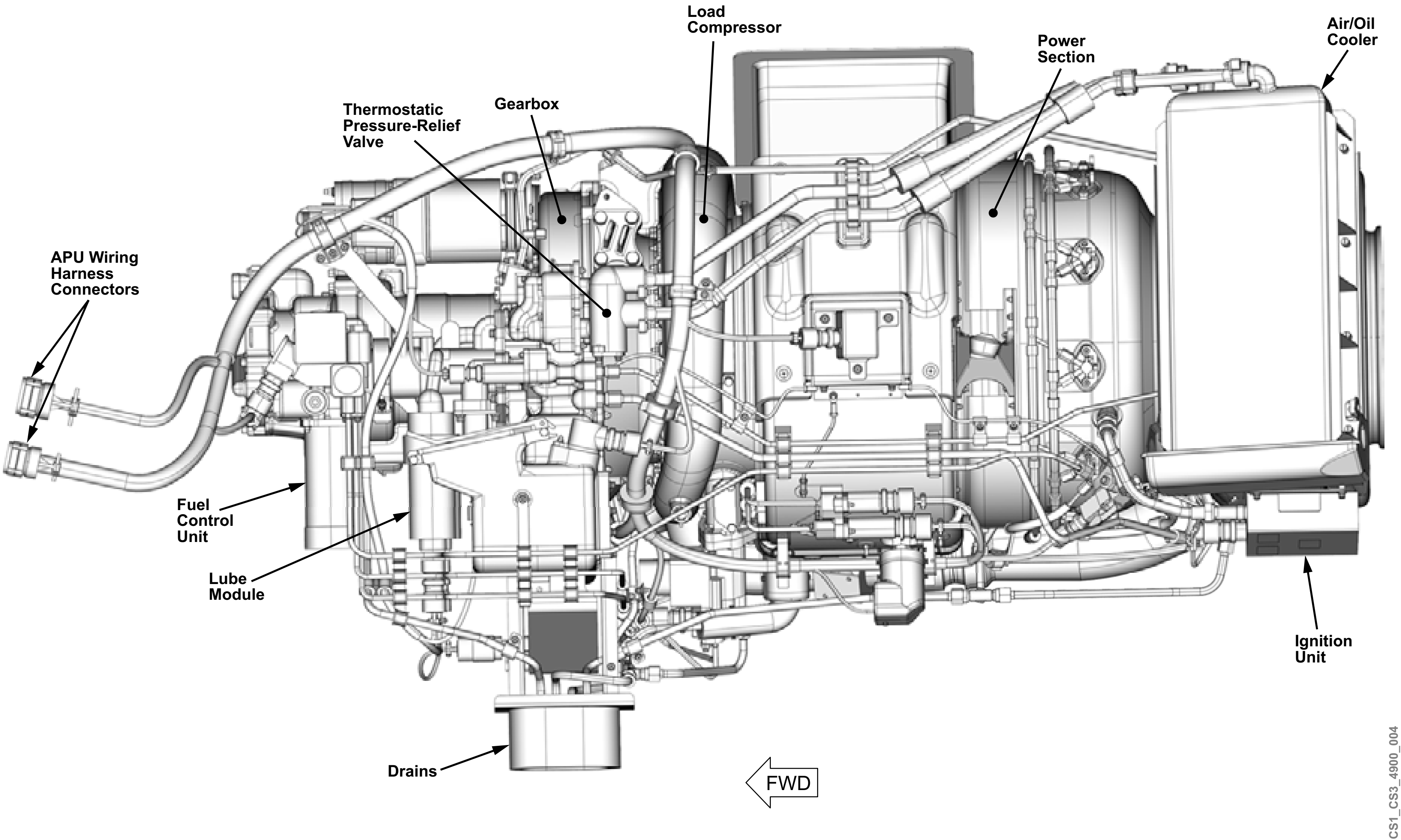


Figure 1: APU Left Side View

RIGHT SIDE VIEW

The following major components are installed on the APU right side:

- APU generator
- Starter motor
- Bleed air valve (BAV)
- Surge control valve
- Fuel manifold and nozzles

Compartment cooling air is drawn through an eductor that surrounds the exhaust to cool the APU oil and provide compartment ventilation.

The bleed air duct provides the interface with the aircraft pneumatic system.

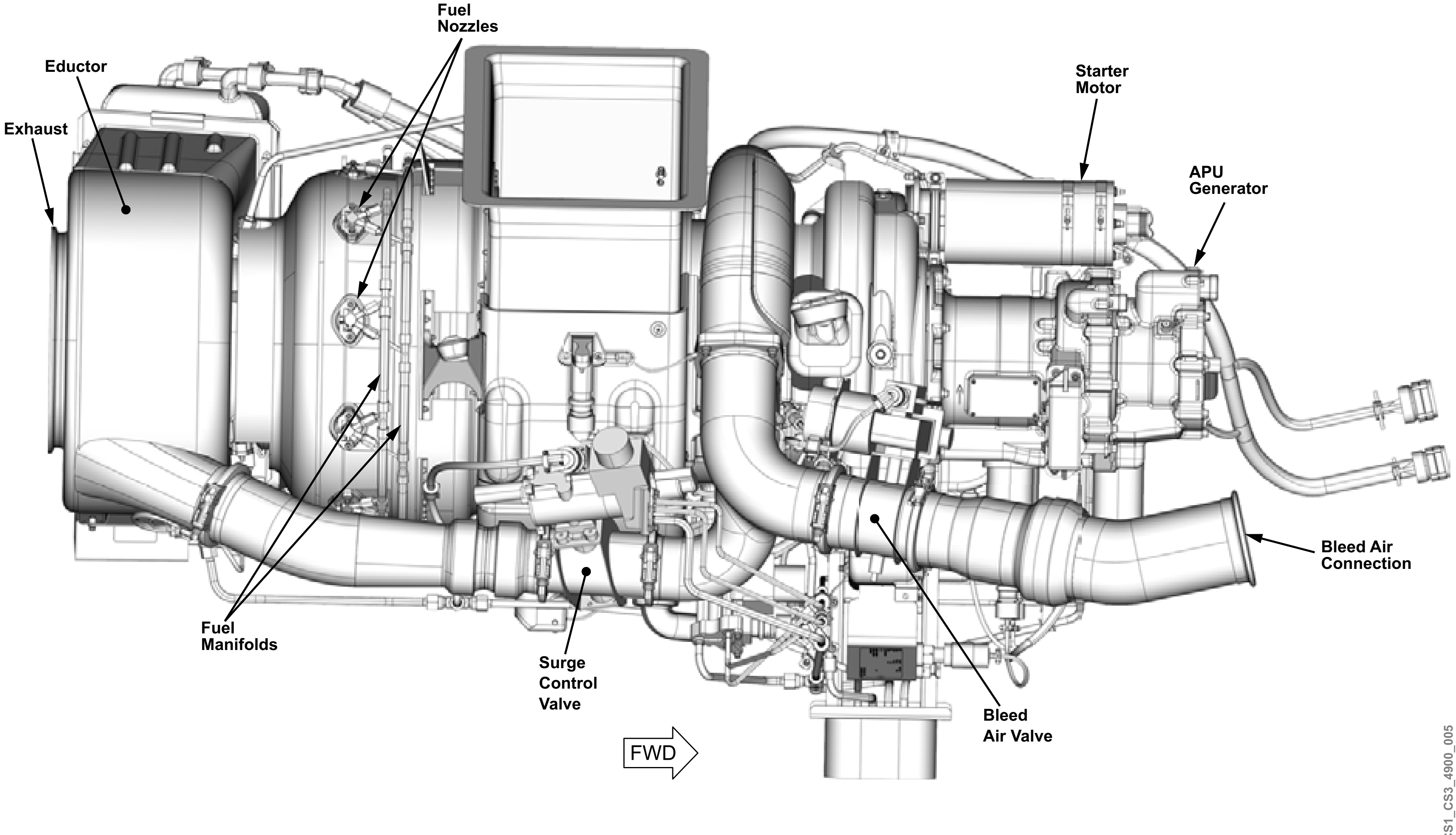


Figure 2: APU Right Side View

ACCESSORY GEARBOX SECTION

The accessory gearbox is driven through a quill shaft by the high speed torque of the power section. The gearbox contains a series of spur gears to drive the APU accessories.

The following accessories are installed on the gearbox:

- Generator
- Lubricating module
- Fuel control unit (FCU)
- Starter motor
- Air/oil separator
- Oil fill port
- High oil temperature sensor

The gearbox also serves as an oil reservoir for the wet-sump lubrication system. An oil drain plug located near the bottom of the gearbox incorporates a magnetic chip collector that can be removed separately.

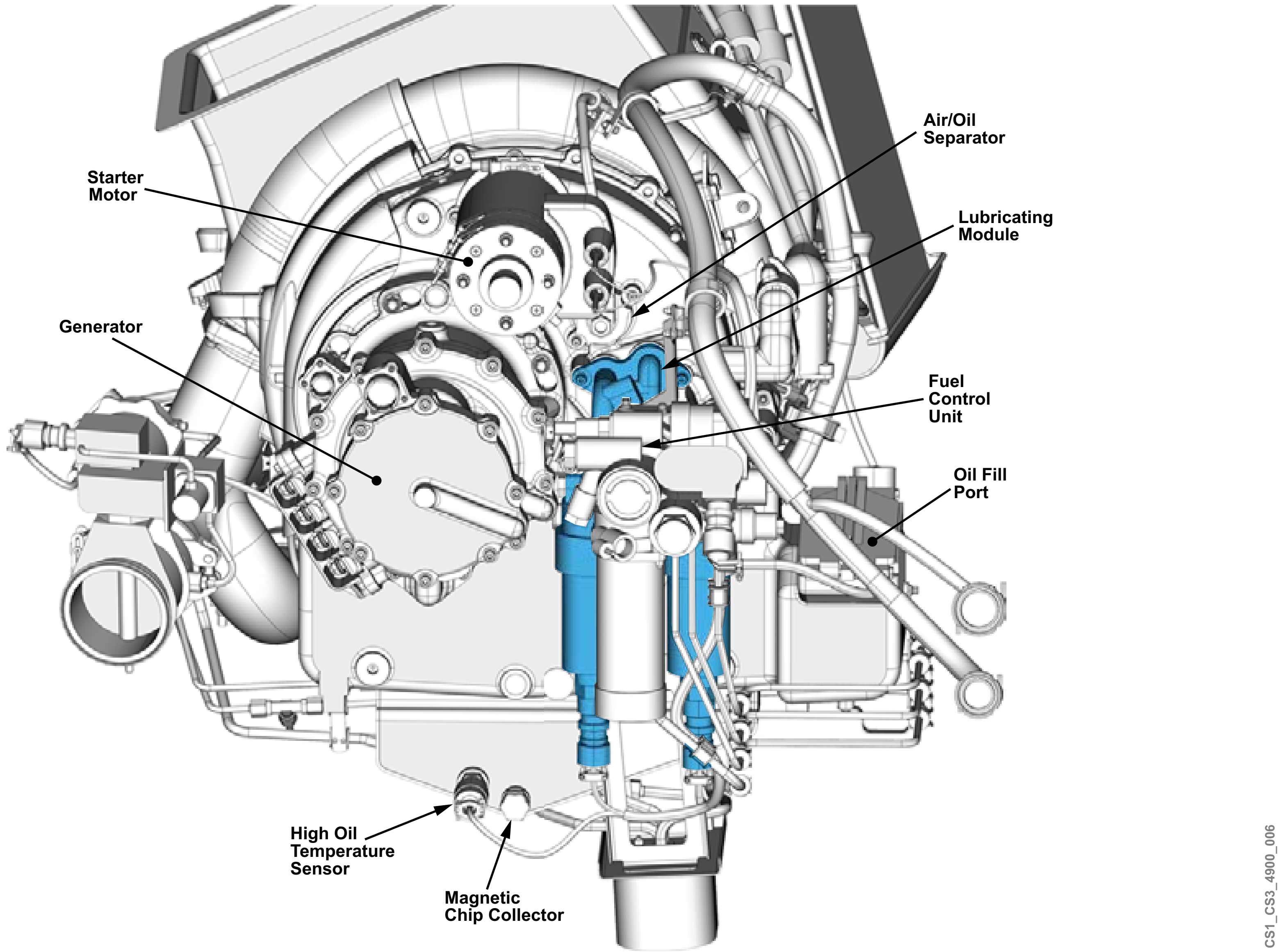


Figure 3: APU Gearbox

49-10 APU INSTALLATION

GENERAL DESCRIPTION

The auxiliary power unit (APU) sits in an enclosure that consists of five titanium firewalls and a pair of hinged composite access doors. The APU is suspended in the enclosure at a 13° nosedown attitude. The enclosure isolates the APU from the rest of the structure and acts as a load path to support the APU. Seals are provided around the firewall and door interface.

There are three access panels on both the right side firewall and left side firewall and two access panels on the top firewall, which are used to gain access to the system components and structures. There is also one access panel located on the aft firewall, to facilitate the removal of the muffler.

APU compartment cooling and ventilation is provided by the APU air inlet duct when the APU inlet door opens during APU operation.

A pair of hinged access doors provides access for servicing and maintenance of the APU. A drain allows accumulated fluids to exit through a small opening in the left hand access door.

The electrical wiring harnesses provide the interfaces between the electronic control unit (ECU) and the APU. The APU wiring harness extends from the firewall connector to the front of the APU, and connect all electrical components. APU harness connectors are threaded, stainless steel, and self-locking.

A bleed air duct connects the APU pneumatic output to the aircraft.

COMPONENT LOCATION

The APU installation consists of the following components:

- Access doors
- APU drains
- Engine mounts and struts
- Eductor
- Exhaust

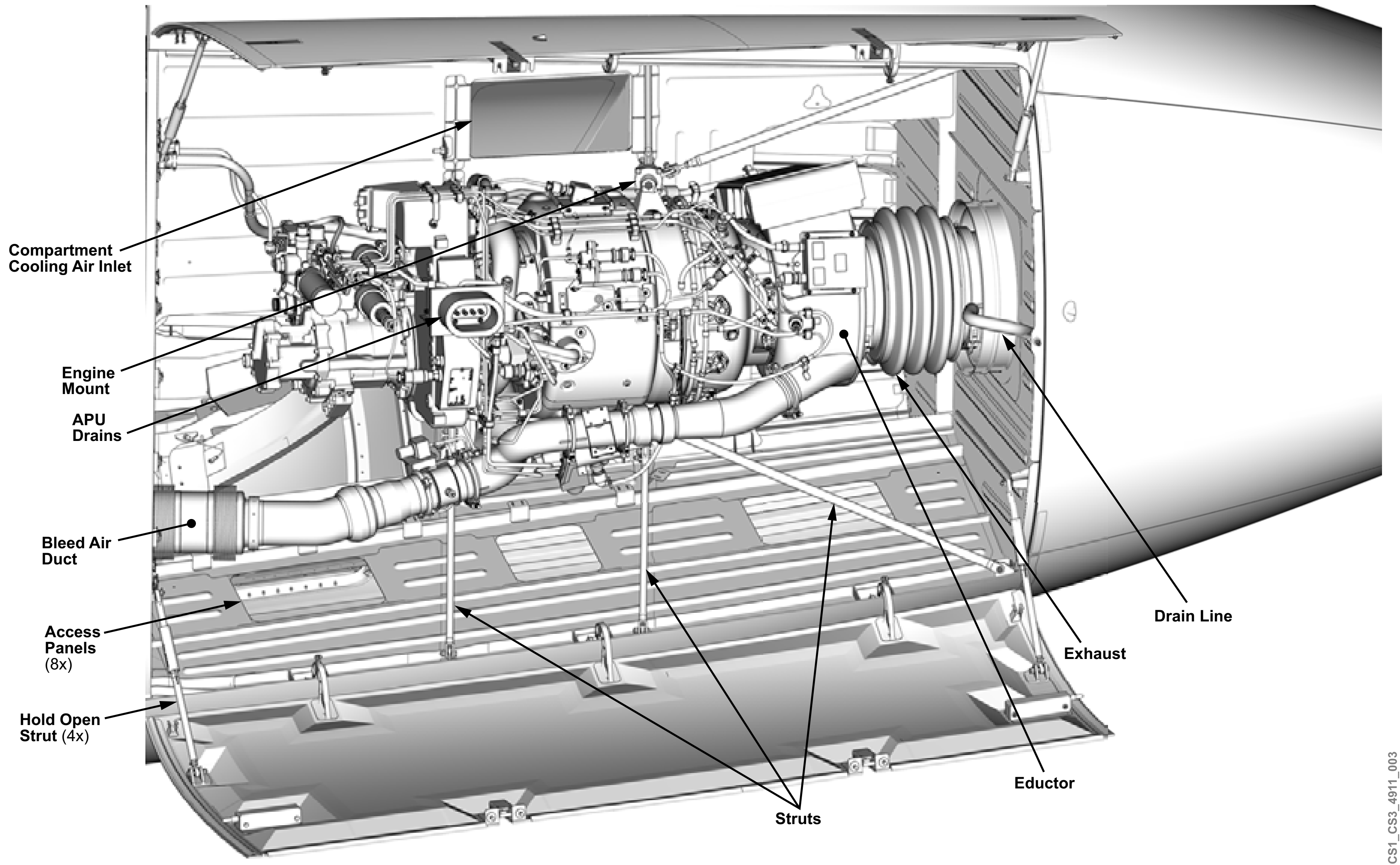


Figure 4: APU Enclosure and Installation

COMPONENT INFORMATION

ACCESS DOORS

The APU doors are attached to the aircraft structure by goose neck hinges. Fireproof door seals located at the interface of the APU firewall and access doors help to prevent fire propagation and provide containment of Halon when extinguisher is activated.

The doors are secured closed by two transverse hook latches and four pin latches. A door seal runs the length of the door interface. There are drain holes along the forward edge of the doors. Accumulated fluid from the APU drains exit through a hole near the forward transverse latch.

CAUTION

Do not keep the doors in the full open position if winds are more than 65 kt (120 kph). Damage to door structures, hinges, and supports can occur.

NOTE

Open the left door first if the two doors are to be opened. Close the right door first when the two doors are open.

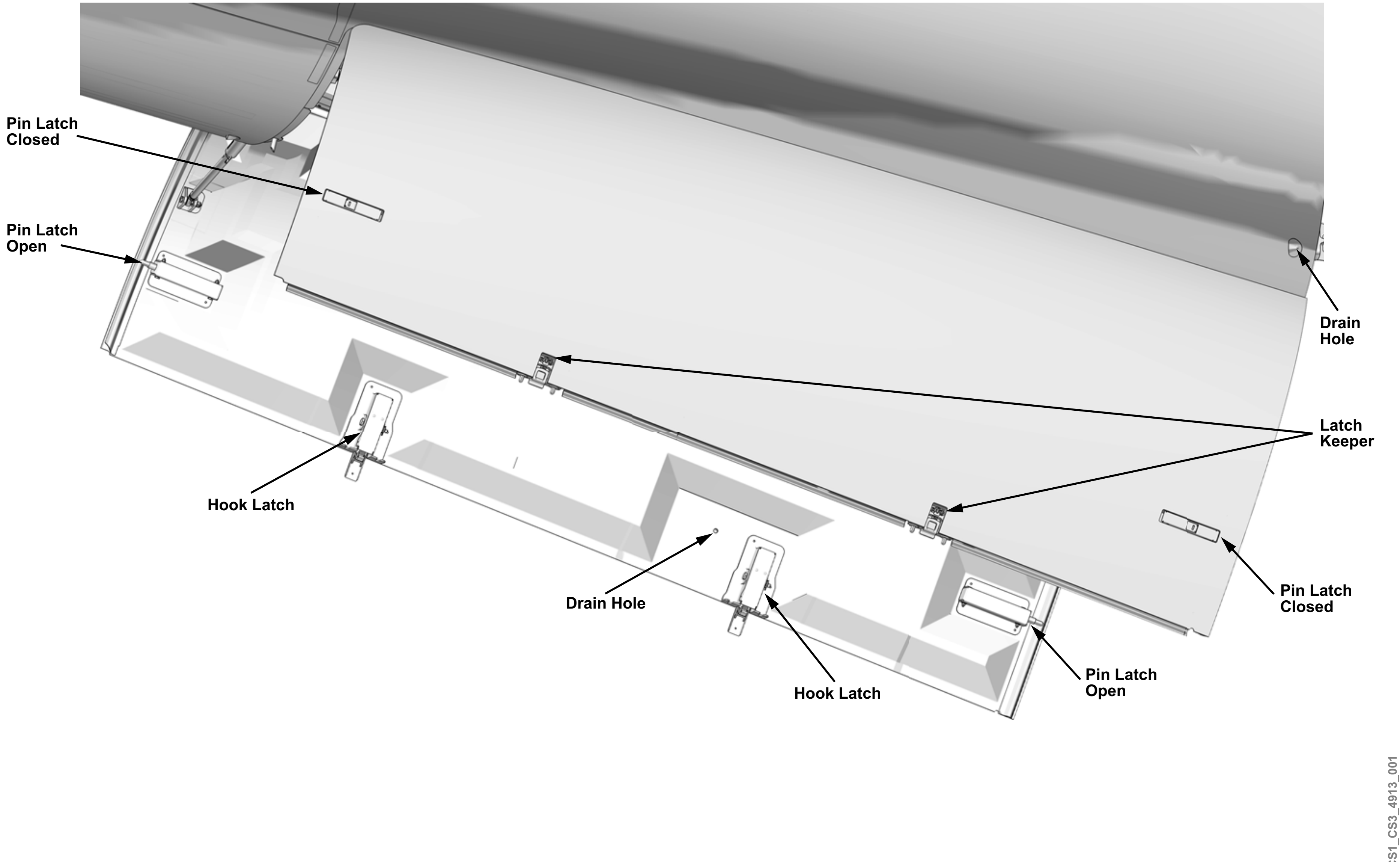


Figure 5: APU Access Doors

ACCESS DOOR STRUTS

Four struts hold the access doors open. The struts are fixed to the doors and enclosure. When the doors are opened or closed, the struts fold and stow in clips mounted on the doors.

A sleeve locks the strut in the extended position when the doors are opened.

Each strut has an adjustable rod end to ensure the door is flush with the fuselage.

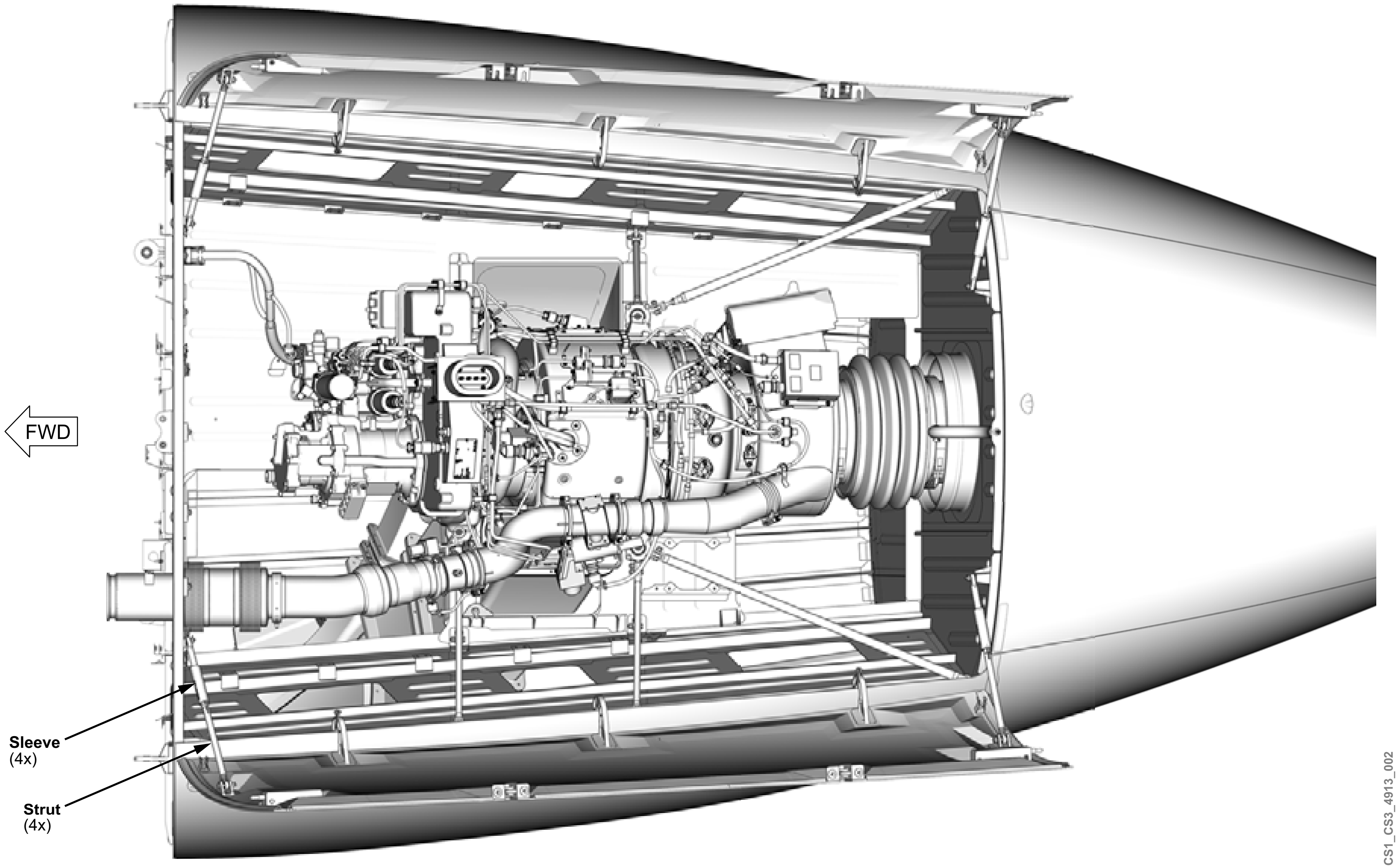


Figure 6: APU Access Door Struts

APU DRAINS

The overboard drains are used to drain the APU inlet plenum and exhaust. The drains aid in detecting fuel leaks at the fuel control unit, inlet guide vane actuator and surge control valve, and oil leaks at the load compressor seal. The last drain allows fluid accumulation in the muffler to drain as well as fuel from an unsuccessful start to drain from the combustor plenum drain orifice.

The APU Inlet plenum drain tube disposes of the fluids that enter the APU inlet. The fluids drain on the interior of the APU compartment door near a drain hole.

All of the APU drains, except for the inlet plenum drain, are grouped at a drain manifold that is supported off the APU structure. The drain manifold provides sealing via a silicone-rubber bathtub seal to the compartment access door. From this manifold, the system drains into a cavity created by the seal and the APU compartment access door. Fluids drain overboard through a drain hole in the compartment access door.

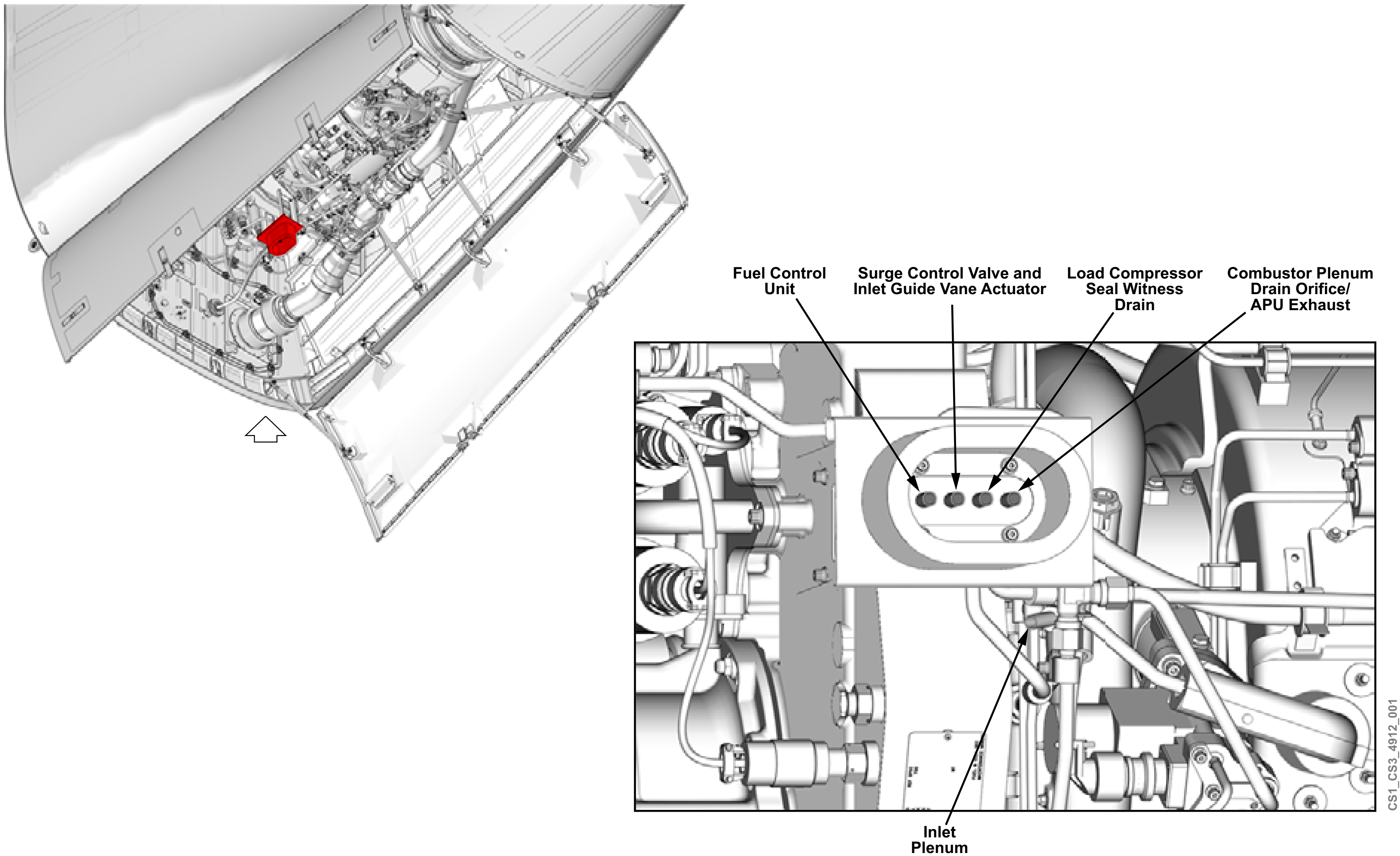


Figure 7: APU Drains

DETAILED COMPONENT INFORMATION

MOUNTS AND STRUTS

The APU is suspended in the enclosure at a 13° nosedown attitude by a series of struts and isolators. The system includes four groups of struts that connect to the three mounting points on the APU. The four groups of struts have a suspension system with seven active struts and one catcher strut. A vibration isolator connects each group of struts to the mounting points.

The mount system consists of a right side forward isolator with two struts, a left side forward catcher with an adjustable strut, a right side aft isolator with three fixed struts, and a left side aft isolator with one fixed strut, and one adjustable strut.

The mount system provides a redundant load path to maintain the APU in position, and provides redundant vibration and shock isolation from the APU to the aircraft. The mount system maintains the APU in position even if one strut fails, or an entire mount is damaged by fire.

All isolators react to vertical and lateral loads. The aft isolators also react to thrust loads. The forward left link only reacts to loads in failed condition.

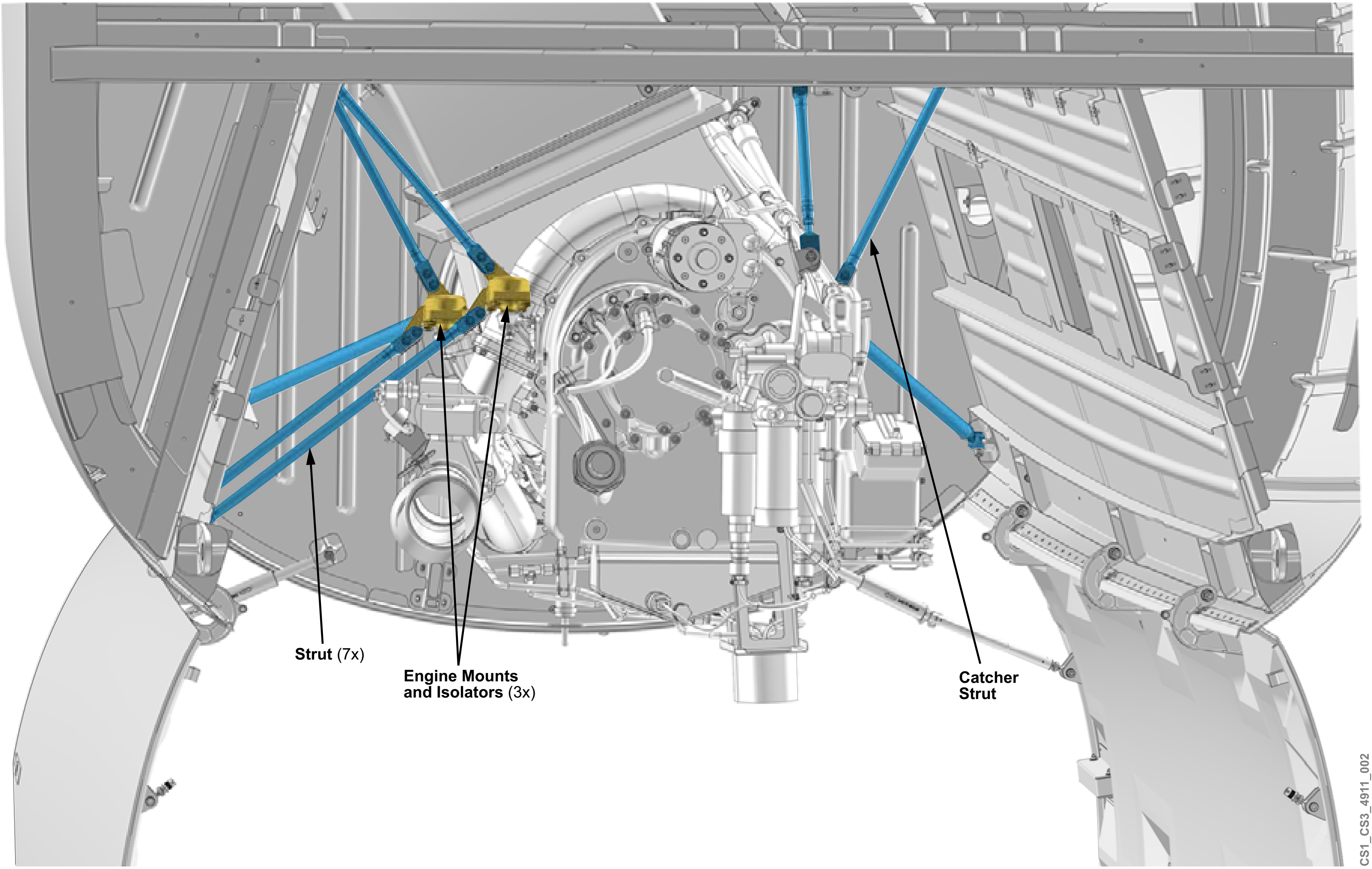


Figure 8: APU Mounts and Struts

EDUCTOR

The eductor provides APU compartment cooling and oil cooling. The eductor uses APU exhaust gas to passively create APU compartment cooling flow. The kinetic energy of the high-velocity exhaust gas draws the flow of lower velocity compartment cooling airstream in a mixing duct. Airflow from the surge system is also introduced in the eductor.

Airflow from the compartment is pulled across the oil cooler prior to entering the exhaust stream. The mixed exhaust flow results in lower exhaust system surface temperatures.

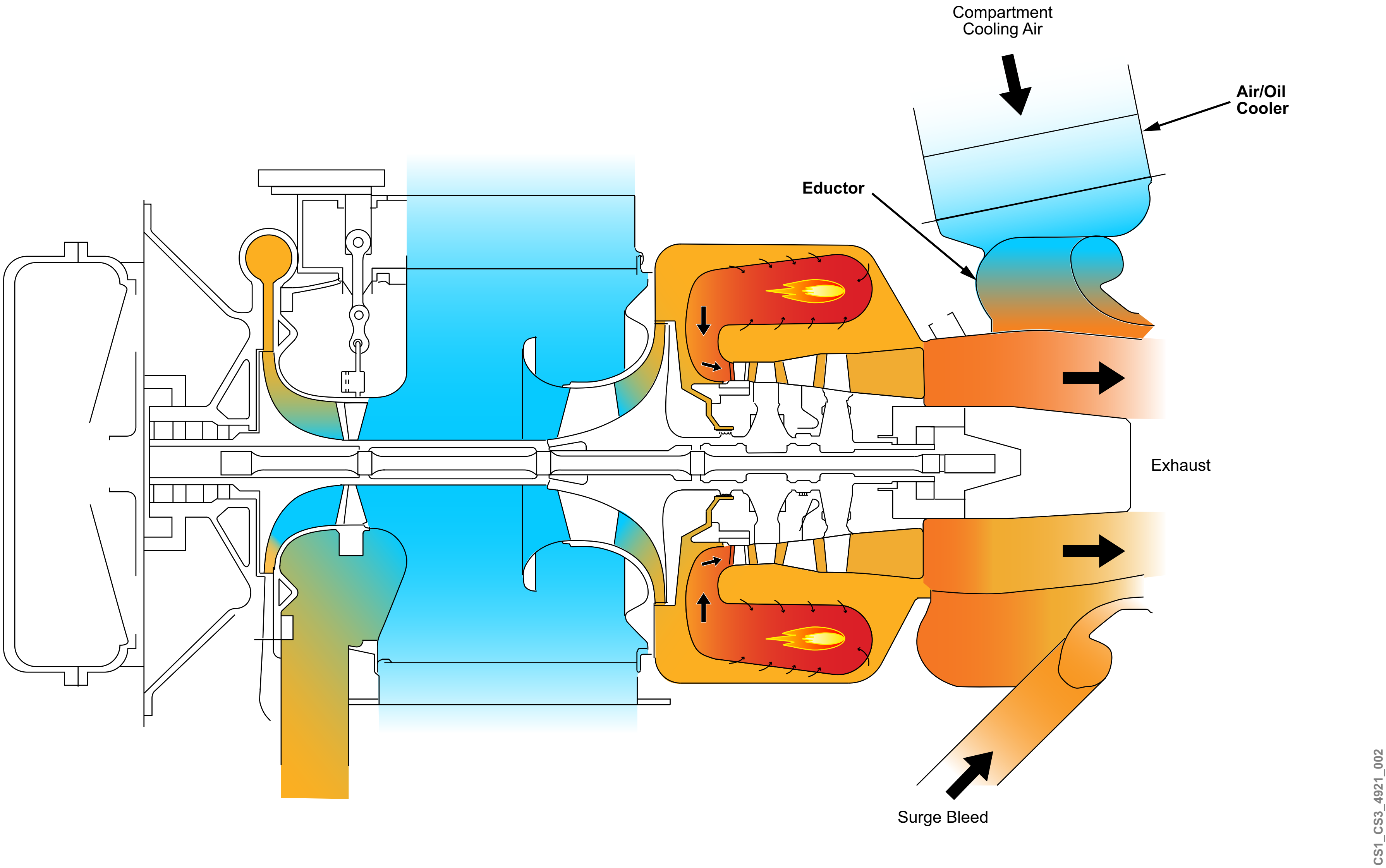


Figure 9: APU Eductor

EXHAUST

The APU exhaust system is located in the tailcone of the aircraft, aft of the APU compartment. The exhaust has a muffler for noise reduction.

The APU exhaust duct attaches to the engine using flexible bellows. The bellows take up the relative motion between the engine and exhaust. A bellows liner provides a smooth surface for the gases to flow through. The bellows bolts to the muffler at the aft firewall.

The muffler consists of an outer shell and an inner liner separated by baffles. The inner liner can be removed separately from the outer shell. The muffler is covered by a thermal blanket. The muffler has two lifting points on top for removal.

The front muffler face has a fire seal and mounting tabs to mount the exhaust duct to the collar frame at the aft firewall. The muffler has a forward facing drain that expels any fluid accumulation. The drain connects to a flexible tube that drains overboard through the APU access doors

The tailpipe is attached to the muffler. The tailpipe is covered by an upper and lower thermal blanket, where it extends through a cutout in the tailcone.

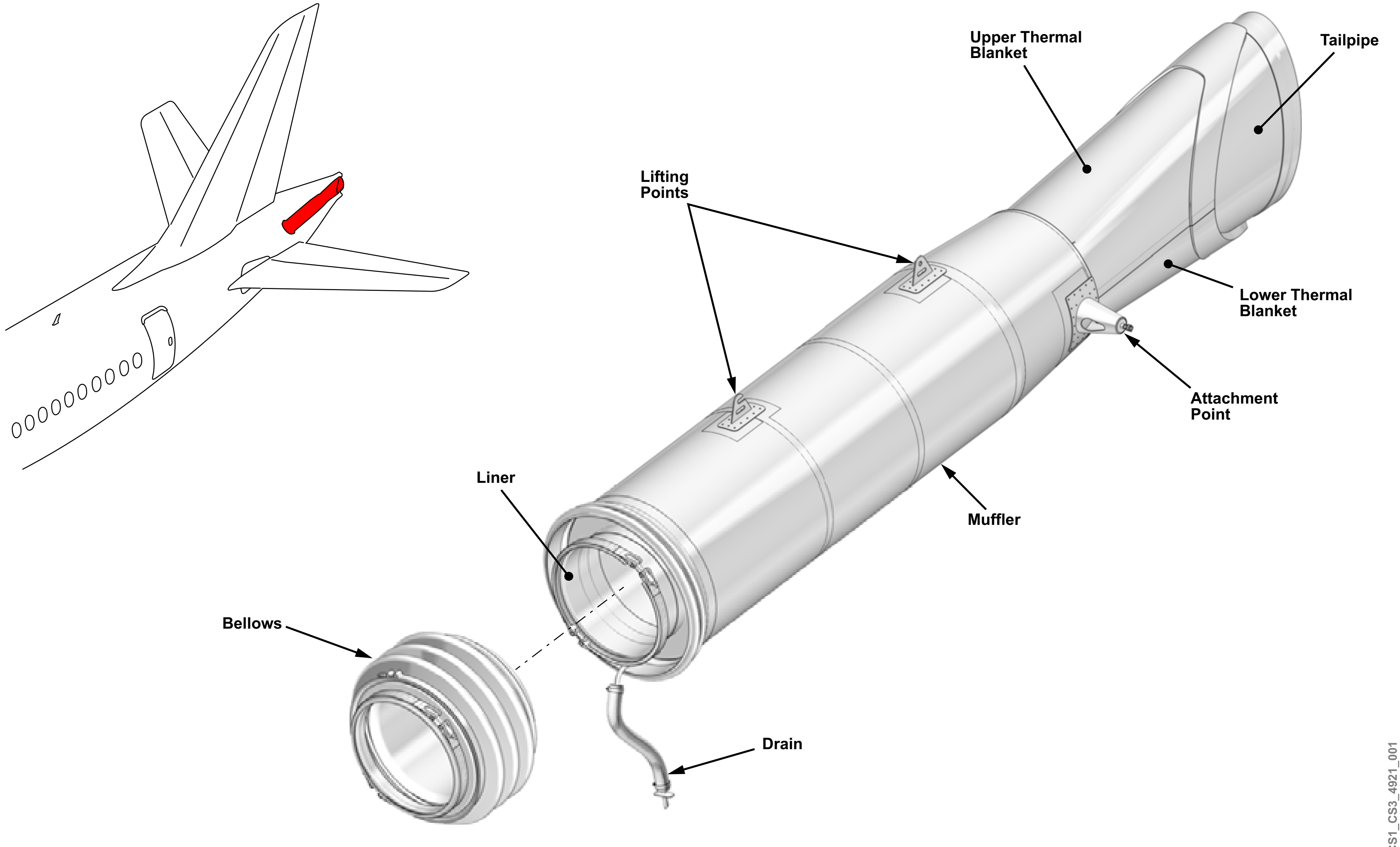


Figure 10: APU Exhaust

APU DRAINS SCHEMATIC

The forward drain disposes of the fluids that enter the APU inlet. The second drain detects degraded seal performance in the fuel control unit (FCU), inlet guide vane (IGVA) actuator, and surge control valve (SCV) actuator. The third drain detects degraded load compressor main shaft seal performance. The aft drain provides a means for disposing of excess fuel in the combustor after an aborted start. It also serves as a drain for any fluid accumulation in the APU exhaust. During normal operations, no fuel is discharged from the aft drain.

Telltale drains help find fuel seal failures. They are located in the drain lines from the FCU, IGVA, and SCV. A cap at each drain can be removed and checked for the presence of fuel as part of the troubleshooting procedure for isolating a fuel leak.

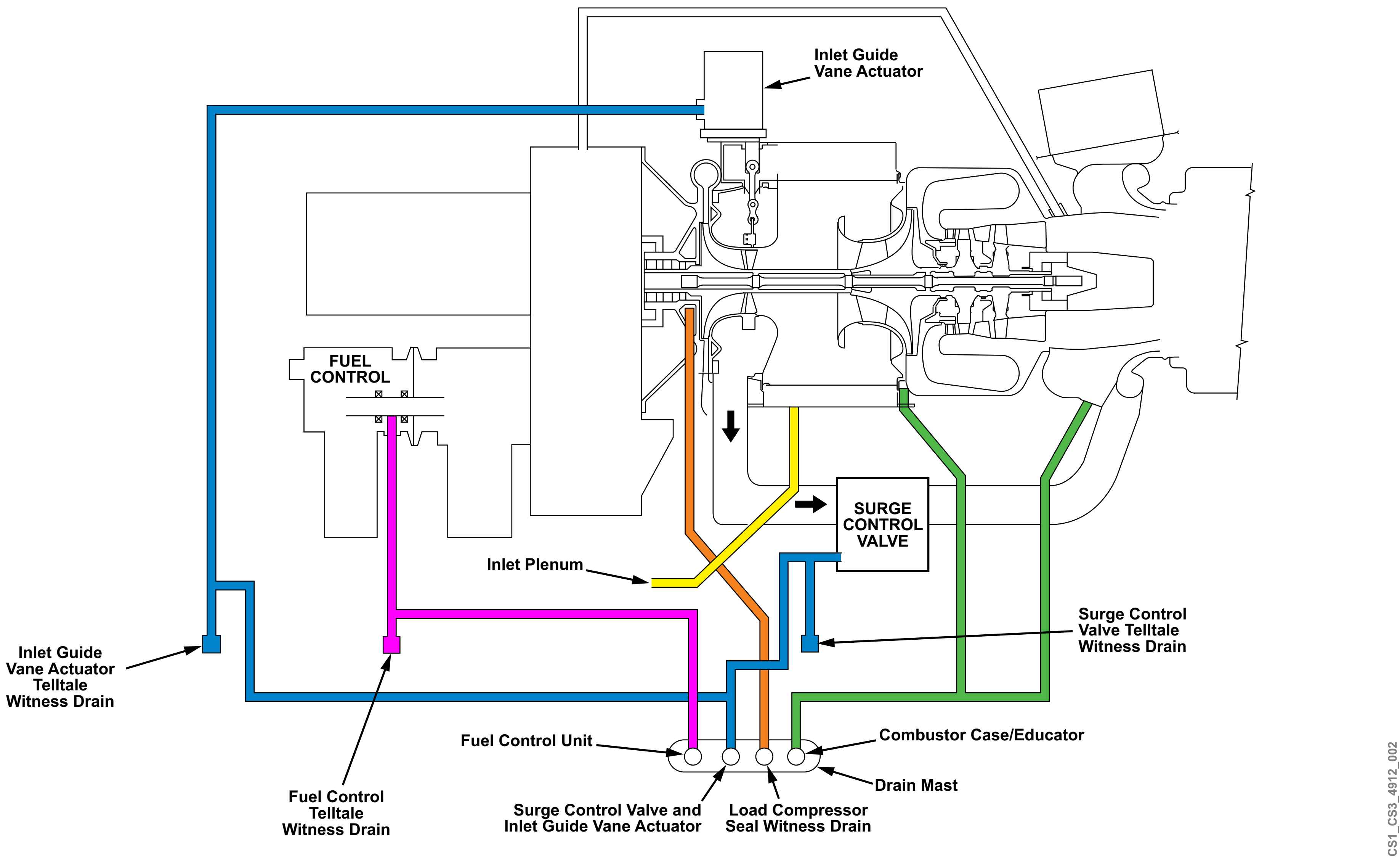


Figure 11: APU Drains Schematic

49-12 APU AIR INTAKE

GENERAL DESCRIPTION

AIR INLET

The auxiliary power unit (APU) air inlet supplies air for APU operation, cooling, and ventilation. The APU inlet door directs air into the air inlet duct. Cooling air is ducted to the APU compartment by splitting the APU inlet air at the inlet door on the aircraft skin. The inlet duct provides an airflow path to the APU plenum. The compartment cooling air ducting provides cooling air to the APU enclosure. The air flows into the APU compartment, circulates around the APU to vent toxic and combustible gases, before exiting through the exhaust-mounted oil cooler and eductor.

During ground operation, the velocity of the hot gas exhaust discharge of the APU through the eductor creates a low-pressure around the APU, pulling the cooling air through the system. During flight operation, ram air at the inlet door provides additional force to move the cooling air.

APU AIR INLET DOOR

The APU inlet door is an outward opening door that scoops air within the boundary layer and directs the air flow to the APU compressor and oil cooler inlet. In the closed position, the door prevents the APU from windmilling in flight.

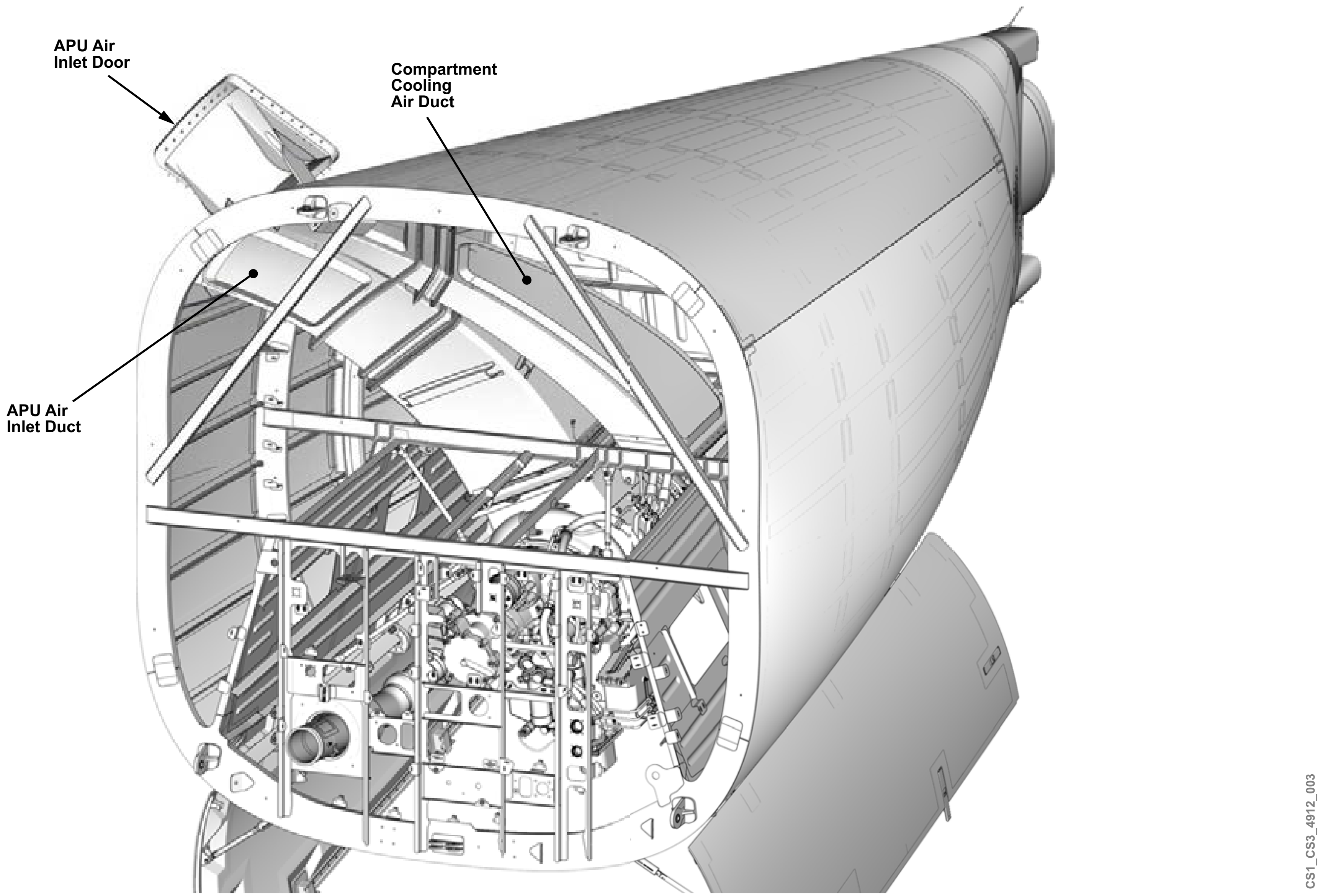


Figure 12: APU Air Intake

APU AIR INLET DOOR OPERATION

The door opens to 45° for ground operations and 30° in flight. When the door is open, an air gap between the inlet door and the fuselage reduce inlet losses, improve flow stability, and reduce door buffeting.

The inlet door actuator moves the lever arm to open and close the inlet door. The inlet door has a mechanical stop that limits the door travel to 50°.

The APU electronic control unit (ECU) controls the door position. The actuator motor is powered from the 28 VDC EMER BUS. An air inlet door position sensor provides door positioning information to the ECU for door control.

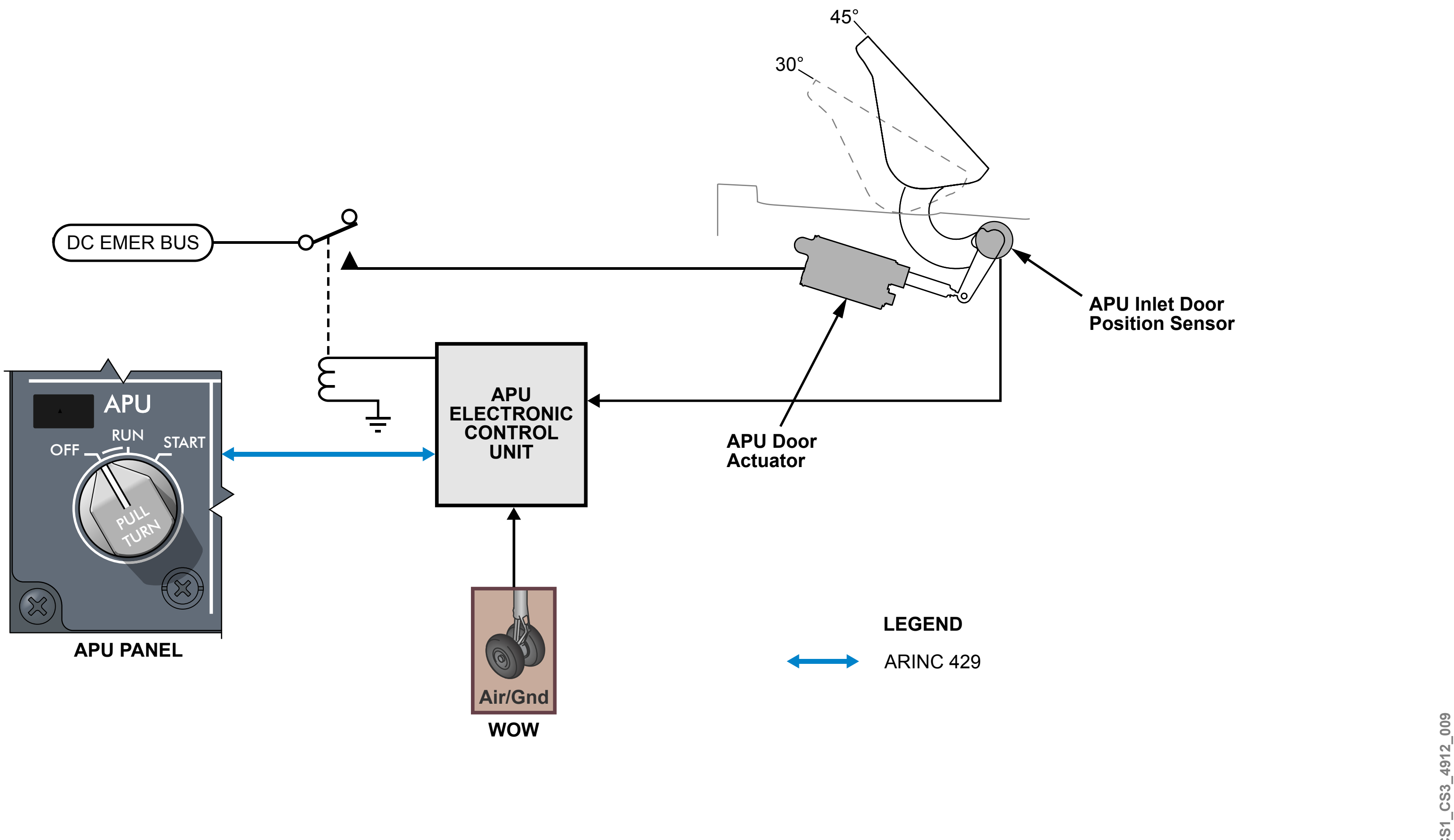


Figure 13: APU Air Inlet Door Operation

COMPONENT LOCATION

The following components are part of the APU air intake:

- Air inlet door
- Air inlet door actuator
- Air inlet door position sensor

AIR INLET DOOR

The air inlet door is located on the right side of the vertical stabilizer above the APU compartment.

AIR INLET DOOR ACTUATOR

The air inlet door actuator is installed on the left side of the air inlet door.

AIR INLET DOOR POSITION SENSOR

The air inlet door position sensor is installed on the right side of the APU air inlet door.

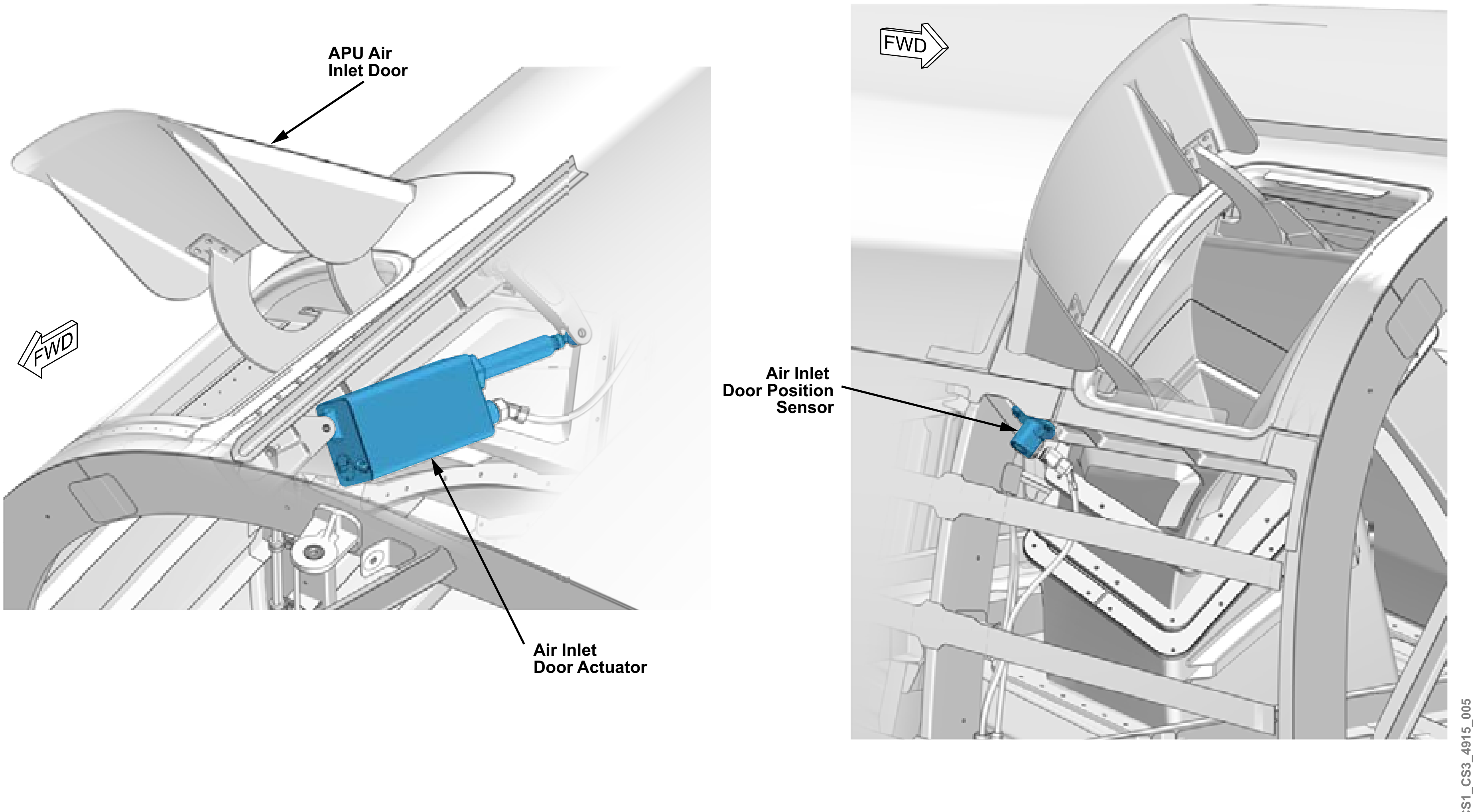
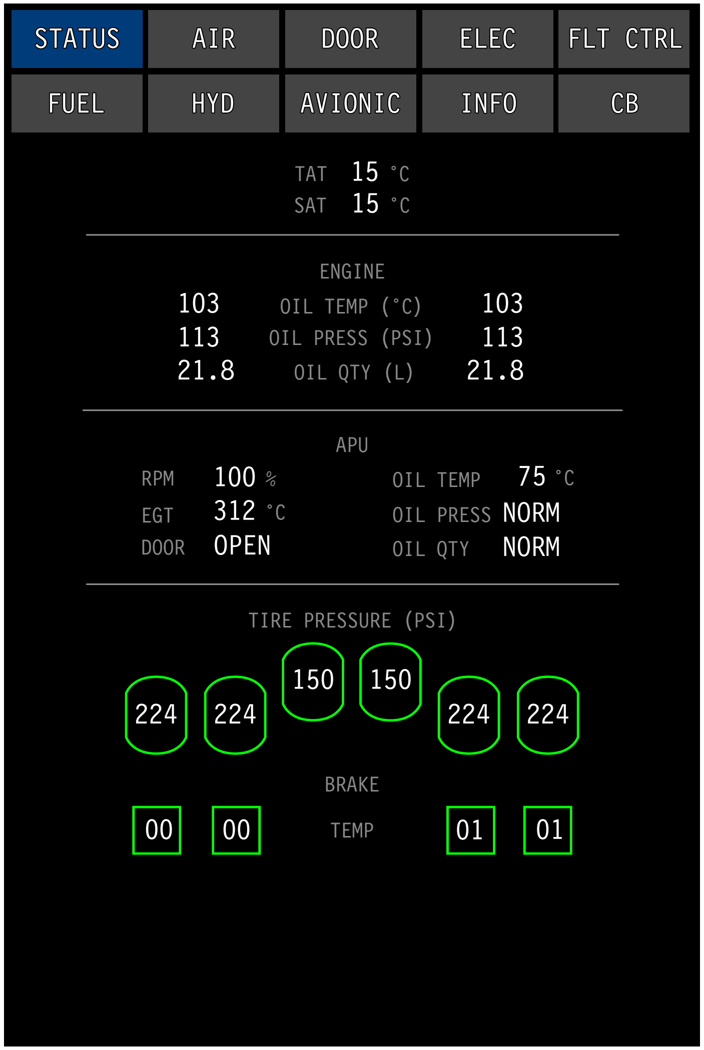


Figure 14: APU Air Intake Components

CONTROLS AND INDICATIONS

The APU inlet door status is shown on the STATUS page.



SYNOPTIC PAGE - STATUS

APU DOOR	
Symbol	**Condition**
OPEN	APU door open.
CLOSED	APU door closed.
OPEN	APU door failed open.
CLOSED	APU door failed closed.
--	APU door position invalid.

CS1_CS3_4915_006

Figure 15: APU Door Indication

DETAILED DESCRIPTION

When the ECU receives the APU switch START command, the APU inlet door opens. The ECU supplies power to energize the APU relay and excitation for the air inlet door position sensor.

The 28 VDC power is removed from the actuator when the air inlet door position sensor reports the door is in the correct position, based on weight-on-wheels (WOW) sensing. The actuator is driven by the electronic control unit (ECU) logic to the correct position for starting and operating conditions. There are three inlet positions: closed, 30° open for flight, and 45° open for ground operation.

If the ECU does not determine that the inlet door is open within 30 seconds after issuing the command, APU starting is inhibited.

The ECU supplies 28 VDC to close the inlet door during the normal APU shutdown sequence as well as protective shutdowns. The 28 VDC power is removed when the ECU detects that the inlet door is in the closed position.

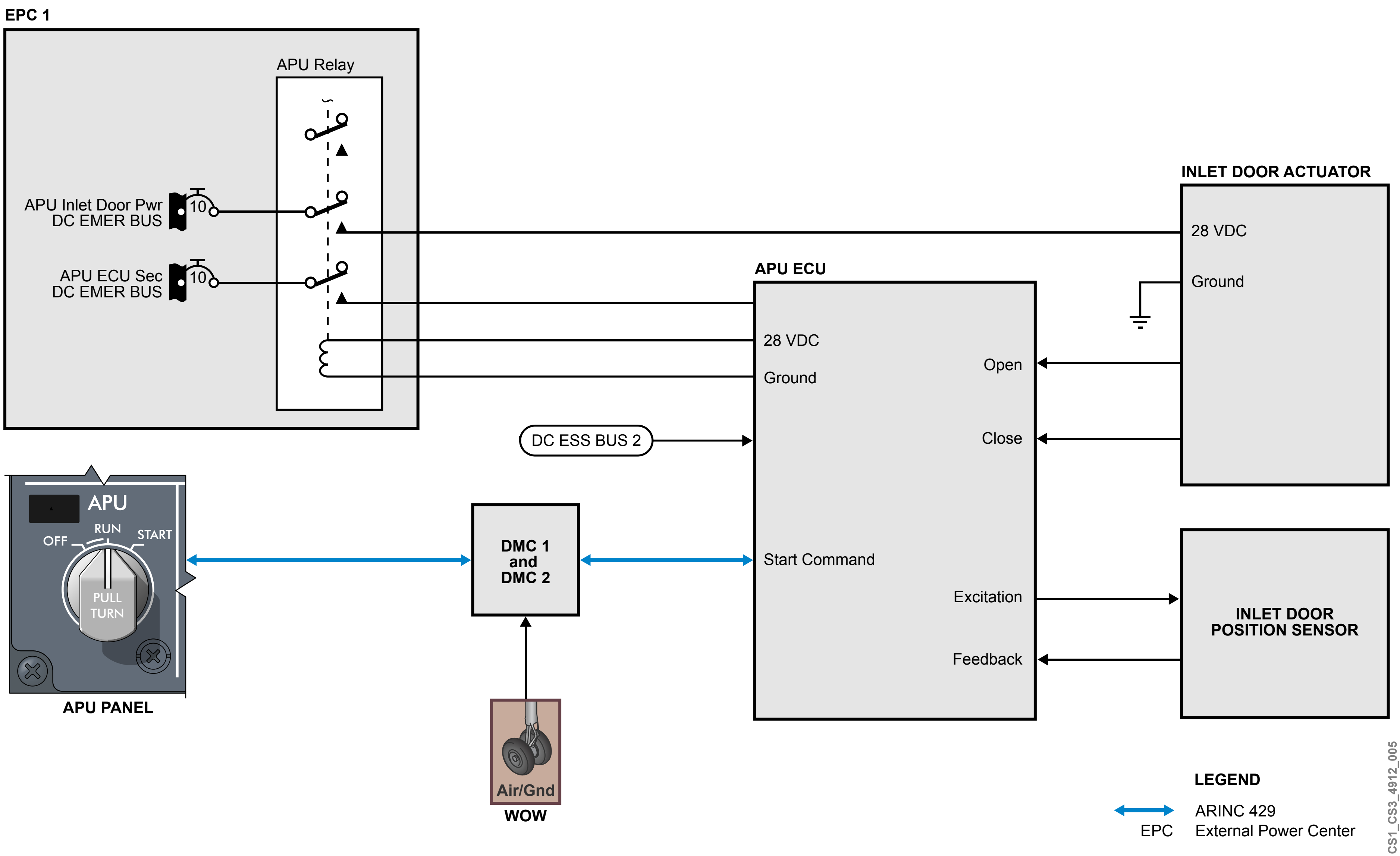


Figure 16: APU Inlet Door Operation

MONITORING AND TESTS

The following page displays the crew alerting system (CAS) and INFO messages for the APU air intake system.

CAS MESSAGES

Table 1: CAUTION Message

MESSAGE	LOGIC
APU DOOR OPEN	APU door failed to close when required.

Table 2: INFO Message

MESSAGE	LOGIC
49 APU FAULT - APU DOOR INOP	APU door failed to open or APU door closed after open or WOW and APU door RVDT fault.

PRACTICAL ASPECTS

APU INLET DOOR MECHANICAL RIGGING

The door rigging procedure ensures the door is in the proper closed position in respect to the fuselage and door seal, as well as providing the correct door open positions for ground and flight. Always refer to the Aircraft Maintenance Publication (AMP) for the latest information.

Access to the APU inlet door actuator is through the forward upper access panel on the APU enclosure. The bleed air duct between the bleed air valve (BAV) and the forward firewall must be removed first for access.

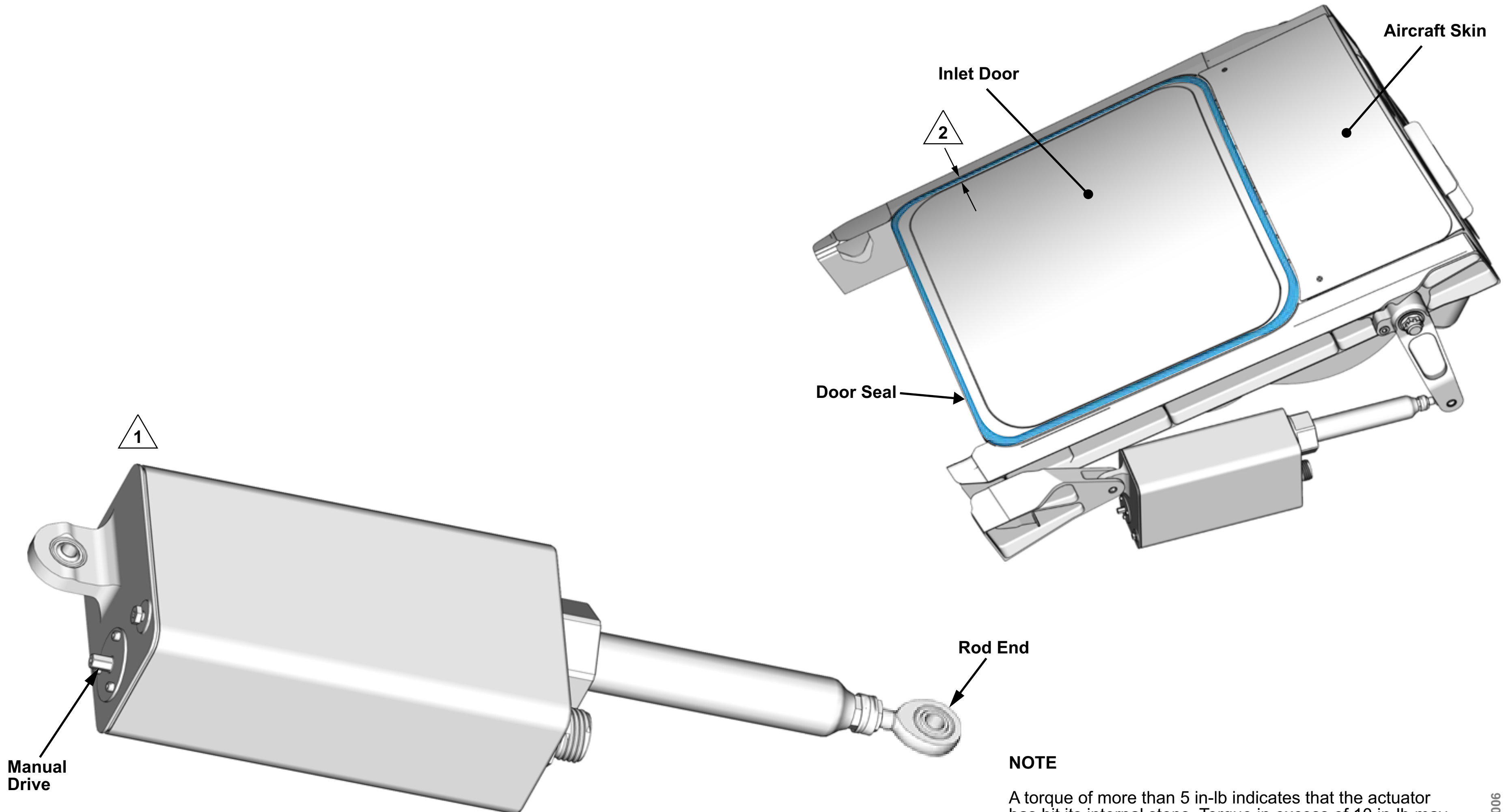
The door can be manually driven using the 1/4 in. hex manual drive on the door actuator. Turning the manual drive in a clockwise direction to retracts the actuator and opens the door. Turning the manual drive counterclockwise extends the actuator and closes the door.

To mechanically rig the door:

- Disconnect the actuator from the inlet door lever arm
- Manually move the inlet door from fully open to fully closed and ensure the inlet door moves freely
- Reconnect the lever arm to the inlet door and manually drive the actuator counterclockwise to close the inlet door until it contacts the door seal
- Manually drive the actuator clockwise to check that the inlet door opens without interference from any components. Manually drive the actuator to close the inlet door.
- Measure between the inlet door and aircraft skin to make sure a gap of approximately 1 cm (0.4 in) exists.

CAUTION

Use caution when manually driving the actuator. A torque of more than 5 in-lb indicates that the actuator has hit its internal stops. Torque in excess of 10 in-lb may damage the APU inlet door actuator, APU inlet door seal or APU inlet door mechanism. Never exceed this torque.



NOTE

1. A torque of more than 5 in-lb indicates that the actuator has hit its internal stops. Torque in excess of 10 in-lb may damage the APU inlet door actuator, APU inlet door seal or APU inlet door mechanism. Never exceed this torque.

2. Clearance between inlet door and fuselage should be 1 cm (0.4 In.)

CS1_CS3_4912_006

Figure 17: APU Inlet Door Mechanical Rigging

OMS APU INLET DOOR RIGGING

The door rigging test can be accessed through the onboard maintenance system (OMS) LRU/System operations APU ECU - DOOR RIGGING page.

The electrical door rigging test is accomplished after successful completion of the mechanical door rigging. The test lists preconditions that must be followed. The test can only be carried out with the aircraft on the ground and the APU off.

When the test is carried out, the door moves. Ensure that personnel and equipment are clear of the door before starting the test. Follow the test instructions on the screen. The door position can be monitored on the OMS and the STATUS synoptic page, and should indicate 45° when open, and 0° when closed. Unsatisfactory results require further mechanical rigging.

CS1_CS3_4912_007

Figure 18: Onboard Maintenance System Inlet Door Rigging

APU DOOR LOCKOUT

The APU inlet door lockout pin supports the dispatch of the aircraft with an inoperative APU inlet door actuator or position sensor. The door can be manually driven to the open or closed position from the actuator.

The lockout pin is inserted into the actuator arm to lock the door at 0º to keep the door closed if the APU is not going to be operated, or at 30º if the APU is required for flight.

The APU can be dispatched for flight with the inlet door in either position. If the door is in the open position, the APU must be operating in flight to prevent the APU from windmilling. If the APU is not operated in flight, the airspeed is limited to 250 KIAS.

When the door is locked out in the desired position, the onboard maintenance system (OMS) APU door inhibit is applied.

NOTE

The APU can operate with full electrical and pneumatic load on the ground with the door open at 30°.

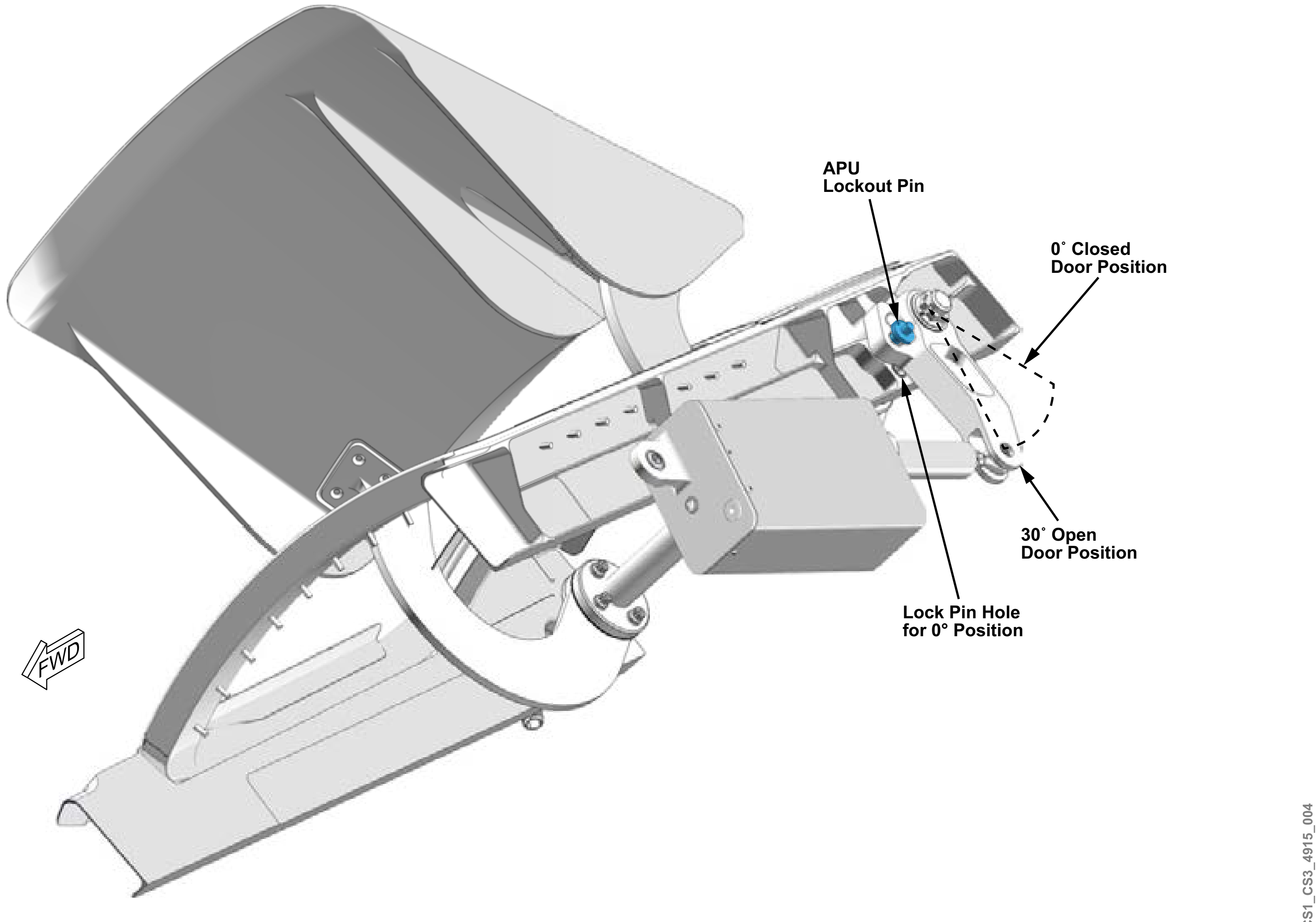


Figure 19: APU Door Lockout

ONBOARD MAINTENANCE SYSTEM APU DOOR INHIBIT

The onboard maintenance system (OMS) LRU/System Operations APU ECU-Inhibit Tests page is used to inhibit the APU inlet door in the open or closed position when the APU inlet door actuator or position sensor is inoperative.

When the APU inlet door has been moved to the desired position, select the applicable door inhibit and then select the Continue soft key. Wait for the TEST OK and either the DOOR SUCCESSFULLY INHIBITED OPEN or the DOOR SUCCESSFULLY INHIBITED CLOSED messages to be displayed.

Reactivation of the APU Inlet Door

To reactivate the APU inlet door, remove the lock pin from the APU inlet door. If the APU inlet door is not closed, manually drive the door to the fully closed position.

On the APU ECU-Inhibit Tests page, select APU DOOR INHIBIT RESET. Confirm the lock pin is removed and select the Continue soft key. Stay on the INHIBIT RESET PAGE for a minimum of five minutes. It is possible that a No response from LRU message may appear. After a successful reset, TEST OK and DOOR INHIBITS SUCCESSFULLY REMOVED messages are displayed.

Failure to remain on the INHIBIT RESET PAGE for a minimum of five minutes can result in a failed reset sequence.

When the reset is complete, do the OMS APU inlet door rigging as per the aircraft maintenance publication (AMP) to confirm the APU inlet door operation.

CAUTION

When the APU inlet door is in the APU DOOR INHIBIT OPEN mode in the OMS, make sure that the door is physically open before an APU start. If the APU is started when the APU inlet door is physically closed while in the APU DOOR INHIBIT OPEN mode, damage to the APU can occur.

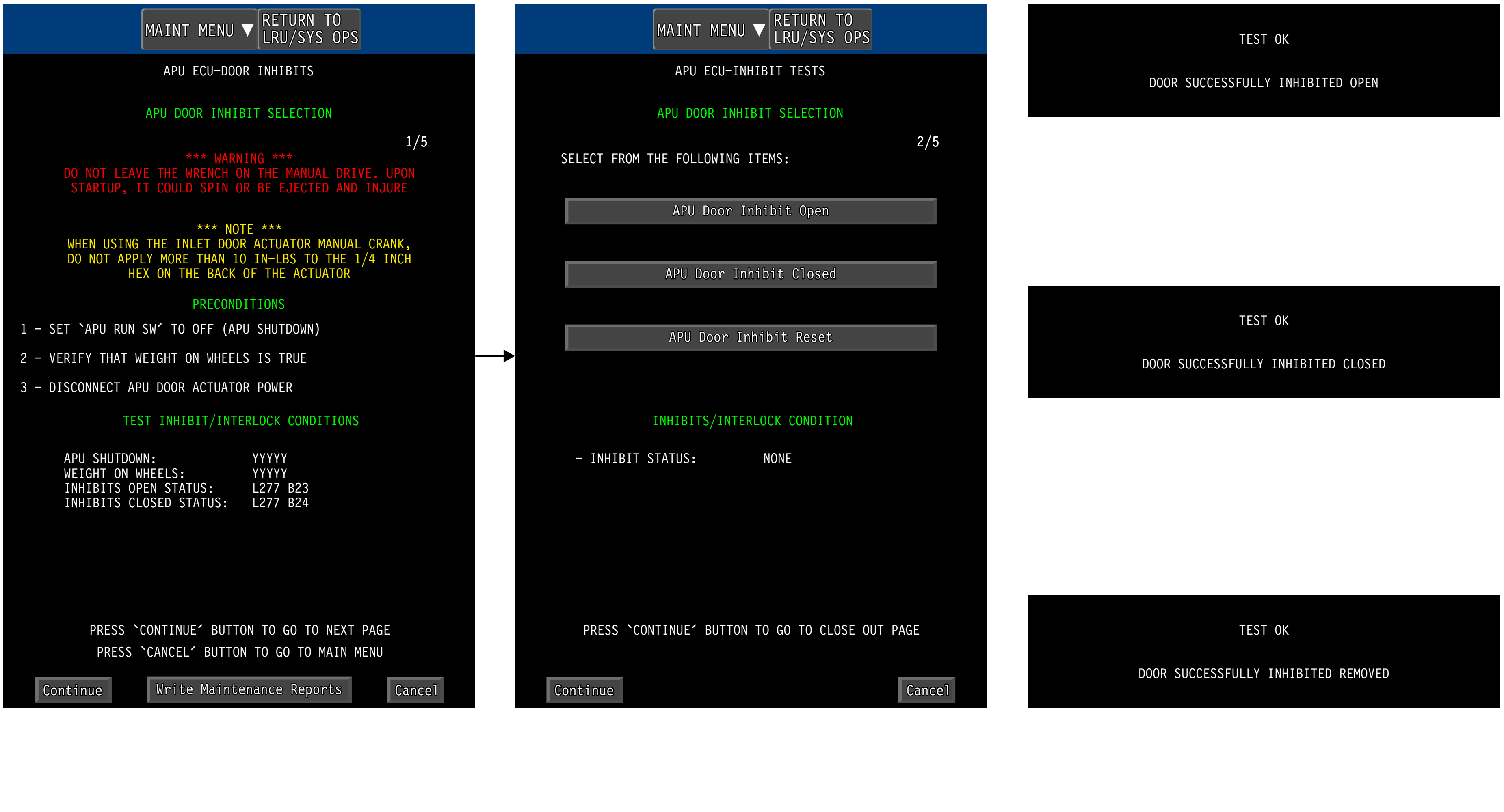


Figure 20: Onboard Maintenance System APU Door Inhibit

49-90 APU OIL SYSTEM

GENERAL DESCRIPTION

Oil is drawn from the gearbox through a protective screen into the supply pump. After being discharged from the pump, the oil is regulated to maintain a constant pressure throughout auxiliary power unit (APU) operation. The oil then flows to the thermostatic pressure-relief valve (PRV) and on to the oil cooler. If the oil temperature is low or the oil pressure is high, the oil bypasses the cooler and is distributed to the engine and components.

The oil is filtered and distributed to the bearings, gears, and generator. Oil is distributed through internal passages and external lines to the various parts of the APU. The oil is monitored for pressure and temperature. If pressure is too low or temperature is too high, the APU shuts down.

The generator is scavenged by elements in the lube module. Oil returning from the generator first passes through inline screens. The pumps discharge oil into a generator scavenge filter and empties it into the reservoir. The generator scavenge and lube module filters have bypass switches that cause the APU to shut down if the filter becomes restricted.

Oil from the forward bearings and gearbox returns to the reservoir by gravity. The turbine bearing area is scavenged by a single pump element. Oil passes through an inlet screen, through the pump and into the reservoir. Air and oil are separated by an air/oil separator. The reservoir is vented to the exhaust section which prevents overpressurization of the reservoir during APU operation.

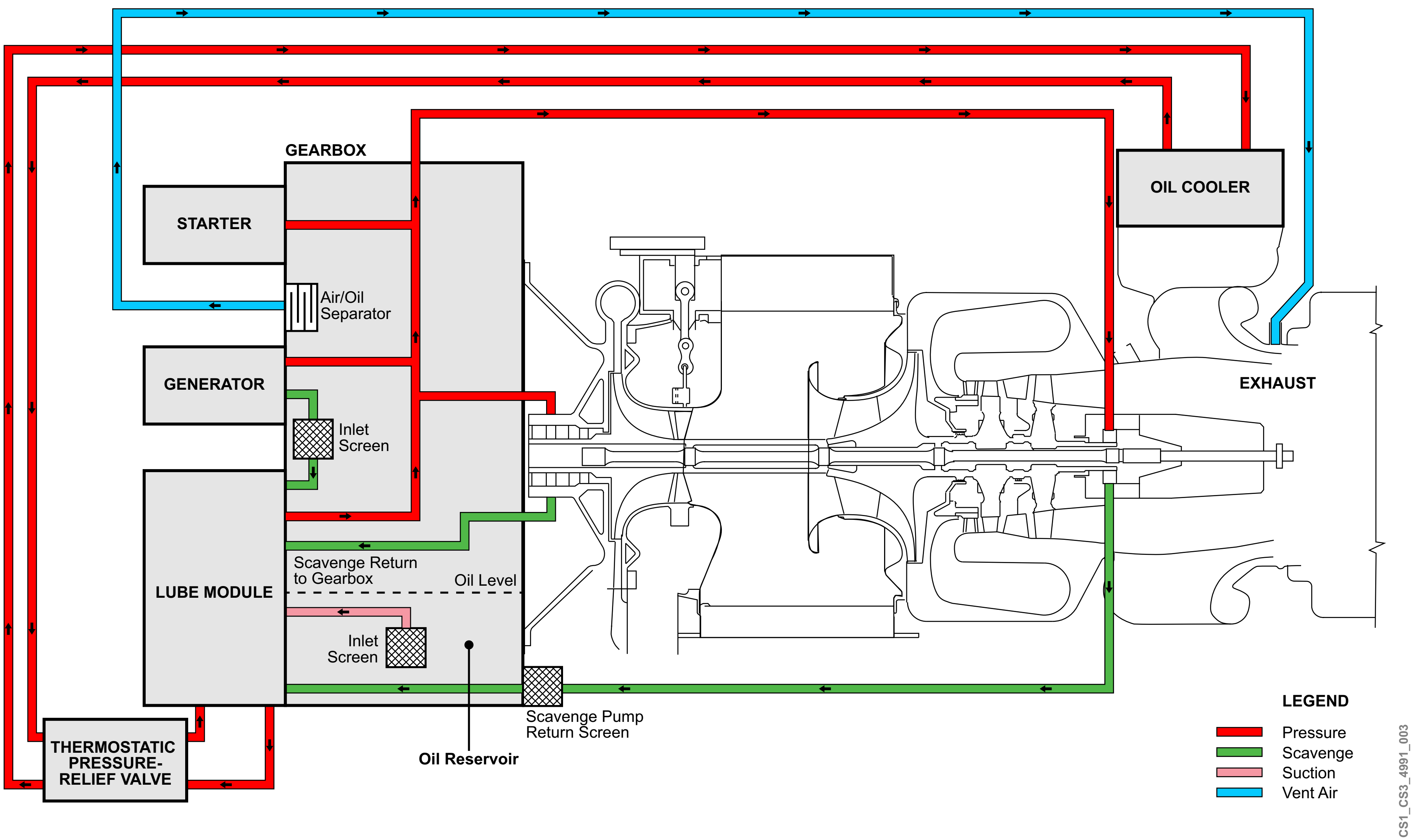


Figure 21: APU Oil System

COMPONENT LOCATION

The oil system components consist of the following:

- Lubricating module
- Oil cooler
- Thermostatic pressure-relief valve
- Low oil pressure (LOP) switch
- Oil level sensor
- Gearbox
- Magnetic chip collector
- High oil temperature sensor
- Air/oil separator

LUBRICATING MODULE

The lube module is installed on the gearbox.

OIL COOLER

The oil cooler is an air/oil heat exchanger mounted on the left side of the eductor.

THERMOSTATIC/PRESSURE-RELIEF VALVE

The thermostatic pressure-relief valve is located on the left side of the gearbox.

LOW OIL PRESSURE SWITCH

The low oil pressure (LOP) switch is located on the front face of the gearbox next to the lubricating module.

OIL LEVEL SENSOR

The oil level sensor is located on the gearbox, aft of the oil servicing door.

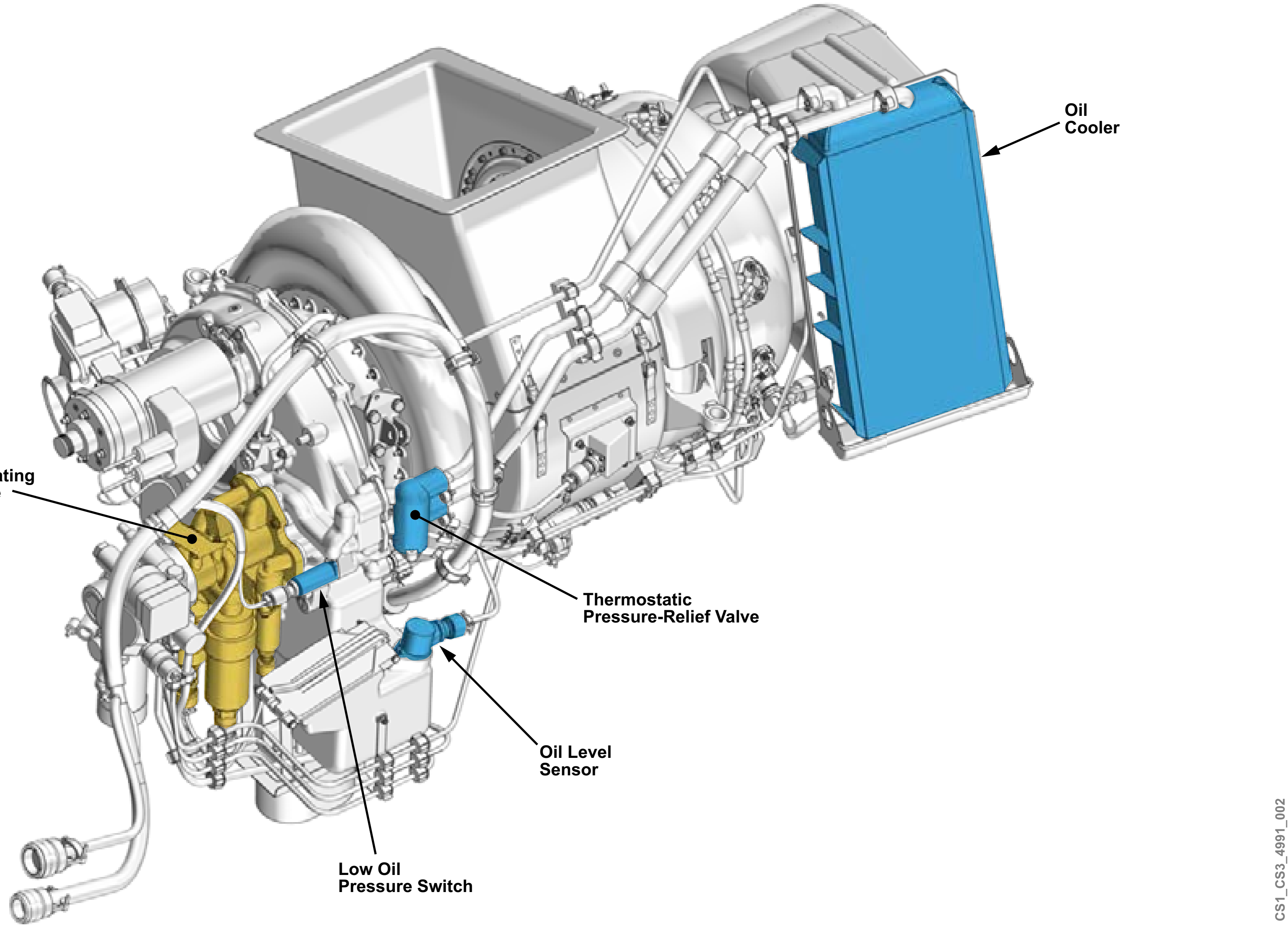


Figure 22: APU Oil Components

GEARBOX

The gearbox is mounted to the APU load compressor section. The gearbox holds the APU oil supply.

MAGNETIC CHIP COLLECTOR

The magnetic chip collector is located on the APU gearbox drain plug.

OIL TEMPERATURE SENSOR

An oil temperature sensor mounted on the gearbox next to the magnetic chip collector.

AIR/OIL SEPARATOR

The air/oil separator is located right above and to the left of the lubrication module.

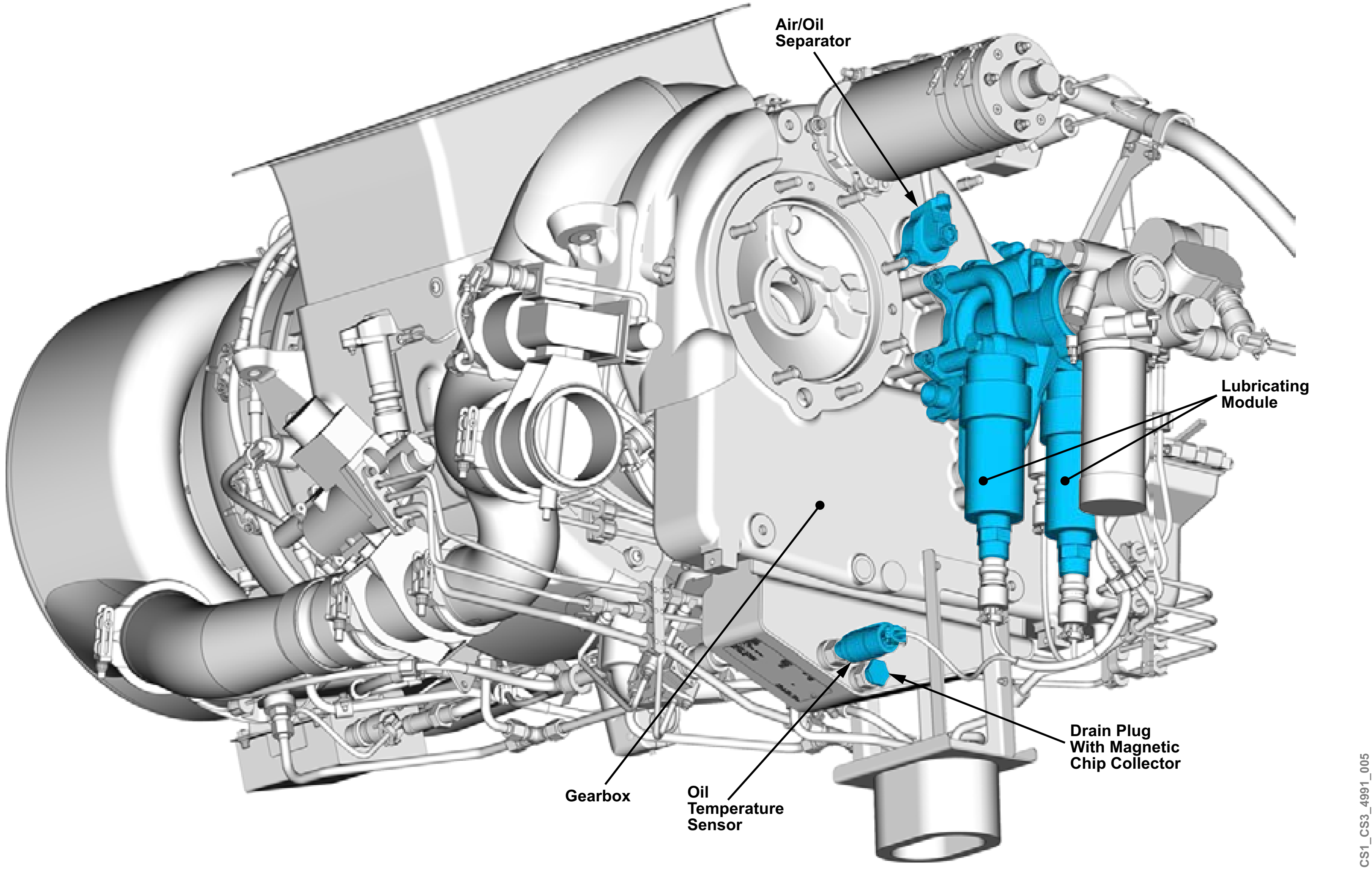


Figure 23: APU Gearbox Oil Components

DETAILED COMPONENT INFORMATION

LUBRICATING MODULE

The lubricating module is a self-contained unit that provides lubrication and scavenge functions to the generator, gearbox, and main shaft bearings.

The lubricating module incorporates a three-element gerotor pressure pump, a three-element gerotor scavenge pump for clearing oil from the generator, and a single element gerotor scavenge pump for clearing oil from the APU turbine bearing cavity.

A pressure regulator maintains a constant lube supply pressure to the engine and generator and a relief valve prevents overpressurization of the oil system.

The module incorporates a seal plate to provide sealing features at each oil passageway. This eliminates the need for external tubes at the lubricating module.

Deprime Solenoid

The deprime solenoid valve is located next to the oil filters on the lubricating module. The deprime solenoid valve opens during cold weather starts to allow air into the oil system. The air helps reduce the drag on the APU during cold start conditions. During APU shutdown, the valve opens to de-oil the lubrication pump.

Filters

There are two filters installed on the lubricating module. The APU oil filter removes contaminants from the oil before it is distributed to the APU oil system. The a generator scavenge filter removes contaminants from the oil coming from the generator. The oil filter elements are throwaway types contained in a housing that is screwed into the lubricating module housing.

Each filter incorporates a filter bypass valve and filter bypass switch to indicate when filter contamination occurs. The filter bypass valve opens when the oil temperature is cold or the filter is clogged. The filter bypass switches signal the ECU when a filter clogs.

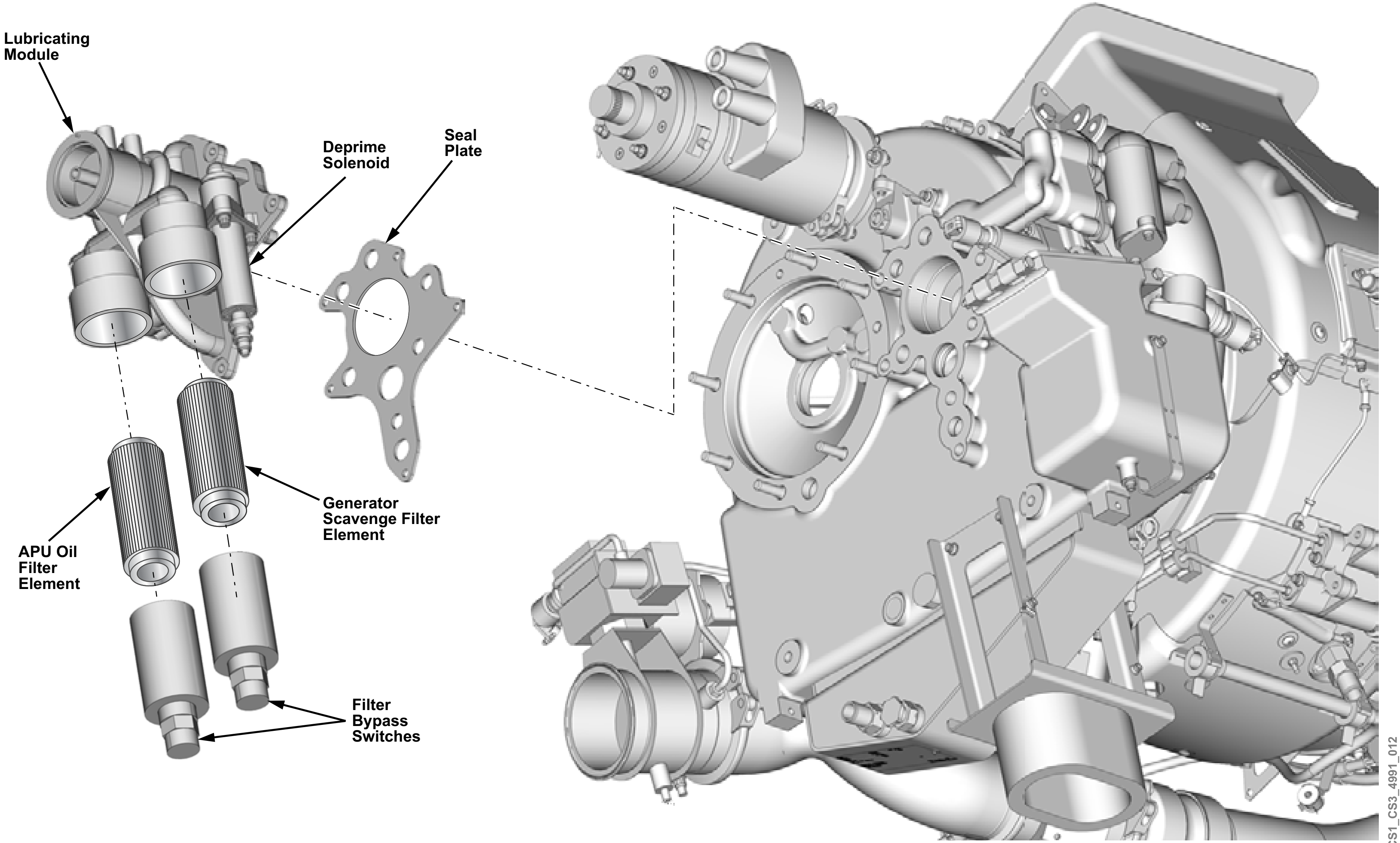


Figure 24: APU Lubricating Module

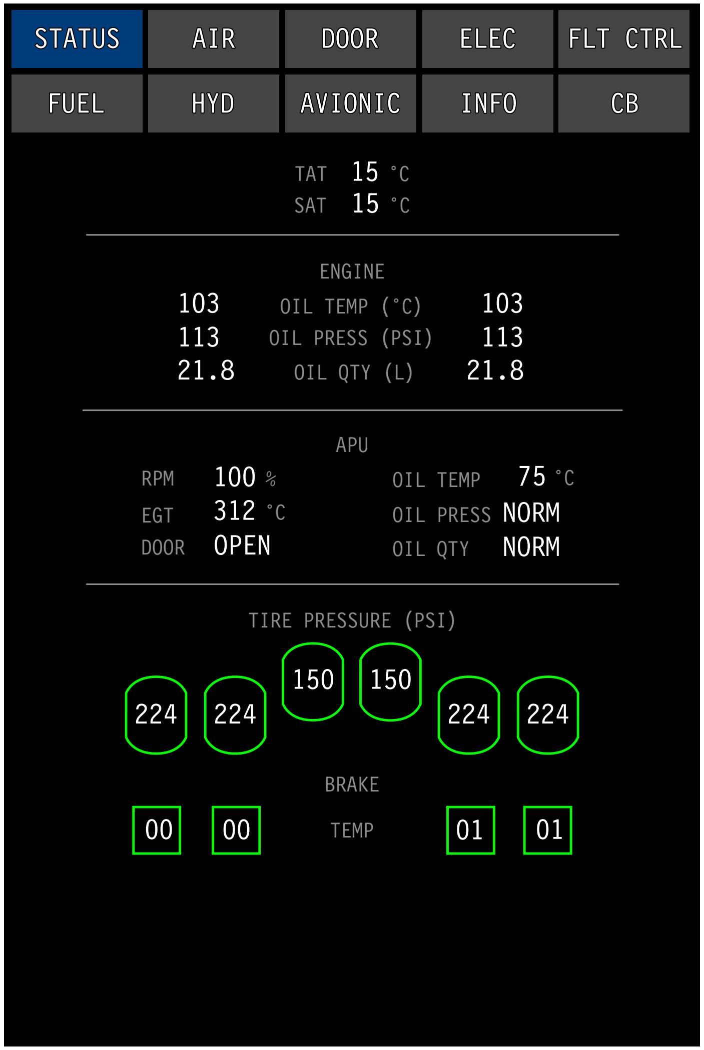
CONTROLS AND INDICATIONS

The APU oil indications are displayed on the STATUS page. The oil temperature, pressure and quantity are monitored by the ECU.

The APU oil pressure shows NORM when the APU is not running.

NOTE

The APU oil pressure indication shows NORM when the APU is off. The APU continues to show NORM as the APU starts, and only indicates LOW if there is insufficient oil pressure when the APU is operating above 34% rpm.



SYNOPTIC PAGE - STATUS

APU OIL TEMPERATURE	
Symbol	**Condition**
32 °C	APU oil temperature in normal range.
144 °C	Oil temperature at or above high oil temperature amber line threshold.
---	Oil temperature invalid.

APU OIL PRESSURE	
Symbol	**Condition**
NORM	Oil pressure in normal range.
LOW	Oil pressure below low oil press threshold.
--	Oil pressure invalid.

APU OIL QUANTITY	
Symbol	**Condition**
FULL	Oil quantity full.
LOW	Oil quantity below low threshold.
NORM	Oil quantity in normal range.
----	Oil quantity invalid.

CS1_CS3_4991_004

Figure 25: APU Status Page Oil Indications

DETAILED DESCRIPTION

Oil is drawn from the gearbox case reservoir through a coarse inlet screen into the lubrication supply pump elements. The deprime solenoid valve improves APU starting after cold soaked conditions. The deprime solenoid valve is energized to open when the APU start command is received, and the oil temperature is less than -6.6°C (20°F). When the deprime solenoid is opened, air is introduced into the lube pump inlet and the lubrication module loses the ability to circulate oil to the APU. This reduces starter motor drag during cold starting. The valve closes at 60% rpm. On a normal or automatic shutdown, the solenoid is energized at 50% to de-oil the oil system.

The oil pressure is regulated to 68 psi. A pressure-relief valve (PRV) opens if the system pressure exceeds 240 psi. The output of the pump is fed to the oil cooler.

The thermostatic pressure-relief valve allows the flow of oil to either flow through or bypass the oil cooler, depending on oil temperature. The thermostatic pressure-relief valve allows oil to bypass the oil cooler at temperatures less than 60°C (140°F). Oil flows through the cooler when temperatures are greater than 77°C (170°F). The valve also opens to bypass the oil cooler when there is a differential pressure of 50 psi across the oil cooler assembly. If the thermostatic pressure-relief valve (PRV) fails to close, a high oil temperature shutdown may occur.

The oil is filtered and then internally distributed within the gearbox to the oil-cooled generator, engine gears, and bearings. An external line delivers oil to the rear turbine bearing. Both the generator scavenge and APU lube oil filters have filter bypass switches to indicate when filter contamination occurs. On the ground, a contaminated filter causes an APU shutdown and an APU SHUTDOWN advisory message on the engine indication and crew alerting system (EICAS). In flight, an APU caution message appears and the APU does not shut down.

After lubricating the generator, oil is scavenged from the generator by the generator scavenge elements, and filtered prior to returning to the gearbox. The APU oil drains directly to the gearbox. The turbine bearing compartment oil returns via a dedicated APU scavenge pump element and drains into the gearbox.

In scavenging the oil from the bearing cavities, some air is drawn with it since the scavenge pump capacity is greater than the oil flow to the turbine cavity. The scavenged air/oil mixture must be separated and the gear case is vented to prevent the buildup of pressure. The air/oil separator removes the air from the oil as it returns to the gearbox reservoir. The air vents overboard through a line going back to the APU exhaust duct.

An oil level sensor, located inside the gearbox adjacent to the oil level sight glass is used to check the APU oil level when the APU is not running.

A low oil pressure (LOP) switch monitors the oil flow to the engine bearings. The LOP switch provides protection against low oil pressure conditions. The LOP is a normally closed switch that is checked during prestart and self-test ECU modes. During a prestart built-in test (BIT), the LOP switch has failed if the APU speed is less than 7% and the switch is open. If the LOP switch has failed during the prestart BIT, the APU can start and run but has no LOP protection.

An oil pressure auto-shutdown occurs if the APU is on speed, and the oil pressure is less than 35±5 psi for 20 seconds.

The oil temperature sensor provides indication, high oil temperature protection, and activation of the deprime solenoid. The normal operating oil temperature is approximately 96°C (205°F) on a standard sea level day. A high oil temperature shutdown occurs when the APU is on speed and the oil temperature is greater than 143°C (290°F) for 10 seconds.

A magnetic chip collector collects debris as the oil returns to the gearbox.

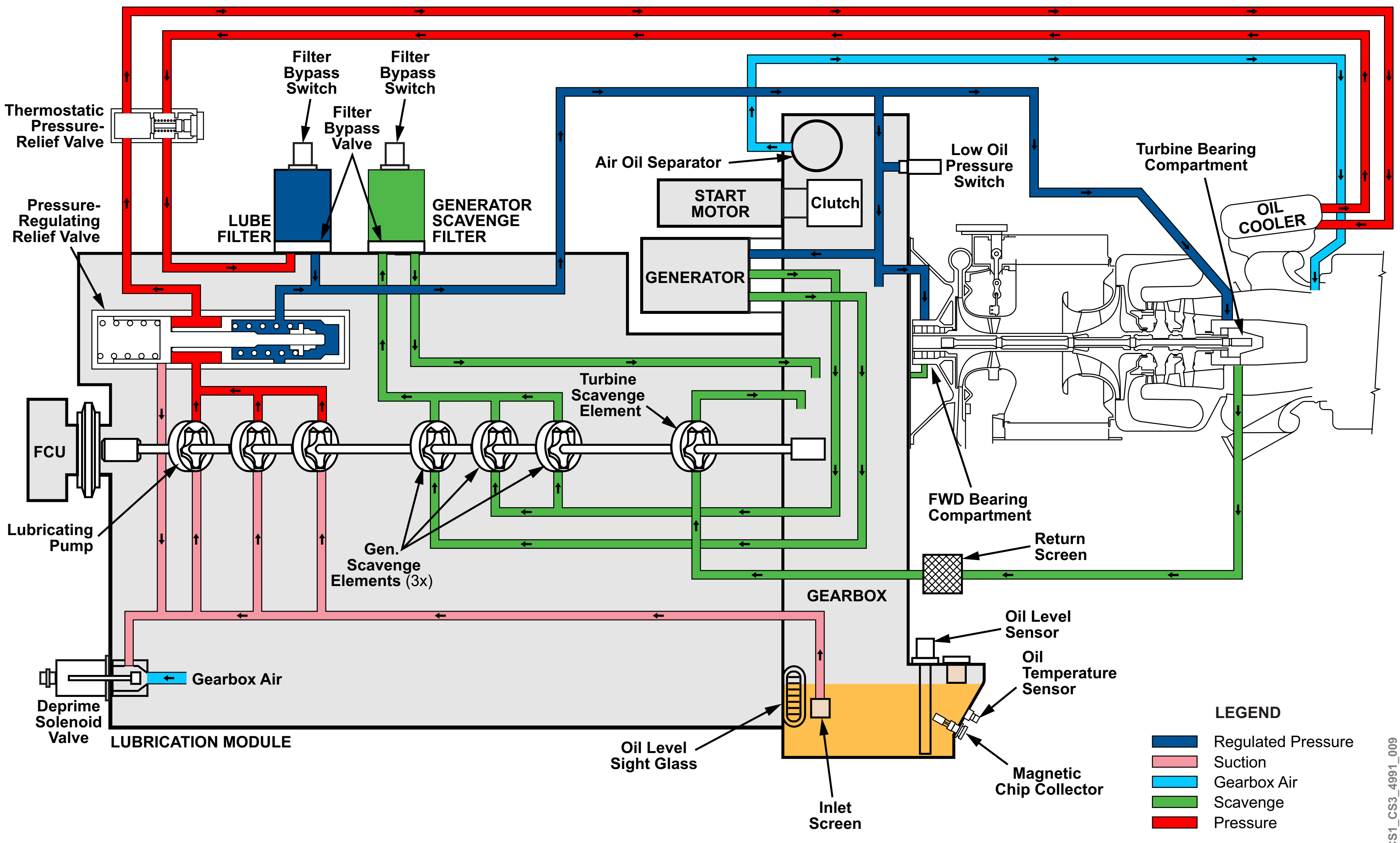


Figure 26: APU Oil System Schematic

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) messages for the APU oil system.

CAS MESSAGES

Table 3: Advisory Message

MESSAGE	LOGIC
APU OIL LO QTY	APU oil low quantity detected.

PRACTICAL ASPECTS

OIL SERVICING

The gearbox oil supply provides 700 hours of APU operation without oil servicing. The gearbox sump capacity is 7.32 L (7.74 qt).

The APU oil servicing is accomplished when the APU access doors are open. The oil quantity can be checked using the sight glass on the side of the gearbox. The sight glass is located on the aft side of the gearbox near the oil service door. The oil level should be checked 30 minutes after shutdown.

NOTE

If the oil level is in the ADD mark, oil must be added to the APU gearbox no later than 30 hours of APU operation.

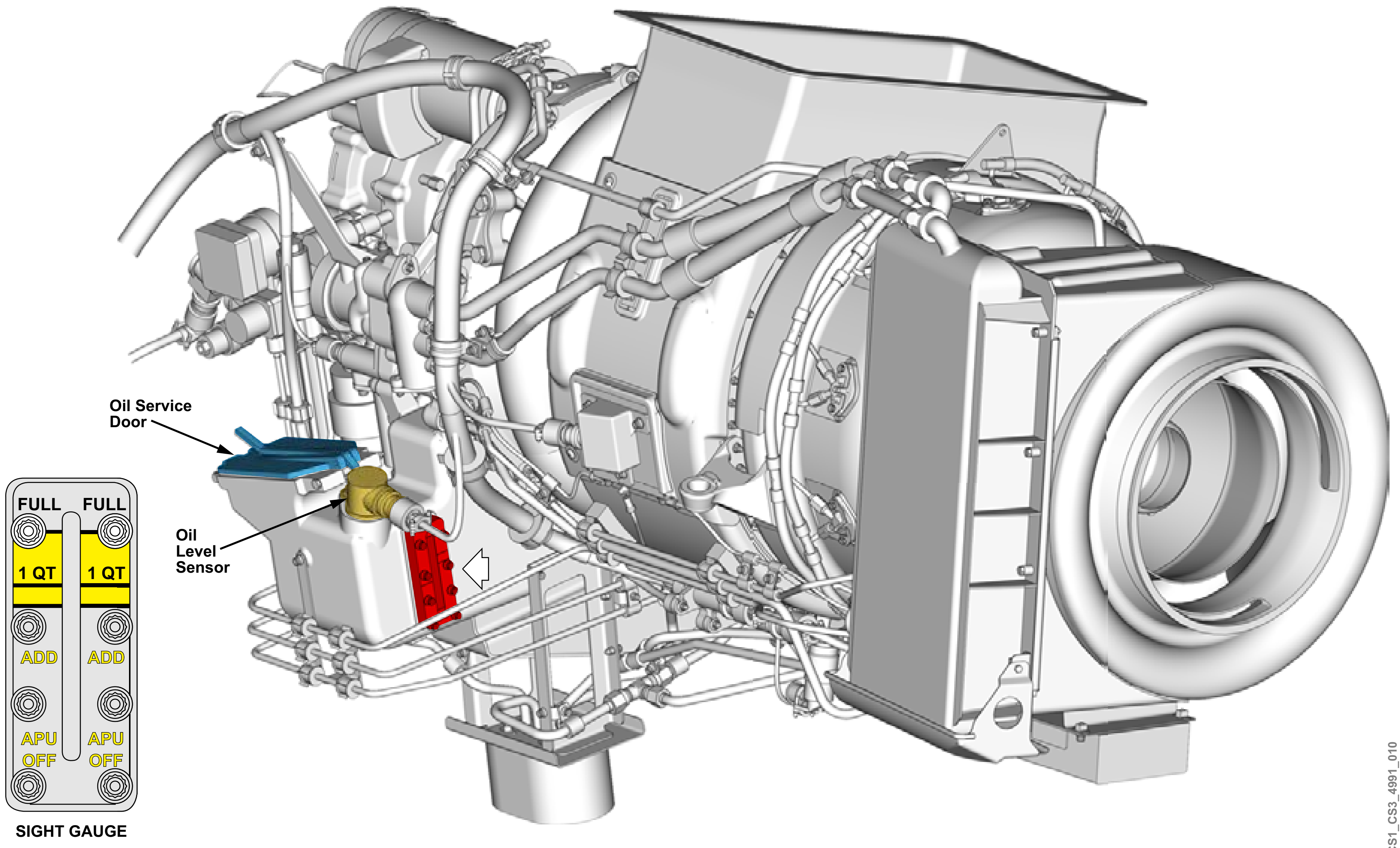


Figure 27: APU Oil Servicing

49-30 APU FUEL SYSTEM

GENERAL DESCRIPTION

The aircraft fuel system supplies the auxiliary power unit (APU) with low-pressure fuel. Fuel enters the fuel control unit (FCU) through the fuel shutoff solenoid valve and is filtered, pressurized, and metered. The FCU supplies metered fuel to the APU for combustion, and high-pressure fuel to operate the surge control valve (SCV) and inlet guide vane (IGV) actuator.

The metered fuel flows from the FCU to the fuel flow divider when the fuel shutoff solenoid valve in the FCU is commanded open by the electronic control unit (ECU).

The flow divider distributes the fuel between primary flow and secondary fuel manifolds. The primary flow is used for initial starting of the APU. The secondary flow augments the primary flow for acceleration and on-speed operating conditions. The flow divider solenoid is controlled by the ECU.

Ten fuel nozzles inject fuel into the combustion chamber for APU operation.

When the ECU receives an OFF signal and the APU has completed a cool-down cycle, the fuel shutoff solenoid valve closes and the APU shuts down.

On the bottom of the outer combustion case is a plenum drain orifice. It drains the excess fuel from the combustion chamber in case of a no start condition.

NOTE

The plenum drain valve is always open, whether the APU is operating or not. An arrow on the valve body points in the direction of fuel flow toward the overboard drain line.

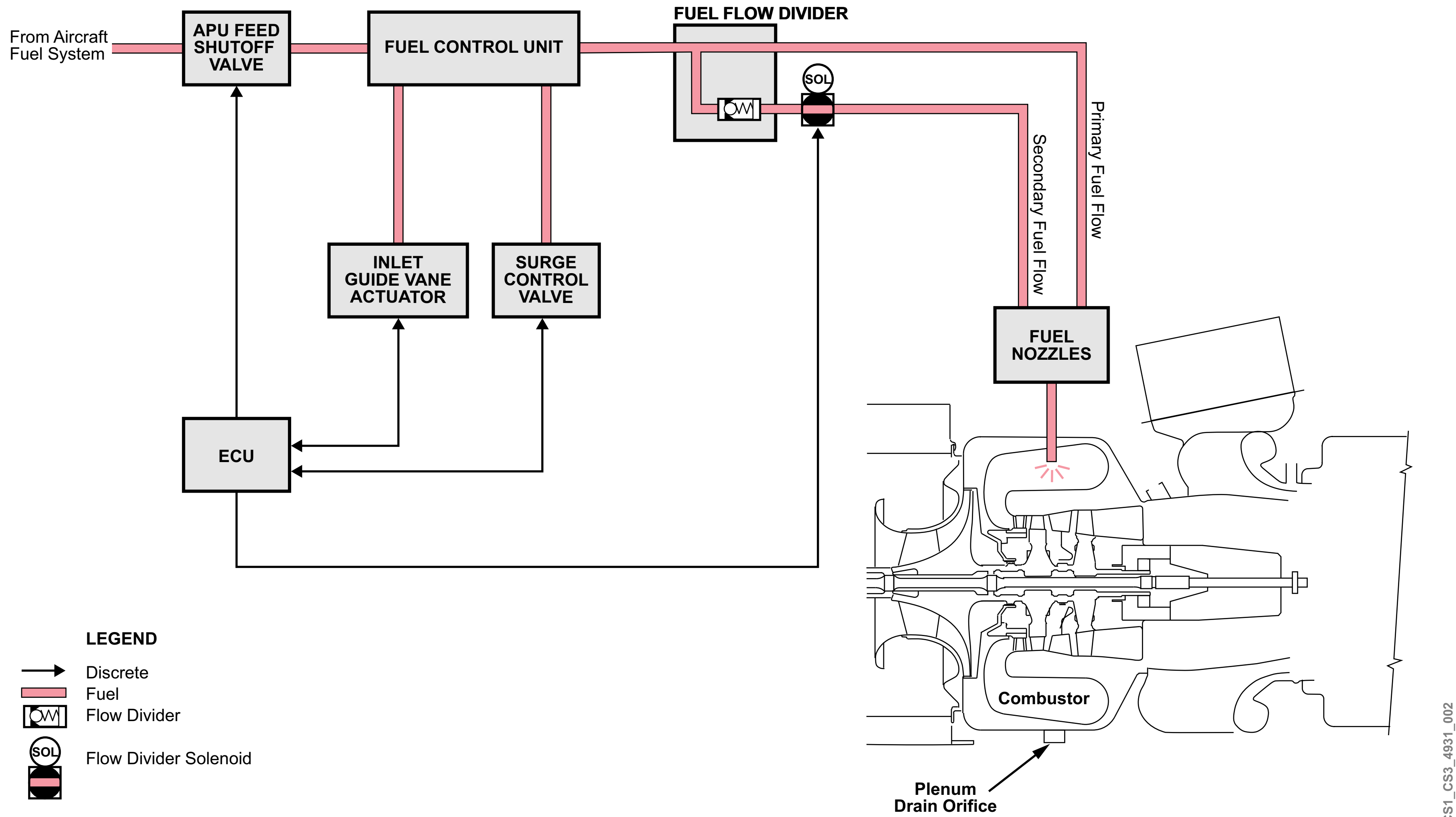


Figure 28: APU Fuel System

COMPONENT LOCATION

The fuel system consists of the following components:

- Fuel control unit
- Flow divider assembly
- Fuel manifolds and fuel nozzles
- Plenum drain orifice

FUEL CONTROL UNIT

The fuel control unit (FCU) is attached to the lubricating module by means of a quick-release v-groove clamp, and is driven by an oil lubricated splined drive shaft.

FLOW DIVIDER ASSEMBLY

The flow divider is mounted on the lower left side of the turbine plenum.

FUEL MANIFOLDS AND FUEL NOZZLES

The primary and secondary fuel manifolds surround the combustor plenum.

PLENUM DRAIN ORIFICE

The plenum drain orifice is located at the lowest point on the plenum.

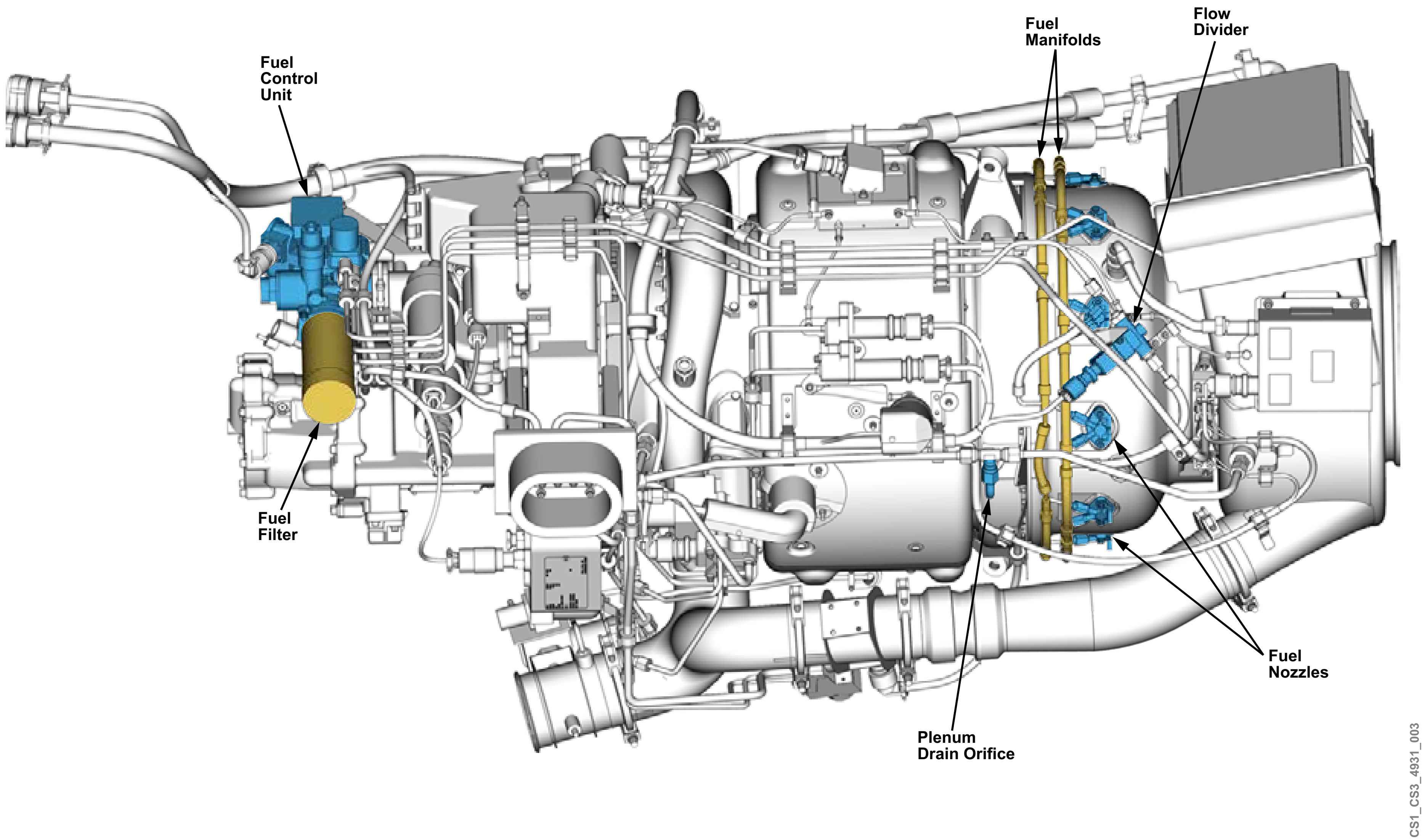


Figure 29: APU Fuel Control Unit, Fuel Manifold, Flow Divider, Fuel Nozzles, and Plenum Drain Valve

DETAILED COMPONENT INFORMATION

FUEL NOZZLES

There are 10 dual-orifice fuel nozzles equally spaced and mounted on the combustor plenum. The primary and secondary fuel manifolds route fuel to the fuel nozzles. The fuel nozzles inject metered fuel into the combustor.

The primary and secondary fuel inlets have screens to prevent contaminants from entering the spray tip.

A locating pin positions the air shroud on the nozzle to ensure proper installation in the engine.

Fuel for ignition and initial acceleration is supplied by the primary nozzles. The secondary nozzles provide the additional fuel flow as required for operation up to 100% rpm.

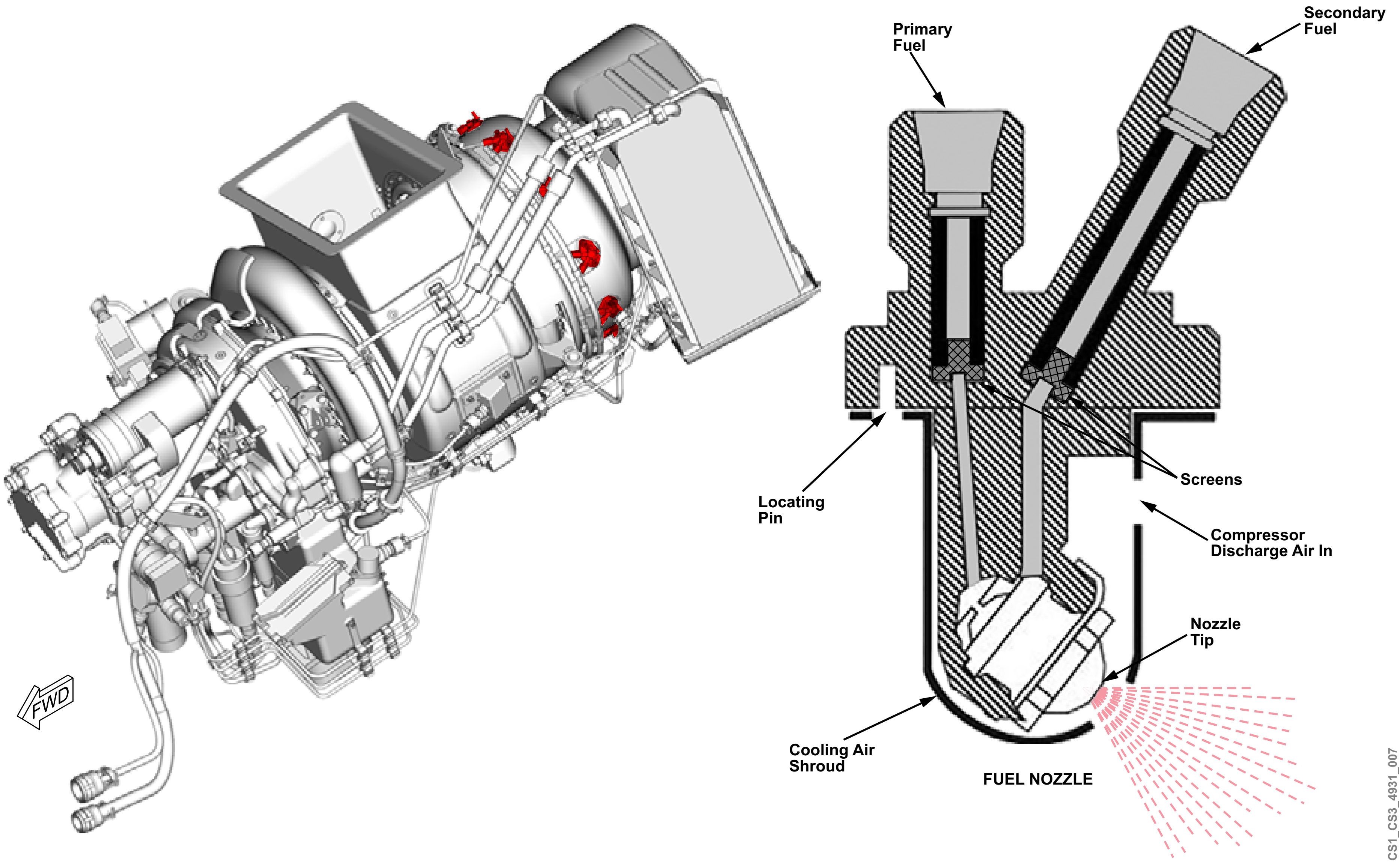


Figure 30: Fuel Nozzles

DETAILED DESCRIPTION

FUEL CONTROL UNIT

The fuel control unit is attached to the lubrication module by means of a quick release v-band clamp. The FCU is driven by an oil lubricated spline drive shaft.

The APU fuel system is automatically controlled by the electronic control unit (ECU). The aircraft fuel system supplies the APU with low-pressure fuel from the left engine feed line. Fuel enters the FCU through the inlet filter. The filters trap normal fuel pump wear debris. The entire pump discharge flow passes through the filter and feeds a gear pump that supplies the actuator pressure regulator and the metering valve through the high-pressure fuel filter.

A high pressure-relief valve protects the FCU. The relief valve opens at 1150 psi and also relieves pump pressure on all FCU shutdowns.

The pressure regulating valve provides 250 ± 25 psi hydraulic pressure used to activate the inlet guide vane actuator and the surge control valve.

The fuel metering valve meters fuel flow using a torque motor. The metered fuel is a function of ECU current signal. An increase in current increases fuel flow. Whenever a protective shutdown is initiated, the current to the torque-motor is automatically removed.

At 7% rpm, the fuel shutoff solenoid allows fuel to flow to the flow divider and flow divider solenoid. When the solenoid is de-energized, APU operation is terminated.

NOTE

If the fuel shutoff solenoid fails to open at 7% speed, a protective shutdown occurs. A leaking fuel shutoff solenoid that allows fuel into the APU before 7% speed causes torching.

A temperature sensor signals the ECU to adjust flow rates for starting the APU in cold conditions and at high altitude. A flow meter resolver provides fuel flow feedback to the ECU. Once rated speed is reached, fuel flow is modulated to meet the demands of varying pneumatic and electrical loads.

When the ECU receives an OFF signal and the APU has completed a cooldown cycle, power is removed from the fuel shutoff solenoid and the APU shuts down. Whenever a protective shutdown is initiated, the signal to the FCU torque motor is automatically removed as well.

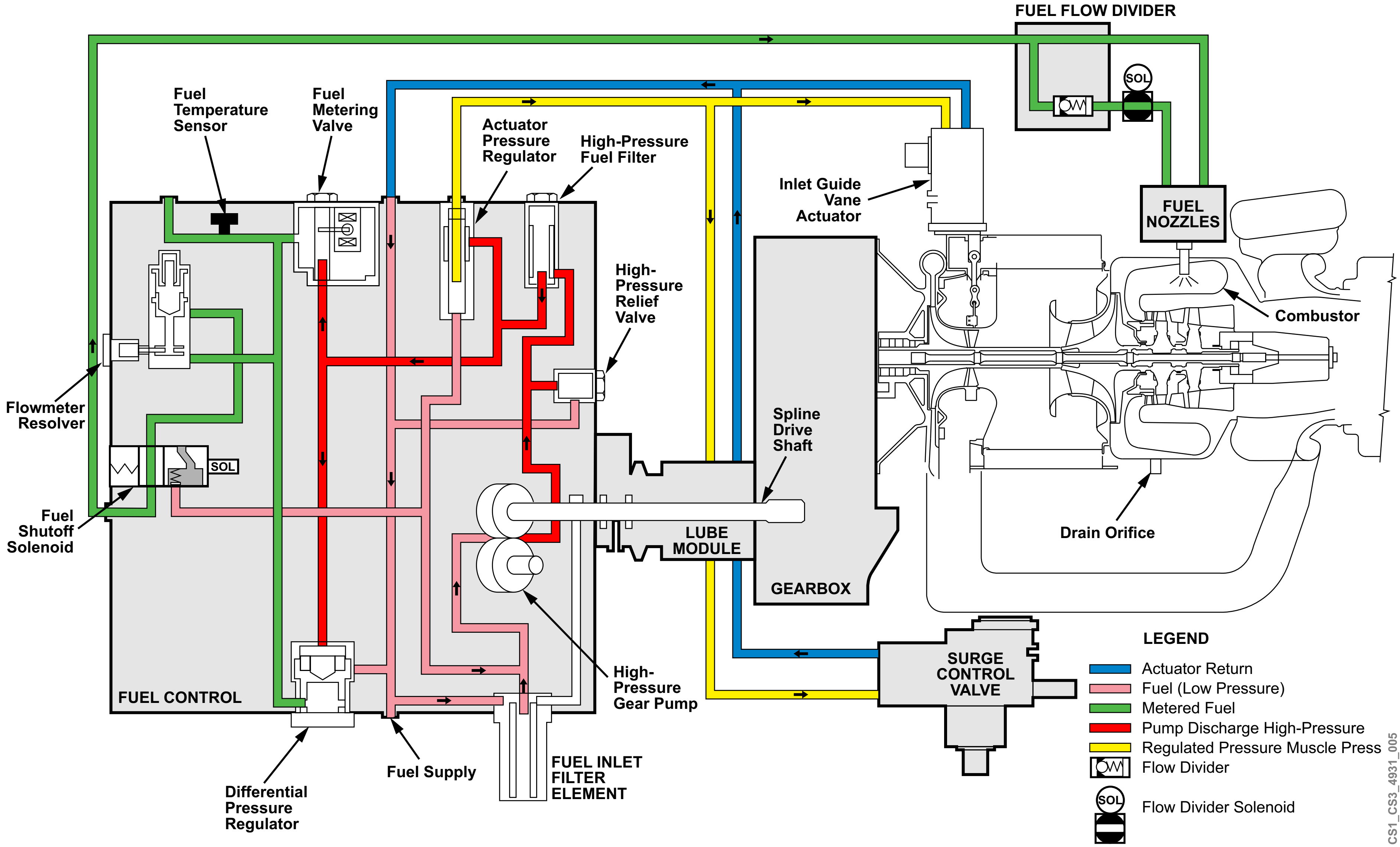


Figure 31: APU Fuel Control Unit Schematic

FUEL FLOW DIVIDER AND FUEL FLOW DIVIDER SOLENOID

The flow divider and flow divider solenoid distribute the fuel between primary flow and secondary fuel manifolds. The flow divider solenoid also improves start capability in cold weather. Fuel flows to the primary manifold for APU light-off and initial acceleration. The ECU uses P2, T2, and speed signals to control the fuel flow divider solenoid valve.

During an APU start on the ground or at low altitude, the normally open flow divider solenoid is energized closed until 30% rpm. The solenoid also energizes closed when the APU is operating above 23,000 ft or T2 less than 13°C (55°F). This allows the primary fuel manifold pressure to increase above 125 psi and provide better atomization for increased combustion performance.

Fuel flow to the secondary manifold is controlled by the flow divider check valve that opens at 120 psi (25-40% rpm). Incorrect flow divider sequencing results in an APU protective shutdown.

The flow divider assembly is a line replaceable unit.

NOTE

If flow divider sequencing is incorrect, no acceleration, no flame, overtemperature and underspeed, auto-shutdowns could occur.

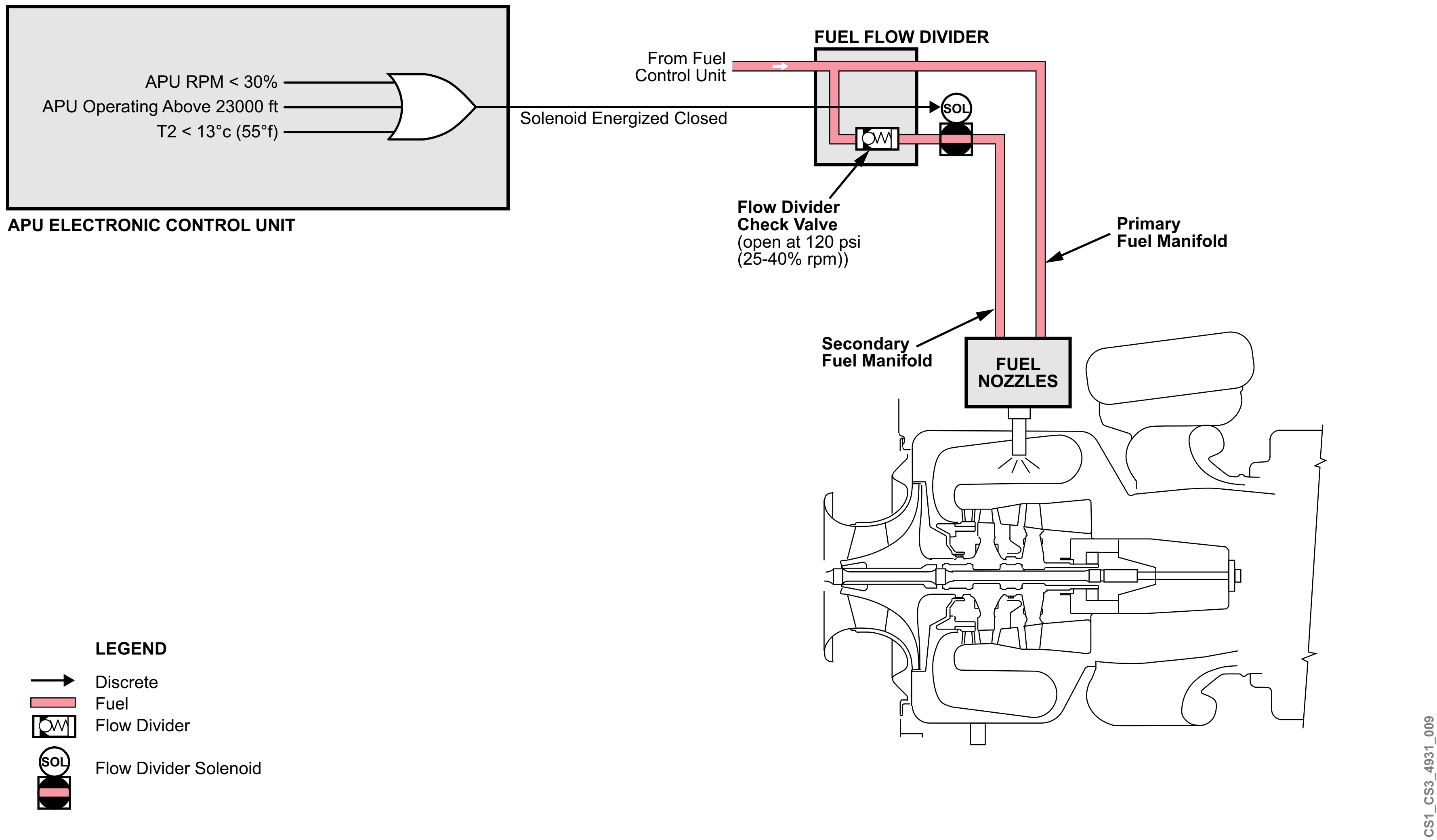
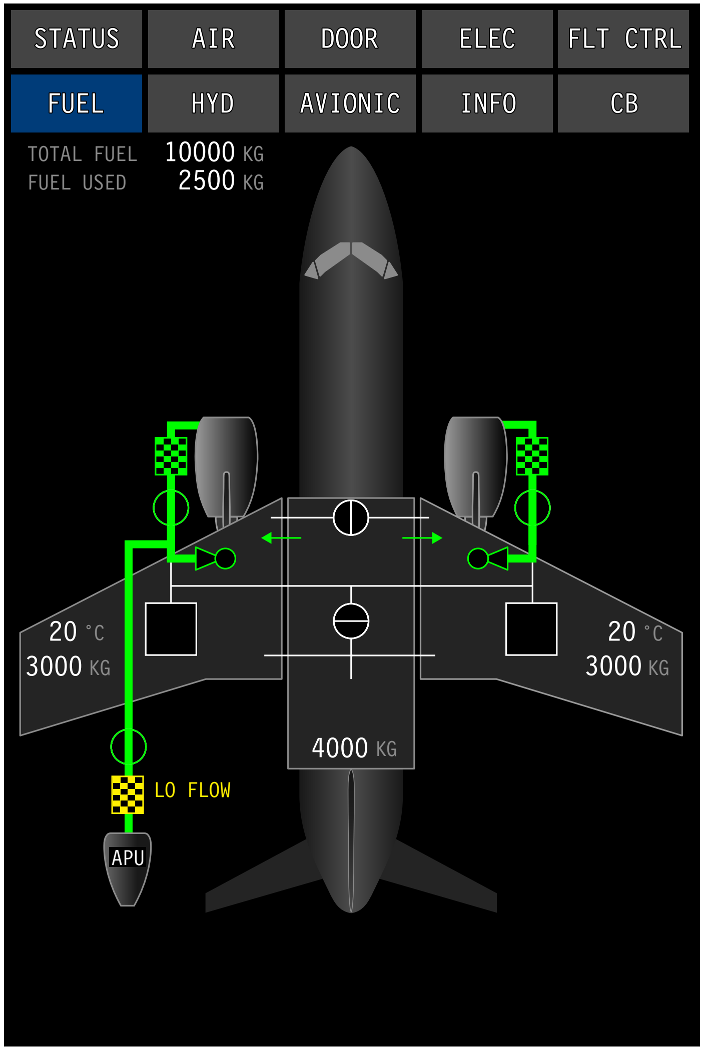


Figure 32: APU Fuel Flow Control

CONTROLS AND INDICATIONS

The APU actual fuel flow is compared to the ECU commanded fuel flow. If a disagreement occurs, the LO FLOW indication on the FUEL synoptic page appears and the filter symbol turns from green to yellow.



SYNOPTIC PAGE - FUEL

CS1_CS3_4931_006

Figure 33: Fuel Low Flow Indication

49-40 APU START AND IGNITION SYSTEMS

GENERAL DESCRIPTION

The auxiliary power unit (APU) starting and ignition system accelerates the APU to the starting speed and ignites the fuel/air mixture in the combustor.

The ECU provides a start command to BPCU 2 when the APU panel START switch is moved to START and the APU inlet door has reached the open position. An APU start uses electrical power supplied by TRU 3 via the TRU start contactor (TSC) or BATT 2 via the battery start contactor (BSC).

Once a power source is selected, BPCU 2 closes the APU start contactor (ASC) and the APU starter motor drives the APU through a sprag clutch mounted on the gearbox.

The APU start request terminates at 50% rpm and power is removed from the starter. The APU continues to accelerate to a self-sustaining speed.

The ignition system provides electrical energy to create a spark at the igniter plug for starting the APU. The ECU controls and powers the ignition system. The ignition system is energized once the start request is made and terminates at 60% rpm. The ignition unit delivers 18,000 V to the igniter plug.

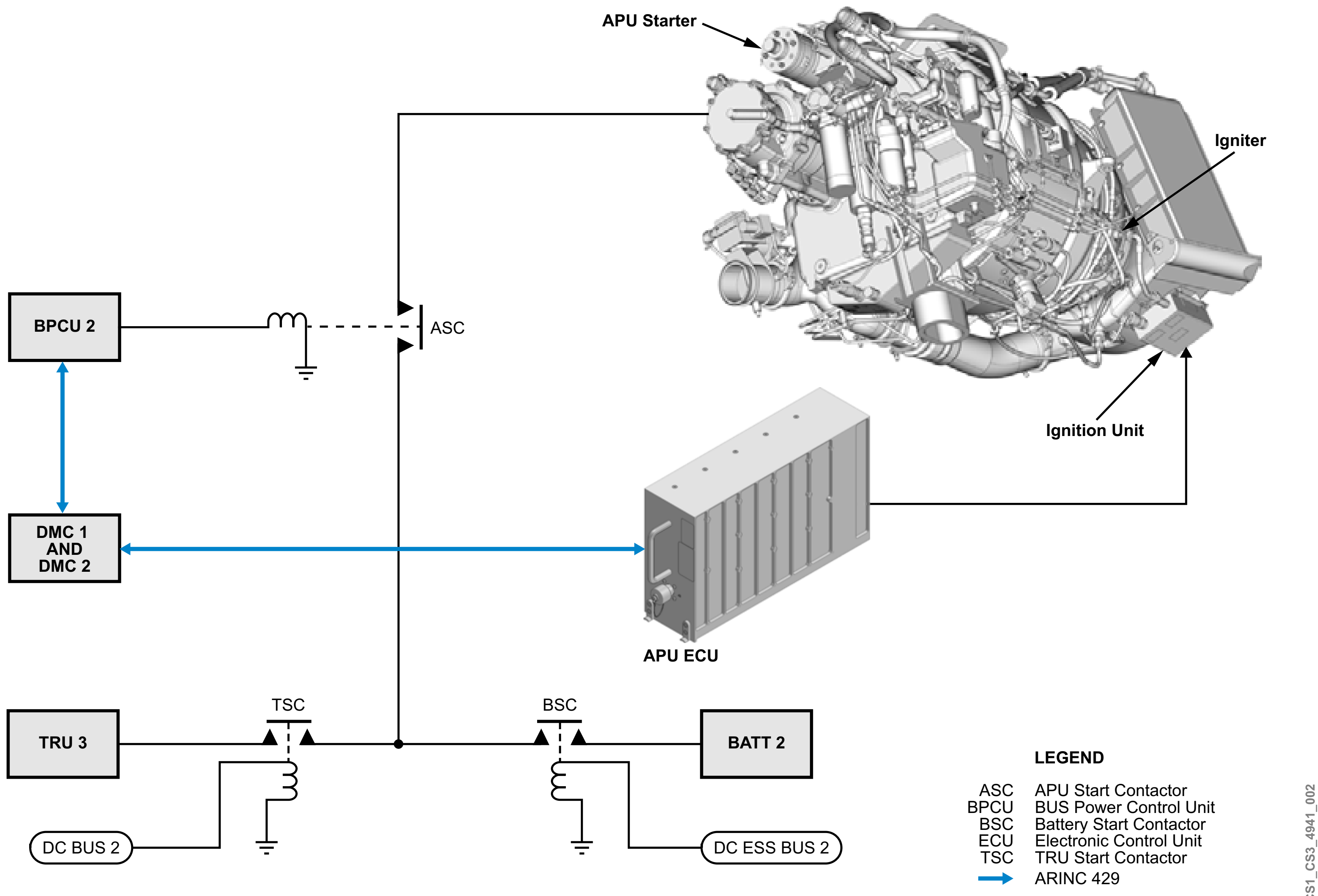


Figure 34: APU Starting and Ignition

COMPONENT LOCATION

The starting and ignition system consists of the following components:

- Starter motor
- Ignition unit
- Igniter plug and igniter lead

STARTER MOTOR AND CLUTCH ASSEMBLY

The starter motor and clutch assembly is installed on the gearbox, above the generator.

The DC starter motor has a brush wear indicator pin. The brush wear indicator pin must be seen through the side window of the indicator housing. If any part of the white indicator pin is visible, the brushes are acceptable for continued use.

IGNITION UNIT

The igniter unit is located on the bottom of the APU, below the oil cooler.

IGNITER PLUG AND IGNITER LEAD

The igniter plug is installed on the left side of the combustion section.

A single air-gap-type igniter provides ignition for the APU. The igniter consists of an center electrode, an insulator, and an outer shell. The outer shell has cooling holes to aid in reducing tip temperatures. An igniter lead connects the igniter to the ignition unit.

CAUTION

Voltage produced by the ignition system is lethal. Caution must be observed when working with the system.

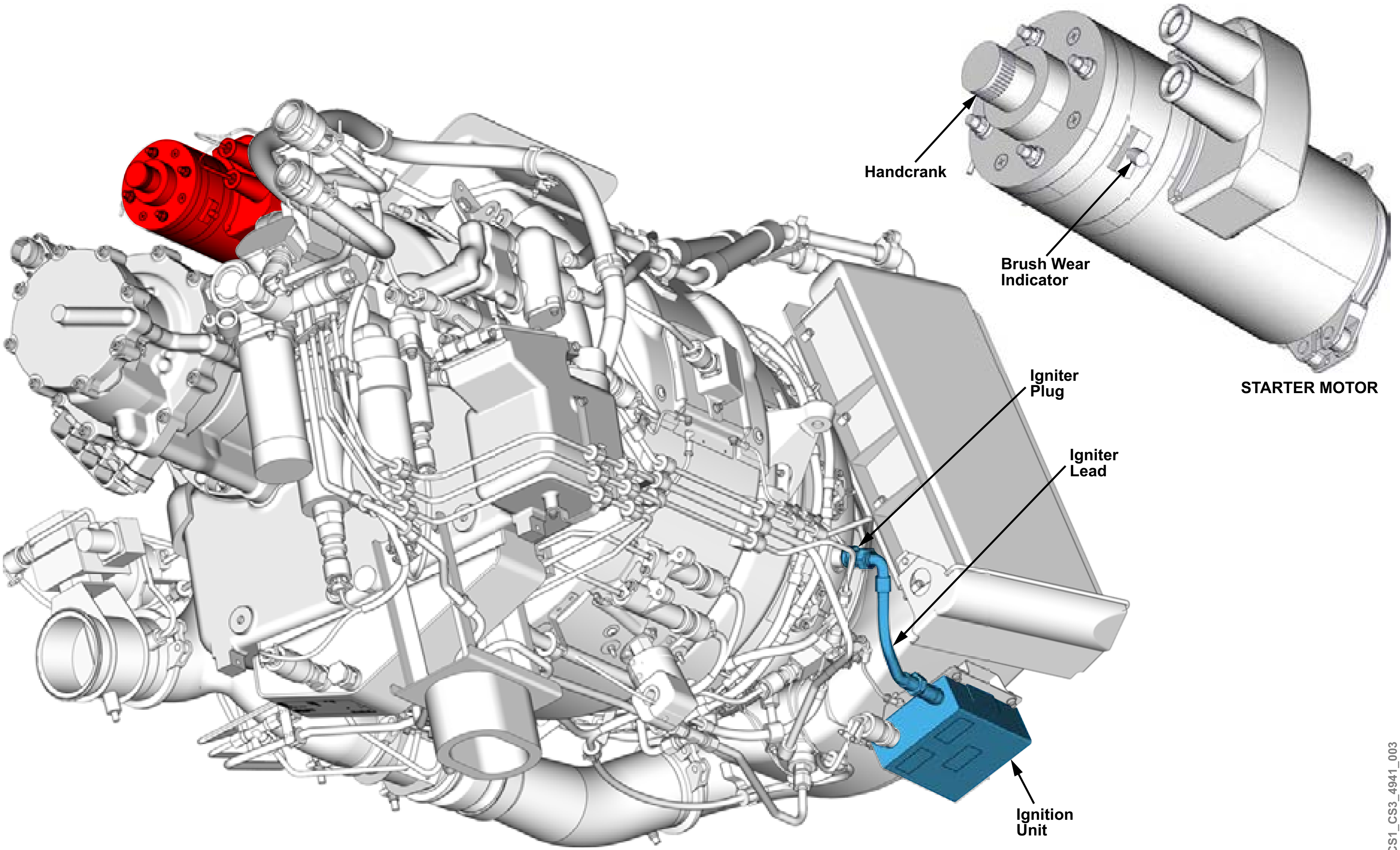


Figure 35: APU Starting and Ignition Starter Motor, Ignition Unit, Igniter Plug, and Igniter Lead

DETAILED DESCRIPTION

APU START POWER

The APU can be started using transformer rectifier unit (TRU) 3 or BATT 2. A TRU APU start has priority over a battery APU start if there is no TRU failure in the system.

The APU starter motor power is determined by the electrical system configuration. The electrical system applies power to the APU starter motor when BPCU 2 receives an APU start request from the APU ECU. BPCU 2 configures the electrical system for start. The ECU monitors the voltage available to the starter. If an open circuit is detected, a maintenance message is generated.

The APU start request discrete signal sent by the APU ECU becomes inactive during a normal start when APU speed exceeds 50% or if the ECU detects an aborted start condition. BPCU 2 aborts and inhibits the APU start function if the start request from the ECU is not terminated after 60 seconds.

Two failed APU start attempts followed by one successful attempt are permitted during normal operation. Do not perform more than three starts/start attempts in one hour. A two-minute delay must be observed between start attempts. A cooldown period is required for the TRU after three failed starts. The battery must cooldown and recharge after three failed start attempts. If any power contactor failure is detected, or there is an AC power transfer, the APU start is aborted.

TRU START

APU starts from TRU 3 is inhibited if any TRU is not powered.

For a TRU APU start, BPCU 2 commands CDC 4 via CDC 2 to close the TRU start contactor (TSC). CDC 4 commands the TSC coil to close. CDC 4 monitors and sends the status of the TSC contactor to BPCU 2. Following confirmation the TSC is closed, BPCU 2 closes the APU start contactor (ASC).

The BPCU commands the ASC open and commands CDC 4 to open the TSC when the APU start from TRU 3 is terminated or aborted.

BATTERY START

An APU start from battery occurs when both batteries are available and the TRU 3 start is inhibited.

Two batteries are in use during the APU starting interval. Battery 2 is dedicated to the APU starter, and battery 1 powers the aircraft.

An APU start from battery 2 is inhibited when it is the only source of power on the aircraft or any associated APU start power contactors are failed.

For a battery start, BPCU 2 commands battery line contactor (BLC) 2 to open. Following confirmation the BLC 2 status is open, BPCU 2 requests CDC 2 to close the BSC. CDC 2 monitors the status of BSC contactor and reports it to BPCU 2. When the BSC is confirmed closed, BPCU2 commands the ASC to close.

When the APU start from battery 2 is terminated or aborted, the BPCU commands the ASC open and commands CDC 2 to open the BSC.

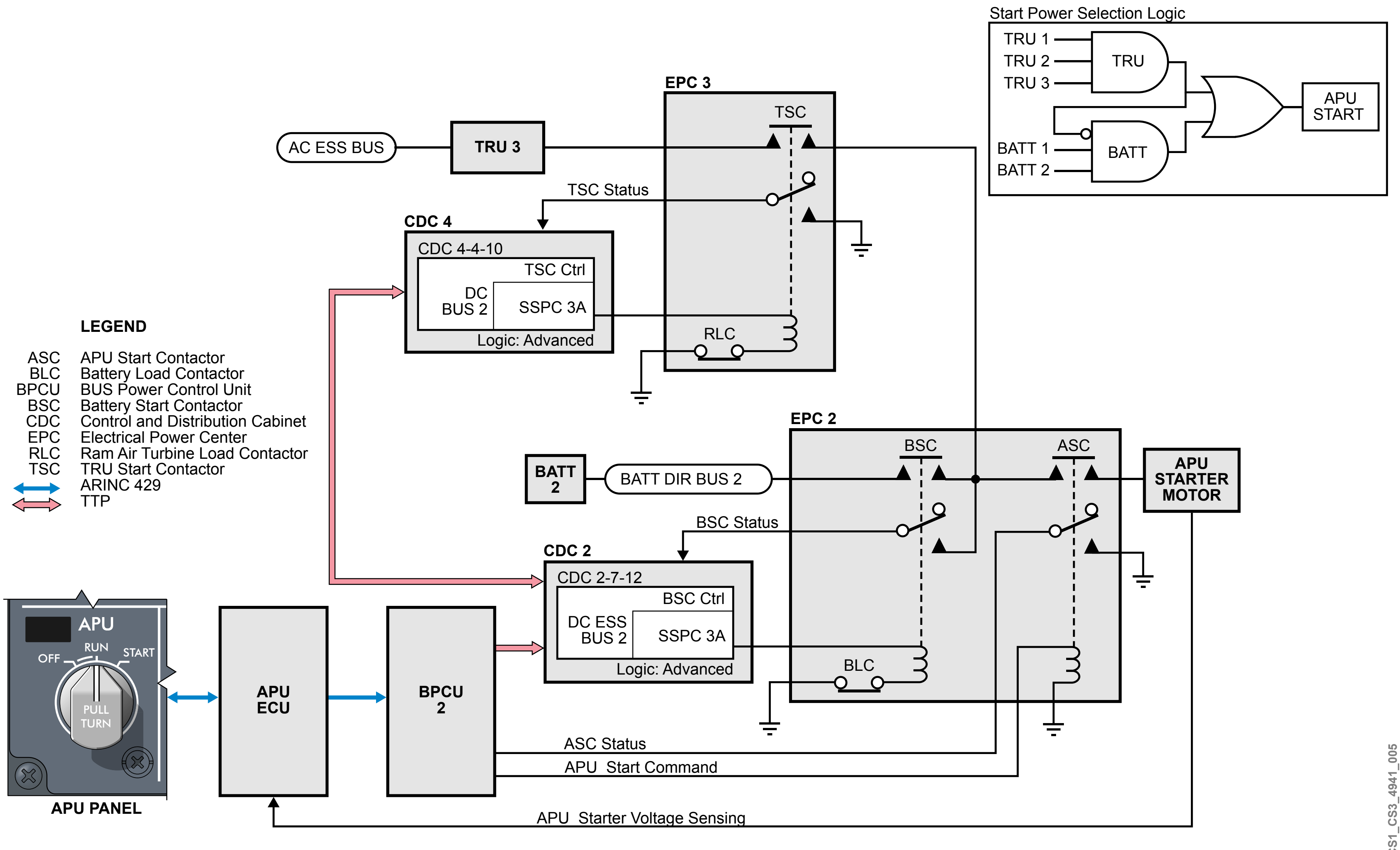


Figure 36: APU Start Power

49-50 APU AIR SYSTEMS

GENERAL DESCRIPTION

The auxiliary power unit (APU) supplies compressed air for air conditioning and main engine starting from the load compressor. The pneumatic output is through the bleed air valve (BAV).

The load compressor takes air from the same inlet as the power section compressor. It compresses the air and supplies it to the aircraft through a bleed air valve. The bleed air valve shuts off the air flow from the load compressor when there is no demand placed on the APU by the aircraft pneumatic system.

The load compressor provides the maximum amount of air flow that the aircraft demands. Inlet guide vanes (IGVs), located between the load compressor and the air inlet, match the compressor flow to the demand. The inlet guide vanes are positioned by an inlet guide vane actuator (IGVA).

The ECU controls the IGVA and surge control valve (SCV), using environmental data from P2 (pressure) and T2 (inlet temperature), load compressor output (P_t and ΔP), and load demand from the aircraft. When there is no demand, the ECU closes the IGVs to the lowest possible setting, so the load compressor imposes a minimum load on the power section. In this condition, the bleed air valve is closed and all air is directed through the SCV.

The SCV prevents load compressor surge for all operating modes, altitudes, and inlet temperatures. Surge is prevented by opening the SCV when the aircraft pneumatic system takes little or no airflow, or when the BAV is closed. During the APU shutdown sequence, the ECU initiates pneumatic unloading by closing the IGVs and BAV, and opening the SCV.

The BAV is a pneumatically actuated butterfly valve used to control bleed flow to the airplane. The BAV is normally in the closed position and uses load compressor bleed air controlled by a solenoid to provide the muscle for valve opening.

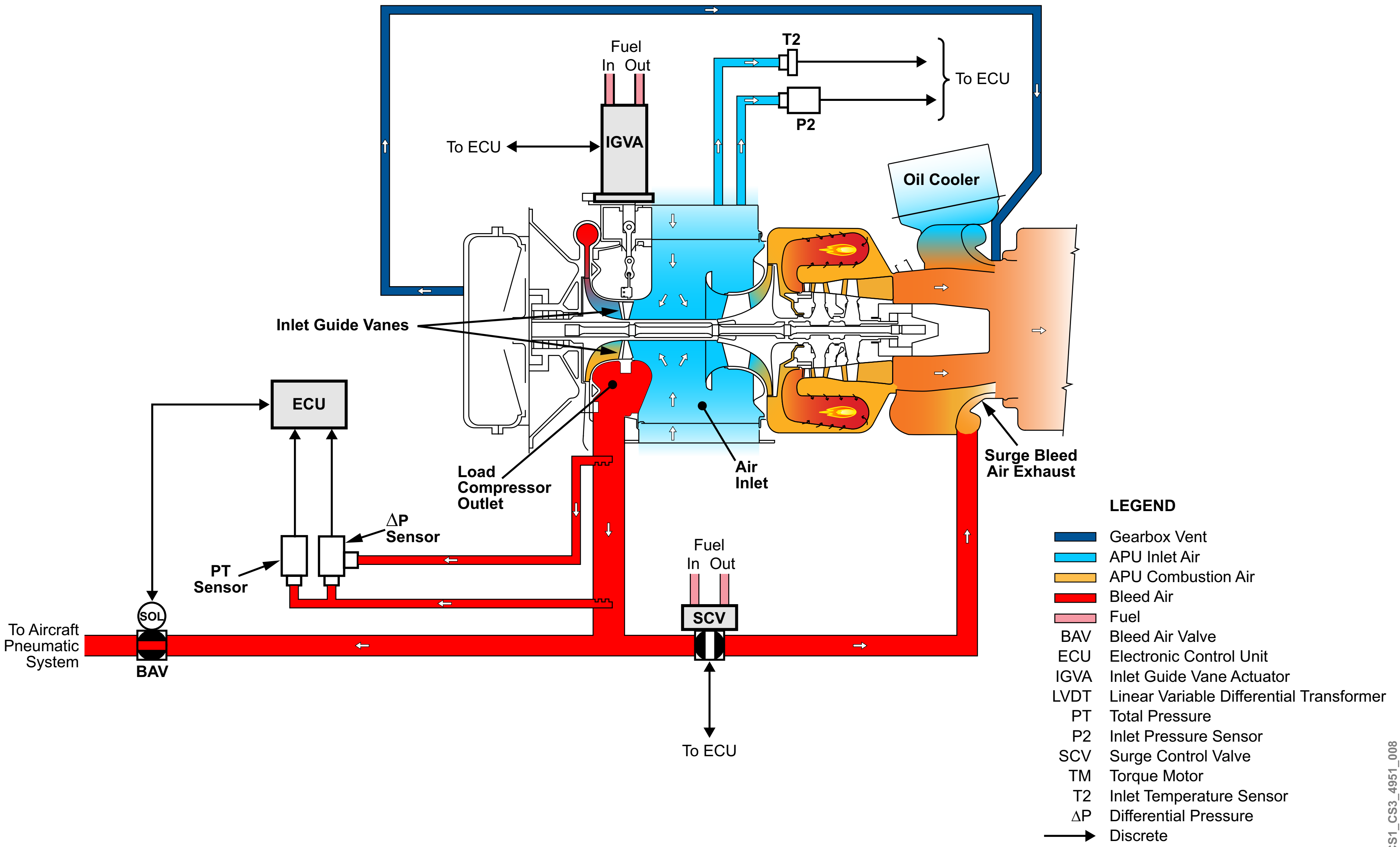


Figure 37: APU Air System

COMPONENT LOCATION

The air system consists of the following components:

- Surge control valve
- P2 sensor
- Total pressure sensor
- Differential pressure sensor
- Inlet temperature sensor
- Inlet guide vane actuator
- Bleed air valve

SURGE CONTROL VALVE

The surge control valve is located in the surge bleed duct on the right side of the APU.

P2 SENSOR

The P2 sensor is installed on the right side of the inlet plenum.

TOTAL PRESSURE SENSOR

The total pressure (P_t) sensor is located on the bottom of the APU compressor plenum.

DIFFERENTIAL PRESSURE SENSOR

The differential pressure (ΔP) sensor is located on the bottom of the APU compressor plenum.

INLET TEMPERATURE SENSOR

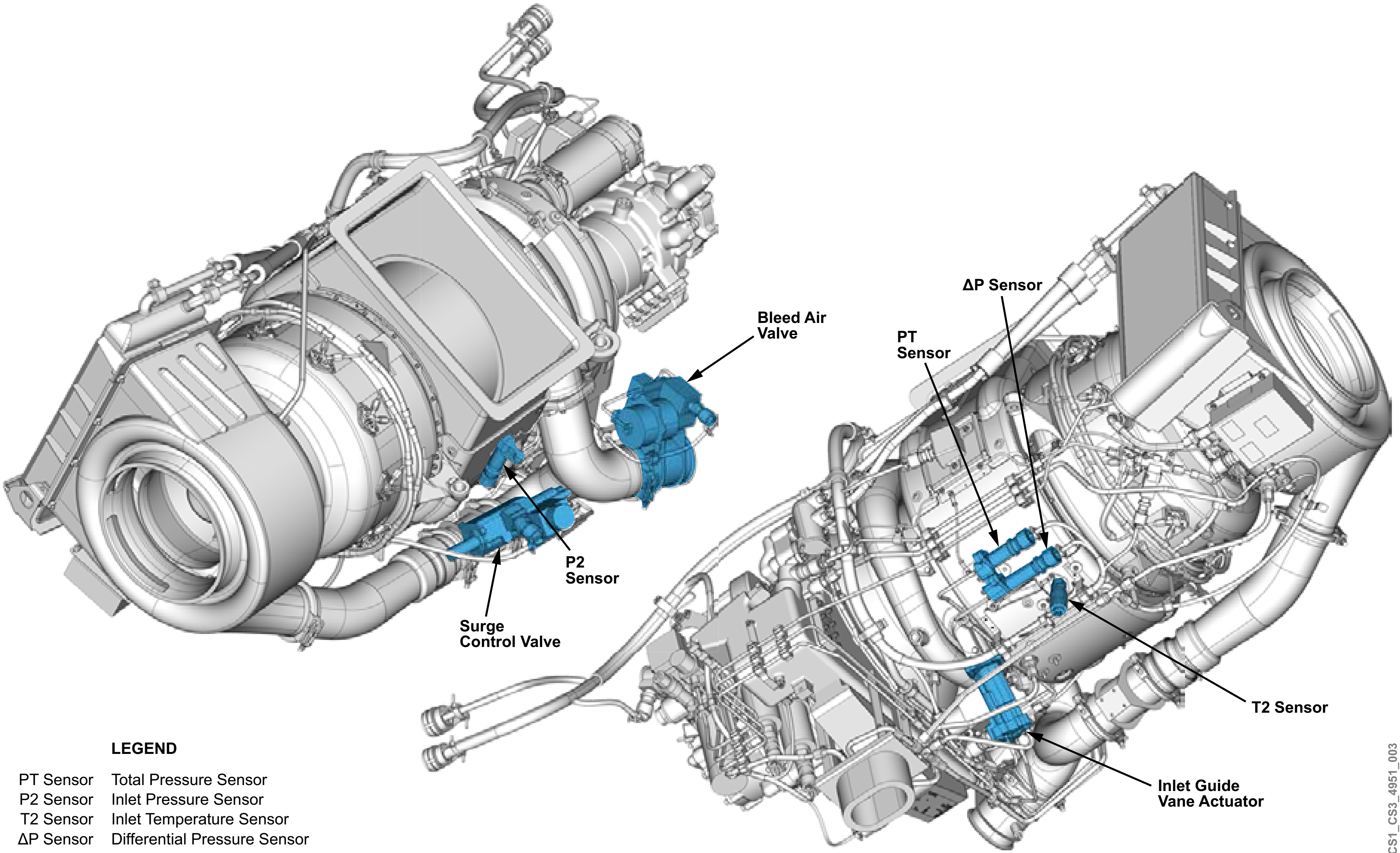
The inlet temperature (T2) sensor is located on the bottom of the APU compressor plenum.

INLET GUIDE VANE ACTUATOR

The inlet guide vane actuator is located below the load compressor section.

BLEED AIR VALVE

The bleed air valve is on the right side of the APU compartment in the bleed air duct.



LEGEND

PT Sensor	Total Pressure Sensor
P2 Sensor	Inlet Pressure Sensor
T2 Sensor	Inlet Temperature Sensor
ΔP Sensor	Differential Pressure Sensor

Figure 38: Air System Surge Control Valve, Bleed Air Valve, Inlet Guide Vane Actuator, and Flow Sensors

DETAILED DESCRIPTION

APU AIR INTERFACE

The APU electronic control unit (ECU) communicates with the aircraft systems via the data concentrator module cabinet (DMC) 1 and DMC 2 over ARINC 429 BUSES.

The APU generator control unit (GCU) supplies electrical load information to the ECU and receives a ready-to-load signal when the APU is on-speed.

The ECU receives environmental control system (ECS) demands from the integrated air system controller (IASC) 1 and IASC 2. The ECS demands are based on air data system information and whether the aircraft is on the ground or in flight.

The left electronic engine control (EEC) and right EEC provide main engine start (MES) and MES termination commands.

The control of the pneumatic output is based on ECS or MES demands as well as the APU inlet temperature (T2) and inlet pressure (P2).

The pneumatic output is based on the position of the inlet guide vanes. The IGVA is positioned by a torque motor driven by the ECU. The IGVA provides position feedback to the ECU.

When the pneumatic demand changes, due to ECS pack valve closures or the termination of MES, the surge control valve opens to prevent an APU surge.

The surge control valve is positioned by a torque motor driven by the ECU. The ECU receives position feedback from the SCV. The SCV position is based on the load compressor total pressure (P_t) and differential pressure (ΔP). The P_t, ΔP, and P2 sensors receive 10 VDC excitation from the ECU and supply outputs proportional to the sensed pressure.

The pneumatic output of the APU is supplied to the aircraft through a bleed air valve. The bleed air valve is a solenoid controlled valve, receiving 28 VDC form the ECU. A microswitch in the valve provides an indication when the valve is open.

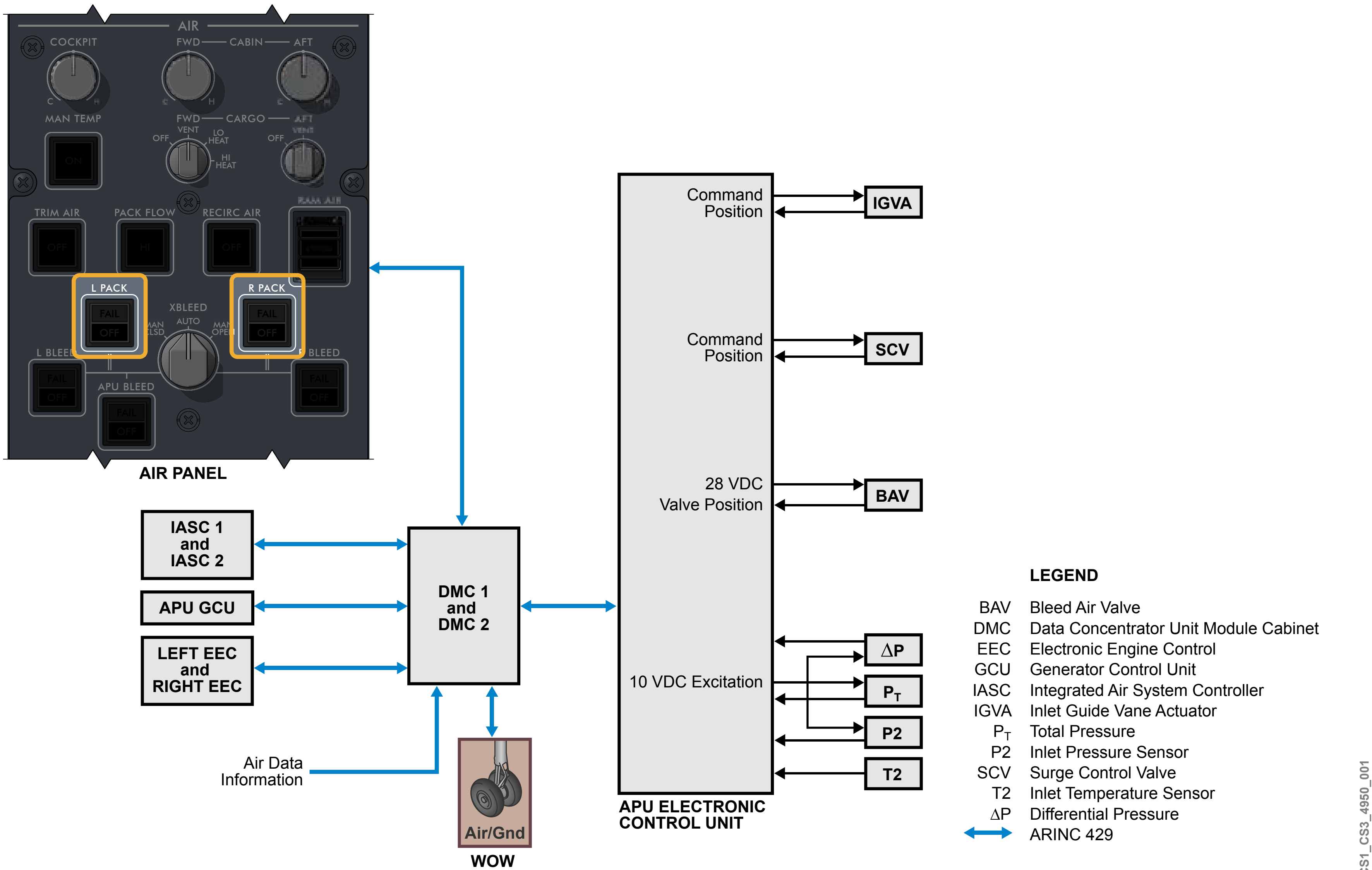


Figure 39: APU Air Interface

APU AIR CONTROL

The load compressor output is regulated by variable IGVs located just upstream of the load compressor impeller. Output is based on aircraft environmental control system (ECS), main engine start (MES), and electrical load demands as indicated by aircraft signals to the APU electrical control unit (ECU).

The IGV position is a function of P2, T2, and load demand. During an APU start, the IGVs are closed to 15° up to 60% rpm and then opened to 22° if altitude is below 25,000 ft. This removes any pneumatic load from the load compressor. Above 25,000 feet, the IGVs are held at 15°.

To minimize fuel consumption, when there is no demand, the ECU closes the IGVs to the lowest possible setting that provides a constant airflow through the compressor during all loading conditions so the load compressor imposes minimum load on the power section. The ECU logic for determining IGV position includes a compensation factor for load compressor deterioration, based on the APU operating hours recorded by the data memory module (DMM).

To optimize fuel consumption in the ECS mode, the pneumatic load provided by the APU is matched to the aircraft requirements. The pressure and flow requirements vary with ambient conditions and whether the aircraft is on the ground or in flight.

As the combined loads of the generator and the load compressor are imposed on the power section, the speed governing function of the ECU responds by increasing fuel flow. Under some extremely high combination loads or with significant deterioration of the power section, the turbine temperature limit is reached.

When a load change results in a temperature higher than its limit, the ECU moves the IGVs toward a more closed position, reducing load compressor airflow, and thus reducing the pneumatic load and decreasing the turbine temperature. The electrical load remains unchanged and has priority over the pneumatic load.

The surge control valve (SCV) ducts air from the load compressor to the APU exhaust. The SCV makes sure there is a minimum flow of air through the load compressor.

The surge margin set point is the minimum quantity of air that should flow through the load compressor to prevent load compressor surge. The ECU calculates the surge margin set point using:

- Inlet pressure (P2)
- Inlet temperature (T2)
- IGV position
- Bleed mode
- Air/ground mode

The ECU uses total pressure (P_t) and differential pressure (ΔP) to maintain the load compressor output at or above the stall margin set point.

During MES, the APU ECU opens the IGVs to 90° and modulates the SCV to provide maximum bleed for engine starting. The priority is given to the pneumatic load during engine start. If required, some APU generator load sheds.

To prevent damage to air conditioning components as well as overpressure and overtemperature conditions, the APU software logic limits the APU bleed pressure and temperature to:

- 65 psi maximum bleed pressure
- 246ºC (475ºF) maximum bleed temperature

In the event of a failure, the ECU opens the SCV and closes the IGVs. This gives priority to electric operation over bleed air demand in the event of a surge system malfunction.

Table 4: APU AIR System Operating Modes

MODE	DEFINING CONDITIONS	ELECTRICAL LOAD AVAILABILITY	PNEUMATIC SYSTEM STATUS	COMMENT
Startup	• APU ON signal present • MOMENTARY START signal initiates start	Not available	• IGVs positioned at 15° during engine start • SCV fully open	• Timed acceleration starting logic applies • Mode ends when APU reaches 95% speed
READY-TO-LOAD (RTL) (Generator only)	• Speed >95% +2 seconds • BLEED-AIR COMMAND OFF	Available	• Bleed air not available • IGVs positioned at 15° for altitude >25,000 ft • SCV fully open	• Bleed air not available, only because APU is not selected as bleed air source at conditions where bleed air would normally be available
Environmental Control System	• Speed >95% +2 seconds • BLEED-AIR COMMAND ON • Main engine start is not requested • One or both packs on • Altitude <23,000 ft	Available	• Bleed air available • IGVs set to ECS position schedule by logic in the ECU, as a function of inlet temperature (T2), inlet pressure (P2), A/C configuration, and flow mode • Surge control system active	• Electrical load has priority • If power capability is exceeded, pneumatic load is cut back, as needed, by closing the IGVs
Main Engine Start	• Speed >95% +2 seconds • BLEED-AIR COMMAND ON • Either left or right main engine start signal is true • Altitude <23,000 ft	Available	• Bleed air available • IGVs set at 90° (full open) • Surge control system active	• Pneumatic power has priority. If power capability is exceeded, load shed is sent to airplane to reduce electrical load, so that IGVs do not have to be cut back

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MONITORING AND TEST

The following page provides the crew alerting system (CAS) messages for the APU air system.

CAS MESSAGES

Table 5: CAUTION Message

MESSAGE	LOGIC
APU BLEED FAIL	Failure of APU providing bleed air when requested or bleed leak detection fail.

49-60 APU CONTROL

GENERAL DESCRIPTION

The auxiliary power unit (APU) electronic control unit (ECU) controls and monitors the APU operation and provides an interface with the aircraft. The APU ECU uses inputs from the exhaust gas temperature (EGT) thermocouples and the speed sensor to control the start and acceleration sequence, speed governing, load sequencing, and the shutdown sequence. The ECU has field-loadable software.

The EGT thermocouple senses the temperature of the exhaust gas that exits out of the turbine section.

The speed sensor provides two independent speed signals to the ECU.

The APU ECU operates on 28 VDC power provided by DC ESS BUS 2, with the DC EMER BUS supplying a secondary source of power.

The APU is started using the APU switch on the APU panel. The ECU controls the ignition and fuel automatically, based on environmental conditions. The APU runs at a variable speed (91-98%) on the ground, and is governed to 100% rpm in flight.

The ECU sends a ready-to-load signals to allow electrical or pneumatic loading when the APU is approaching its governed speed.

The data memory module (DMM) is an electronic data storage device mounted on the APU and is used for recording parameters, such as APU operating hours, starts, number of unsuccessful starts, and the APU serial number.

The ECU monitors the APU components for faults, recording, and reporting. A failed ECU causes an immediate APU shutdown and an APU SHUTDOWN EICAS advisory message to be displayed.

The ECU shuts down the APU when the built-in test (BIT) identifies conditions that can damage the engine or the aircraft for continued APU operation. The ECU also has the authority to inhibit starting of the APU when the BIT indicates APU operation may cause damage.

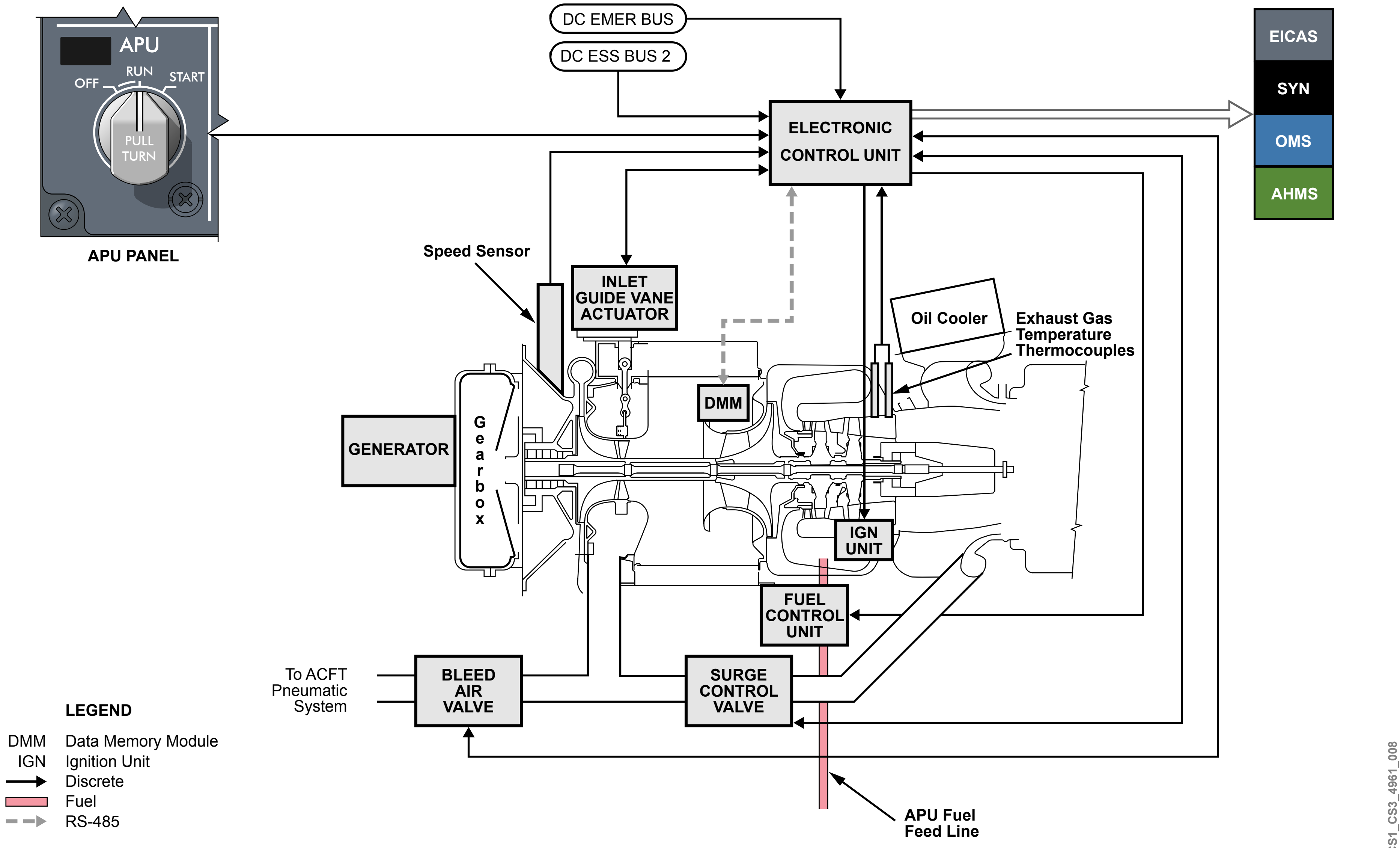

Figure 40: APU Control

APU ECU CONTROL FUNCTIONS

Prestart Sequence

The ECU performs integrity checks to verify that the APU is safe to start and operate. The requirements for an APU start are:

- ECU BIT self-test passed
- ECU BIT test of sensors and controls
- APU door full open signal

Start Sequence

When the inlet door reaches the proper position, the ECU sends a start enable command to BPCU 2 to power the APU starter. The ECU powers the ignition unit, and then energizes the fuel solenoid.

As the APU accelerates to operating speed, the ECU controls fuel flow via the fuel control unit torque motor as a function of acceleration, fuel schedule, and EGT. The ignition unit and starter command are deactivated when the speed thresholds are attained.

On-Speed Governing

Exhaust gas temperature and engine speed are continuously monitored by the ECU. Once on-speed, the ECU maintains engine speed at 100% rpm. On the ground, the APU is governed between 91% and 98% rpm based on environmental conditions. Ready-to-load signals are provided for electrical and pneumatic systems.

Pneumatic Loading

Pneumatic loading for the aircraft environmental control system and main engine start operation are provided by the APU through the ECU control of the IGVs. The ECU positions the IGVs in response to bleed demands. The ECU regulates the APU pneumatic output by sensing the APU EGT and comparing it to a predetermined schedule within the ECU.

APU surge is prohibited through the ECU control of the SCV actuator.

Normal Shutdown

The ECU initiates APU shutdown by injecting a false overspeed signal into the programmable logic device (PLD), which provides overspeed protection. The ECU uses two independently operating PLD devices. The function of each overspeed PLD is to determine if an APU overspeed condition exists and cause a shutdown.

The primary mode for shutdown is by removing power from the fuel solenoid. Each PLD is capable of shutting down the APU in the event of an overspeed. In order to limit potential fuel solenoid dormancy, at every normal APU shutdown, the APU fuel shut-off solenoid is exercised, alternating the commanding PLD at each APU shutdown.

Protective Shutdowns

The ECU shuts down the APU when the BIT identifies conditions that can damage the engine or the aircraft.

APU protection is provided by logic schemes that determine whether the APU is safe to operate and protective shutdown logic that shuts down an operating APU.

Fault Detection

The ECU monitors the APU components for faults, recording and reporting. A failed ECU causes an immediate APU shutdown and an APU SHUTDOWN EICAS advisory message to be displayed.

The ECU reports faults to the onboard maintenance system (OMS).

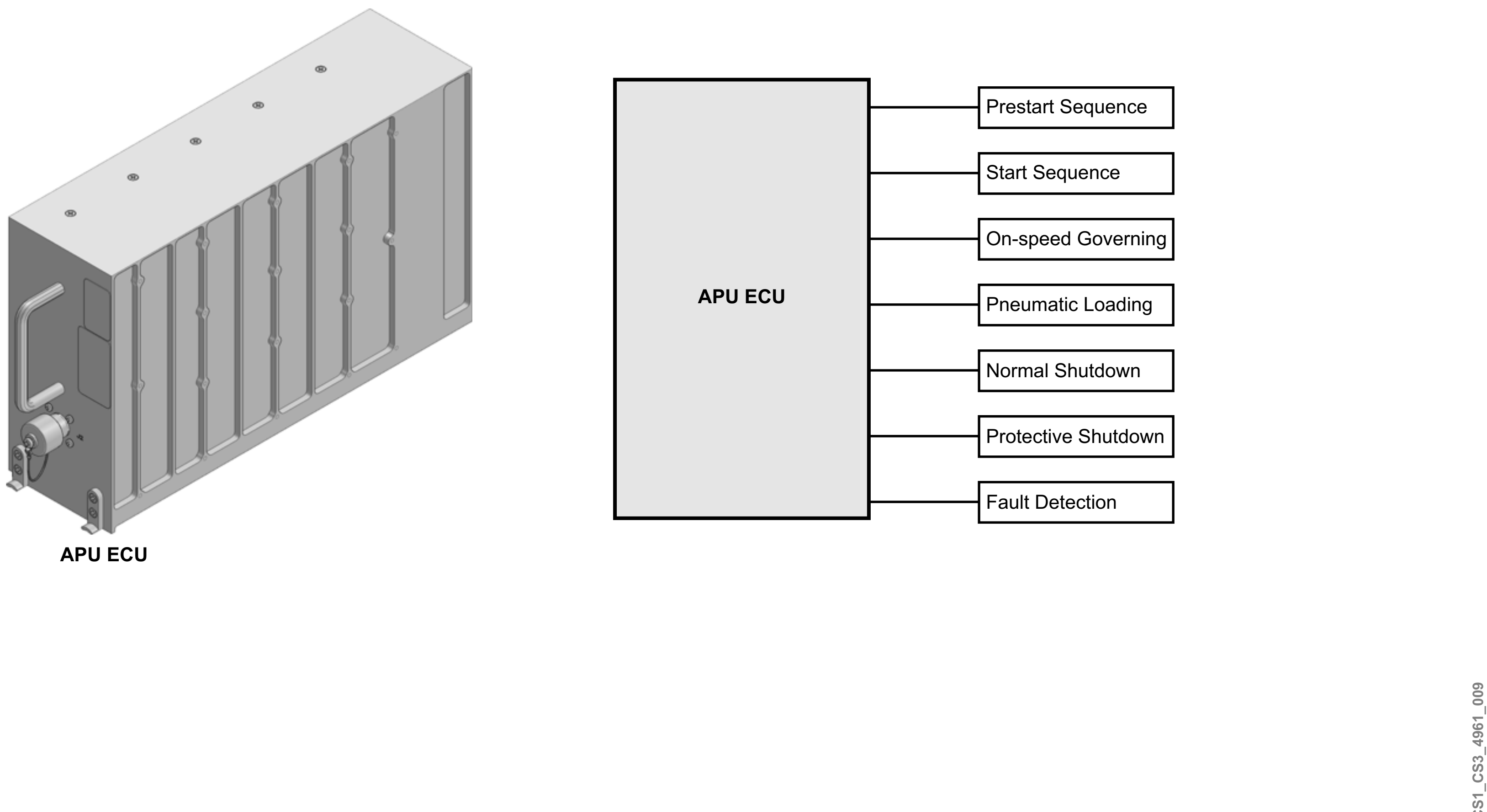

Figure 41: APU ECU Control Functions

APU START

The APU switch signals the ECU when it is selected to the START position. The ECU opens the APU feed shutoff valve and the APU inlet door. When the air inlet door is fully open, the inlet door position sensor sends a door open signal to the ECU.

The APU start sequence is as follows:

- 0% rpm, the ECU energizes the starter and ignition circuit
- 0% rpm, the IGVs positioned to 15°
- 7% rpm, the fuel solenoid valve opens
- 50% rpm, the starter de-energizes
- 60% rpm, the ignition unit de-energizes
- On ground, ready to load signal is 3% below governed speed signal (88% rpm to 95% rpm)
- In flight, ready to load signal is 95% rpm
- 91-98% rpm, governed speed range on ground
- 100% rpm, governed speed in flight
- 106% rpm, overspeed shutdown is initiated

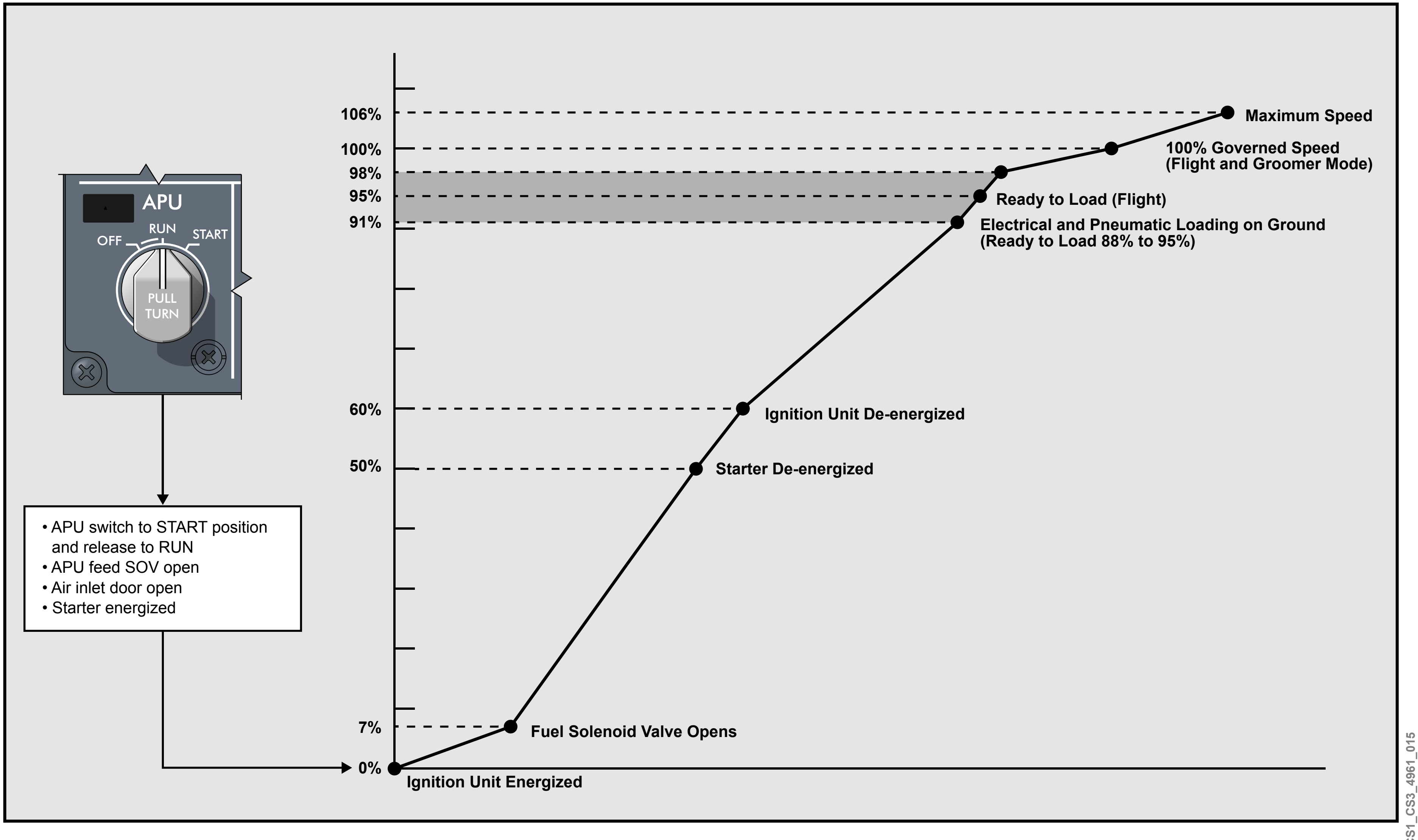

Figure 42: APU Start Sequence

APU SHUTDOWN

The ECU controls the APU shutdown. There are two types of shutdown conditions: normal and protective.

Normal Shutdown

A normal shutdown is initiated from the APU panel when the APU switch is moved to the OFF position. The normal shutdown sequence is as follows:

- 28 VDC ON signal removed from ECU
- ECU receives the OFF signal
- Generator ready to load signal is removed
- 60 second cooldown period starts
- After cooldown, the ECU tests the overspeed circuit closes the FCU shutoff valve and removes power from the metering valve
    - ECU tests the overspeed circuit which closes the fuel solenoid shutoff valve and shuts down the APU
    - During this time, the ECU tries to maintain the APU speed by modulating the FCU torquemeter current
    - If the APU continues to run, the fuel shutoff solenoid valve has failed to close
    - If the fuel solenoid shutoff valve has failed, the ECU reduces the FCU torquemotor current to 0 to shut down the APU
- 20% speed, the APU inlet door is commanded closed
- 7% rpm, the APU fuel shutoff valve (SOV) closes
- At less than 7% speed, an APU restart can be initiated

Cooldown Cycle

The cooldown period prevents oil coke at the turbine bearing and fuel nozzles. During cooldown, the ECU:

- Removes the bleed air ready-to-load signal
- Closes the bleed air valve
- Closes the inlet guide vanes to 22°
- Opens the surge control valve
- De-energizes the generator
- Starts the 60 second timer

Protective Shutdown

A protective shutdown is similar to the normal shutdown but has no cooldown cycle.

NOTE

The APU fuel shutoff valve and air inlet door close for both normal and protective shutdowns.

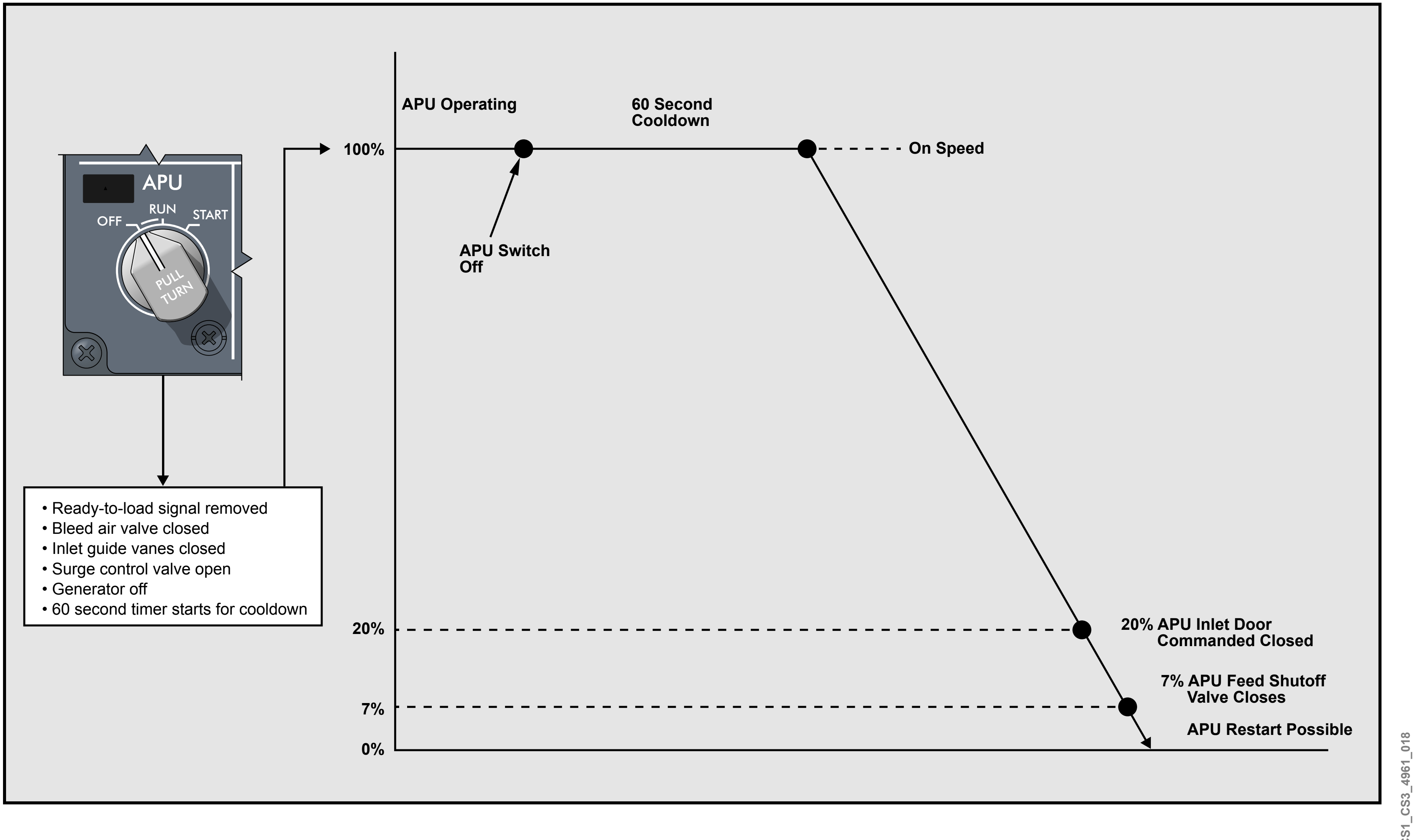

Figure 43: APU Shutdown Sequence

COMPONENT LOCATION

The control system consists of the following components:

- Exhaust gas temperature thermocouple
- Speed sensor
- Data memory module
- Electronic control unit

EXHAUST GAS TEMPERATURE THERMOCOUPLE

The APU consists of two exhaust gas temperature thermocouples probes that are located below the APU turbine casing and extend into the exhaust gas stream inside the APU.

SPEED SENSOR

The speed sensor is located on the bottom of the load compressor case.

DATA MEMORY MODULE

The data memory module is installed on the left side of the APU air inlet plenum.

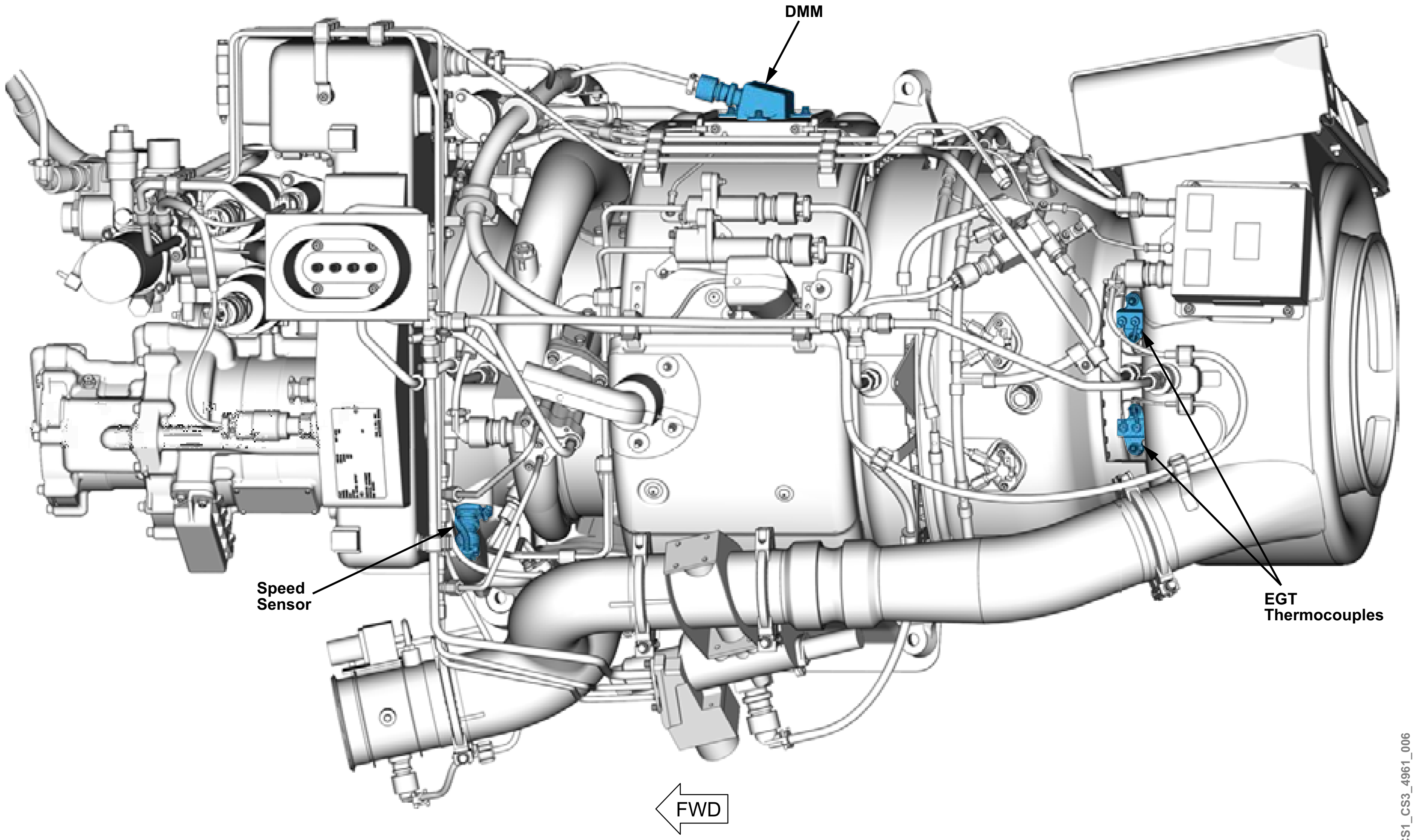

Figure 44: Data Memory Module, Speed Sensor, and EGT Thermocouples

ELECTRONIC CONTROL UNIT

The electronic control unit (ECU) is located in the mid equipment bay.

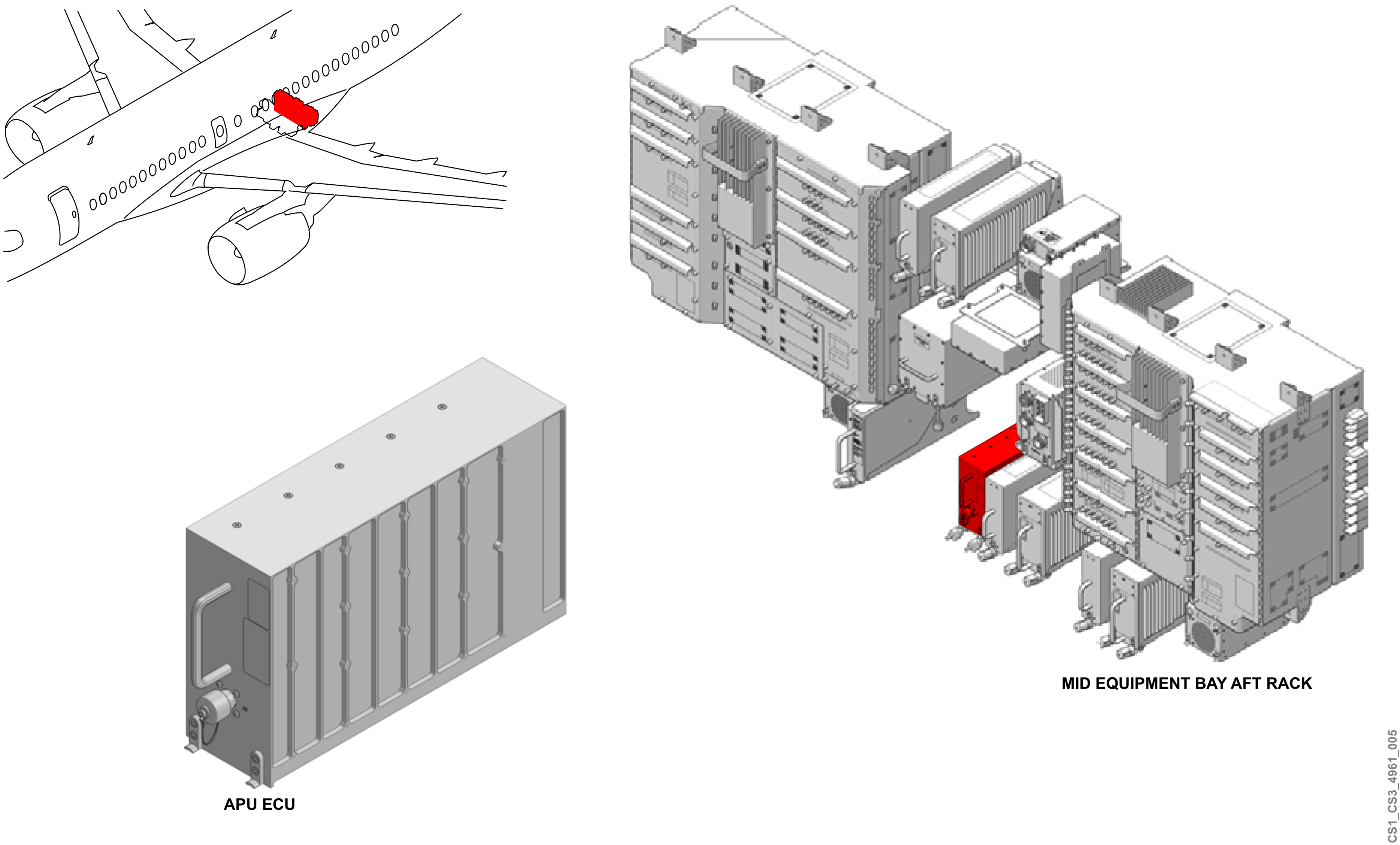

Figure 45: APU Electronic Control Unit Location

DETAILED COMPONENT INFORMATION

DATA MEMORY MODULE

The data memory module (DMM) interfaces with the APU ECU. The APU ECU manages the communication between the OMS and DMM.

The DMM stores information regarding the health monitoring and life usage of the APU. The DMM interfaces with the ECU via a serial link and stores the following information:

- APU serial number and model number
- APU operating hours and starts
- APU low cycle fatigue and creep life
- APU accumulated aborted starts

At power-up, the ECU reads the DMM and performs the following:

- If the information is the same as the ECU, power-up initialization occurs
- If the information is different, the ECU record is updated from the DMM
- Upon completion of an APU cooldown cycle, the DMM is automatically updated from the ECU

The APU data is also stored in the ECU in case of DMM failure.

Replacement DMMs are delivered with all zeros stored in their memory. Once recognized, the ECU records in memory that a DMM change has occurred. The new DMM is then loaded with all the totaled APU data that is stored in the ECU.

If the ECU is replaced, the DMM retains all the APU data and automatically re-initializes the new ECU with the old accumulated APU data.

CAUTION

Do not remove the APU electronic control unit (ECU) and data memory module (DMM) at the same time. Always do the APU BIT check between replacement of each of the units or data may be erased.

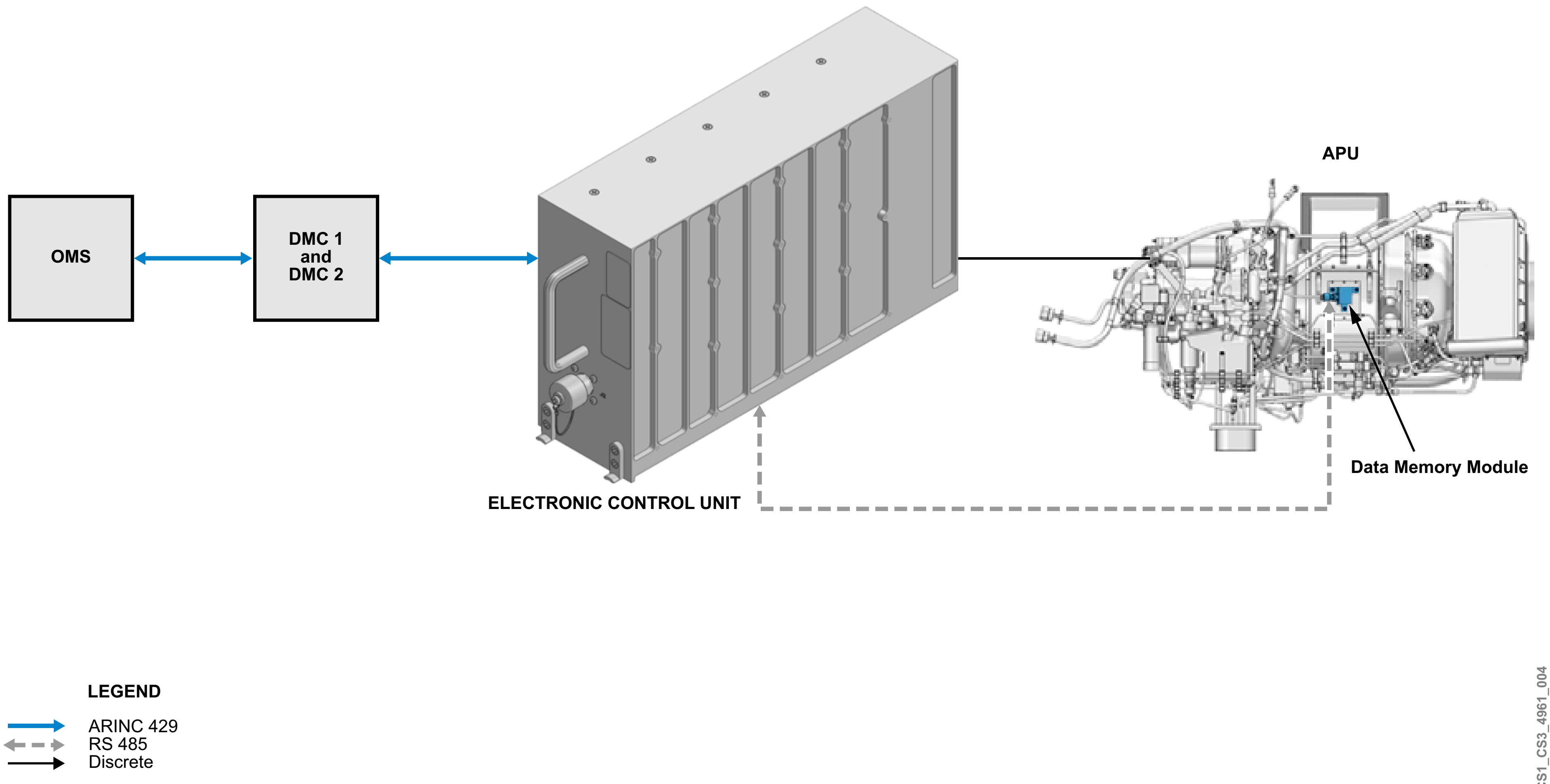

Figure 46: Data Memory Module

DETAILED DESCRIPTION

ECU POWER

The ECU primary power is supplied by DC ESS BUS 2. The ECU also has a secondary power input that is connected to the aircraft DC EMER BUS through the APU relay controlled by the ECU.

The dual input power configuration allows the ECU to select the highest voltage power source, remain powered ON whenever the DC EMER BUS is switched on, and allows the DMC to continuously communicate with the ECU whenever the aircraft is powered.

APU Start

During normal operation, the ECU is powered on via the DC ESS BUS 2 when the aircraft battery switch is closed. After the APU switch is turned to RUN and the ECU completes successful prestart BIT, the ECU applies power to the APU relay and provides secondary power to the ECU via the DC EMER BUS.

APU Shutdown

On the ground, once the APU switch is turned to OFF and the APU 60 second cooldown, shutdown, and memory management activities are completed, the ECU de-energizes the relay. The APU ECU power is only supplied by the DC ESS BUS 2.

In flight, the relay remains activated at APU shutdown to retain power input redundancy.

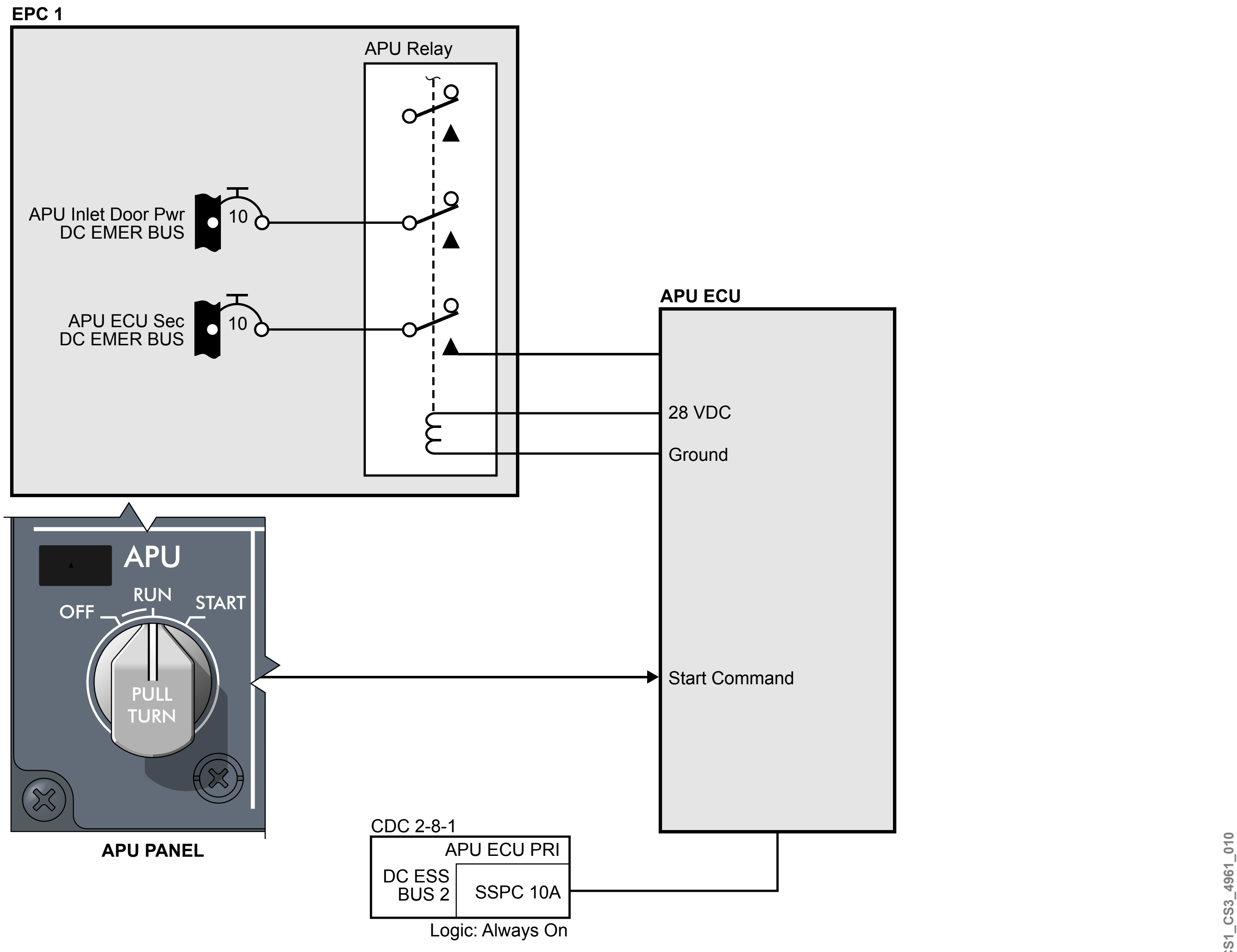

Figure 47: ECU Power

ECU INTERFACE

The ECU communicates with the IASC to determine the APU bleed mode required. The ECU positions the IGVs and SCV based on this information.

The ECU receives DC electrical power from DC ESS BUS 2 and the DC EMER BUS.

The ECU receives a signal from the fire detection and extinguishing (FIDEX) system when an APU fire is detected. When pressed, the APU FIRE pushbutton annunciator (PBA) also closes the shutoff valve (SOV), independent of the ECU.

The ECU controls the APU fuel shutoff valve during normal operation.

The ECU provides information to the EICAS and synoptic pages and receives input signals from aircraft systems via an ARINC 429 BUS from the data concentrator unit module cabinets (DMCs). The DMCs provide:

- Electrical load
- Electrical load shed requests
- Main engine start requests
- Engine status
- Weight-on-wheels (WOW) status
- Air data
- APU switch commands

Unless otherwise specified, when duplicate data is available from both DMCs, the ECU selects DMC 1 as the primary data source.

The ECU controls and receives position feedback from APU inlet door actuator.

The APU can be shut down externally by a switch located in the external service panel or from the switch in the APU compartment.

The groomer mode is entered when the maintenance outlets in either galley 1 or 4 are used. In the groomer mode, the APU is governed to a frequency of 400 Hz.

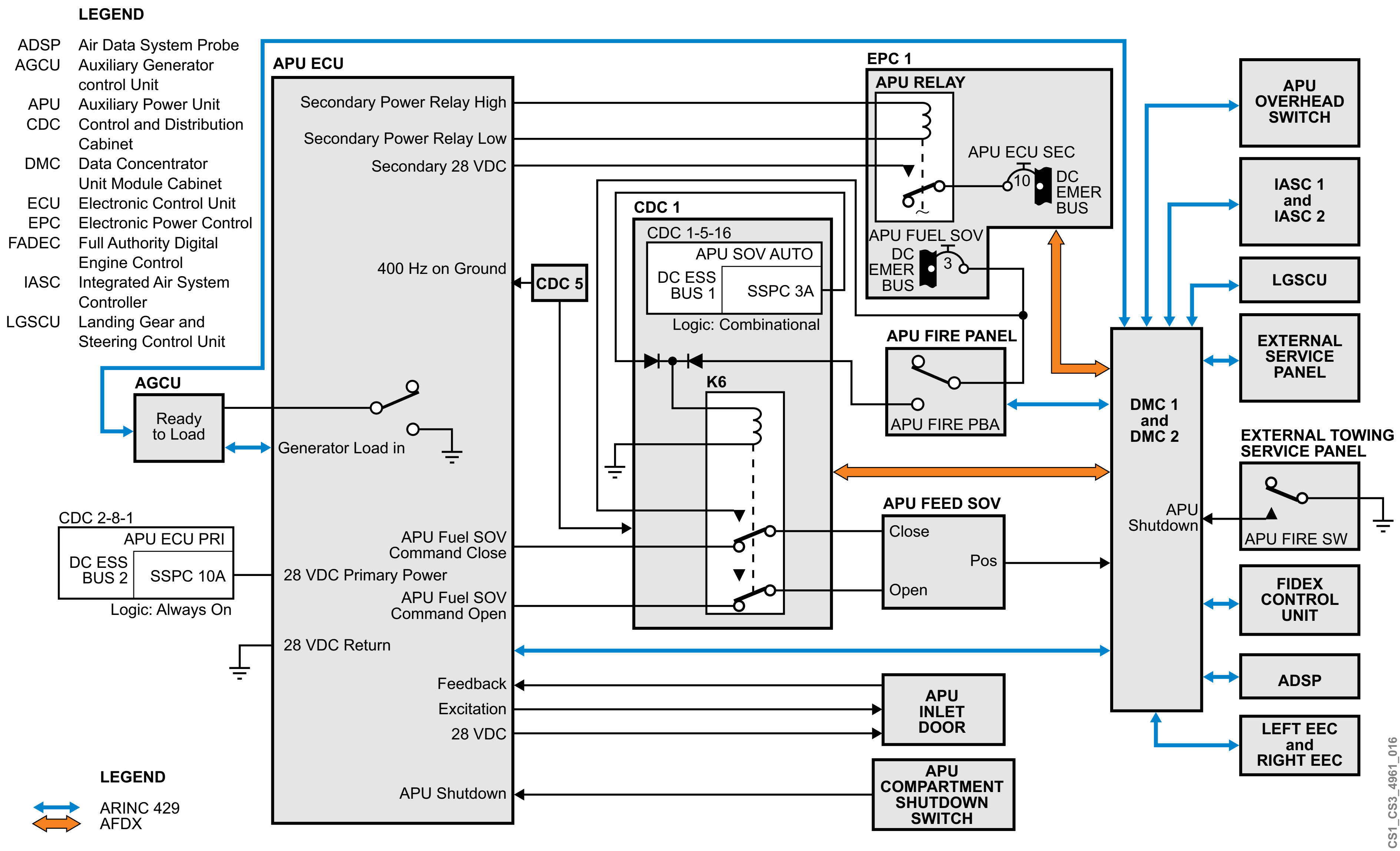

Figure 48: Electronic Control Unit Interface

APU PROTECTIVE SHUTDOWNS

The ECU provides protective, automatic shutdowns to prevent APU or aircraft damage. The weight-on-wheels (WOW) signal is used to inhibit some of the auto-shutdown functions when the APU is in flight mode.

The ECU automatically shuts down the APU when certain fault conditions are detected. Protective shutdowns are indicated on the engine indication and crew alerting system (EICAS) by an APU SHUTDOWN advisory message with the exception of OVERSPEED, which is a dedicated warning message.

In flight, the ECU inhibits some protective shutdown. When a protective shutdown is inhibited in flight, an APU caution message is displayed. The APU can be shut down manually if it is not required.

During a protective shutdown, there is no cooling period and the APU shuts down immediately. When a protective shutdown occurs, the electrical and pneumatic loads are removed immediately. The bleed air valve closes and the SCV opens. Power is removed from the APU fuel control unit fuel solenoid and torque motor, and then the APU fuel SOV closes. The APU switch must be manually selected OFF when a protective shutdown occurs.

Table 6: APU Protective Shutdown

CONDITION	AUTO SHUTDOWN ON GROUND	AUTO SHUTDOWN IN FLIGHT
Failed/slow/hot/hung start	YES	YES
RPM underspeed	YES	NO
RPM overspeed/signal loss	YES	YES
Critical ECU fault (i.e. loss of DC power)	YES	YES
Loss of ARINC inputs to the ECU	YES	YES
APU fire	YES	NO
EGT overtemperature/signal loss	YES	NO
APU door position uncommanded/signal loss	YES	NO
Excessive oil temperature	YES	NO
Low oil pressure	YES	NO
Oil filter clogged	YES	NO
APU inlet overheat/reverse flow	YES	NO

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MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the APU control system.

CAS MESSAGES

Table 7: WARNING Message

MESSAGE	LOGIC
APU OVERSPEED	APU overspeed and no auto-shutdown.

Table 8: CAUTION Message

MESSAGE	LOGIC
APU	APU failure in flight that does not lead to an automatic shutdown.

Table 9: ADVISORY Message

MESSAGE	LOGIC
APU SHUTDOWN	Failures detected that cause the APU to shut down (mostly used on ground).
APU FAULT	Loss of redundant or noncritical function for the APU systems.

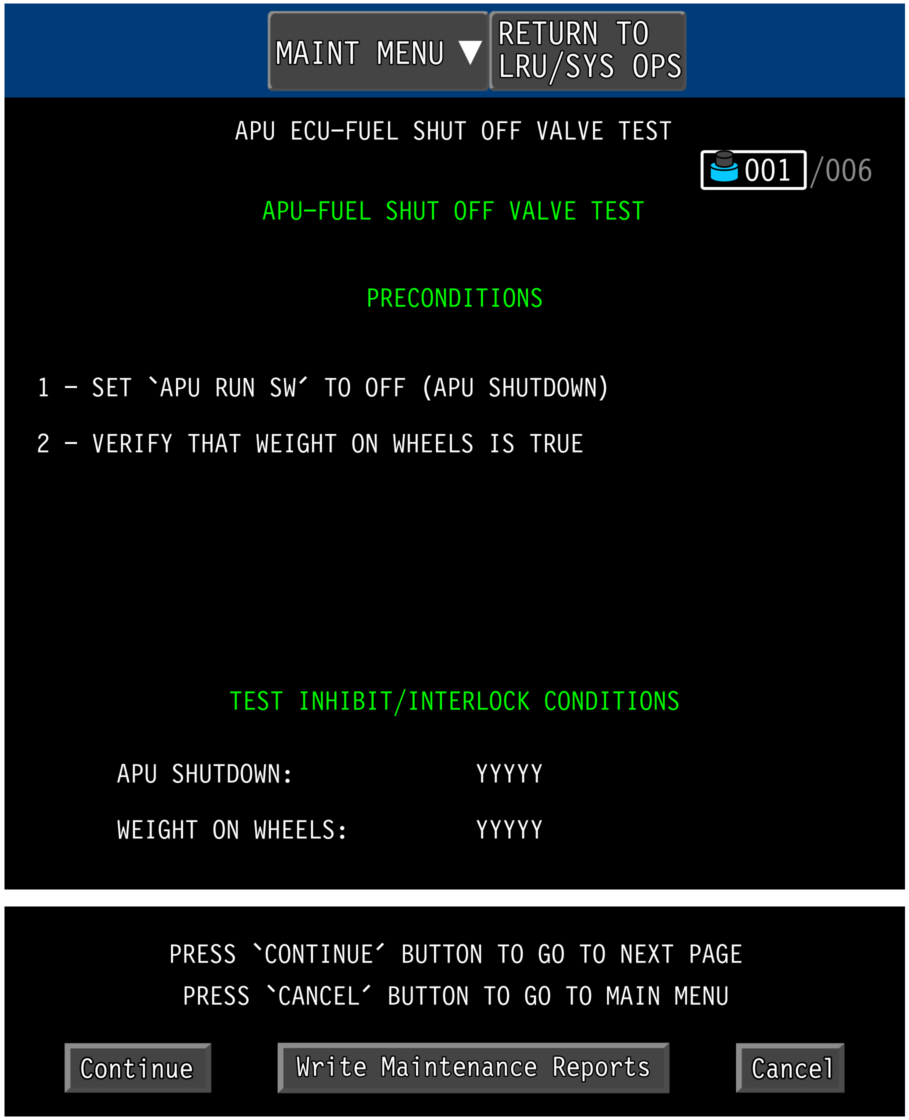
Table 10: INFO Messages

MESSAGE	LOGIC
49 APU FAULT - APU INOP	APU inoperative (APU fuel solenoid did not close when commanded fuel solenoid or loss of ARINC 429 input from DMC) or shutdown condition is detected.
49 APU FAULT - APU DEGRADED	Oil temp sensor fault or inlet temp sensor fault or ECU noncritical fault or APU low fuel flow and not inflight or oil pressure switch is lost.
49 APU FAULT - APU REDUND LOSS	APU secondary power relay command open short or APU 28 VDC secondary low or APU 28 VDC primary low and not APU 28 VDC secondary power relay command open short or APU 28 VDC secondary low and APU 28 VDC primary low or APU speed sensor failure and not loss of both speed sensors or EGT1 loss or EGT 2 loss and not EGT 1 and EGT 2 loss or APU ARINC 429 to DMC 1 no data received or APU ARINC DMC 2 no data received and not loss of both ARINC 429 signals.

PRACTICAL ASPECTS

ONBOARD MAINTENANCE SYSTEM

The APU fuel shutoff valve is tested through the onboard maintenance system (OMS). The fuel shutoff valve test is found on the APU test page.



CS1_CS3_4931_008

Figure 49: OMS Fuel Shutoff Valve Test

49-00 APU OPERATION

GENERAL DESCRIPTION

The auxiliary power unit (APU) is started from the flight deck using the APU switch located on the overhead panel.

When the APU switch is momentarily held in the START position, the APU begins the start sequence:

- Built-in test (BIT)
- APU door open
- APU feed shutoff valve (SOV) open

When released, the selector automatically returns to RUN position.

If the BIT tests are successful, the APU start is initiated. The electronic control unit (ECU) controls the APU start sequence.

The ECU continuously monitors the APU start sequence and shuts down the APU if a fault is detected. If a BIT fails or a fault is detected during start, the APU switch is selected to OFF and a restart can be carried out after a two minute cooldown period.

When the rotary selector is placed in the RUN position after START, the APU runs automatically, providing air and electrical loads as required.

On ground, moving the APU switch to the RUN position without going to START enables the maintenance mode, which opens the APU inlet door and the aircraft APU feed shutoff valve.

When the APU is running, selecting the APU switch to OFF initiates the cooldown sequence and the APU shuts down.

NOTE

Moving the selector between OFF and RUN position requires a pull-to-turn action.

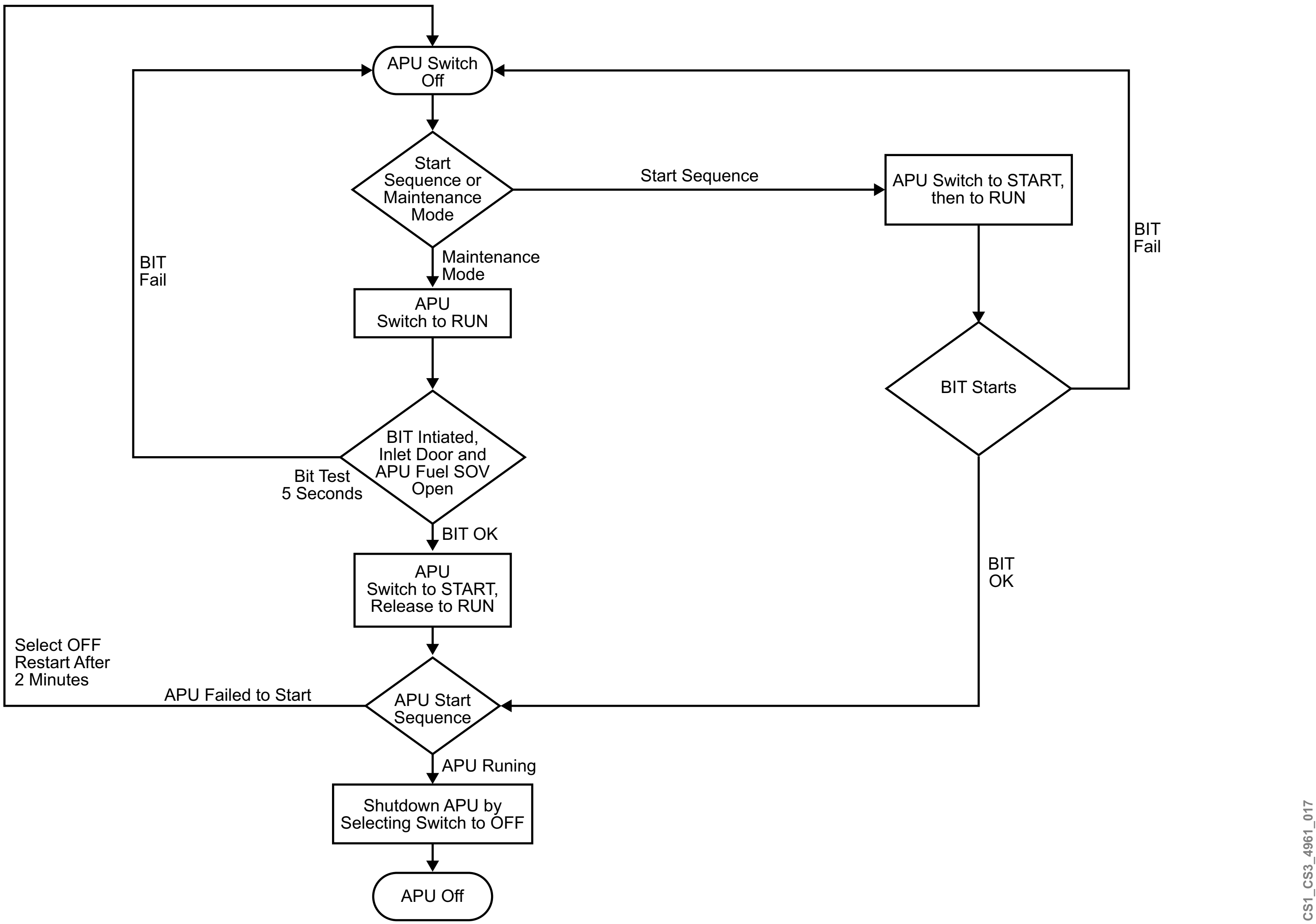


Figure 50: APU Operation

CONTROLS AND INDICATIONS

FLIGHT DECK CONTROLS

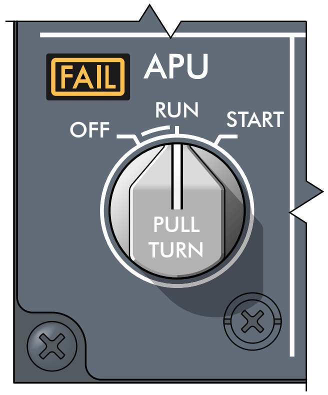
APU Fire Panel

The APU FIRE PBA is located on the ENGINE and APU FIRE panel on the overhead integrated cockpit control panel (ICCP). The APU FIRE PBA provides 28 VDC to close the APU feed shutoff valve through hardwired switch contacts.

APU Panel

The APU master switch is located on the left inboard ICCP module of the overhead panel. The switch position is sent to the ECU via an ARINC 429 BUS.

ENGINE AND APU FIRE PANEL



APU PANEL

CS1_CS3_4961_011

Figure 51: APU Flight Deck Controls

EXTERNAL CONTROLS

There are two external APU shutdown switches. Operating either switch causes the APU to shut down without a cooldown period. The switch inputs are inhibited when the aircraft is in flight.

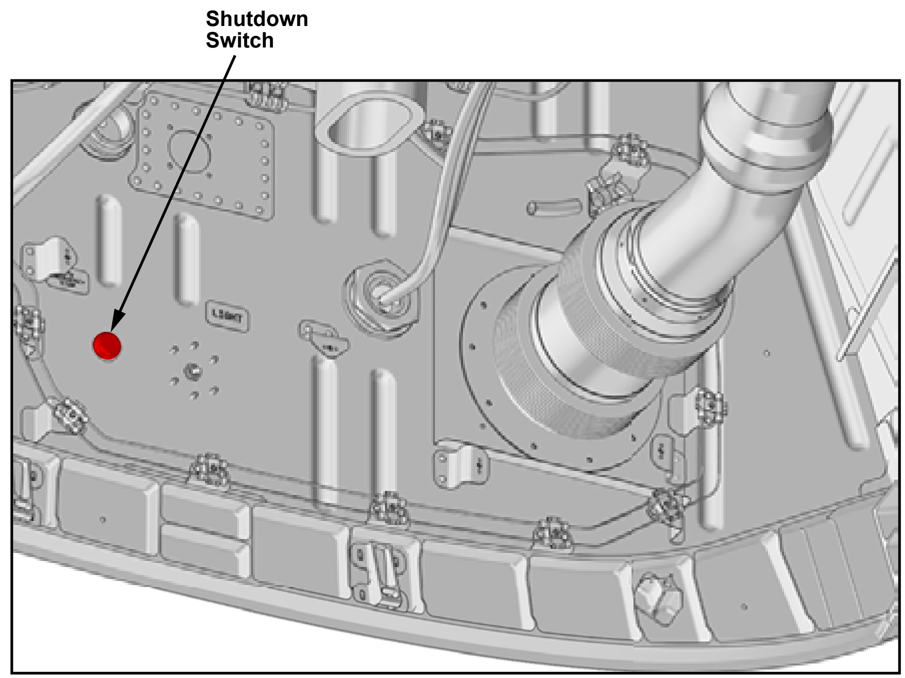
ELECTRICAL/TOWING SERVICE Panel

The APU SHUT OFF toggle switch on the external service panel is a guarded momentary contact switch with a center OFF position. The switch is hardwired to the data concentrator unit module cabinet (DMC).

APU Compartment Shutdown Switch

The APU compartment shutdown switch is mounted on the forward firewall. The switch is hardwired to the APU ECU.

ELECTRICAL/TOWING SERVICE PANEL



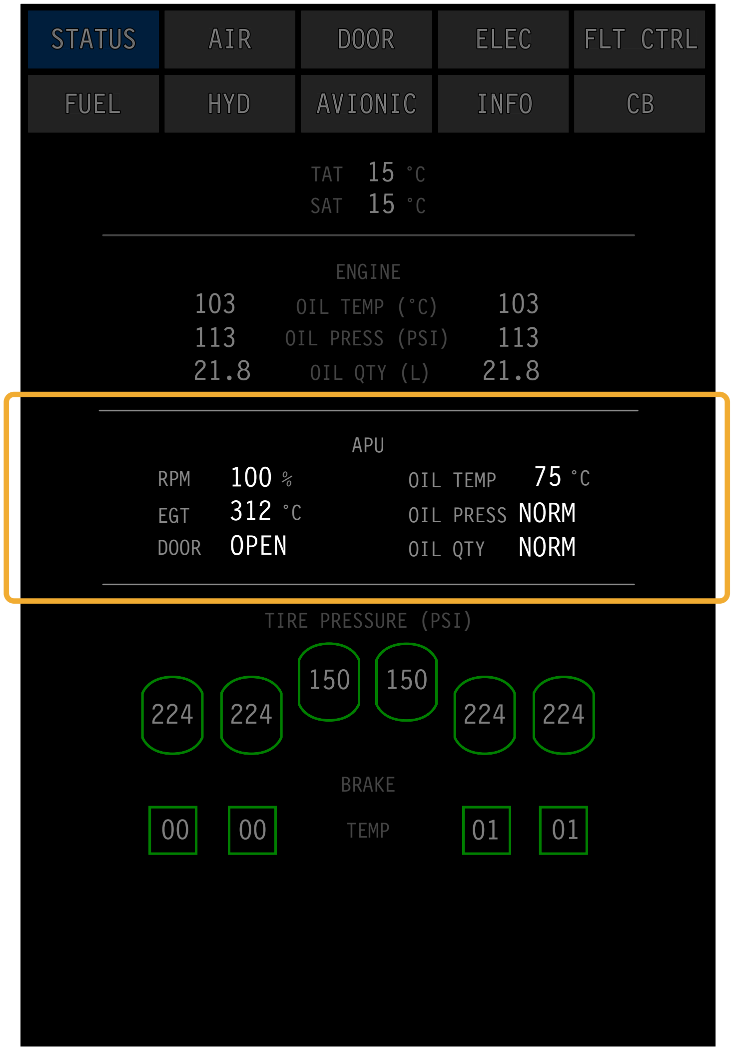
APU COMPARTMENT SHUTDOWN SWITCH

CS1_CS3_4961_013

Figure 52: APU External Controls

SYNOPTIC PAGE

The APU performance information is displayed on the STATUS synoptic page.



SYNOPTIC PAGE - STATUS

APU RPM	
Symbol	**Condition**
85	APU operation or below redline.
107	APU operation above redline (overspeed).
---	APU rpm value invalid.

APU EGT	
Symbol	**Condition**
650	APU operating at or below redline.
960	APU operating above redline (overtemperature).
---	APU EGT invalid.

CS1_CS3_4991_011

Figure 53: APU Status Performance Indications

OPERATION

APU START

On the APU control panel, pull and turn the APU switch to the START position and release to RUN.

- APU inlet door opens
- On the STATUS synoptic page:
  - APU DOOR indicates open
  - APU RPM increases
- On EICAS:
  - APU IN START status message
- On the FUEL synoptic page:
  - APU feed SOV open
  - APU flow line green
  - APU fuel filter green

NOTE

APU IN START is displayed until APU reaches 70% rpm.

At 70% rpm, the following message is shown on EICAS:

- APU ON status message

At 100% rpm:

- On EICAS
  - APU ON status message
- On STATUS synoptic page:
  - RPM 91 to 100% (ground) 100% (flight)
  - EGT less than 950°C
  - DOOR OPEN
  - OIL TEMP NORM
  - OIL PRESS NORM
  - OIL QTY FULL

WARNING

BEFORE OPERATING THE APU, MAKE SURE THAT THE ACCESS DOORS ARE LOCKED IN THE OPEN OR CLOSED POSITION. DO NOT OPEN OR CLOSE THE ACCESS DOORS DURING APU OPERATION. THE APU AIR SUCTION CAN CAUSE THE DOORS TO CLOSE QUICKLY IF THEY ARE NOT LOCKED AND CAUSE INJURY TO PERSONS AND DAMAGE TO THE EQUIPMENT.

CAUTION

1. **The APU start from battery shall only be allowed if there are two batteries available and TRU 3 is not available. A battery start is allowed with one battery available and a TRU feeding the DC BUSES when TRU 3 is not available.**
2. **The APU fire extinguishing system is not effective when the access doors are open. If the APU is operated with the access doors open, make sure that fire fighting equipment is available.**

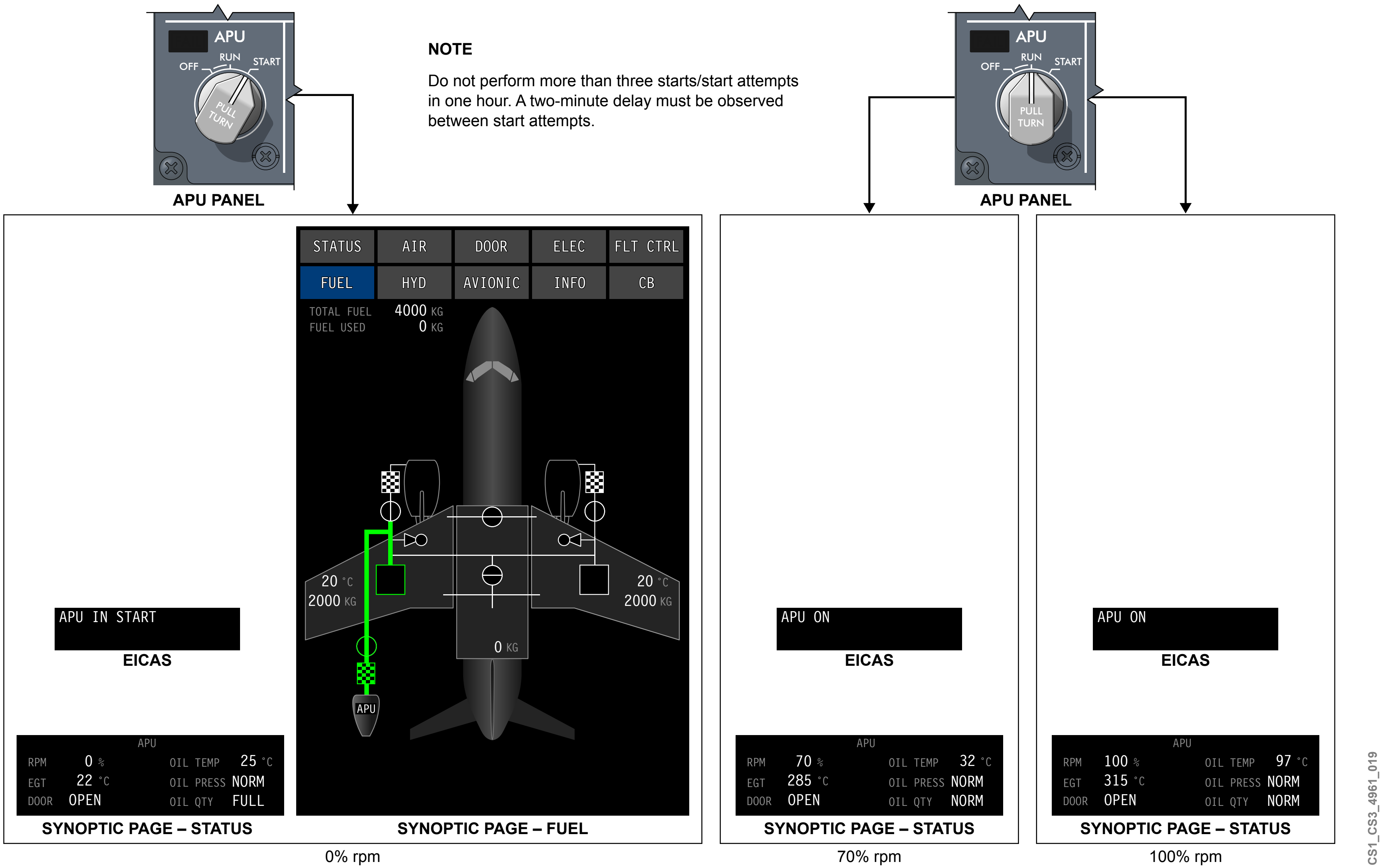


Figure 54: APU Start

NORMAL APU SHUTDOWN

The normal shutdown is carried out using the APU switch in the flight deck.

Before shutting down, remove the electrical and pneumatic loads.

On the APU control panel, pull and turn the APU switch to the OFF position. After one minute:

- APU stops, RPM decreases to 0%, and EGT decreases towards ambient temperature
- On the FUEL synoptic page, the APU SOV shows CLOSED, and the APU flow line shows white
- On the STATUS page DOOR CLOSED is shown in white

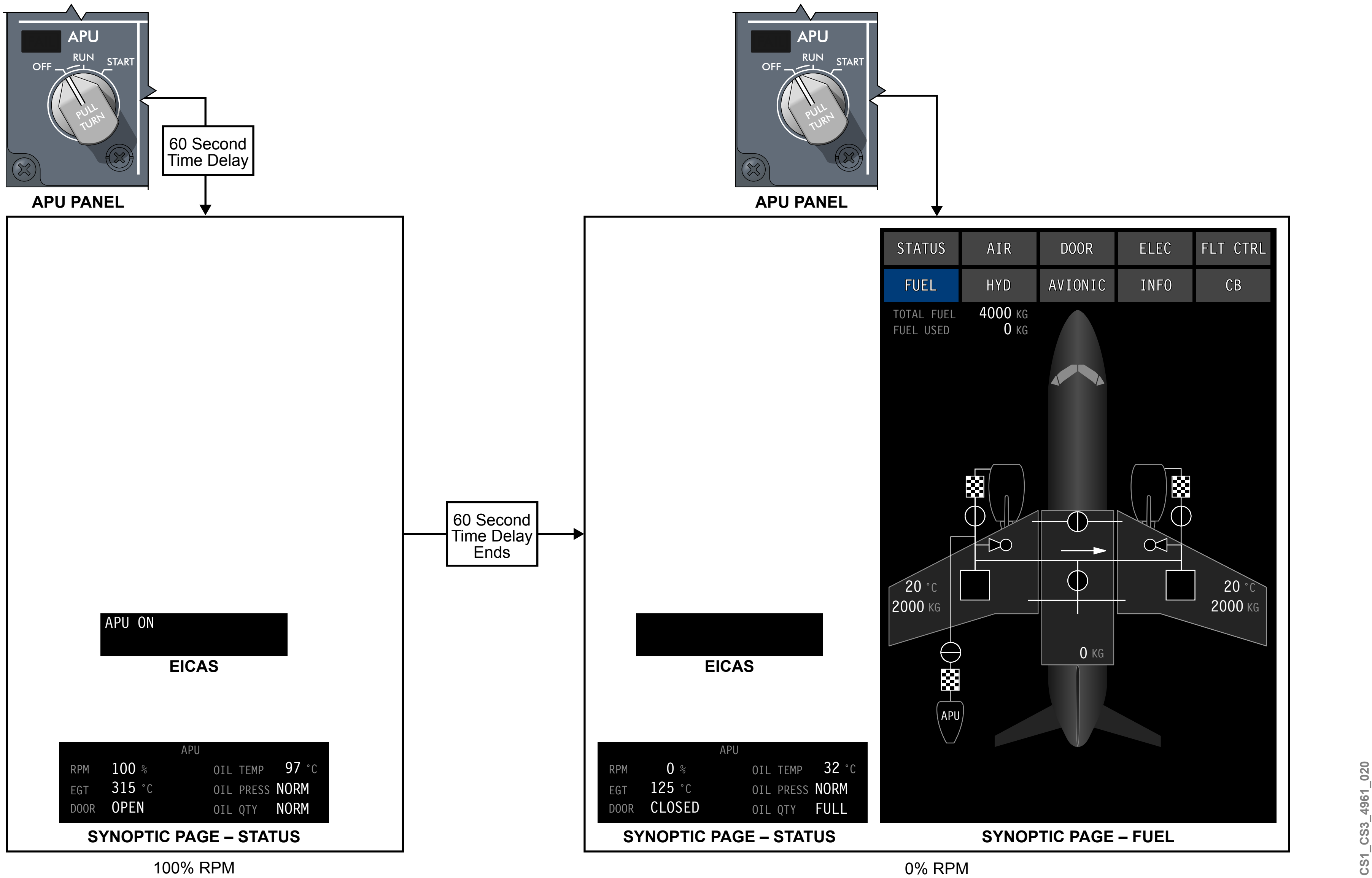


Figure 55: APU Normal Shutdown

EMERGENCY APU SHUTDOWN

The emergency shutdown is carried out from the external shut down switches or the APU FIRE PBA in the flight deck.

Pressing any emergency shutdown switch:

- Immediately APU stops with no cooldown period, RPM decreases to 0%, and EGT decreases toward ambient temperature
- On FUEL synoptic page, APU SOV shows CLOSED and APU flow line shows white
- On STATUS page DOOR CLOSED is shown in white

On the APU control panel, pull and turn the APU switch to the OFF position.

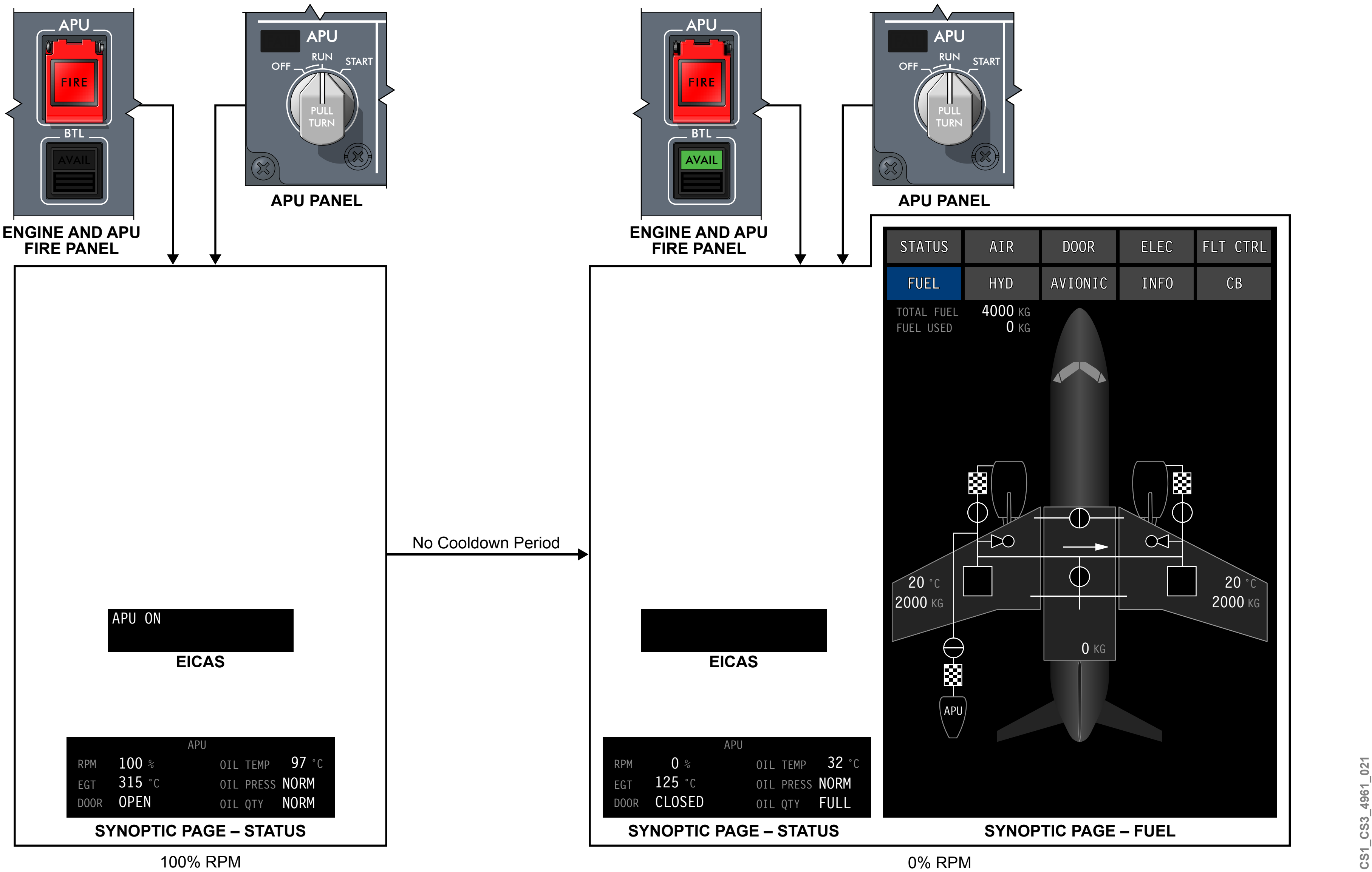


Figure 56: APU Emergency Shutdown

MONITORING AND TEST

The following page provides the crew alerting system (CAS) messages for the APU operation.

CAS MESSAGES

Table 11: STATUS Messages

MESSAGE	LOGIC
APU ON	APU is running.
APU IN START	APU performing startup sequence.

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ATA 71 - Power Plant

BD-500-1A10
BD-500-1A11

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POWER PLANT - CHAPTER BREAKDOWN

71-10 NACELLES

GENERAL DESCRIPTION

The engine cowlings minimize airflow turbulence around the engine. They provide an aerodynamically smooth surface around the engine and protect the engine mounted components.

The engine inlet cowl assembly directs the airflow to the engine fan. The inlet cowl is acoustically treated for noise reduction.

The fan cowls, installed between the inlet cowl and the thrust reverser doors, are attached to the pylon fan cowl support beam. Each inboard fan cowl has a strake installed to decrease the turbulence caused by the airflow between the fan cowl and the fuselage.

The thrust reverser doors enclose the core engine and contain the necessary components for the thrust reverser system. The thrust reverser doors attach to the hinge beam on the pylon. Both the fan cowls and thrust reverser doors can be opened to provide access to the engine.

The turbine exhaust system controls and directs the exhaust gases.

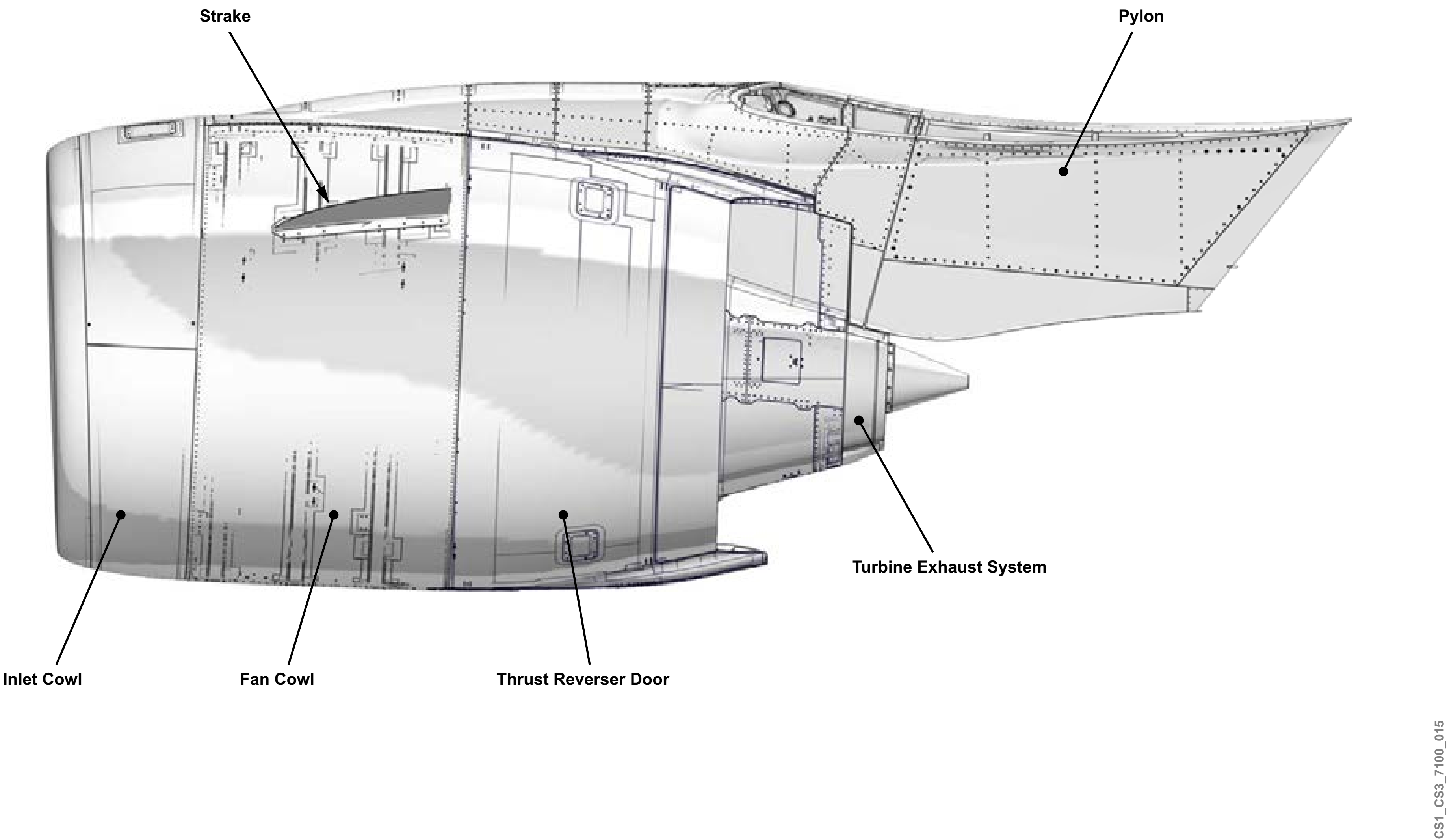


Figure 1: Nacelle Systems

71-60 INLET COWL

GENERAL DESCRIPTION

The inlet cowl provides smooth undisturbed airflow to the fan in all operating conditions, and reduced noise. The inner barrel of the inlet cowl is made of titanium and composite with 360° of interior acoustic treatment. Lightning strike protection is provided by a copper screen layer impregnated into the outer barrel.

The outer barrel is a two-piece composite structure extending from the inlet lip skin to the leading edge of the fan cowl, providing a smooth surface for airflow across the nacelle exterior. Two hoist points are provided for the removal of the inlet cowl.

A NACA scoop at the 1 o’clock position is used for fan compartment cooling. A panel, located at the 11 o’clock position, provides access to the P2T2 probe, sense line, and wiring harness.

The inlet cowl is anti-iced by bleed air. The air is exhausted out of the bottom of the inlet cowl through a series of slots. An access panel for the cowl anti-ice (CAI) duct is located at the 5 o’clock position. Drain holes in the core of the inner barrel and at the bottom of the outer barrel allow fluids to drain.

The inlet cowl is bolted to the front flange of the engine at the attachment ring.

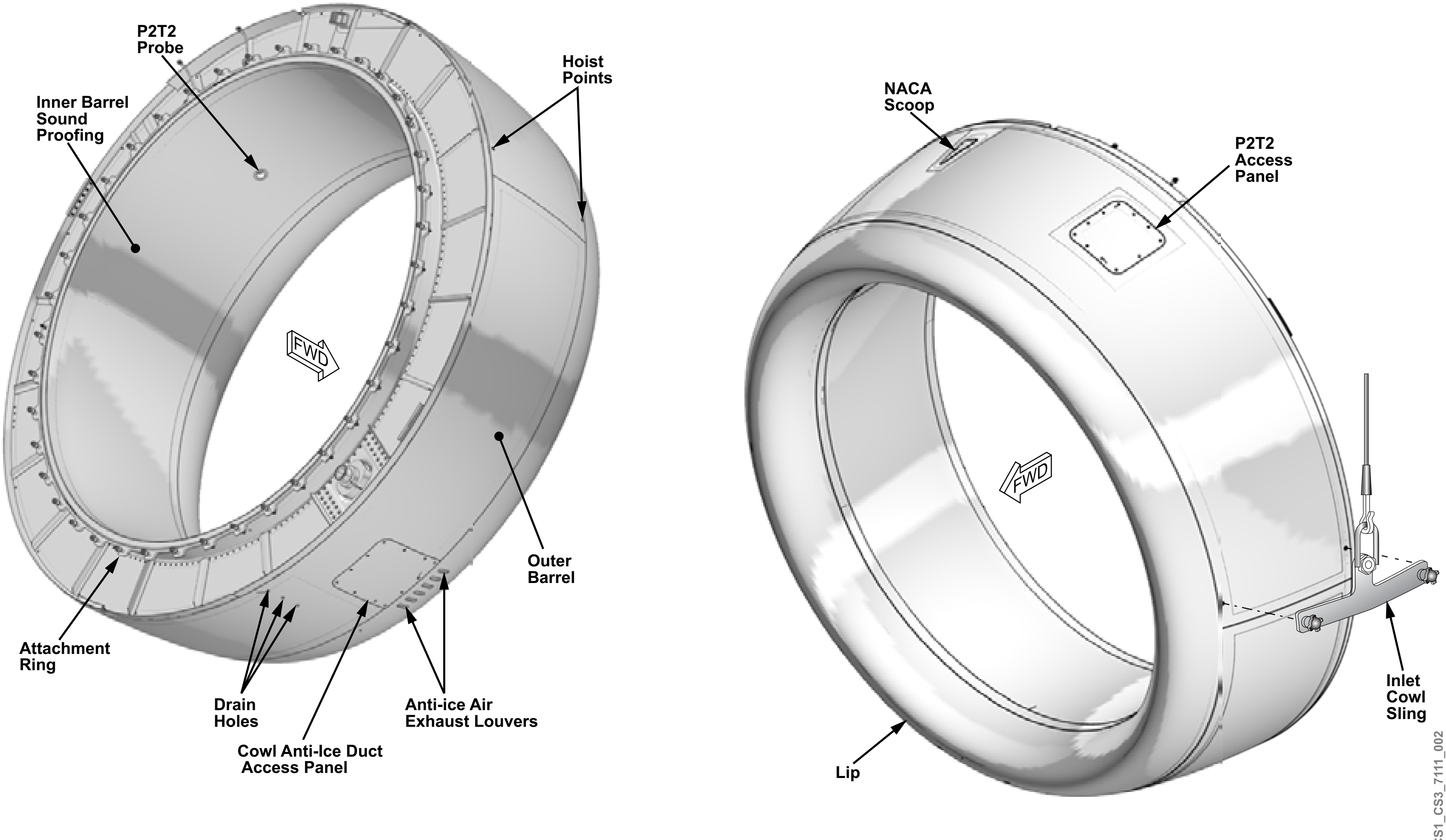


Figure 2: Inlet Cowl

71-11 FAN COWLS

GENERAL DESCRIPTION

The fan cowls cover the fan case, providing a protective enclosure for engine mounted components and a smooth aerodynamic outer surface for the nacelle. They are hinged at the top and latched at the bottom centerline with three flush mounted latches.

The fan cowl doors are manufactured using carbon fiber reinforced polymer (CFRP), strengthened with radial stiffeners and longerons. Copper mesh is embedded in the fan cowl laminate for lightning strike protection. The removable strakes are installed on the inner cowls only.

A pressure-relief door, located on the outer fan cowls, relieves high-pressure in the event of a duct failure. During maintenance, hold open rods (HORs) provide a means to support the cowls in the open position. Drain holes, located near the lower centerline, ensure fluid drainage.

Three hoist points of two screw holes are on each panel. They are used to attach the fan cowl door assembly to the ground handling sling. The ground handling sling is used to lift the cowl during removal or installation.

Wear strips ensures aerodynamic smoothness is maintained between the cowls.

Three hinges attach the cowls to the pylon. These hinges are secured by pins held in place by retaining clips. The pins facilitate removal of the fan cowl from the pylon.

Two HORs on each panel are used to hold the cowls in the open position. Quick-release fittings secure one end of the rods to the fan case when the cowls are opened. The other end of the rods are attached to a pivot bracket assembly. Stow brackets and clips on the cowl provide a means to secure the rods when the cowl is closed.

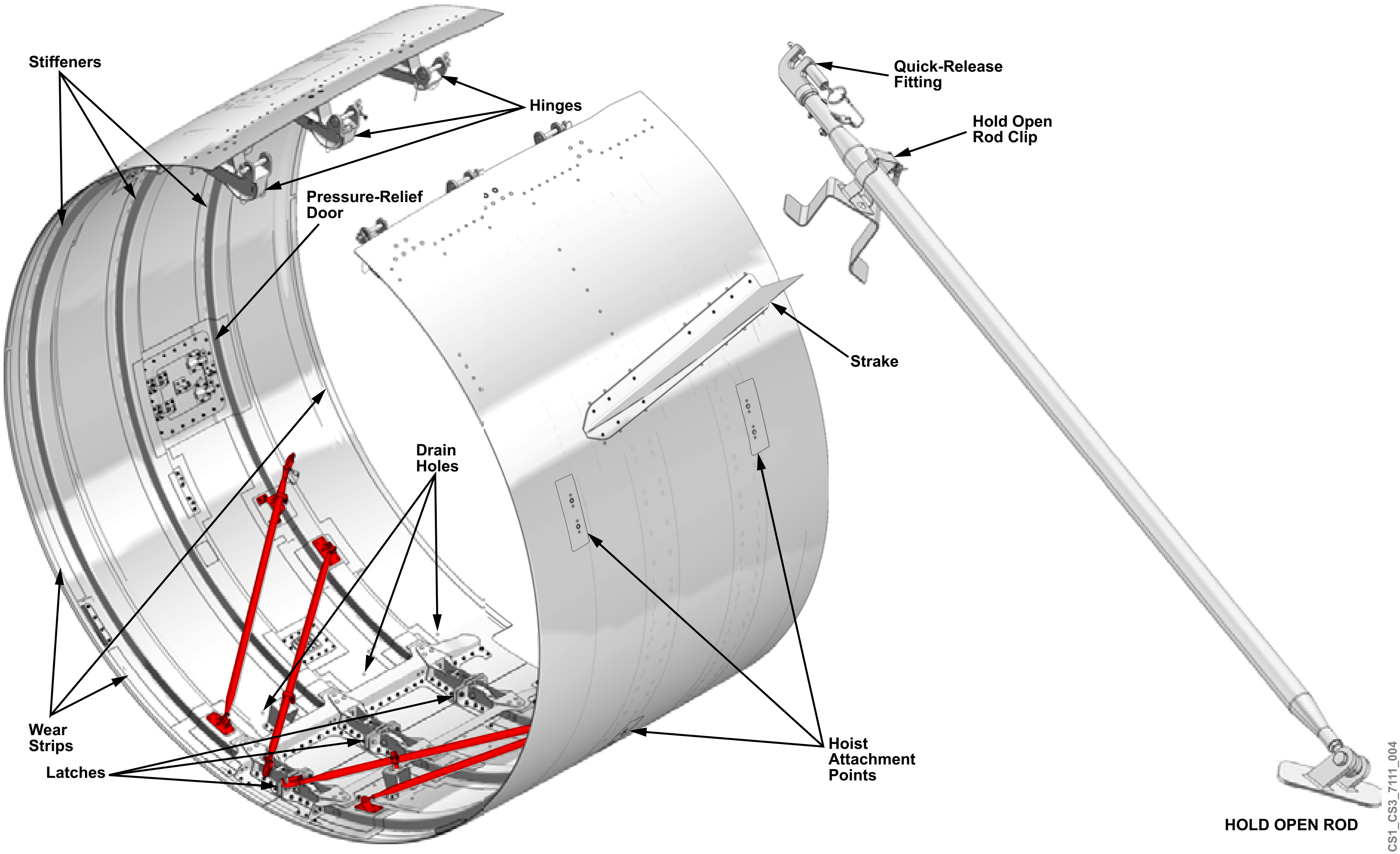


Figure 3: Fan Cowls

PRACTICAL ASPECTS

Always refer to the latest Aircraft Maintenance Publication (AMP) before carrying out any procedures on the aircraft.

OPEN THE FAN COWL

To open the fan cowl:

- Push the fan cowl door latch triggers to release the three latch handles. Pull down each handle to open the three latch handles
- Move the latches away from the three latch keepers in the sequence L1, L3, and L2
- Manually lift and hold a fan cowl door at the lower edge
- Remove the forward hold open rod (HOR) from the fan cowl and install it in the forward HOR bracket on the fan case. Secure it with the quick-release pin on the end of the HOR
- Remove the aft HOR from the fan case and secure it to the power door operating system (PDOS) bracket on the fan case. Secure it with the quick-release pin on the end of the HOR
- Slowly lower the fan cowl door until the HORs hold the weight of the door

CLOSE THE FAN COWL

To close the fan cowl:

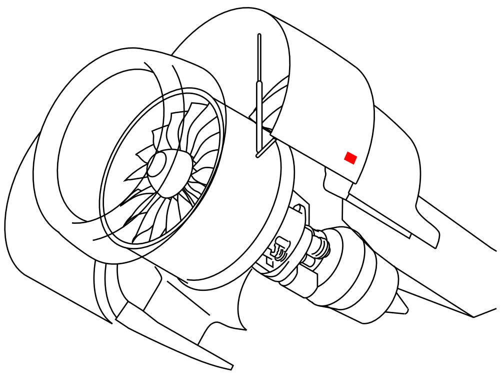
- Manually lift and hold the fan cowl door at the lower edge so that the weight is not on the HORs
- Remove the aft HOR from the PDOS bracket on the fan case and install it on the fan cowl. Secure the HOR with the quick-release pin
- Remove the forward HOR from the forward HOR bracket on the fan case and install it on the fan cowl. Secure the HOR with the quick-release pin
- Put the forward HOR on to the HOR clip. Secure the HOR with the snap spring of the clip
- Carefully lower the fan cowl and push the fan cowl door together at the bottom centerline. Make sure the alignment pins enter the holes adjacent to the latches
- Engage the hooks on all three latches with the eye-bolts on the three latch keepers in the sequence L1, L3, L2
- Close the three latch handles until they are flush with the door surface and locked in position. When the latches are properly closed, the red marks on the latches do not show

WARNING

1. **TWO OR MORE PEOPLE ARE REQUIRED TO OPEN THE FAN COWL DOOR. THE FAN COWL DOOR WEIGHS MORE THAN 45.36 KG (100 LB). IF THE DOOR FALLS, INJURY AND/OR DAMAGE TO THE ENGINE OR PERSON(S) CAN OCCUR.**
2. **DO NOT KEEP OPEN A FAN COWL DOOR WHEN THE WIND SPEED IS OVER 96 KM/H (60 MPH). IF THE WIND MOVES THE FAN COWL DOOR, INJURY AND/OR DAMAGE TO THE ENGINE OR PERSON(S) CAN OCCUR.**
3. **CAUTION IS REQUIRED IF YOU OPEN A FAN COWL DOOR WHEN THE WIND SPEED IS 74 KM/H (46 MPH) OR MORE. IF THE WIND MOVES THE FAN COWL DOOR, INJURY AND/OR DAMAGE TO THE ENGINE OR PERSON(S) CAN OCCUR.**

CAUTION

Lifting the fan cowl door more than 55° vertical can cause fan cowl door damage or pylon damage.



FAN COWL OPEN (VIEW LOOKING FORWARD)

A

CS1_CS3_7111_010

Figure 4: Fan Cowl Opening and Closing

FAN COWL LATCH AND KEEPER ADJUSTMENT

The fan cowl latches require adjustment anytime the fan cowl does not close properly and the latches are safely engaged.

The latches engage adjustable latch keepers on the opposite cowl. If the fan cowl latches require adjustment, the keepers are adjusted by inserting an allen key into the star wheel adjustment nut.

The latches are adjusted to provide a closing force of 9.1 to 10.9 kgf (20 to 24 lbf).

Shim plates on latches and keepers are used to adjust the cowl gap. Make sure the gap between the left and right fan cowls is 0.51 - 5.08 mm (0.020 - 0.200 in.).

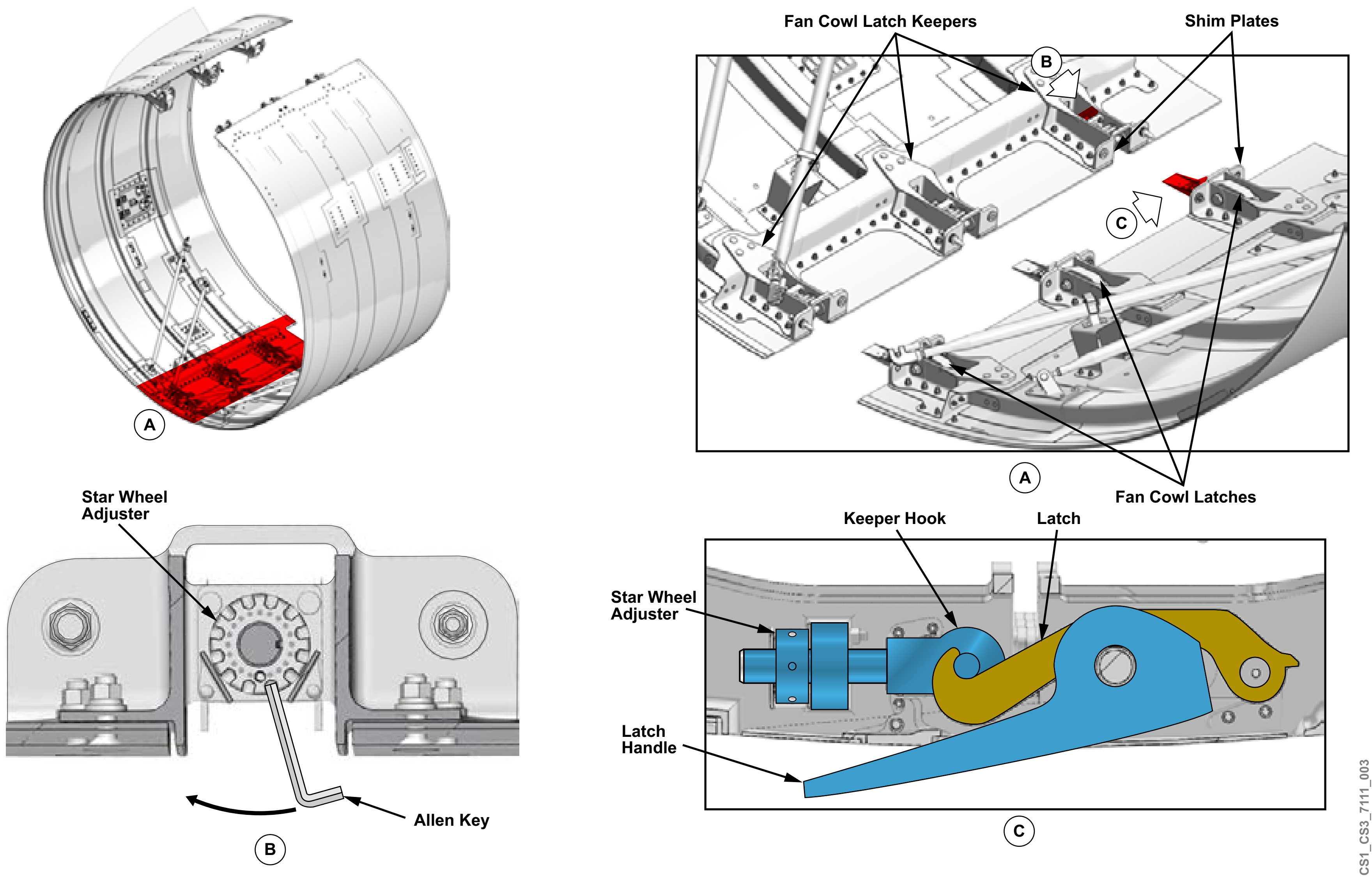


Figure 5: Fan Cowl Latch and Keeper Adjustment

78-30 THRUST REVERSER DOORS

GENERAL DESCRIPTION

The power plant is equipped with a cascade type thrust reverser (T/R) assembly consisting of left and right T/R doors. The left and right T/R doors are installed on a hinge beam on the pylon. The hinge beam has four attachment points. The T/R doors are latched together at the bottom of the engine and form an enclosure for the core engine.

The T/R doors can be opened using the power door operating system (PDOS), or manually using a hydraulic pump. When opened, the engine core can be accessed.

Each T/R door consists of two fixed structures that direct fan air aft. The inner structure forms an aerodynamic surface from the fan case exit to the aft nozzle. The outer structure contains the T/R translating sleeve, cascades, and blocker doors that provide reverse thrust. The inner and outer fixed structures are connected by the torque box. The torque box is the attachment point for the T/R components and the PDOS.

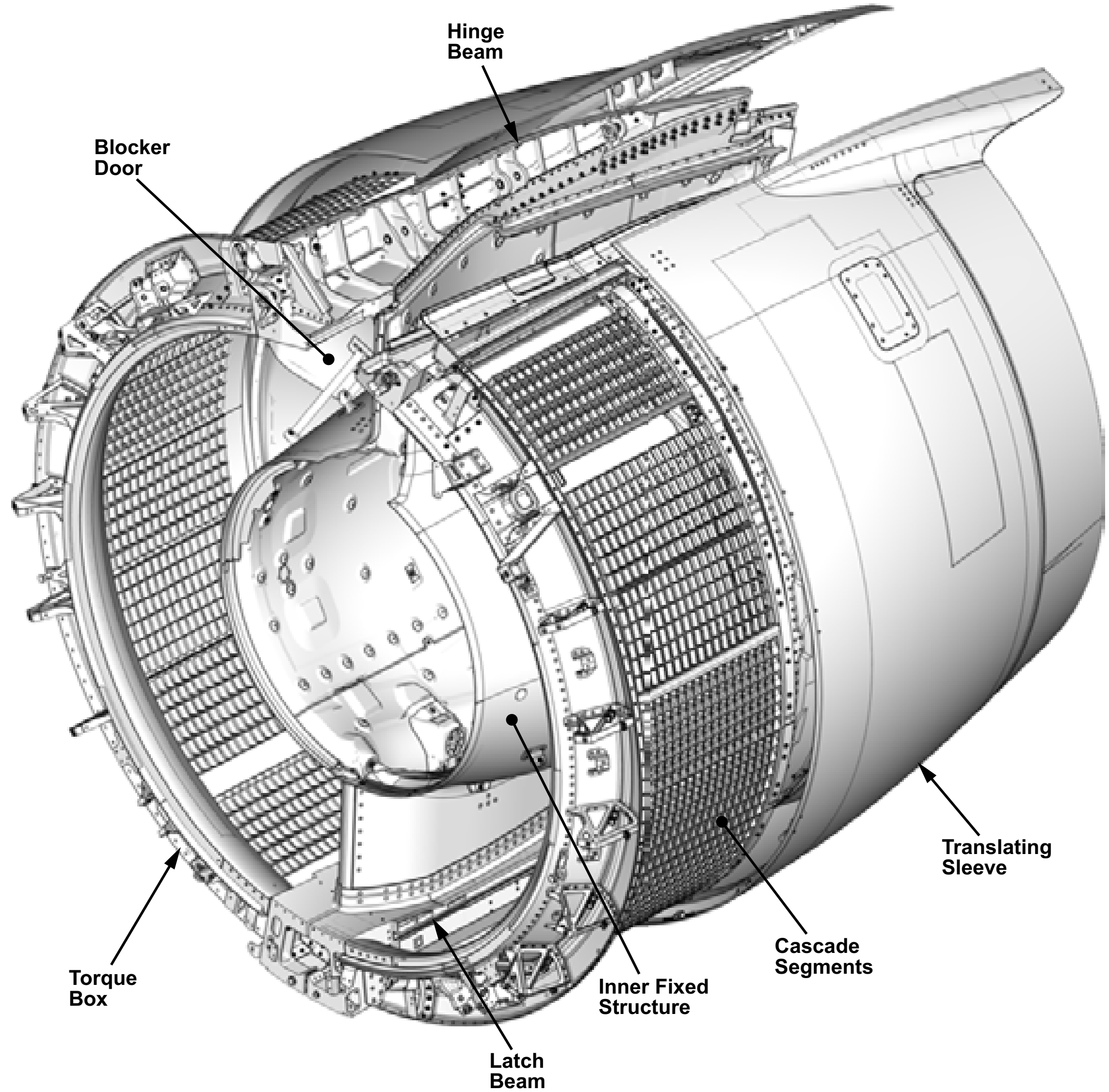


Figure 6: Thrust Reverser Doors

THRUST REVERSER FIXED STRUCTURE

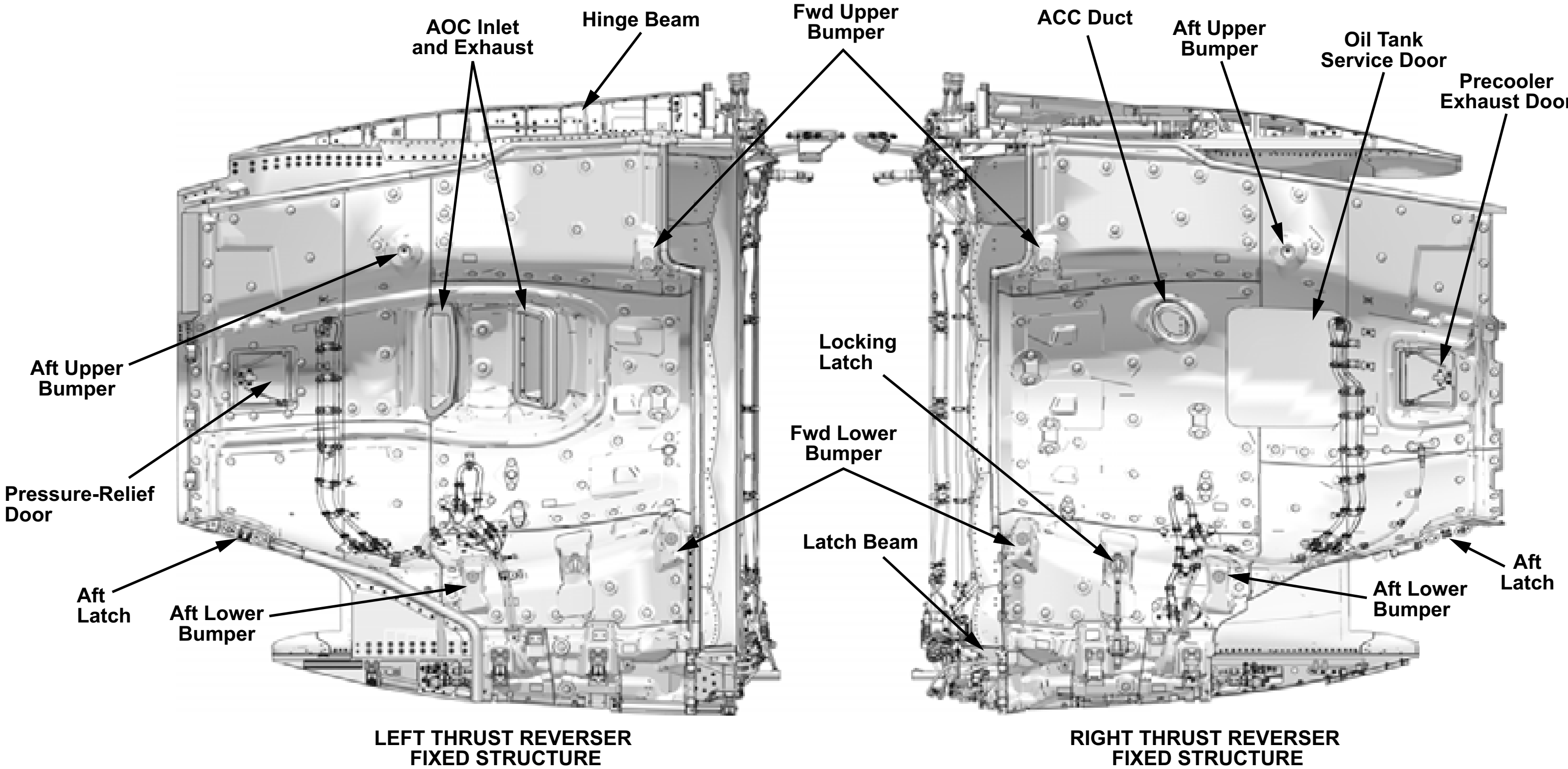
The thrust reverser fixed structure covers the engine core. It provides core compartment ventilation, and forms the fan duct inner aerodynamic surface from the fan case exit to the exhaust nozzle. The thrust reverser attaches to the pylon at the hinge beam.

An air/oil cooler (AOC) inlet and exhaust is located on the left side of the fixed structure. A pressure-relief door, located behind the AOC, opens in the event of a burst duct in the bleed air system. The pressure relief door is located on the left side of the inner fixed structure behind the air/oil cooler (AOC) exhaust, at the 9 o'clock position. The latches release at 3.5 psi. The door is manually closed if it has opened.

The right side of the fixed structure has an oil tank service door and a cutout for the active clearance control (ACC) ducting. A precooler exhaust door is located behind the oil tank service door.

Forward and aft bumpers, located at the top and bottom of the structure, are used to align the doors as they are closing. A locking latch is used to ensure the doors remain closed in the aligned position.

Latches, installed on the latch beam, are used to secure the T/R door in the closed position.



LEGEND

ACC	Active Clearance Control
AOC	Air/Oil Cooler

Figure 7: Thrust Reverser Fixed Structure

THRUST REVERSER DOOR LATCHES

Latches are quick-release mechanisms integral to the latch beam that provide quick access to the engine core. The latches provide resistance to loads that might otherwise cause the T/R to disengage or open during the flight cycle.

Latches no. 1, no. 2, no. 3, and no. 4 are installed on the latch beam. Latch no. 5 is located on the lower aft section of the inner fixed structure. The latches are takeup latches that pull the two T/R halves together when securing.

Latches no. 2 and no. 3 are located inside the latch access door. The latch access door cannot be closed if latch 2 and latch 3 are not properly secured.

A closure assist assembly consists of a turnbuckle, which is used to draw the door together before engaging the latches. An eye on one end of the turnbuckle allows it to pivot in order for the pin to engage the opposite cowling. When engaged, the turnbuckle is turned manually to draw the two doors together. The closure assist assembly is stowed after use.

CAUTION

1. **Do not turn the body of the closure assist assembly to make it shorter than the minimum length. The closure assist assembly can be damaged if the minimum length is decreased.**
2. **Adjust the closure assist assembly to the minimum length before placing the retention clip of the stow bracket assembly. The closure assist assembly can move out of position if this step is not followed.**

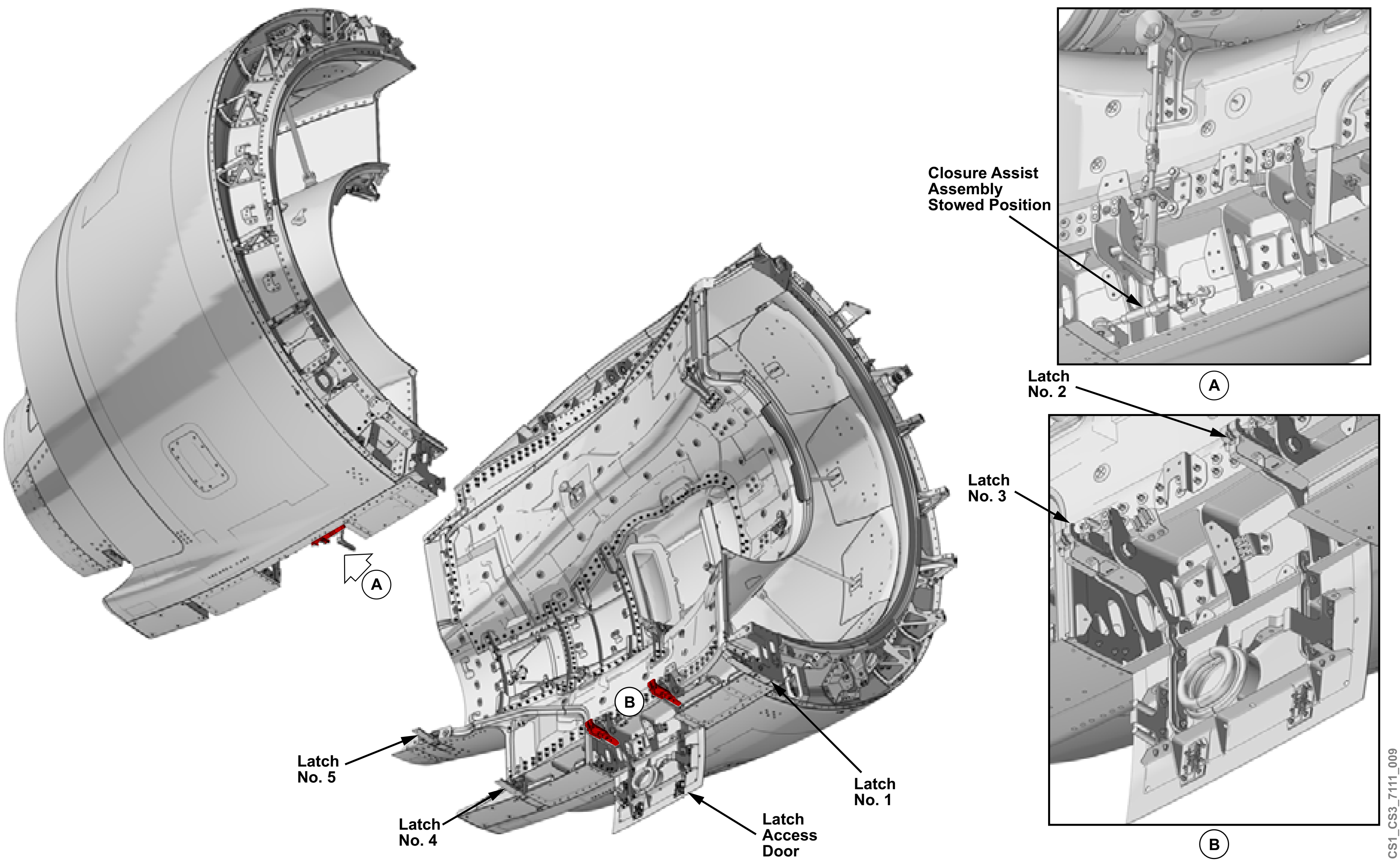


Figure 8: Thrust Reverser Door Latches

Bumpers and Bifurcation Latch System

To maintain a correct fit of the ducts to the engine, the inner fixed structure (IFS) has upper and lower bumpers to assist alignment as the two thrust reverser doors are closed.

The upper bumper includes a compression strut to compensate for the larger gap between the two cowls.

The bifurcation latch system (BLS) is similar to the bumpers but has a locking feature to keep it engaged. The handle is accessed through the latch access door. The BLS is used to limit the deflection of the IFS, and maintain its structural integrity in the event of a burst air duct.

When engaged, the bifurcation latch uses a rod and cam mechanism to insert a locking pin through the latch. As the thrust reverser (T/R) doors close, a pin on the left side is inserted into a receiver located on the right side. The pin mechanism is locked together using a rod. The rod is inserted through the pin by turning, and pushing the BLS lever up.

The bifurcation latch lever is painted red, providing a visual indication of the position. If the bifurcation latch lever is not locked, the latch access panel cannot be closed.

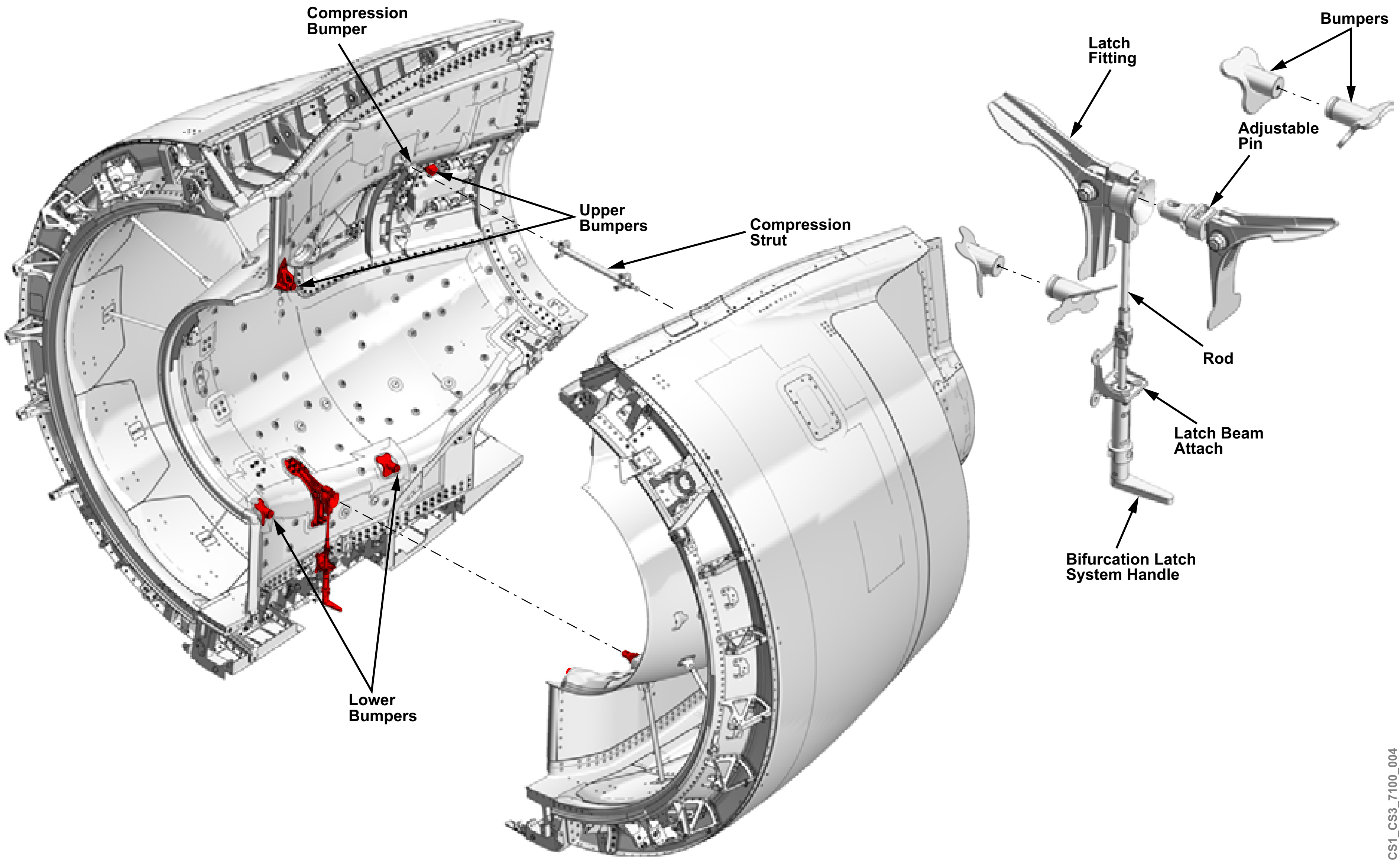


Figure 9: Bumpers and Bifurcation Latch System

POWER DOOR OPERATING SYSTEM

The thrust reverser (T/R) power door operating system (PDOS) is used to open the T/R doors and support them in the open position. Each T/R door is opened independently of the other. The PDOS powerpack consists of a hydraulic gear pump driven by an AC motor, and a reservoir capable of supplying fluid to two locking actuators. The PDOS is operated by switches located on each side of the fan case.

Each T/R door is raised and lowered using a locking actuator connected between the engine fan case and the T/R door torque box. The locking actuator has an internal locking mechanism that is engaged when the actuator is fully extended. A hold open rod (HOR) provides additional support for the T/R door. The locking actuator takes 100% of the load under normal operation, with the HOR as a failsafe mechanism.

Each locking actuator is connected to the hydraulic powerpack using a hydraulic line with a quick-disconnect fitting. If the powerpack fails, the quick-disconnect fitting can be removed from the locking actuator and a hand pump connected in its place.

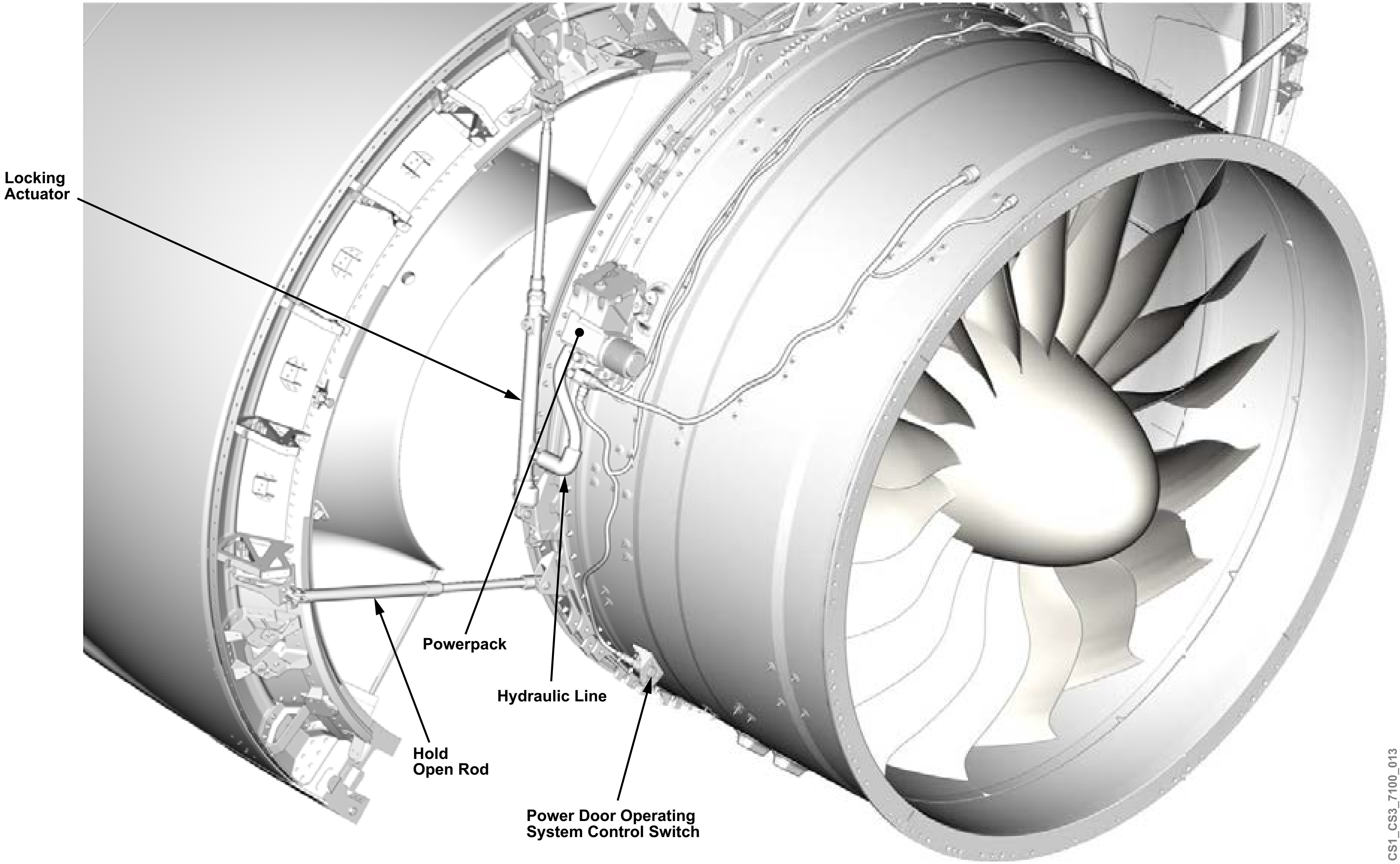


Figure 10: Power Door Operating System

Power Door Operating System Control

The PDOS can be operated when the aircraft is on the ground and the engine is shut down. The left engine powerpack motor is powered by 115 VAC from AC BUS 1. The PDOS operates when the thrust reverser (T/R) door switch is pressed. Only one door can be opened at a time. The switch supplies 28 VDC from DC BUS 1 to energize solenoids within the powerpack. When the solenoid is energized, the powerpack supplies 2300 psi hydraulic pressure to the locking actuator. The right engine PDOS motor is powered from AC BUS 2, and the control power is supplied from DC BUS 2.

When the actuator fully extends, a pressure-relief valve (PRV) in the actuator opens to ensure the maximum system pressure is not exceeded. The T/R door switch is held for 5 seconds after full actuator travel, then released. The PRV closes, and the internal lock engages and holds the door open.

To close the door, the T/R door switch is held. The actuator extends slightly and the internal lock releases. When the lock is released, the switch is released. The actuator retracts due to the weight of the T/R door. An internal restrictor limits the rate of closing.

If no power is available, a hand pump can be connected to the actuator quick-disconnect fitting to open the door.

CAUTION

1. **The inboard thrust reverser doors can contact the slats if the slats are in the extended position.**
2. **The actuators can be opened manually using a hand pump. The introduction of additional fluid in the system may cause the powerpack reservoir to overflow.**

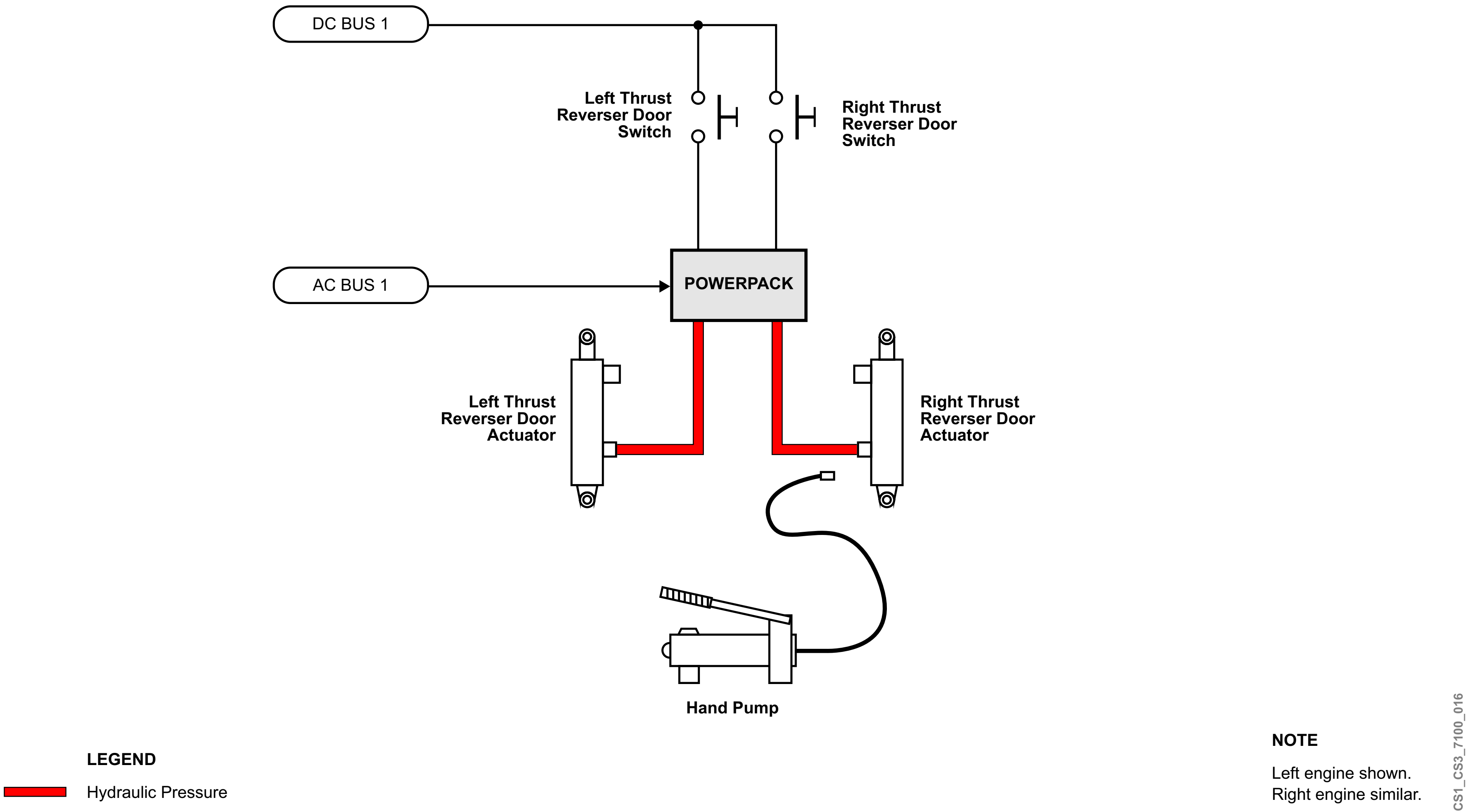


Figure 11: Power Door Operating System Control

COMPONENT LOCATION

The power door operating system (PDOS) consists of the following components:

- Powerpack
- Locking actuators
- Control switches
- Hold open rods (HORs)

POWERPACK

The powerpack is located on the fan case at the 10 o’clock position.

LOCKING ACTUATORS

The locking actuators are located on the fan case at the 3 o’clock and 9 o’clock positions.

CONTROL SWITCHES

The control switches are located on the fan case at the 4 o’clock and 8 o’clock positions.

HOLD OPEN RODS

The hold open rods (HORs) are installed on each thrust reverser (T/R) door torque box.

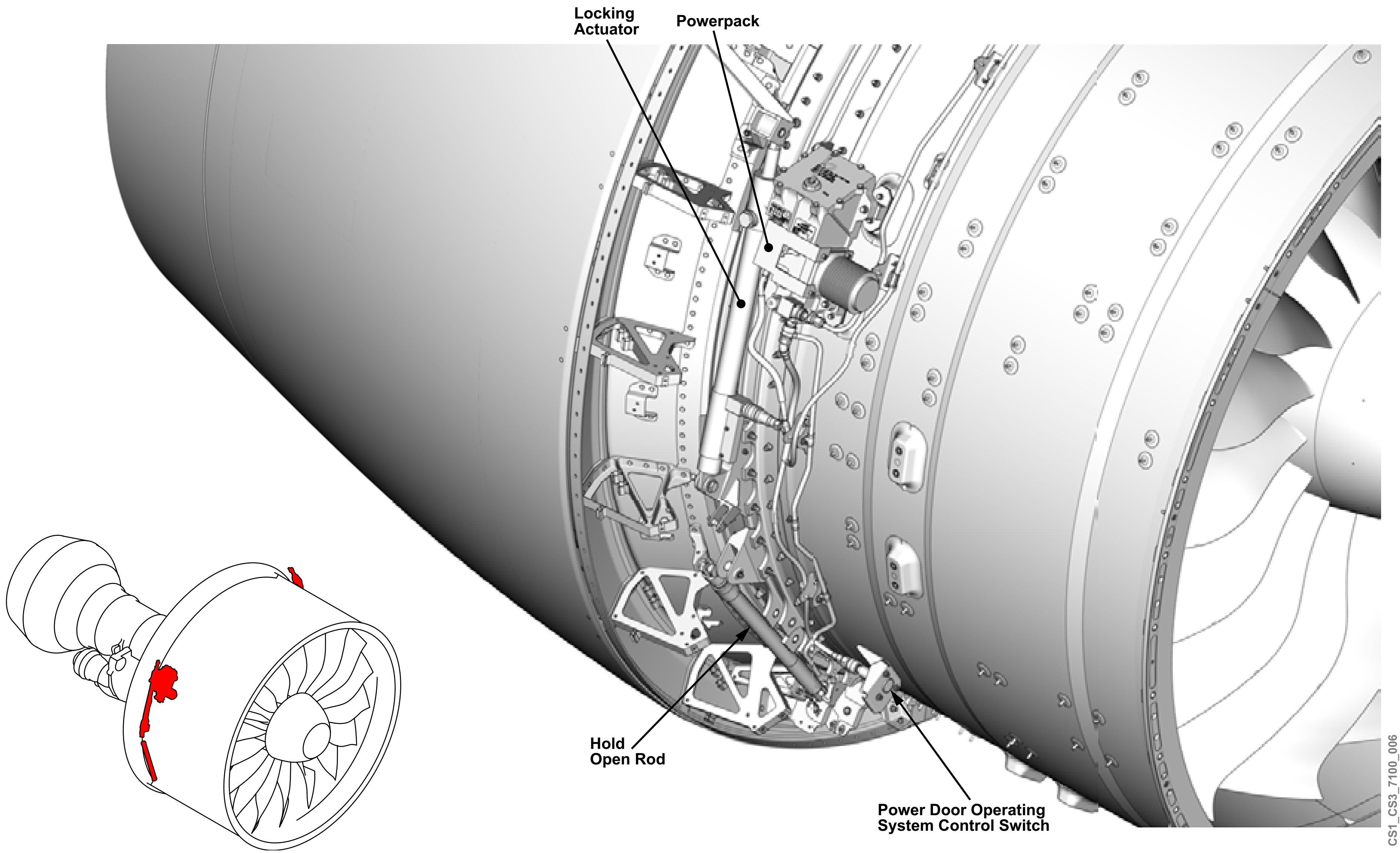


Figure 12: Power Door Operating System Component Location

COMPONENT INFORMATION

POWERPACK

The self-contained power door operating system (PDOS) is powered by an electrohydraulic powerpack producing 2300 psi. The powerpack houses an AC motor and a gear pump assembly, which provides hydraulic pressure to two actuators during system operation.

The powerpack includes a reservoir, solenoid valves, filter, and a filler cap/strainer. A drain/vent port on the lower half of the powerpack provides venting and overflow drainage.

The system is serviced with engine oil. A filler cap on the reservoir lid is used to replenish the oil.

A dipstick is provided under the filler cap for checking the oil level. A strainer prevents any contaminants from entering the reservoir. Oil levels are marked on the dipstick as follows:

- L - Low 0.74 L
- N - Normal 0.82 L
- H - High 0.89 L

CAUTION

Before servicing the PDOS powerpack, the thrust reverser (T/R) doors should be fully closed.

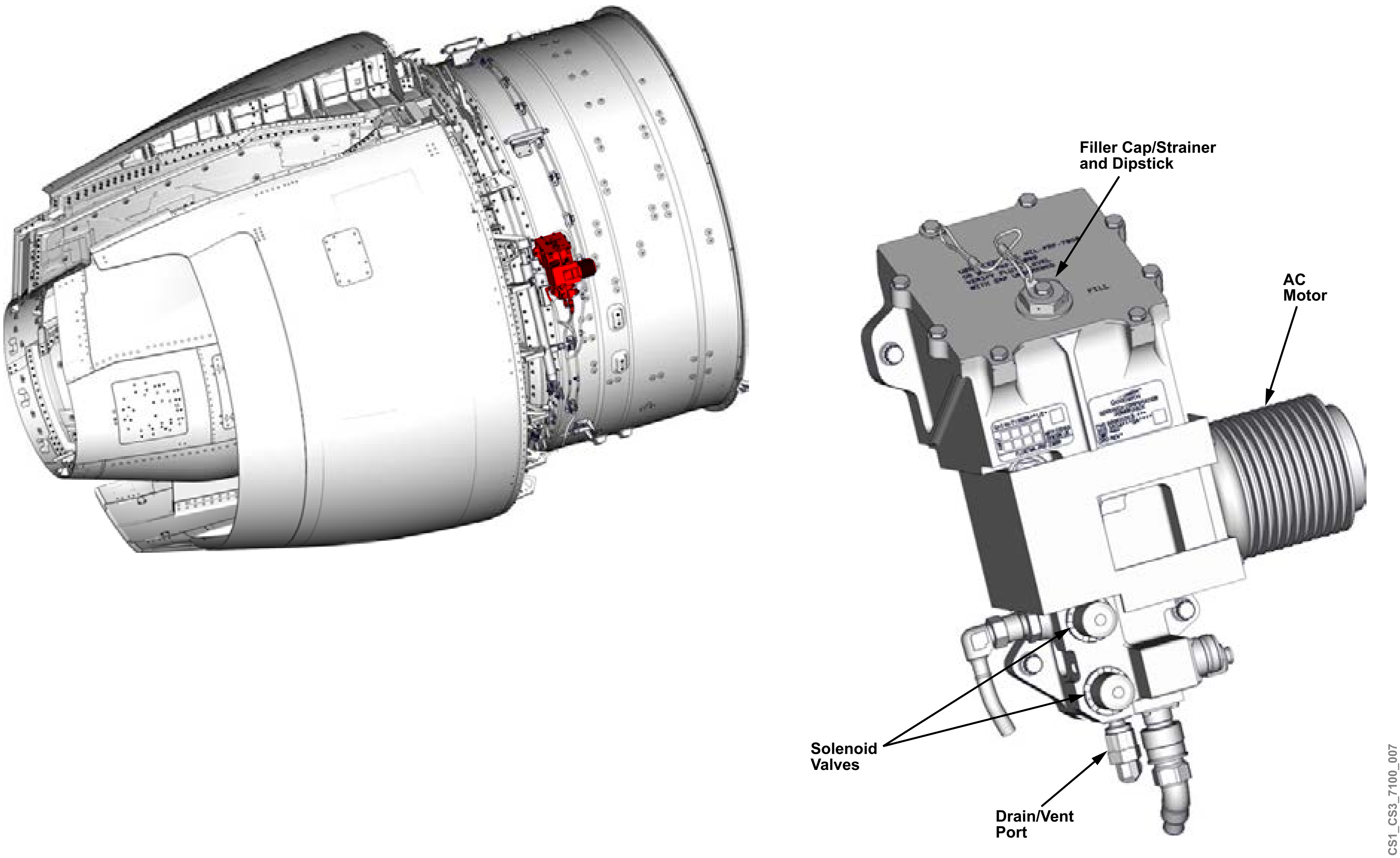


Figure 13: Power Door Operating System Powerpack

LOCKING ACTUATORS

Two locking actuators are used independently to open each thrust reverser (T/R) door.

Each actuator has a flow control valve (FCV), quick-disconnect connector, pressure-relief valve (PRV), mechanical lock, and a manual bleed feature.

The actuators internal flow control valve provides the system with the means to control the closure rate of the cowl when the lock is released. Both actuators have quick-disconnect fittings to accommodate a hand pump in the event of a powerpack failure. When opened, an automatic mechanical lock inside the actuator provides positive retention of the C-duct. A manual bleed port on the actuator is used to expel air from the system.

HOLD OPEN RODS

Each T/R door has a telescopic hold open rod (HOR) which is attached between the T/R door and the fan case to help hold the T/R doors open. Lock collars are used to extend and retract the HOR. The HOR is used in conjunction with the locking actuator to hold the T/R door open, and provide a safe environment during maintenance.

When not in use, the HORs are stored on the T/R door torque box.

WARNING

1. **MAKE SURE THAT THE HOLD OPEN RODS (HORS) ARE INSTALLED. FAILURE OF THE POWER DOOR OPERATING SYSTEM (PDOS) CAN CAUSE THE THRUST REVERSER (T/R) TO CLOSE IF THE HORS ARE NOT INSTALLED. THIS CAN CAUSE INJURY TO PERSONNEL OR DAMAGE TO EQUIPMENT.**

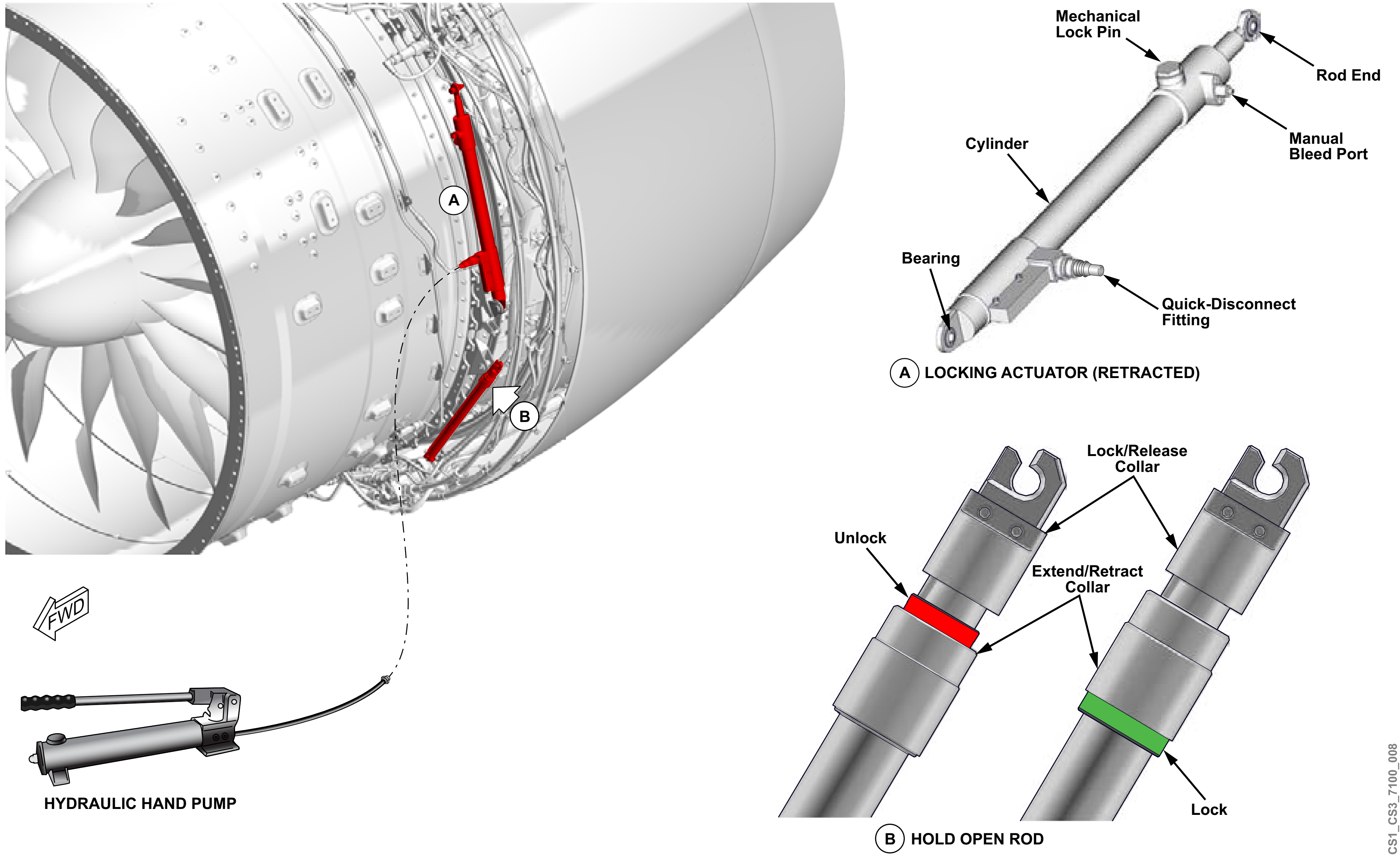


Figure 14: Power Door Operating System Locking Actuator and Hold Open Rods

DETAILED DESCRIPTION

The power door operating system (PDOS) is powered when the aircraft is on the ground and the engine is shut down. For the left engine, the following logic is supplied to the control and distribution cabinet 1 (CDC 1).

- Weight-on-wheels (WOW) signal from the landing gear and steering control units (LGSCUs)
- Engine shutdown signal from the electronic engine control (EEC)

When the thrust reverser (T/R) door switch is pressed, the solenoid valve is energized, allowing hydraulic fluid to be supplied to the hydraulic pump.

When solenoid is energized, logic turns on the L (R) PDOS Motor SSPC and the motor drives the hydraulic pump. The thrust reverser door hydraulic actuator extends to open the door. If the motor should overheat, a thermal switch shuts the motor off.

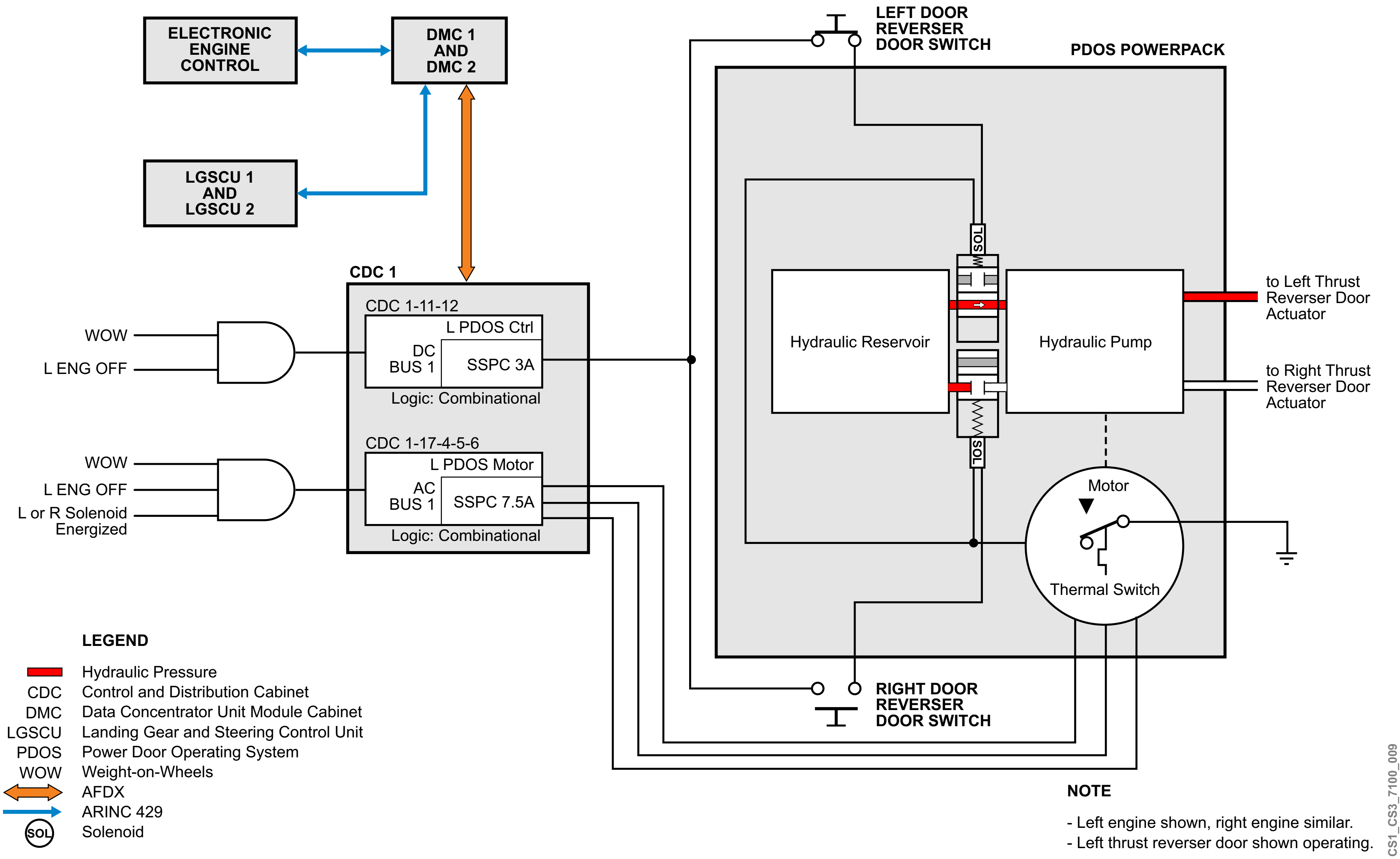


Figure 15: Power Door Operating System Schematic

PRACTICAL ASPECTS

THRUST REVERSER DOOR BIFURCATION LATCH SYSTEM AND LATCH OPERATION

Open the Bifurication Latch System

1. Open the latch access door.
2. Turn the bifurication latch system (BLS) latch handle 90° clockwise.
3. Pull the BLS latch handle down until it comes to a stop.
4. Turn the BLS latch handle 90° counterclockwise.

Open the Thrust Reverser Door Latches

1. Remove the closure assist assembly from the stow bracket assembly and adjust it to its minimum length.
2. Engage the closure assist assembly in the hook on the fixed panel support on the other thrust reverser (T/R) door.
3. Turn the body of the closure assist assembly with a wrench to pull the two T/R doors together and relieve the tension on the T/R latches.
4. Open the T/R door latches in the sequence that follows.
   - LATCH No. 5
   - LATCH No. 4
   - LATCH No. 3
   - LATCH No. 2
   - LATCH No. 1
5. Put the closure assist assembly into its stowed position. Ensure the closure assist assembly is at its minimum length.

CAUTION

The closure assist assembly must be at its minimum length before being put into the retention clip of the stow bracket assembly. If this is not done, the closure assist assembly can move out of position.

Close the Thrust Reverser Door Latches

1. Make sure that the T/R doors are fully closed.
2. Adjust the closure assist assembly so that it can engage the hook on the fixed panel support on the other T/R half.
3. Turn the body of the closure assist assembly with a wrench to pull the two T/R halves closer together.
4. Close the T/R door latches in the sequence that follows:
   - LATCH No. 1
   - LATCH No. 2
   - LATCH No. 3
   - LATCH No. 4
   - LATCH No. 5
5. Put the closure assist assembly into its stowed position

Close the Bifurication Latch System

1. Turn the BLS latch handle 90°clockwise.
2. Push the BLS latch handle up until it comes to a stop.
3. Turn the BLS latch handle 90° counterclockwise.
4. Close the latch access door.

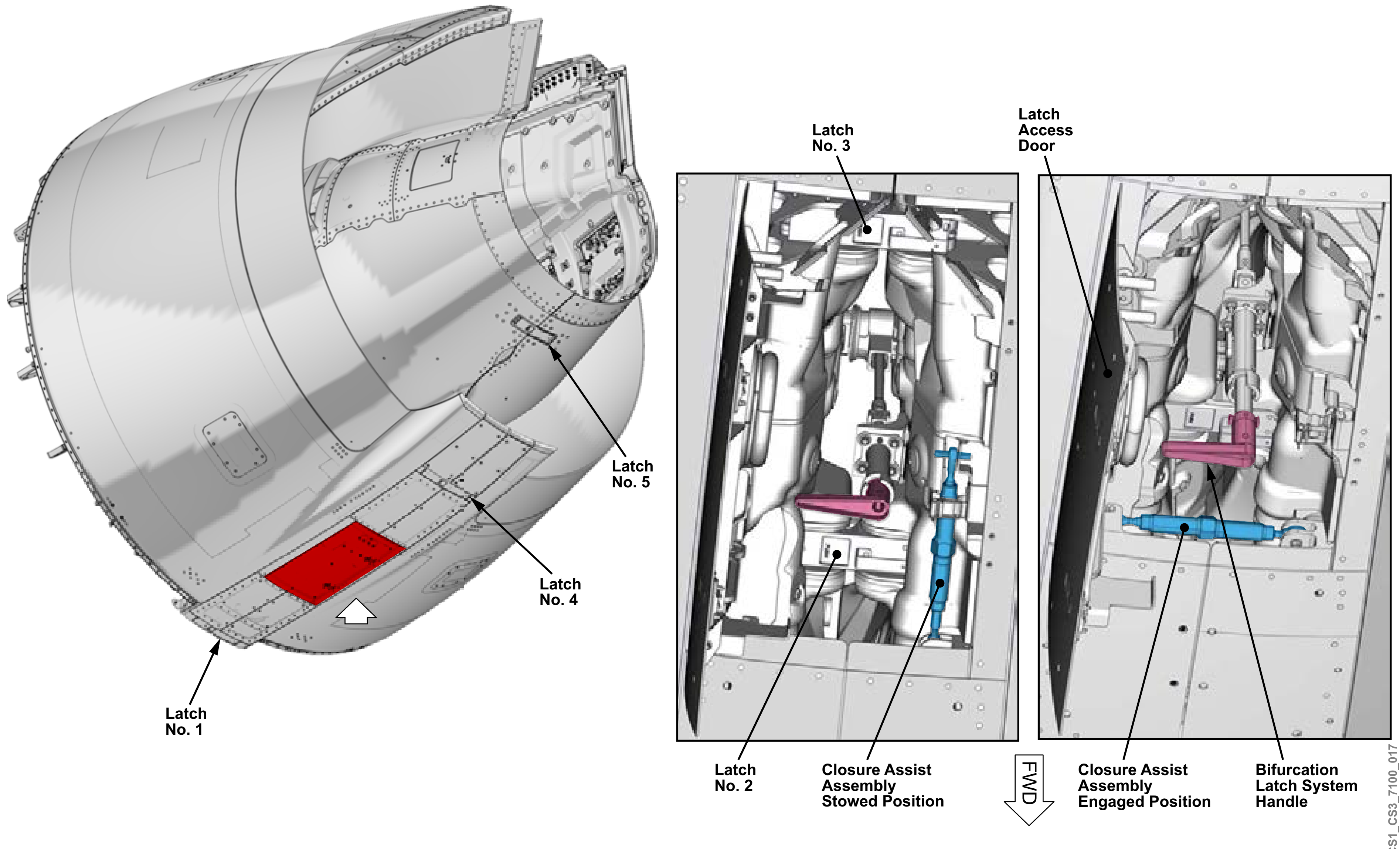


Figure 16: Thrust Reverser Door Bifurcation Latch System and Latch Operation

THRUST REVERSER DOOR OPENING AND CLOSING

Open the Thrust Reverse Door

1. Push PDOS operating switch until the locking actuator is fully extended.
1. Hold the switch for a minimum of 5 seconds after the locking actuator is fully extended and the powerpack pressure-relief valve (PRV) opens.
2. Release the switch and the locking actuator retracts in a controlled descent until the mechanical lock is engaged.
3. Release the top end of the hold open rod (HOR) from the HOR stow bracket and install it in the HOR open fitting on the fan case.

WARNING

1. **DO NOT MOVE BETWEEN THE ENGINE AND THE OPEN THRUST REVERSER (T/R) UNTIL THE THE HOLD OPEN ROD (HOR) IS INSTALLED. THE T/R IS HEAVY AND CAN CLOSE QUICKLY IF THE PDOS FAILS.**
2. **TO PREVENT INJURY TO PERSONNEL OR DAMAGE TO EQUIPMENT, CAUTION IS REQUIRED WHEN OPENING THE T/R IN HIGH WIND.**

Close the Thrust Reverser Door

1. Release the HOR from the HOR open fitting on the fan case and install the HOR in the HOR stow bracket.
2. Push the PDOS operating switch until the locking actuator is fully extended.
3. Hold the switch for a minimum of 5 seconds after the locking actuator is fully extended and the powerpack PRV opens.
4. Release the switch. The powerpack PRV closes and the locking actuator mechanical lock disengages.
5. Make sure that the thrust reverser outer V-groove blade engages with the outer V-groove on the engine during closing.
6. Make sure that the alignment pins correctly engage the receptacles before the thrust reverser doors are fully closed.

CAUTION

Make sure that the T/R half area is clear of tools and equipment before closing the door. Failure to do so can cause damage to the T/R half and to the engine.

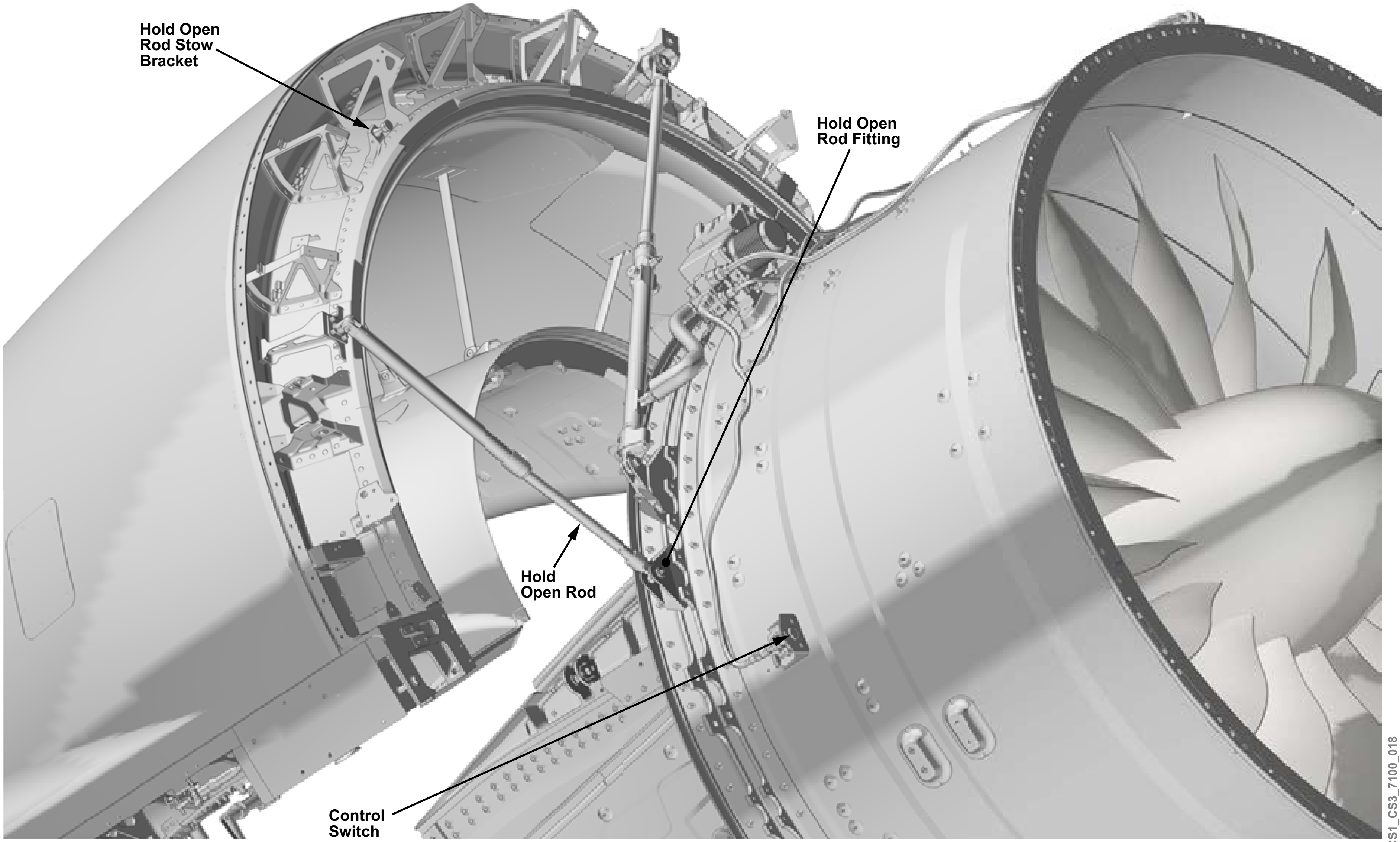


Figure 17: Thrust Reverser Door Opening and Closing

BIFURICATION LATCH SYSTEM ADJUSTMENT

When engaged, the bifurcation latch uses a rod and cam mechanism to insert a locking pin through the latch. As the inner fixed structure closes, a pin on the left side is inserted into a receiver located on the right side.

The pin and latch mechanism are locked together using a rod. The rod is inserted through the pin by turning, and pushing the bifurcation latch system (BLS) lever up. The bifurcation latch is locked in place when the lever is aligned with the fixed structure by pushing the lever in. The pin latch mechanism is adjustable up to 0.635 cm (0.250 in.) in either direction.

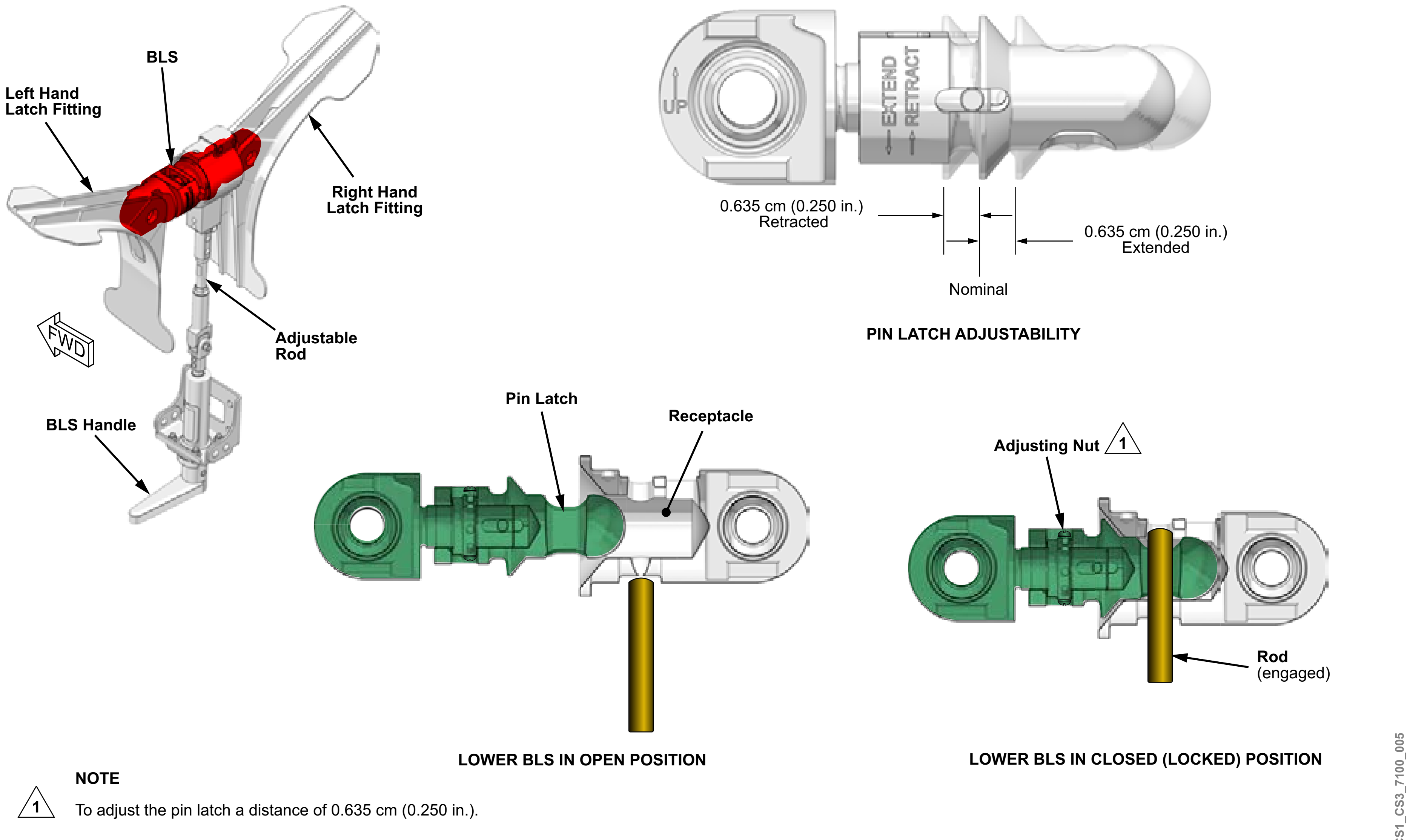


Figure 18: Thrust Reverser Door Bifurcation Latch System Adjustment

78-11 TURBINE EXHAUST SYSTEM

GENERAL DESCRIPTION

The turbine exhaust system helps control the flow of the exhaust gases. It consists of an exhaust nozzle and centerbody. The centerbody is made up of the forward centerbody and aft centerbody. The exhaust nozzle and centerbody are bolted to the aft engine flange.

Crossflow blockers, mounted on the exhaust nozzle at the 2 o’clock and 10 o'clock positions, enhance performance by reducing secondary air flow disturbances across the pylon area.

Fireseal fingers, also known as turkey feather seals are mounted on the top of the exhaust nozzle to prevent flames from exiting or entering the core compartment area.

During installation, two alignment pins on the forward centerbody facilitate proper clocking of the nozzle.

Drain holes help remove any fluid accumulation in the turbine exhaust.

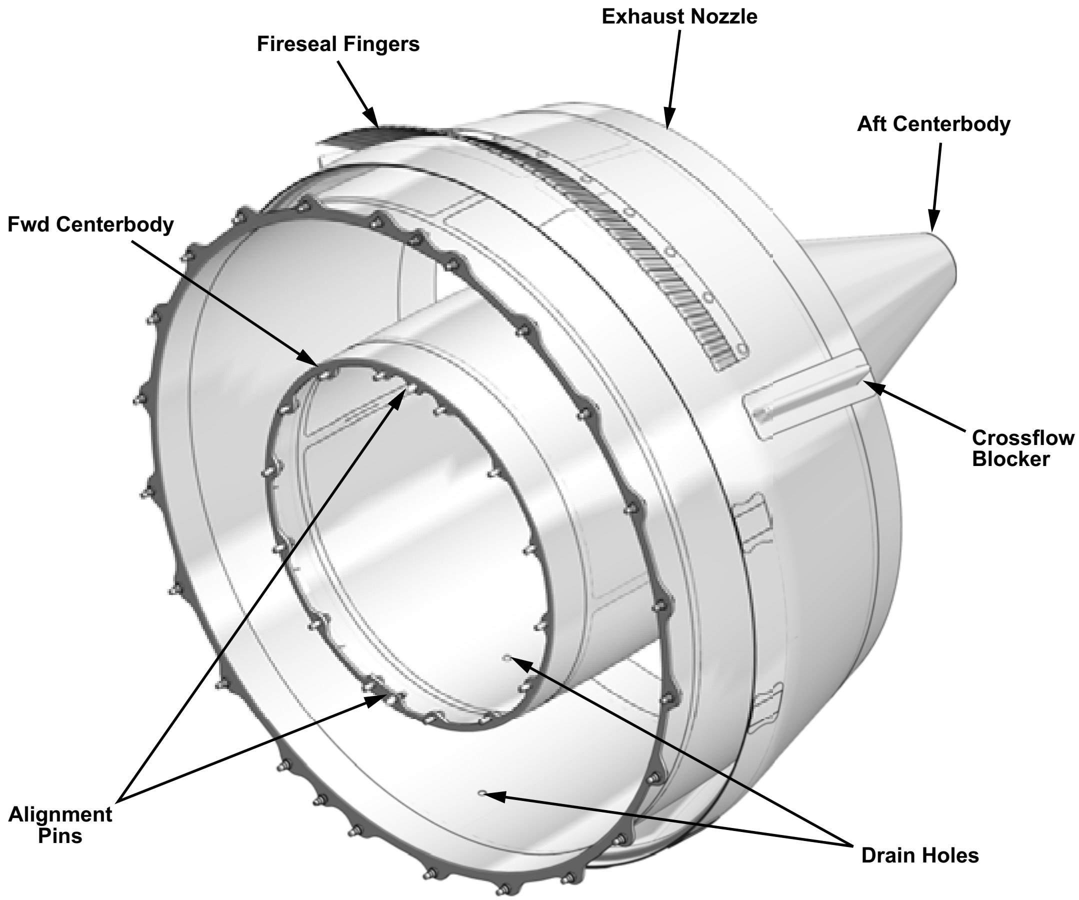


Figure 19: Turbine Exhaust System

71-40 ENGINE MOUNTS AND PYLON

GENERAL DESCRIPTION

The engine mounts attach the engine to the aircraft pylon. The nacelles do not carry any significant structural loads. Thrust loads are transferred through the engine mounts. Mounts are disconnected to remove the engine.

The function of the engine mount system is to transmit engine and nacelle loads to the pylon. The pylon then transfers these loads to the wing. The main sources of engine mount loads are engine thrust, aerodynamic pressures, and inertia loads from maneuvers and gusts.

Thrust links transfer thrust loads to the aft mount and prevent any bending of the engine core.

All mounts are designed to be failsafe. Excess clearances in holes along secondary load paths allow the secondary paths to remain inactive, while the primary load paths are intact.

COMPONENT LOCATION

Engine mounts located on the fan and turbine exhaust cases are used to attach the engine to the aircraft and pylon.

The engine mount system consists of the following components:

- Forward engine mount
- Aft engine mount
- Thrust links

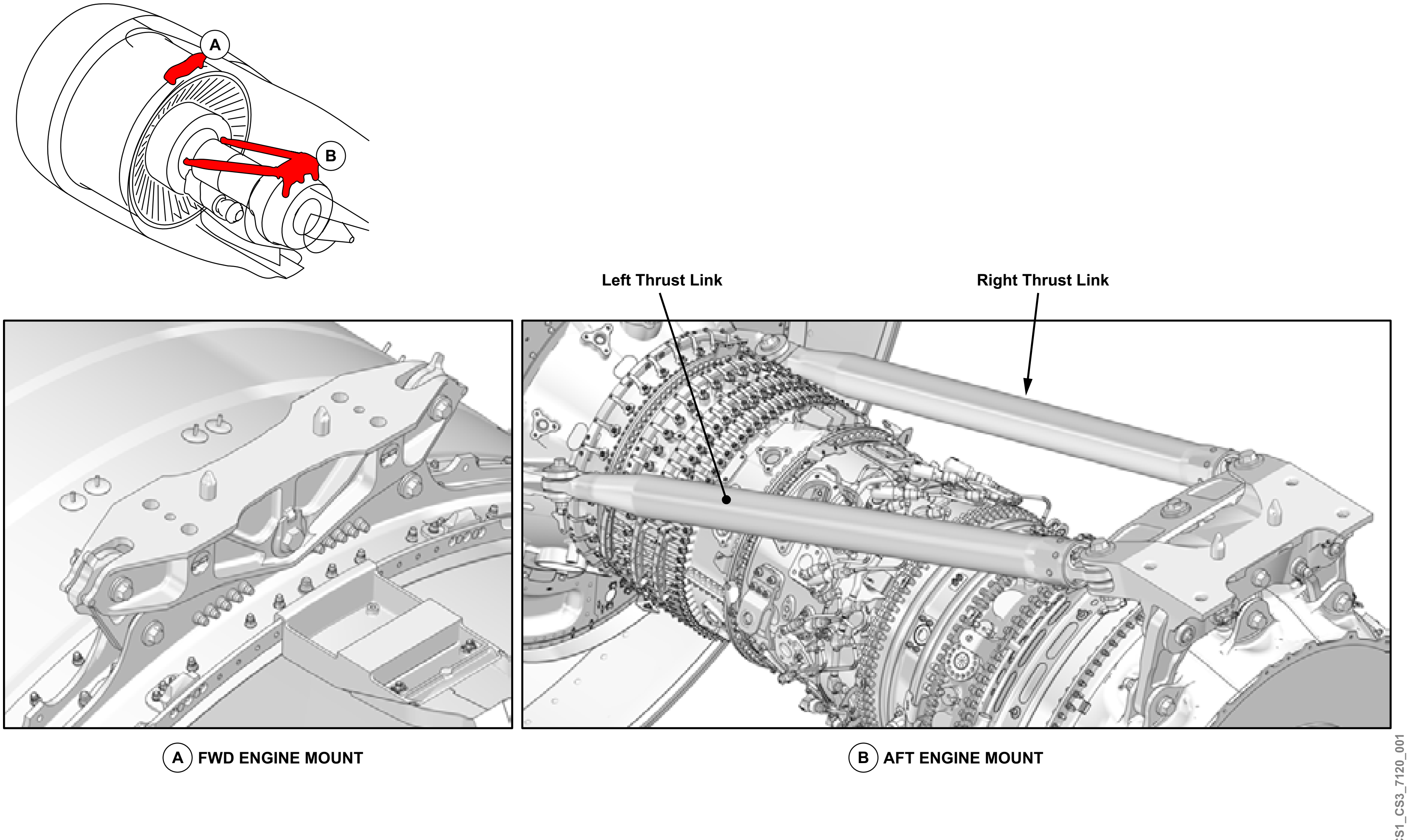


Figure 20: Engine Mounts and Pylon

DETAILED COMPONENT INFORMATION

FORWARD ENGINE MOUNT

The forward engine mount is attached to the engine fan case, and is connected to the pylon by four bolts. Shear pins aid in aligning the mount with the pylon during engine installation. The forward engine mount supports side and vertical loads using a two-link arrangement with bolts loaded in shear.

The main components of the forward mount assembly are the:

- Main beam
- Side links
- Shear pins
- Failsafe bolt

The side and vertical loads at the front mount couple with the aft mount loads to support overall engine pitch and yaw.

Side links provide the primary load paths from the fan case into the front beam. The front mount is attached to the pylon with four bolts and two shear pins, which transmit vertical and shear loads into the pylon. The mount bolts use captive barrel nuts to ease removal.

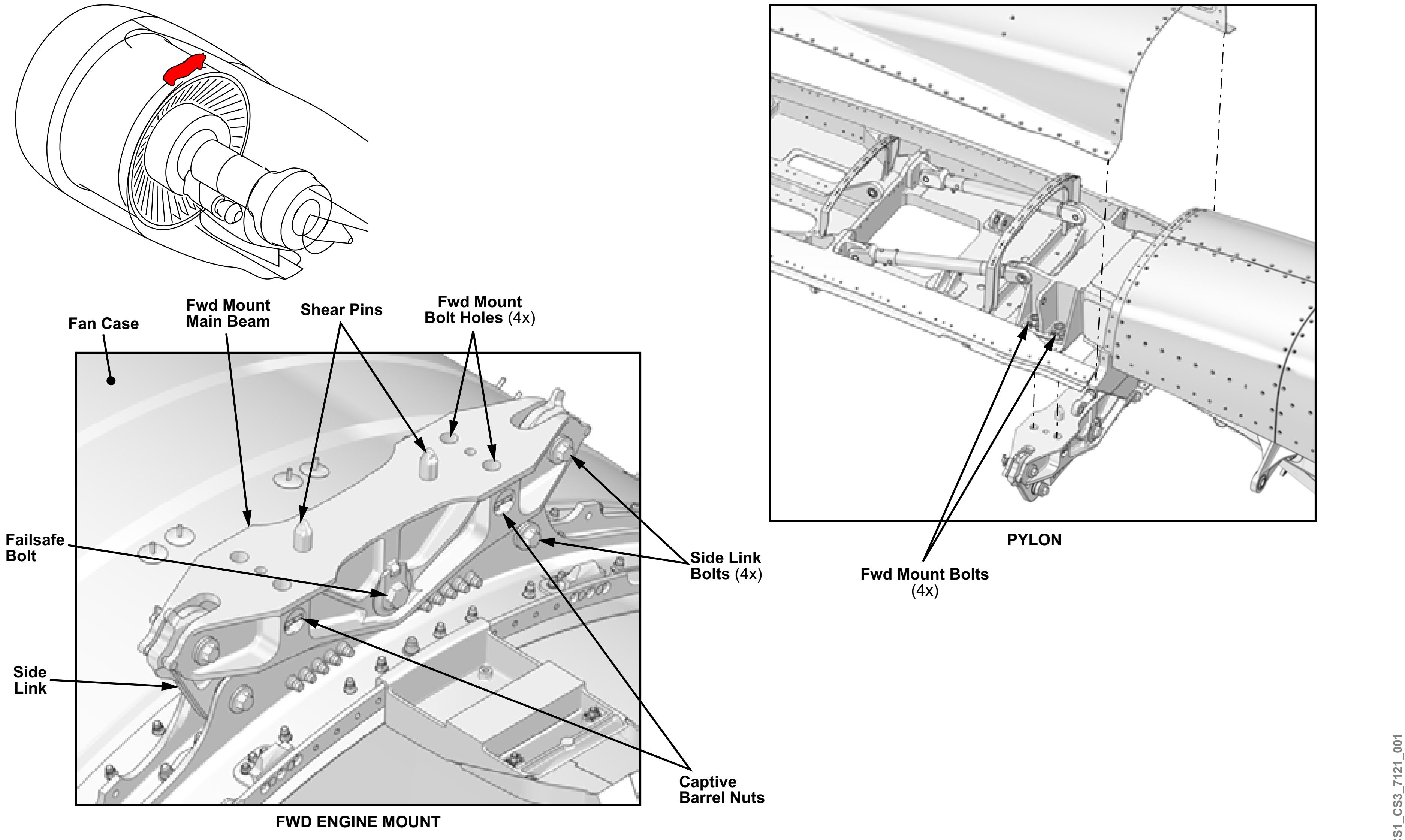


Figure 21: Forward Engine Mount

AFT ENGINE MOUNT

The aft engine mount is attached to the engine turbine exhaust case (TEC) and reacts to engine fore-aft, side, vertical, and roll loads.

The main components of the aft mount assembly are:

- Main beam
- Side links
- Shear pins
- Failsafe link
- Balance beam

The two side links are assembled to the beam with four shear bolts. Each side link is attached to the turbine exhaust case with one shear bolt. The center link and the two attaching bolts are referred to as the failsafe link and bolts.

THRUST LINKS

Two thrust links attach the compressor intermediate case (CIC) to the aft mount through a balance beam. The thrust links transfer thrust loads to the aft mount. Any load differential in the two thrust links are balanced around the center pivot of the balance beam.

The failure of any one link on the rear mount, including the thrust links, cause the system to transfer loads to a secondary load path.

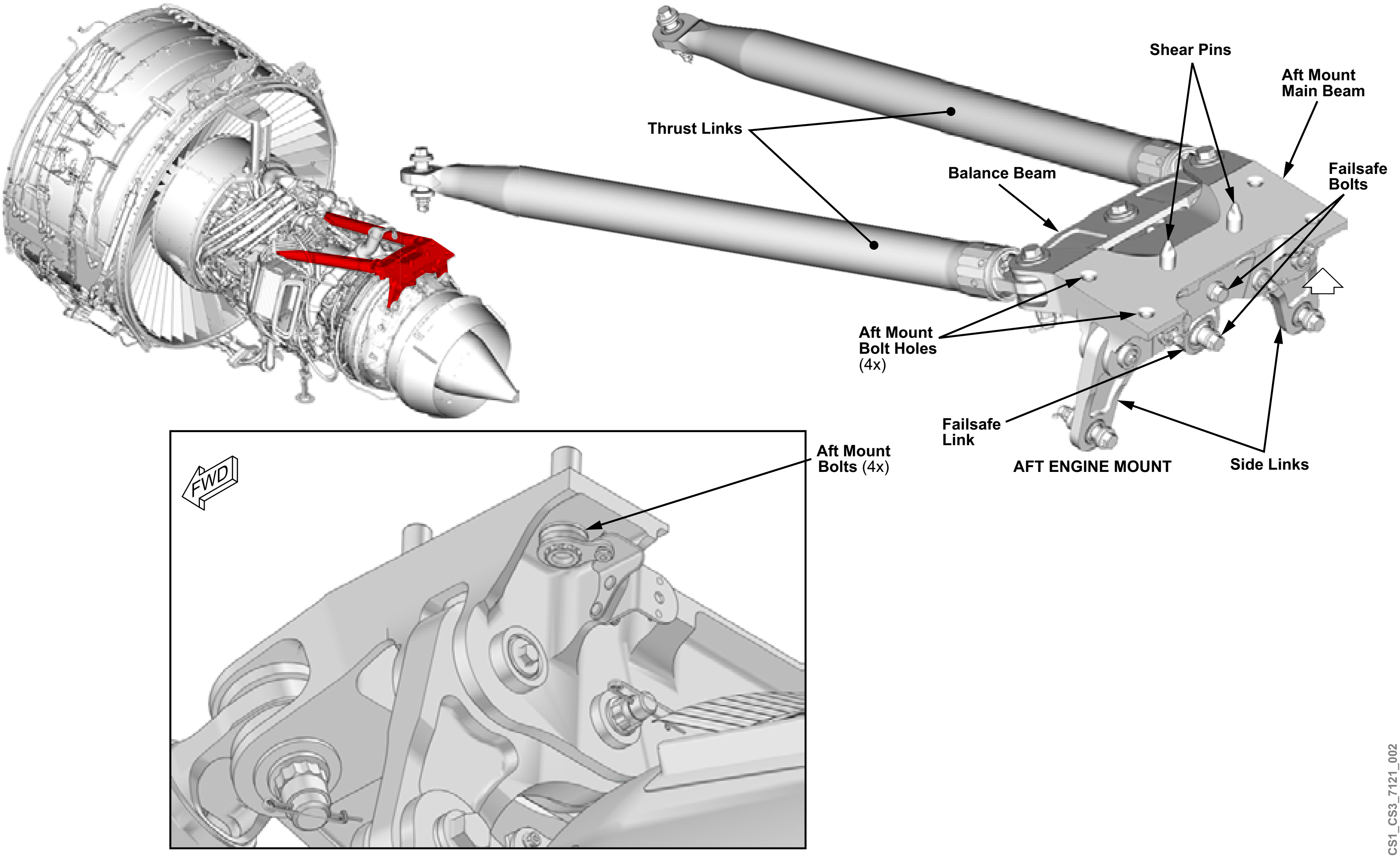


Figure 22: Aft Engine Mount and Thrust Links

PRACTICAL ASPECTS

ENGINE REMOVAL BOOTSTRAP EQUIPMENT

The engine is removed using bootstrap equipment. The equipment consists of forward and aft bootstrap assemblies. Each assembly consists of arms and pylon mount attach points. Hoists, fitted to each arm, are used to raise and lower the engine cradle.

The arms and pylon mount attach points are assembled and attached to the pylon fittings using quick-release pins.

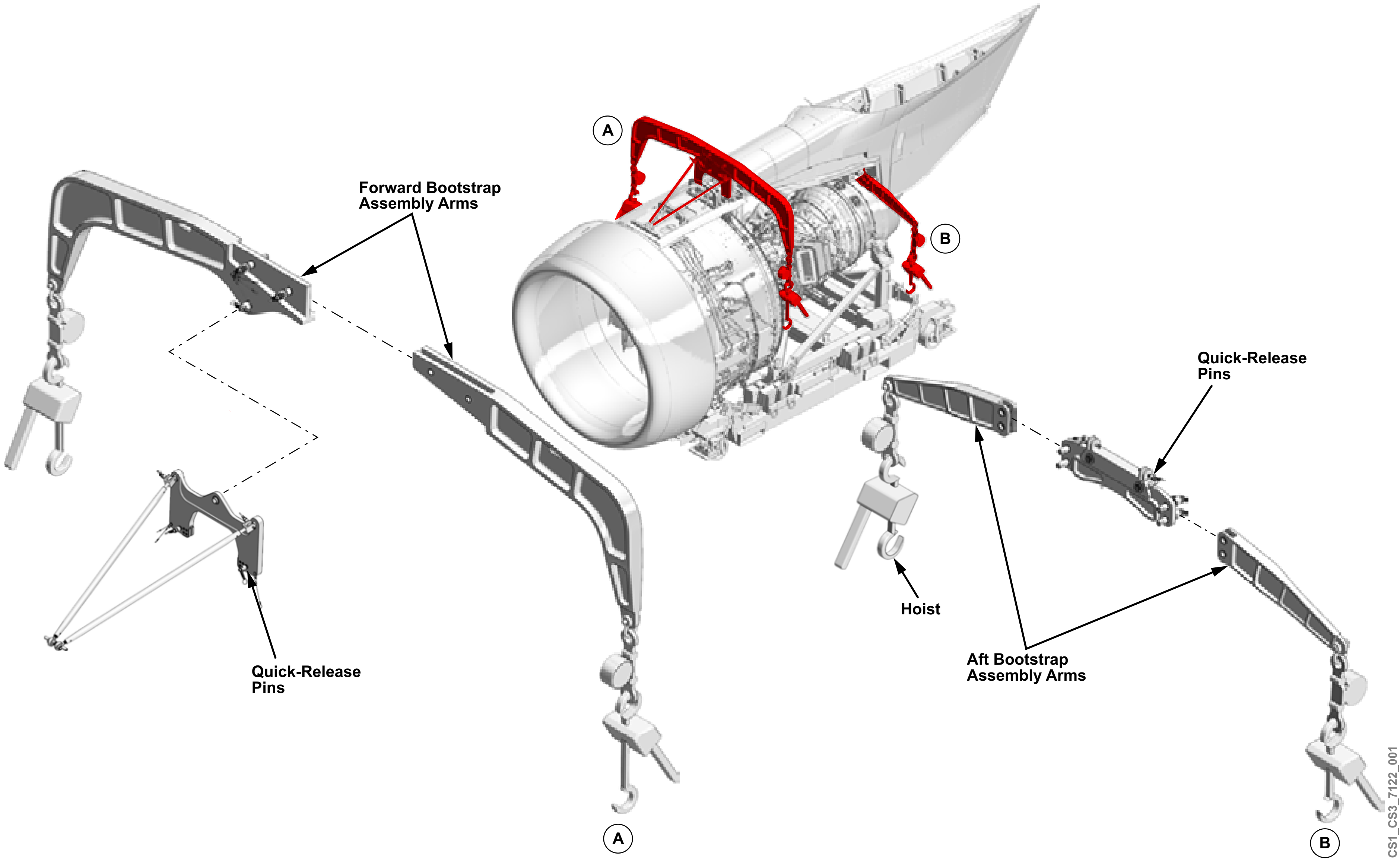


Figure 23: Engine Removal Bootstrap Equipment

ENGINE CRADLE AND TRANSPORT STAND

The engine cradle sits on a transport stand so that the engine can be moved. To remove the engine, the transport stand and cradle are positioned under the engine.

The bootstrap hoists attach to a cradle that holds the engine. The cradle is raised up to the engine using the hoists and is attached to the engine. When the engine is disconnected from the pylon, the cradle and engine are lowered back onto the transport stand.

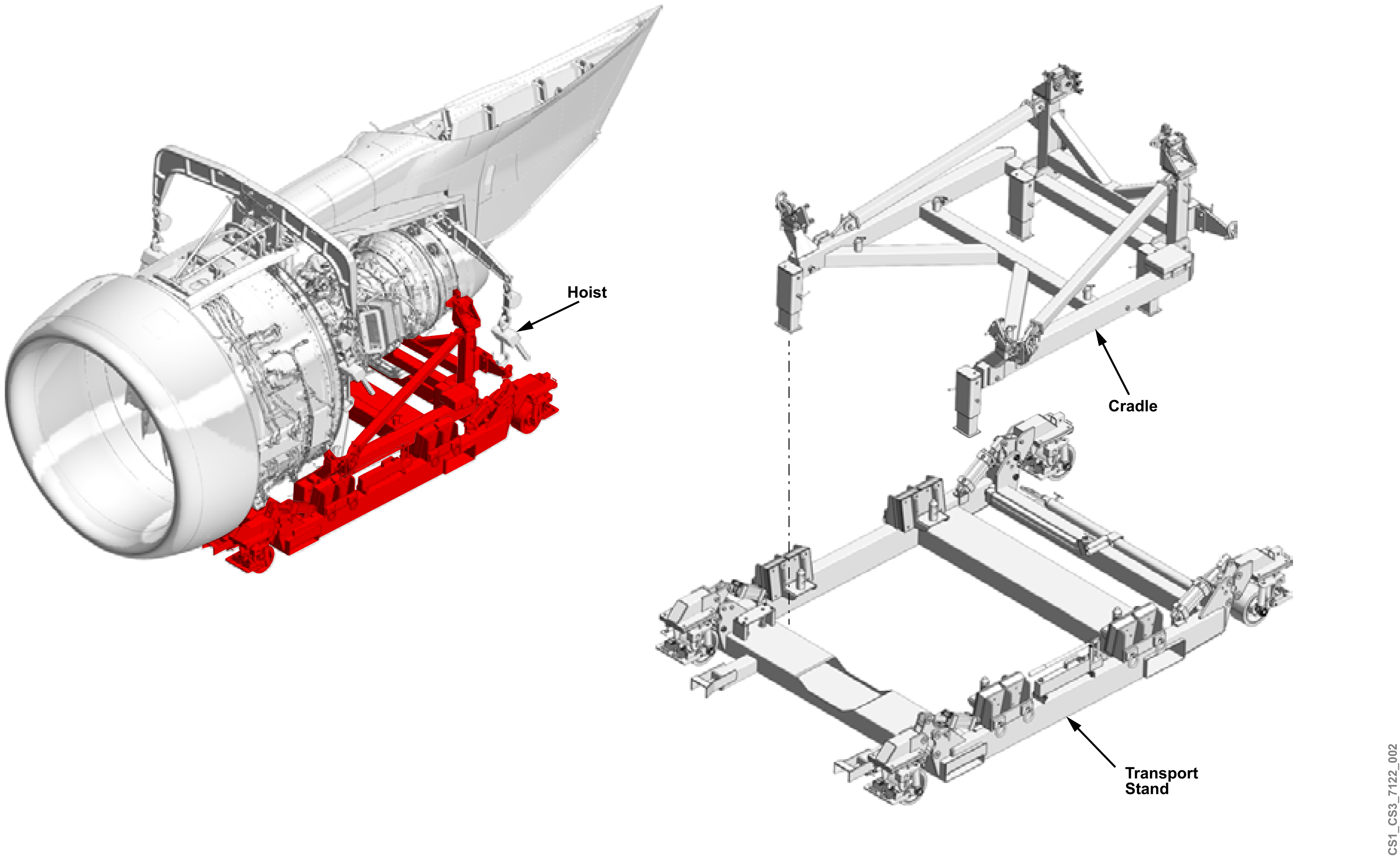


Figure 24: Engine Cradle, and Transport Stand

ELECTRICAL HARNESSES

Electrical connectors are located at three positions on the forward pylon. There is a bracket with four connectors on each side of the engine fan case, and one bracket with three connectors on top of the engine near the precooler. All of these connectors need to be disconnected using soft jaw pliers prior to removing the engine.

The generator power feeder harness connects to a terminal block, mounted near the precooler. The terminal block is accessed by removing the cover from the cooling conduit. The ring terminals need to be removed from the terminals and marked prior to removing the engine.

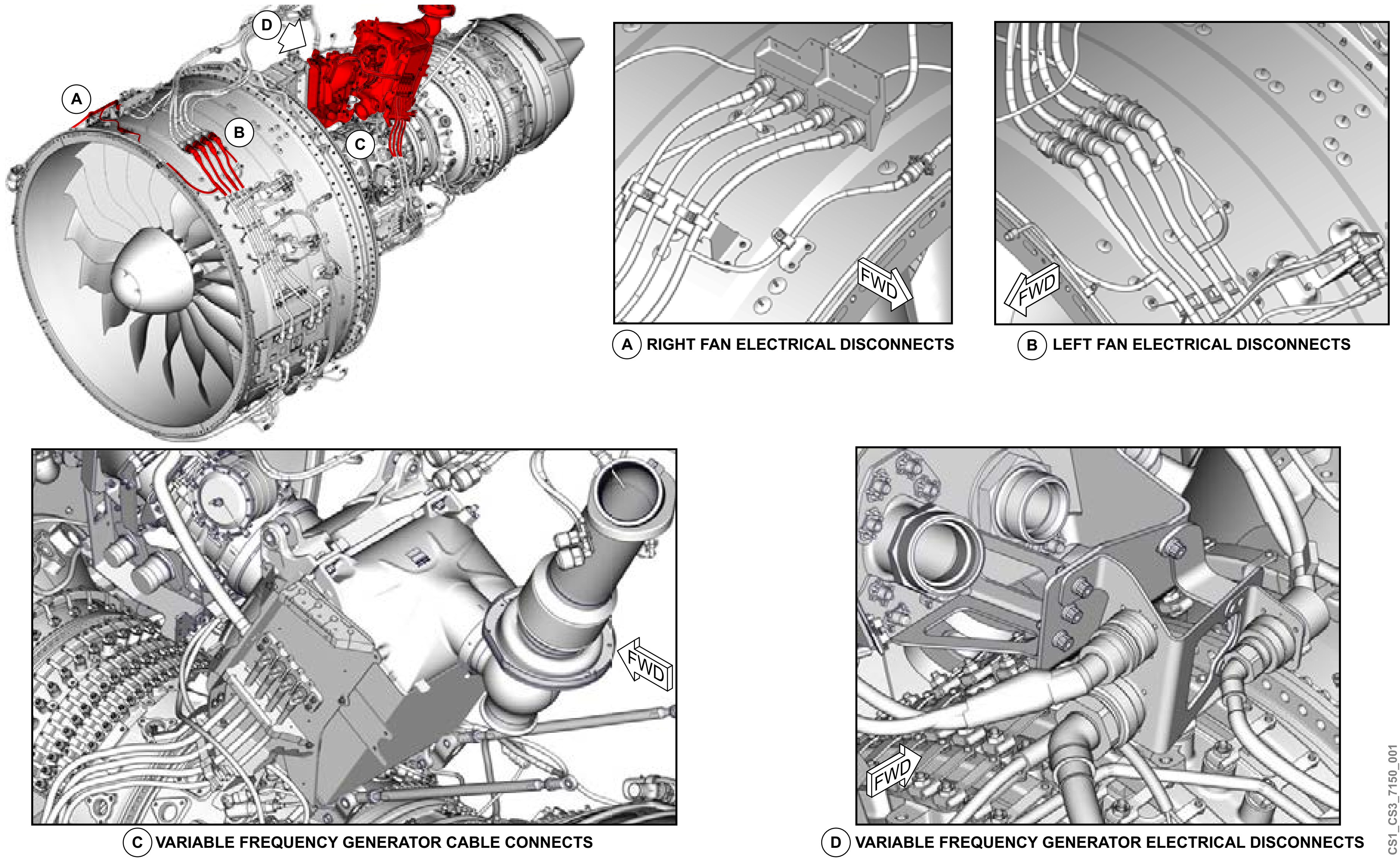


Figure 25: Electrical Harness Disconnects

HYDRAULIC AND FUEL CONNECTIONS

A bracket mounted on the precooler inlet duct supports three hydraulic line quick-disconnect fittings, and two fuel line connections.

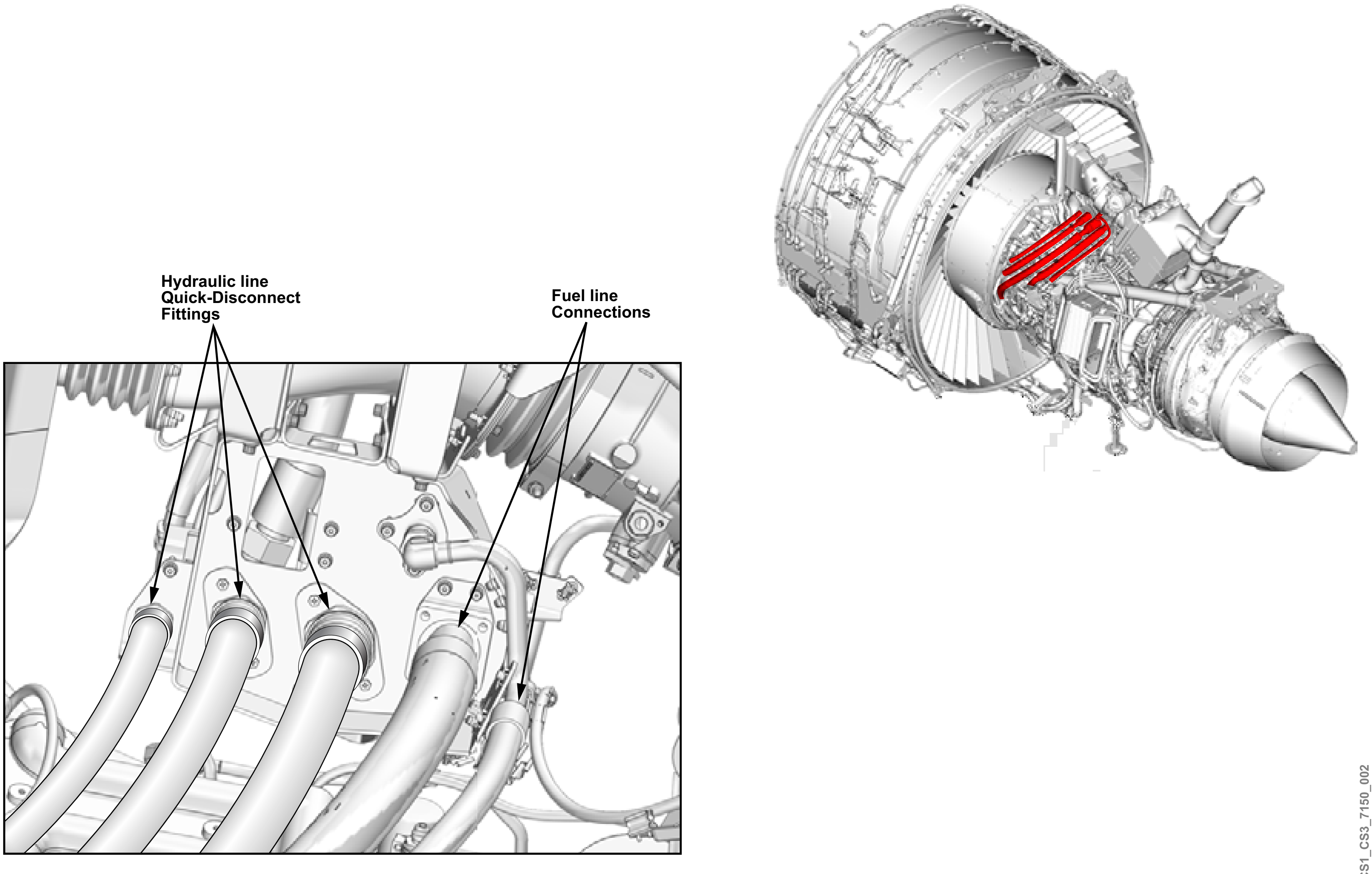


Figure 26: Hydraulic and Fuel Connections

71-70 ENGINE DRAINS

GENERAL DESCRIPTION

The engine drain system collects and discharges oil, fuel, air, and hydraulic fluid from the engine through various tubes, forming a drain mast.

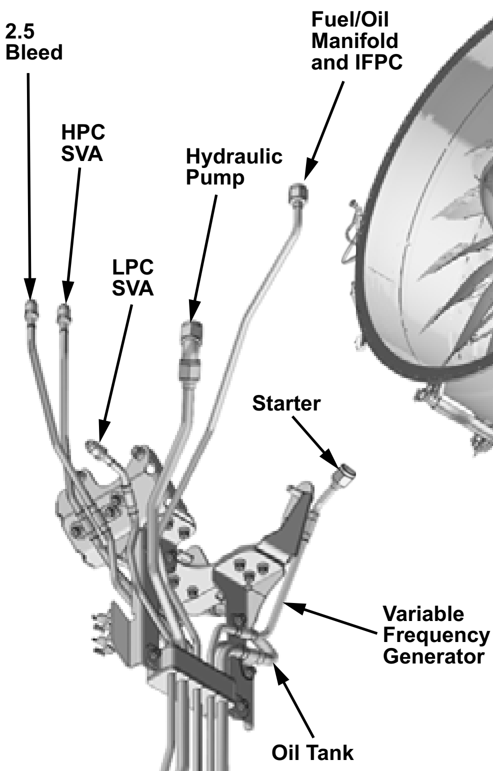
The drain mast terminates at the latch access panel at the 6 o'clock position on the bottom of the engine cowling.

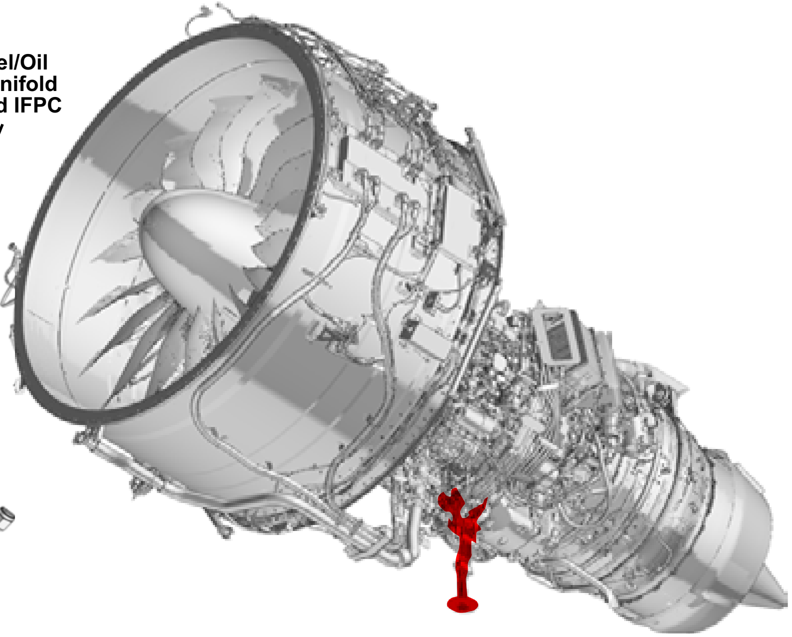
The nacelle is also designed to drain fluid. Drain paths include gaps around the latch access panel, and drain holes in the lower bifurcation fixed/access panels.

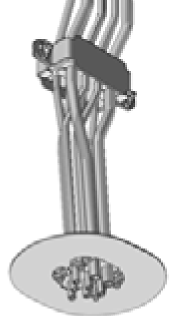
Components that drain through the drain mast include:

- 2.5 bleed
- High-pressure compressor (HPC) stator vane actuator
- Low-pressure compressor (LPC) stator vane actuator
- Variable frequency generator (VFG)
- Starter
- Fuel oil manifold and integrated fuel pump and control (IFPC)
- Oil tank scupper
- Hydraulic pump

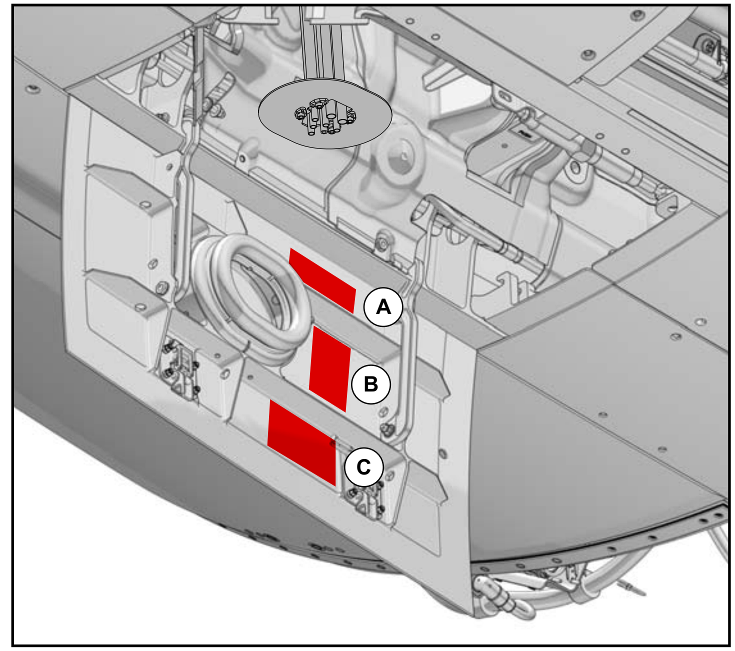
A map of how the drain tubes are situated in the mast assists in troubleshooting. The map is located on the inside of the latch access panel door.







DRAIN MAST

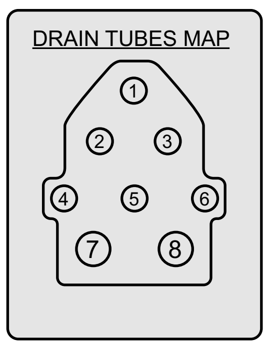


COWLING LATCH ACCESS PANEL

DRAIN TUBES MAP

1. 999-5822-1 DR11, 2.5 BLEED
2. 999-5827-1 DR11, HPC-SVA
3. 999-5823-1 DR01, LPC-SVA

A



B

DRAIN TUBES MAP

4. 999-5825-1 DR61, VFG
5. 999-5832-1 DR91, STARTER
6. 999-5829-1 DR41, FOM-FUEL
7. 999-5826-1 DR71, OIL-TANK
8. 999-5821-1 DR51, HYDRAULIC

C

LEGEND

HPC	High-Pressure Compressor
IFPC	Integrated Fuel Pump and Control
LPC	Low-Pressure Compressor
SVA	Stator Vane Actuator

CS1_CS3_7171_001

Figure 27: Engine Drains

71-00 ENGINE STORAGE AND PRESERVATION

GENERAL DESCRIPTION

Turbine engines are designed to operate on a regular basis. Corrosion of vital engine parts can occur very quickly, especially in harsh environments such as coastal areas where the engine is exposed to salt air.

Storage and preservation procedures are designed to minimize the way the environment may effect the engines.

Engine covers must be installed on both engines to protect engine parts during shipping and handling, and from environmental conditions.

Engine covers should always be installed if the engine is outdoors for extended periods without being operated.

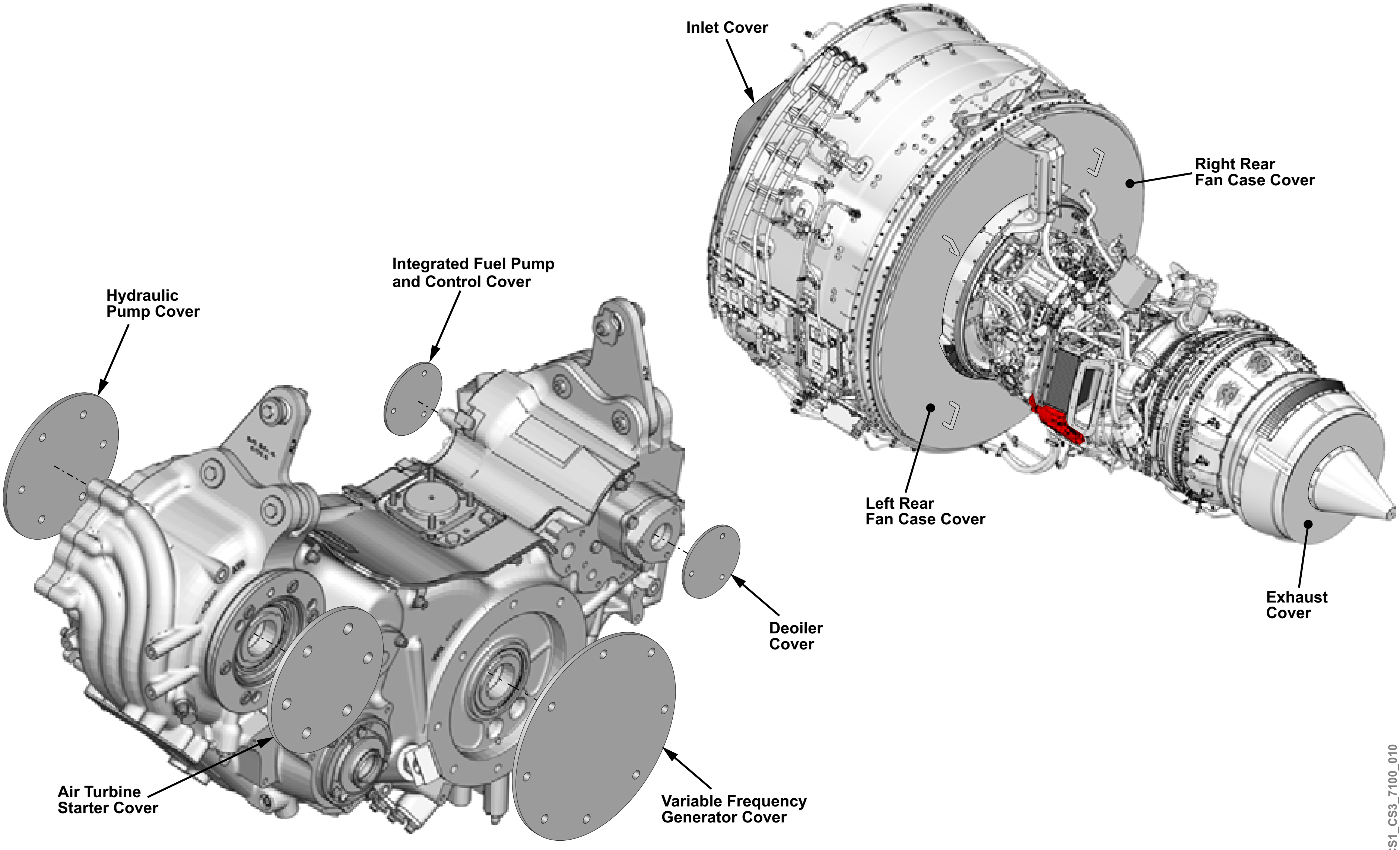


Figure 28: Engine Storage and Preservation

ENGINE PRESERVATION CHART

The engines should be operated at least weekly. For engines that are not operated at least once a week, the following engine preservation guidelines are provided.

Always refer to the aircraft maintenance publication (AMP) for preservation requirements and instructions.

Method I (60 days or less)

Whenever engine preservation is going to be 60 days or less the following steps need to be carried out:

- Clean and degrease the engine
- Check for oil leaks
- If components are removed from the engine or gearbox, ensure they are properly protected and preserved from corrosion
- Sample the engine oil for water using ASTM D6304-00 procedure, if the sample fails then flush the engine oil system
- Drain the engine oil system
- Change the starter oil
- Install tarp or PVC plastic sheets over inlet cowl, fan exhaust area and engine exhaust area
- Make a record of the engine preservation

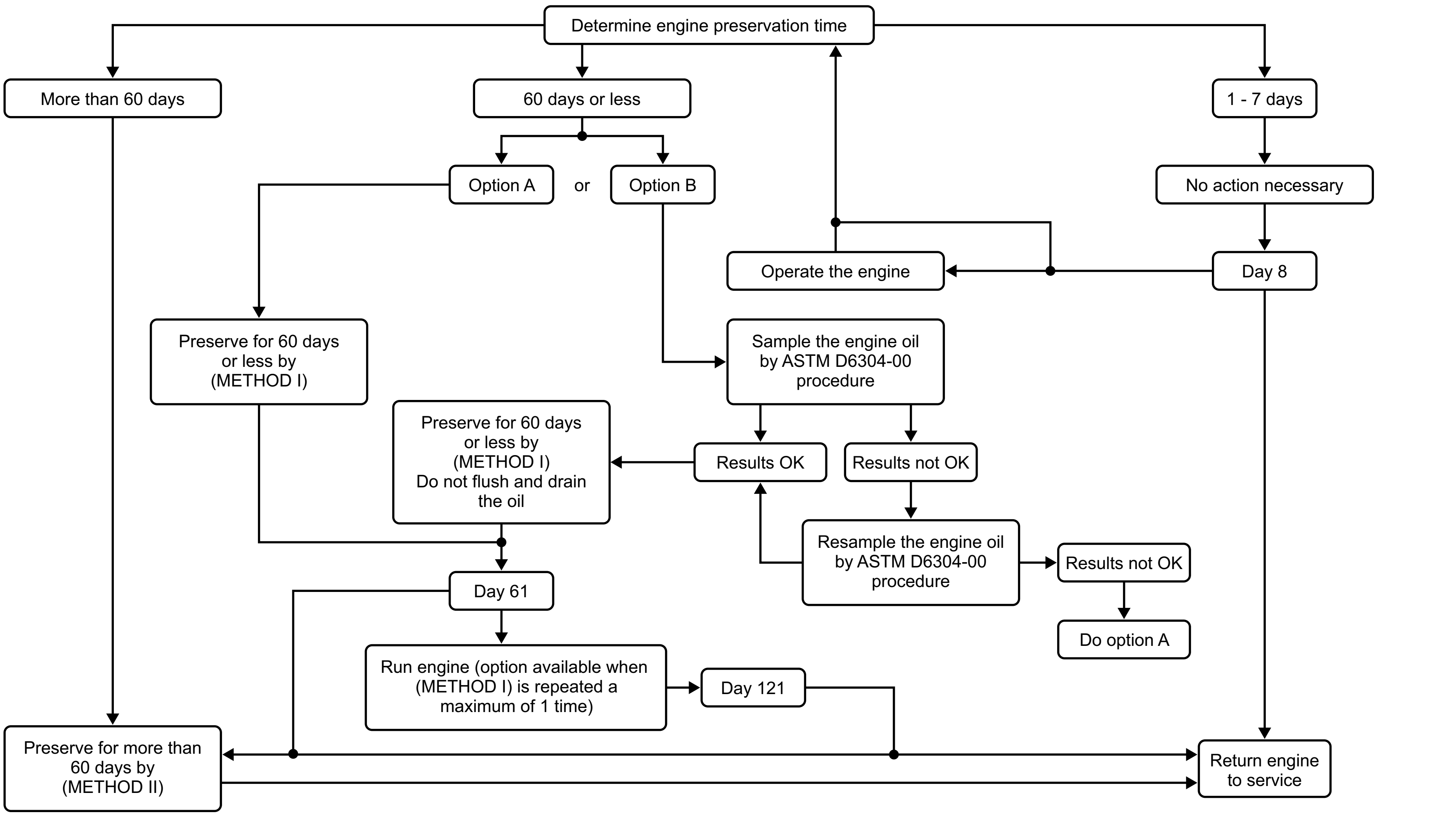
There is a provision to extend the engine preservation an extra 60 days if the following procedures are carried out:

- Remove the engine covers
- Service the engine oil system
- Run the engine at idle for five minutes once the oil temperature is above 220 °F (104.4 °C)
- Shut down the engine
- Check fuel for water using the ASTM D4176 procedure
- If required, drain fuel from the tanks until fuel passes the test procedure

Method II (More than 60 days)

Whenever engine preservation is going to extend beyond 60 days, the following steps need to be carried out:

- Drain the engine oil system
- Replace engine oil filter and fill the oil system
- Start the engine and run at idle or dry motor for 3 minutes using the engine starter. It is necessary that the fan turn so that oil reaches through all of the engine oil system
- Preserve the fuel system using preservation oil
- Drain the engine oil
- Change the starter oil
- Clean and preserve the gearbox
- Place some desiccant and some relative humidity indicators inside engine covers
- Install tarp or PVC plastic sheets over inlet cowl, fan exhaust area and engine exhaust area. Make sure the relative humidity indicators are visible through windows in the protective covers
- Make a record of the engine preservation



NOTE

1. This chart does not specify all of the steps in the preservation method.
2. Sampling option is restricted to one 60 day period.
3. In option B, oil system flush is not necessary (ref. METHOD I) if sample test results are OK.
4. When preserving for more than 60 days, fuel system preservation is also necessary.
5. If the engine remains in storage after 60 days (METHOD I), preserve the engine again after the initial 60 day period.

Figure 29: Engine Preservation Chart

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ATA 72 - Engine

BD-500-1A10
BD-500-1A11

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ENGINE - CHAPTER BREAKDOWN

72-00 PW1500G ENGINE

GENERAL DESCRIPTION

The PW1500G series engines are high bypass ratio turbofans, utilizing two spools with an integral primary and fan duct exhaust systems.

The N1 rotor consists of a single-stage low-pressure ratio fan, driven at a reduced speed through a fan drive gear system (FDGS), a 3-stage low-pressure compressor (LPC), and a 3-stage low-pressure turbine (LPT) on a common shaft.

The FDGS is a planetary star gear reduction unit that takes torque from the LPT and reduces the fan speed to 30% of the LPT speed.

The LPC has variable inlet guide vanes, which are positioned automatically by the electronic engine control (EEC) to optimize engine operation.

The N2 rotor consists of the high-pressure compressor (HPC) with eight compressor stages, and a high-pressure turbine (HPT) consisting of two stages. The HPC has variable inlet guide vanes and variable stator vanes in the 1st, 2nd, and 3rd stages which are positioned automatically by the EEC to optimize engine operation.

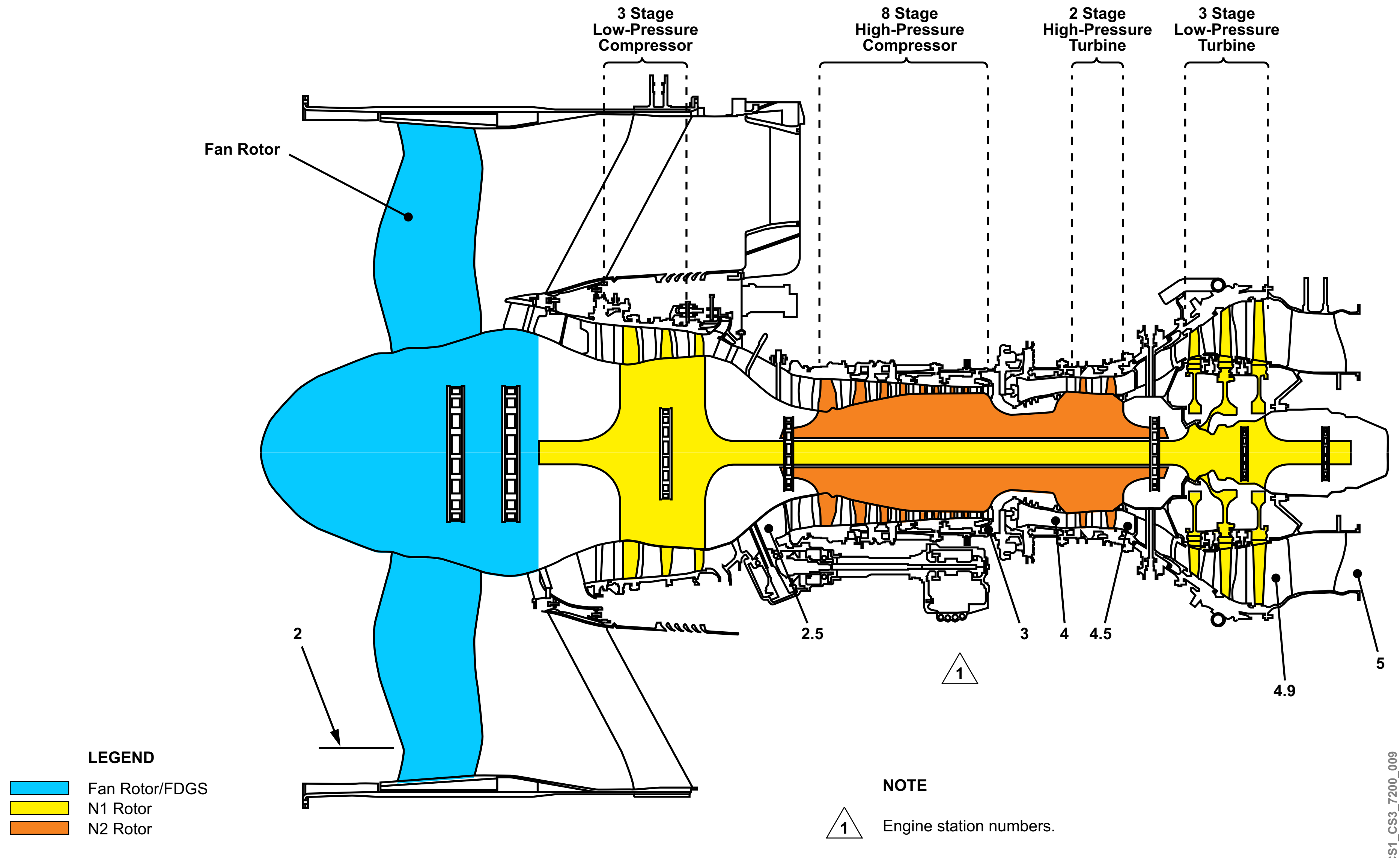
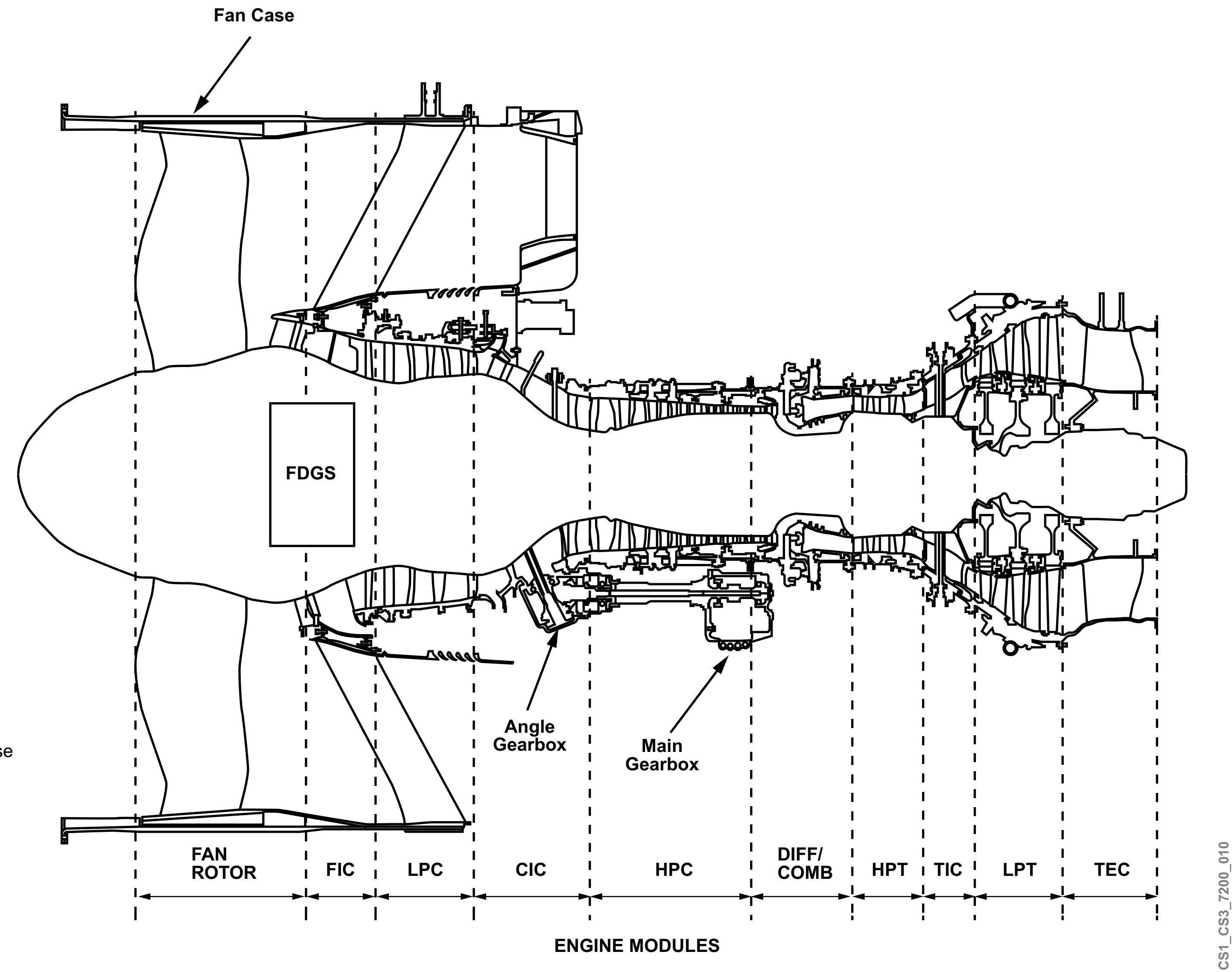


Figure 1: PW1500G Engine

COMPONENT LOCATION

The PW1500G engine is made up of the following modules:

- Fan rotor
- Fan case
- Fan drive gear system (FDGS)
- Fan intermediate case (FIC)
- Low-pressure compressor (LPC)
- Compressor intermediate case (CIC)
- High-pressure compressor (HPC)
- Diffuser and combustor
- High-pressure turbine (HPT)
- Turbine intermediate case (TIC)
- Low-pressure turbine (LPT)
- Turbine exhaust case (TEC)
- Angle gearbox
- Main gearbox (MGB)



LEGEND

CIC	Compressor Intermediate Case
DIFF/COMB	Diffuser/Combustor
FDGS	Fan Drive Gear System
FIC	Fan Intermediate Case
HPC	High-Pressure Compressor
HPT	High-Pressure Turbine
LPC	Low-Pressure Compressor
LPT	Low-Pressure Turbine
TEC	Turbine Exhaust Case
TIC	Turbine Intermediate Case

Figure 2: PW1500G Engine Assembly Component Location

72-11 FAN ASSEMBLY

GENERAL DESCRIPTION

The fan assembly is composed of the fan rotor and the fan case assembly.

The fan rotor is located inside the fan case assembly. It turns clockwise and produces 90% of the engine total thrust.

The fan case assembly directs the fan air through and around the engine core.

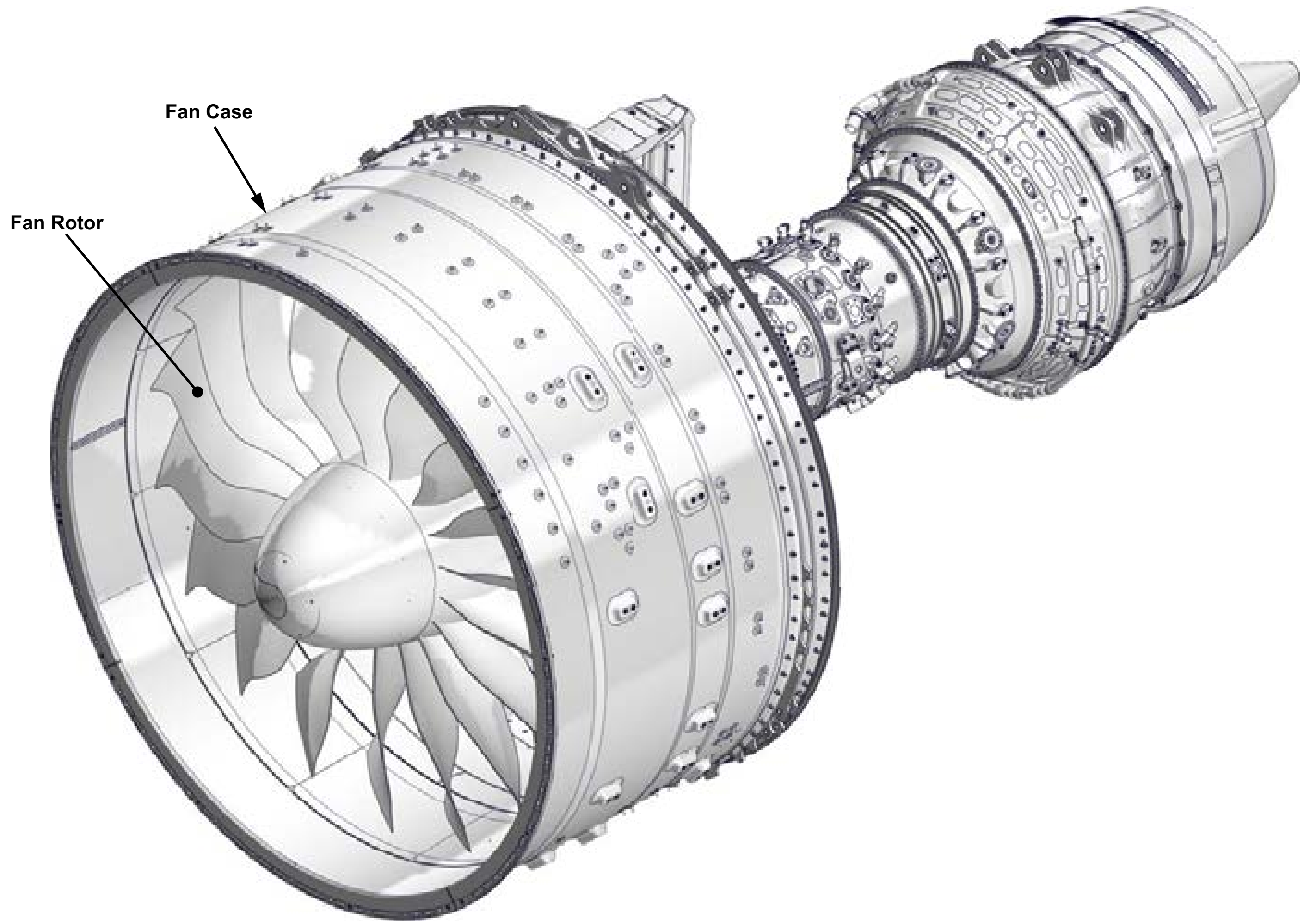


Figure 3: Fan Assembly

DETAILED COMPONENT INFORMATION

FAN HUB

The fan hub retains the fan blades on its circumference. The hub has attachment points on the front face for the inlet cone and fan rotor trim balance weights. The no. 1 fan blade slot is located between two no.1 marks on the hub. Numbering of the fan blades and fairings is in the direction of rotation, beginning at the no. 1 blade position.

Details on fan rotor balancing can be found in ATA 77, Engine Indicating.

FAN BLADES

The fan rotor has 18 wide-chord hollow aluminum fan blades.

The hollow aluminum fan blades have titanium leading edges for wear resistance and foreign object debris (FOD) tolerance. A protective polyurethane black paint coating provides erosion protection for the fan blade airfoil.

Each fan blade has a dovetail shaped root that fits into slots on the fan hub. Composite wear strips are bonded to the fan blade dovetail sides.

Composite fan blade fairings are located between the fan blades to provide an inner flow path surface. Each fan blade fairing is held in place by a fan blade retaining pin, inserted into pin lugs on the fan hub.

A spacer is installed between each blade and the hub to minimize wear, and provide a tight fit. The fan blades are held in place on the hub by a front locking ring.

INLET CONE

The composite inlet cone is installed in front of the fan hub to smooth the airflow into the engine. The inlet cone cover provides access to the inlet cone retaining hardware. The inlet cone is removed to gain access to the fan blades lock ring.

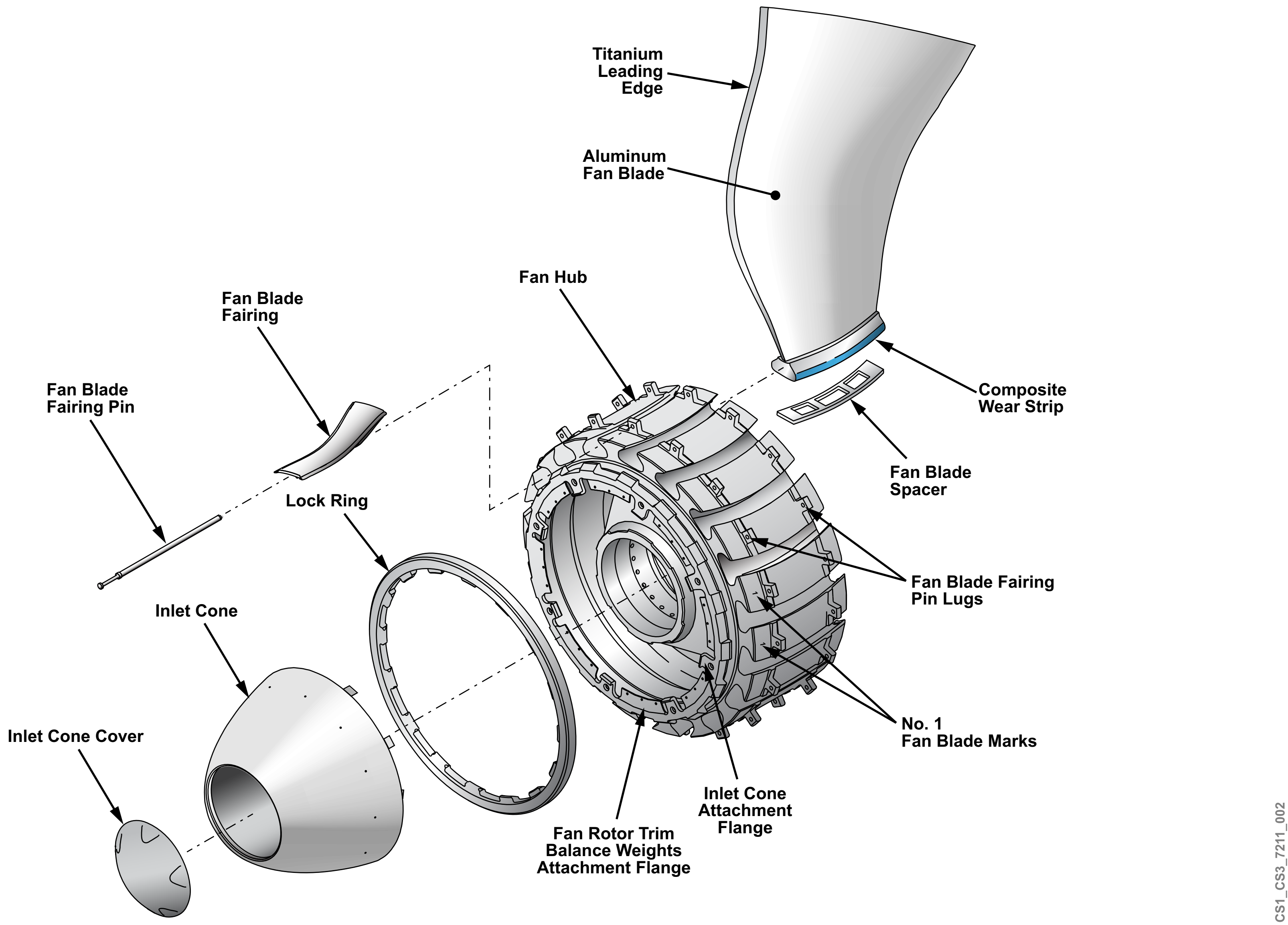


Figure 4: Fan Hub, Blades, and Inlet Cone

FAN CASE ASSEMBLY

The fan case serves as the structural link between the inlet cowl and the engine core by providing an attachment point to the airframe. It consists of a one-piece composite case with a titanium mount ring attached around the aft outer surface. A Kevlar containment ring surrounds the fan case.

Fan case components include:

- Fan blade rub strips
- Acoustic panels
- Ice liners
- Precooler duct inlet
- Fan exit guide vanes (FEGV)
- Fan exit liner segments

Fan Blade Rub Strips

The fan blade rub strips provide an abradable surface for the fan blade to rub against when the blades elongate under power. They also protect the fan blades from contacting the fan case and ensure a tight clearance between the case and the rub strips. The surface of the rub strips can be refurbished if any significant rubbing occurs.

Acoustic Panels

Acoustic panels, forward of the fan rub strip, and acoustic liner segments aft of the fan blades, decrease noise generated from the fan.

Ice Liners

Replaceable ice liners aft of the fan, provide protection from ice for the case structure.

Precooler Duct Inlet

The precooler duct inlet is a four-piece titanium casting mounted at the rear top of the fan case. It is used to direct fan air into the pylon mounted precooler, provides seal lands for the thrust reverser (T/R) door firewall seal, and sealing of the upper nacelle flow path. The precooler duct inlet is attached to the titanium mount ring at the rear of the case. The mount ring provides the front mount attachment points for the engine mounts, and includes a V-groove around the entire circumference, providing alignment and support for the thrust reverser doors.

Fan Exit Guide Vanes

On the inner surface of the fan case, 44 hollow aluminum fan exit guide vanes (FEGVs) extend rearward to the fan intermediate case (FIC). The FEGVs straighten the fan air and provide structural support between the FIC and the fan case assembly. A polyurethane coating covers most of the FEGV surfaces to protect against corrosion and erosion.

Fan Exit Liner Segments

Six fan exit liner segments, immediately aft of the FEGVs, surround the low-pressure compressor (LPC). Stationary louvers in the fan exit liner segments provide an exit path for the 2.5 bleed air into the fan air stream.

The segments can be removed individually for component access and inspection of lines and fittings.

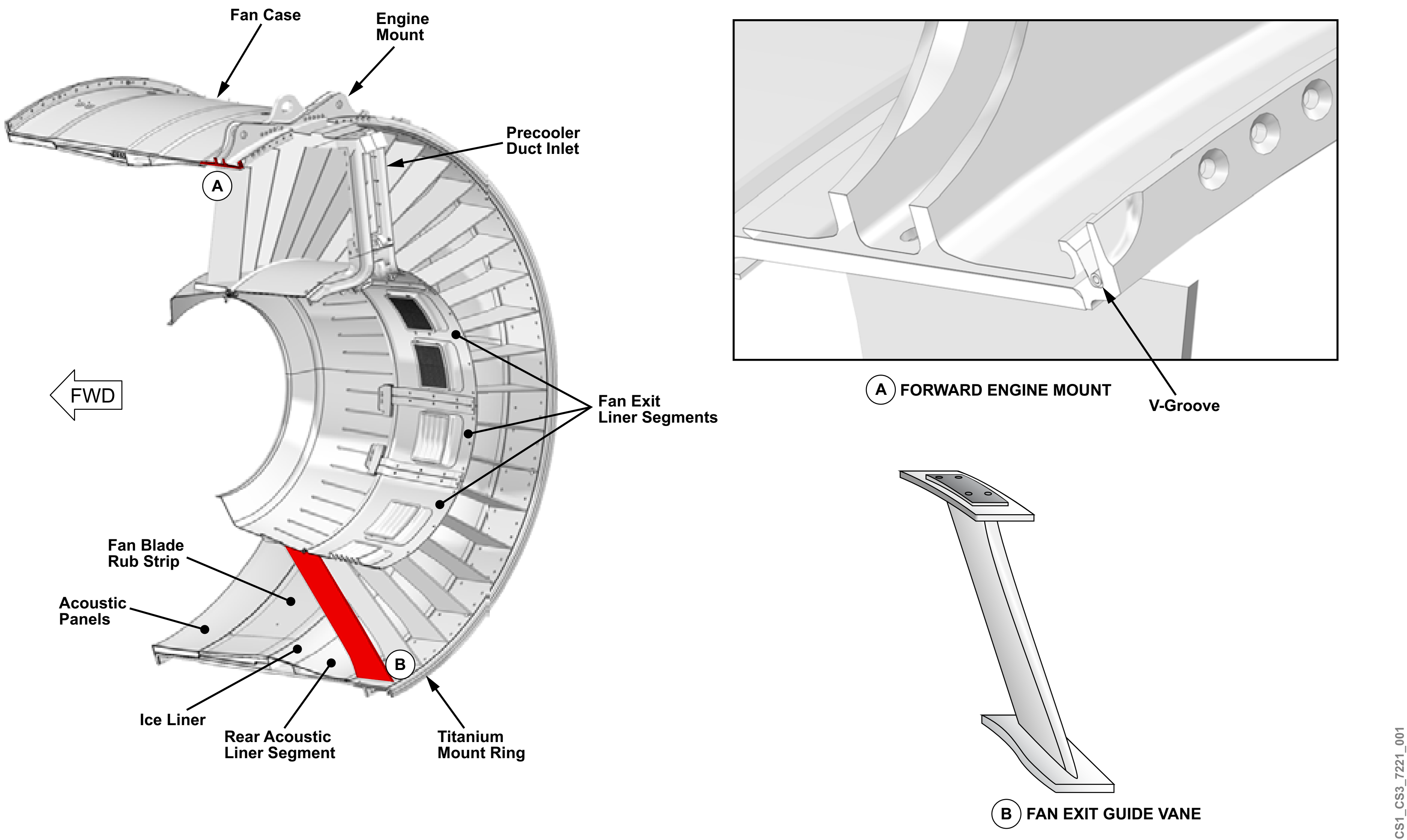


Figure 5: Fan Case Assembly

72-15 FAN DRIVE GEAR SYSTEM

GENERAL DESCRIPTION

The fan drive gear system (FDGS) is a reduction gear assembly that uses torque from the low-pressure turbine (LPT) N1 to turn the fan assembly at approximately one third the speed of the LPT. This results in a lower fan speed and a higher low-pressure compressor (LPC) speed, thus increasing compressor efficiency.

The FDGS is located between the fan rotor assembly and the fan intermediate case (FIC). It is radially and axially supported by the no.1 and no. 1.5 bearing support assembly.

The LPC shaft connects to the input of the FDGS. The input speed is reduced by the fan drive gearbox (FDG) and then transmitted to a gearbox shaft connected to the fan rotor hub.

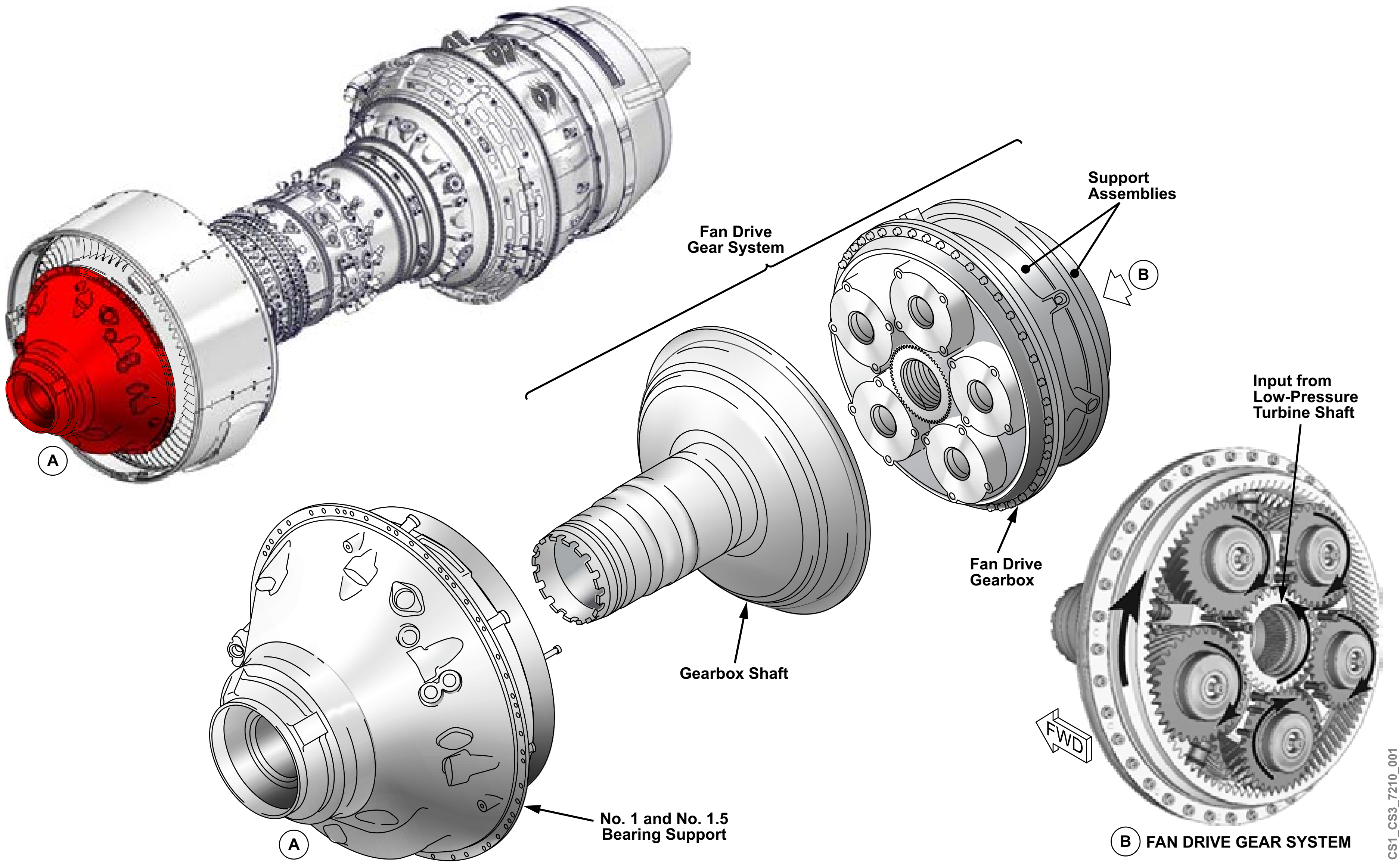


Figure 6: Fan Drive Gear System

DETAILED COMPONENT INFORMATION

FAN DRIVE GEARBOX

The fan drive gearbox (FDG) is the most aft section of the FDGS. It houses the sun gear, the five star gears, and the rotating ring gear halves.

The FDG takes the torque from the low-pressure turbine (LPT) FDG titanium coupling to turn a sun gear. This sun gear turns five stationary star gears against an outer ring gear set. The outer ring gear set is turned by the star gears and is connected to the gearbox shaft, which connects to the fan hub.

Each star rotates around its own journal bearing mounted in the carrier. The journal bearings provide the rotational hinge for the star gears. The oil manifold supplies oil to no. 1 and no. 1.5 bearing support assembly, which supplies the lubrication for the FDG gears and bearings.

The torque frame links the carrier to the flex support that is bolted to the fan intermediate case (FIC). This flex support ensures the FDGS remains aligned with the FDG coupling. The FDG coupling connects the LPC shaft to the FDGS. The FDG coupling is a specially designed flex shaft that is used to reduce pressure on the FDGS. It transmits the torque from the LPC to the FDGS input.

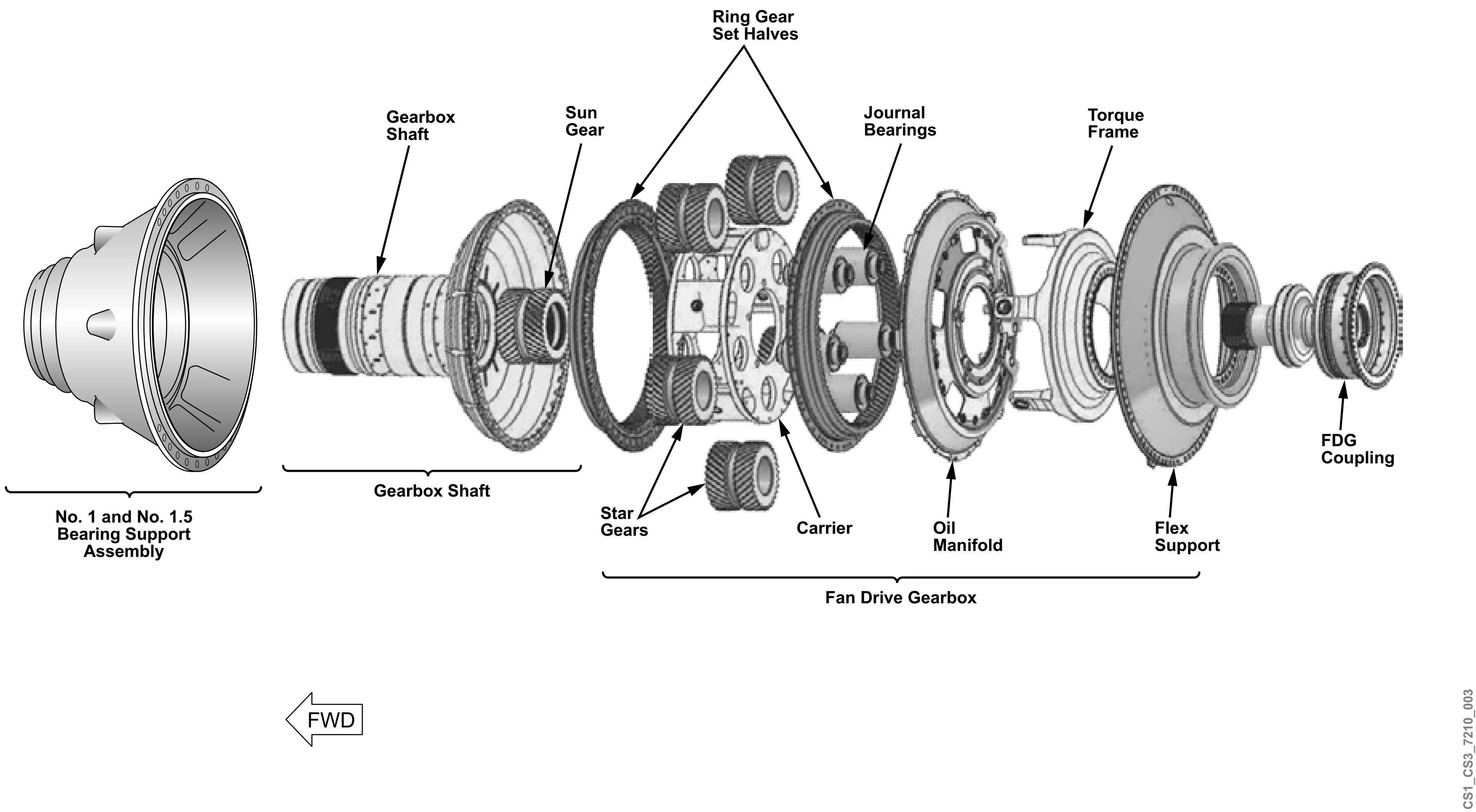


Figure 7: Fan Drive Gearbox

NO. 1 AND NO. 1.5 BEARING SUPPORT ASSEMBLY

The no. 1 and no. 1.5 bearing support assembly supports the gearbox shaft using two tapered roller bearings. The no. 1 bearing is oil damped to absorb fan vibration. The aft section of the bearing support assembly is attached to the fan intermediate case (FIC).

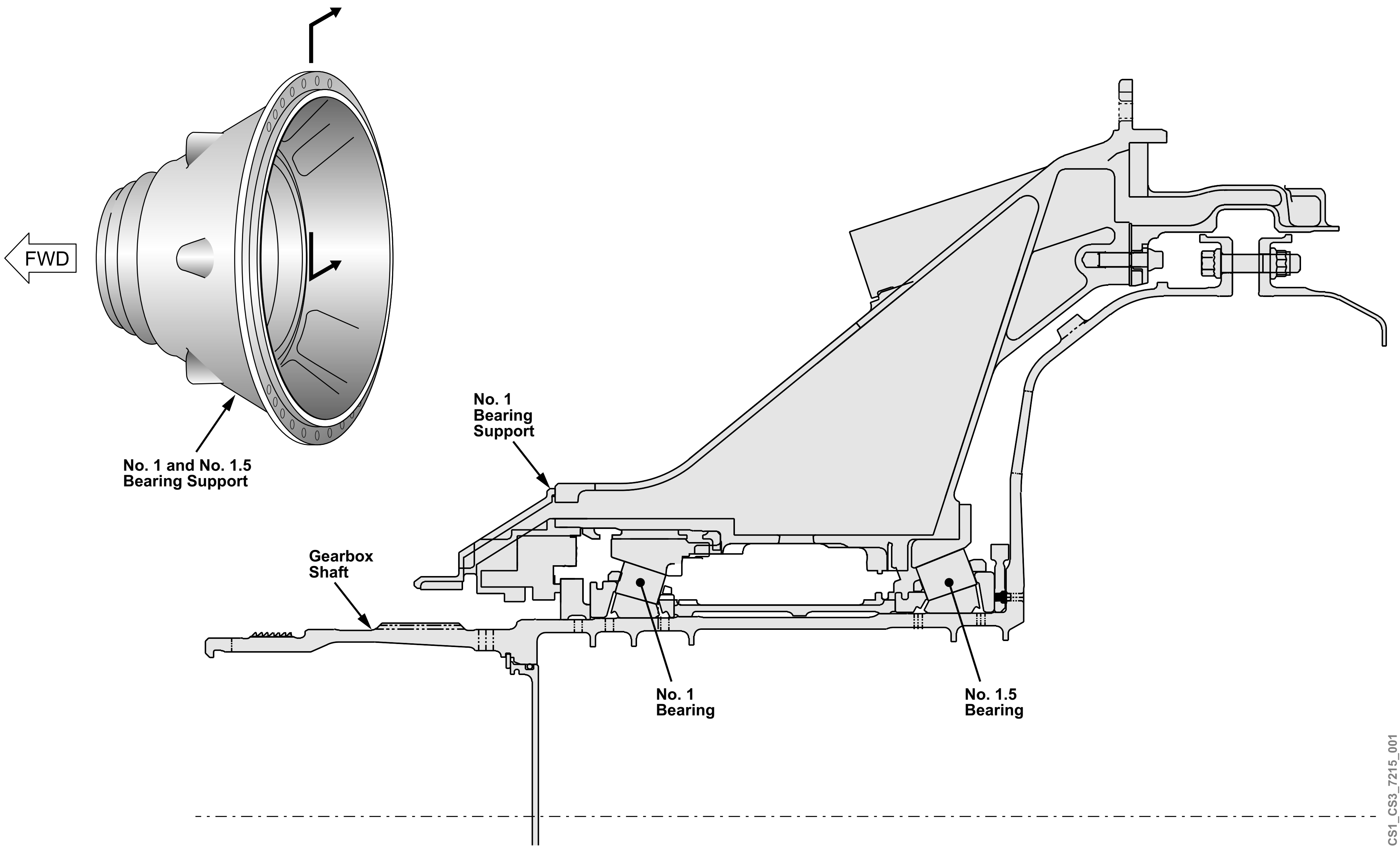


Figure 8: No. 1 and No. 1.5 Bearing Support Assembly and Cross-Section

72-22 FAN INTERMEDIATE CASE

GENERAL DESCRIPTION

The fan intermediate case (FIC) provides the support structure between the fan case, the fan drive gear system (FDGS), and the low-pressure compressor (LPC). The fan drive gearbox (FDG) is housed inside the FIC. It also provides the aerodynamic flow path between the fan rotor and the LPC.

The fan exit fairing splits the air between the fan duct and the low-pressure compressor. Air entering the fan exit stators is directed to the LPC variable stator vanes.

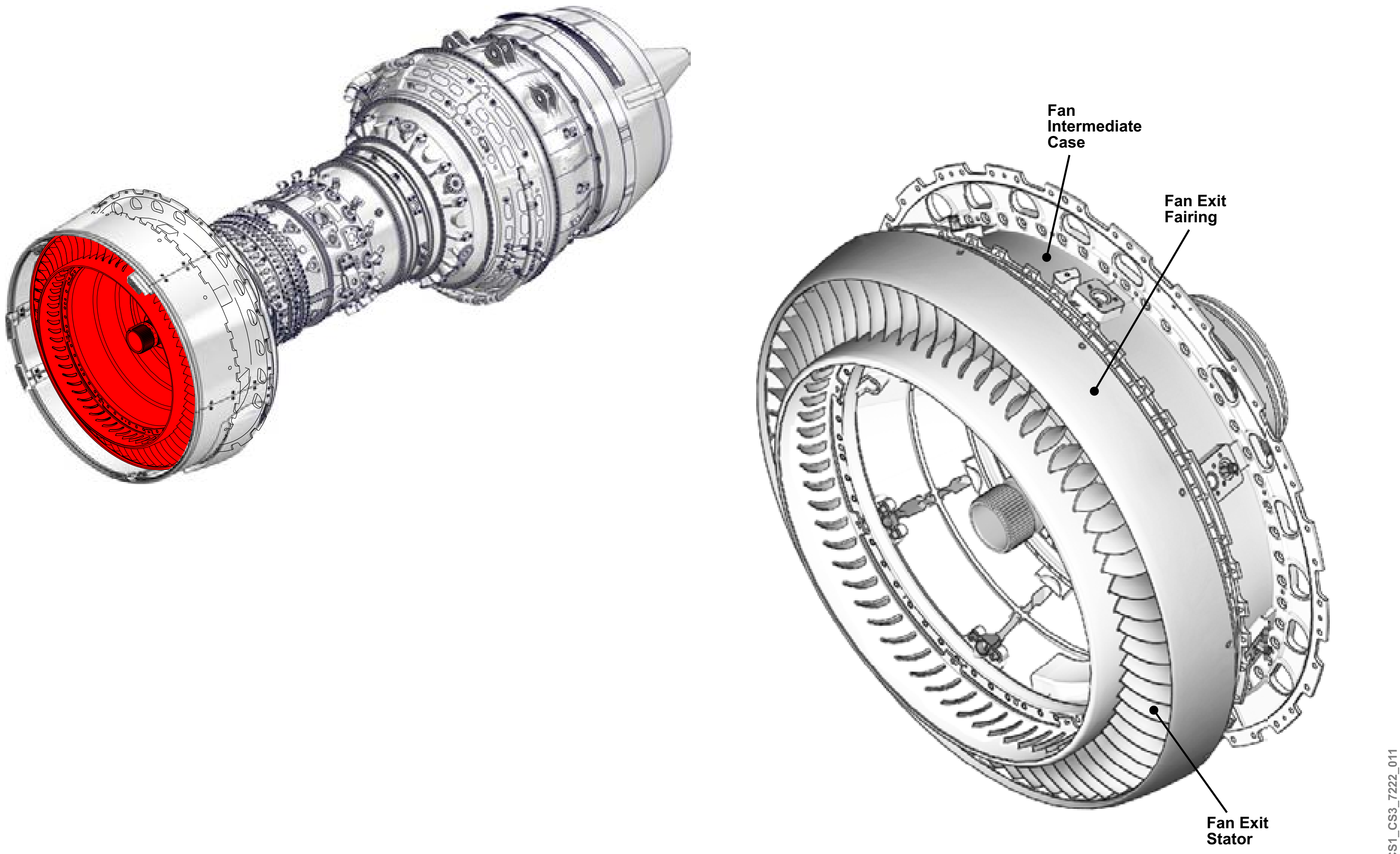


Figure 9: Fan Intermediate Case

DETAILED COMPONENT INFORMATION

FAN EXIT FAIRING

The fan exit fairing is installed on the front flange of the fan intermediate case (FIC). It splits the airflow between the engine core section, and the fan.

FAN EXIT STATOR

The fan exit stator is installed on the front flange of the FIC, just inside of the splitter. It straightens the fan air flow and directs it to the low-pressure compressor (LPC) variable vane stator.

FAN INTERMEDIATE CASE

The fan intermediate case (FIC) keeps the fan drive gear system (FDGS) and the LPC in alignment, and also provides support for the no. 1 and no. 1.5 bearing support assembly.

NO.2 BEARING AND SUPPORT ASSEMBLY

The no. 2 bearing and support assembly is attached to the rear center of the FIC. It absorbs rotor radial movement, and is oil damped on the outer race to decrease vibration.

FAN DRIVE GEARBOX COUPLING

The fan drive gearbox coupling (FDG coupling) is located in the heart of the FIC. Its aft end sits inside of bearing no. 2 and extends forward to couple with the fan drive gearbox system (FDGS).

LOW-PRESSURE COMPRESSOR VARIABLE STATOR VANES

The low-pressure compressor (LPC) variable stator vanes are located at the FIC aft end and mate with the LPC case front end. The LPC variable stators vane improve engine stability and operation.

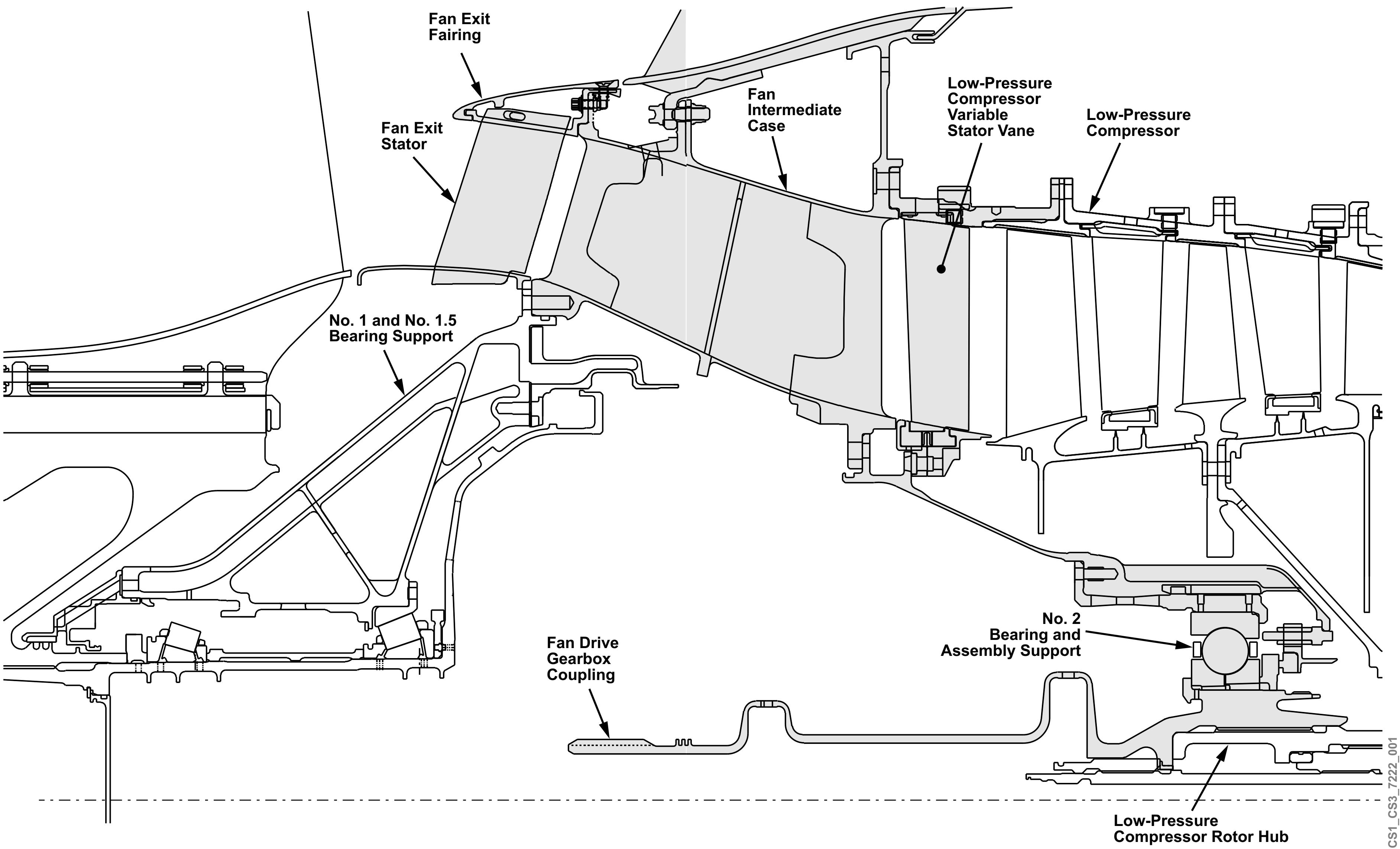


Figure 10: Fan Intermediate Case Cross-Section

72-31 LOW-PRESSURE COMPRESSOR

GENERAL DESCRIPTION

The low-pressure compressor (LPC) has three stages. Each stage has an integral bladed rotor (IBR). An IBR consists of a disc and blades manufactured as a single unit, resulting in increased efficiency and less weight than conventional bladed discs.

An annular bleed valve at the rear of the LPC compressor case, designated as the 2.5 bleed valve, provides operational stability for the LPC.

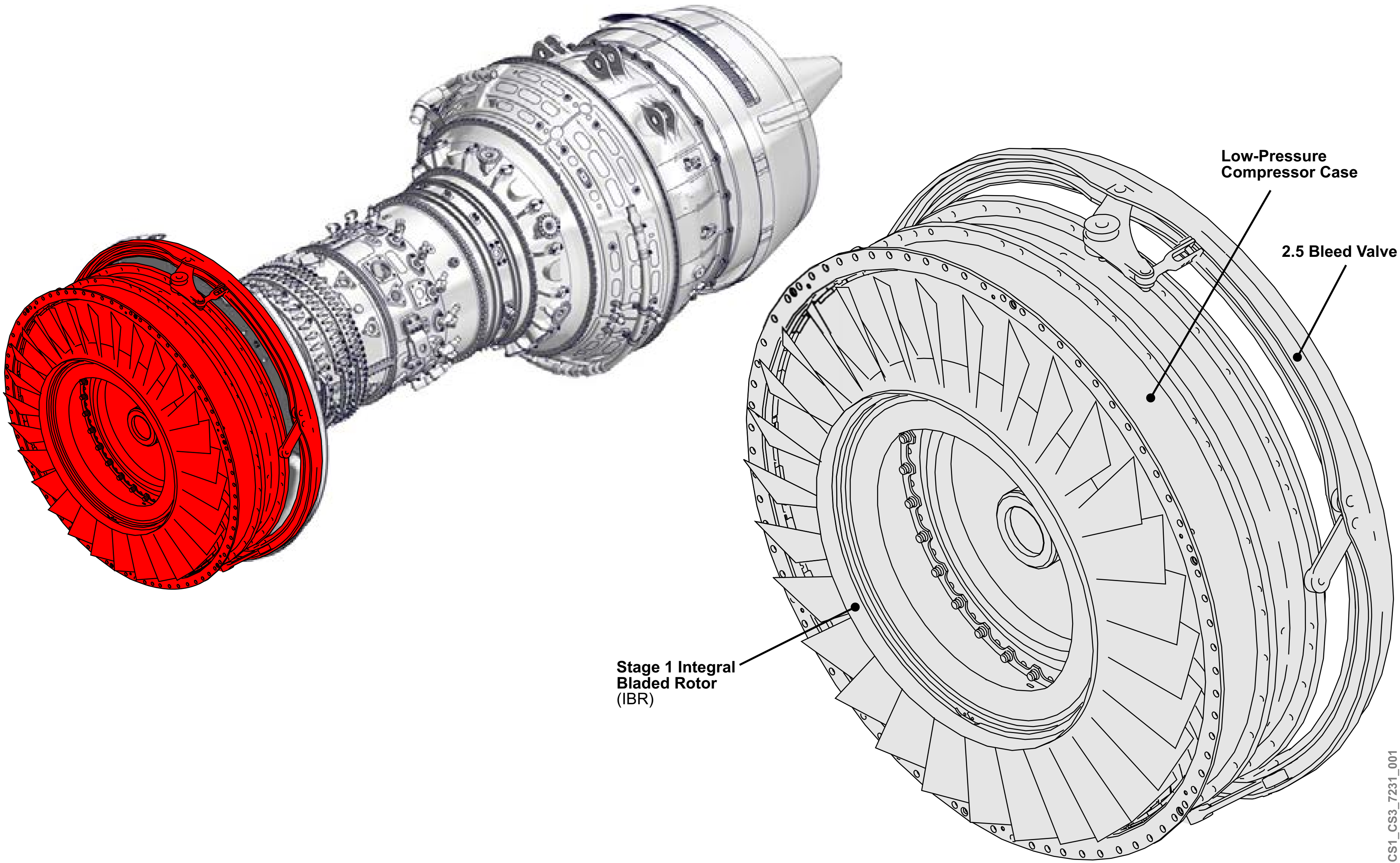


Figure 11: Low-Pressure Compressor

DETAILED COMPONENT INFORMATION

LOW-PRESSURE COMPRESSOR CASE

The 2.5 bleed valve is mounted on the LPC case. Removable fan exit liner segments cover the LPC case.

LOW-PRESSURE COMPRESSOR STATORS

There are two stages of stator vanes in the LPC. Each stage consists of fixed aluminum segment stators.

INTEGRAL BLADED ROTOR

The three integral bladed rotors (IBRs) are tied to the LPC rotor hub outer diameter using tie rods. The LPC rotor hub is splined onto the low-pressure turbine (LPT) shaft, just aft of the oil damped no. 2 bearing. It also couples to the fan drive gearbox (FDG) coupling.

NO.2 BEARING

The no. 2 bearing supports the LPC rotor hub, the FDG coupling, and the LPT shaft.

TIE RODS

There are 48 tie rods that connect the IBRs to the LPC rotor hub. They are located on the second IBR.

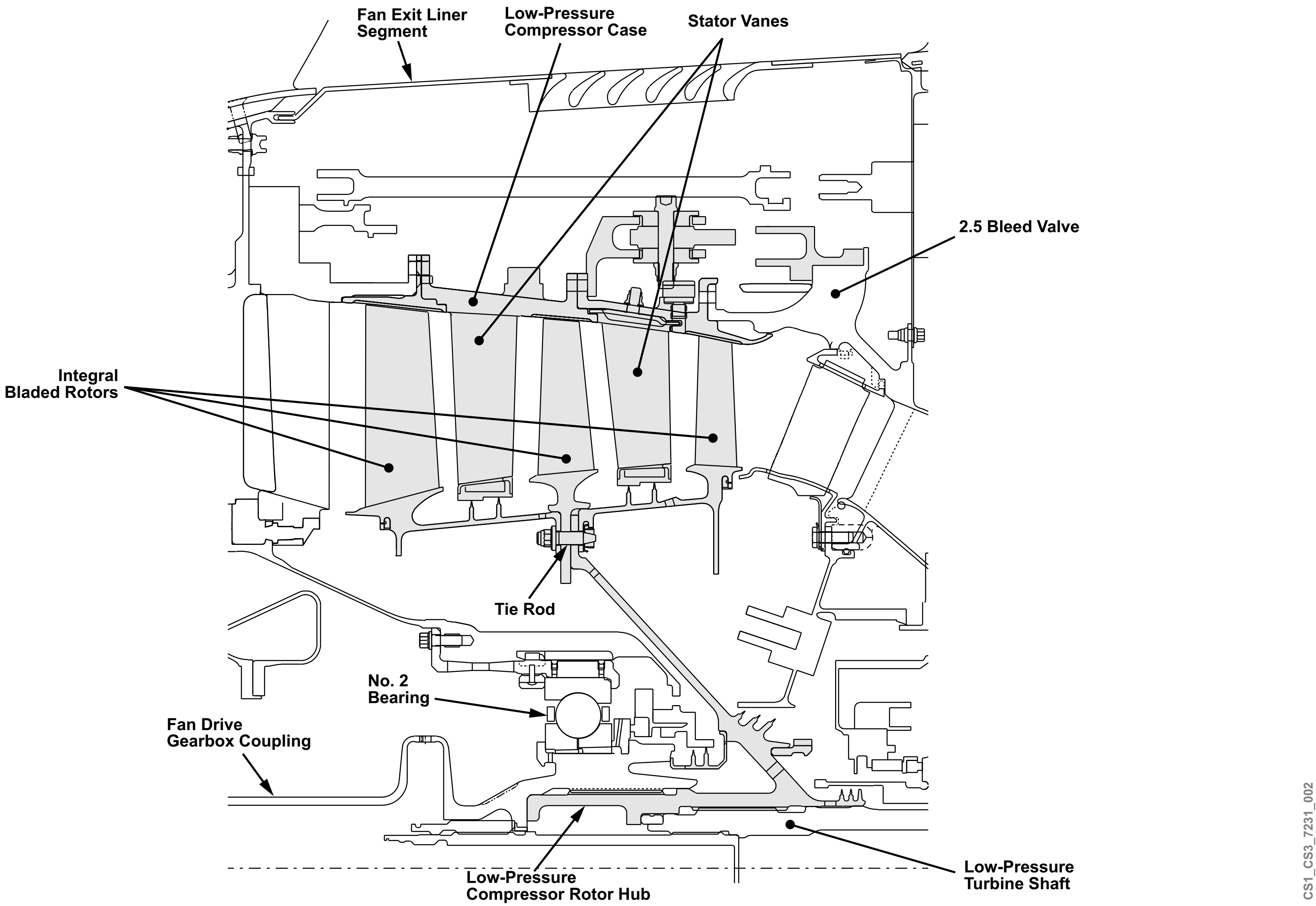


Figure 12: Low-Pressure Compressor Cross-Section

72-34 COMPRESSOR INTERMEDIATE CASE

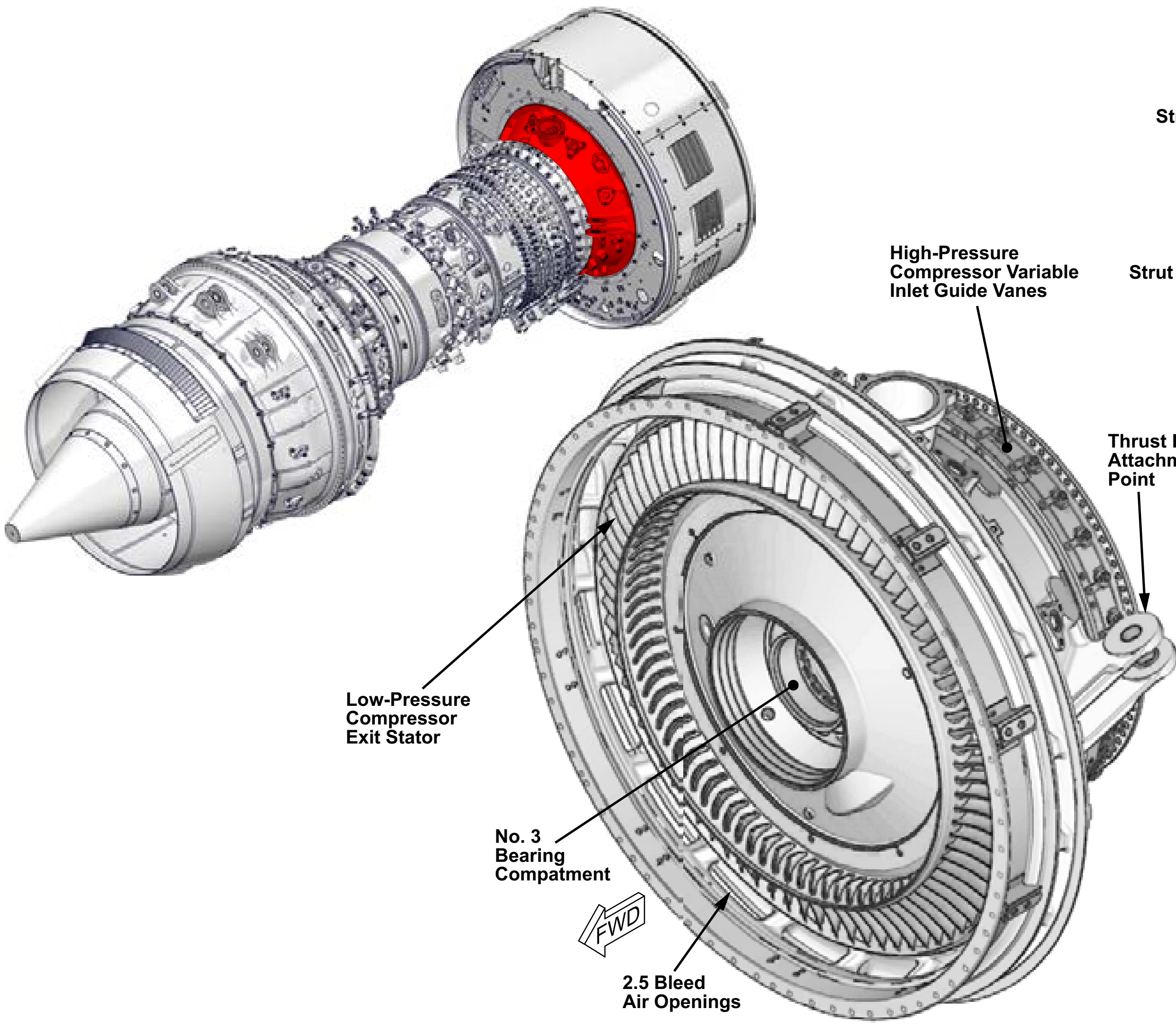
GENERAL DESCRIPTION

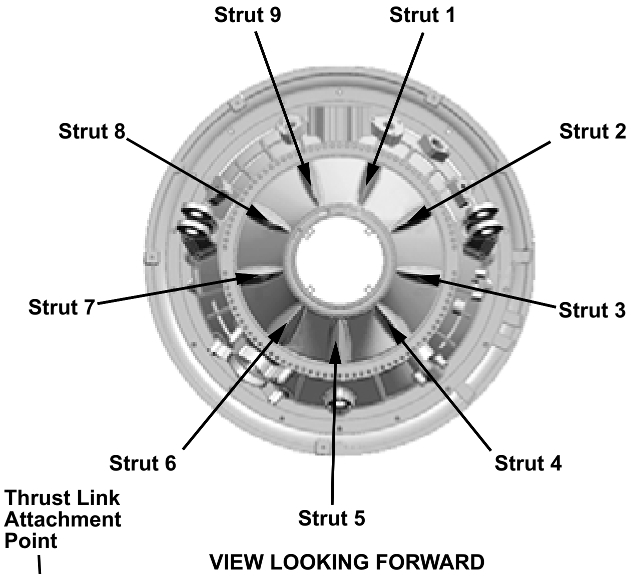
The compressor intermediate case (CIC) acts as the support structure for the low-pressure compressor (LPC), and the high-pressure compressor (HPC).

The CIC carries the thrust loads from the engine through the two thrust links attachment points to the aft engine mounts. The two thrust link attachment points are on the CIC at the 2 o'clock and 10 o'clock positions. The CIC also houses the no. 3 bearing compartment.

The case structure includes nine struts and an opening for a gearbox drive shaft that transmits torque to the angle gearbox. The case is equipped with an opening for the discharge of the 2.5 bleed air.

The hollow struts provide support for the no.3 bearing compartment. The struts interior provide access to the no.3 bearing for lubrication supply and scavenge lines. The struts are numbered clockwise beginning at the 1 o'clock position. A table lists all the struts internal passage usage.





COMPRESSOR INTERMEDIATE CASE STRUTS	
Strut No.	Used For
1	No. 2 bearing buffer air supply
2	Breather tube
3	No. 3 bearing oil supply
4	N1 probe
5	No. 3 bearing oil scavenge
6	Gearbox drive shaft
7	Empty
8	No. 3 bearing buffer air supply
9	No. 3 bearing buffer air supply

CS1_CS3_7234_001

Figure 13: Compressor Intermediate Case

DETAILED COMPONENT INFORMATION

COMPRESSOR INTERMEDIATE CASE

The compressor intermediate case (CIC) includes the no. 3 bearing, the 2.5 bleed air outlet, and the thrust reverser (T/R) door inner fixed structure (IFS) support.

NO. 3 BEARING COMPARTMENT

The no. 3 bearing compartment is located in the aft section of the compressor intermediate case (CIC). The no. 3 bearing supports the front of the high-pressure compressor's (HPC) rotor shaft axial movement and absorbs vibrations.

GEARBOX DRIVE BEVEL GEAR

The gearbox drive bevel gear is located at the 8 o'clock position in the no. 3 bearing compartment. The no. 3 bearing bevel gear engages the gearbox drive bevel gear to supply torque to the angle gearbox drive shaft. This shaft drives the angle gearbox, which in turn, supplies torque to the main gearbox.

FIRE SHIELD

The fire shield is attached to the outer perimeter of the CICs aft section. It is a metal containment shield used to protect the LPC structure from fire originating in the engine core compartment. The outer section of the shield acts as a support for the thrust reverser (T/R) door inner fixed structure (IFS).

LOW-PRESSURE COMPRESSOR EXIT STATOR

The low-pressure compressor (LPC) exit stator segments are located at the front of the CIC. The LPC exit stators divert air from the LPC, and direct it towards the HPC.

HIGH-PRESSURE COMPRESSOR VARIABLE INLET GUIDE VANES

The high-pressure compressor variable inlet guide vanes are located at the aft end of the CIC. They are adjusted during engine operation to optimize engine performance.

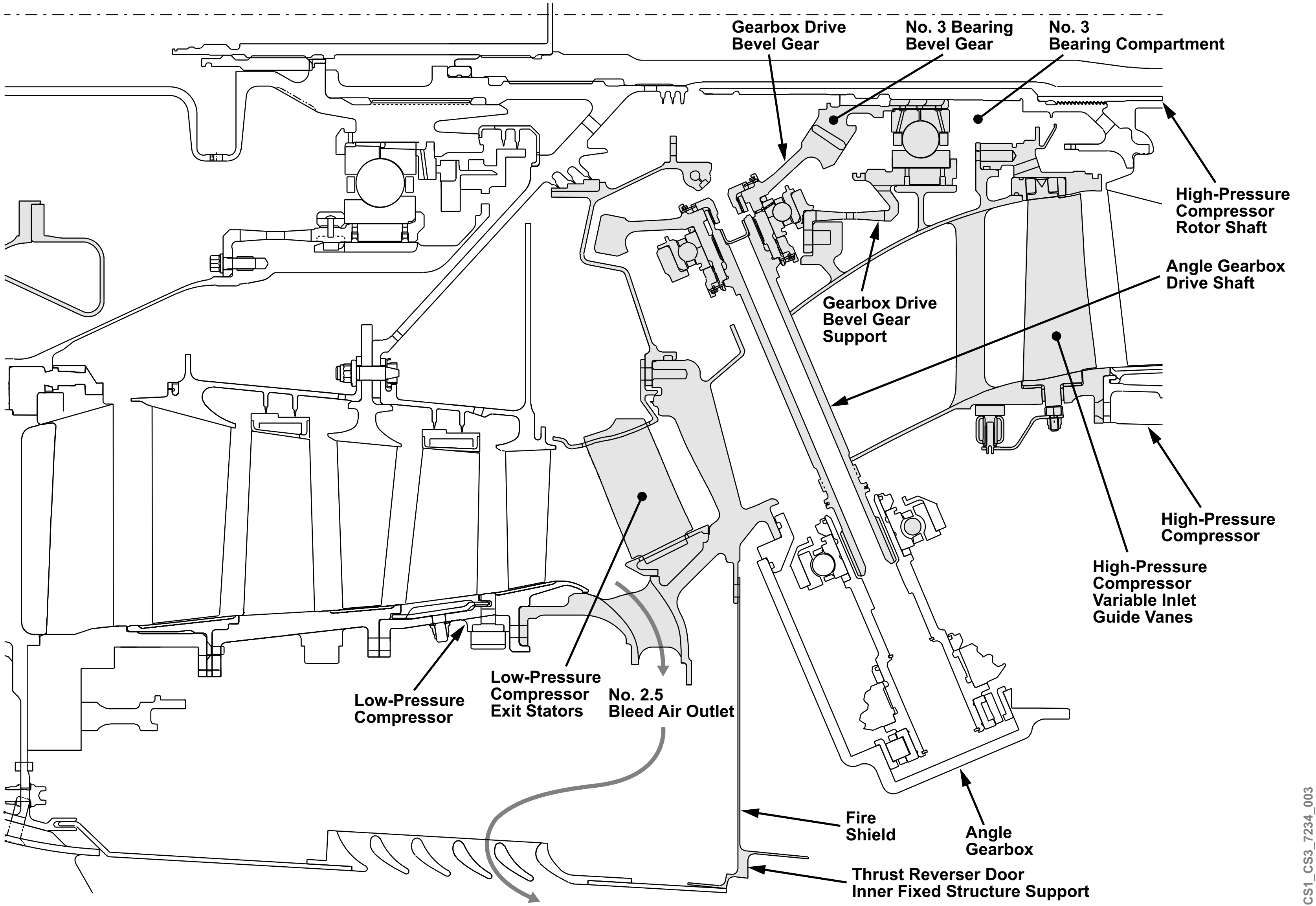


Figure 14: Compressor Intermediate Case Cross-Section

72-35 HIGH-PRESSURE COMPRESSOR

GENERAL DESCRIPTION

The high-pressure compressor (HPC) increases the speed and pressure of the primary gaspath air. The HPC is driven by the high-pressure turbine (HPT) through the HPC rotor shaft.

The first seven rotor stages are integral bladed rotors (IBRs). The first three stator stages are variable and actuated through unison rings on the outside of the HPC case.

The forward HPC case is split horizontally and mated with a one piece aft ring case.

The HPC supplies 4th and 8th stage air to the bleed air system. The 4th stage bleed air port is located on the HPC. Engine cooling is supplied by 4th stage air through cooling air ports on the HPC.

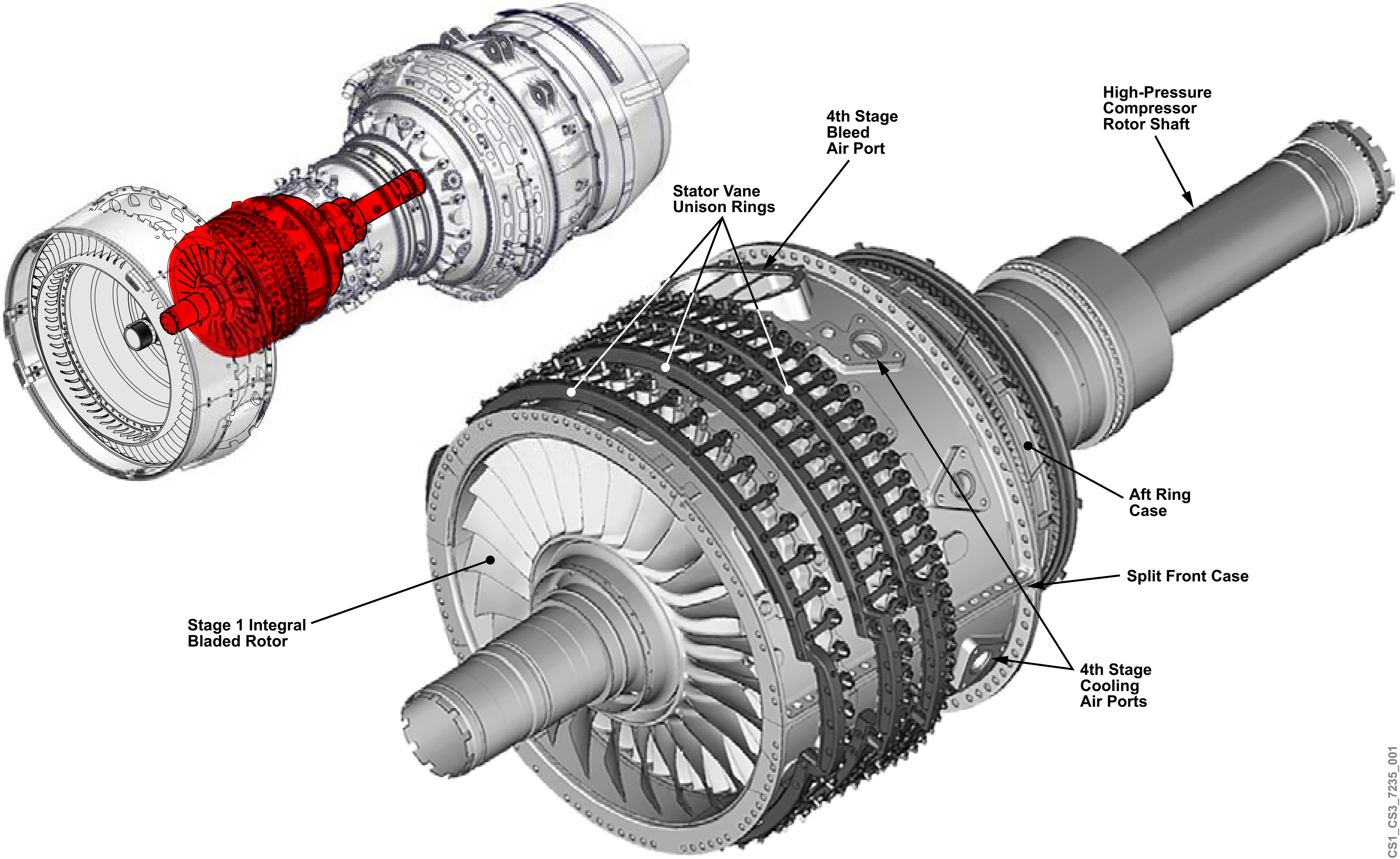


Figure 15: High-Pressure Compressor

DETAILED COMPONENT INFORMATION

ROTORS

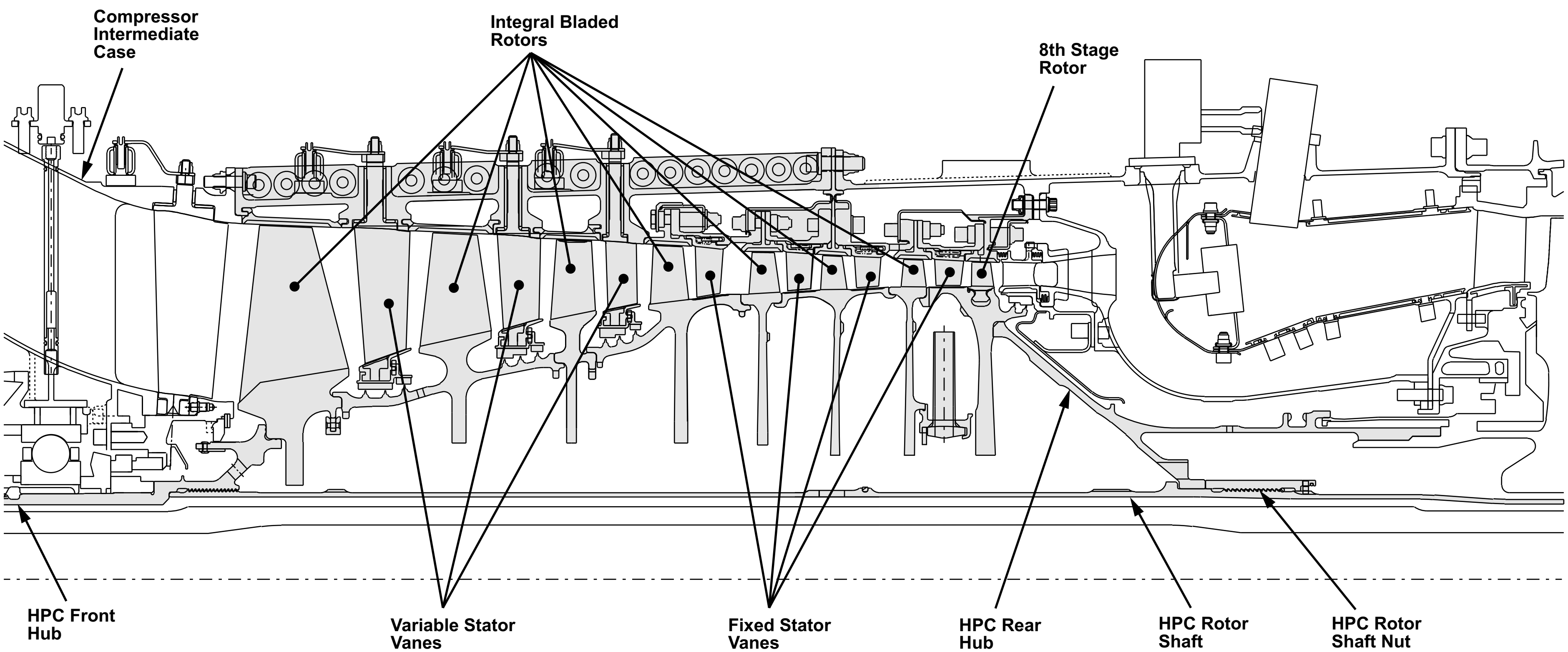
There are eight rotor stages in the high-pressure compressor (HPC). The first seven rotor stages are integral bladed rotors (IBRs). The 8th stage rotor is a standard blade and disk assembly.

HIGH-PRESSURE COMPRESSOR ROTOR SHAFT

The rotor stage hubs are held together on the HPC rotor shaft. The HPC front hub acts as the forward stop for the IBRs and the HPC rear hub secures the 8th stage rotor. This rotor hub is held by the HPC rotor shaft nut on the HPC rotor shaft.

STATORS

The first three stator stages of the HPC are variable, while the remaining four stages are fixed stages.



LEGEND

HPC High-Pressure Compressor

Figure 16: High-Pressure Compressor Cross-Section

72-41 DIFFUSER, COMBUSTOR AND TURBINE NOZZLE ASSEMBLY

GENERAL DESCRIPTION

The diffuser case, combustion chamber, and turbine nozzle assemblies work together to provide the internal combustion that drives the turbines.

Compressed air flows from the high-pressure compressor (HPC) exit stator into the diffuser and then into the combustor section. The airflow is straightened and diffused, reducing the velocity, while maintaining pressure before it enters the combustion chamber. The diffuser case has the bleed air ports for the 6th stage bleed air, cooling air, and cowl anti-ice (CAI).

The combustion chamber inner perimeter and the turbine nozzles are attached to the diffuser inner case. The combustion chamber consists of an inner and outer chamber liner assembly. The diffuser case also has provisions for fuel nozzles and igniter plugs.

Metered fuel is supplied to the combustion chamber through 16 fuel nozzles, where it is mixed with the air. The heat of the combustion process causes the airflow to expand and accelerate rearward.

Turbine nozzle guide vanes then direct the high-temperature, high-velocity gases out of the combustion chamber to drive the turbines.

The diffuser outer case supports the main gearbox (MGB) using two mount brackets, located at the 3 o'clock and 9 o'clock position.

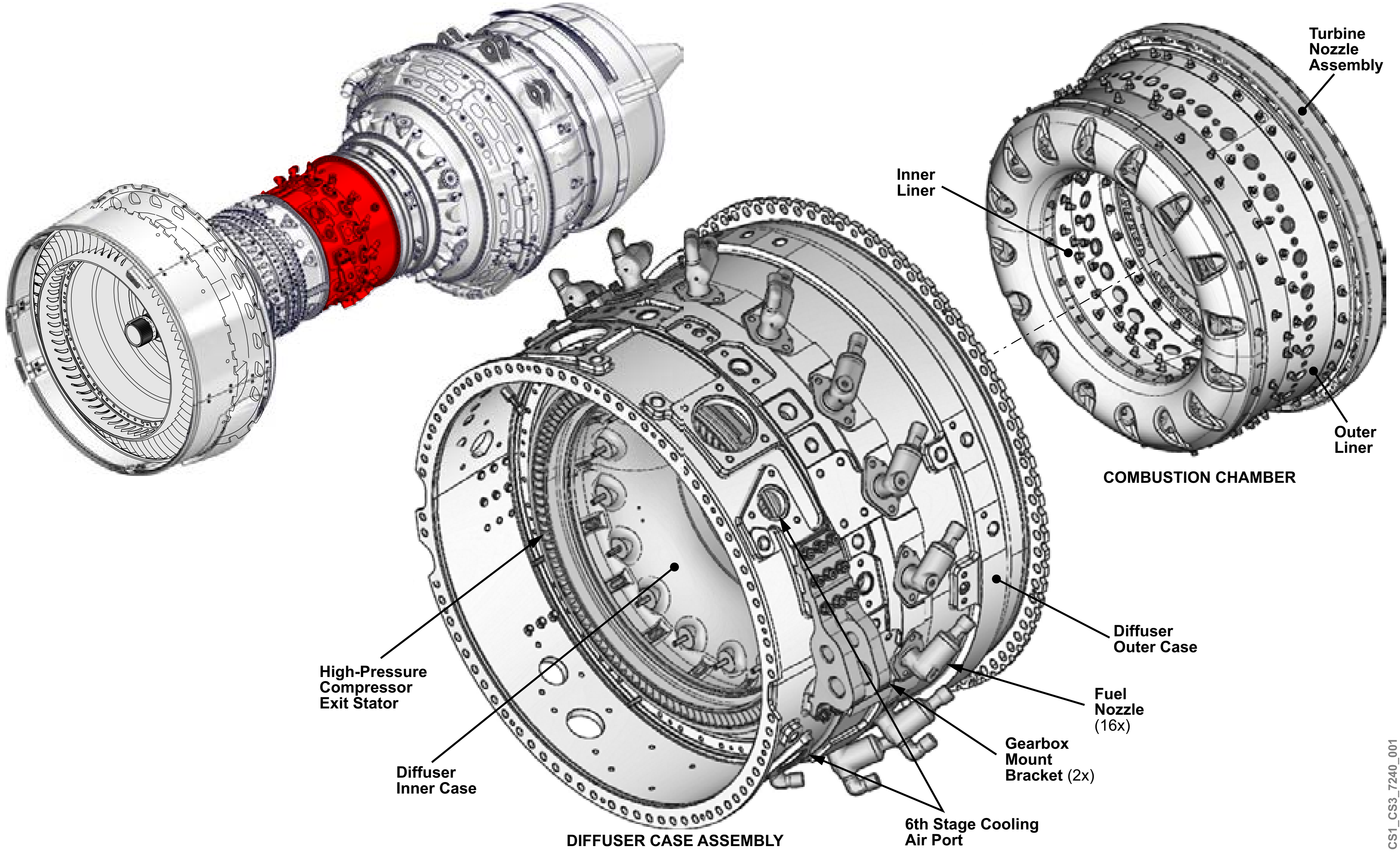


Figure 17: Diffuser, Combustor, and Turbine Nozzle Assembly

DETAILED COMPONENT INFORMATION

DIFFUSER CASE ASSEMBLY

The diffuser inner case attaches to a common diffuser outer case flange with the high-pressure compressor (HPC). The inner case attaches to the high-pressure turbine (HPT) flange.

Two igniter plugs located at the 4 o’clock and 5 o'clock position extend into the combustion chamber.

COMBUSTION CHAMBER

The HPC air enters the diffuser and flows through the hood and cooling holes around the inner and outer liners of the combustion chamber.

A bulkhead is used to attach both combustion chamber inner and outer liners, and the hood. Fuel nozzles inject fuel through orifices in the bulkhead where swirlers spin the air/fuel mixture inside the combustion chamber.

HIGH-PRESSURE TURBINE NOZZLE ASSEMBLY

The turbine nozzle assembly attaches to the cooling air duct, which is bolted with the combustion chamber inner liner flange, to the inner diffuser case.

The nozzle assembly has 32 air-cooled thermal-barrier coated guide vanes between the inner and outer combustion chamber liners. Located at the rear of the combustion chamber, they direct the hot gases to the 1st stage high-pressure turbine blades.

Each hollow vane is cooled by diffuser air and use cooling air exit holes on the airfoil external surface to reduce vane temperature.

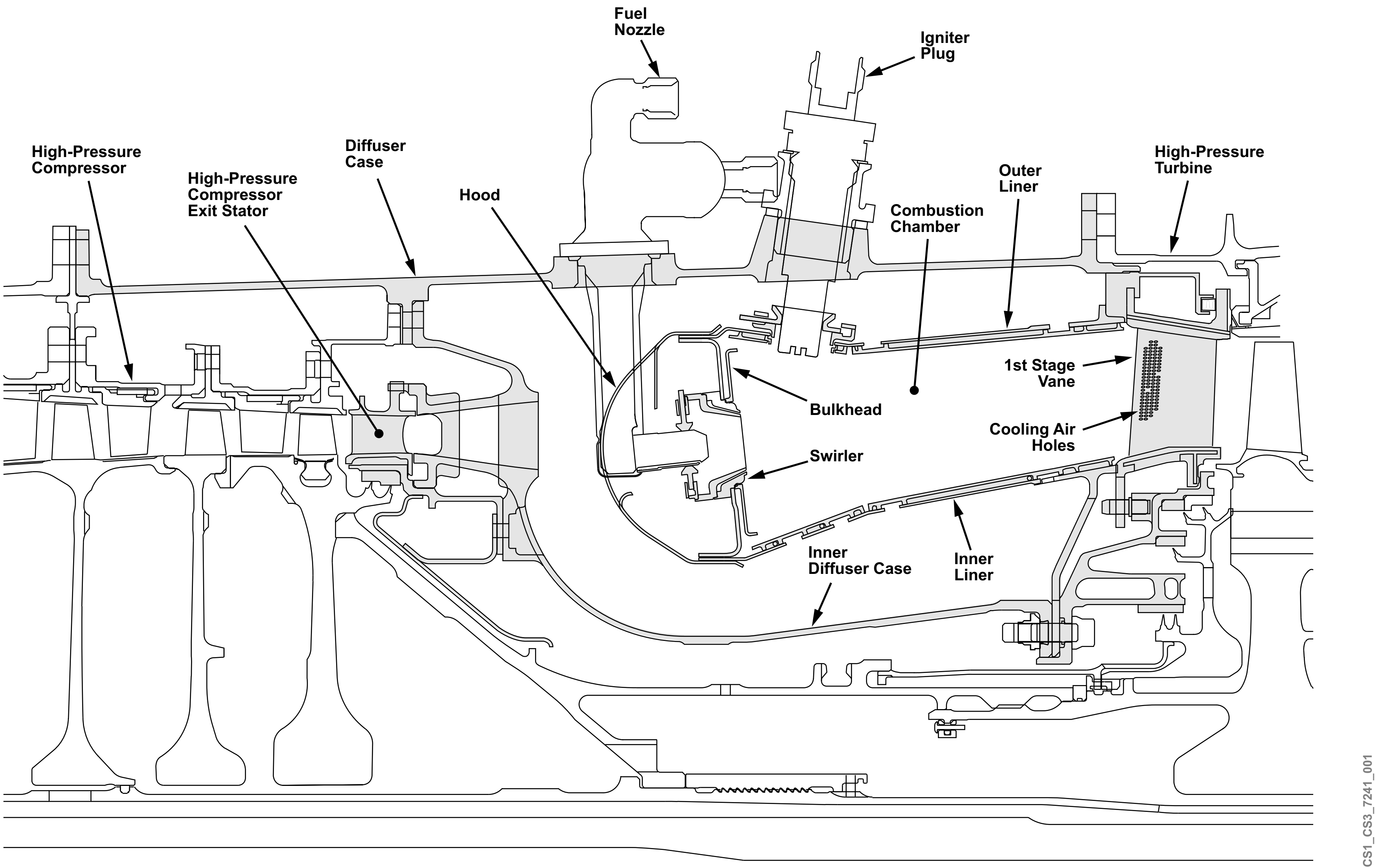


Figure 18: Diffuser and Combustor Cross-Section

72-51 HIGH-PRESSURE TURBINE

GENERAL DESCRIPTION

The high-pressure turbine (HPT) provides the rotational force to drive the HPC by extracting energy from the hot combustion gases. The HPT case provides containment of the HPT rotors. Cooling air flows through the hollow 2nd stage stator vanes to keep them cool.

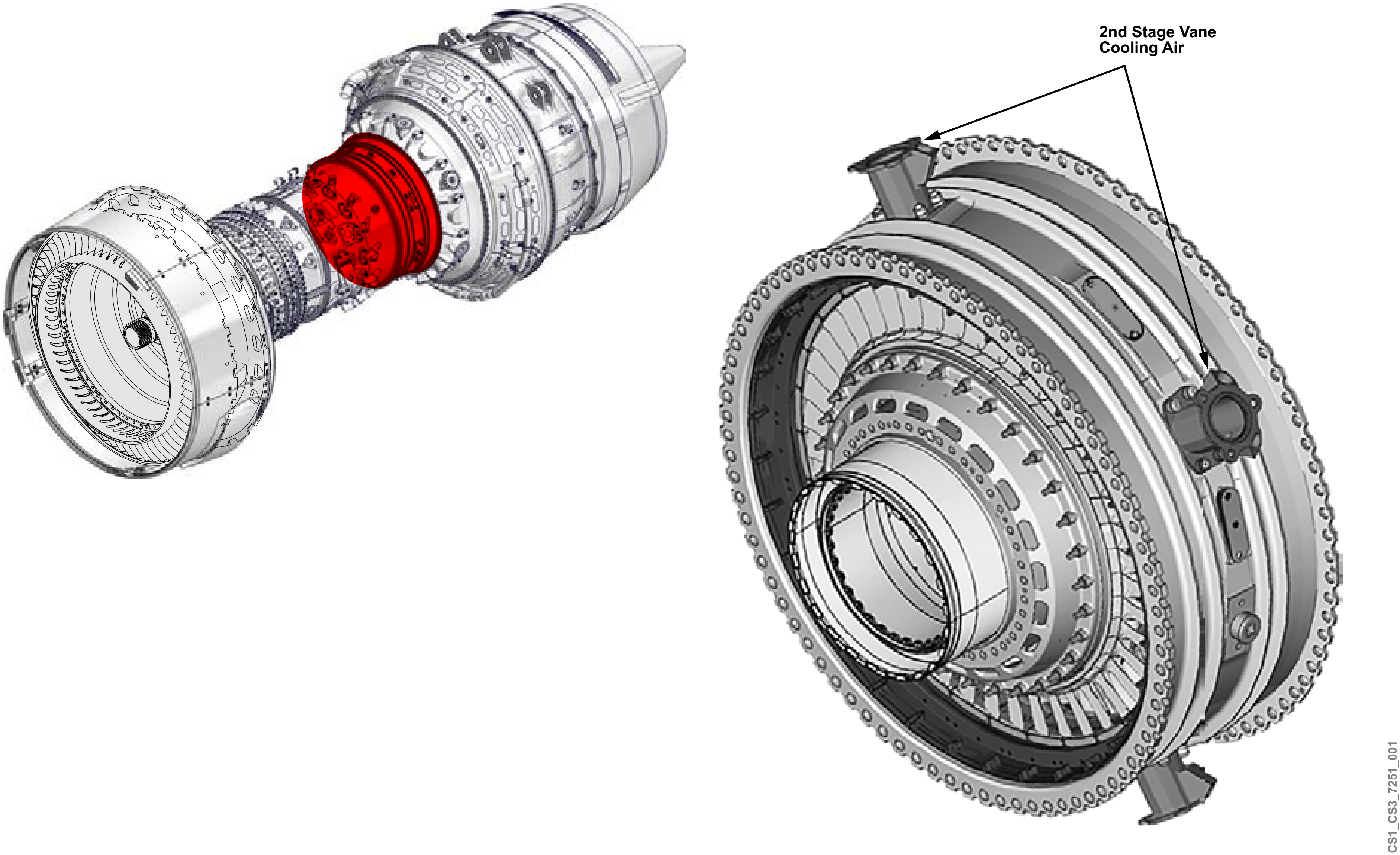


Figure 19: High-Pressure Turbine

DETAILED COMPONENT INFORMATION

HIGH-PRESSURE TURBINE

Each of the two turbine rotor stages contain 44 blades coated with a thermal barrier and internally cooled. The 1st stage hub engages a spline in the HPC rear hub at the front. The 2nd stage hub is attached by a nut on the HPC rotor shaft at the rear.

The blades are attached to their hubs in fir tree type slots and held in position with retaining plates. The blades outer air seal segments are ground as a set with the turbine blades to maximize rotor tip sealing.

Internally cooled 2nd stage stator vanes are mounted inside the case between the two rotor stages. Fan air from the active clearance control (ACC) system cools the HPT case, thus decreasing its inner diameter and increasing the turbine efficiency. The HPT case has mounting points for the turbine ACC manifold.

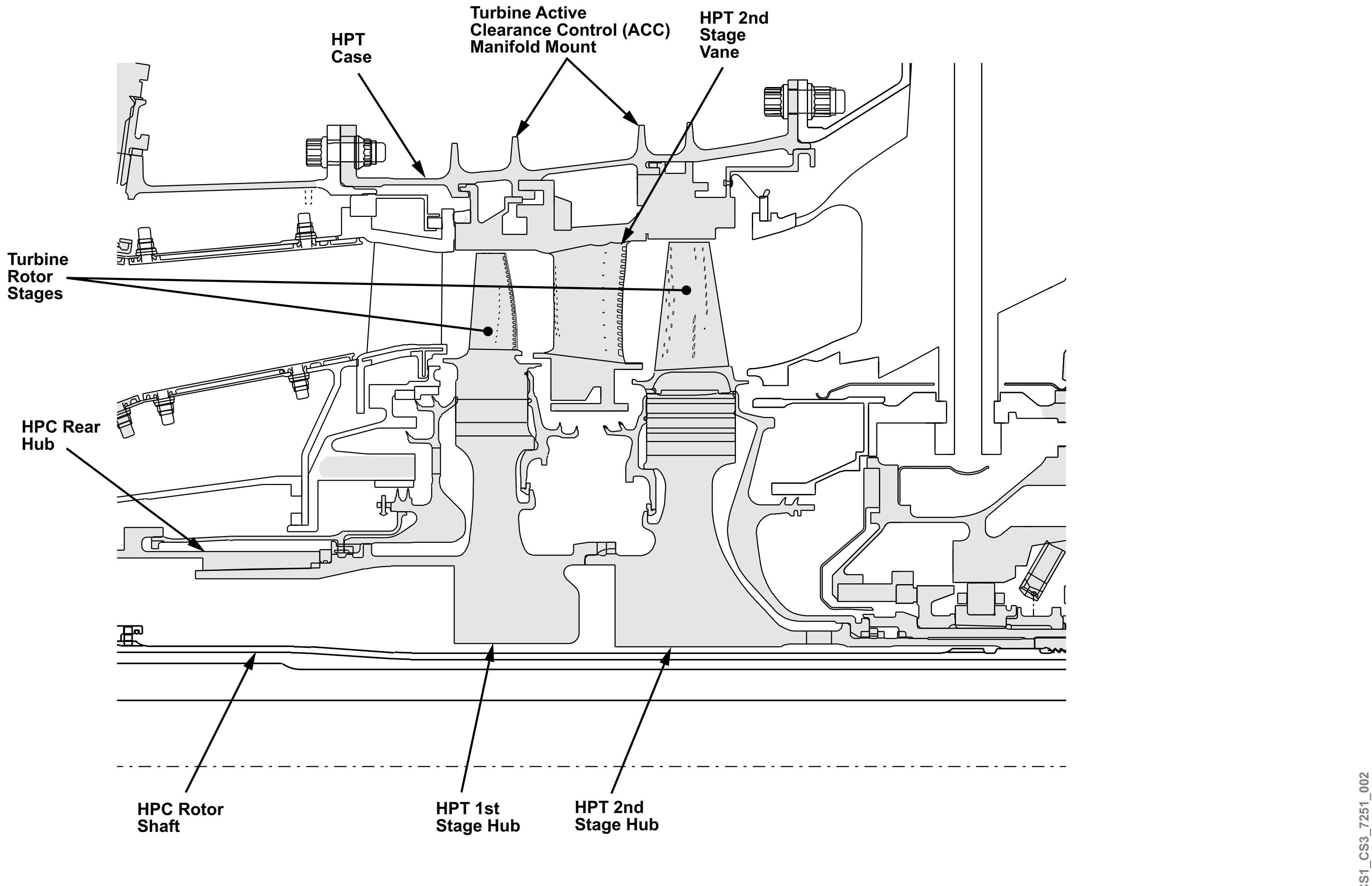


Figure 20: High-Pressure Turbine Cross-Section

72-52 TURBINE INTERMEDIATE CASE

GENERAL DESCRIPTION

The turbine intermediate case (TIC) is used to turn the high-pressure turbine (HPT) airflow to align it with the counter rotating low-pressure turbine (LPT). The no. 4 roller bearing housing assembly is located within the TIC.

The turbine intermediate case houses a turbine stator assembly which connects to the intermediate case using seven support rods.

The turbine stator assembly is composed of 14 airfoils that direct HPT airflow to the LPT rotor.

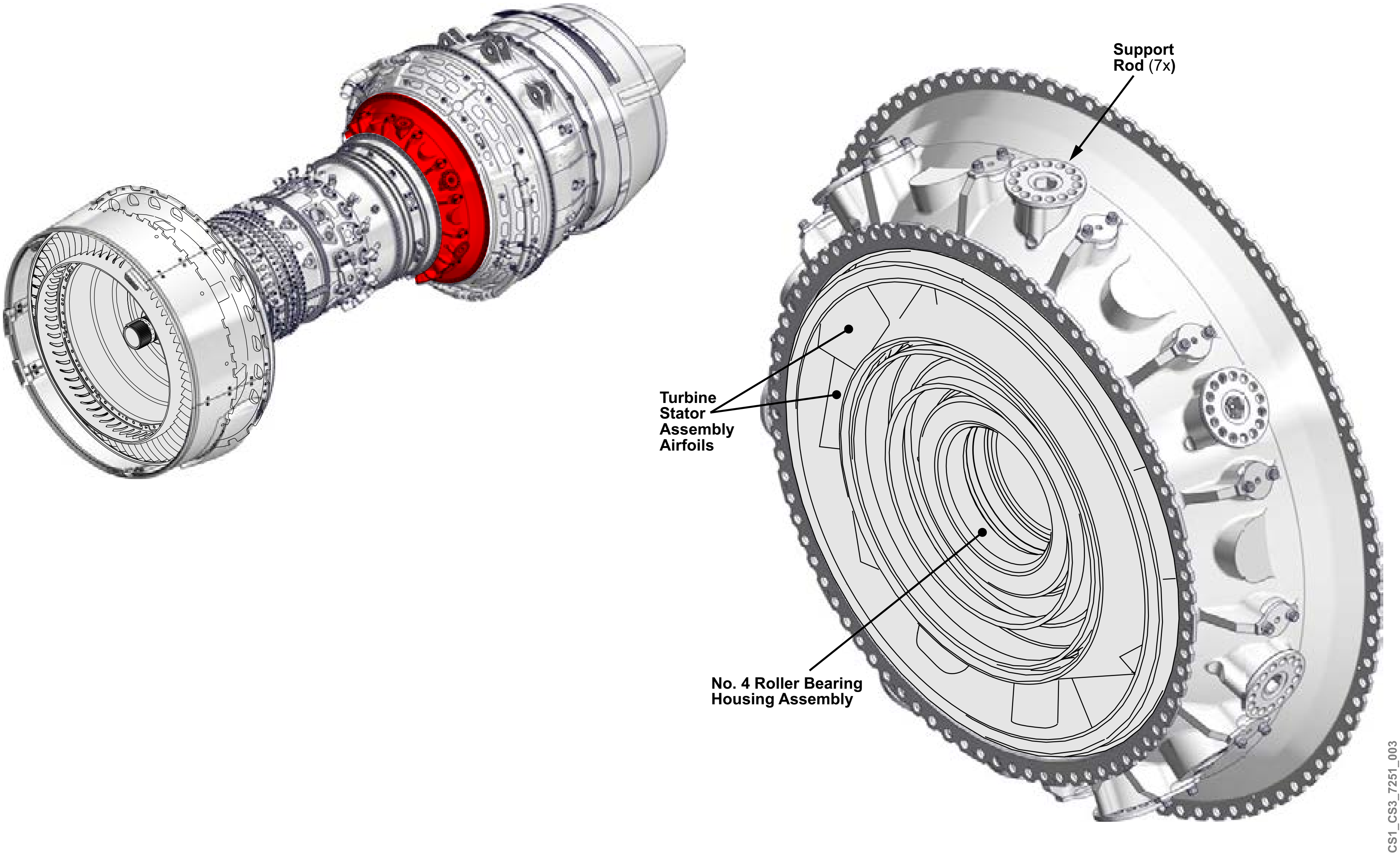


Figure 21: Turbine Intermediate Case

DETAILED COMPONENT INFORMATION

TURBINE INTERMEDIATE CASE

The turbine intermediate outer case holds the turbine intermediate inner case in a center position with seven rods. These rods run through the turbine stator vanes and provides support to the no.4 roller bearing housing assembly. It is oil damped in order to absorb rotor vibration, and supports the HPT drive shaft axially and radially. Heat shields are provided as thermal barriers to protect the no. 4 roller bearing housing assembly.

Buffer air from the 4th stage provides sealing and cooling for the no. 4 bearing compartment, and prevents coking of engine oil.

Lubrication and air cooling tubes are routed through the hollow turbine stator vanes to and from the no.4 roller bearing housing assembly.

An outer heat shield provides a thermal barrier to the TICs outer case.

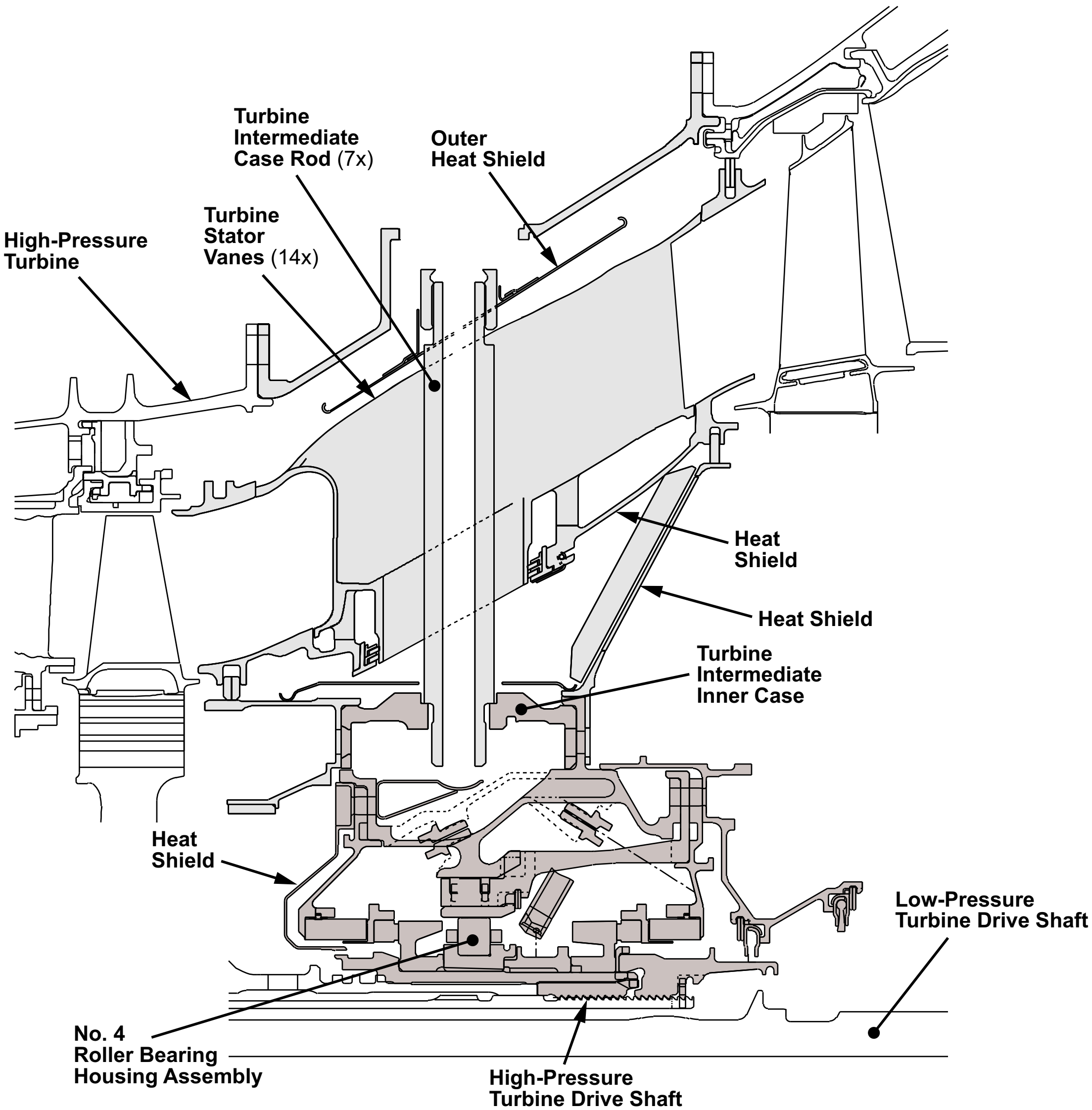


Figure 22: Turbine Intermediate Case Cross-Section

72-53 LOW-PRESSURE TURBINE

GENERAL DESCRIPTION

The low-pressure turbine (LPT) drives the LPT shaft. The LPT case provides an attachment for the active clearance control (ACC) manifold.

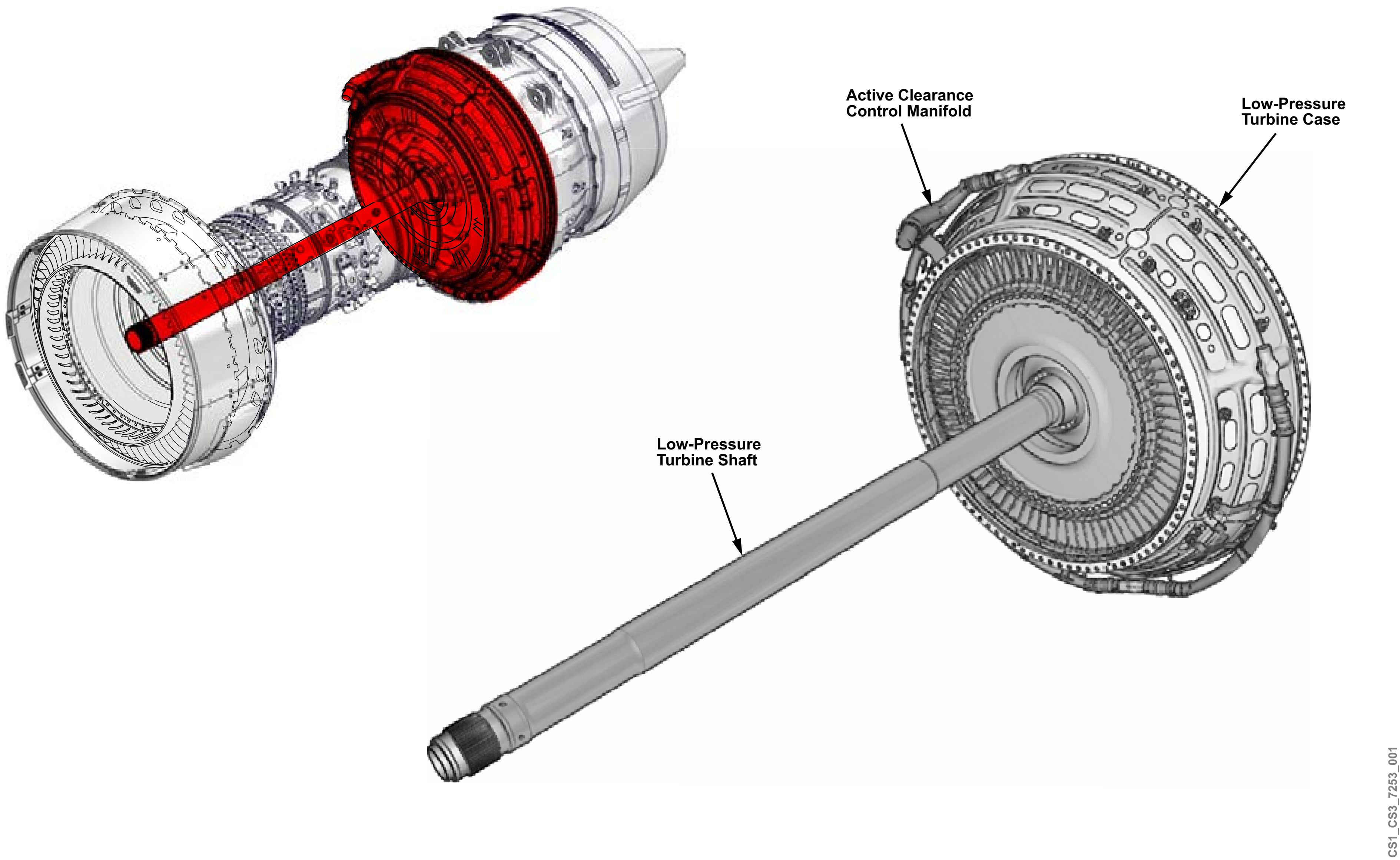


Figure 23: Low-Pressure Turbine

DETAILED COMPONENT INFORMATION

LOW-PRESSURE TURBINE

The low-pressure turbine (LPT) consists of 3 rotor stages and 2 stages of stator vanes.

The LPT drive shaft runs through the center of the engine.

The LPT 2nd stage rotor hub is splined to the LPT drive shaft. The low-pressure turbine (LPT) 1st stage and 3rd stage rotor disks are bolted together on the LPT 2nd stage hub using tie rods.

The LPT 1st stage rotor blades are hollow, while the 2nd and 3rd stage rotor blades are solid alloy. The LPT 2nd and 3rd stator vanes are segment arrangement type.

Fan air is sprayed around the exterior of the LPT case from the active clearance control (ACC) cooling manifold. This reduces the LPT case diameter, reducing the turbine blades tip clearance, thus improving fuel efficiency.

Insulation segments, located on the inside periphery of the LPT, protect the LPT case from heat.

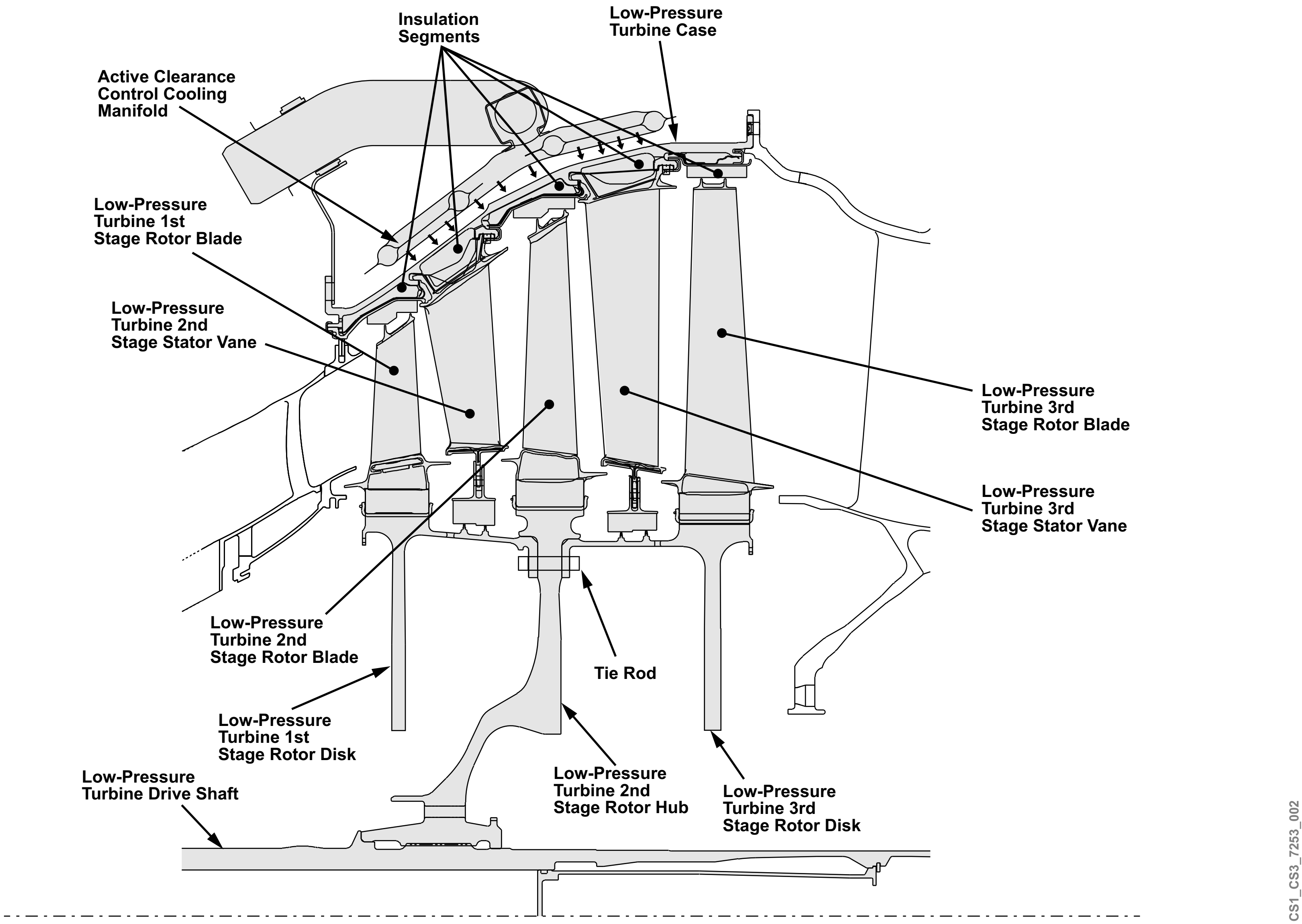


Figure 24: Low-Pressure Turbine Cross-Section

72-54 TURBINE EXHAUST CASE

GENERAL DESCRIPTION

The turbine exhaust case (TEC) is the aft most engine structural element. It is formed by the outer duct surface, the inner duct surface, eleven airfoil shaped struts, and the no.5 and no.6 bearing housings.

The TEC is attached to the rear flange of the low-pressure turbine (LPT) at the front end, and to the exhaust system at the aft end. It forms a transition duct between the LPT and the exhaust case.

Two ground handling bosses are provided at the 3 o’clock and 6 o'clock position.

The TEC also houses the aft engine mount lugs at its top surface.

Eight exhaust gas temperature (EGT) probes bosses are provided on the case outer surface.

Oil supply and scavenge tubes use the hollow interior of two lower struts to reach no. 5 and no. 6 bearing housings.

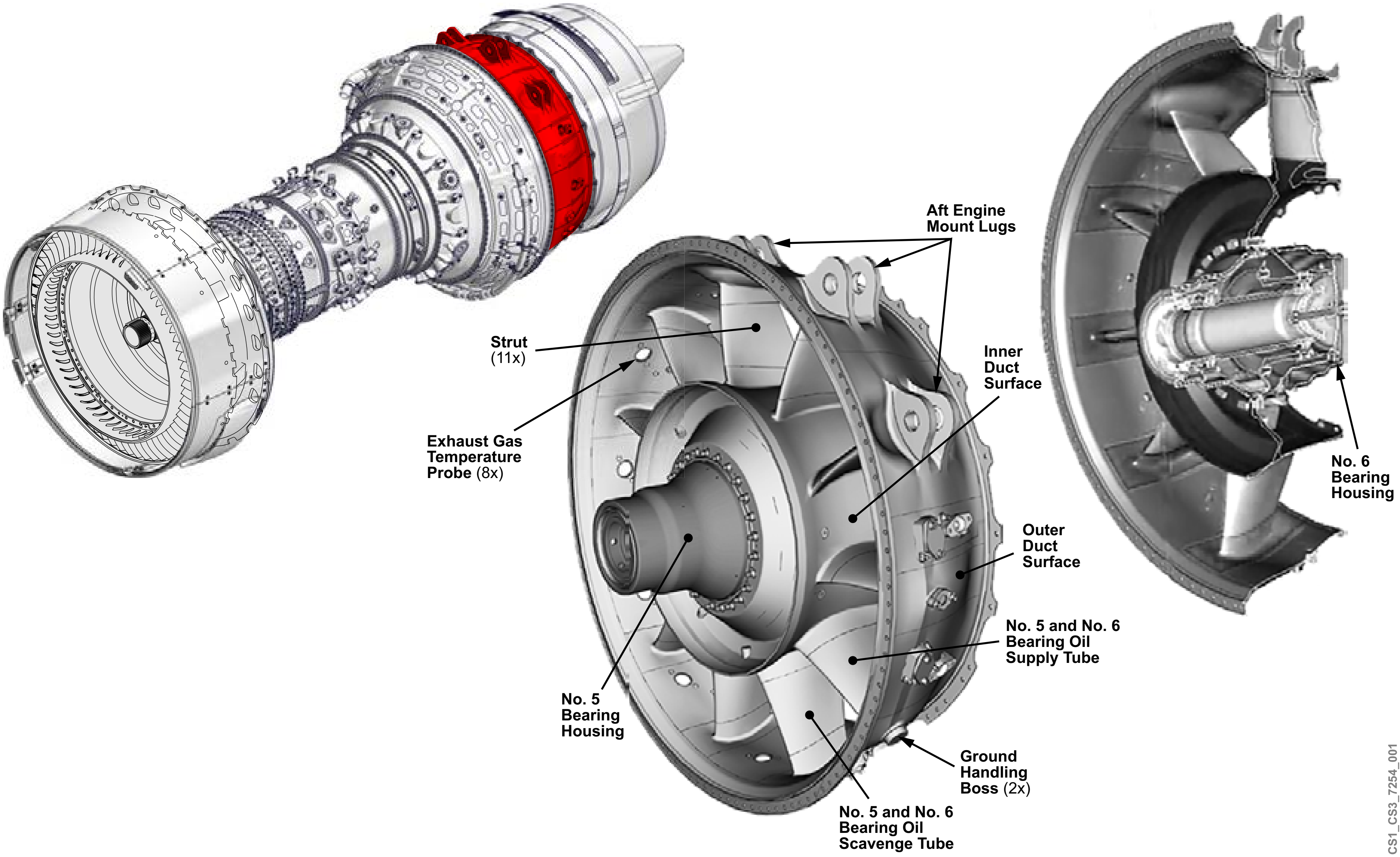


Figure 25: Turbine Exhaust Case

DETAILED COMPONENT INFORMATION

TURBINE EXHAUST CASE

The turbine exhaust case (TEC) and struts attach to the no. 5 and no. 6 bearing housings. Heat shields are installed to protect both bearing housings from excessive hot exhaust gas flow. The no. 5 and no. 6 bearings are roller bearings that provide the radial support for the LPT shaft. Both bearings are oil-damped to absorb vibration.

The no. 5 bearing rests against its forward seat and is properly kept separated from no.6 bearing by a shaft spacer. The no. 6 bearing retaining nut is installed on the low-pressure turbine (LPT) shaft to keep both bearing assemblies together.

The three aft engine mount lugs carry part of the engine weight and loads to the pylon structure.

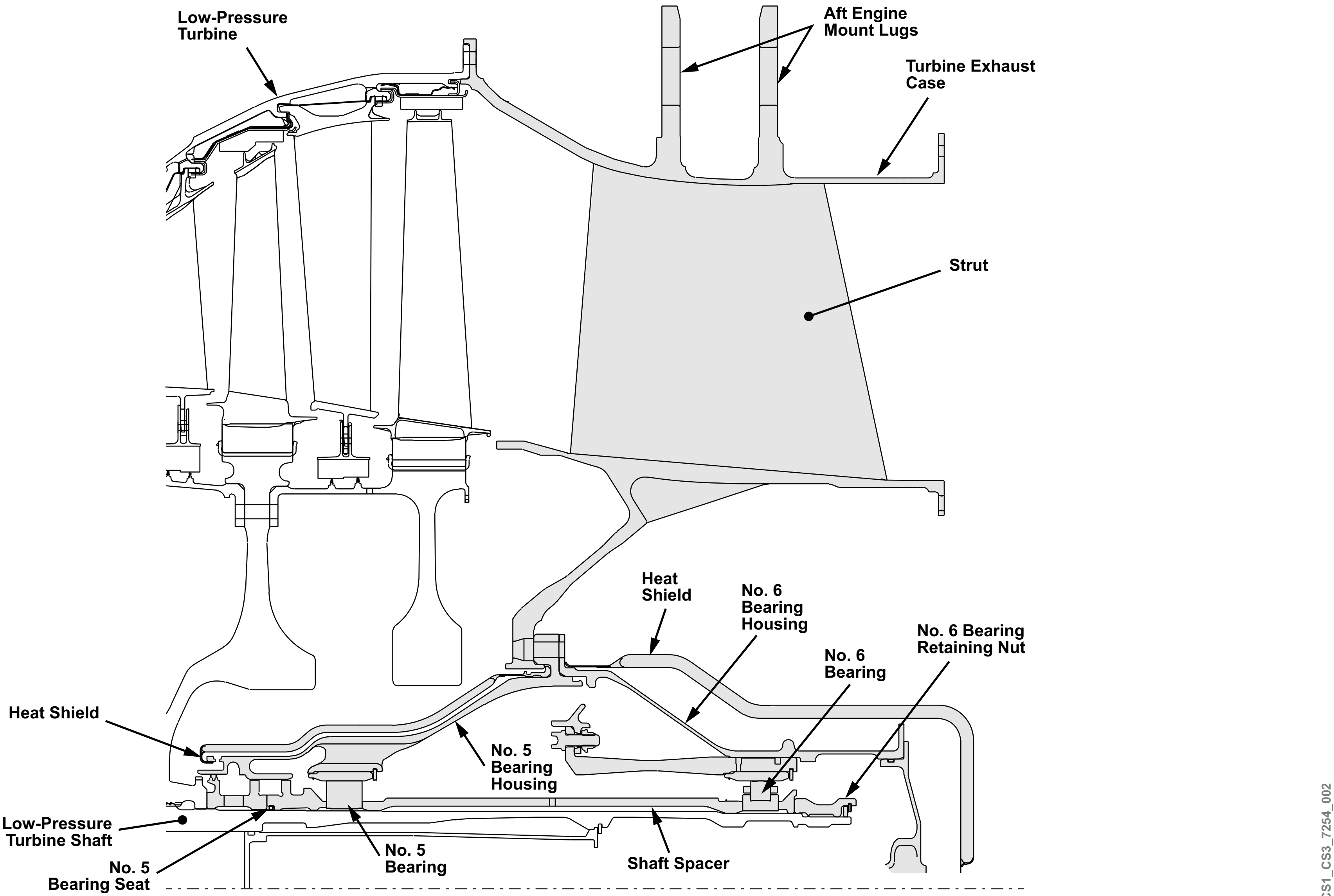


Figure 26: Turbine Exhaust Case Cross-Section

72-60 MAIN AND ANGLE GEARBOXES

GENERAL DESCRIPTION

MAIN GEARBOX

The main gearbox (MGB) housing is a cast aluminum structure mounted to the engine core. Accessories are mounted on the forward and aft faces of the gearbox.

The gearbox housing has the following components:

- Replaceable carbon seals
- De-oiler
- N2 crank pad port
- Core drain plugs
- Permanent magnet alternator generator (PMAG) drive

The MGB uses gears and shafts to drive accessories mounted to it. The angle gearbox layshaft drive is the rotational input to the MGB. The N2 speed is reduced before reaching the angle gearbox in order to have the MGB input drive in the proper rotational speed range.

The gearbox aft face is axially supported by a rod link assembly that connects to the turbine intermediate case (TIC).

A centering puck is installed on the gearbox top surface to help align the gearbox to the engine when it is installed.

Two mounting links are used to mount the gearbox to the diffuser case.

A heat shield protects the gearbox from engine core radiating heat. It is clipped at the front side and bolted to brackets at the aft end top surface of the gearbox.

The angle gearbox mechanically connects the high-pressure compressor (N2) and the main gearbox through the angle gearbox layshaft.

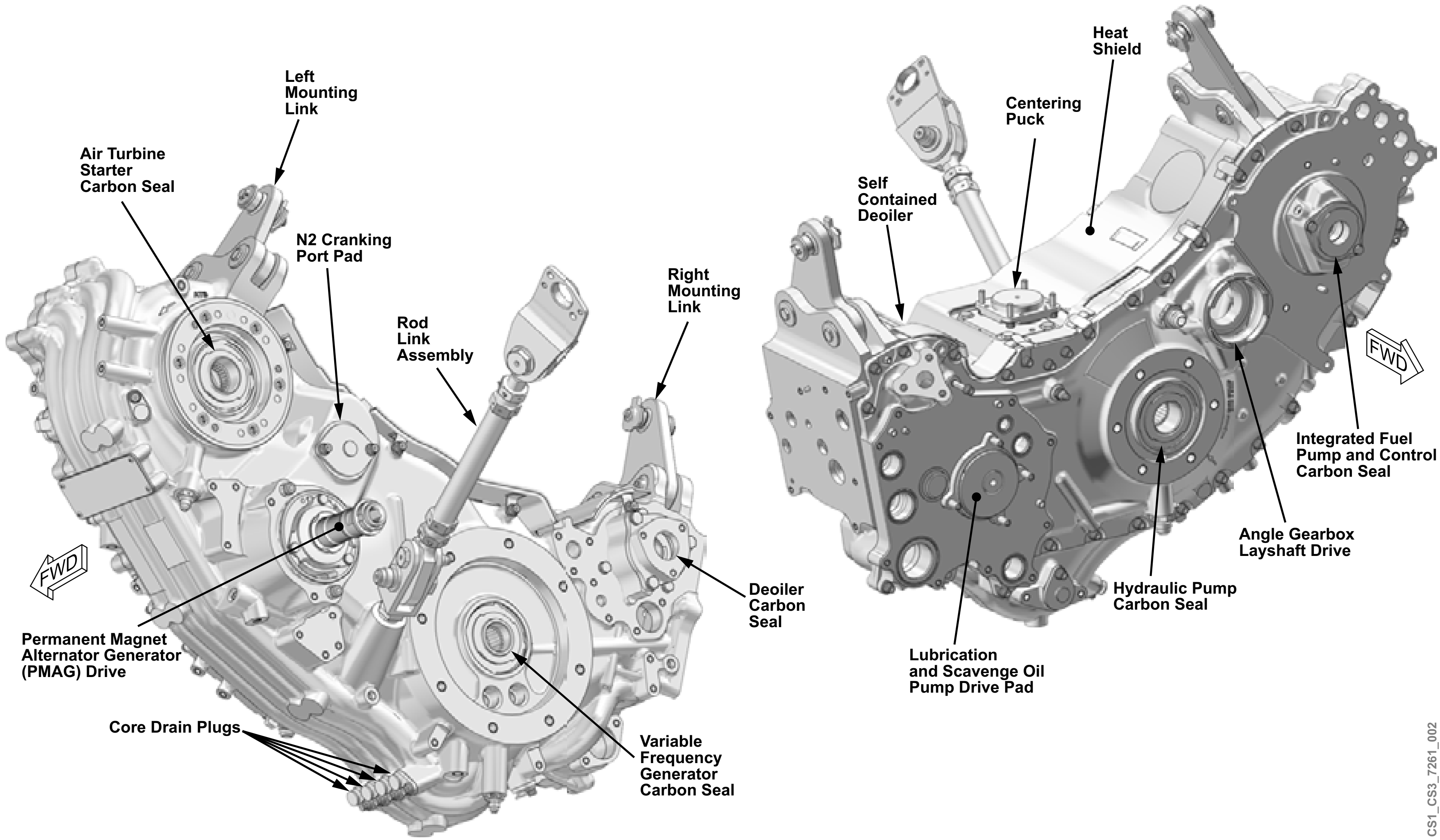


Figure 27: Main Gearbox

MAIN GEARBOX MOUNTED COMPONENTS

The following components are mounted on the main gearbox:

- Hydraulic pump
- Lubrication and scavenge oil pump (LSOP)
- Oil control module (OCM)
- Variable frequency generator (VFG)
- Permanent magnet alternator generator (PMAG)
- Air turbine starter
- Fuel oil manifold
- Integrated fuel pump and control (IFPC)

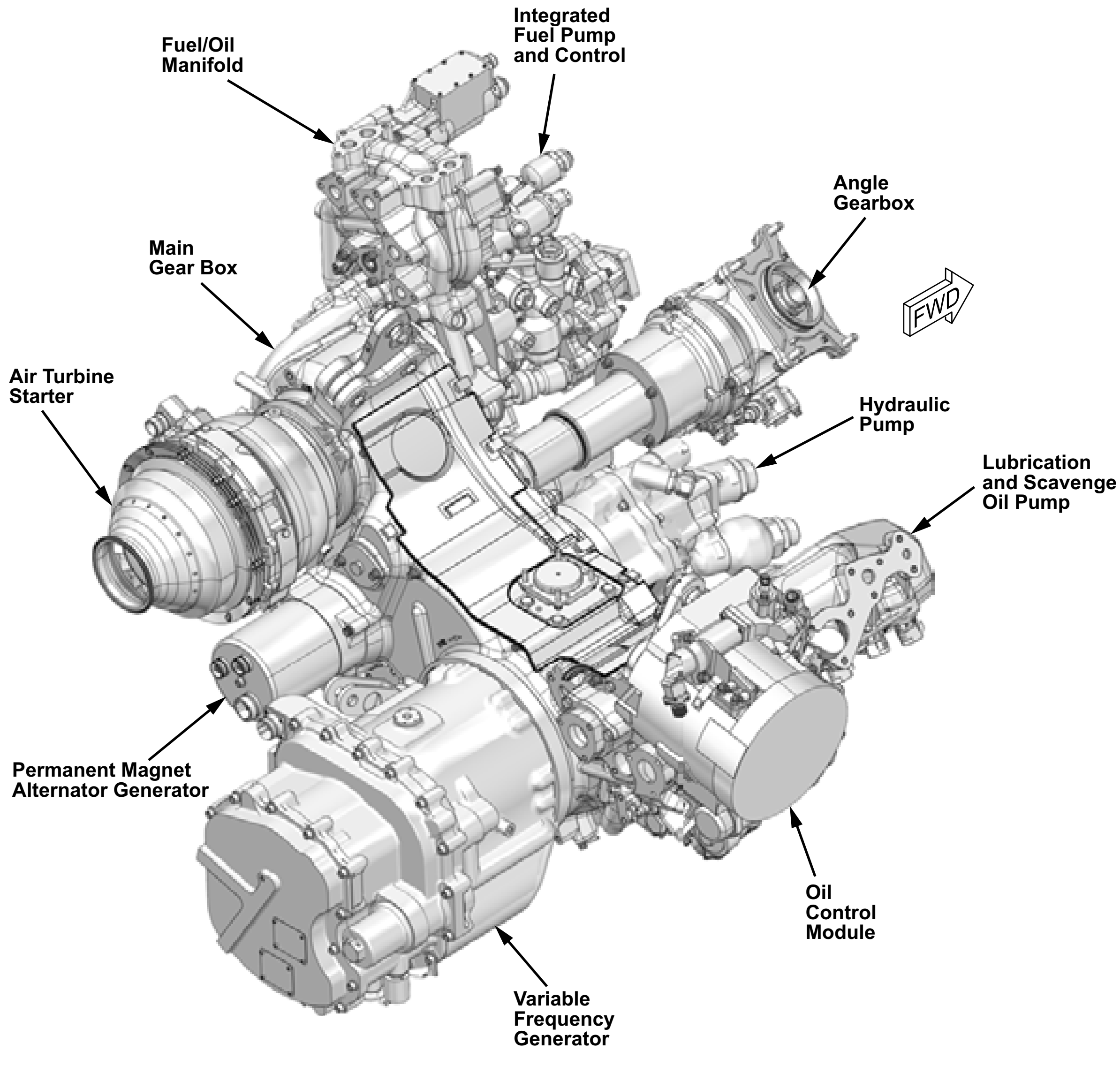


Figure 28: Main Gearbox Mounted Components

ANGLE GEARBOX

The angle gearbox mechanically connects the HPC (N2) and the main gearbox (MGB) through the layshaft. The angle gearbox is installed at the 8 o'clock position on the compressor intermediate case (CIC).

The shaft is covered by two layshaft covers. The forward layshaft cover front end is secured to the angle gearbox, and the rear cover aft end is secured to the MGB. Both layshaft covers slide one over the other at the mating center area to allow for thermal growth during engine operation.

When the engine is being started, the air starter drives the MGB, which is connected to the angle gearbox. The angle gearbox drives the HPC. When the engine is running, the HPC drives the MGB through the angle gearbox.

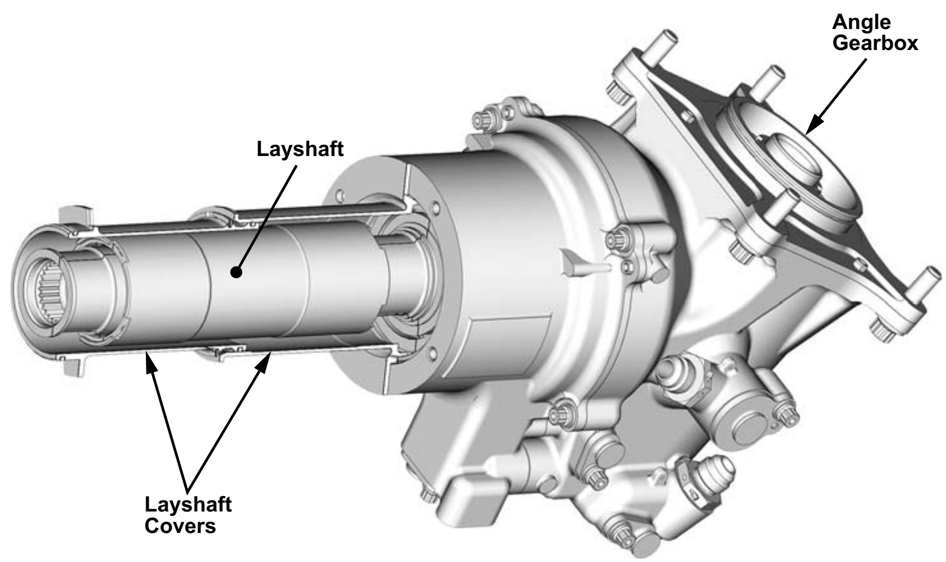


Figure 29: Angle Gearbox

72-00 BORESCOPE ACCESS

GENERAL DESCRIPTION

The borescope permits visual inspection of internal gaspath parts without the need for disassembly.

A borescope probe is inserted in one of several access ports to inspect parts for damage, cracks, wear and missing material.

Some damage to blades or stators can be blended in-situ using borescope blade blending. Consult the Aircraft Maintenance Publication (AMP) for airfoil damage and blend limitations.

The following tables show the borescope access ports and inspection areas.

ENGINE LEFT SIDE BORESCOPE ACCESS

Table 1: Left Side Borescope Access Ports and Inspection Areas

PORT	LOCATION	INSPECTION AREA
AP-LPC 3	9:30 position	LPC 2nd stage stator vanes and 3rd stage IBR blades.
AP-HPT 1	9 o'clock position	HPT 1st stage rotor blades, 2nd stage vanes, and 2nd stage rotor blades.
AP-LPT 1	8:30 position	LPT 1st stage rotor blades, LPT 1st stage vanes, and LPT 2nd stage rotor blades.
AP-LPT 2	8:30 position	LPT 2nd stage rotor blades, LPT 2nd stage vanes, and LPT 3rd stage rotor blades.
AP-DIFF 1	11 o'clock position	Combustion chamber inner and outer liners, bulkhead, fuel nozzles, HPT 1st stage rotor blades, and HPT 1st stage vanes.

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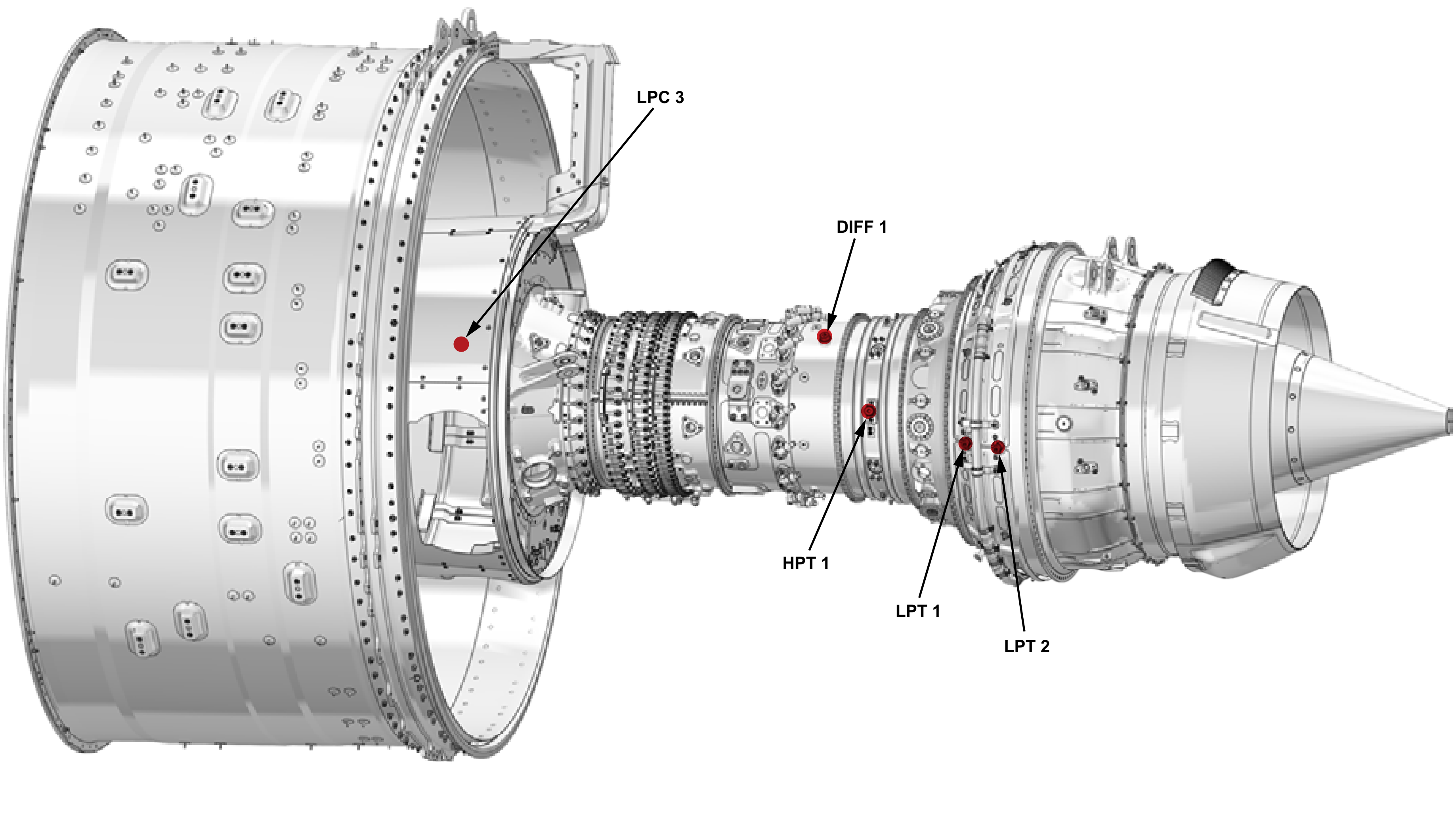

Figure 30: Engine Left Side Borescope Access

ENGINE RIGHT SIDE BORESCOPE ACCESS

Table 2: Right Side Borescope Access Ports and Inspection Areas

PORT	LOCATION	INSPECTION AREA
AP-LPC 1	3:30 position	LPC variable inlet stator vanes 1st stage integral bladed rotor (IBR) airfoils.
AP-LPC 2	4 o’clock position	LPC 1st stage stator vanes and 2nd stage IBR airfoils.
AP-HPC 1	4 o’clock position	HPC variable inlet guide vanes and 1st stage IBR airfoils.
AP-HPC 2	4 o’clock position	HPC 1st stage variable stator vanes (VSV) and 2nd stage IBR airfoils.
AP-HPC 3	3:30 position	HPC 2nd stage VSV and 3rd stage IBR airfoils.
AP-HPC 4	4 o’clock position	HPC 3rd stage VSV and 4th stage IBR airfoils.
AP-HPC 5	4.o’clock position	HPC4th stage stator and 5th stage IBR airfoils.
AP-HPC 6	4 o’clock position	HPC 5th stage stator and 6th stage IBR airfoils.
AP-HPC 7	2 o’clock position	HPC 6th stage stator and 7th stage IBR airfoils.
AP-HPC 8	2 o’clock position	HPC 7th stage stator and 8th stage rotor blade.
Igniter Plug Ports	3:30 and 4 o’clock positions	Combustion chamber inner and outer liner, bulkhead, fuel nozzles, HPT 1st stage rotor blades and vanes.
AP-TIC 1	1:30 position	HPT 2nd stage rotor blades. LPT 1st stage rotor blades.

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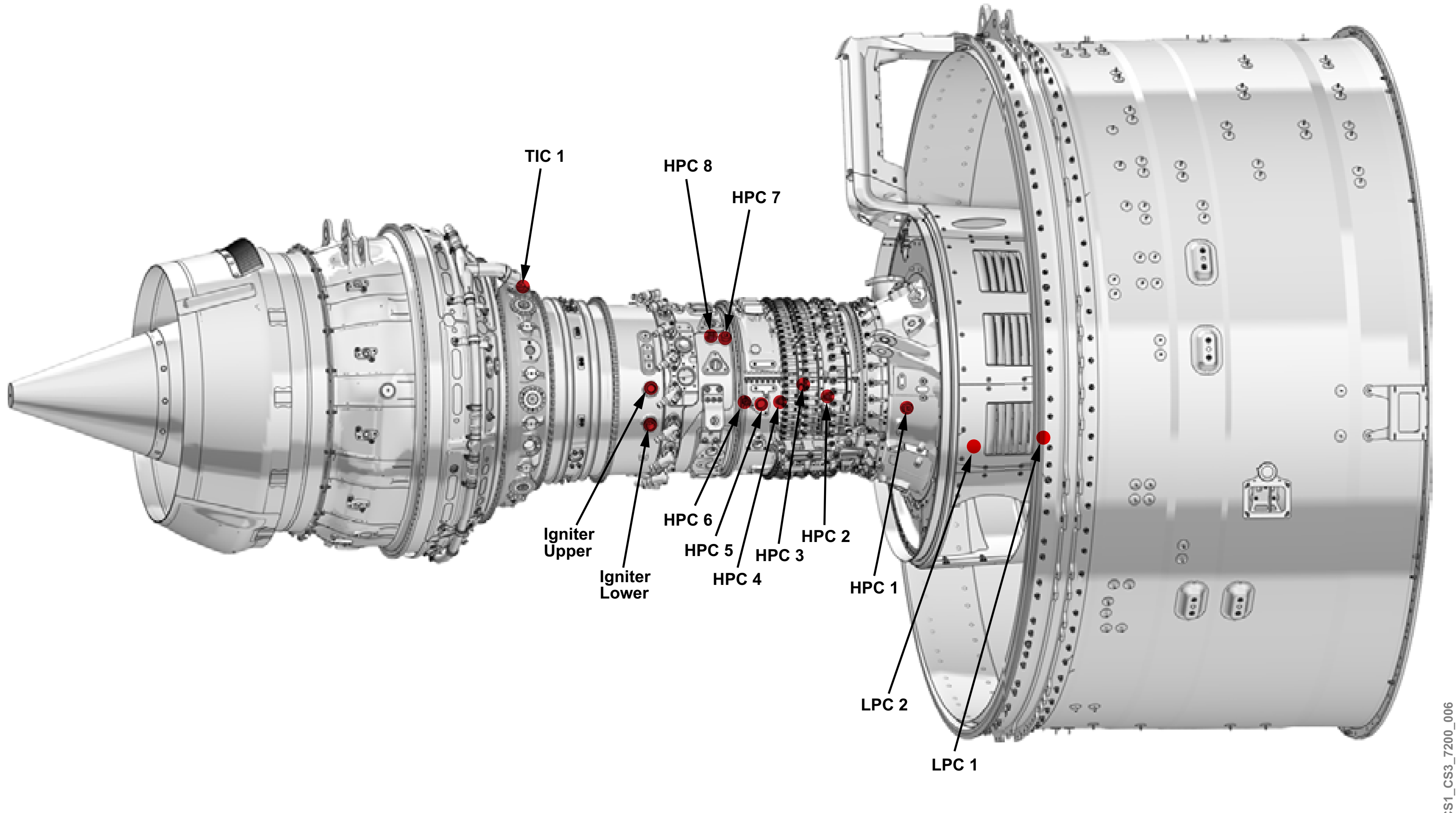

Figure 31: Engine Right Side Borescope Access

BORESCOPE ACCESS - MAIN GEARBOX

The gearbox also has two access ports to facilitate inspection.

An N2 crank pad is used to turn the N2 rotor during the borescope procedure.

BORESCOPE ACCESS - ANGLE GEARBOX

The angle gearbox has one borescope access port for internal inspection.

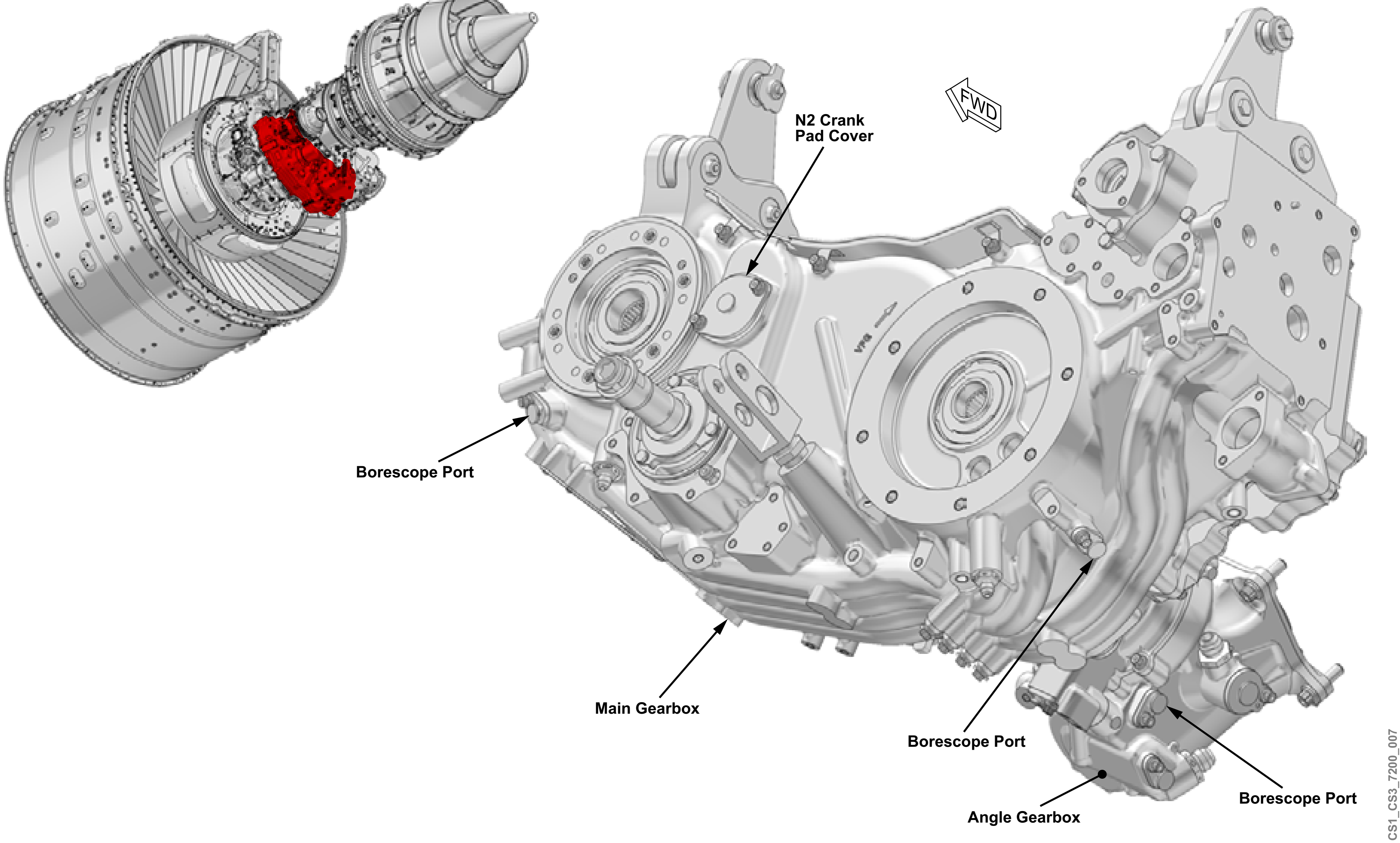

Figure 32: Borescope Access - Gearbox and Angle Gearbox

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ATA 73 - Engine Fuel and Control

BD-500-1A10
BD-500-1A11

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ENGINE FUEL AND CONTROL - CHAPTER BREAKDOWN

73-11 DISTRIBUTION

GENERAL DESCRIPTION

The engine fuel distribution system supplies fuel to the engine for combustion and for air system actuator operation. The aircraft fuel system supplies fuel to the integrated fuel pump and control (IFPC).

A fuel oil manifold provides mounting for the IFPC, fuel flow transmitter and the fuel filter. The manifold simplifies maintenance by minimizing the number of external fuel lines needed to connect the components.

The IFPC has integral pumps driven by the engine gearbox:

- Centrifugal, low-pressure boost pump
- High-pressure main gear pump to provide pressurized fuel
- Motive flow gear pump

Fuel from the aircraft is supplied to the IFPC boost pump under pressure. Should the aircraft fuel system pumps fail, the boost pump is capable of suctioning fuel from the aircraft fuel tanks. The boost pump provides fuel at low-pressure to the motive flow pump and to a fuel/oil heat exchanger (FOHX).

The motive flow pump provides fuel to operate the aircraft fuel system ejector pumps.

The FOHX heats engine fuel to prevent icing and cools the engine oil for proper engine bearing seal operation. The heated fuel then passes through the fuel filter, which is installed in a housing attached to the fuel manifold.

The filtered fuel then flows to the main pump where it is pressurized. Some of the pressurized fuel is supplied to the air system actuators for engine cooling and stability control. The main pump also supplies the fuel to the fuel metering valve (FMV). The FMV is controlled by the electronic engine control (EEC) to provide the correct fuel for a given thrust setting. The metered fuel flow is measured by the fuel flow meter. The fuel is then sent to the flow divider.

The flow divider valve (FDV) allows fuel to flow to the primary fuel nozzles for engine start. As the engine accelerates to idle, the flow divider valve provides flow to the secondary fuel nozzles.

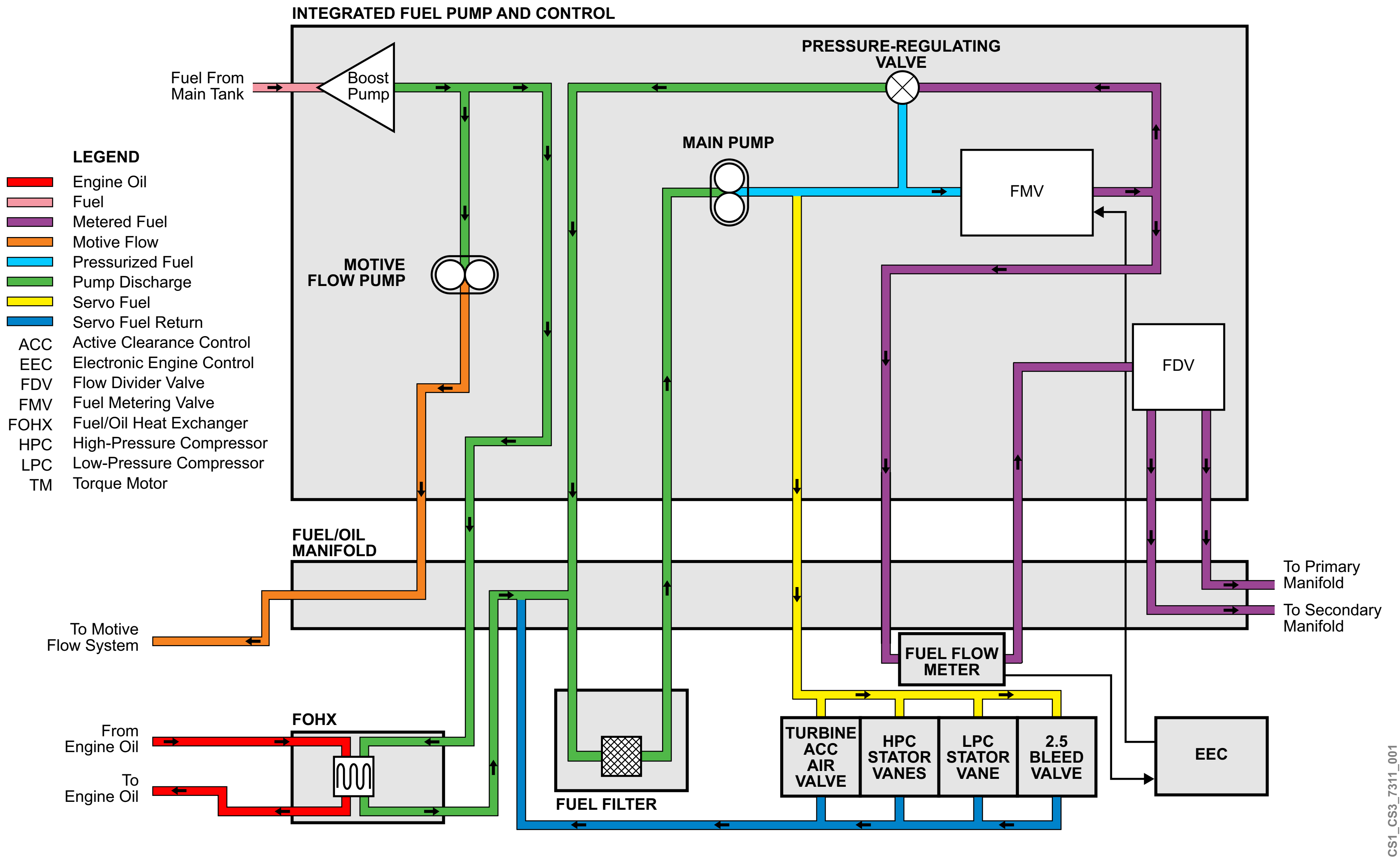


Figure 1: Fuel Distribution System General Description

COMPONENT LOCATION

The engine fuel distribution system consists of the following components:

- Integrated fuel pump and control
- Fuel/oil heat exchanger
- Fuel/oil heat exchanger bypass valve
- Fuel filter
- Fuel nozzles
- Fuel supply manifolds
- Fuel/oil manifold
- Fuel flow meter

Integrated Fuel Pump and Control

The integrated fuel pump and control (IFPC) is mounted on the fuel/oil manifold on the front face of the main gearbox at the 8 o’clock position.

Fuel/Oil Heat Exchanger and Fuel/Oil Heat Exchanger Bypass Valve

The fuel/oil heat exchanger (FOHX) and fuel/oil heat exchanger bypass valve are located at the 12 o’clock position on the compressor intermediate case.

Fuel Filter

The fuel filter is located below the IFPC on the fuel/oil manifold.

Fuel Nozzles and Primary and Secondary Fuel Manifolds

The fuel nozzles and primary and secondary fuel manifolds are located on the circumference of the combustor.

Fuel/Oil Manifold

The fuel/oil manifold is located on the front face of the gearbox on the left side of the engine.

Fuel Flow Meter

The fuel flow meter is attached to the fuel oil manifold on the front face of the main gearbox at the 9 o’clock position.

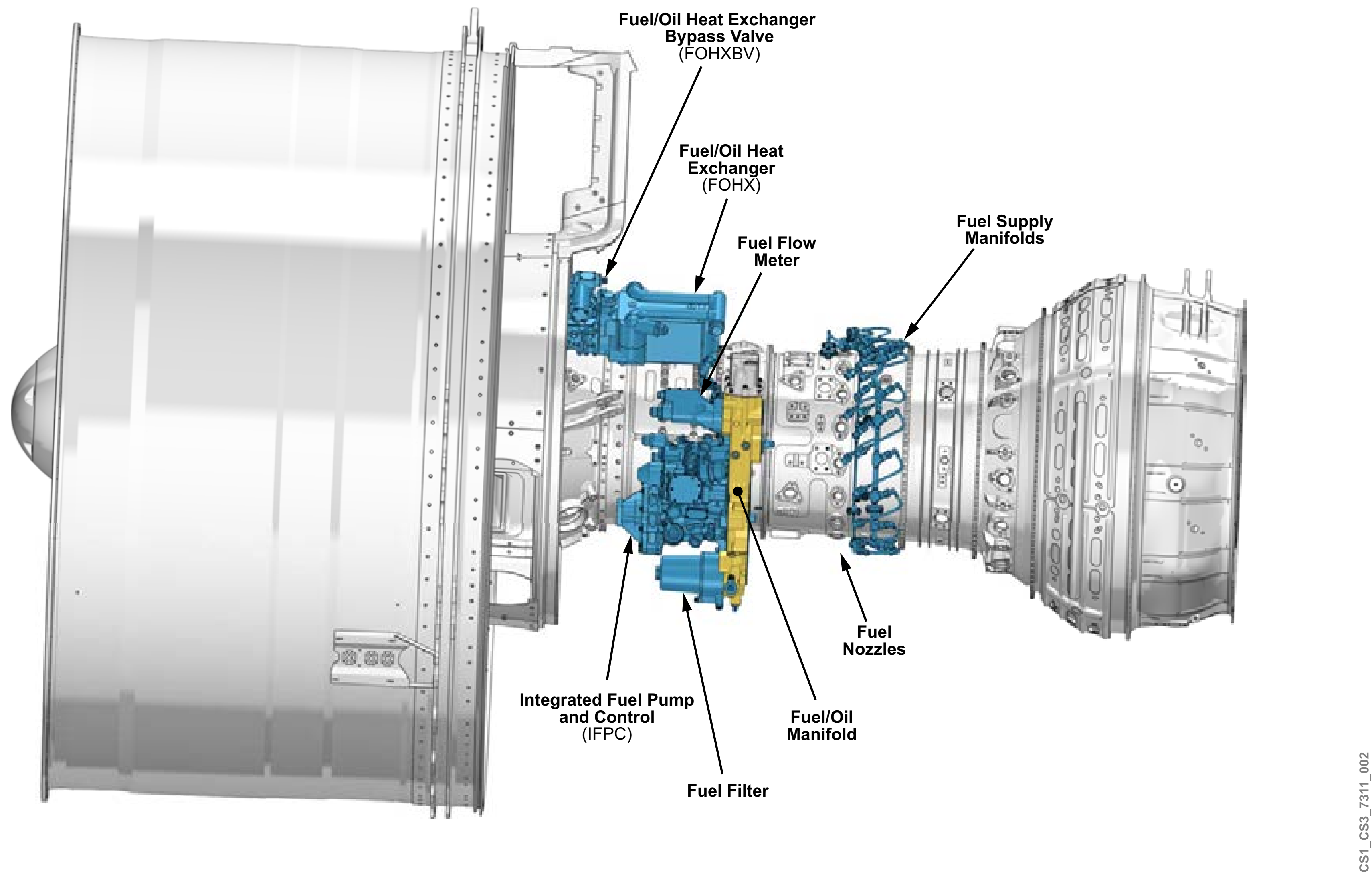


Figure 2: Fuel Distribution System Component Location

COMPONENT INFORMATION

INTEGRATED FUEL PUMP AND CONTROL

The IFPC supplies metered fuel based on inputs from the EEC in response to changes in thrust demand and environmental conditions.

The IFPC performs the following major functions:

- Supply motive fuel for aircraft fuel system
- Meter fuel for combustion
- Pressurize fuel supply
- Fuel shutoff for engine shutdown
- Control primary and secondary flow distribution to fuel nozzles
- Supply pressurized servo fuel for air system actuators
- Bypass excess fuel flow

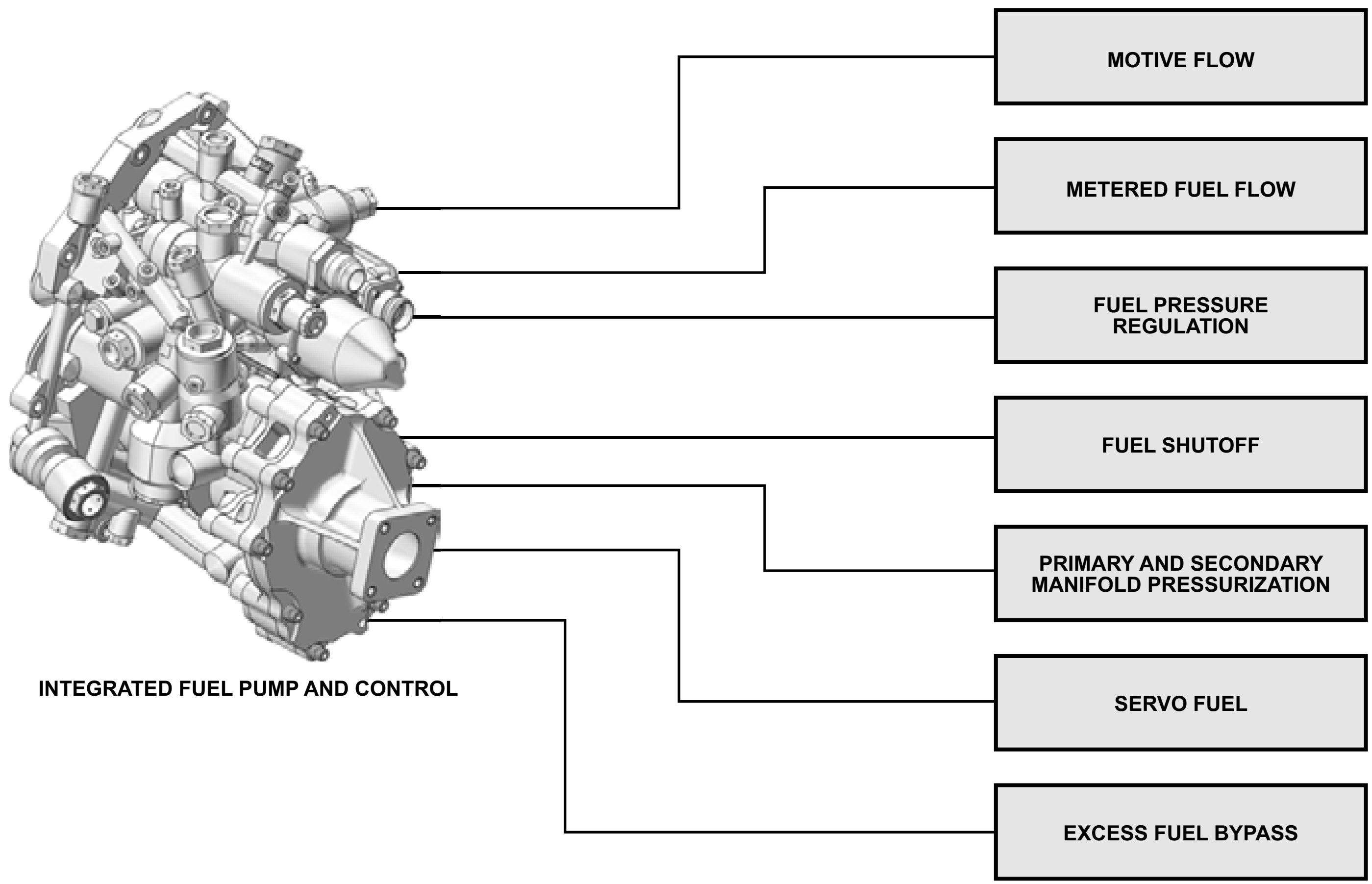


Figure 3: Integrated Fuel Pump and Control

FUEL FILTER

The main fuel filter element is used to remove contaminants from the fuel sent from the IFPC. The canister type disposable fuel filter is installed in a housing on the fuel oil manifold below the IFPC.

The fuel filter bypass valve opens when the filter clogs and the differential pressure exceeds 25 psid.

The filter housing is equipped with a drain plug for draining the housing prior to removing the filter.

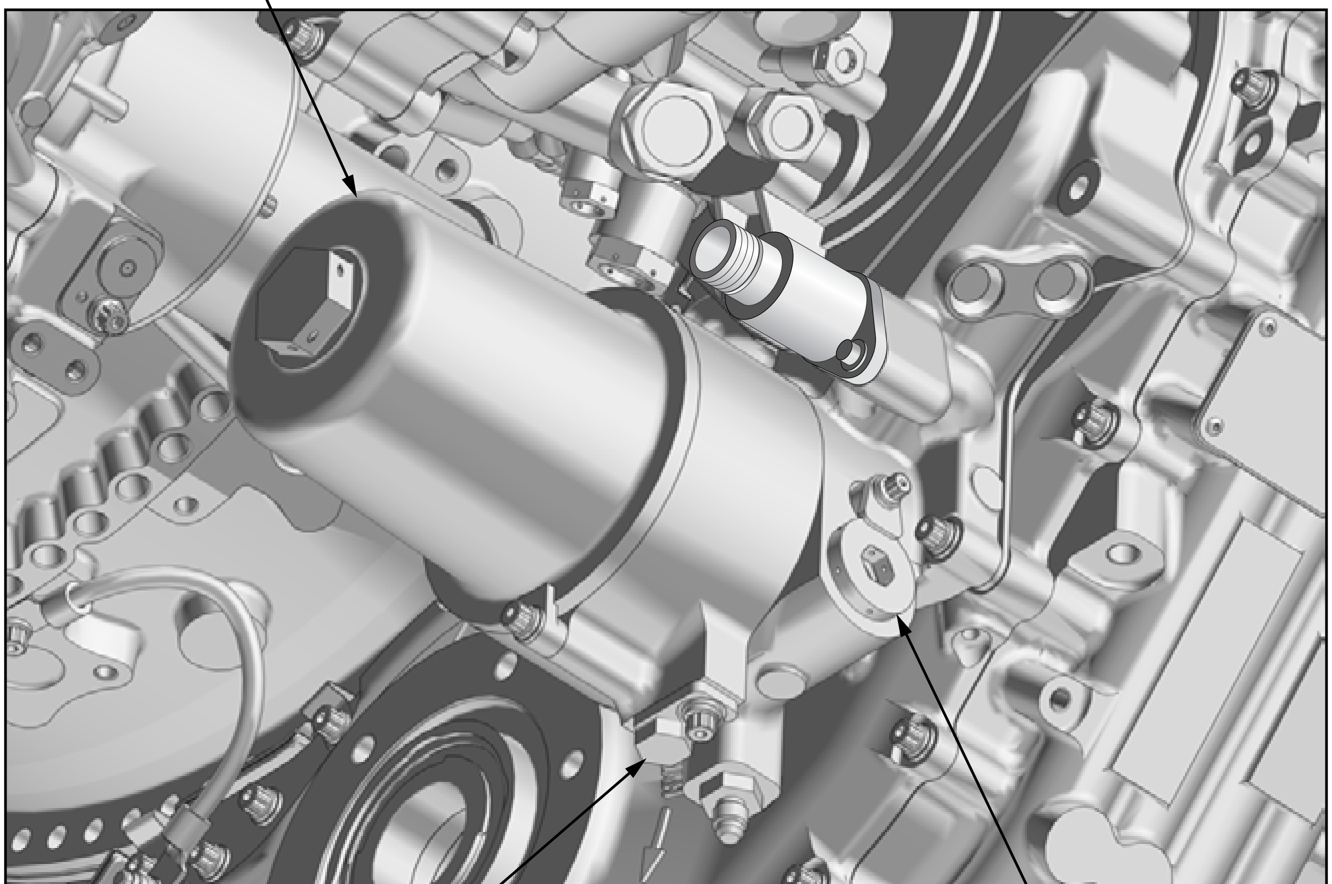


Figure 4: Fuel Filter

FUEL NOZZLES

There are 16 fuel nozzles, 10 duplex, and 6 simplex. They are arranged in a pattern of 4 duplex, 4 simplex, 6 duplex, and 2 simplex nozzles. The IFPC and fuel/oil manifold send fuel to the primary and secondary fuel nozzle manifolds that supply the fuel nozzles.

The duplex nozzles contain both primary fuel nozzles and secondary fuel nozzles. The simplex nozzles only contain secondary fuel nozzles.

All of the fuel nozzles contain check valves to prevent fuel in the primary and secondary fuel nozzle manifolds from draining into the engine following an engine shutdown. These valves in the primary and secondary circuits of the fuel nozzles shut when the fuel system pressure drops below a minimum threshold during shutdown conditions, preventing fuel from draining into the combustor.

During an engine start, the IFPC delivers fuel to the primary manifold to provide combustion for initial spool up of the engine. During engine acceleration to ground idle, the FDV in the IFPC distributes fuel flow to both the primary and secondary nozzles.

An alignment pin on each nozzle prevents incorrect installation.

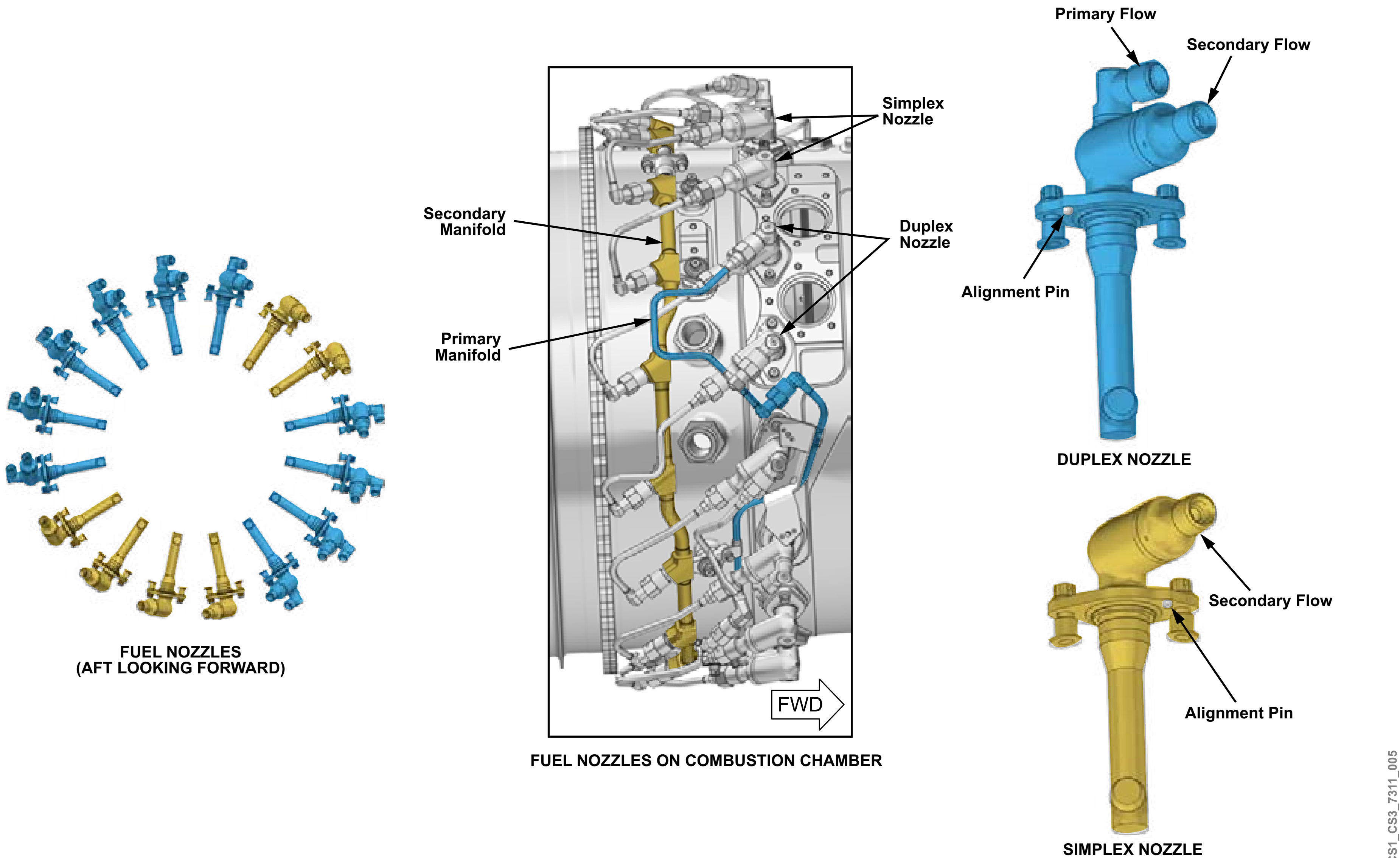


Figure 5: Fuel Nozzles

DETAILED DESCRIPTION

MOTIVE FLOW FUEL

Motive flow for the aircraft fuel system is provided by a motive flow gear pump downstream of the centrifugal boost pump exit. The inlet of the motive flow pump is protected from fuel contamination by a screen with a bypass valve. The outlet of the motive flow pump is protected from overpressure by a high-pressure relief valve.

SERVO FUEL

The servo pressure regulator adjusts the pressure to a suitable level for actuator operation. Servo fuel is supplied to the turbine active clearance control (ACC) valve, high-pressure compressor (HPC) stator vane actuator, low-pressure compressor (LPC) stator vane actuator, and the 2.5 bleed valve actuator. The actuators are bi-directional fully modulated linear actuators used to optimize overall engine performance and to prevent engine compressor surges. The servo fuel is then returned from the actuators upstream of the fuel filter before being recirculated in the IFPC.

FUEL FILTER

The fuel filter differential pressure sensor measures the pressure drop across the fuel filter. If the fuel filter is completely clogged, a bypass valve opens to divert fuel around the fuel filter. The EEC monitors the fuel filter and provides an L(R) ENG FUEL FILTER advisory message when the fuel filter clogs. An accompanying INFO message indicates a filter clogging or an actual bypass condition.

METERED FUEL

The pressure-regulating valve (PRV) modulates to provide a constant fuel flow to the FMV regardless of the FMV position. The EEC computes the required metered fuel using the fuel temperature sensor for compensation. In the case of a fuel temperature sensor failure, the EEC uses a default fuel temperature of 75°C (167°F).

The metered fuel output of the FMV then flows to the fuel flow meter as long as the windmill bypass valve (WBV) remains closed. The fuel flow meter sends a flow signal to the EEC for fuel flow indication.

Fuel is supplied to the minimum pressure-shutoff valve (MPSOV) and flow divider valve (FDV) for distribution to the fuel nozzles. When the engine is shutdown, the WBV opens and connects fuel from upstream of the minimum pressure-shutoff valve (MPSOV) with the fuel filter inlet. The MPSOV closes and fuel is shut off from the engine.

FUEL/OIL HEAT EXCHANGER OPERATION

The fuel/oil heat exchanger maintains the fuel temperature above 0°C (32°F) at the fuel filter inlet during steady-state operation by transferring engine oil heat to the fuel. The EEC controls the fuel/oil heat exchanger bypass valve (FOHXBV) based on the fuel temperature sensor signal. As the temperature decreases, the EEC opens the FOHXBV to supply more hot oil to warm the fuel. If the fuel side of the heat exchanger is blocked, a bypass valve opens to divert fuel around the heat exchanger.

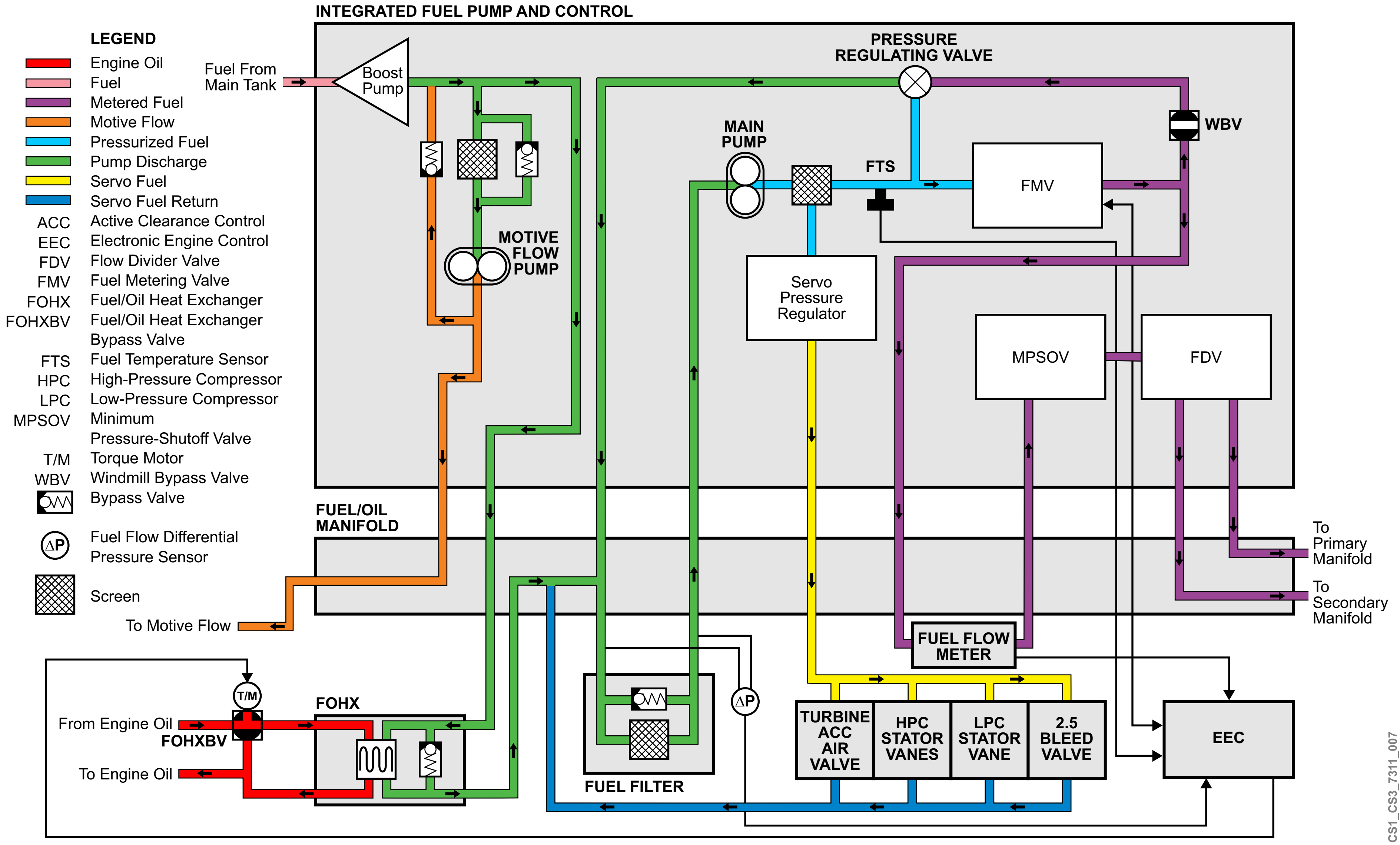


Figure 6: Fuel Distribution System Detailed Description

73-20 CONTROL SYSTEM

GENERAL DESCRIPTION

The engine is controlled by a full authority digital engine control (FADEC). A dual-channel electronic engine control (EEC) controls the engine thrust, using N1 as a reference, in accordance with the position of the throttle quadrant assembly (TQA) thrust lever, environmental conditions, and the amount of bleed air being extracted from the engine. This is done by the EEC using the data received from both aircraft systems and engine sensors, together with the engine trim data inputs from the data storage unit (DSU).

The FADEC consists of three main components:

- Electronic engine control (EEC)
- Prognostics and health management unit (PHMU)
- Data storage unit (DSU)

The data storage unit (DSU) is installed on the EEC and used for storage of engine configuration and identification data, including information related to the engine's specific serial number.

The prognostics and health management unit (PHMU) is a single channel line replaceable unit (LRU) that monitors engine parameters to provide the engine health status, vibration, auxiliary oil pressure sensing, and oil debris monitoring.

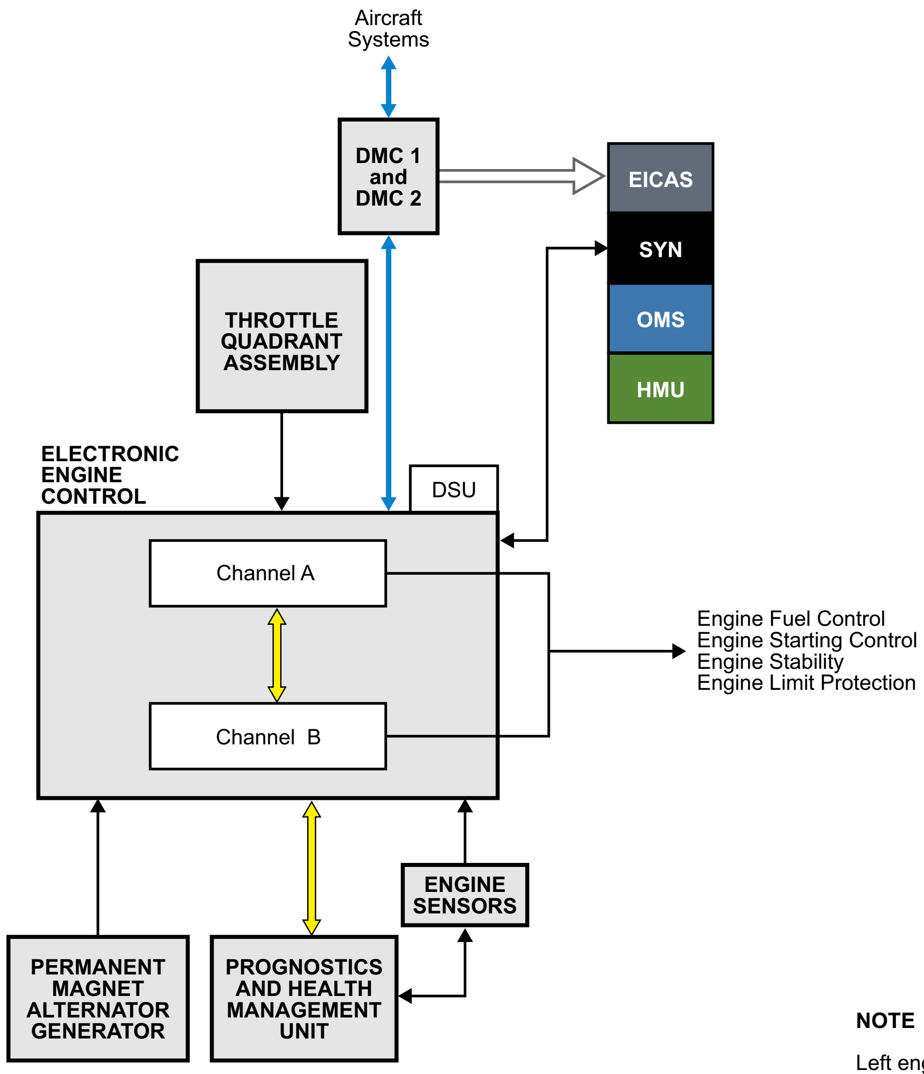
The software in the EEC provides the logic required for engine control, fault detection and accommodation. The engine indication and crew alerting system (EICAS) display parameters, and engine diagnostics and maintenance data to the onboard maintenance system (OMS). The EEC also provides health monitoring information from the PHMU to the health management unit (HMU). All of the FADEC components have field-loadable software.

A gearbox-driven permanent magnet alternator generator (PMAG) provides electrical power to the EEC during normal operation.

The EEC channel A and channel B, each have their own processor, power supply, program memory, and inputs and outputs. Either EEC channel is capable of controlling all of the critical engine functions:

- Engine fuel control
- Engine starting control
- Engine stability
- Engine limit control

During normal operation, one channel is in control and the other channel is in standby. Both channels calculate and provide the same commands, however, only the outputs from the channel in control are active. In case of an output failure on the channel in control, the equivalent output from the standby channel is used and controlled by the standby channel.



LEGEND

DMC Data Concentrator Unit Module Cabinet
DSU Data Storage Unit
ARINC 429
CAN BUS

NOTE

Left engine shown. Right engine similar.

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Figure 7: Engine Control System General Description

ELECTRONIC ENGINE CONTROL

The EEC channels are normally powered by the PMAG when engines are running but are powered by ESSENTIAL DC BUSES when the engines and the PMAG fails or the engine is shutdown. The left EEC channel A is powered by DC ESS BUS 3 and channel B is powered by DC ESS BUS 1. The right EEC channel A is powered by DC ESS BUS 3, and channel B is powered by DC ESS BUS 2.

Fuel Control

The EEC schedules engine thrust in response to thrust lever angle from the throttle quadrant assembly (TQA), aircraft inputs from the integrated air management system (IAMS), flight management system (FMS) and engine sensor inputs. Engine N1 and N2 limits are protected at all times and the exhaust gas temperature (EGT) is limited at start. The main input for fuel flow scheduling is thrust lever angle from the TQA.

Additional parameters the EEC uses to calculate fuel flow are ambient pressure (Pambient), inlet air temperature (T2), and pressure (P2), burner pressure (Pb), exhaust gas temperature (EGT), N1 speed and N2 speed.

Starting and Ignition System

The control system provides automatic starting sequencing of the starter air valve for the air turbine starter, igniters, and hydraulic pump depressurization during the starting phase.

Engine Airflow Control System

The EEC controls bleed valves, and the low-pressure compressor (LPC) and high-pressure compressor (HPC) stator vane system for engine stability and operability. The EEC also controls the turbine active clearance control systems for improved engine efficiency.

Cowl Anti-Ice

The cowl anti-ice (CAI) valves are controlled by the EEC to provide anti-icing for the inlet cowl. They are also opened to improve HPC stability during engine start and at low engine power settings.

Air Systems

The engine interfaces with the IAMS to provide the correct bleed demand based on air conditioning and wing anti-ice demands.

Thrust Reverser

The EEC controls the deploy and stow functions of the thrust reverser actuation system (TRAS). The EEC control is based on thrust lever position from the TQA when the aircraft is on the ground.

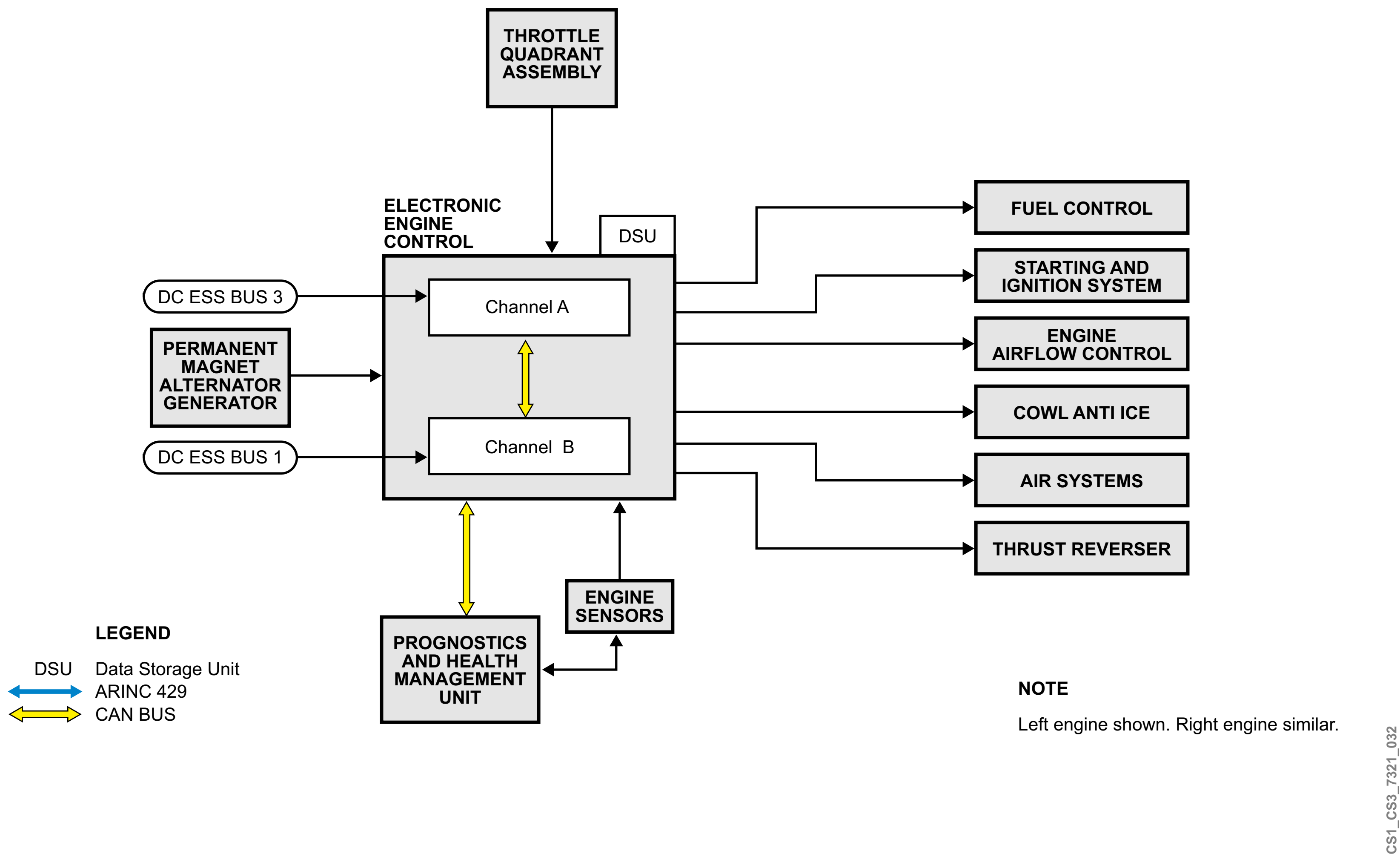


Figure 8: Electronic Engine Control

FUEL CONTROL DESCRIPTION

Each engine is controlled by a thrust lever on the TQA. The thrust lever controls both forward and reverse thrust. The ENG switches provide engine run and shutdown commands for normal operation. The ENGINE panel has a START switch to initiate the start sequence in manual mode.

If a fire is detected, the engine is automatically shut down when the L(R) FIRE PBA is pressed.

The EEC uses information from the air data system for engine control. The air data is validated against data received from the engine sensors P2T2 and Pamb. Engine sensor data is used in the event the air data is not available, however, engine performance is degraded.

The flight management system (FMS) provides thrust requirements based on the flight plan information entered. The FMS also provides an interface to manually control the thrust reference.

The integrated air management system supplies the bleed air requirements from the aircraft systems. The thrust requirement is adjusted to ensure that the minimum bleed air requirements are always met.

When the thrust requirements are set, the autothrottle can be engaged to control the thrust levers automatically, based on the FMS flight plan inputs.

The EEC provides fuel scheduling and engine start commands to the integrated fuel pump and control (IFPC). The IFPC provides metered fuel for combustion and servo fuel for control of the engine air systems.

The fuel scheduling and limit protection is based on the following sensors:

- N1 speed
- N2 speed
- Ambient air pressure (Pamb)
- Inlet air temperature (T2)
- Inlet pressure (P2)
- Burner pressure (Pb)
- Exhaust gas temperature (EGT)

Engine rotor speeds are calculated by the EEC using the N1 and N2 signals. The EEC schedules fuel according to the calculated N1 target and limits fuel if N1 or N2 limits are reached. If an overspeed is detected on N1 or N2, the engine is shutdown.

Burner pressure (Pb) is sent to the EEC to schedule fuel, detect stalls, and hung starts.

The exhaust gas temperature (EGT) is monitored by the EEC. During engine start, the maximum EGT is limited. The EGT is used for temperature exceedance determination, light off detection, hot start abort and engine health monitoring.

The EEC limits fuel flow when redline limits are reached for N1, N2 or burner pressure. In addition to redline limiting, if the high or low rotor speed exceeds the overspeed limit of 105% N1 or 105% N2, the EEC will command an overspeed solenoid in the integrated fuel pump and control unit (IFPC) to cut engine fuel flow.

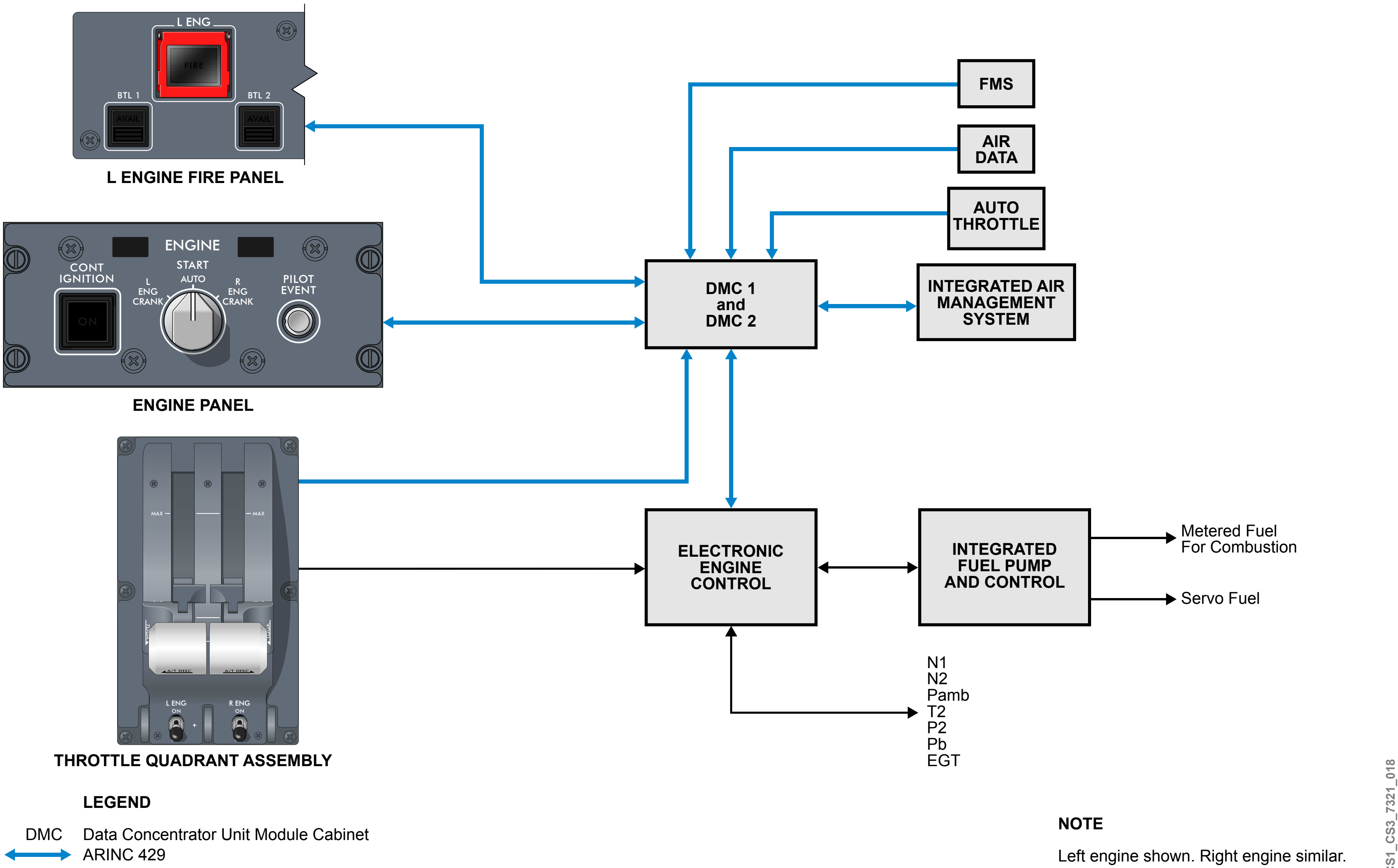


Figure 9: Fuel Control

PROGNOSTICS AND HEALTH MANAGEMENT UNIT

The prognostics and health management unit (PHMU) provides information about the health of the engine to help optimize maintenance operations.

The PHMU is a single channel system with health monitoring functions using data from the EEC and engine sensors. The PHMU also performs the following functions:

- Engine vibration monitoring
- Fan trim balance
- Oil debris monitoring
- Fan drive gear system (FDGS) auxiliary oil pressure sensor monitoring

The PHMU is powered from DC BUS 1. The PHMU communicates with the EEC through a CAN BUS. The engine health monitoring information is supplied to the HMU where is can be downloaded for analysis.

The engine vibration monitoring helps locate sources of vibration and provides information for fan trim balance. The fan trim balance is carried out through the onboard maintenance system (OMS).

The PHMU monitors the fan drive gear system (FDGS) to ensure that it receives a constant supply of oil under all flight conditions.

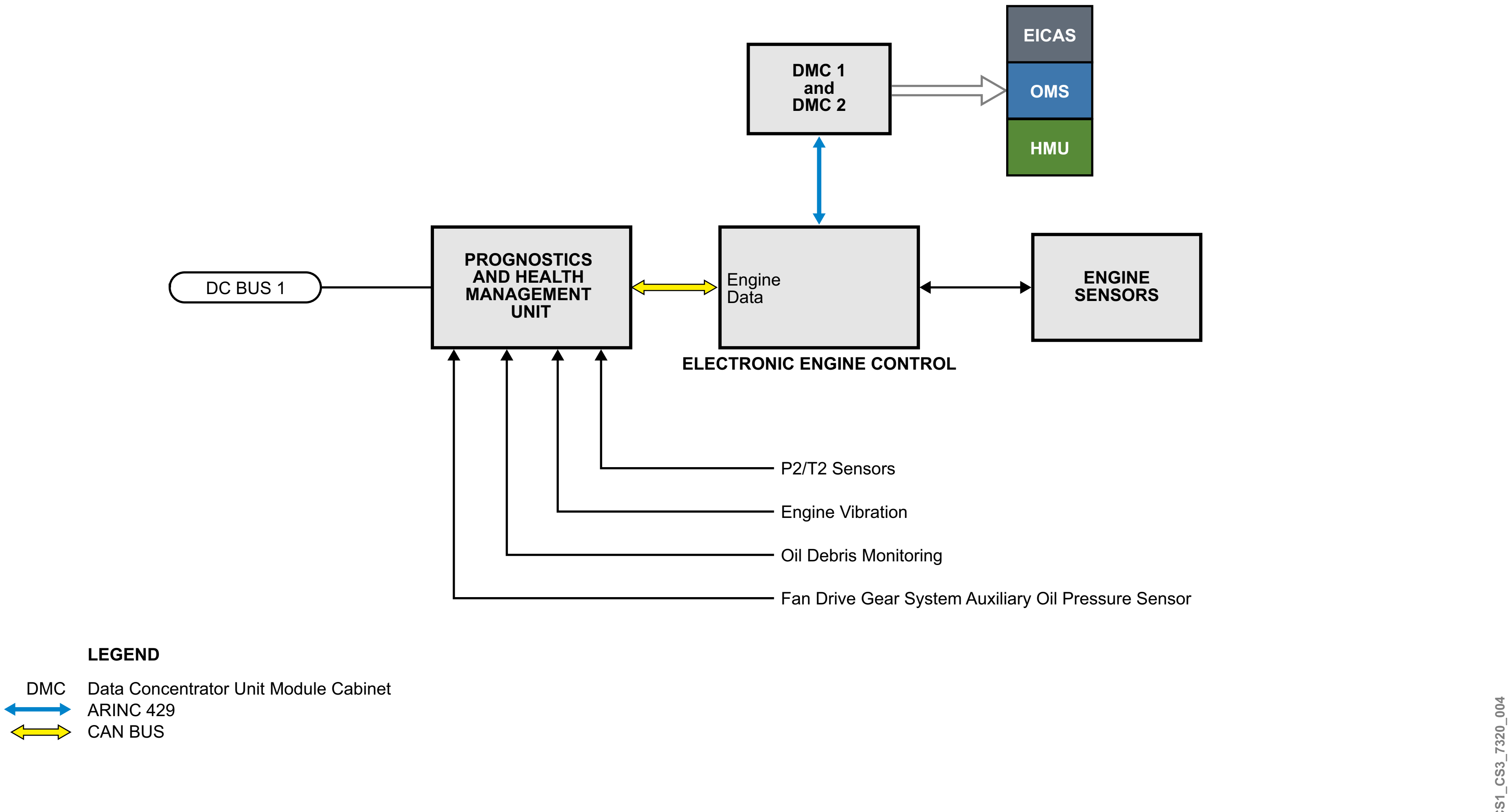


Figure 10: Prognostics and Health Management Unit

COMPONENT LOCATION

Components of the engine control system include:

- P2/T2 probe
- Burner pressure (Pb) Sensor
- Electronic engine control (EEC)
- Prognostics and health management unit (PHMU)
- Data storage unit (DSU)
- Permanent magnet alternator generator (PMAG)
- P2.5/T2.5 probe (Refer to Figure 12)
- T3 probe (Refer to Figure 12)

P2/T2 PROBE

The P2/T2 probe is mounted to the inlet cowl at the 11 o’clock position.

BURNER PRESSURE SENSOR

The Pb sensor is attached to the compressor intermediate case (CIC) fire seal at the 10 o’clock position.

ELECTRONIC ENGINE CONTROL UNIT

The EEC is located on the fan case at the 9 o’clock position.

PROGNOSTICS AND HEALTH MANAGEMENT UNIT

The PHMU is located beside the EEC on the engine fan case at the 9 o’clock position.

DATA STORAGE UNIT

The data storage unit (DSU) is located on the EEC.

PERMANENT MAGNET ALTERNATOR GENERATOR

The permanent magnet alternator generator (PMAG) is located on the main gearbox.

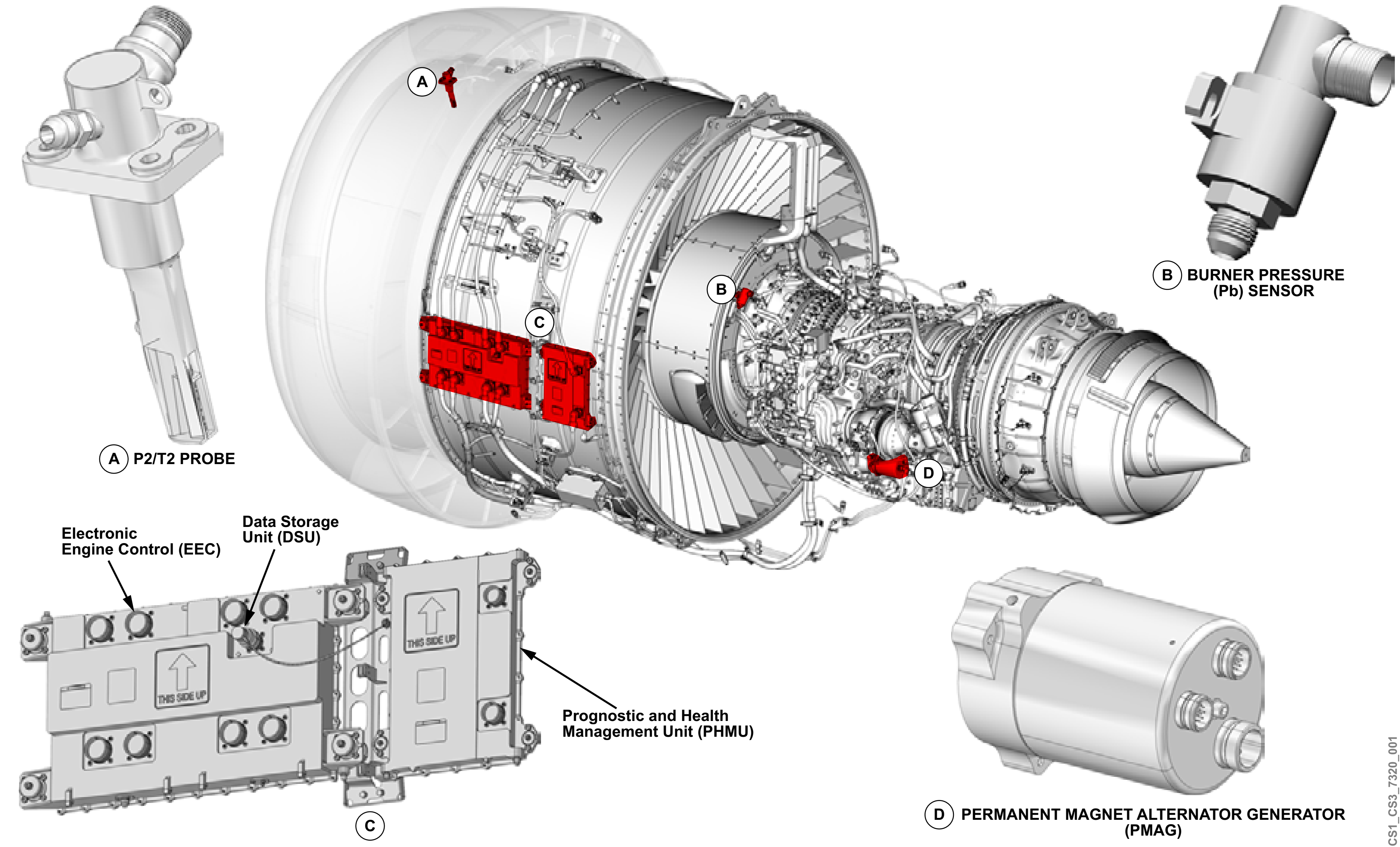


Figure 11: Engine Control System Component Location

P2.5/T2.5 PROBE

The P2.5/T2.5 probe is located on the CIC at the 1 o’clock position.

T3 PROBE

The T3 probe is located on the diffuser case, forward of the fuel nozzles, at approximately the 1 o’clock position.

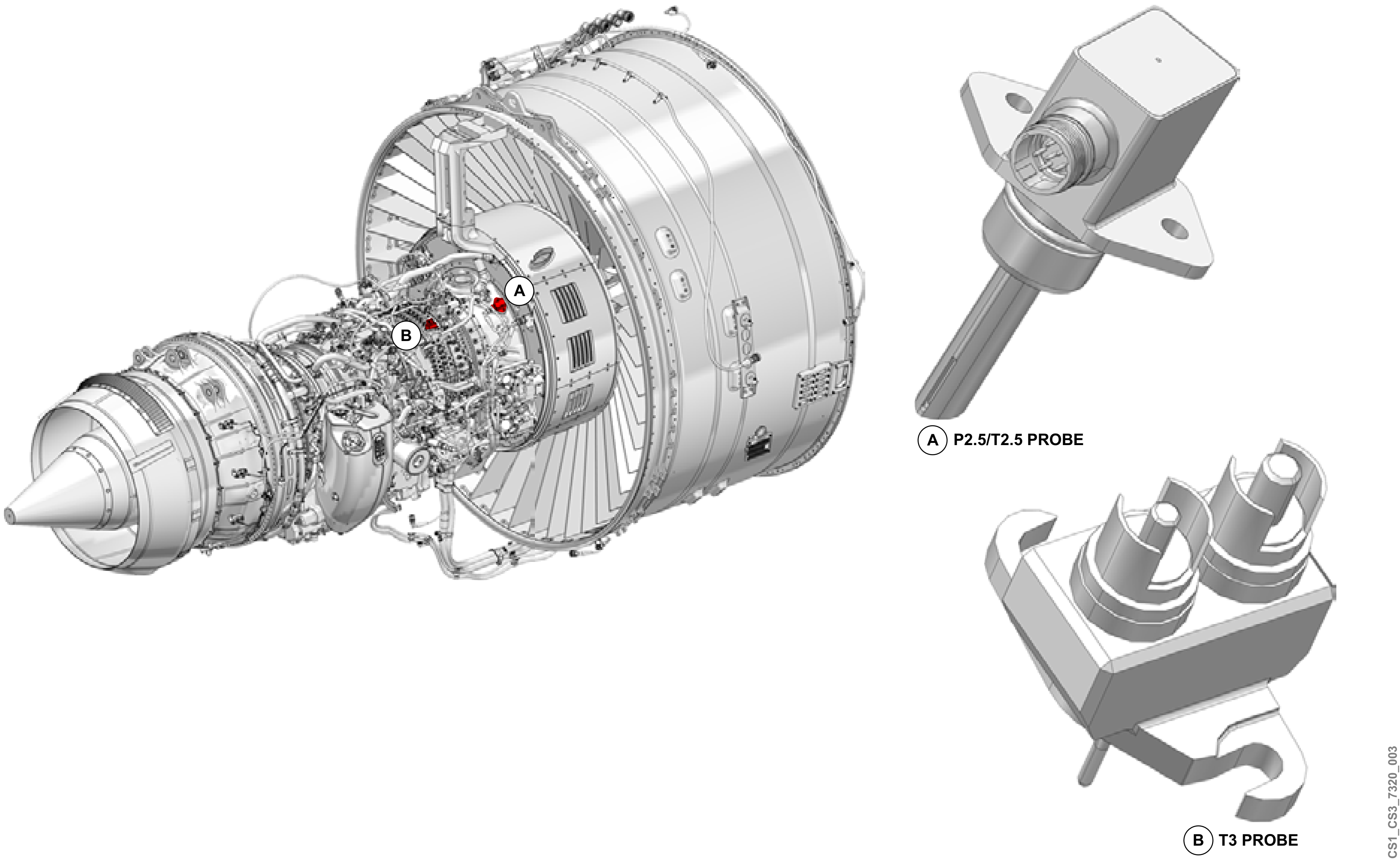


Figure 12: P2.5/T2.5 Probe and T3 Probe Location

COMPONENT INFORMATION

ELECTRONIC ENGINE CONTROL

The EEC electronic components are installed in a cast aluminum housing that is cooled by natural convection using fan compartment air.

The EEC is installed on the fan case using four vibration isolators. The EEC has eight electrical connectors that allow it to interface with the aircraft systems and engine sensors, a data storage unit (DSU), and two pneumatic pressure ports.

The EEC has an internal ambient pressure sensor that receives ambient pressure through a port on the lower right side of the EEC. The EEC also receives inlet pressure (P2) through the P2 port.

DATA STORAGE UNIT

The DSU is used for storage of engine configuration and identification data, engine historical and exceedance information, thrust ratings data, and fan trim balance data. If an EEC or prognostics and health management unit (PHMU) is replaced, the data stored in the DSU is supplied to the new unit.

The DSU is mounted on the EEC A channel housing. The DSU is attached to the engine fan case by a lanyard and remains with the engine if the EEC is replaced.

CAUTION

The DSU must remain with the engine when the EEC is removed. An incorrectly installed DSU can affect engine operation.

PROGNOSTICS AND HEALTH MANAGEMENT UNIT

The PHMU electronic components are installed in a cast aluminum housing that is cooled by natural convection using fan compartment air.

The PHMU is installed on the fan case using four vibration isolators. The PHMU has two electrical connectors that allow it to interface with the EEC and engine sensors.

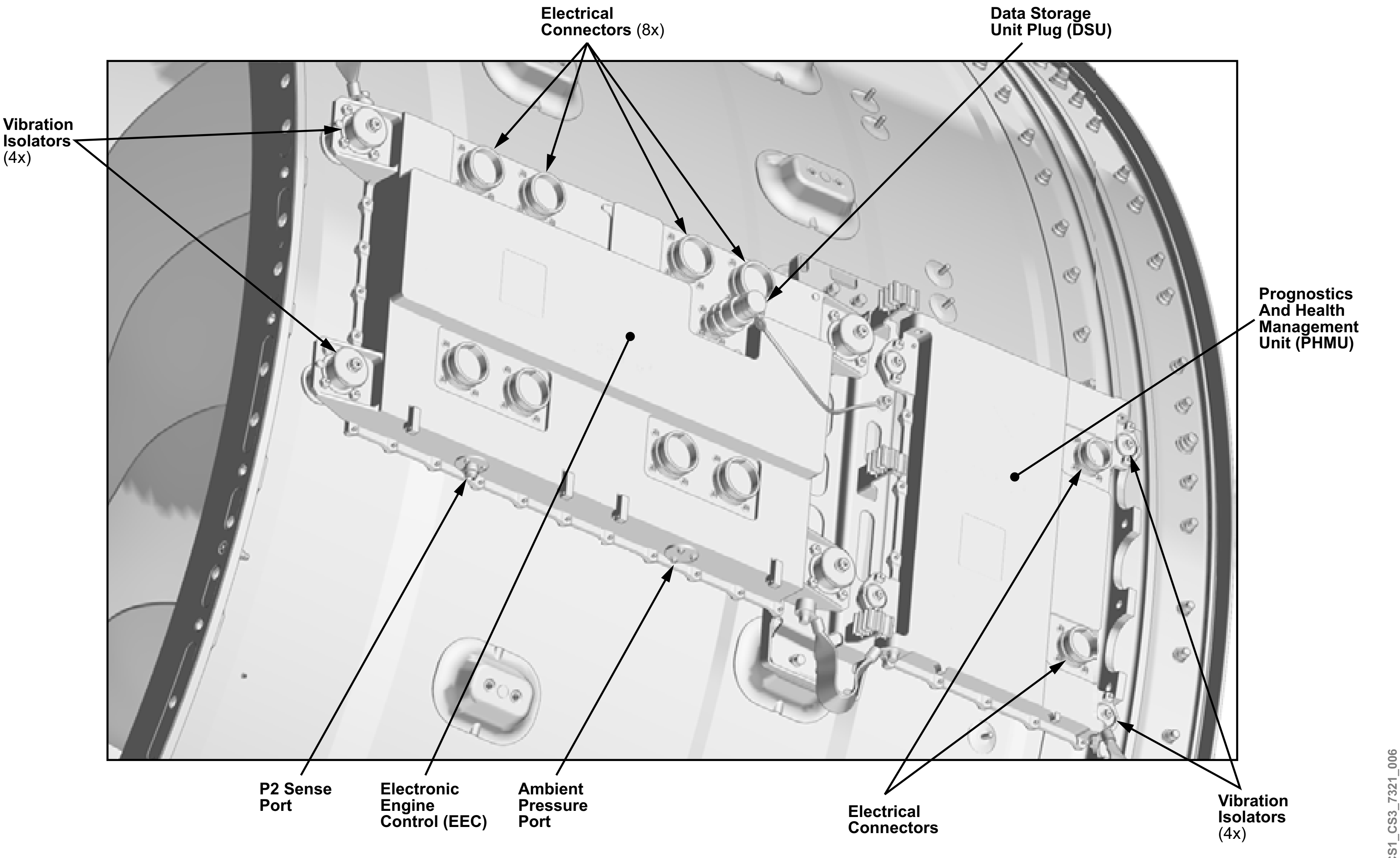


Figure 13: Electronic Engine Control, Data Storage Unit, and Prognostics and Health Management Unit

PERMANENT MAGNET ALTERNATOR GENERATOR

The permanent magnet alternator generator (PMAG) consists of a stator and a rotor. The stator and electrical windings are attached to the gearbox with captive bolts. The rotor remains attached to the gearbox shaft when the stator is removed.

The PMAG stator has dual-channel EEC AC power windings, dual-channel N2 windings and a single generator winding for AC power to the fly-by-wire (FBW) controller.

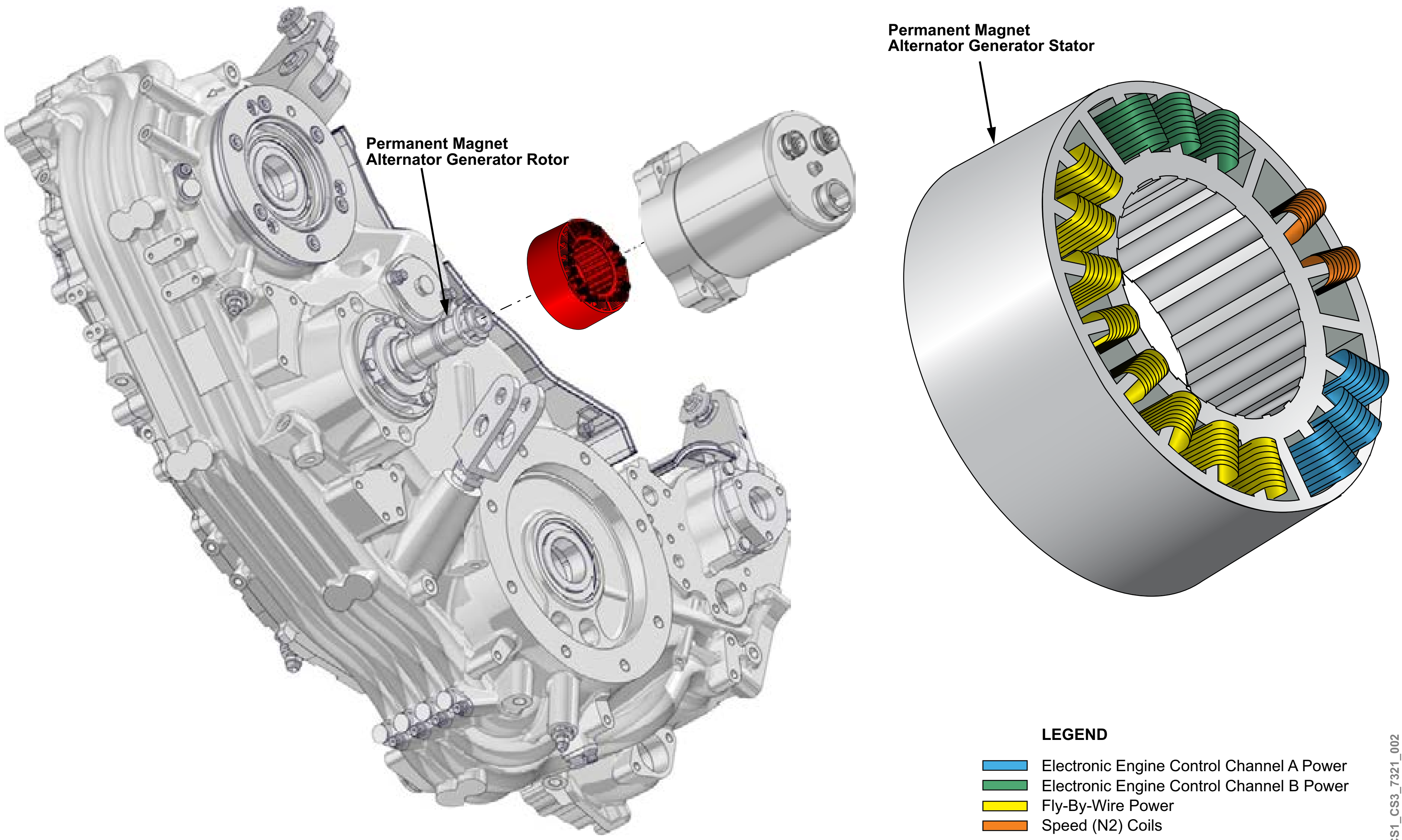


Figure 14: Permanent Magnet Alternator Generator

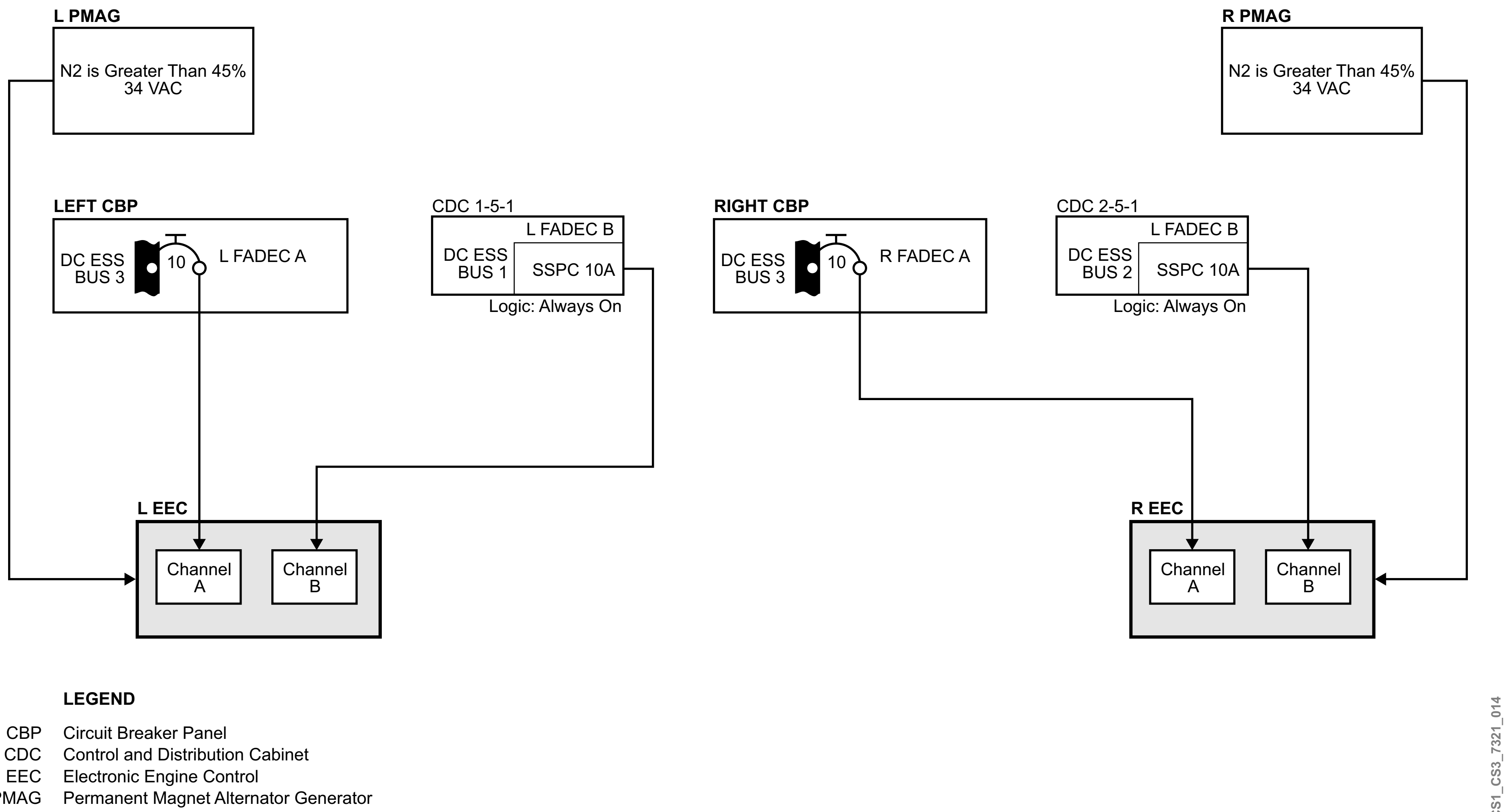
DETAILED COMPONENT INFORMATION

ELECTRONIC ENGINE CONTROL POWER

During normal operating conditions with the engine operating above 45% N2 speed, the EEC is powered by the permanent magnet alternator generator (PMAG). The PMAG provides a separate output for each EEC channel.

When the engine is not running, both EEC channels are powered by the aircraft electrical system. Channel A of each EEC is powered by DC ESS BUS 3. Channel B of the left EEC is powered by DC ESS BUS 1. Channel B of the right EEC is powered by DC ESS BUS 2.

When the engine is at or above minimum idle, a failure of one PMAG output results in the corresponding EEC channel switch its aircraft electrical power source. The other channel of the EEC continues operation using power from the PMAG.



LEGEND

CBP	Circuit Breaker Panel
CDC	Control and Distribution Cabinet
EEC	Electronic Engine Control
PMAG	Permanent Magnet Alternator Generator

Figure 15: Electronic Engine Control Power

ELECTRONIC ENGINE CONTROL FAULT DETECTION

The EEC is designed to be fail-safe by detecting and accommodating component failures within the system. This is accomplished by the fault accommodation logic in the software. The main goal of the fault accommodation logic is to make the system operational upon the detection of all single electrical faults when the system is dispatched with two channels operational. The secondary goal is to maintain as much engine functionality as possible when multiple faults exist.

Failures detectable by the control system are classified by their effect on the ability of the channel to control the engine. Many input failures have redundant backup sources and therefore do not require a channel switchover. The system is designed such that the minimum number of channel switchovers will occur due to faults.

The fault accommodation is based on:

- If there are no detected faults in either channel, use average of the two channels
- If an input fault or faults exist, use data from the other channel
- If an output fault or faults exist, switch to other channel output
- In the case of a dual failure, the EEC fails to a designated safe mode

The active-standby control scheme is used until the first failure is detected. At that point, control switches over to the active driver control scheme.

The channel switchover logic has active driver and active-standby control schemes. The active driver control scheme allows either channel to control any of the output drivers independently, regardless of which channel is in control.

In the active-standby control scheme, the system gives active control to the healthiest channel. When the engine is shutdown, control of the engine switches to the other channel for the next engine start. This aids in the detection of potential hidden failures and provides equal output driver usage in the EEC. The channel switchover function is tested on each EEC power application.

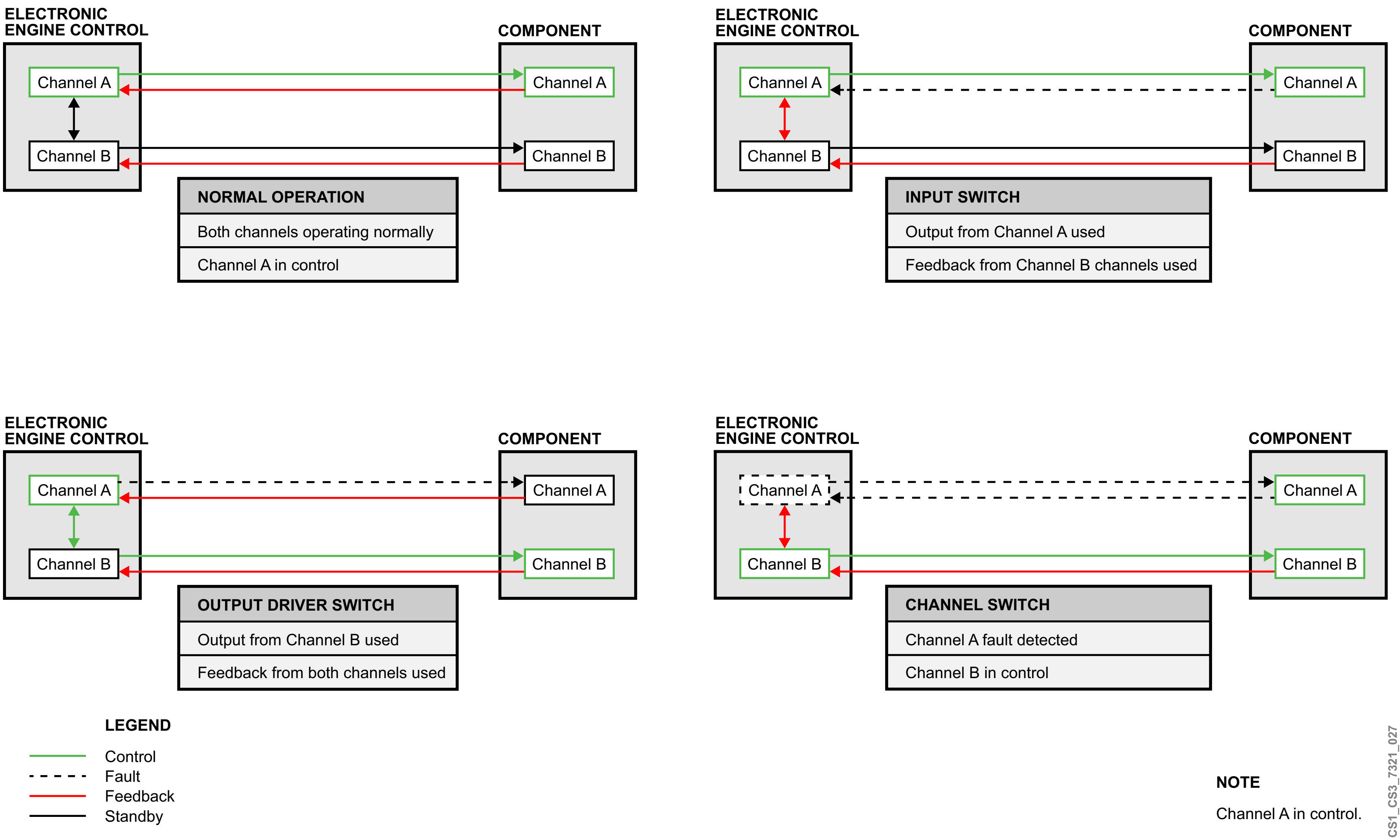


Figure 16: Electronic Engine Control Fault Detection

PROGNOSTICS AND HEALTH MANAGEMENT UNIT

The prognostics and health management unit (PHMU) performs engine health monitoring functions based on data provided by the EEC. The PHMU also performs the following functions:

- Engine vibration monitoring
- Oil debris monitoring
- Fan drive gear system (FDGS) oil monitoring

The PHMU provides engine health monitoring by comparing streaming engine data from engine sensors to stored models of engine gas path and subsystem parameters. The P2.5/T2.5 sensor provides P2.5 pressure to the PHMU. The other engine sensors provide information through the EEC. The information is processed by the PHMU and the appropriate output is sent to the EEC. The EEC then sends this information to the onboard maintenance system (OMS) and the health management unit (HMU).

The PHMU receives signals from engine vibration sensors to assess the health of rotating and mechanical components. The vibration signals are compared against the engine speed signals supplied by the EEC to determine the location of the vibration. The processed vibration signals are sent to EICAS and also provide additional vibration information for fan balancing.

The oil debris monitoring system checks for metal particles in the oil system and provides an early indication of engine wear. The PHMU supplies the information to the EEC as part of the data report supplied to the HMU. The PHMU also monitors the FDGS oil supply to ensure that a continuous supply of oil is being provided.

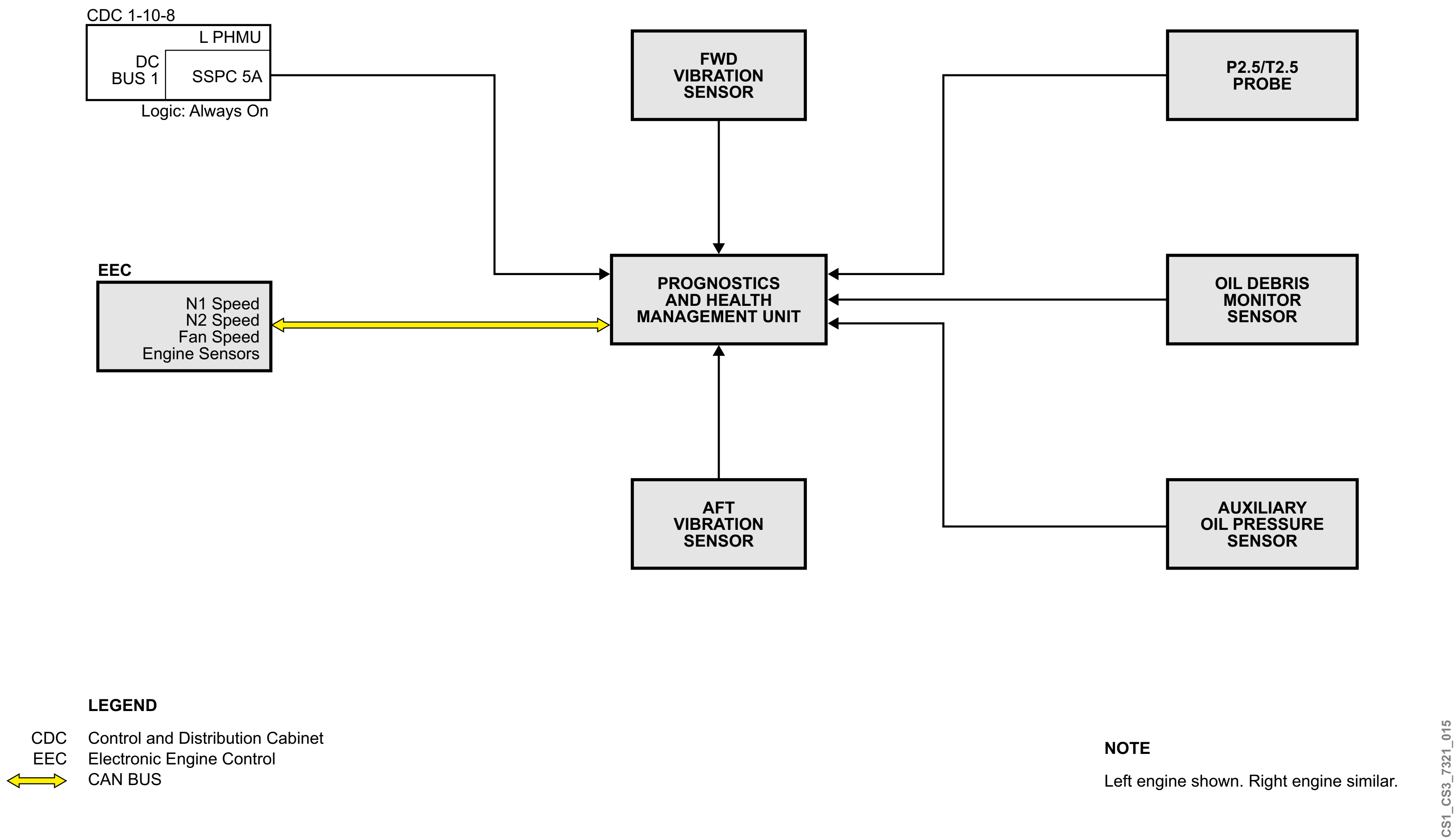


Figure 17: Prognostics and Health Management Unit

DATA STORAGE UNIT

The digital storage unit (DSU) connects directly to EEC channel A. It communicates with both channels through a serial data bus. Data is stored in the DSU memory, EEC channel A, and EEC channel B.

The DSU is capable of storing up to 8 MB of data. Data from the current flight overwrites older information when the memory section is full. The DSU stores the following information:

- Engine identification data
- Trim balance data
- Engine snapshot data
- Engine historical data

Engine configuration and identification data is programmed on the DSU during production acceptance testing and is not modifiable on the aircraft. The DSU remains with its specific engine for the life of the engine, unless the DSU fails.

Engine identification data is read from the DSU during EEC initialization. Each channel performs an integrity check on the engine identification data in the DSU and in its own data flash. The DSU as the primary source, and uses its own data as a back-up source.

The engine serial number is used by the prognostics and health monitoring unit (PHMU) for engine condition monitoring.

N1 Modifier

To ensure that all engines have uniform N1/thrust characteristics, an N1 class modifier is defined and its value is set in the DSU for each engine by tests during production. The EEC adds the N1 class modifier to the mechanical N1 speed to get the N1 feedback used in the N1 control loop and the N1 speed Indication displayed on EICAS.

Engine Thrust Rating

The EEC uses the engine nameplate identification and aircraft type on the DSU to select which set of thrust ratings data to transfer to processor memory. During synchronization, the thrust ratings data corresponding to the synchronized engine nameplate identification and aircraft type are transferred to the EEC processor memory.

Engine Configuration Selector

The engine configuration selector is used by EEC software to select control laws parameters compatible with the engine hardware configuration.

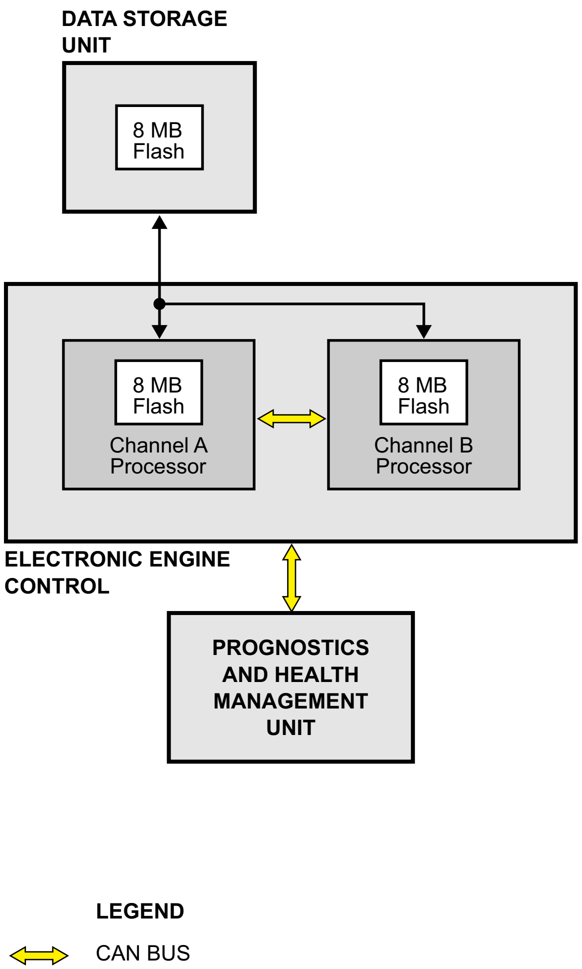
DSU Failures

If the DSU is not installed, and the aircraft is on the ground, the EEC does not allow the engine to start. If the DSU disconnects in flight, the EEC uses the programming information stored in the EEC memory to continue controlling the engine.

DSU critical faults such as nameplate and aircraft type incompatibility with the selectable engine ratings in one or both EEC channels result in an ENGINE FAULT advisory message and an 73 L(R) ENGINE FAULT - FADEC FAULT 1 INFO message. The aircraft cannot be dispatched.

A DSU noncritical fault such as a failure to write data to the DSU detected by one channel only results in an ENGINE FAULT advisory message and an 73 L(R) ENGINE FAULT - FADEC FAULT 2 INFO message.

The DSU dispatch failures are only annunciated on the ground to prevent an ENGINE FAULT advisory message and associated INFO message from being displayed in flight should a DSU failure be detected after an in flight shutdown.



DATA STORAGE UNIT DATA			
Engine Identification Data	**Trim Balance Data**	**Engine Snapshot Data**	**Engine Historical Data**
DSU Hardware Part Number DSU Data Part Number Engine Serial Number Engine Nameplate Thrust (PW1525G, PW1524G, PW1521G, PW1521GA, or PW1519G) Aircraft Type (CS100, or CS300) N1 Modifier CRC32 Checksum	Fan Weight Data Low Pressure Turbine Weight Data Trim Balance Solution Data Vibration History Data Data for Trim Balance Calculations	Engine Faults (such as sensor or actuator failures) Engine Events (such as exceedances and surges)	Total Engine Running Time Engine Run Time in Flight Start Counter Shutdown Counter Aborted Start Counter Number of Flights Preferred Channel in Control Preferred Igniter Next Overspeed Shutdown Test Set-up (channel A, channel B, or channel A and B) Overspeed Shutdown Test Failed Journal Oil Shuttle Valve Shutdown Test Failed NVM Engine Position HPT Disk Temperature at Shutdown

CS1_CS3_7321_001

Figure 18: Data Storage Unit

CONTROLS AND INDICATIONS

THROTTLE QUADRANT ASSEMBLY

The throttle quadrant has two thrust levers to control the engines. Each thrust lever controls the forward and reverse thrust functions.

The L(R) ENG switches provide engine run and shutdown commands.

ENGINE PANEL

The ENGINE panel CONT IGN PBA provides a manual start request signal to the EEC for engine starting.

The START switch is spring loaded to the AUTO position for normal automatic starts. The switch has L(R) ENG CRANK positions for manual engine starts.

ENGINE FIRE PANEL

The ENGINE FIRE panel FIRE PBA provides a shutdown signal to the EEC when pressed. The FIRE PBA is used to shut down the engine should the L(R) ENG switch shutdown command fail.

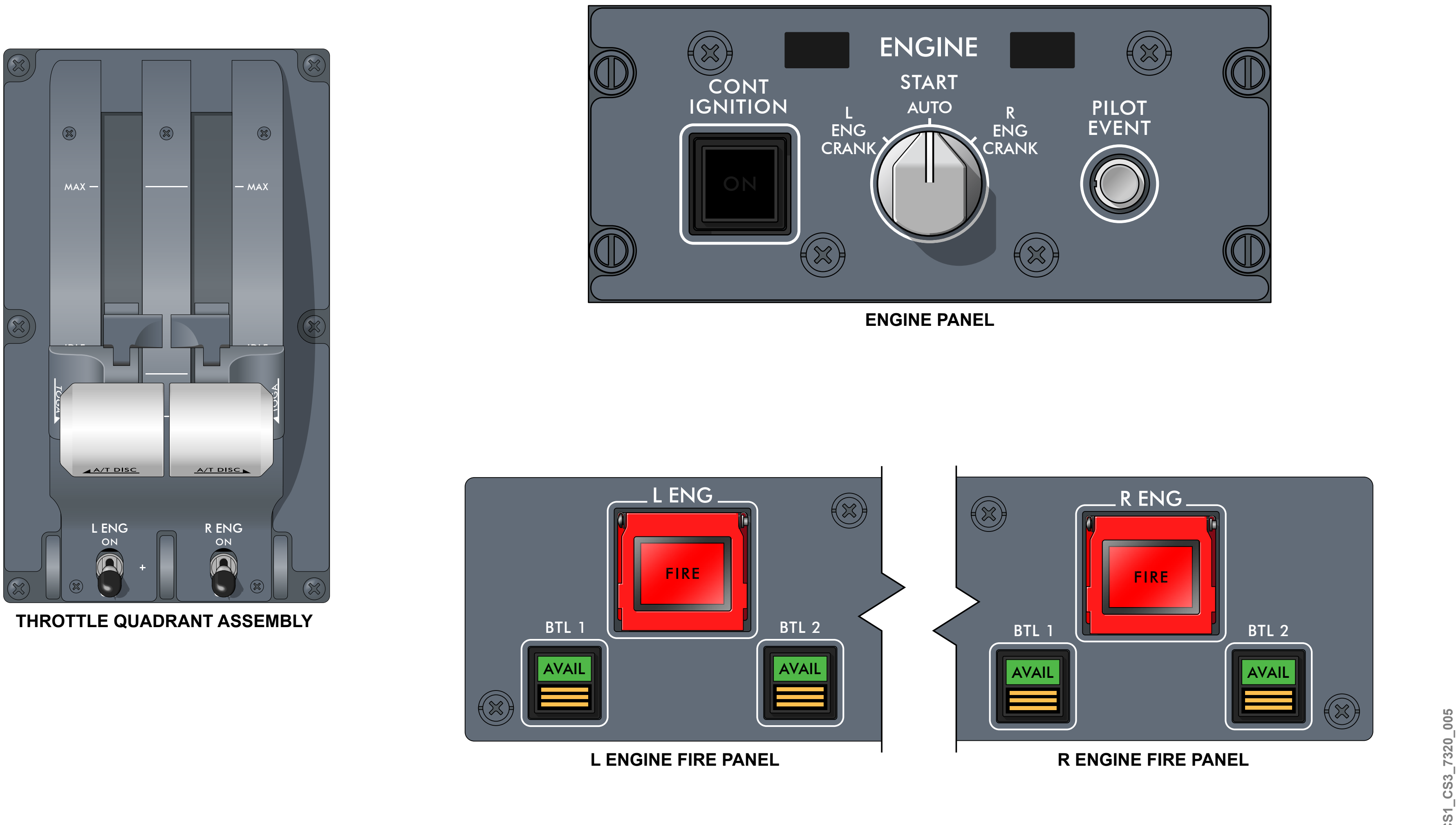
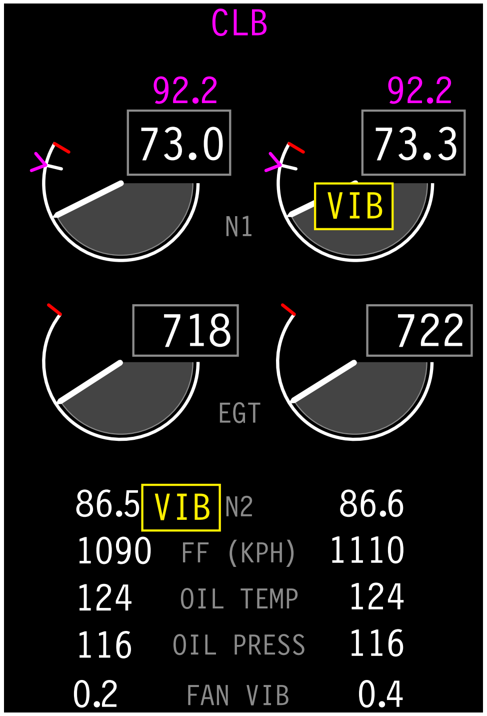


Figure 19: Fuel Control System Controls

INDICATIONS

The fuel control system indications are displayed on the engine indication and crew alerting system (EICAS) page.



EICAS

CS1_CS3_7320_008

Figure 20: Fuel Control System Indications

DETAILED DESCRIPTION

AIRCRAFT AND ELECTRONIC ENGINE CONTROL INTERFACE

The following aircraft systems interface with the EEC for engine control.

Flight Management System

The flight management system (FMS) provides flight plan information used to calculate the engine thrust reference setting. The FMS also supplies the thrust mode selection automatically and is the interface for manual selection of thrust mode, automatic power reserve (APR) arming, N1 synchronization arming, and FLEX temperature selection.

Integrated Air System Controllers

The integrated air system controllers (IASCs) supplies the EEC with environmental control system (ECS) and wing anti-ice system configuration. The IASCs receive bleed status information from the EECs.

Air Data Information

The EEC uses aircraft air data as primary source of air data because of its accuracy and redundancy. The air data system supplies total air temperature, pressure, and Mach information. The air data system receives T2 information for use in the event of a TAT failure.

Landing Gear and Steering Control Units

The EEC use the weight-on-wheels (WOW) signal for engine idle speed control, engine control, and thrust reverser control.

Brake Data Concentrator Units

The brake data concentrator units (BDCUs) supply wheel speed signals for control of the thrust reversers.

Primary Flight Control Computers

The primary flight control computers supply a steep approach signal used to modify the engine idle speed for a steep approach.

Ice Detection

The EEC uses the ice detector signal to control the cowl anti-ice valve (CAIV) operation.

Engine Panel

The manual or automatic engine start mode is selected from the ENGINE panel.

Autothrottle

The EEC provides the N1 target command to the autothrottle system. The autothrottle controls the thrust levers based on the N1 target information. The autothrottle also provides engine trim signals for engine synchronization.

ENGINE FIRE Panel

The ENGINE FIRE panel supplies engine shutdown commands when an engine fire PBA is pressed.

Throttle Quadrant Assembly

The throttle quadrant assembly (TQA) provides the EEC with thrust lever position information and engine ON/OFF commands. The EEC supplies a signal to energize the baulk solenoid when the thrust reverser translating sleeves are deploying or stowing.

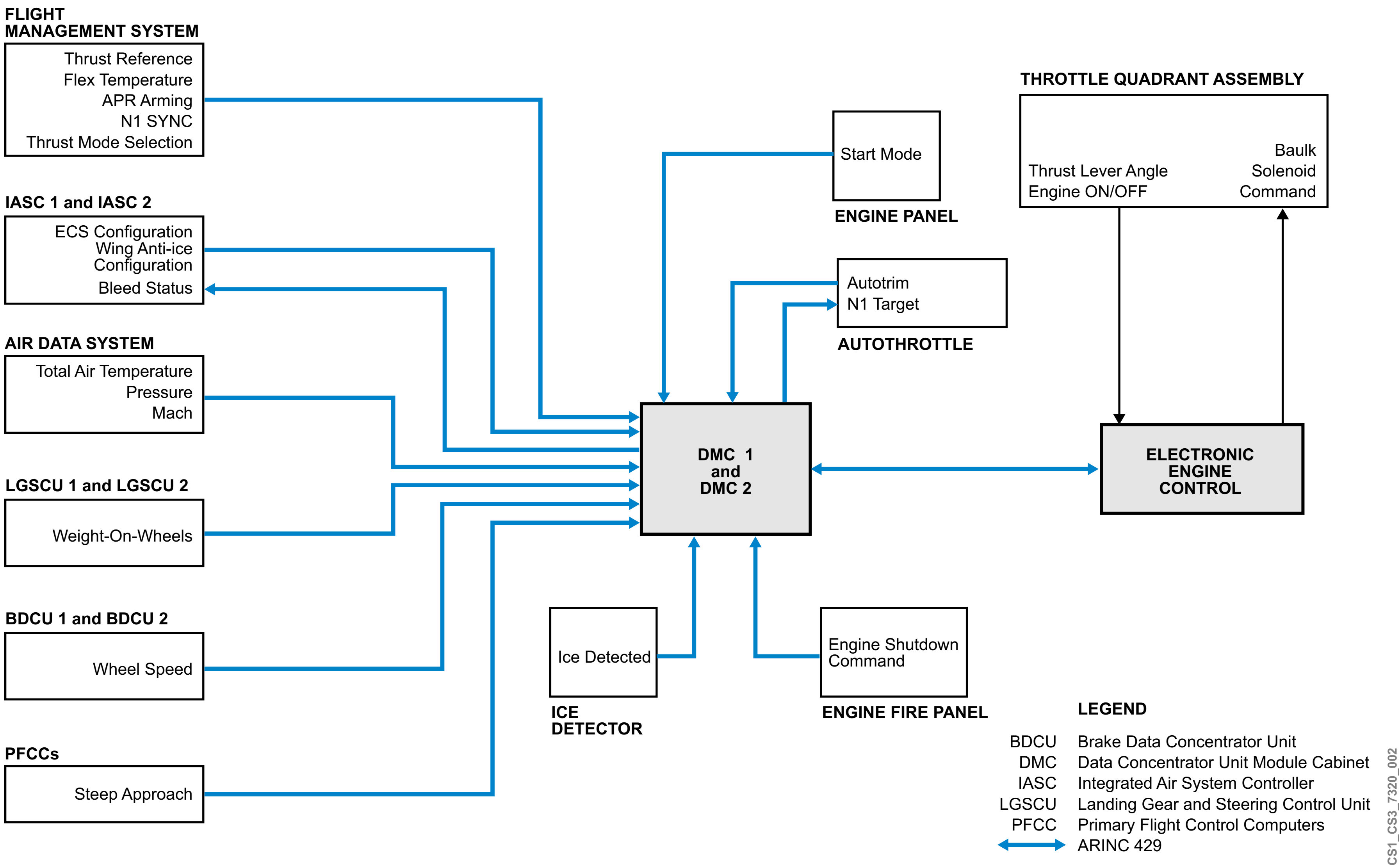


Figure 21: Aircraft and Electronic Engine Control Interface

ENGINE SENSOR INTERFACE

The engine sensors are used to provide fuel control and engine health monitoring.

P2/T2 Probe

The P2/T2 probe consists of a single channel P2 probe that connects to channel B of the EEC and a dual channel T2 sensor. The P2 probe measures the engine inlet total pressure. P2 is used to validate the air data from the air data system, control scheduling functions, engine rating calculation and Mach number calculation.

The P2 probe is heated. The left P2/T2 probe heater is powered by AC BUS 1 and the right P2/T2 is powered by AC BUS 2. The EEC commands the probe heat on when the engine is running above idle and the temperature is below 9°C (48°F). When the probe heat is on, the EEC software compensates for the effect of the probe heater.

The EEC turns the P2/T2 heater off when the engine is shutdown or the temperature is above 9°C. If the EEC detects a heater failure, a 73 L ENGINE FAULT - P2/T2 HEATER INOP INFO message is displayed.

The T2 sensor measures the engine total inlet temperature. The T2 sensor feeds both channels of the EEC. T2 is used to validate the total air temperature information from the air data system, control scheduling functions including engine rating calculation and Mach number calculation. The T2 sensor information is used in place of air data information when the aircraft is on the ground and below 60 kt.

The P2T2 probe is used if air data from the air data system is not valid or not available. If the P2T2 probe also fails, the EEC uses default values:

- Inlet total temperature: synthesized total temperature based on standard day temperature, Mach and pressure
- Total pressure: 14.7 psi
- Pressure altitude: 17000 ft

Pamb Sensor

The single-channel Pamb sensor measures the ambient pressure through a port on EEC. The Pamb sensor is connected to channel A of the EEC. Channel B receives Pamb data via the EEC CAN BUS.

The engine sensed altitude is calculated based on Pamb. Pamb is used to validate the air data from the air data system, control scheduling functions, engine rating calculation and Mach number calculation.

P2.5/T2.5 Sensor

The P2.5/T2.5 sensor consists of a pressure probe and a temperature sensor.

The P2.5 pressure probe measures the pressure at the low-pressure compressor (LPC) exit. The probe receives an excitation signal from the PHMU and provide an output used to monitor high-pressure compressor (HPC) and LPC health.

The T2.5 temperature sensor measures total temperature at the LPC exit. The sensor receives excitation from channel A of the EEC and provides an output used to monitor high-pressure compressor (HPC) and LPC health. Channel B of the EEC receives T2.5 sensor data via the EEC CAN BUS.

T3 Sensor

The T3 thermocouple sensor provides a single output to channel B of the EEC. It measures the HPC exit temperature. T3 is used to assess engine performance and compressor health.

Pb Sensor

The Pb (burner pressure) sensor is a dual channel pressure sensor. The EEC provides excitation voltage to the sensor. Each channel has a low and a high range pressure transducer. Pb is used in fuel scheduling, stall detection, autostart logic, continuous ignition, and shaft shear detection.

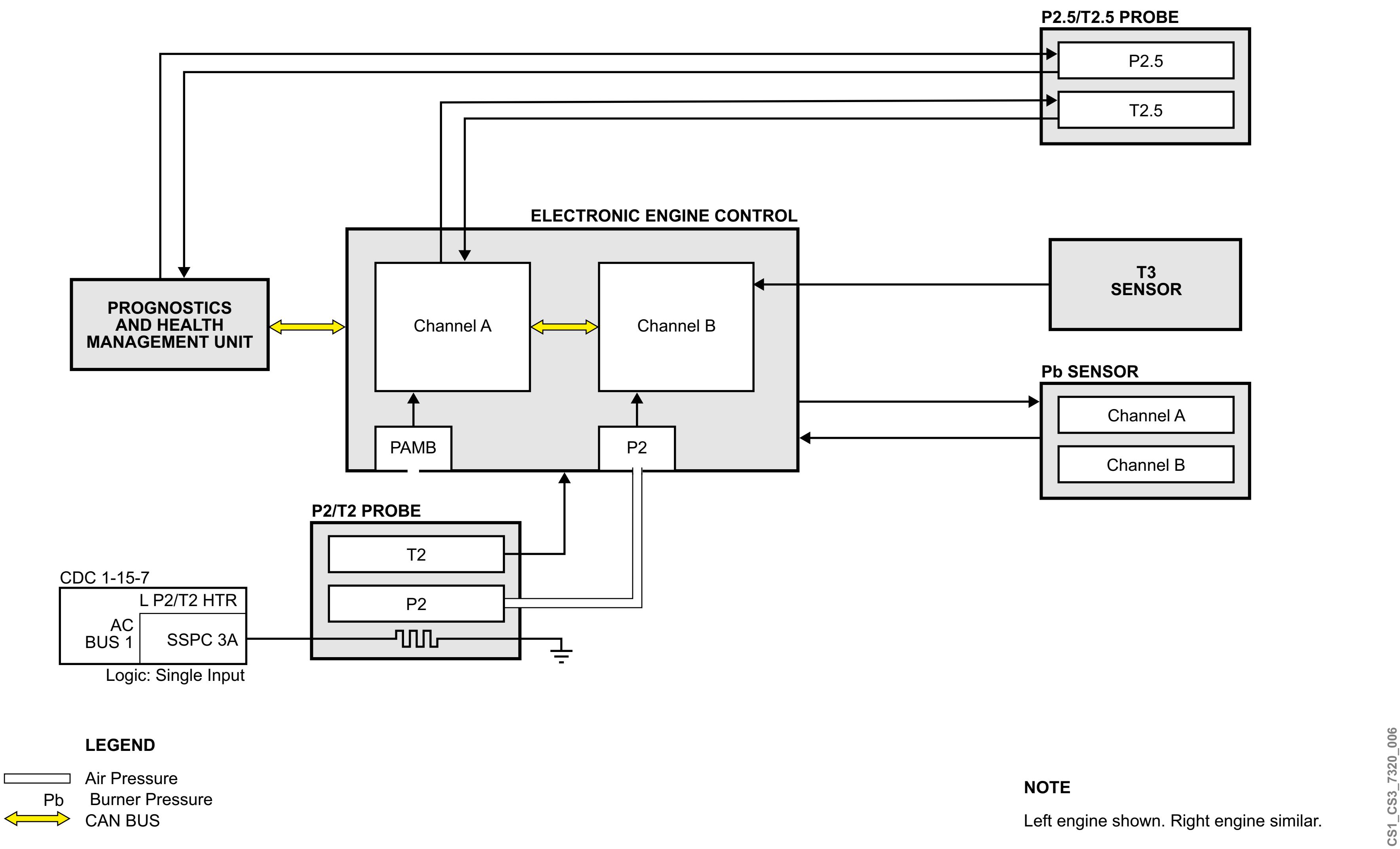


Figure 22: Engine Sensor Interface

ENGINE OPERATING LIMITS

The engine operating limits ensure that the engine operates within the design parameters. The EEC uses feedback from engine sensors and inputs from aircraft systems to ensure that the design limits are not exceeded.

A governing loop sets the primary engine thrust requirements using N1 as a reference. The governing loop is set based on aircraft system inputs for the thrust requirements.

The N1 is limited to maximum speed of 10,600 rpm (100%) and a minimum of 1,616 rpm (15%). The minimum N1 limit is dependant on environmental conditions, minimum thrust level, ice accumulation and bleed air requirements. The governing loop is constrained by EEC limiting loops.

The following limiting loops are monitored by the EEC:

- N2
- Fuel flow
- Fuel flow/burner pressure ratio
- Burner pressure
- Fuel flow rate

The N2 is limited to maximum speed of 24,470 rpm (100%) and a minimum of 13,087 rpm (53%). The minimum N2 limit ensures that the engine remains above generator trip speed.

If N1 or N2 increase beyond 100%, but do not reach the overspeed shutdown condition of 105%, a L(R) ENG EXCEEDANCE caution message is displayed and the engine thrust must be reduced manually using the thrust levers.

The fuel flow maximum limit protects the IFPC. There is a fuel flow minimum that assures a minimum amount of fuel is delivered to the combustor during starts. While in flight, the minimum fuel flow provides flameout protection at high altitude and during high G maneuvers.

The fuel flow to burner pressure ratio limit loop makes sure that the fuel to air ratio inside the combustor is maintaining a margin against flameout.

The burner pressure maximum limit loop protects the engine from over pressurization of the combustion chamber. The burner pressure minimum limit loop also ensures enough pressure for the environmental control system. The minimum burner pressure depends on the demands for engine air made by the following systems:

- Cowl anti-ice
- Wing anti-ice
- Environment control system

The fuel flow rate maximum and minimum limit loops are provided to prevent excessive fuel command changes during acceleration and deceleration. During engine starting, the EEC modifies the fuel schedule once the starter air valve (SAV) closes and the air turbine starter (ATS) disengages.

Thrust Limitation at Low Airspeed

A thrust limitation at low air speed (TLLS) function is available in TO mode only. TLLS maintains a minimum nose landing gear loading during the takeoff phase.

The TLLS function is enabled when the aircraft is on the ground and the airspeed is less than 80 kt. As the airspeed increases, the TLLS input is reduced and the required thrust commanded by the EEC is provided by 80 kt.

The TLLS function is inhibited when the aircraft is on the ground and the parking brake is set. This ensures full power can be developed during a high-power ground run.

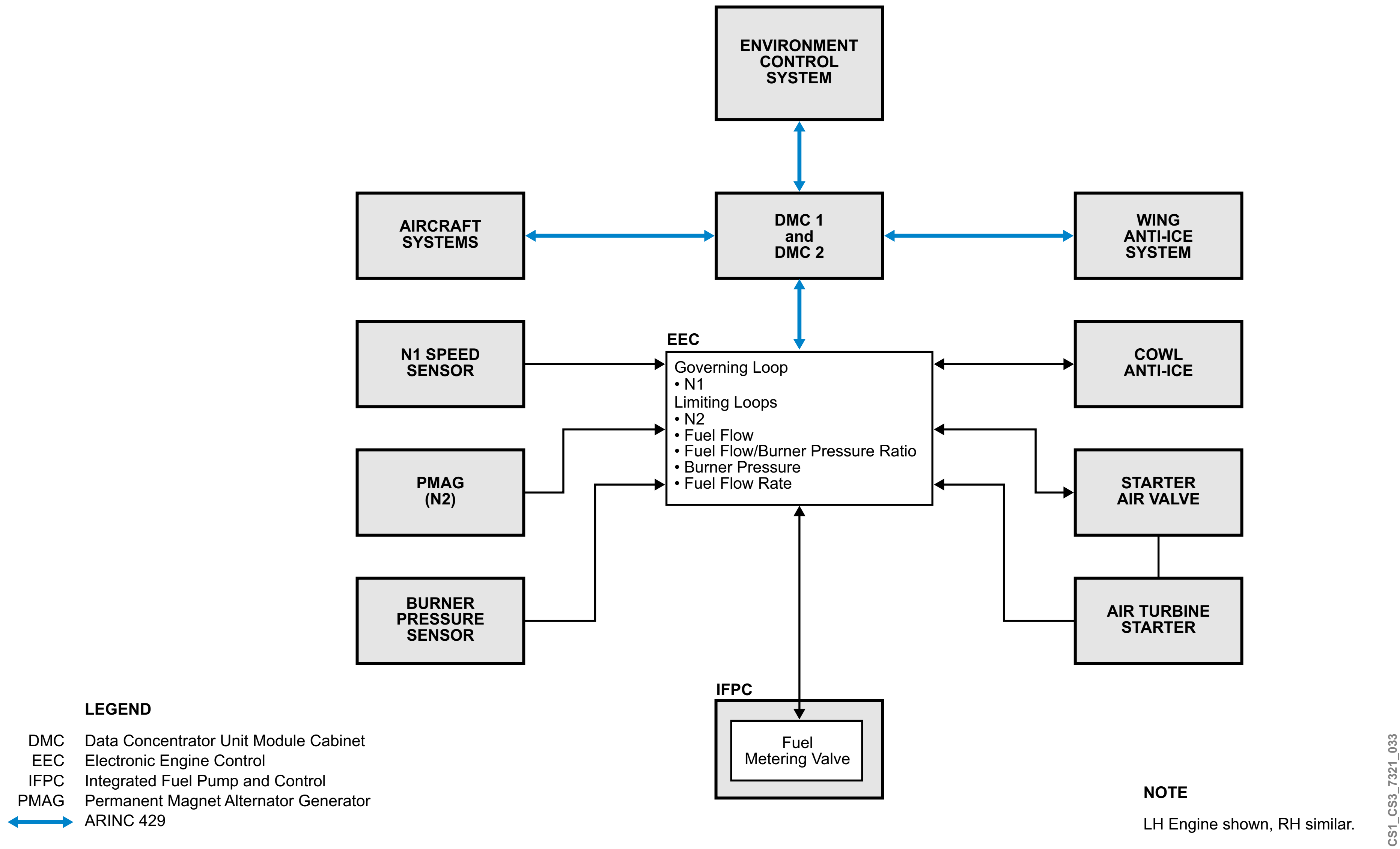


Figure 23: Electronic Engine Control Engine Operating Limits

OVERSPEED PROTECTION, SHAFT SHEAR PROTECTION AND THRUST CONTROL MALFUNCTION

Overspeed protection

Overspeed protection is provided for both high (N2) and low (N1) pressure rotors. This is done via dedicated protection computers within the EEC independent of the channel A and B control computers.

The protection computers control the overspeed shutdown solenoid (OSS). When either N1 or N2 engine speeds exceed the overspeed limit, at any time the engine is running, the engine immediately shuts down.

The protection computers share the N1 and N2 signals provided to the control computers within the EEC and have a separate crosstalk CAN BUS.

The EEC limits fuel flow when redline limits are reached for N1 or N2. When either of the protection computer detects an overspeed limit of 105% N1 or 105% N2, the OSS is energized.

Once an overspeed limit exceedance is detected, the system is latched and must be reset. The system can be reset by cycling the power supply to the EEC, or by selecting the L(R) ENG switch to ON to initiate the engine start sequence. To ensure that there are no undetected faults, the OSS is tested at shutdown.

Shaft Shear Protection

The protection computers also provide protection against low and high pressure rotor shaft shear. If the protection computers determine that either shaft shear conditions have occurred, the OSS is energized to shutdown the engine.

Thrust Control Malfunction

A thrust control malfunction is an uncontrollable high thrust event. A thrust control malfunction is characterized by:

- Uncommanded engine thrust increase
- Engine thrust remaining at a high thrust setting when a lower thrust setting is commanded

In either case, the thrust can not be controlled by the thrust lever. When the EEC detects a thrust control malfunction, the OSS is energized to shut down the engine

Thrust control malfunction detection is inhibited during:

- Thrust reverse operation
- Approach
- Steep approach

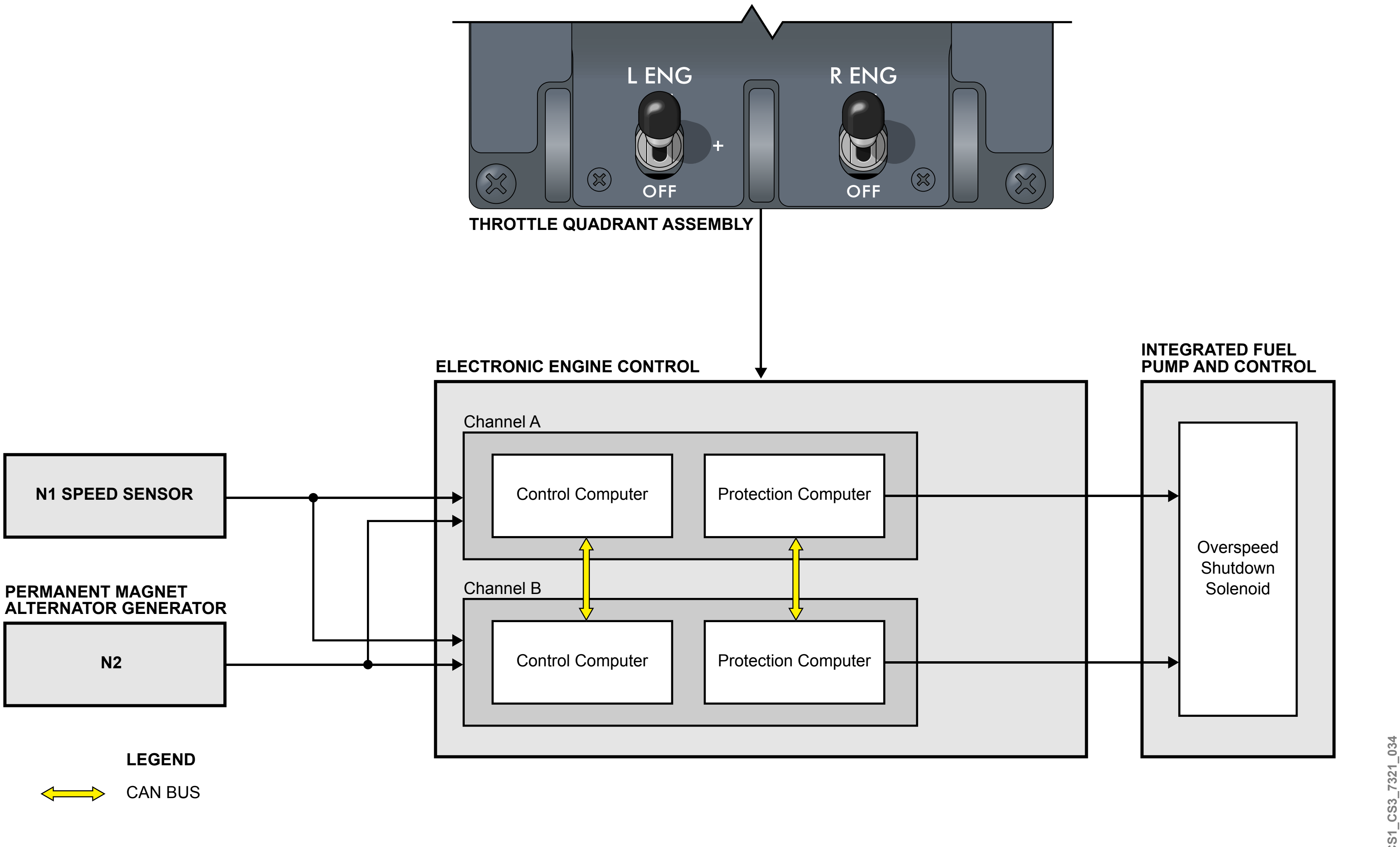


Figure 24: Overspeed, Shaft Shear Protection, and Thrust Control Malfunction

INTEGRATED FUEL PUMP AND CONTROL OPERATION - START

The IFPC is controlled by the EEC to provide metered fuel for combustion based on thrust lever position and ambient operating conditions. The IFPC also provides servo fuel for the fuel actuated system on the engine. The IFPC consists of both hydromechanical and electromechanical valves to control the fuel flow.

The fuel metering valve (FMV) is a dual channel component controlled by the EEC. A dual channel torque motor allows either channel of the EEC to control the FMV. FMV feedback is provided to the EEC using a dual coil LVDT.

The pressure-regulating valve (PRV) maintains a constant pressure drop across the FMV by returning excess pump flow to the inlet of the high-pressure pump. The excess flow is dependent on the servo fuel demand and metered fuel requirements.

The fuel flow meter provides the EEC with an indication of the amount of fuel being used for combustion.

When the engine is commanded to shutdown, the windmill bypass valve (WBV) opens to connect fuel from upstream of the minimum pressure-shutoff valve (MPSOV) with the fuel filter inlet.

The minimum pressure-shutoff valve (MPSOV) is a spring-loaded valve that prevents fuel from entering the primary and secondary fuel manifolds until it is sufficiently pressurized.

The overspeed shutdown solenoid (OSS) is a dual-coil solenoid valve that operates the WBV and MPSOV to shutoff metered flow to the engine.

The flow divider valve (FDV) schedules the fuel flow to primary and secondary manifolds depending on engine start requirements. The FDV is normally spring-loaded to the closed position, blocking fuel to the secondary fuel manifold. Once the fuel pressure between the primary and secondary fuel manifolds exceeds 100 psi, the FDV opens allowing fuel into the secondary manifold.

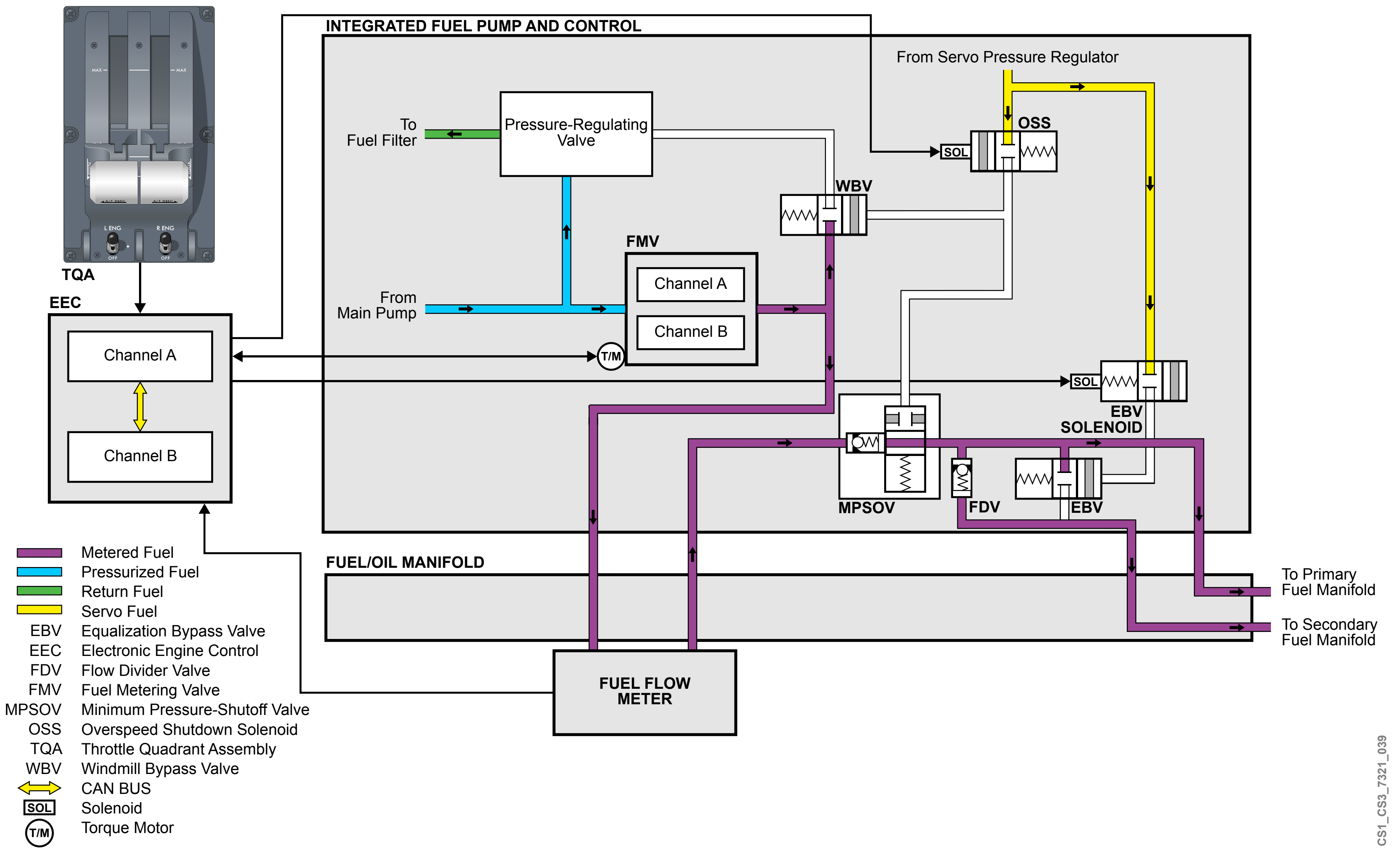


Figure 25: Integrated Fuel Pump and Control Operation - Engine Start

INTEGRATED FUEL PUMP AND CONTROL OPERATION - RUN

The EEC energizes the EBV solenoid when the engine start is completed. The equalization bypass valve solenoid controls the equalization bypass valve (EBV). The EBV energizes to equalize the fuel flow in the primary and secondary fuel manifolds. The FDV closes and the secondary fuel manifold is supplied through the EBV.

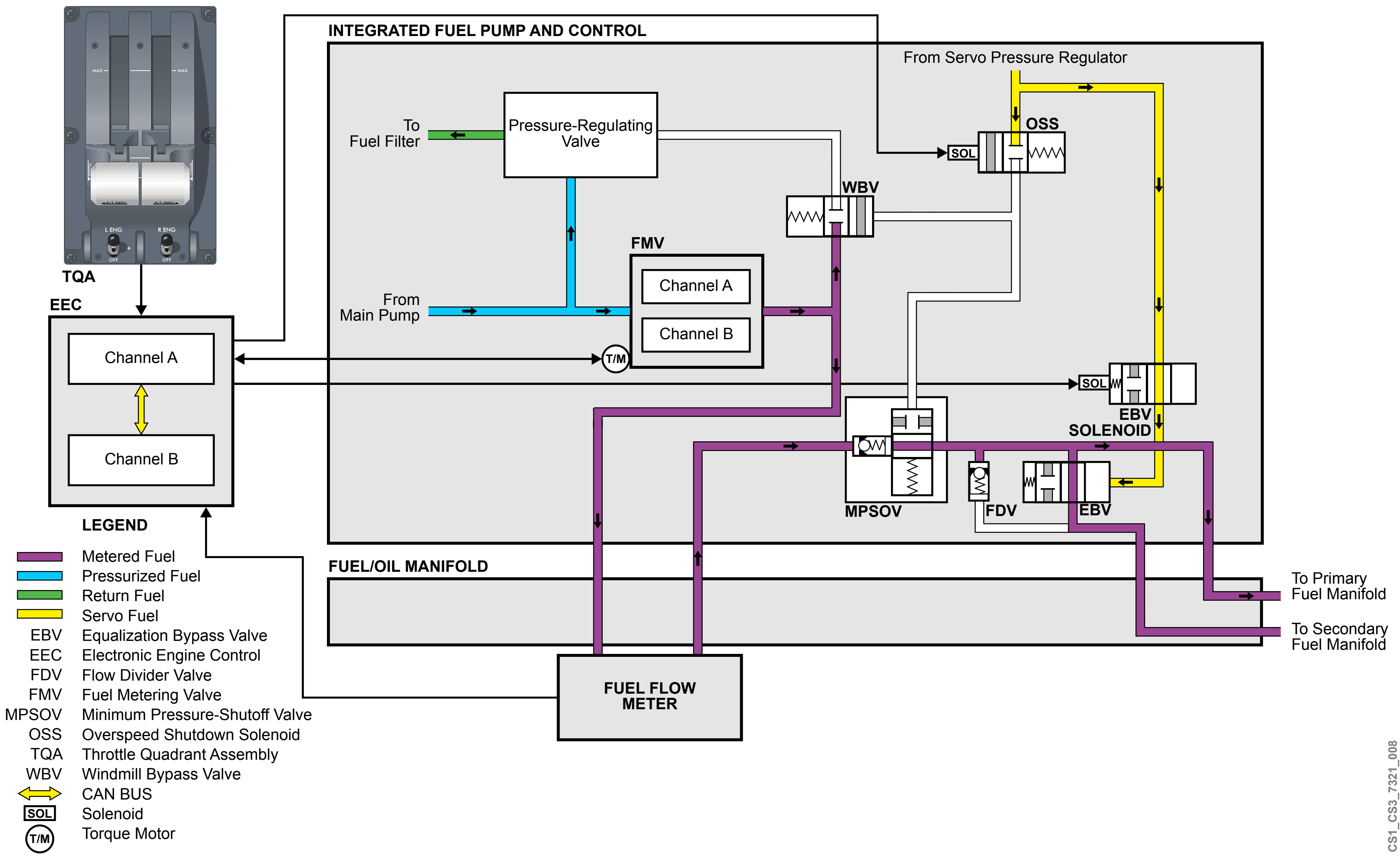


Figure 26: Integrated Fuel Pump and Control Operation - Engine Run

NORMAL ENGINE SHUTDOWN

On Ground

The EEC commands shutdown of the engine using the overspeed shutdown solenoid (OSS) whenever one of the following conditions is present:

- L(R) ENG SWITCH is moved to the OFF position on the throttle quadrant assembly
- An excessive engine overspeed is detected by the EEC
- A shaft shear is detected by the EEC

The EEC uses the overspeed protection system to shut down the engine. The EEC channel in control commands the overspeed shutoff solenoid to energize. On every third shutdown both EEC channels command the overspeed shutoff solenoid to fully test the system.

When the EEC energizes the OSS, servo fuel pressure opens the WBV to allow fuel to flow back to the pressure-regulating valve. The pressure-regulating valve opens so that the fuel flows back to the fuel filter and is recirculated. At the same time, the MPSOV moves to the closed position and stops fuel from flowing to the engine fuel nozzles. The EEC de-energizes the equalization bypass valve solenoid to the closed position.

Once the overspeed protection system has shut down the engine, or if the overspeed protection system fails to shut down the engine, the EEC commands the fuel metering valve closed. When the fuel metering valve is confirmed closed, the overspeed shutdown solenoid is de-energized and all EEC processors are reset.

In Flight

In flight the EEC commands the engine to shutdown by driving the FMV to the minimum flow position first to allow time for the high-pressure compressor (HPC) stator vane actuator to track to the optimal start position.

When the HPC stator vane actuator is in position, the fuel metering valve is commanded closed and the OSS is energized.

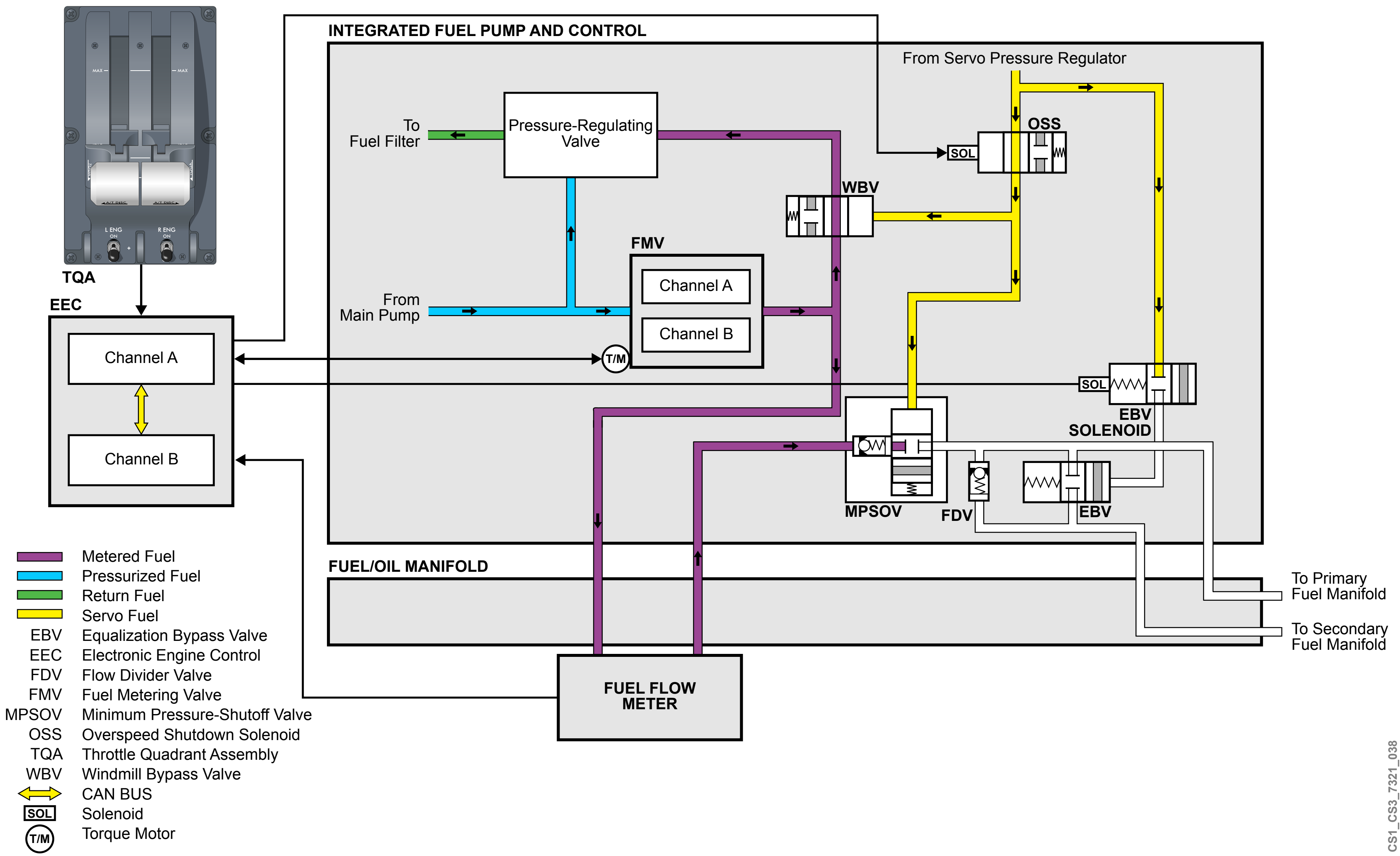


Figure 27: Normal Engine Shutdown

EMERGENCY SHUTDOWN

An emergency shutdown can be carried out using the L ENG or R ENG FIRE PBA. Pressing the fire PBA initiates the following:

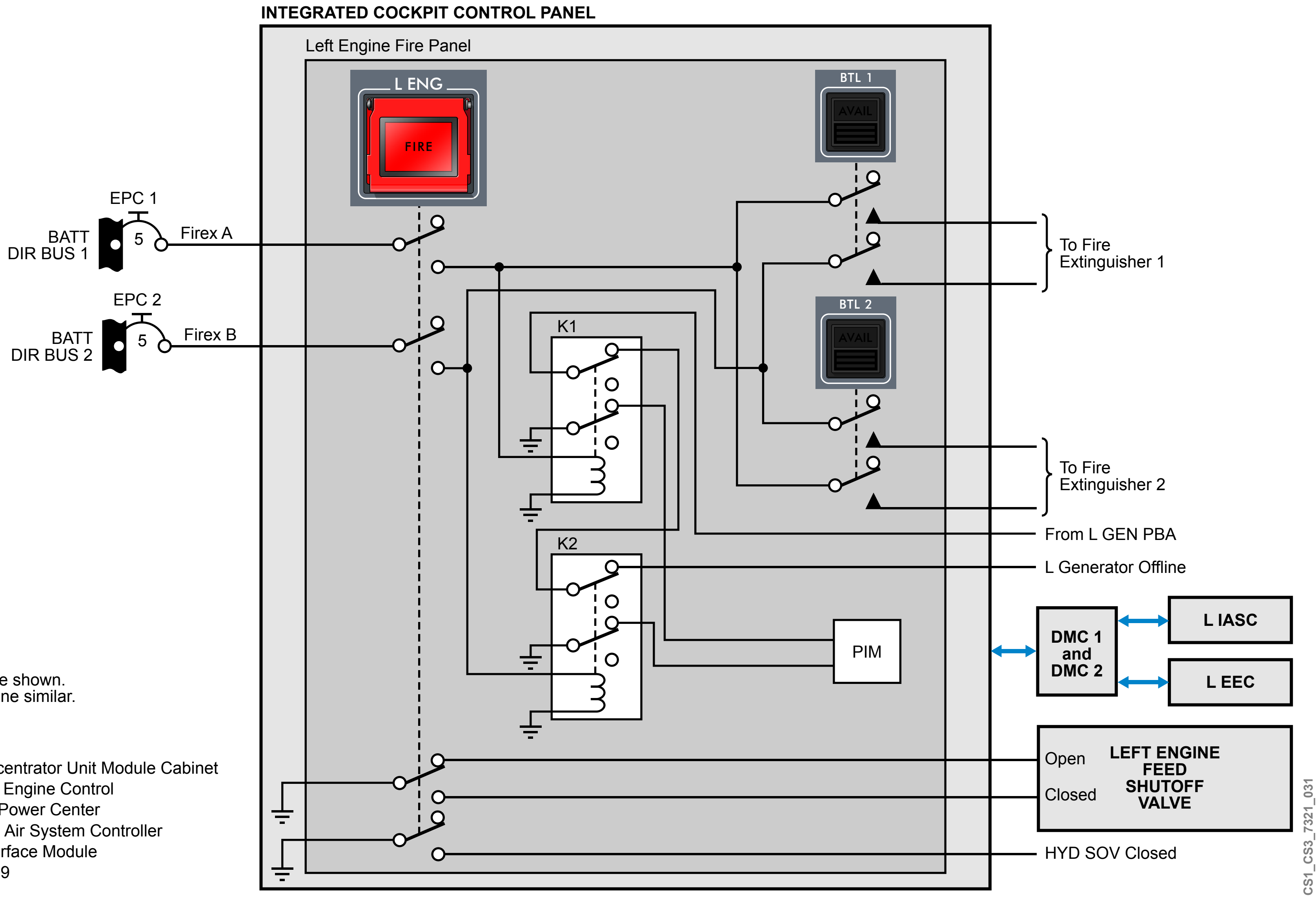
- Engine feed shutoff valve closes
- Pressure-regulating shutoff valve (PRSOV) closes
- Fan air valve closes
- Hydraulic shutoff valve closes
- Generator trips offline
- EEC commands engine shutdown
- Both engine fire extinguisher bottles are armed

The integrated air system controller (IASC) and electronic engine control (EEC) receive digital inputs from the FIRE PBA via an ARINC 429 BUS.

The fan air valve and pressure-regulating shutoff valves are closed by the IASC.

The EEC provides a shutdown command to the engine.

The engine feed shutoff valve is closed through switch contacts in the FIRE PBA.



NOTE

Left engine shown.
Right engine similar.

LEGEND

DMC	Data Concentrator Unit Module Cabinet
EEC	Electronic Engine Control
EPC	Electrical Power Center
IASC	Integrated Air System Controller
PIM	Panel Interface Module
↔	ARINC 429

Figure 28: Emergency Shutdown

ENGINE POWER MANAGEMENT

Engine Bleed Management

Each EEC receives information from the integrated air system controllers (IASCs). IASC 1 provides information to the left EEC. IASC 2 provides information to the right EEC.

- High-pressure valve (HPV) open or closed status
- Environmental control system (ECS) configuration
- Anti-ice configuration:
  - Wing anti-ice status
  - Nacelle anti-ice status

Idle Speed

The EEC selects engine idle speed based on engine requirements and bleed demand. The minimum idle speed is based on the following priorities:

- Minimum fuel flow for combustor stability
- Minimum pressure to provide the required aircraft bleed flow, and maintaining the minimum pressure to not exceed a 20% core flow bleed limit
- Minimum N1 to achieve acceleration time requirements
- Minimum N2 speed to maintain the PMAG power generation

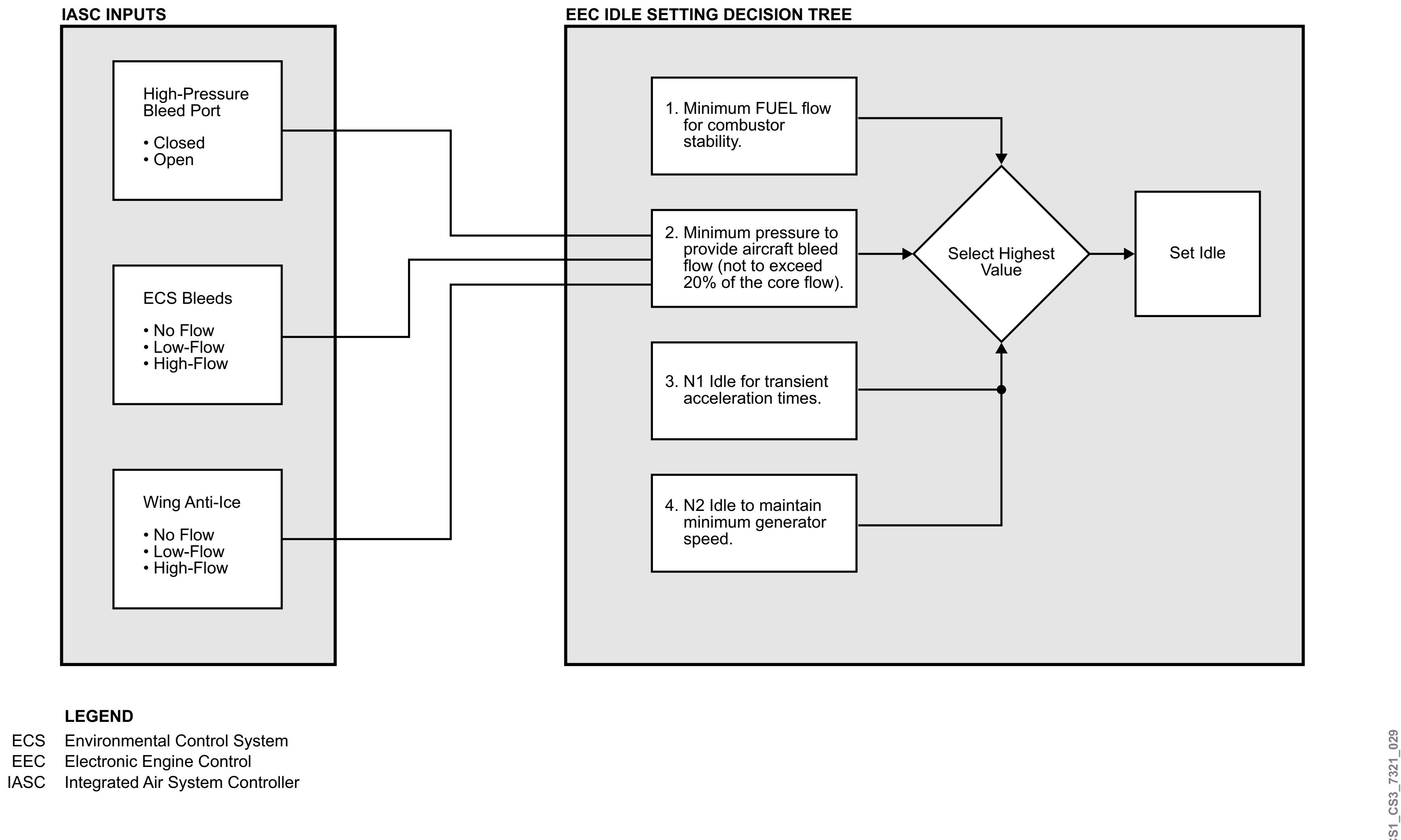


Figure 29: Engine Bleed Management and Idle Speed

ENGINE POWER SETTINGS AND MODES

Using flight plan performance data from the flight management system (FMS), environmental data, and engine sensors, the EEC calculates the N1 target for maximum takeoff and go-around, maximum continuous thrust, derated takeoff, reduced takeoff (FLEX), and climb thrust.

Maximum Thrust

The EEC sets the maximum thrust available at the MAX thrust position to maximum takeoff thrust (MTO) when the aircraft is operating in the takeoff or go around envelope. Outside the takeoff envelope or go around envelope, the EEC schedules maximum continuous thrust (MCT). The MCT rating supports one engine inoperative (OEI) conditions and is designed to be lower or equal to MTO thrust and higher than or equal to climb thrust (CLB).

Mode Selection

The thrust lever is adjusted automatically to the selected thrust mode rating when the autothrottle is engaged. Thrust mode changes are managed automatically based on the FMS active flight plan.

The thrust mode can be controlled manually on the FMS PERF page DEP tab using the THRUST button. A window pops up to enable manual selection.

Derated Takeoff

Derated takeoffs TO-1, TO-2, or TO-3 are used to reduce fuel consumption, noise, and to extend engine life and reliability. The derated TO thrust selection is computed using the mode selection on the FMS DEP page. The takeoff derate is limited to a minimum of 17,000 pounds of thrust.

FLEX Thrust

FLEX thrust is computed using the assumed temperature method, based on the value entered in the FMS. On the ground, when the EEC confirms the assumed temperature input is higher than the actual temperature, the FLEX reduction in thrust is applied for that takeoff. FLEX is available from any selected takeoff thrust mode.

Climb Thrust

The FMS calculates the optimal climb (CLB) thrust and N1 reference based on the takeoff thrust mode (derated, FLEX), and the outside air temperature (OAT). Derated climb CLB-1 or CLB-2 allow the thrust settings for the climb envelope of the flight to be reduced by the FMS. There are two levels of climb derate. Each subsequent level reduces the thrust during the climb.

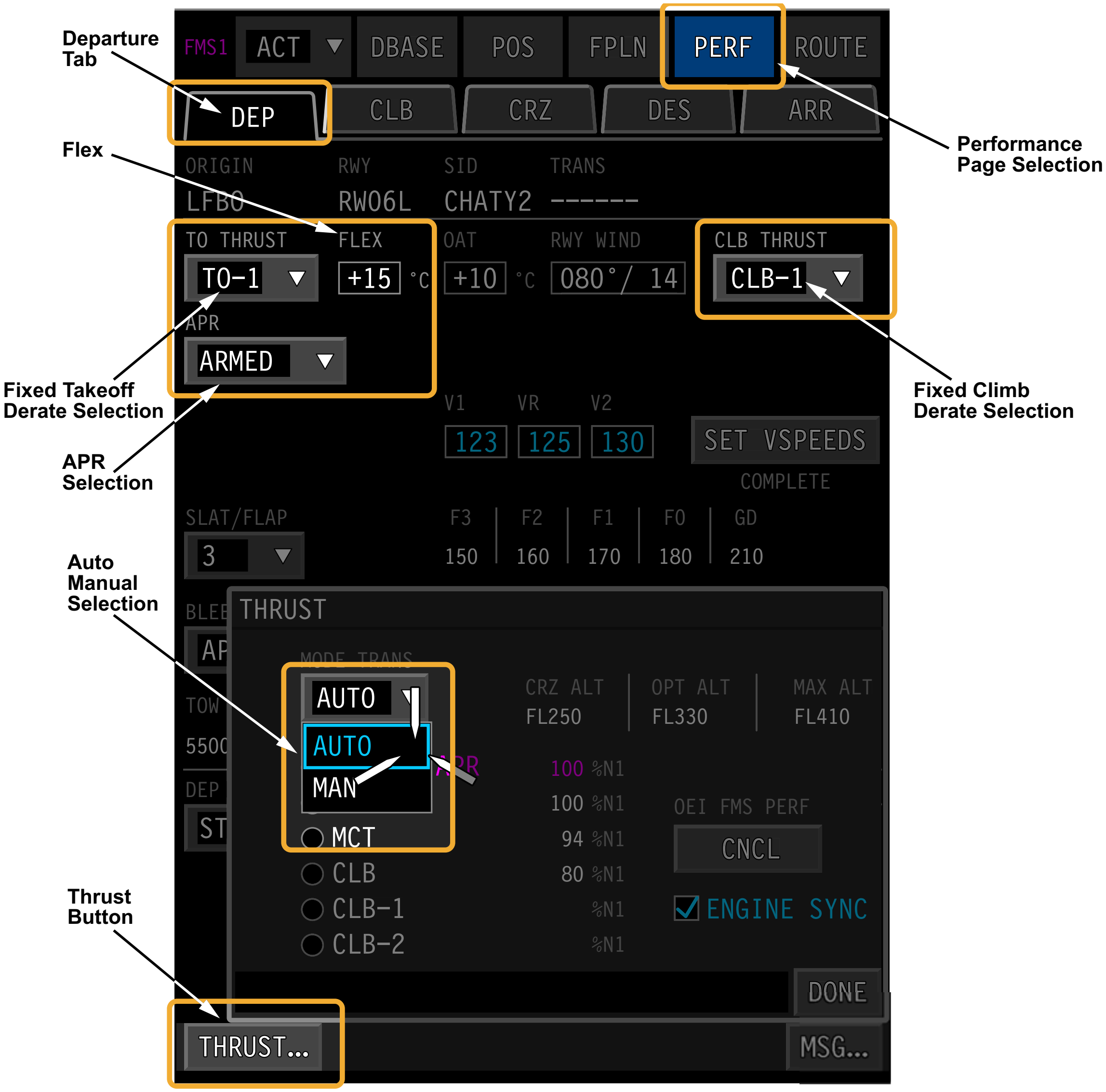
Go-Around

Go-around thrust is set as a thrust reference when the aircraft is on approach with the slats and flaps extended and the landing gear down. Go-around thrust is equivalent to MTO. Go-around thrust is initiated when the aircraft is in flight and the takeoff/go-around (TOGA) switches on the thrust levers are pressed.

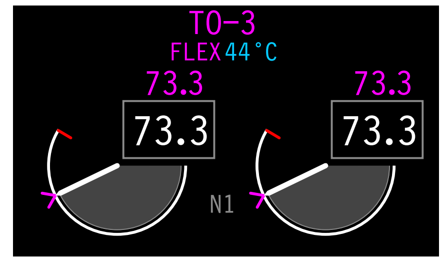
Idle Speed

The EEC determines the idle setting based on engine stability requirements, bleed demand, and phase of flight. The minimum idle speed setting minimizes the aircraft landing distance and brake wear during taxi. Flight idle is used in flight to minimize fuel consumption during descent. When the landing gear is selected down or the flaps are extended, the engine idle speed increases to the approach idle setting to allow a faster engine response during a go-around.

During a steep approach, a lower than normal approach idle setting is used. The steep approach idle is adjusted for bleed air demands and go-around thrust requirements.



FMS PERFORMANCE PAGE



EICAS

Symbol	Description
>	FMS managed thrust reference value.
>	Manually entered thrust reference value.
TO-3 FLEX 44°C 73.3 73.3	Thrust mode. Assumed temperature for FLEX XX°C. Digital thrust reference.
TO or TO TO-1 or TO-1 TO-2 or TO-2 TO-3 or TO-3 CLB or CLB CLB-1 or CLB-1 CLB-2 or CLB-2 MCT or MCT GA or GA FLEX 44°C 73.3 73.3 FLEX 44°C 73.3 73.3	Magenta: FMS managed Cyan: manual selection

CS1_CS3_7321_036

Figure 30: Engine Power Settings and Modes

AUTOMATIC POWER RESERVE

The automatic power reserve (APR) automatically provides a higher thrust setting to the operating engine should one of the engines fail during a derated takeoff. APR provides improved single-engine acceleration and climb gradient performance.

APR is only available when a derated takeoff thrust reference is selected. APR is not available for any other thrust mode. Should one engine fail, the EEC automatically increases the thrust on the operative engine by canceling the engine fixed derate by one level and removing any FLEX entry. When the thrust increase is commanded, the thrust levers angle (TLA) does not change.

If TO-1 derate is selected, the engine thrust increases to MTO. If TO-2 or TO-3 are selected, the thrust is commanded to a level below MTO in order to maintain aircraft controllability. MTO is available by advancing the thrust levers to MAX.

When armed, APR activates when both thrust levers are advanced beyond 23° and one engine experiences a loss of power greater than 15% N1.

The APR can be manually disarmed from the FMS performance page DEP tab. When APR is manually disarmed selected, an APR DISARM status message is displayed.

Scenario 1

The engines are derated to TO-1 prior to takeoff. This cancels one fixed derate level and maximum thrust is commanded on the operative engine.

Scenario 2

The engines are derated to TO-1 and FLEX is selected. When the EEC detects one engine has lost thrust, the derate and FLEX are canceled. Maximum thrust is commanded on the operative engine.

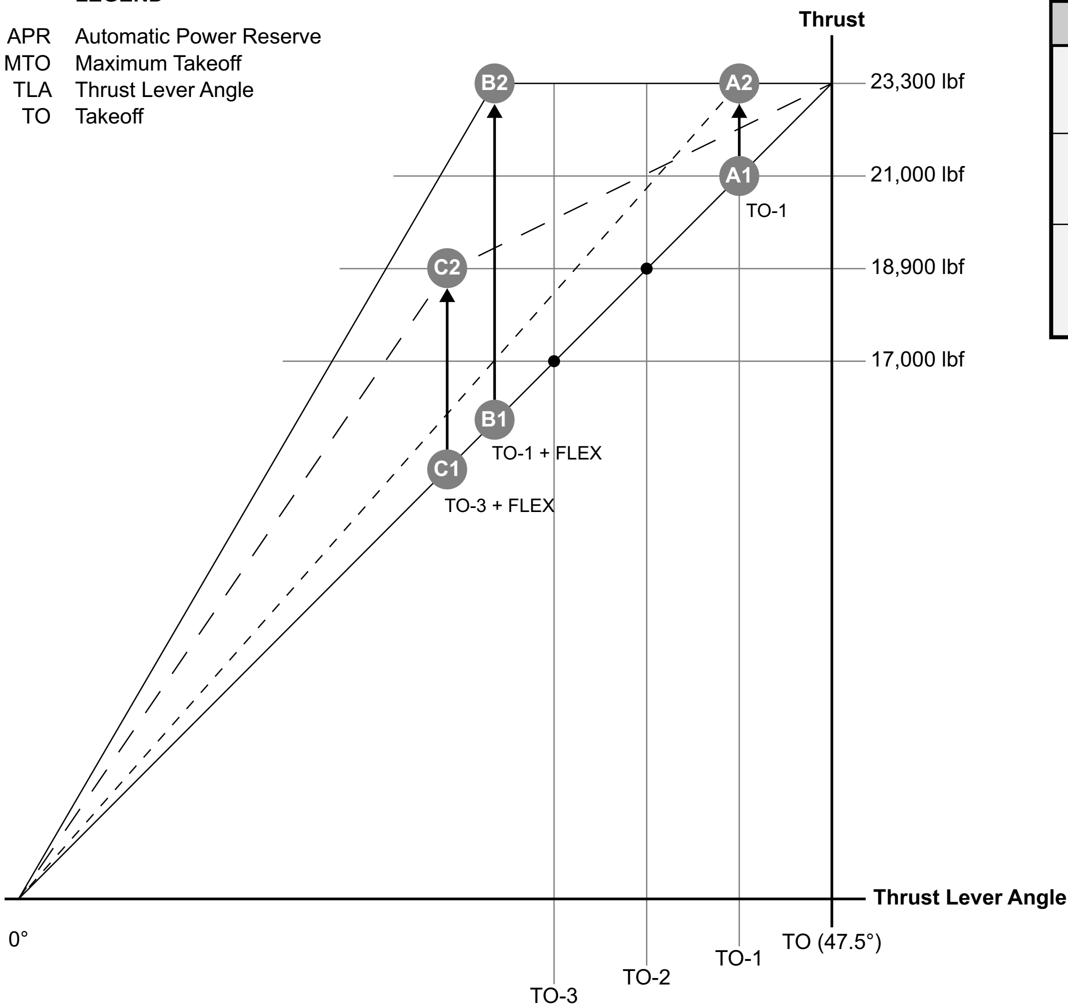
Scenario 3

The engines are derated to TO-3 and FLEX is selected. When the EEC detects one engine has lost thrust, the derate changes to TO-2 and FLEX is canceled. The operative engine is commanded to supply TO-2 thrust.

Additional thrust is available by advancing the thrust levers. The EEC recalculates the TLA versus N1 command relationship to ensure the full range of thrust is available from the engine.

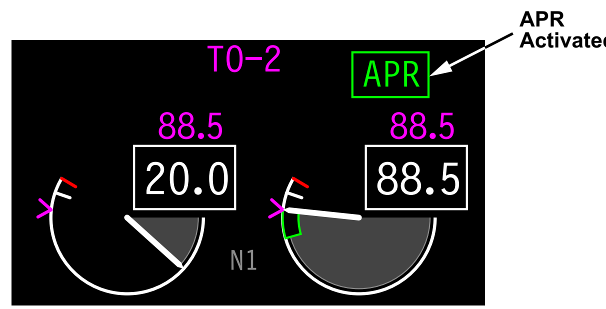
LEGEND

- APR Automatic Power Reserve
- MTO Maximum Takeoff
- TLA Thrust Lever Angle
- TO Takeoff

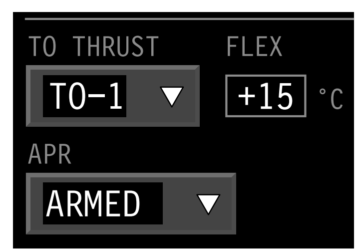


THRUST LEVER ANGLE VERSUS APR RELATIONSHIP

SCENARIOS		
Scenario 1	A1 to A2	The derate is canceled and thrust is increased to MTO.
Scenario 2	B1 to B2	Both the derate and the FLEX are canceled, thrust is increased to MTO.
Scenario 3	C1 to C2	The derate is reduced to TO-2 and FLEX are canceled. Engine thrust automatically increases to TO-2. Additional thrust is available by advancing the thrust levers.



EICAS PAGE



FMS PERFORMANCE PAGE

CS1_CS3_7321_030

Figure 31: Automatic Power Reserve

N1 SYNCHRONIZATION

The N1 synchronization (N1 SYNC) function is used to synchronize the engine speeds to reduce cabin noise. The N1 SYNC function is selectable through the aircraft flight management system (FMS) and is armed by default.

The N1 SYNC function maintains the difference between the sensed N1 values of both engines to less than 0.2% during steady state conditions. N1 SYNC is enabled by the EEC, provided specific engine operating conditions are satisfied.

The left engine is the master engine for the N1 SYNC function. The N1 command for the slave engine is biased to match that of the master engine.

With the N1 SYNC mode set in the FMS, the EEC activates N1 SYNC if the following conditions are met:

- Both engines are operating in a steady-state
- The aircraft is not in takeoff, or go-around
- The difference between the two engines N1 commands is less than 1.5% N1
- Throttle lever angle (TLA) is greater than 6°
- No opposite engine low-thrust condition or automatic power reserve selection is detected

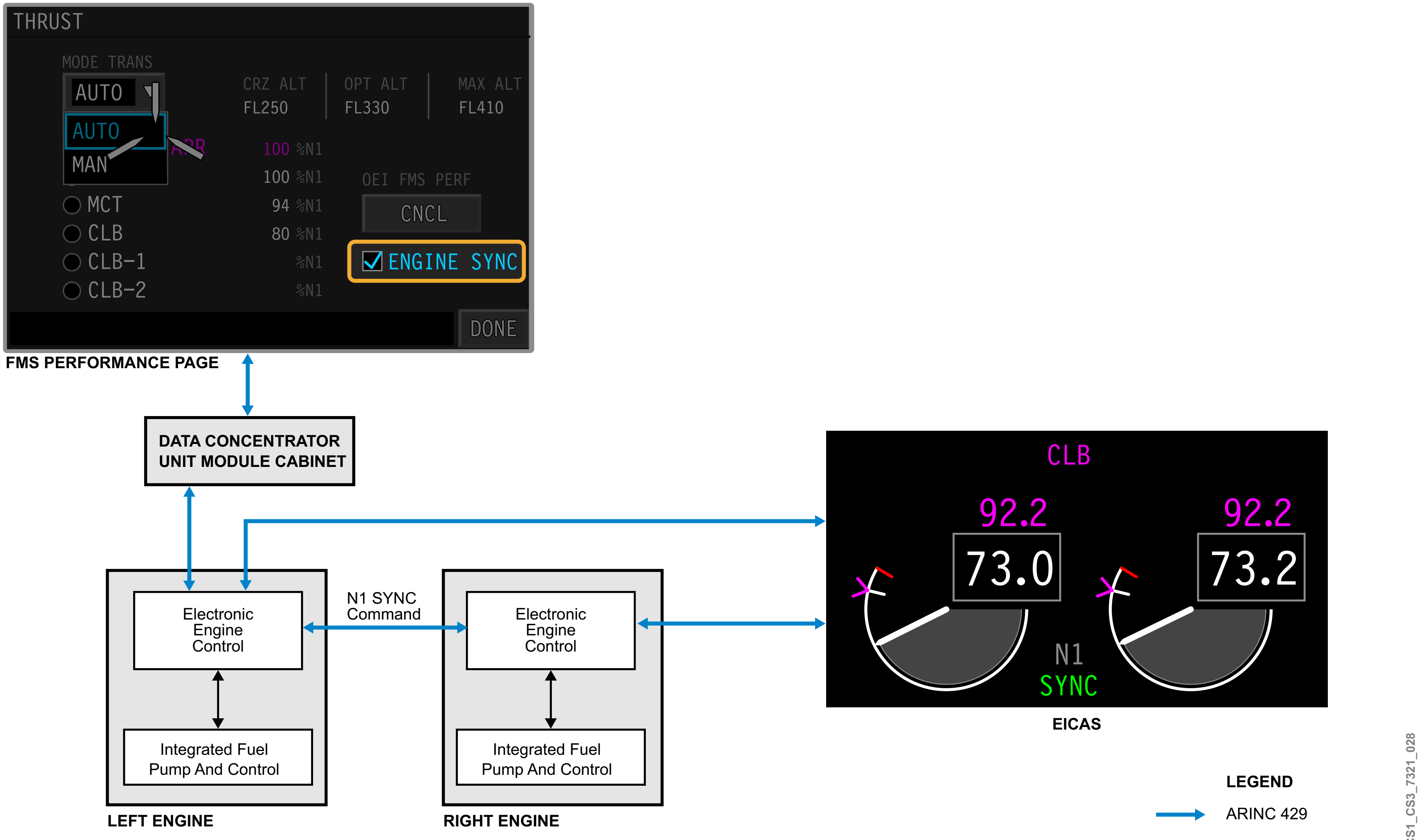


Figure 32: N1 Synchronization

FAULT REPORTING AND TESTING INTERFACE

Each EEC channel has a non-volatile memory (NVM) that can store snapshots and transient records of parameters. Each EEC channel records its own faults or events that are detected. Old data is overwritten by new data when the memory is full.

The PHMU transmits data to the health management unit (HMU) via the EEC. The EEC sends the data via an ARINC 429 bus for distribution to the health management unit (HMU) and the onboard maintenance system (OMS). The EEC also provides information to generate CAS and INFO messages on EICAS and the INFO synoptic pages. The CAS and INFO messages are used to determine the capability to dispatch the aircraft.

The health management unit is capable of recording the following:

- Fault data such as failures detected within FADEC system components or its interfaces with the aircraft such as sensor and actuator failures
- Event data consisting of abnormal operation such as surges and exhaust gas temperature exceedances
- Health trending data consisting of parameters collected on every flight to track engine performance over time. The data is used for trend monitoring and can provide early detection of an engine failure. The data can also be used to generate reports such as takeoff reports and stable cruise reports

The OMS records faults annunciated by the EEC. The OMS also provides the capability of testing EEC functions and displaying real-time data that support line maintenance operations. (see ATA 45).

Dispatch

The EEC monitors all faults and classifies them according to the operability of the engine. Faults are classified as follows:

- Time limited dispatch
    - No dispatch
    - Short time dispatch
    - Long time dispatch
- Master minimum equipment list (MMEL) faults

Time limited dispatch faults include:

- Failures directly used for thrust control
- Malfunctions resulting in an auto-idle or auto-shutdown condition
- Malfunctions resulting in a thrust lever moved to idle or an engine shutdown in response to a CAS message

No dispatch faults are major failures that prevent operation or control of the engine. A no dispatch fault has an L(R) ENGINE FAULT advisory message and a 73 L (R) ENGINE FAULT - FADEC FAULT 1 INFO message.

Short time dispatch faults usually relate to a single channel failure that can be compensated for by alternate means on the engine. Short time dispatch faults are limited to 125 flight hours. A short time fault has an L(R) ENGINE FAULT advisory message and a 73 L (R) ENGINE FAULT - FADEC FAULT 2 INFO message.

Long time dispatch faults usually relate to a failure that does not affect the operability of the engine such as a sensor dedicated to engine performance monitoring. Long time dispatch faults are limited to 425 flight hours. A long time fault has an L(R) ENGINE FAULT advisory message and a 73 L (R) ENGINE FAULT - FADEC FAULT 3 INFO message.

MMEL faults are normally related to secondary systems or indication system not essential for engine operation. MMEL faults are generally associated with caution messages.

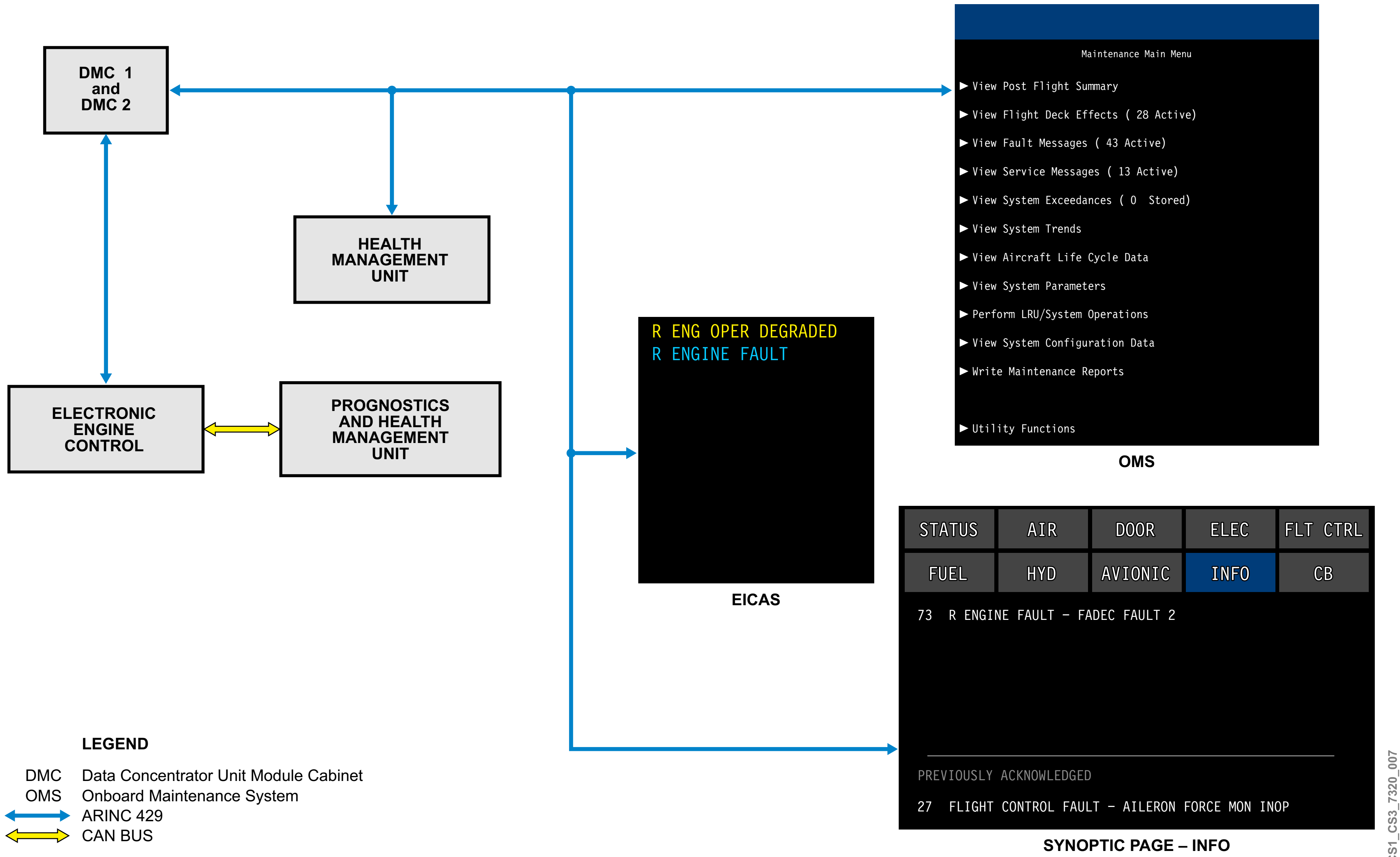


Figure 33: Electronic Engine Control Fault Reporting and Testing Interface

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the engine controls system.

CAS MESSAGES

Table 1: CAUTION Messages

MESSAGE	LOGIC
L (R) ENG EXCEEDANCE	Left (R) N1 or left (R) N2 or left ITT or left oil temperature above threshold.
L (R) ENG OPER DEGRADED	Uncertainty on T2/TAT (degraded or default value), or HPC stator vane actuator not tracking as it should.
L (R) ENG FAIL	L 9R) engine Sub-idle.
ENG SETTINGMISMATCH	Thrust management data received or fed back by FADEC not matching the APR setting, N1 target, etc.

Table 2: ADVISORY Messages

MESSAGE	LOGIC
L (R) ENGINE FAULT	Loss of redundant or non-critical function for the left engine.

Table 3: STATUS Messages

MESSAGE	LOGIC
APR DISARMED	APR is disarmed in the FMS and a derated takeoff is being used.

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Table 4: INFO Messages

MESSAGE	LOGIC
73 L (R) ENGINE FAULT - EEC A CTRL CPU INOP	EEC channel A control processor has failed.
73 L (R) ENGINE FAULT - EEC B CTRL CPU INOP	EEC channel B control processor has failed.
73 L (R) ENGINE FAULT - PT/T2 HEATER INOP	FADEC logic has determined that the P2/T2 heater has a logical failure and is indicated as failed.
73 L (R) ENGINE FAULT - FADEC FAULT 1	Collector bit for all non-dispatchable FADEC faults.
73 L (R) ENGINE FAULT - FADEC FAULT 2	Collector bit for all FADEC short term TLD faults.
73 L (R) ENGINE FAULT - FADEC FAULT 3	Collector bit for all FADEC long term TLD faults.
77 L (R) ENGINE FAULT - PHMU INOP	FADEC logic has determined that the PHMU is failed.
73 L (R) ENGINE FAULT - EEC 28VDC REDUND LOSS	Set when only one EEC channel of the left (right) engine has detected loss of 28 VDC input voltage.
73 L (R) ENGINE FAULT - HEALTH MON DEGRADED	Set by the left (right) engine when the P2.5 or T2.5 sensor is failed.
73 L (R) ENGINE FAULT - T3 SNSR INOP	Set by the left (right) engine when the T3 sensor is failed. This INFO message will be masked if the EEC Channel B control processor is failed.

Table 4: INFO Messages

MESSAGE	LOGIC
73 L (R) ENGINE FAULT - FUEL EQUAL INOP	Set by the left (right) engine in any of the following conditions: Both EEC channels are unable to command fuel equalization solenoid. Any channel control processor failed and remaining channel unable to command fuel equalization solenoid.

PRACTICAL ASPECTS

ONBOARD MAINTENANCE SYSTEM

The aircraft onboard maintenance system (OMS) uses the electronic engine control (EEC) to perform functional checks of specific systems. Once the new EEC is installed, the EEC SYSTEM TEST must be carried out. This test is performed through the OMS using one channel at a time. Each system test should be checked through both channels.

The following EEC tests can be performed through the aircraft OMS:

- Actuator Test
- EEC System Test
- HPD Test
- Harness Test
- Ignition Test
- P2 Probe Heat Test
- PCE Door Test
- Thrust Reverser Cycling Test

System Interactive Test

The system interactive test procedure is used to check for satisfactory operation of the EEC system before an engine start.

This procedure is used to verify the presence of existing faults, and to confirm that corrective action has eliminated the fault. The EEC system interactive test electrically verifies the EEC, and prognostics and health management unit (PHMU) input and output circuits.

NOTE

The system interactive test does not test igniters, thrust reverser, or hydraulic pumps. Separate interactive tests are provided for these systems.

Harness Interactive Test

The harness interactive test checks for harness and connector failures. During the test suspected harnesses must be moved to help reproduce the fault messages reported by the EEC.

This procedure is used to perform harness checks in order to isolate intermittent EEC monitored faults.

Probe Heat Test

The P2/T2 probe heat test verifies the operation of the probe heater circuit.

The EEC turns the probe heat on, and after 60 seconds verifies a rise in T2 temperature. The EEC then turns the probe heat off.

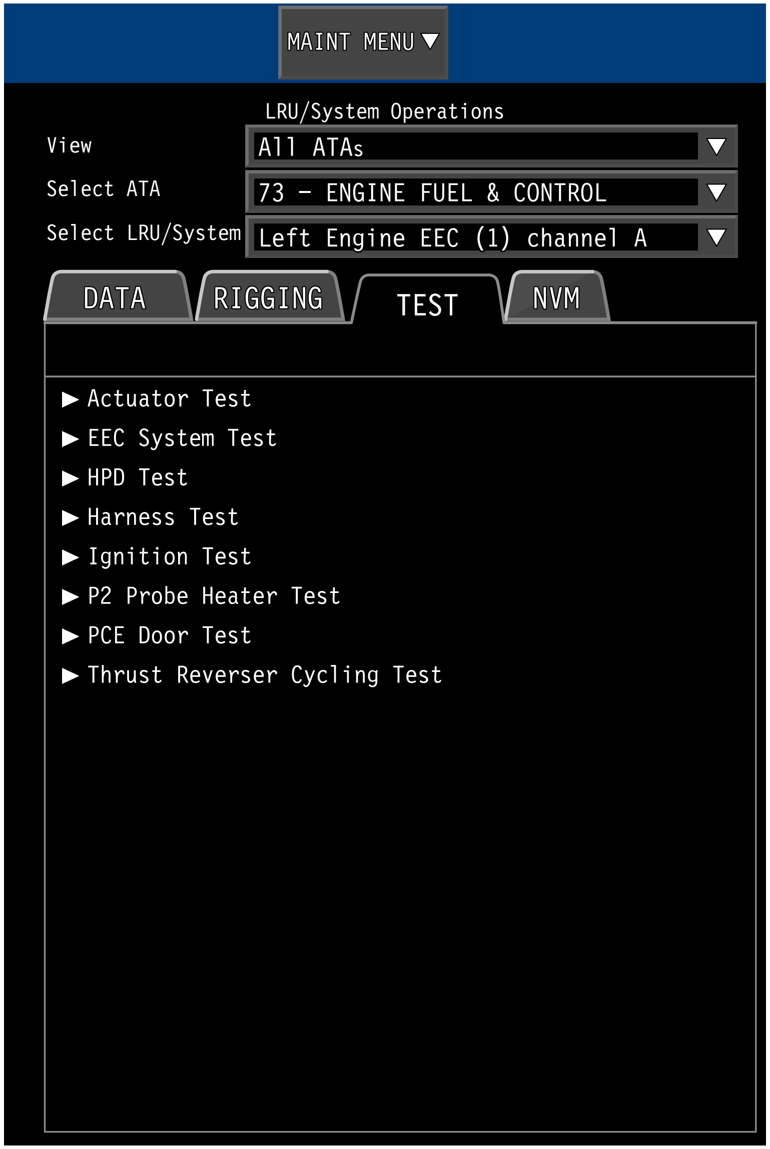


Figure 34: OMS EEC System Test

HMU Power Assurance Test

The power assurance test can be performed using the ATA 46 HMU CH A or CH B selection on the OMS LRU/System Operations page.

Using ambient pressure, temperature, N1 errors, and nacelle anti-ice (NAI) status, the HMU software calculates and corrects the N2, EGT, and fuel flow parameter range.

During the engine run, the HMU compares the actual engine parameters against the corrected parameter ranges. If all of the actual parameters are all within the corrected parameter ranges, a TEST PASSED message is displayed. A TEST FAIL message is shown if any parameter is out of range.

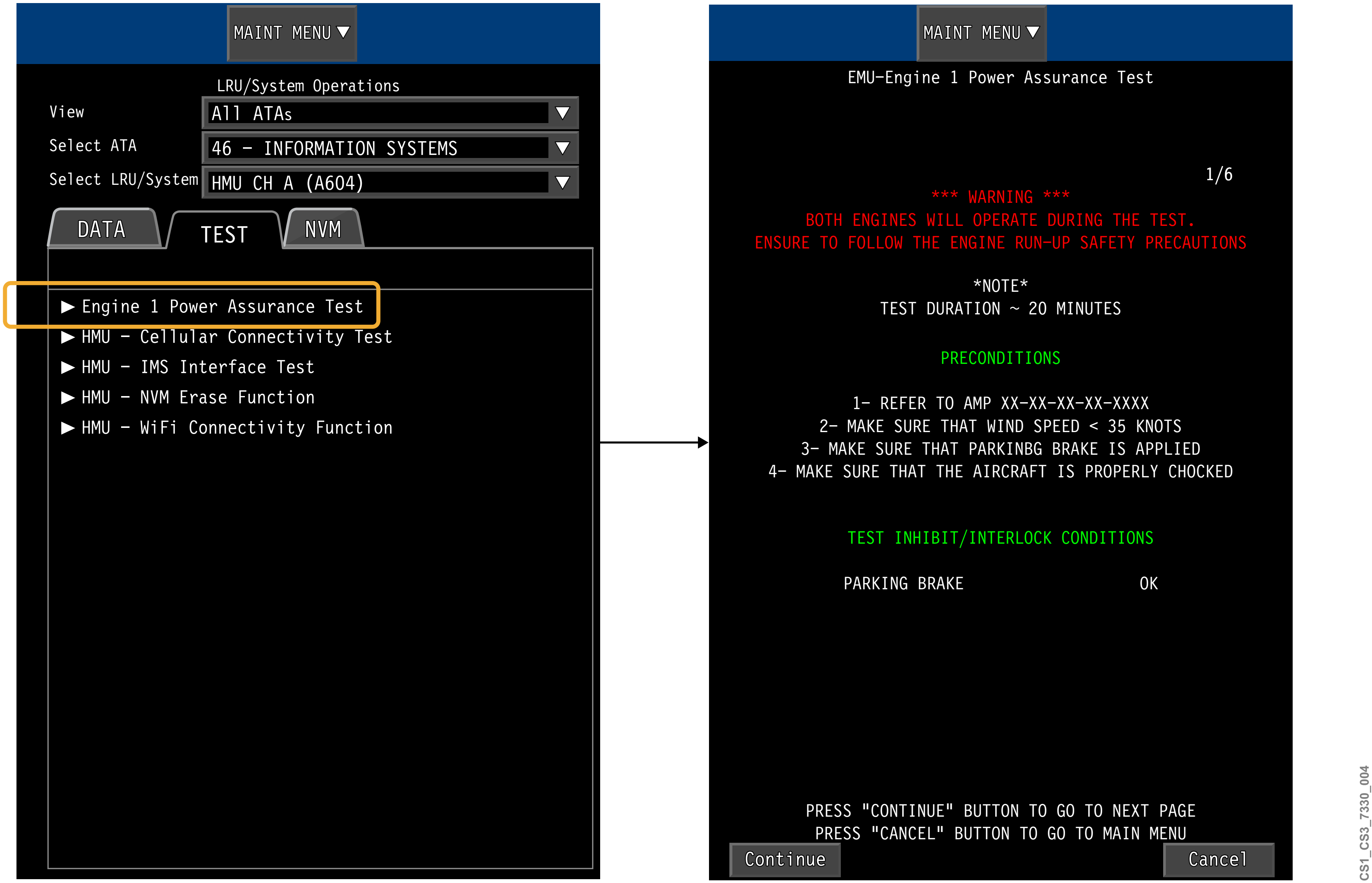


Figure 35: Power Assurance Test

73-30 INDICATING

GENERAL DESCRIPTION

The fuel indicating system consists of a fuel flow meter to monitor fuel flow and a fuel filter differential pressure sensor to provide an indication of a blocked fuel filter.

The fuel flow transmitter measures metered fuel for combustion. The EEC calculates the fuel flow for display on the engine indication and crew alerting system (EICAS) page.

The fuel filter differential pressure sensor measures differential pressure across the fuel filter. The fuel filter status is displayed on the FUEL synoptic page.

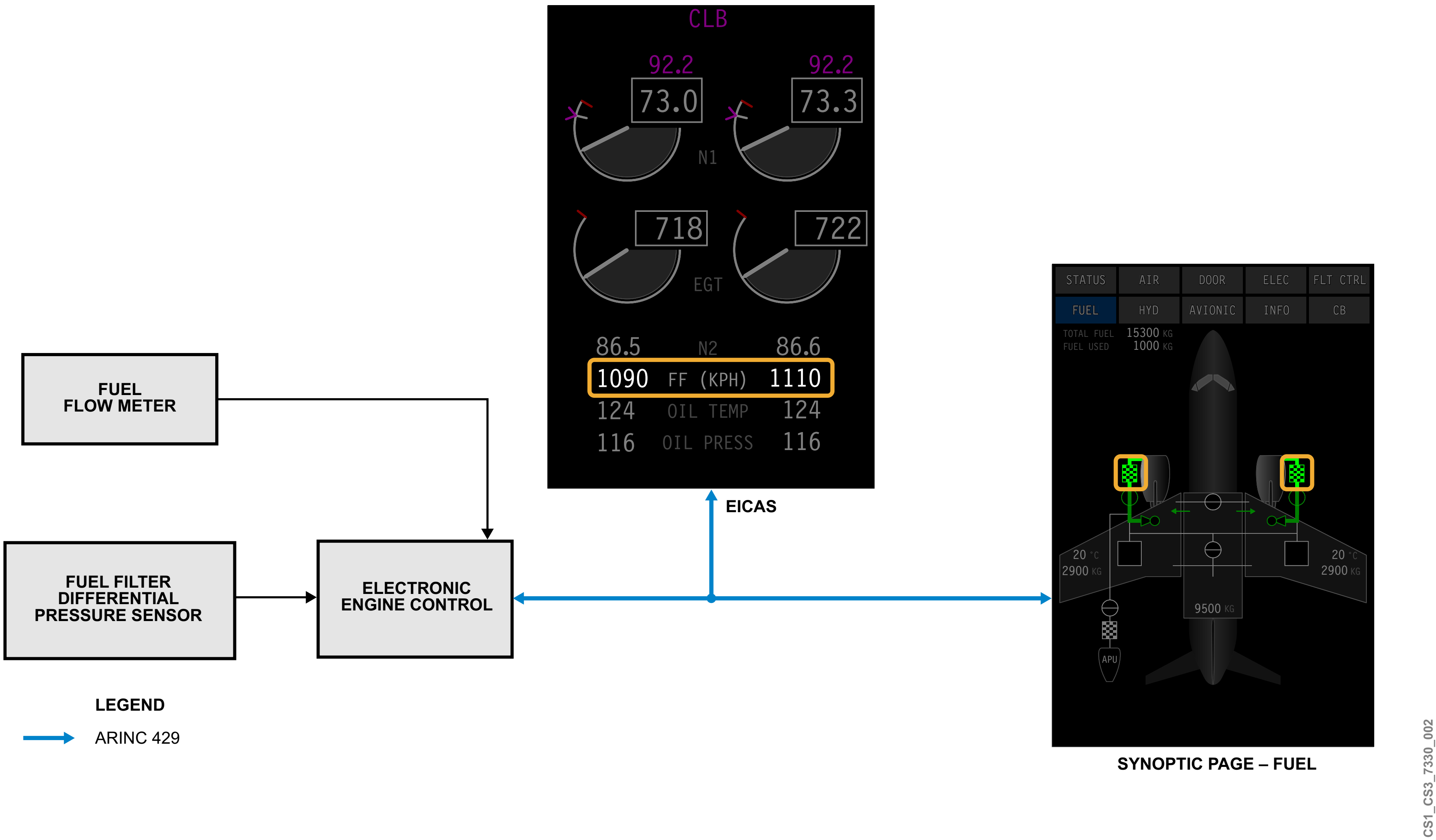


Figure 36: Fuel Control Indicating System

COMPONENT LOCATION

The engine fuel indicating system consists of the following components:

- Fuel flow meter
- Fuel filter differential pressure sensor

Fuel Flow Meter

The fuel flow meter is located above the integrated fuel pump and control (IFPC) on the fuel/oil manifold at the 9 o’clock position.

Fuel Filter Differential Pressure Sensor

The fuel filter differential pressure sensor is mounted on the forward side of the fuel/oil manifold adjacent to the main fuel filter at the 7:30 o’clock position.

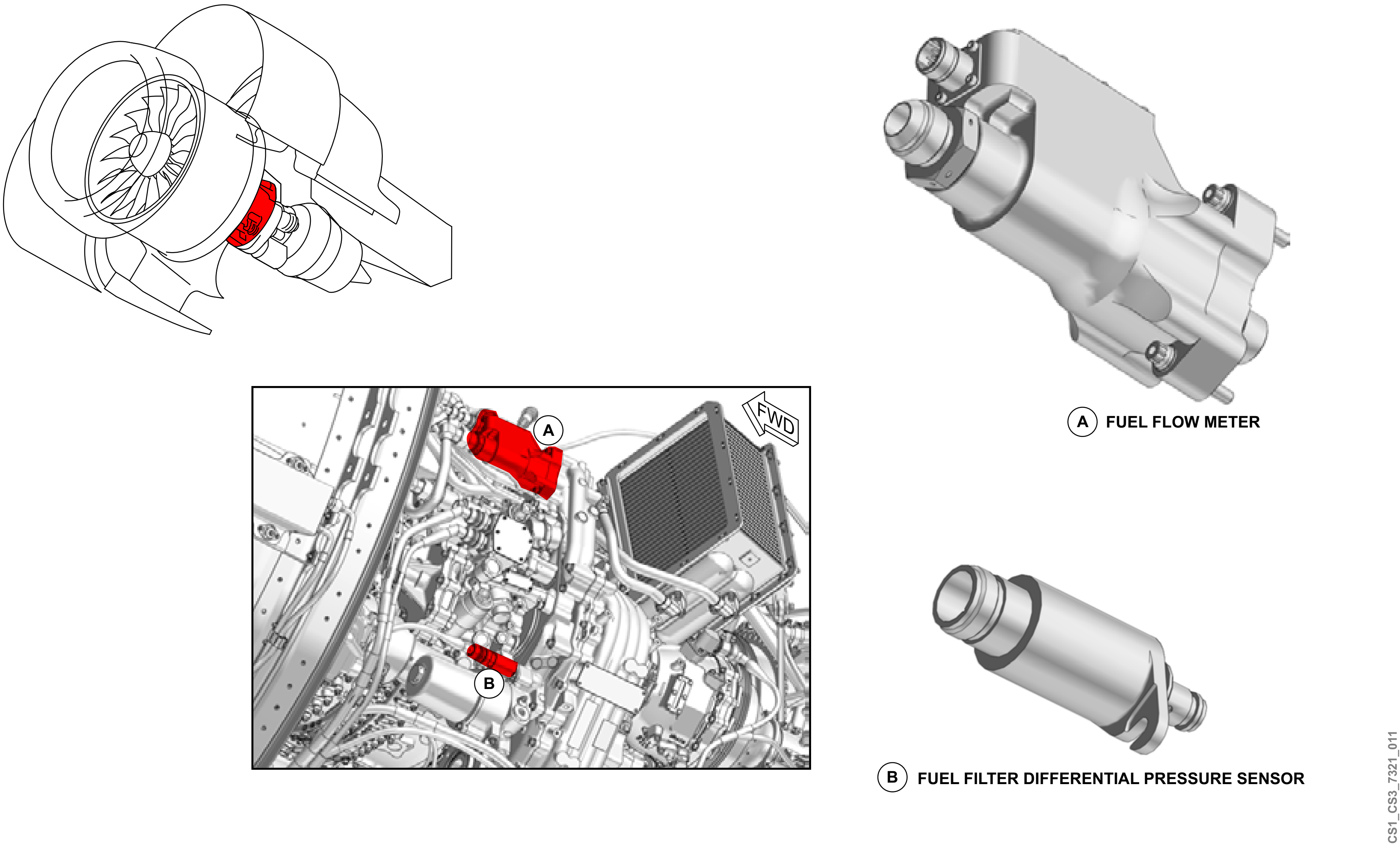
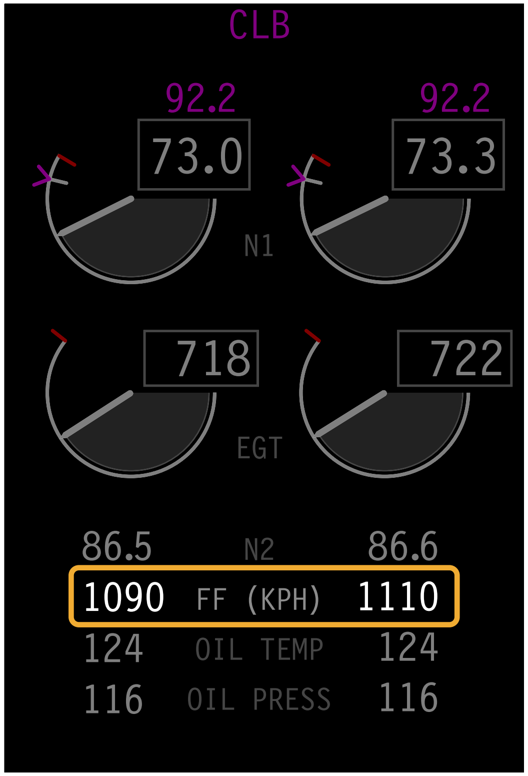


Figure 37: Fuel Control Indicating System Component Location

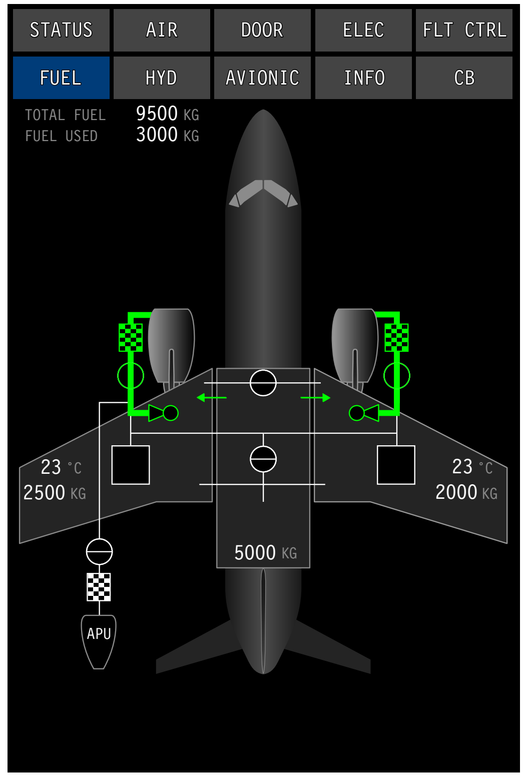
CONTROLS AND INDICATIONS

The fuel flow information is displayed on the EICAS page.

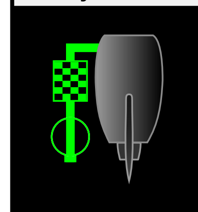
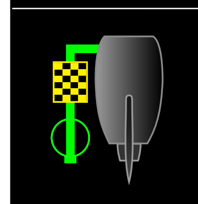
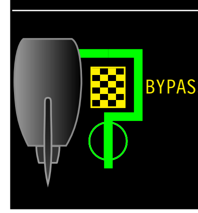
The fuel filter bypass status is indicated on the FUEL synoptic page.



EICAS



SYNOPTIC PAGE – FUEL

FUEL STATUS	
Symbol	**Condition**
	Normal
	Fuel Filter Impending Bypass
	Fuel Filter Bypass

CS1_CS3_7322_003

Figure 38: Fuel Control Indications

DETAILED DESCRIPTION

The fuel flow meter sends a signal to channel A of the EEC where it is used to calculate the metered fuel flow. The signal is supplied to channel B internally via a CAN BUS.

The fuel filter differential pressure sensor provides an output to each channel of the EEC.

The fuel filter differential pressure sensor measures differential pressure across the fuel filter. A fuel filter impending bypass is detected and displayed when the differential pressure exceeds 22 psi on either engine. An L(R) ENG FUEL FILTER advisory message along with an L (R) ENG FUEL FILTER - IMPENDING BYPASS info message.

A fuel filter bypass occurs at 25 psid. This is indicated by the L(R) ENG FUEL FILTER advisory message and an L (R) ENG FUEL FILTER - BYPASS info message.

If both fuel filters are bypassed, fuel contamination can be suspected and an L-R ENG FUEL FILTER caution message is displayed.

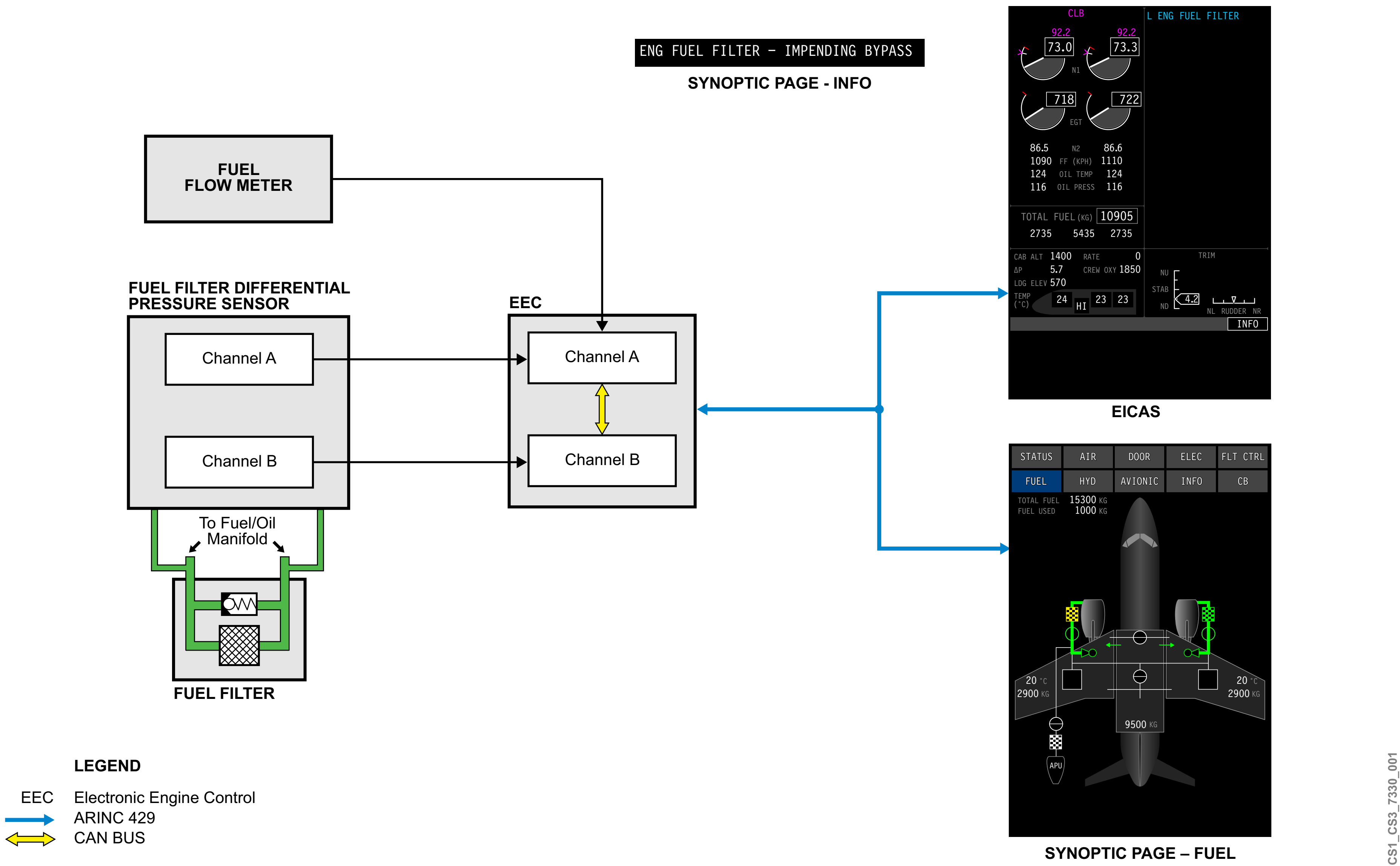


Figure 39: Fuel Control Indicating Detailed Description

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the engine fuel indication.

CAS MESSAGES

Table 5: CAUTION Messages

MESSAGE	LOGIC
L - R ENG FUEL FILTER	Both engine fuel filters are bypassed (clogged filters), or impending, i.e. likely fuel contamination.

Table 6: ADVISORY Messages

MESSAGE	LOGIC
L ENG FUEL FILTER	Left engine fuel filter is impending bypass or is bypassed.
R ENG FUEL FILTER	Right engine fuel filter is impending bypass or is bypassed.
L ENGINE FAULT	Loss of redundant or non-critical function for the left engine.
R ENGINE FAULT	Loss of redundant or non-critical function for the right engine.
L FUEL FLOW DEGRADED	FADEC provides synthesized fuel flow instead of measured fuel flow. Displayed L fuel flow accuracy is degraded.
R FUEL FLOW DEGRADED	FADEC provides synthesized fuel flow instead of measured fuel flow. Displayed R fuel flow accuracy is degraded.

Table 7: INFO Messages

MESSAGE	LOGIC
73 L (R) ENGINE FAULT - FUEL FILTER PRESS SNSR INOP	FADEC logic has determined that the fuel filter delta P sensor signal from channel A and/or B has failed.
73 L (R) ENG FUEL FILTER - IMPENDING BYPASS	INFO message will be set by the left (right) engine when the fuel filter impending bypass is detected but there is no actual bypass.
73 L (R) ENG FUEL FILTER - BYPASS	INFO message will be set by the left (right) engine when the fuel filter actual bypass is detected.

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ATA 74-80 - Ignition and Starting

BD-500-1A10
BD-500-1A11

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IGNITION and STARTING - CHAPTER BREAKDOWN

74-11 IGNITION

GENERAL DESCRIPTION

The engine ignition system consists of two electrically and physically independent circuits located in a single ignition exciter. Each exciter circuit is an AC-powered capacitor discharge type that supplies high voltage to an igniter plug through a shielded, high tension cable.

Each exciter circuit is powered from the aircraft. The left engine ignition power is supplied by DC ESS BUS 1 and DC ESS BUS 3. Right engine ignition power is supplied by DC ESS BUS 2 and 3. The electronic engine control (EEC) provides the control of the ignition.

The ignition system is used during engine start or engine relight. The ignition system is also used any time ambient conditions require a continuous ignition to prevent the risk of flameout.

During a normal automatic engine start, the EEC uses a single igniter plug to start the engine. The igniter plug selected alternates on each normal automatic engine start. The EEC commands the igniter plugs on or turns the igniter plugs off based on N2 speed.

The ignition system can also be operated in a dual-igniter low power continuous ignition mode. The continuous ignition mode operates automatically under the control of the EEC, or manually when selected from the CONT IGNITION PBA on the ENGINE panel.

When the CONT IGNITION PBA is selected before engine start, the EEC enters the manual start mode and both igniter plugs are selected for engine start. In the manual start mode, the ignition is commanded on when the THROTTLE QUADRANT ASSEMBLY L(R) ENG switch is selected to the ON position, and turned off when the engine self-sustaining speed is reached.

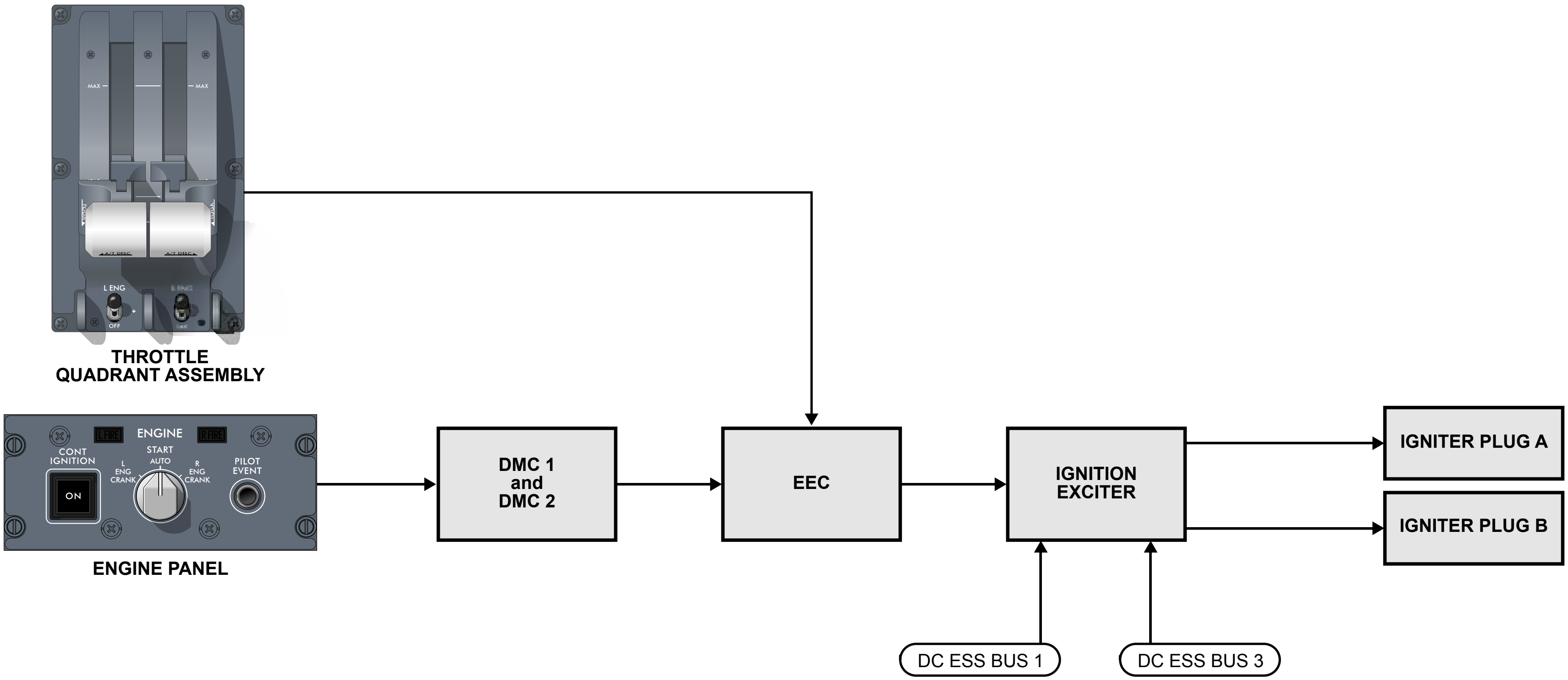
When continuous ignition is selected and the engine is already running, the igniter plugs only fire when the engine is operating at low power settings. This increases the life of the igniter plugs.

Automatic selection of continuous dual-ignition is provided by the EEC during critical or adverse conditions. The EEC also includes an automatic relight system, which energizes both igniters within 2 seconds when an engine flameout is detected.

The EEC automatically selects a dual-igniter continuous ignition if any of the following conditions are true:

- Engine flameout is detected in-flight or during takeoff
- Surge is detected in-flight or during takeoff
- In-flight start

The automatic selection of continuous ignition occurs even if manual continuous ignition is selected.



LEGEND

DMC	Data Concentrator Unit Module Cabinet
EEC	Electronic Engine Control

NOTE

Left engine shown. Right engine similar.

Right engine ignition exciter
power DC ESS BUS 2 and DC ESS BUS 3

CS1_CS3_7400_002

Figure 1: Ignition System

COMPONENT LOCATION

The ignition system includes the following components:

- Ignition exciter
- Ignition cables
- Igniter plugs

IGNITION EXCITER

The ignition exciter is located on the left side of the engine fan case at the 8 o’clock position.

IGNITER PLUGS

The igniter plugs are installed in the diffuser case. Igniter A is located at the 4 o’clock position, and igniter B is located at the 5 o’clock position.

IGNITION CABLES

The ignition cables extend under the engine from the ignition exciter to the igniter plugs.

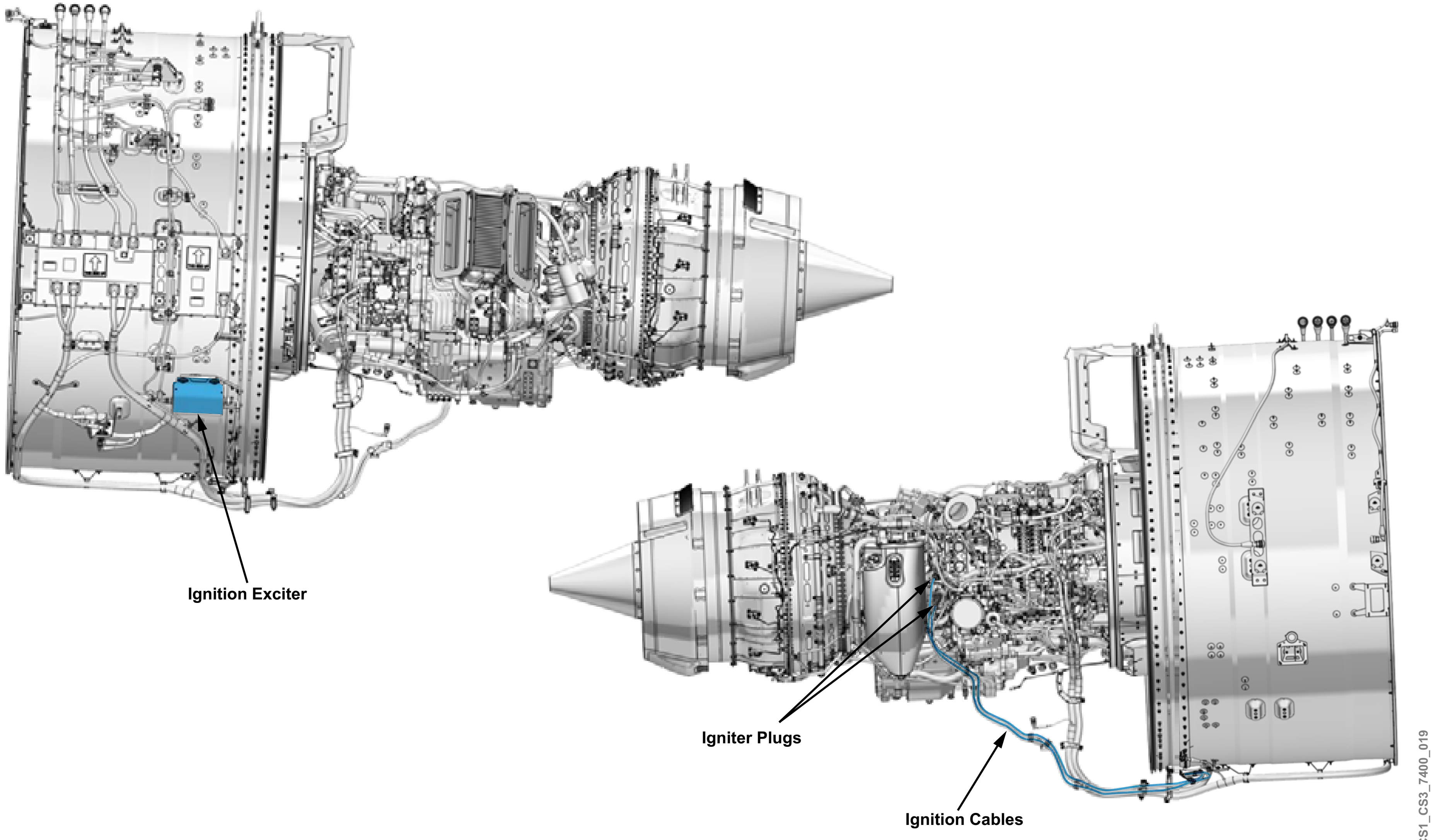


Figure 2: Ignition System Component Location

COMPONENT INFORMATION

IGNITION EXCITER

The dual-channel ignition exciter supplies 5 kV to the igniter plugs.

The ignition exciter has a duty cycle that varies between 1 spark per second to 3 sparks per second.

The ignition exciter is cooled by the fan compartment ventilation air.

WARNING

MAKE SURE THE IGNITION SYSTEM HAS NOT BEEN OPERATED FOR AT LEAST 5 MINUTES PRIOR TO REMOVING THE IGNITER PLUGS OR THEIR CABLES.

IGNITION VOLTAGE CAN BE VERY HIGH AND COULD CAUSE INJURY.

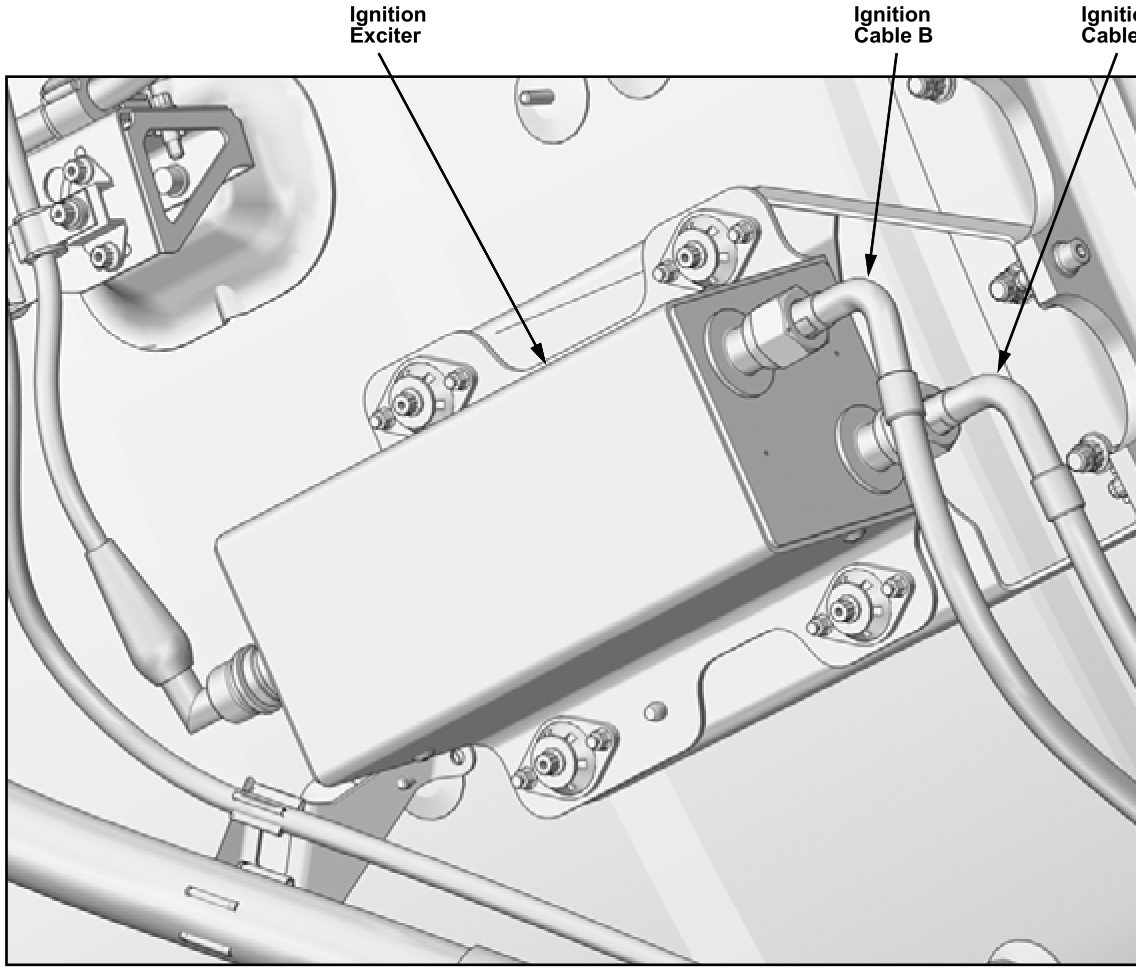


Figure 3: Ignition Exciter

IGNITION CABLES

Ignition cables are interchangeable with flexible braided steel and ceramic insulated terminals at the plug end.

IGNITER PLUGS

The igniter plugs extend through the diffuser case into the combustion chamber. Classified spacers are installed under a mounting boss and are used to control the immersion depth of the igniter plug. The spacer and boss are installed on the diffuser case assembly and are not removed during igniter plug replacement.

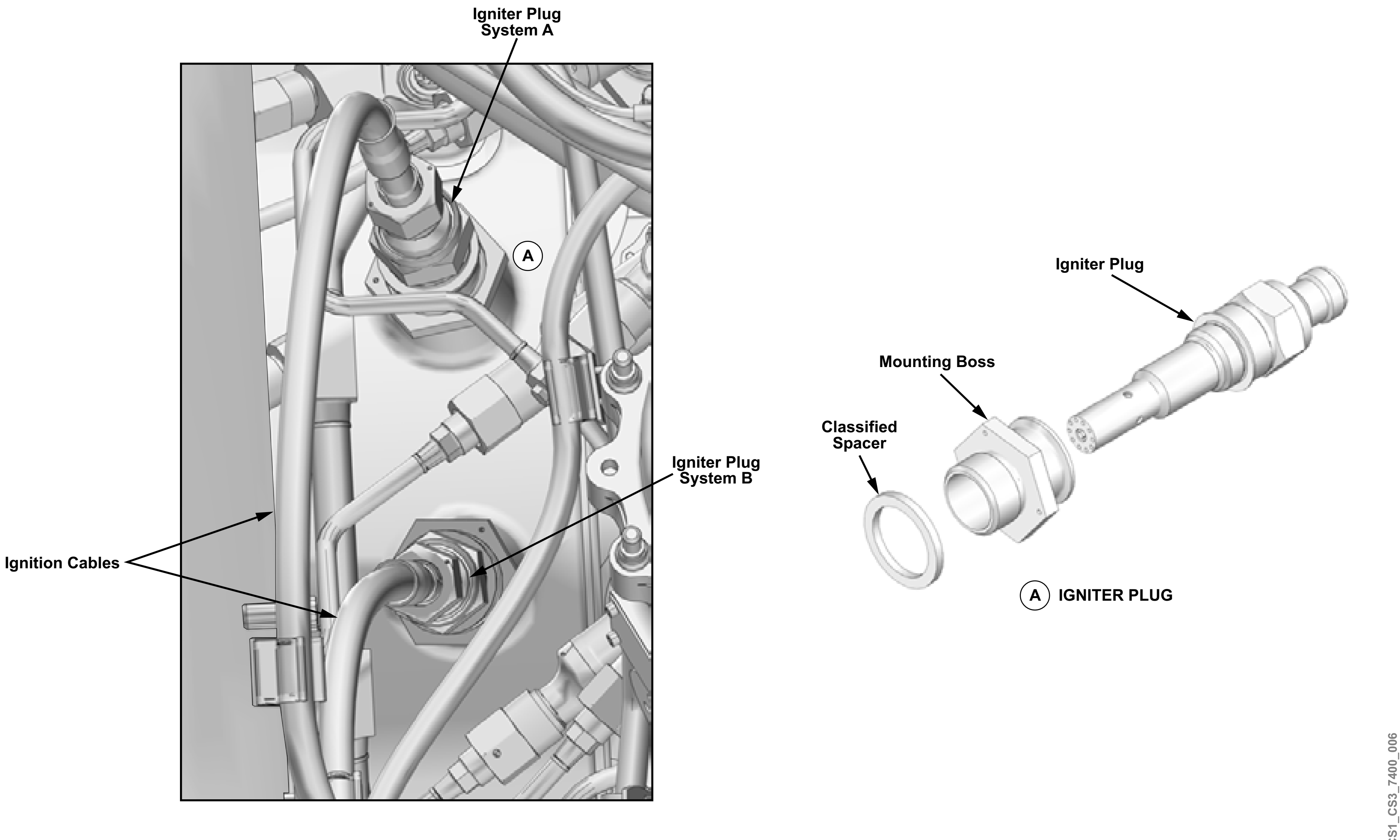


Figure 4: Ignition Cables and Igniter Plugs

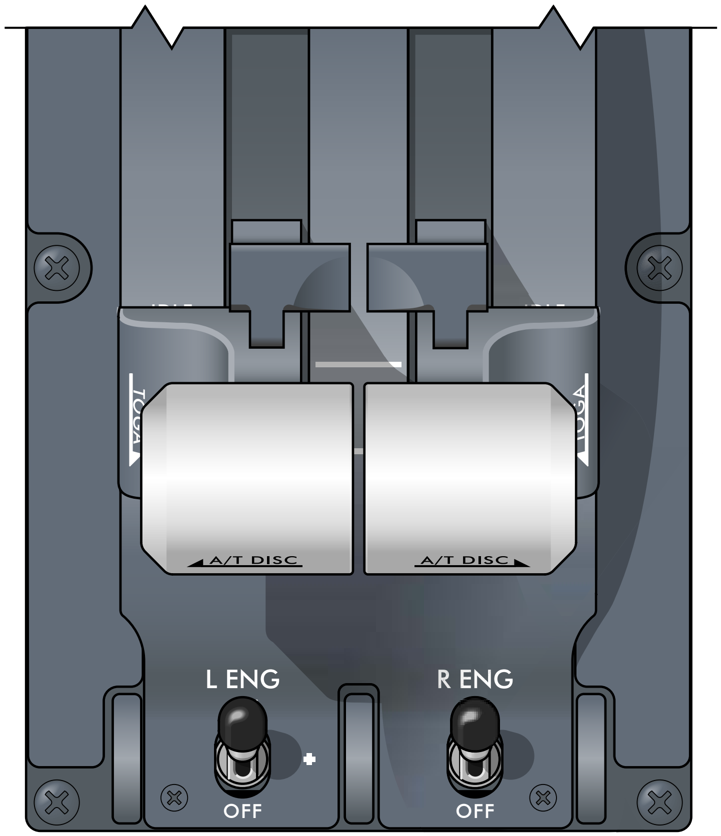
CONTROLS AND INDICATIONS

ENGINE PANEL

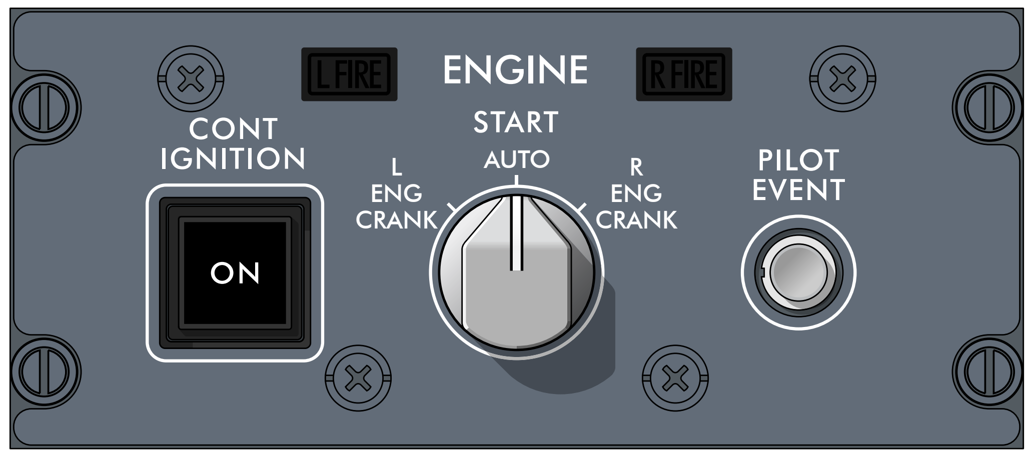
The CONT IGNITION PBA is used to select manual ignition. The manual selection is indicated by the white ON legend on the pushbutton annunciator (PBA).

THROTTLE QUADRANT ASSEMBLY

The L(R) ENG switch signals the EEC to command the ignition ON during a manual engine start when they are selected to the ON position.



THROTTLE QUADRANT ASSEMBLY



ENGINE PANEL

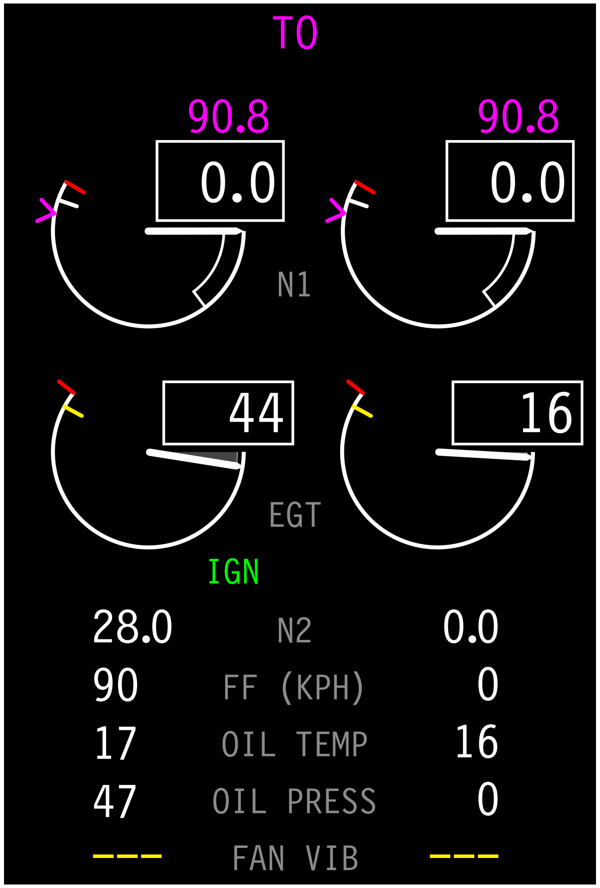
CS1_CS3_7400_007

Figure 5: Ignition System Controls

INDICATIONS

An IGN icon is displayed below the exhaust gas temperature (EGT) gauges when ignition is selected. The IGN icon is green when the ignition system is commanded and the igniters are firing.

The IGN icon is white when continuous ignition is selected ON, but inhibited.



EICAS PAGE

SYMBOL	COLOR	DESCRIPTION
IGN	Green	Ignition active (Firing).
IGN	White	Ignition selected but inhibited due to design constraints.

CS1_CS3_7400_020

Figure 6: Ignition System Indications

DETAILED DESCRIPTION

NORMAL IGNITION FOR AUTOMATIC START

The EEC channel A controls ignition system A, and EEC channel B controls ignition system B. A single igniter plug is used during an automatic start.

During normal starts, the igniter plugs are commanded on at 20.4% to 23.5% N2. When the N2 reaches 49.3 to 53.3%, the EEC commands the igniter plugs off. A green IGN icon is shown on EICAS anytime the EEC activates the igniters. The icon is removed when the igniter plugs are not on.

If the engine fails to start on the first attempt, the EEC dry motors the engine, and then selects both igniter plugs for a second start attempt.

CONTINUOUS IGNITION

After the engine has started, normal engine operation does not require the igniter plugs to sustain combustion. However, under certain conditions continuous ignition is available for use.

Continuous ignition has two modes:

- Manual
- Automatic

Manual Continuous Ignition

Manual continuous ignition is available when the CONT IGNITION PBA is pressed. An ON light illuminates on the PBA to indicate manual continuous ignition is available.

During a manual engine start, continuous ignition is used and both igniter plugs fire once the L(R) ENG switches are selected to ON. The EEC commands the igniter plugs off when the N2 reaches 55%. A green IGN icon appears as long as the igniters are firing.

When the engine is running, manual continuous ignition can be selected by cycling the CONT IGNITION PBA to NORMAL and back to ON.

When the engine is running and manual continuous ignition is selected, a white IGN icon is displayed on the engine indication and crew alerting system (EICAS). In this condition, the igniter plugs may not be firing unless the engine is at low power. The EEC monitors the burner pressure (Pb). If Pb is greater than 150 psi, the ignition is commanded off. Once the engine is running at approach idle or less, Pb is low and the ignition is commanded on.

Automatic Continuous Ignition

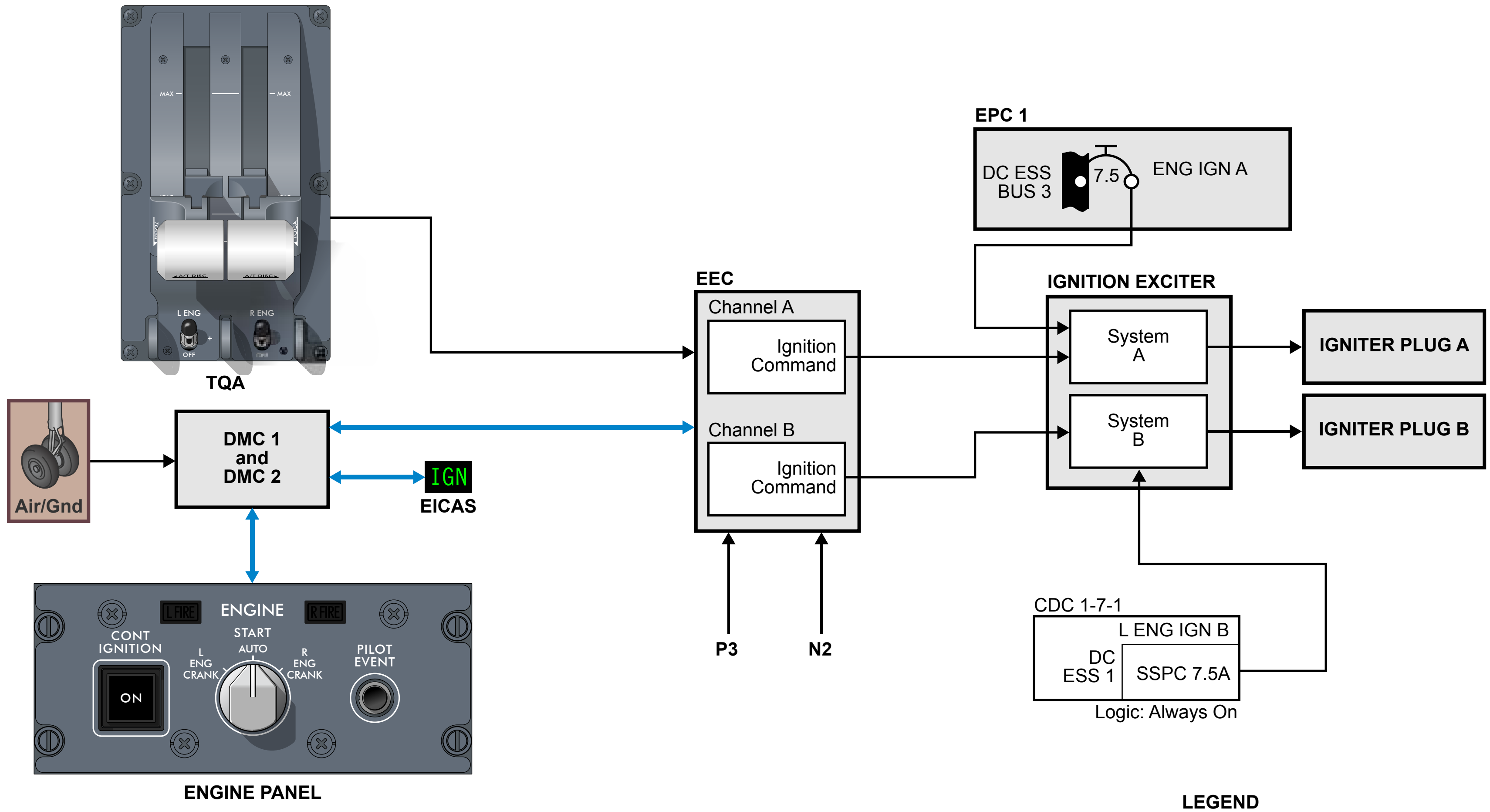
The EEC automatically selects dual-igniter plugs continuous ignition under any of the following conditions:

- Engine flameout detected in-flight or during takeoff above 60 kt
- An engine surge is detected in air
- In-flight start

Quick-Relight Continuous Ignition

The engine performs a quick-relight in-flight only when the L(R) ENG switch is cycled from ON to OFF, and back to the ON position.

When the quick-relight condition occurs, the EEC commands continuous ignition until either the engine has been above idle for 30 seconds, or the L(R) ENG switch is placed in the OFF position.



NOTE

Left engine shown, right engine similar.

LEGEND

CDC	Control and Distribution Cabinet
DMC	Data Concentrator Unit Module Cabinet
EEC	Electronic Engine Control
EPC	Electrical Power Center
P3	Burner Pressure
TQA	Throttle Quadrant Assembly
(blue arrow)	ARINC 429

Figure 7: Ignition System Schematic

Ignition and Starting Modes

The START switch located on the ENGINE panel is in the AUTO position for normal engine starts. The L (R) ENG CRANK position is used for manual engine starts, dry motoring, and wet motoring.

The CONT IGNITION PBA is used to initiate a manual engine start or select both igniters on.

The L (R) ENG switches initiate an automatic start when the START switch is in AUTO. During a manual engine start, moving the L (R) ENG to ON initiates fuel and ignition. When the switch is moved to the OFF position, the engine shuts down.

During an automatic engine start, only one igniter is used. The EEC alternates igniters on consecutive starts.

During a manual engine start on the ground or an engine start in the air, both igniters are on.

When the engine is dry or wet motored, the igniters are off.

In addition to the ignition selection for engine starting, the EEC automatically selects dual-igniter ignition under any of the following conditions:

- In-flight engine start
- Engine flameout detected in-flight or during takeoff above 60 kt
- An engine surge is detected in air

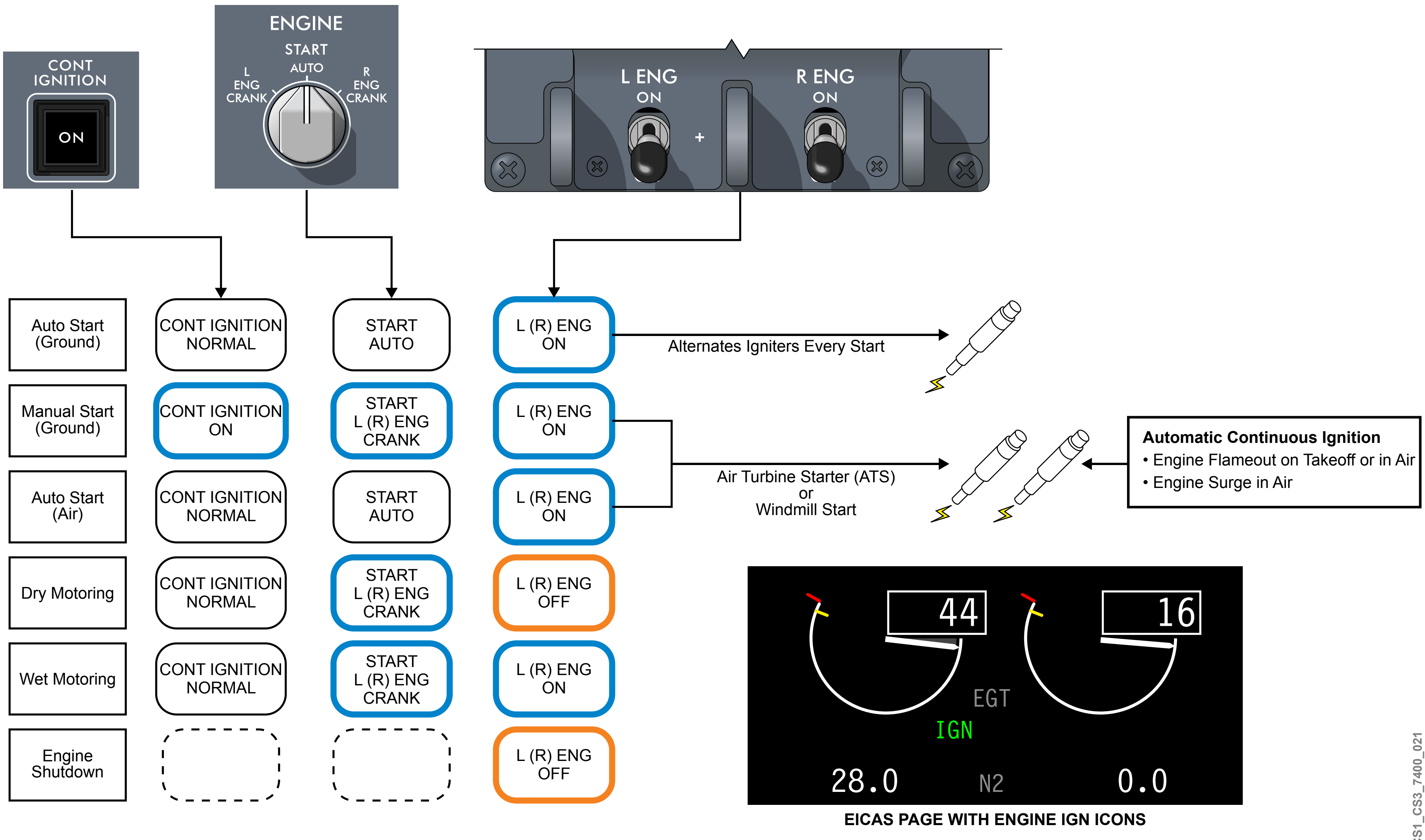


Figure 8: Ignition and Starting Modes

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the starting and ignition system.

CAS MESSAGES

Table 1: STATUS Message

MESSAGE	LOGIC
ENG CONT IGNITION ON	At least one EEC receives a request from the flight deck switch to have continuous ignition.

Table 2: INFO Message

MESSAGE	LOGIC
74 L(R) ENGINE FAULT - IGN REDUND LOSS	An ignition circuit fault, or failure to light using an igniter, or a low side voltage fault has been detected by the left/right full authority digital engine control (FADEC) system.

PRACTICAL ASPECTS

IGNITION SYSTEM TEST

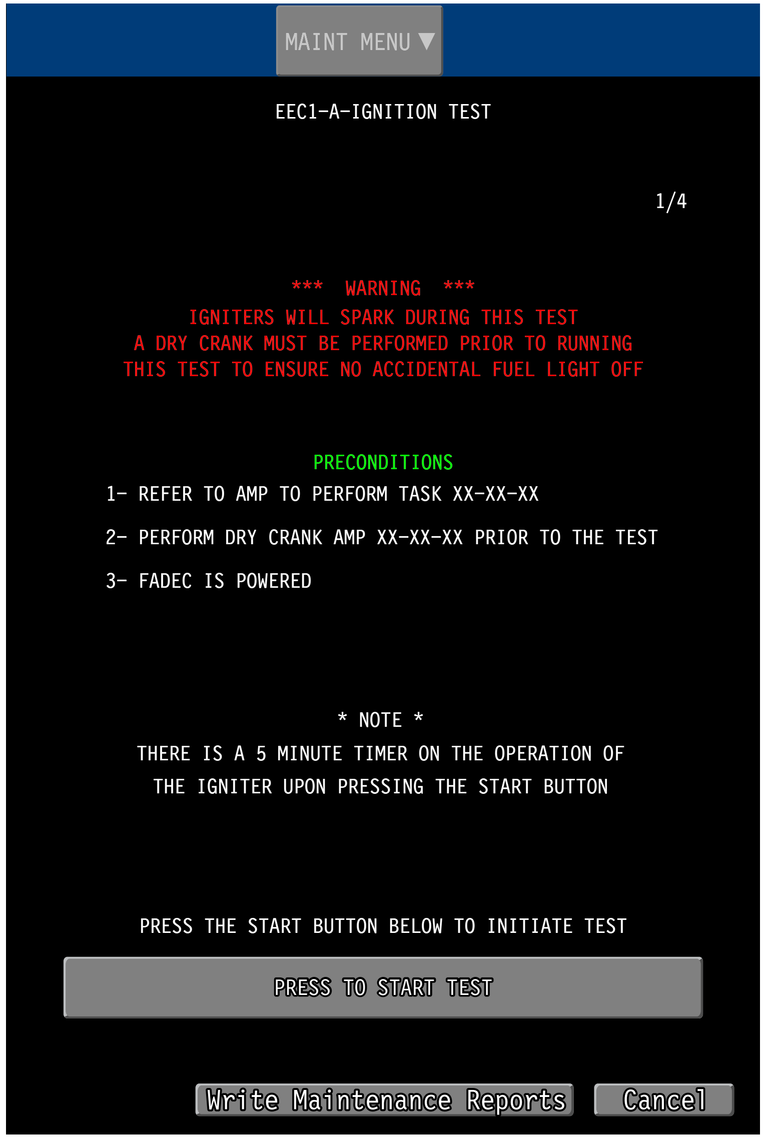
The ignition system test procedure checks for satisfactory operation of the ignition system.

The following functions or units are verified with this test:

- EEC outputs to exciter
- Exciter
- Ignition cables (two per engine)
- Igniter plugs (two per engine)

The engine igniters are wired to one channel each, with igniter 1 going to EEC channel A, and igniter 2 going to EEC channel B. As a consequence, the ignition system test can only verify the functionality of the igniter corresponding to the channel in control during the test. To test both igniters, the maintainer must run the test again on the alternate EEC channel.

This test allows the user to audibly verify if the igniters provide a spark when commanded to do so.



CS1_CS3_7321_023

Figure 9: Ignition System Test

80-11 STARTING

GENERAL DESCRIPTION

The engine starting system consists of an air turbine starter (ATS) and a starter air valve (SAV). The ATS uses bleed air to drive the high-pressure (HP) rotor to a high enough speed to ensure a satisfactory start. The bleed air is provided by the auxiliary power unit (APU), an external ground air cart, or the other operating engine. The SAV controls the flow of bleed air to the ATS.

When the engine has reached a predetermined speed, the SAV closes, shutting off the bleed air to the ATS. As the engine continues to accelerate, the starter clutch automatically disengages the starter. The ATS can also be used to start the engine in flight if a windmill start can not be accomplished.

Engine starting can be accomplished automatically or manually.

AUTOMATIC START

The automatic start mode provides the electronic engine control (EEC) with full control to sequence the SAV, ignition exciter and the fuel metering valve located in the integrated fuel pump and control (IFPC). The automatic start is initiated when the engine is not running, the START switch is in AUTO, and the L(R) ENG switch is set to ON.

When the start command is received from the L(R) ENG switch, the EEC commands the SAV, ignition exciter, and the fuel-on function of the IFPC. Once the engine reaches the starter cutout speed, the EEC commands the SAV closed, and the ignition exciter off.

Switching the L(R) ENG switch OFF during an automatic start shuts off the fuel, closes the SAV, and turns the ignition system off.

The EEC monitors the automatic start and has the capability to abort the startup to the self-sustaining engine speed.

MANUAL START

Manual engine starting is available on the ground only. The manual start mode limits the authority of the EEC so that the starter, ignition, and fuel are controlled manually. This provides the ability to dry motor or wet motor the engine.

The manual start sequence is available when the CONT IGNITION PBA is selected in the ON position. The start is initiated when the START switch is selected to the L (R) ENG CRANK position. The EEC commands the SAV open and the ATS drives the HP rotor. Once the engine is rotating, fuel and ignition exciter are commanded ON by selecting the L(R) ENG switch to ON.

Once the engine reaches the starter cutout speed, the EEC commands the SAV closed, and the ignition exciter off.

The manual start sequence can be terminated by selecting the START switch to the AUTO position if the L(R) ENG switch has not been selected ON, or by selecting the L(R) ENG switch to OFF.

The EEC only provides fault indications for the engine indication and crew alerting system (EICAS) during manual operation. The EEC start abort function is not available. Monitoring of the start parameters is required. The start is terminated by selecting the L(R) ENG switch to OFF.

ROTOR BOW

To minimize the effects of a bowed rotor, a motor-to-start dry crank is incorporated into every engine start sequence. The starter air valve is opened and modulated to maintain an N2 speed between 7% and 13% for a minimum of 30 seconds before continuing with the engine start.

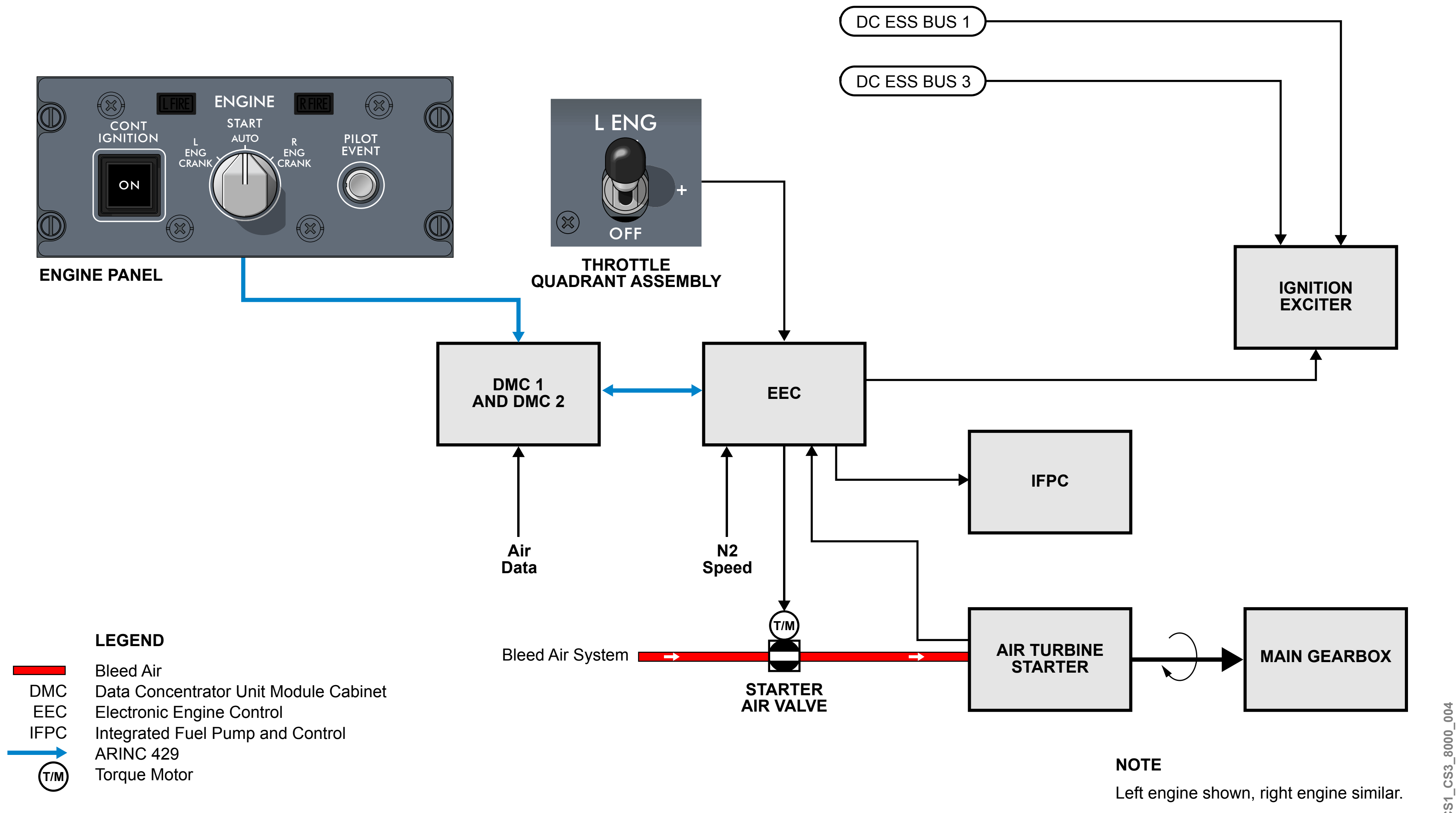


Figure 10: Engine Starting System

COMPONENT LOCATION

AIR TURBINE STARTER

The air turbine starter (ATS) is mounted to the main gearbox (MGB), located on the left side of the engine below the air oil cooler.

STARTER AIR VALVE

The starter air valve (SAV) is mounted to the MGB, located on the left side of the engine behind the air cooler.

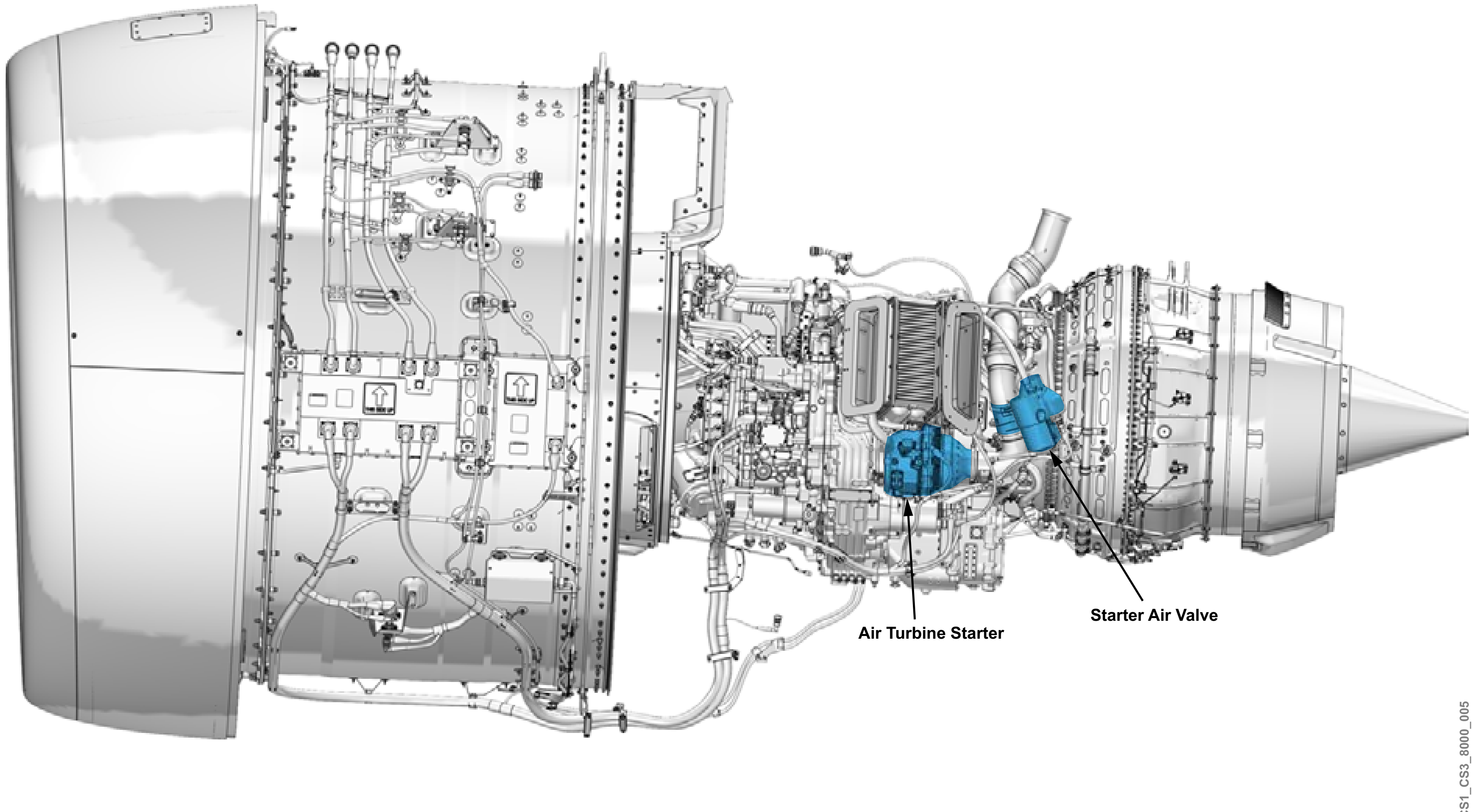


Figure 11: Engine Starting System Component Location

COMPONENT INFORMATION

AIR TURBINE STARTER

The starter converts power from the bleed air system to rotational energy using a turbine and a reduction gearbox. The gearbox drives the output shaft through a ratchet and pawl clutch. At low engine speeds, the pawl springs force the pawls to engage with the ratchet on the output shaft. When the engine speed exceeds that of the output hub, the pawls act as fly weights and disengages from the drive shaft.

To prevent damage to the air turbine stater (ATS) or the engine during start, the output shaft has a shear section between it and the main gearbox (MGB). This is designed to fail within a specific torque range.

Starter speed is sent to the EEC by a dual-channel speed sensor located on the starter housing. The speed sensor is a line replaceable unit (LRU) that provides a starter speed signal to each channel of the engine electronic control (EEC). The EEC uses the starter speed to determine start valve position, and starter engagement and disengagement.

The starter has a sight glass to verify the oil level, a fill port for servicing the oil, and a starter chip collector. The starter chip collector is to prevent any debris from circulating through the ATS oil system.

A locator pin on the ATS gearbox interface is used to ensure the ATS is properly clocked during installation.

The ATS is cooled using fan air from the inlet of the air/oil cooler. The air flows through a cooling ring around the ATS forward flange.

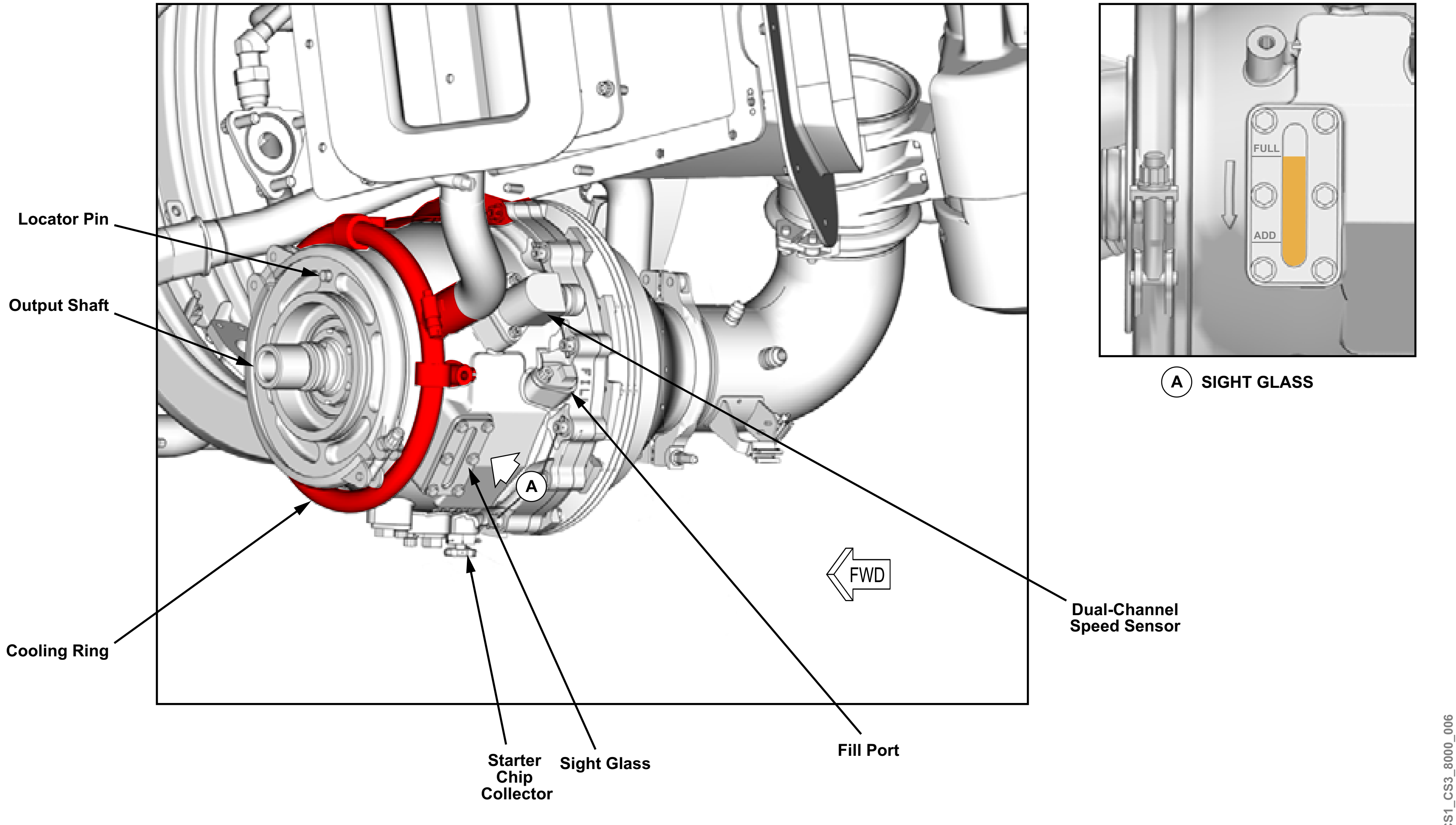


Figure 12: Air Turbine Starter

STARTER AIR VALVE

The starter air valve (SAV) is a pneumatically-actuated butterfly valve. A pressure regulator maintains an output pressure of 30 psi. The valve is spring-loaded to the closed position. The SAV is EEC controlled using a dual-coil torque motor. During start, the valve opening is confirmed through starter speed (Ns) and the N2.

A filter port is accessible once the shroud is removed to access a line replaceable filter.

A cooling shroud surrounds the SAV. A hose provides air from the air/oil cooler to the shroud to protect the valve from the heat of the engine core.

If the SAV fails to operate, a manual override is provided to open the valve. An SAV manual override access hole on the cowls permits a 3/8 in square drive manual override to be inserted into the SAV for manual opening.

Manual Override Start Procedure

The following procedure is used during a manual override start:

1. Establish communication with the flight deck.
2. Ensure bleed air is available to the SAV.
3. Insert the 3/8 in square drive into the SAV manual access hole.
4. Ensure the SAV is in the closed position. Rotate the square drive clockwise to confirm.
5. When directed, open the SAV by turning the manual override counterclockwise until the stop is contacted.
6. Hold the valve in the open position.
7. When directed, allow the valve to return to the closed position. The valve is spring-loaded to the closed position. Rotate the square drive clockwise to close if required.
8. Remove the manual override drive.
9. Move clear of the aircraft, avoiding the engine hazard areas.
10. Contact the flight deck to indicate clear of the aircraft.

NOTE

This is not an approved procedure at this time due to the incorporation of the motor-to-start logic for rotor bow prevention.

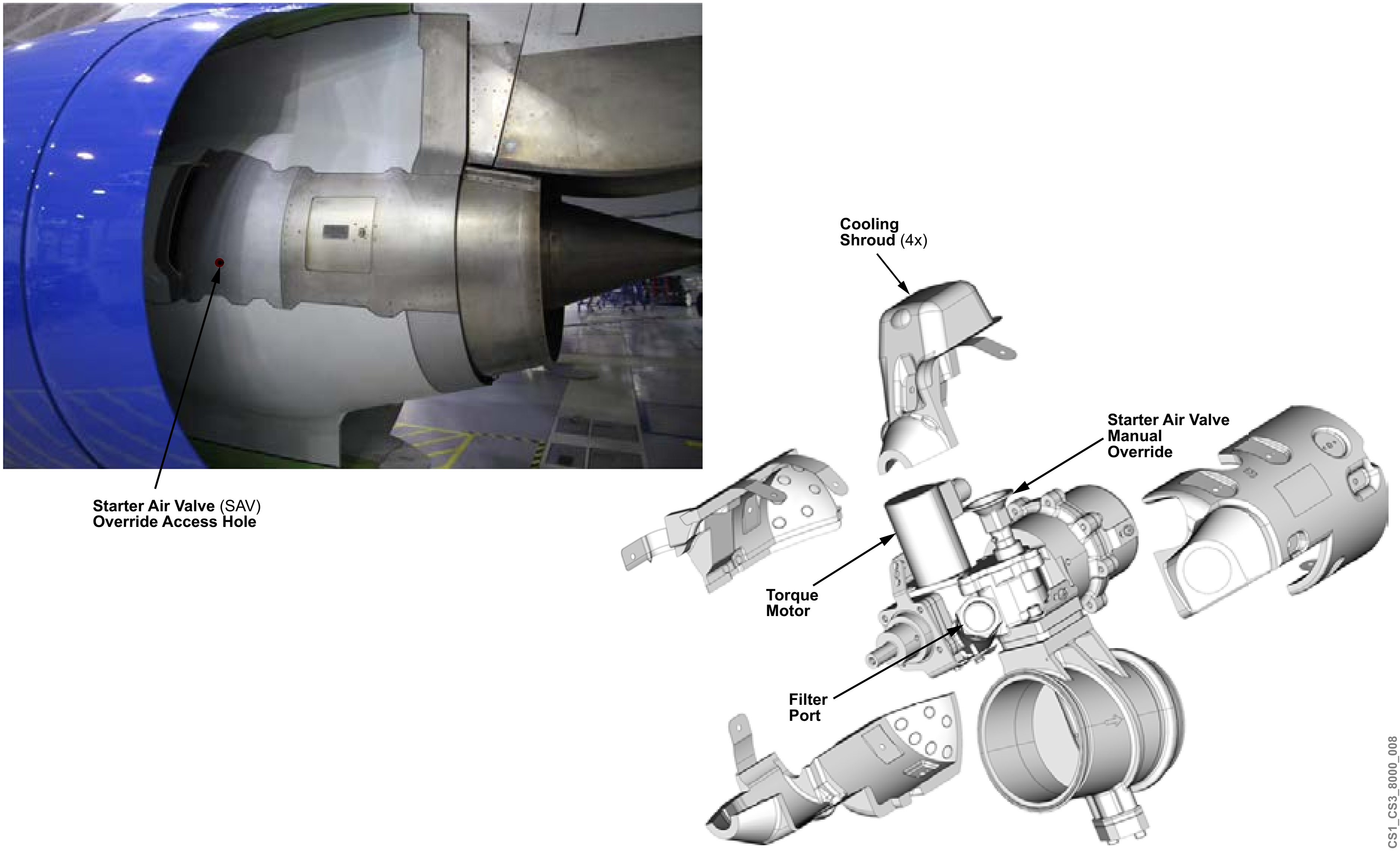


Figure 13: Starter Air Valve

CONTROLS AND INDICATIONS

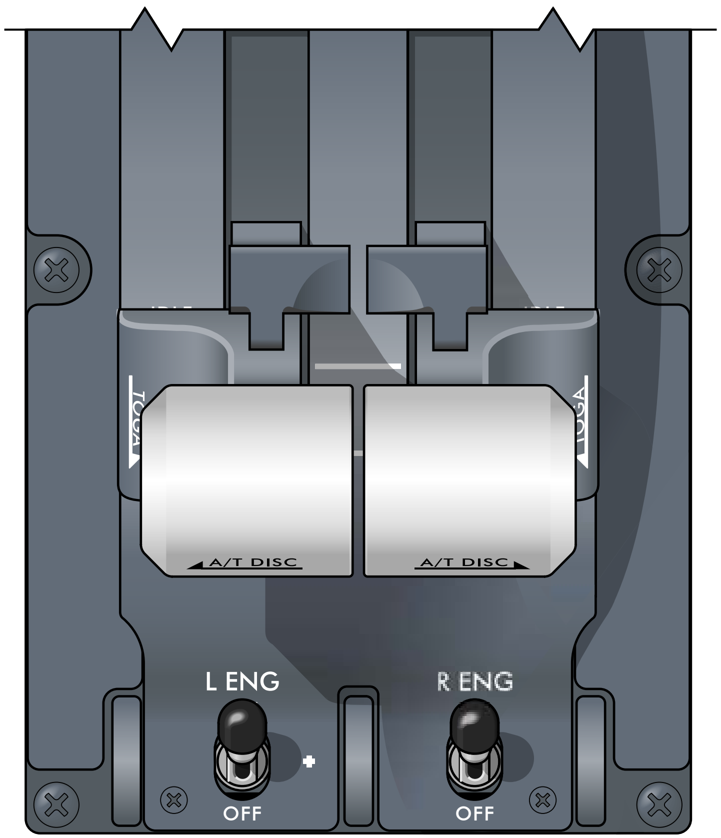
THROTTLE QUADRANT ASSEMBLY

The L(R) ENG switches are used to initiate an automatic start, or command the fuel and ignition during a manual start. The L(R) ENG switches are also used to shut down the engine.

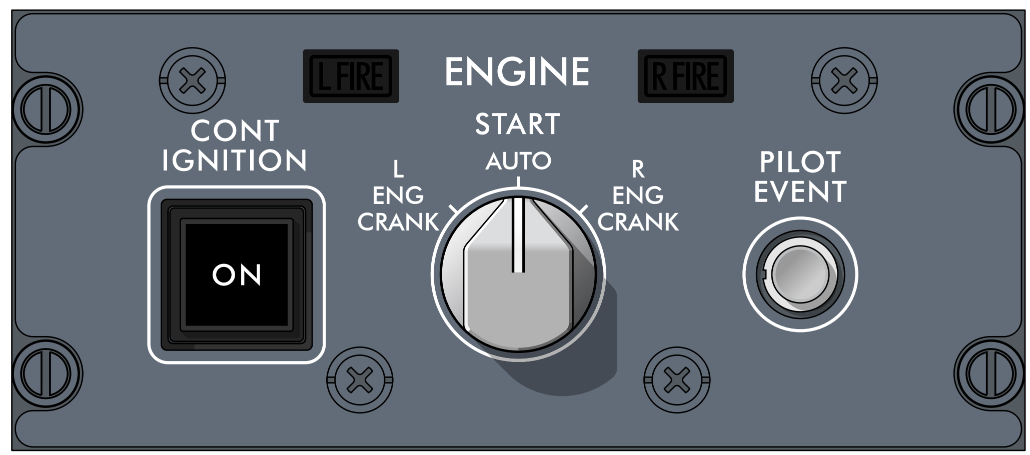
ENGINE PANEL

The engine panel has a three-position START switch marked AUTO and L(R) ENG CRANK. The switch is spring-loaded to the AUTO position. When a start is initiated with the switch in the AUTO position, an automatic start is performed. The L(R) ENG CRANK positions are used for manual starts and wet or dry motoring.

The CONT IGNITION PBA is used to select manual ignition. It also selects the manual start mode for the EEC.



THROTTLE QUADRANT ASSEMBLY



ENGINE PANEL

CS1_CS3_8000_010

Figure 14: Engine Start Controls

INDICATIONS

While the EEC controls the start sequence, the engine indication and crew alerting system (EICAS) page is monitored during the start sequence. Indications that are monitored during engine start include:

- N1
- Exhaust gas temperature (EGT)
- Ignition
- N2
- Fuel flow
- Oil pressure

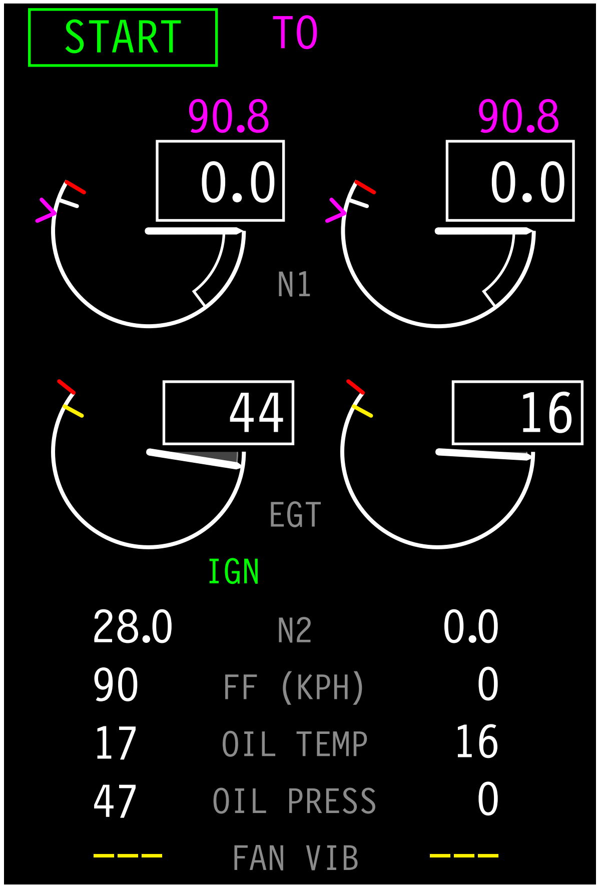
The following indications are displayed above the N1 gauges, depending upon start mode:

- START - Engine is in the start sequence
- RELIGHT - Indicates an automatic relight

In the case of an automatic restart in flight, there are the following indications depending on the type of restart:

- WINDMILL - Altitude from 30,000 ft to sea level, airspeed above 250 KIAS
- ATS - Indicates an air turbine starter start mode. There are two different types of ATS starts
    - Starter assisted APU - Altitude from 23,000 ft to sea level
    - Engine cross-bleed - Altitude from 30,000 ft to sea level, airspeed less than 250 KIAS

The EEC determines what type of in-flight restart to use depending on the altitude and airspeed of the aircraft.



EICAS PAGE

START ICONS	
Symbol	**Description**
START	Engine in start sequence.
RELIGHT	Automatic relight.
ATS	ATS (Air Turbine Start).
WINDMILL	Windmill.

CS1_CS3_7400_008

Figure 15: Engine Starting Indications

OPERATION

ROTOR BOW

The motor-to-start time for rotor bow prevention is based on an altitude/outside air temperature (OAT) graph. Within the altitude/OAT envelope, the motor to start time is 30 seconds. As the distance outside the altitude/OAT envelope increases, the motor-to-start time increases.

An ENG START DELAY advisory message is displayed anytime an engine start is initiated outside of the envelope. The message clears when the motor to start phase is complete.

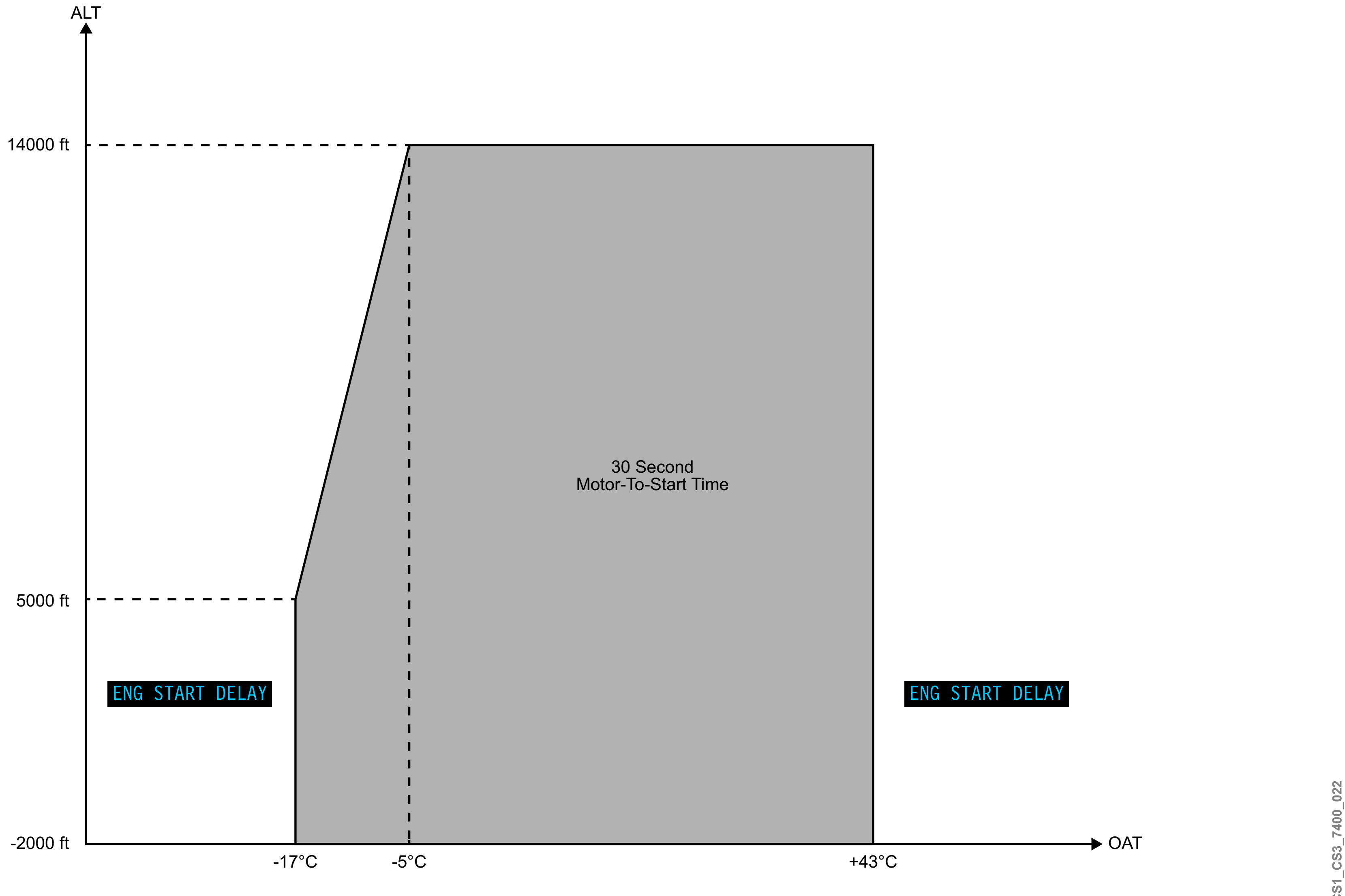


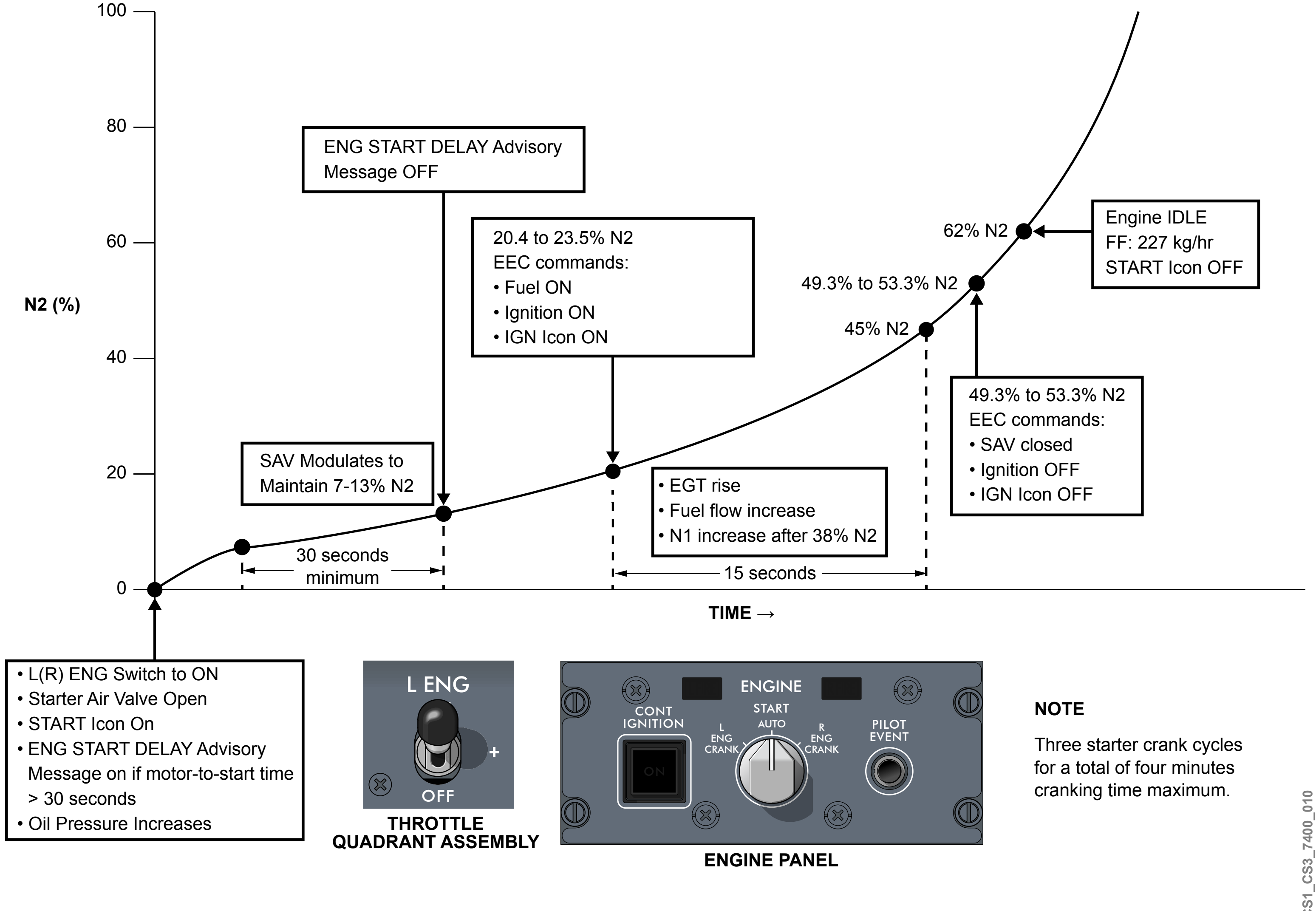
Figure 16: Rotor Bow Altitude/Outside Air Temperature Graph

AUTOMATIC START

When the L(R) ENG switch is moved to the ON position, the electronic engine control (EEC) begins the start sequence:

- Starter air valve (SAV) is opened
- Oil pressure increases
- START icon displayed
- ENG START DELAY advisory message on if motor-to-start time > 30 seconds
- Engine dry cranks between 7%-13% N2 for a minimum of 30 seconds
- ENG START DELAY message off
- At 20.4% to 23.5% N2, the EEC commands ignition using one of the igniter plugs
- IGN icon on
- Commands the fuel metering valve in the integrated fuel pump and control (IFPC) to supply a fuel flow of 91 kg/hr (200 lb/hr)
- Within 20 seconds of fuel flow:
    - EGT increases
    - N1 increases before 45% N2
- At 49.3 to 53.3% N2, the EEC commands the starter air valve closed and shuts off the ignition
- IGN icon off
- At engine idle, 62% N2, fuel flow reaches 227 kg/hr (500 lb/hr)
- START icon off

For automatic starts in flight, the EEC selects both igniters on.



LEGEND

EEC	Electronic Engine Control
EGT	Exhaust Gas Temperature
FF	Fuel Flow
kg/hr	Kilograms per Hour
SAV	Starter Air Valve

NOTE

Three starter crank cycles for a total of four minutes cranking time maximum.

Figure 17: Automatic Start

MANUAL START

In a manual start, the CONT IGNITION PBA is selected to ON. The START switch is held in the L(R) ENG CRANK position, and the following occurs;

- Starter air valve opens
- ENG START DELAY advisory message on if motor-to-start time > 30 seconds
- Engine dry cranks between 7%-13% N2 for a minimum of 30 seconds
- ENGINE START DELAY message off
- At 18.5% to engine maximum motoring speed of 24% N2, the L(R) ENG switch is moved to RUN
  - START icon on
  - The EEC commands the ignition using both igniter plugs
  - IGN icon on
  - Commands the fuel metering valve in the integrated fuel pump and control (IFPC) to open and supply a fuel flow of 91 kg/hr (200 lb/hr)
- Within 20 seconds of fuel flow:
  - EGT increases
  - N1 increases before 45% N2
- At 49.3 to 53.3% N2, the START switch is released to the AUTO position then the EEC commands the SAV closed and shuts off the ignition
  - IGN icon OFF
- At engine idle, 62% N2, fuel flow reaches 227 kg/hr (500 lb/hr)
- Start icon off

When the engine idle speed is reached the CONT IGNITION PBA is selected NORMAL.

ENGINE MOTORING

When on ground, the engine can be motored for maintenance purposes. The engine can run continually for 4 minutes without stopping. A wait time of 30 minutes is required before starting again.

There are two motoring sequences, dry and wet motoring.

Dry Motoring

The EEC initiates a dry motoring when bleed air is available and the following controls are set:

- START switch in L(R) ENG CRANK
- CONT IGNITION PBA in normal
- L(R) ENG switch OFF

The motoring can be interrupted any time by releasing the START switch to AUTO.

Wet Motoring

The EEC initiates a wet motoring when bleed air is available and the following controls are set:

- Pull the applicable ignition circuit breakers or open the ignition solid-state power controllers (SSPCs)
- START switch in L(R) ENG CRANK
- CONT IGNITION PBA in normal
- L(R) ENG switch to RUN

Positioning the L(R) ENG switch to the OFF position shuts off fuel and initiates a dry crank. The motoring can be interrupted any time by releasing the engine START switch to AUTO.

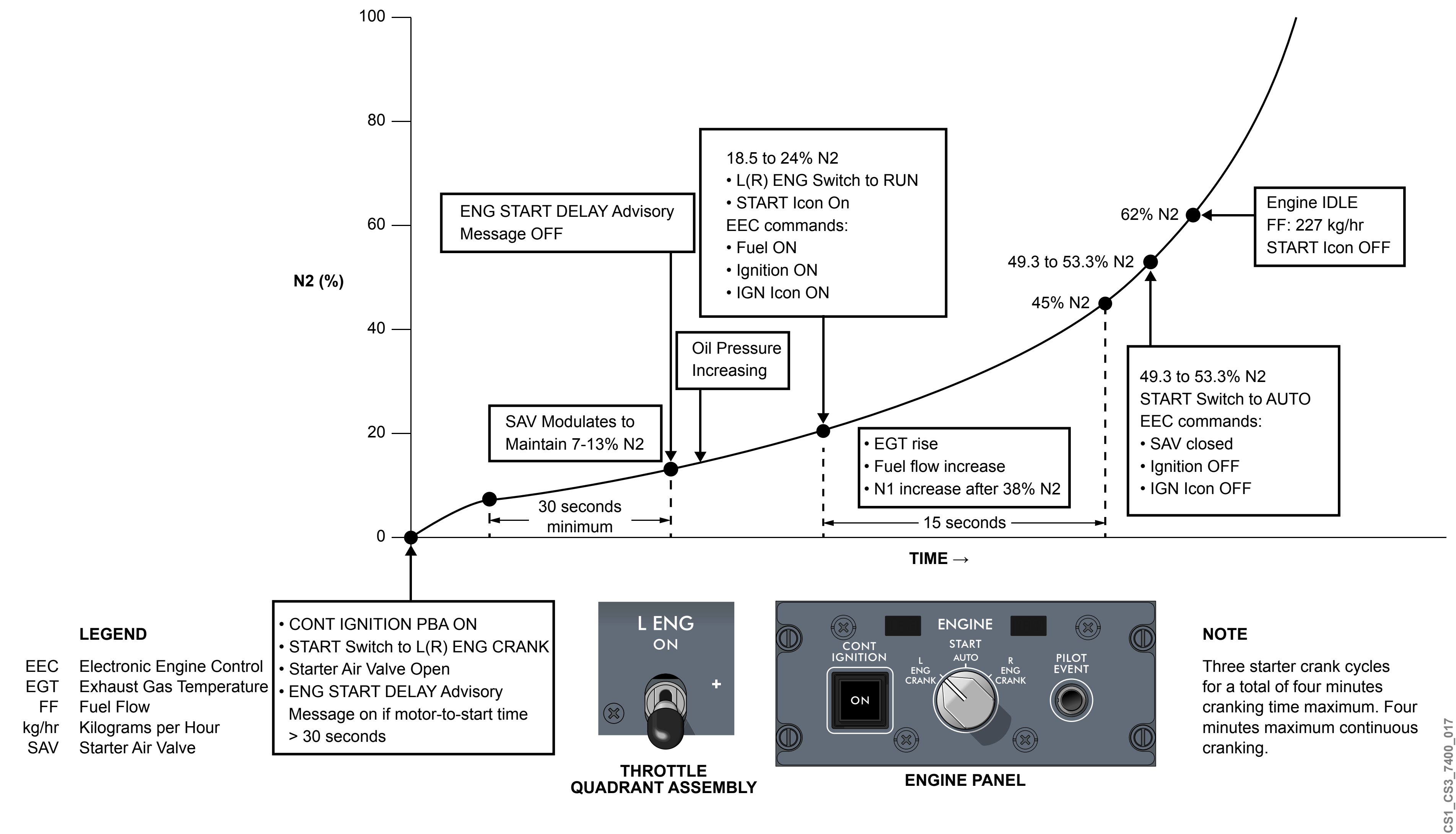


Figure 18: Manual Start

DETAILED DESCRIPTION

AUTOMATIC STARTING

During engine start, the EEC modulates the SAV to maintain the N2 speed between 7% and 13% for a minimum of 30 seconds to prevent rotor bow. An ENG START DELAY advisory message is displayed if the motor to start time exceeds 30 seconds.

During a normal start, the starter air valve (SAV) and ignition exciter are automatically turned off by the electronic engine control (EEC) between 49.3% and 53.3% N2 speed, depending on ambient conditions. A normal engine start is indicated by the START icon above the N1 indicator.

Starter assist is commanded by the EEC for in-flight starts at low Mach numbers where windmilling conditions are insufficient for engine starting. The EEC selects the starter assistance mode if altitude or Mach number are not available. During an in-flight start, the EEC depressurizes the engine-driven hydraulic pump (EDP) to unload the engine and improve starting performance.

A windmill start is carried out if the altitude is between 30,000 ft and sea level with airspeed above 250 KIAS. If the aircraft is not in the windmill start envelope, the starter can be used to assist the start. If the altitude is between 30,000 ft and 23,000 ft, the engine can be restarted using bleed air from the opposite engine. If the altitude and speed are below 23,000 ft and less than 250 kt, the auxiliary power unit (APU) can be used for starting.

On the ground, upon detection of a no light, hot or hung start, loss of exhaust gas temperature (EGT) or a locked low-pressure rotor, the EEC automatically shuts off fuel, ignition, and starter air and provides the appropriate fault indication to the flight deck. An automatic start abort is indicated by an L(R) ENG START ABORT caution message on the engine indication and crew alerting system (EICAS). The message indicates that the engine start procedure has aborted. Idle speed was not reached and the START icon disappears. The message clears when the L(R) ENG switch is selected to OFF.

The EEC start abort is inhibited above 49.3% N2 to 53.3% N2 on the ground and at all conditions in-flight. Following a ground start abort, an automatic 30 seconds engine motoring period is provided to clear fuel vapor and to cool the engine. An automatic start performs one restart attempt on the ground.

STARTER DUTY TIME

A STARTER TIME EXCEEDED caution message is displayed during auto or manual starts when the starter operating time reaches 4 minutes, or if the starter is operated above 44% N2 for 30 seconds. A 30 minute cooling time is required after three start attempts or 4 minutes of continuous cranking. The EEC monitors the starter operating time including during both wet and dry cranking.

During an automatic start on the ground, the start attempt is discontinued and the EEC closes the starter air valve. When the message occurs, the L(R) ENG switch must be selected OFF to reset the EEC timers. For a manual start or an in-flight start, the message remains displayed until the start is terminated, or the engine reaches idle.

STARTER RE-ENGAGEMENT

The L(R) ENG switch must be selected OFF before a second attempt can be made following an aborted manual start attempt or an engine shutdown. If the starter engagement is interrupted during a manual start, the starter can be re-engaged below the starter cut-out speed. During manual or automatic restarts, the EEC prevents starter re-engagement until the starter rotational speed is synchronized with engine N2 speed. The starter re-engagement speed is 20% N2. The EEC compares N2 with the starter speed signal. If the starter speed is unavailable, the start valve must be closed for 1 minute, then the starter can be re-engaged at 20% N2 on the ground, or in the air.

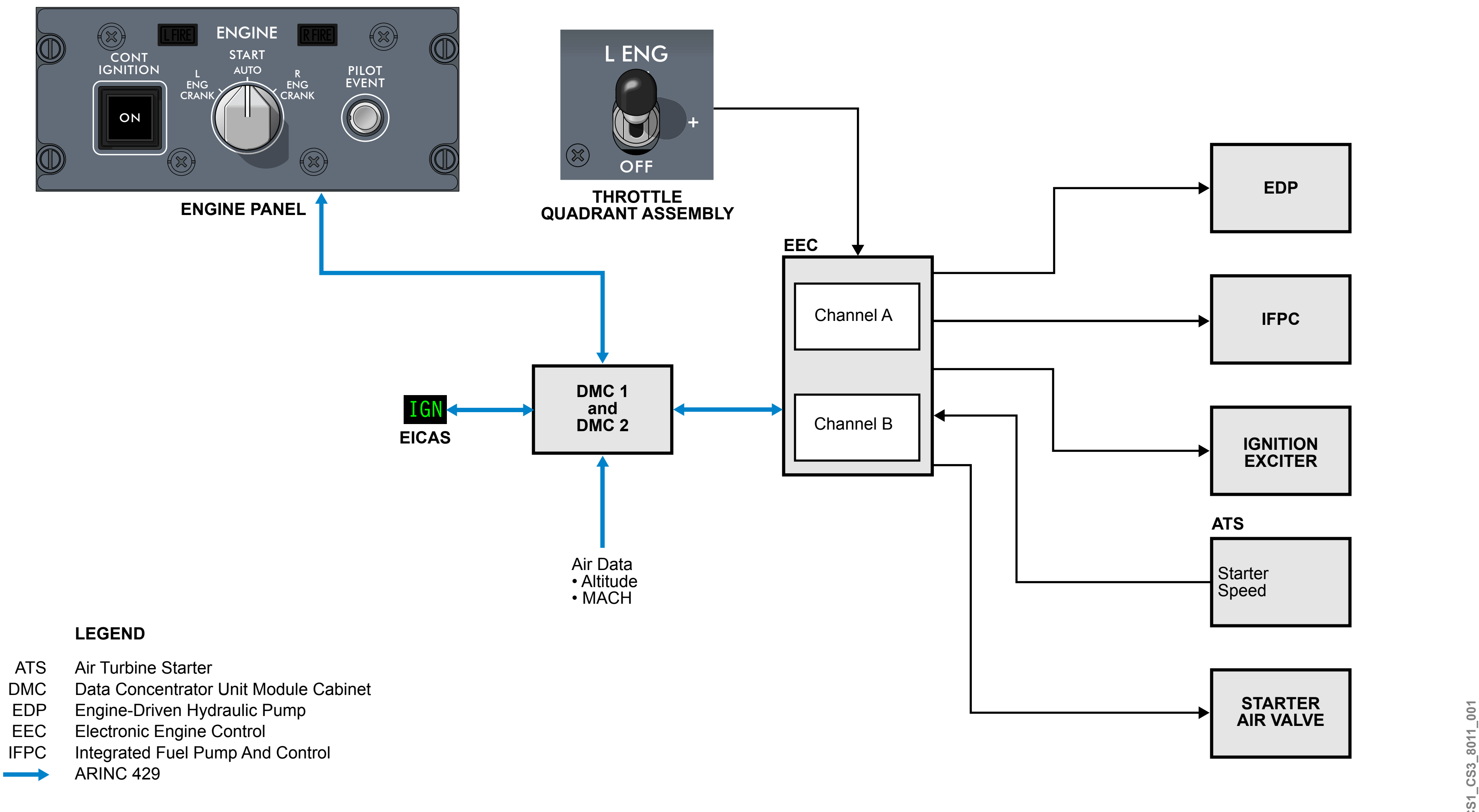


Figure 19: Automatic and Manual Starting Schematic

ABNORMAL STARTS

Normal Engine Start

If the oil pressure is less than 70 psi after 30 seconds at the usual idle speed, shut down the engine using the normal shutdown procedure.

The engine should be operated at idle power for a minimum of 5 minutes before shutdown. If necessary, the minimum idle time can be reduced to 3 minutes.

Automatic Start Failure

The EEC detects an automatic start failure if there are any of the following faults detected:

- Slow N2 spool-up
- Rapidly increasing EGT
- N1 roll back

Auto Restart

On ground, if the EEC senses an automatic start failure, the EEC turns the fuel and ignition OFF, and commands a dry motor to clear fuel from the engine for 30 seconds.

After motoring the engine, the EEC automatically commands an auto restart. If the auto restart attempt fails, the EEC aborts the start. If the EEC detects an overspeed or a fuel metering valve fault, the EEC does not attempt an auto restart.

Automatic Start Abort

The EEC aborts an automatic start if the engine encounters one or more of the following events during the start, and are not recoverable using the fuel depulse or automatic restart logic.

- N1 locked rotor: The EEC determines no N1 engine low-pressure rotor speed after sensing N2 has increased beyond 30%
- EEC unable to command igniters: The EEC is unable to command either igniter in either channel
- Loss of EGT indication
- Inability to control fuel flow

When N2 is above 50%, the automatic start abort function is not available. The L(R) ENG switch must be moved to OFF to abort the engine start.

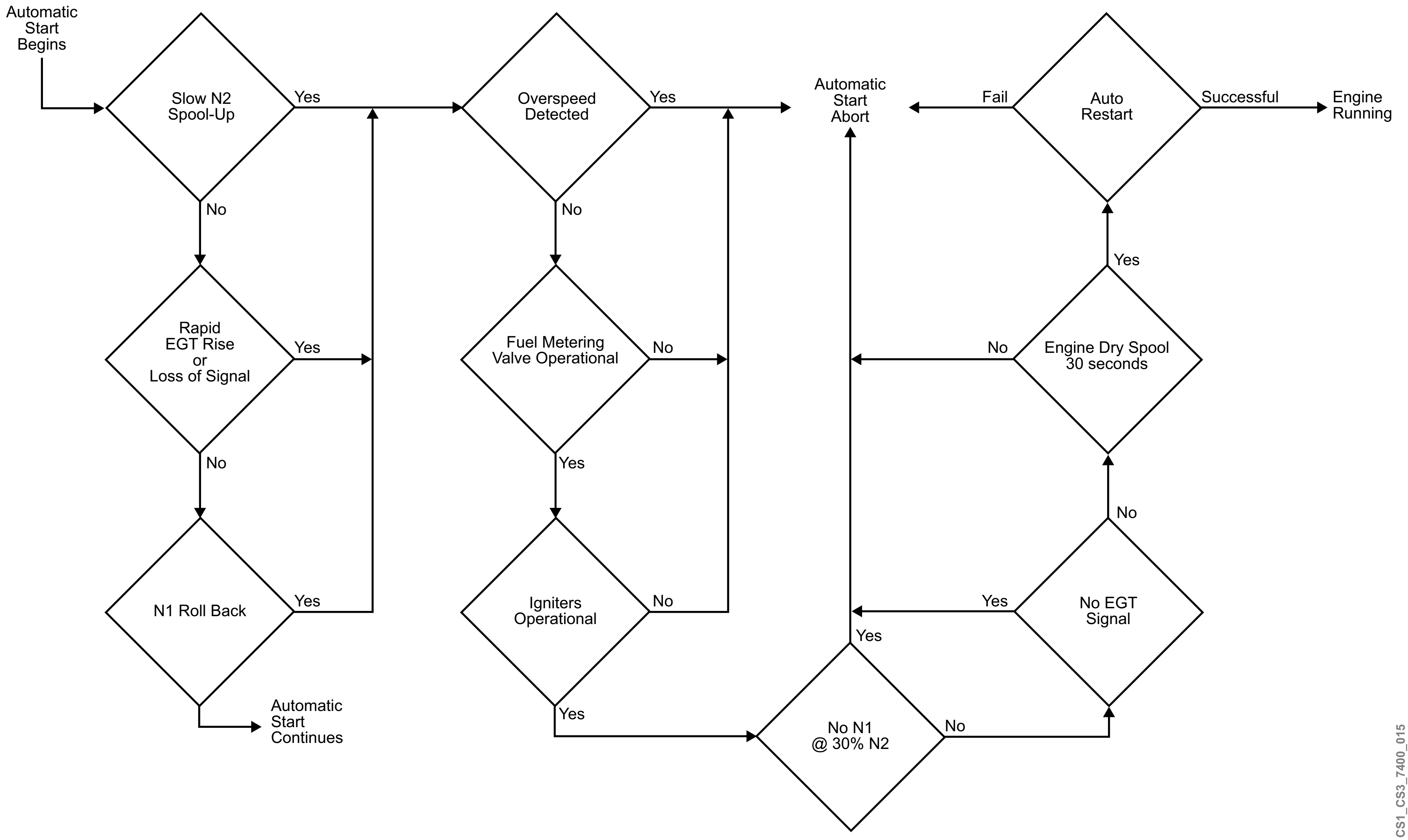


Figure 20: Abnormal Starts

Fuel Depulsing

Fuel depulsing is an EEC function that uses the fuel metering valve (FMV) in the integrated fuel pump and control (IFPC) to turn fuel on and off, in an attempt to clear an impending surge or hot start.

The EEC performs fuel depulsing when a surge or an impending hot start, as indicated by a rapid rise in EGT or an exceedance of 1054°C (1929° F). If an impending hot start is detected, the EEC cycles fuel off for 2 seconds and back on for 12 seconds using the fuel metering valve.

The EEC performs this cycling until EGT drops below the impending hot start detection limit or the surge has cleared.

Fuel depulsing is inhibited whenever any of the following conditions are present:

- The engine N2 is above 50%
- The aircraft is flying above an altitude of 20,000 ft
- EGT signal is lost

A timer in the EEC is used to discontinue fuel depulsing once the maximum allowable time of 28 seconds on the ground or 140 seconds in-flight has been reached, or if EGT increases to the starting amber value. The time limit is sufficient for two depulse cycles if the aircraft is on the ground. If the time limit has been exceeded, or the EGT has reached the starting amber value, the EEC terminates fuel depulsing and dry motors the engine for 30 seconds.

When the aircraft is in flight and below 20,000 ft, the timer in the EEC is changed to 140 seconds for fuel depulsing.

The EEC conducts a ground auto restart when the EGT has decreased below the impending hot start temperature, which the EEC determines depending on aircraft weight-on-wheels (WOW) status and N2 speed.

Engine Start Limits

The following table summarizes the engine starting limits.

Table 3: Engine Starting Limits

START LIMITS	
Maximum EGT	1054°C (1929°F)
Minimum oil temperature for starting	-40°C (-40°F)
Minimum oil temperature for operation above idle	-6°C (21°F)
Maximum Oil Temperature	174°C (345°F)
Starter Duty Cycle	Three starter crank cycles for a total of 4 minutes cranking time maximum followed by 30 minutes of cooling time. 4 minutes maximum continuous cranking.
Start Abort (EEC aborts start automatically below 54% N2)	• No N1 rotation by 30% N2 • N2 fails to reach stabilized idle speed within 120 seconds • No EGT rise after 20 seconds from positive fuel flow indication • Oil pressure does not increase within 30 seconds • EGT greater than 1054°C (Hot Start)

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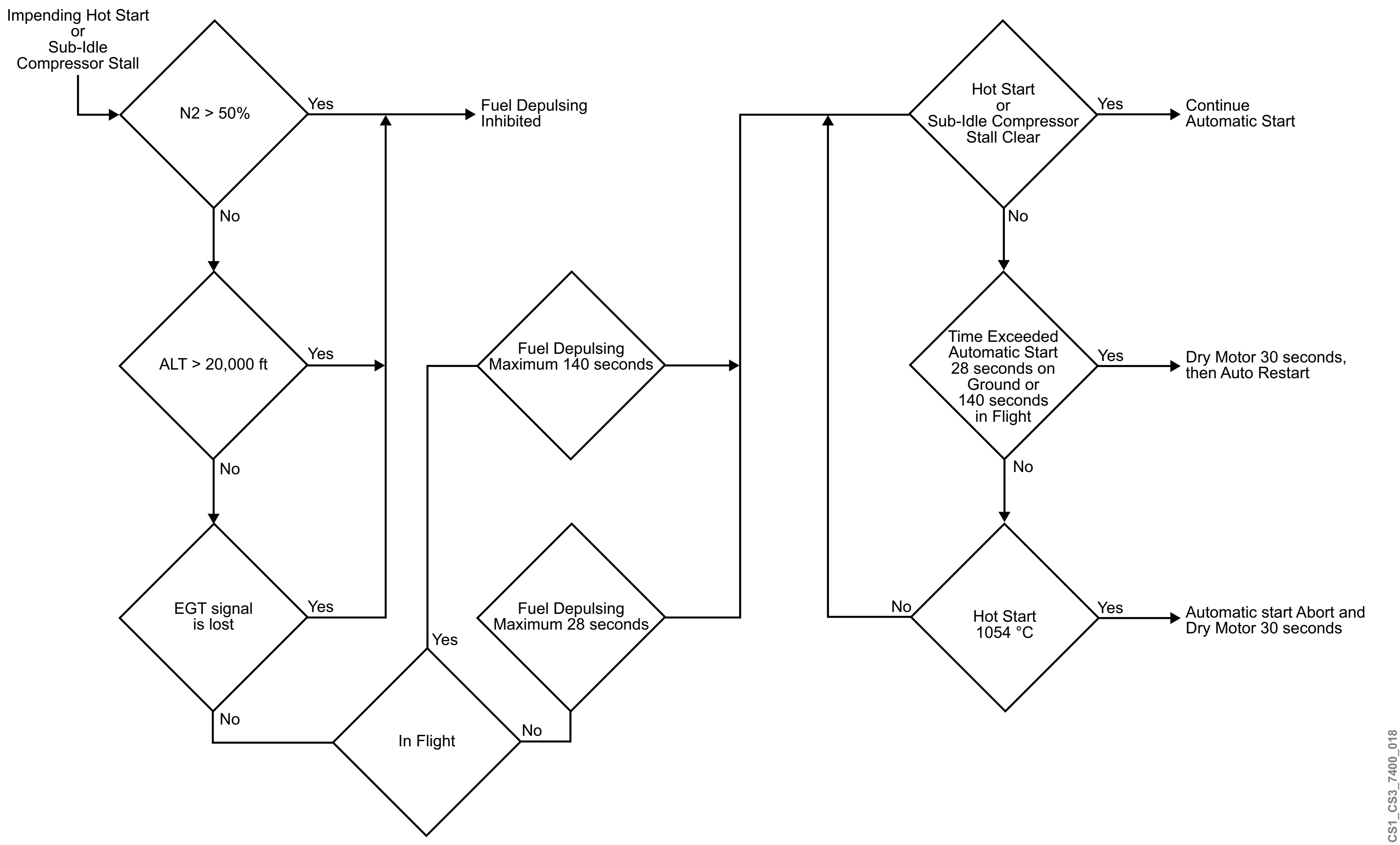

Figure 21: Fuel Depulsing

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) messages for the starting and ignition system.

CAS MESSAGES

Table 4: CAUTION Messages

MESSAGE	LOGIC
L ENG START ABORT	L engine starting procedure aborted.
R ENG START ABORT	R engine starting procedure aborted.
L ENG STARTER FAIL ON	ATS is not disengaging or the L starter air valve (SAV) failed to open.
R ENG STARTER FAIL ON	ATS is not disengaging or the R starter air valve (SAV) failed to open.

Table 5: ADVISORY Messages

MESSAGE	LOGIC
L (R) ENG FAULT	Loss of redundant or non-critical function for the left or right engine.
L (R) ENG STARTER OVHT	Left or right starter usage does not fulfill the duty cycle criteria: A 30 minute cool down time for 4 minutes (low N2), or 30 seconds (high N2).
ENGINE START DELAY	The engine is performing dry crank cooling during a ground start for protection against rotor bow.

Table 6: INFO Messages

MESSAGE	LOGIC
80 L ENGINE FAULT- STARTER SYSTEM INOP	INFO message will be set when one of the following is true: A dual channel starter speed signal fault, OR a dual channel SAV fault. A dual channel SAV failed closed fault has been detected by the right FADEC system. In the case of dual channel sensor fault, if the opposite channel control processor (CPU) has failed, a single channel fault would set this message.

PRACTICAL ASPECTS

STARTER OIL SERVICING

The starter oil is changed at specific intervals as per the aircraft maintenance schedule. To service the oil, both the fill and overfill plugs are removed. The oil is added until it spills from the overfill plug, at which point the starter sight glass should read full.

STARTER CHIP COLLECTOR

The air turbine starter (ATS) chip collector can be removed by hand without draining the oil. A check valve in the fitting prevents the loss of oil when the starter chip collector is removed.

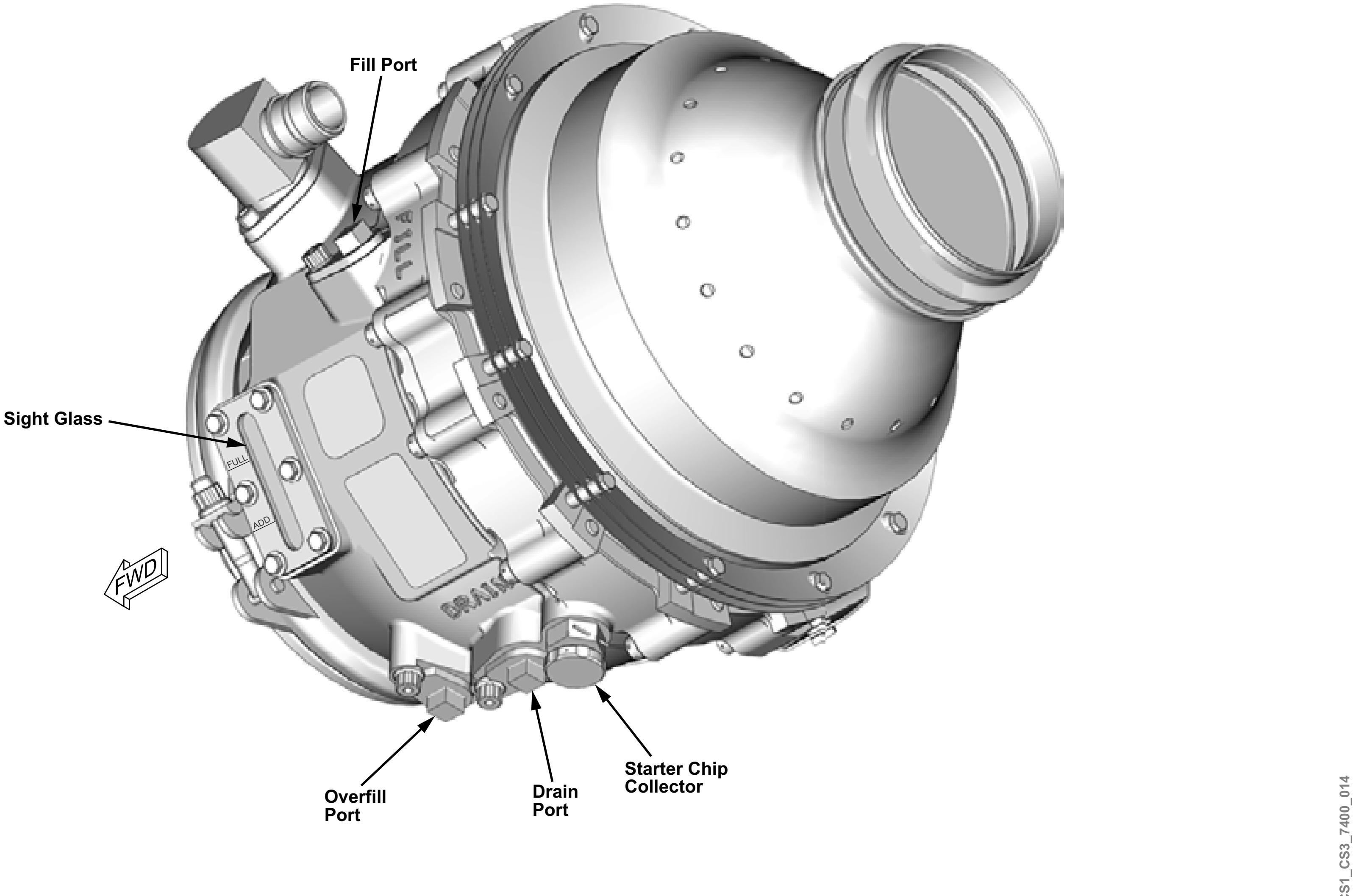

Figure 22: Starter Oil Servicing and Starter Chip Collector

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ENGINE AIR SYSTEMS - CHAPTER BREAKDOWN

75-24 TURBINE ACTIVE CLEARANCE CONTROL SYSTEM

GENERAL DESCRIPTION

The turbine active clearance control system provides fan air to the turbine cases to limit turbine case growth due to thermal expansion. This reduces high-pressure turbine (HPT) and low-pressure turbine (LPT) blade tip clearance and improves fuel efficiency.

The turbine active clearance control (ACC) system consists of a fuel actuated turbine ACC air valve that regulates the fan airflow to the LPT and HPT cooling air manifolds. The cooling air manifolds spray air on the turbine case.

The turbine ACC air valve is controlled by the electronic engine control (EEC). Servo fuel, supplied from the integrated fuel pump and control (IFPC) positions the valve in response to EEC commands.

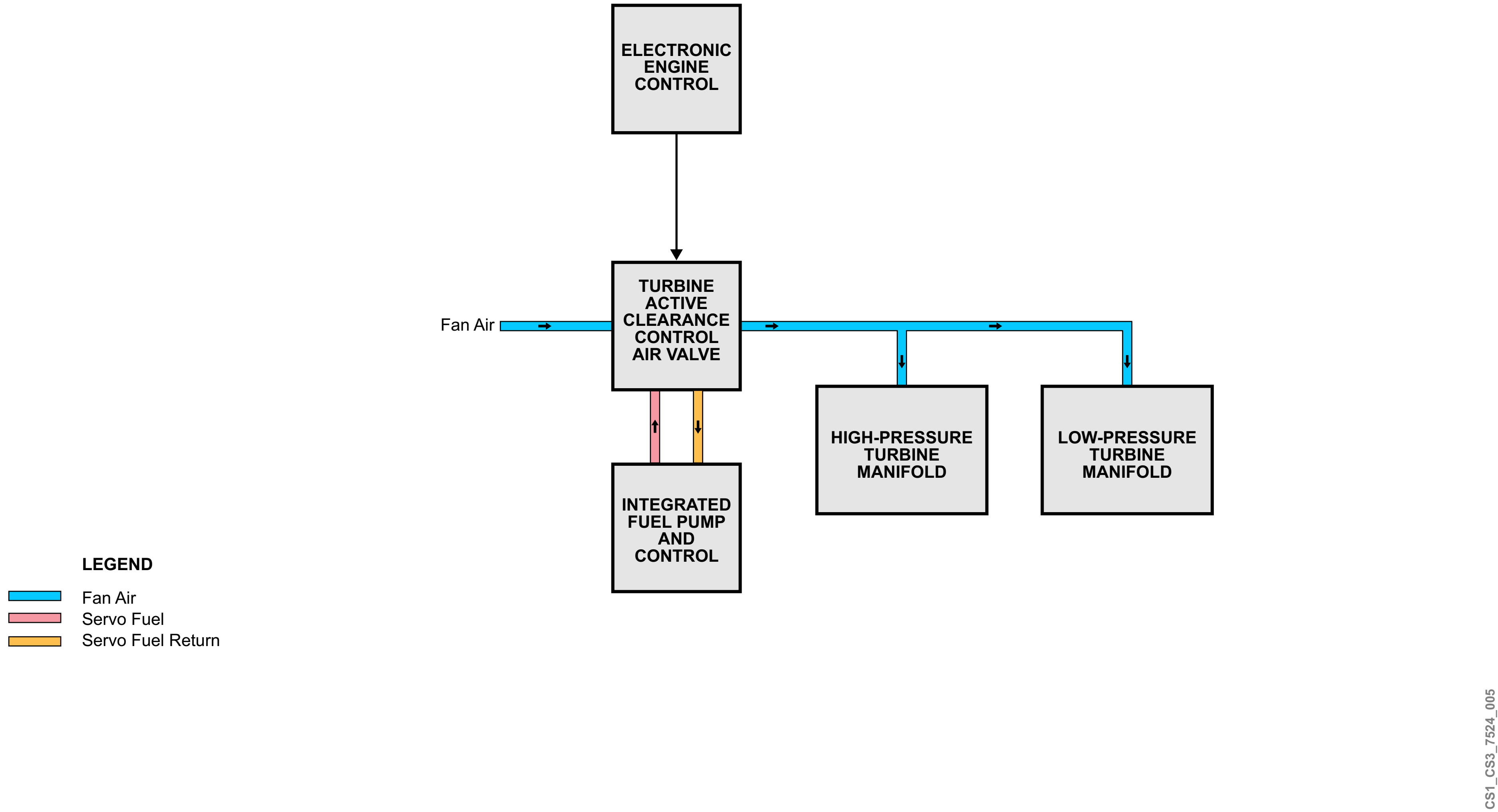


Figure 1: Turbine Active Clearance Control System

COMPONENT LOCATION

The turbine active clearance control system has the following components:

- Turbine active clearance control air valve
- High-pressure turbine (HPT) manifold
- Low-pressure turbine (LPT) manifold

Turbine Active Clearance Control Air Valve

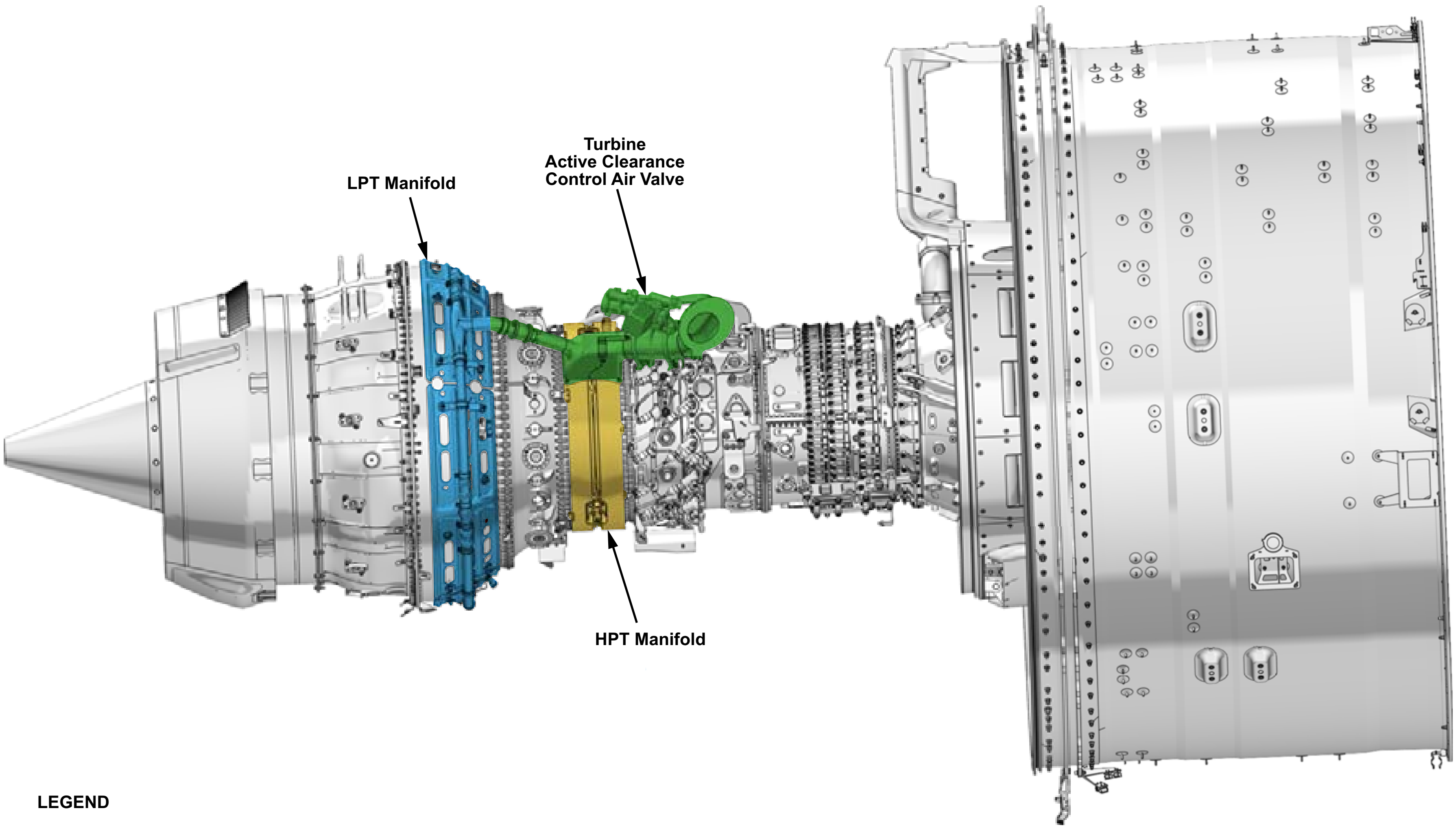
The turbine active clearance control (ACC) air valve is located at the 1 o’clock position, near the rear of the high-pressure compressor (HPC).

High-Pressure Turbine Manifold

The high-pressure turbine (HPT) manifold is located on the HPT case.

Low-Pressure Turbine Manifold

The low-pressure turbine (LPT) manifold is located on the LPT case.



LEGEND

HPT	High-Pressure Turbine
LPT	Low-Pressure Turbine

Figure 2: Turbine Active Clearance Control System Component Location

DETAILED DESCRIPTION

The turbine ACC air valve has a torque motor (T/M) that positions the valve using IFPC servo fuel pressure. The torque motor control is done by the EEC channel in control. The turbine ACC air valve single channel linear variable differential transformer (LVDT) sends position feedback to channel A of the EEC.

The EEC controls the valve according to engine parameters and altitude in combination with the turbine ACC air valve position feedback.

The turbine ACC air valve operation is scheduled as a function of N2 and altitude. The turbine ACC air valve is closed during takeoff and most of climb. It can open during the end of the climb phase and is always open in cruise. The failsafe mode of the turbine ACC air valve is the closed position.

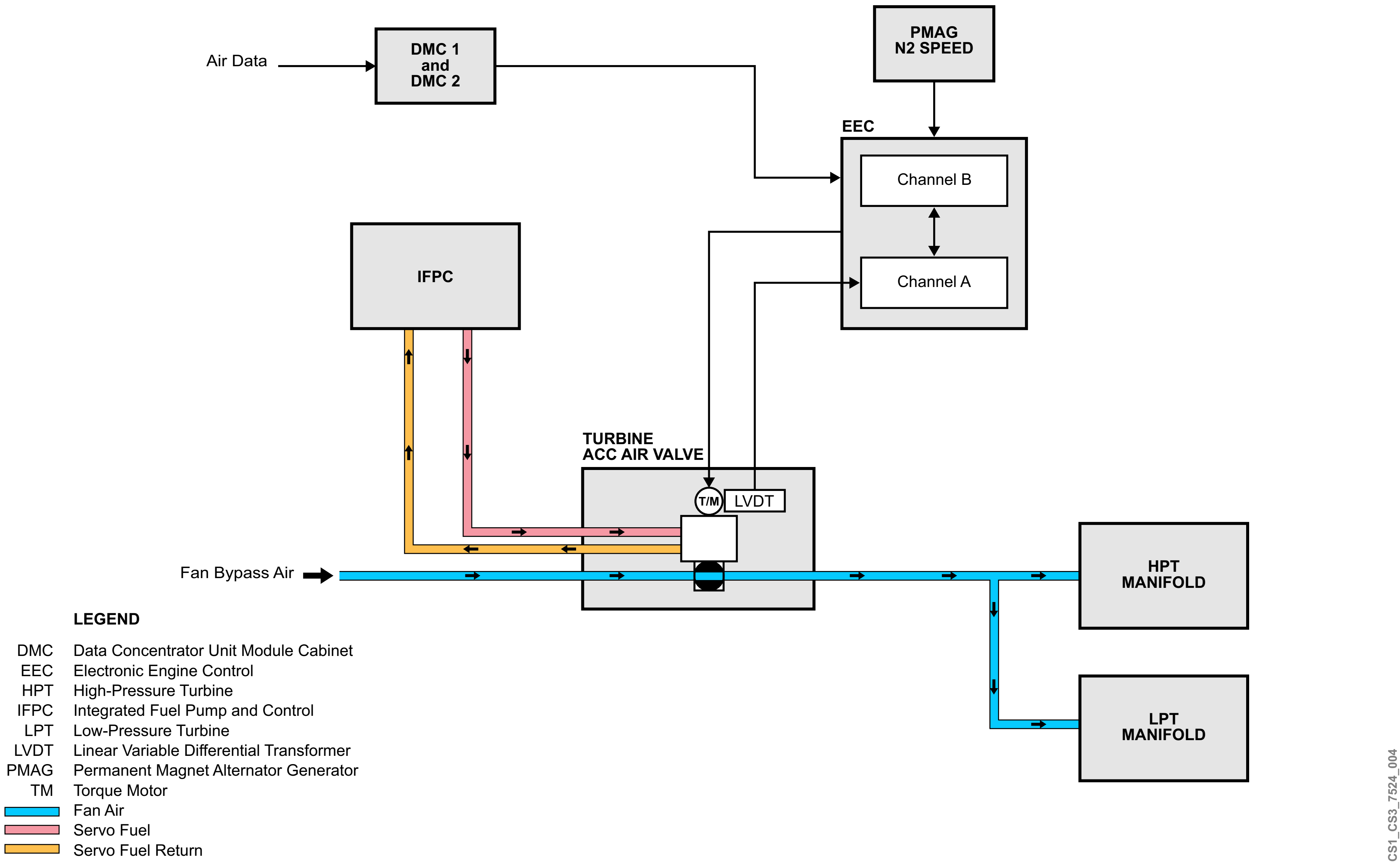


Figure 3: Turbine Active Clearance Control System Schematic

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the active clearance control system.

CAS MESSAGES

Table 1: ADVISORY Message

MESSAGE	LOGIC
L (R) ENG FAULT	Loss of redundant or non-critical function for the left (right) engine. Refer to INFO messages.

Table 2: INFO Messages

MESSAGE	LOGIC
75 L ENGINE FAULT - ACC FAIL CLSD	INFO message is set when one of the following is true: • A dual-channel ACC torque motor fault exists • Single channel ACC feedback fault has been detected by the left FADEC system • The turbine ACC air valve stuck closed
75 R ENGINE FAULT - ACC FAIL CLSD	INFO message is set when one of the following is true: • A dual-channel ACC torque motor fault exists • Single channel ACC feedback fault has been detected by the right FADEC system • The turbine ACC air valve stuck closed

75-25 TURBINE COOLING AIR

GENERAL DESCRIPTION

The turbine cooling air system consists of a series of tubes that provide 4th or 6th stage high-pressure compressor (HPC) air to cool parts of the engine. The 6th stage HPC bleed air cools the following engine parts;

- High-pressure turbine (HPT) 2nd stage vanes
- HPT 2nd stage blade attachment

The 4th stage HPC bleed air cools the following parts;

- Turbine intermediate case (TIC) fairing
- Low-pressure turbine (LPT) case
- LPT rotor and blade attachment

There are four tubes that feed cooling air from the 6th stage HPC to the HPT 2nd stage vanes and blade attachments. The air is metered by plates between the tubes and the HPT case. The cooling air then flows through hollow vanes and exits out the trailing edge.

Seven TIC cooling air tubes cool the TIC fairings, inner walls and outer walls known as transition ducts. Four are supply tubes, and three are jumper tubes. The three jumper tubes feed air through the TIC fairings to the LPT rotors and blade attachment. Three additional jumper tubes feed air into the space between the LPT case and 2nd stage vane.

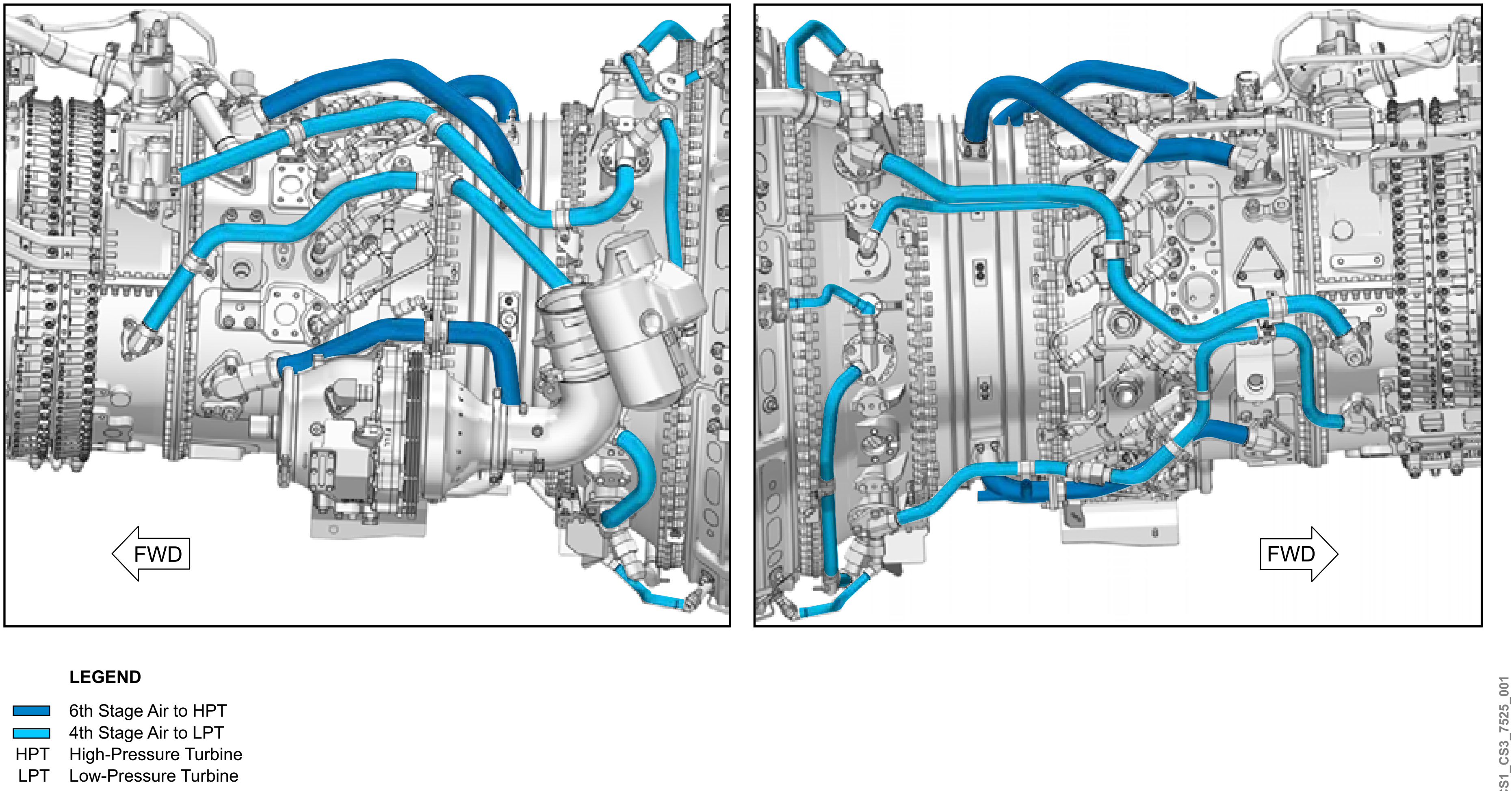


Figure 4: Turbine Cooling Air

75-26 BUFFER AIR SYSTEM

GENERAL DESCRIPTION

The engine bearing compartments are cooled using a split buffer bearing cooling system. This supplies the bearing compartments with cooling and pressurizing air from the 2.5 low-pressure compressor (LPC) exit air, or from the 4th stage high-pressure compressor air (HPC), depending on engine operating conditions.

The system is operated by the controlling electronic engine control (EEC) channel. The EEC controls the system through the buffer air valve solenoid (BAVS). The BAVS opens the buffer air shutoff valve (BASOV) when the BAVS is energized. The BASOV controls the flow of 4th stage air into the low-pressure side of the buffer air system to maintain the required seal pressure in each of the engine's bearing compartments during low power settings.

A buffer air pressure sensor (BAPS) monitors the pressure downstream of the BASOV to confirm that the BASOV is in the correct position for the engine power setting. The BAPS provides a signal to each EEC channel.

The buffer air check valve (BACV) allows 2.5 air to supply the buffer air system at high power settings when the BASOV is closed. The BACV closes when the BASOV is open and supplies 4th stage air to the buffer air system. When the BACV is closed it prevents backflow from the 4th stage into the engine.

The buffer air heat exchanger (BAHX) cools stage 4 air used for the no. 4 bearing compartment cooling and pressurization. The BAHX uses 2.5 air to cool the 4th stage air.

At low power, the 2.5 air does not have sufficient pressure and flow to supply the buffer air system. The EEC energizes the BAVS to open the BASOV. The BASOV allows 4th stage air to supply the front bearing compartments, consisting of the no. 1, no 1.5 and no. 2 bearings, no. 3, and no. 5/6 compartments, with air for cooling and pressurizing the seals. The no. 4 bearing compartment is always supplied with cooled 4th stage air regardless of the engine’s power settings.

At high power, the 2.5 air is sufficient to supply the buffer air system. The high power switchover occurs at 75% N1. The EEC de-energizes the BAVS to close the BASOV. When the BASOV closes, the BACV opens, allowing 2.5 air to supply the front bearing compartment no.3 and compartments no.5/6 with air for cooling and pressurizing the seals.

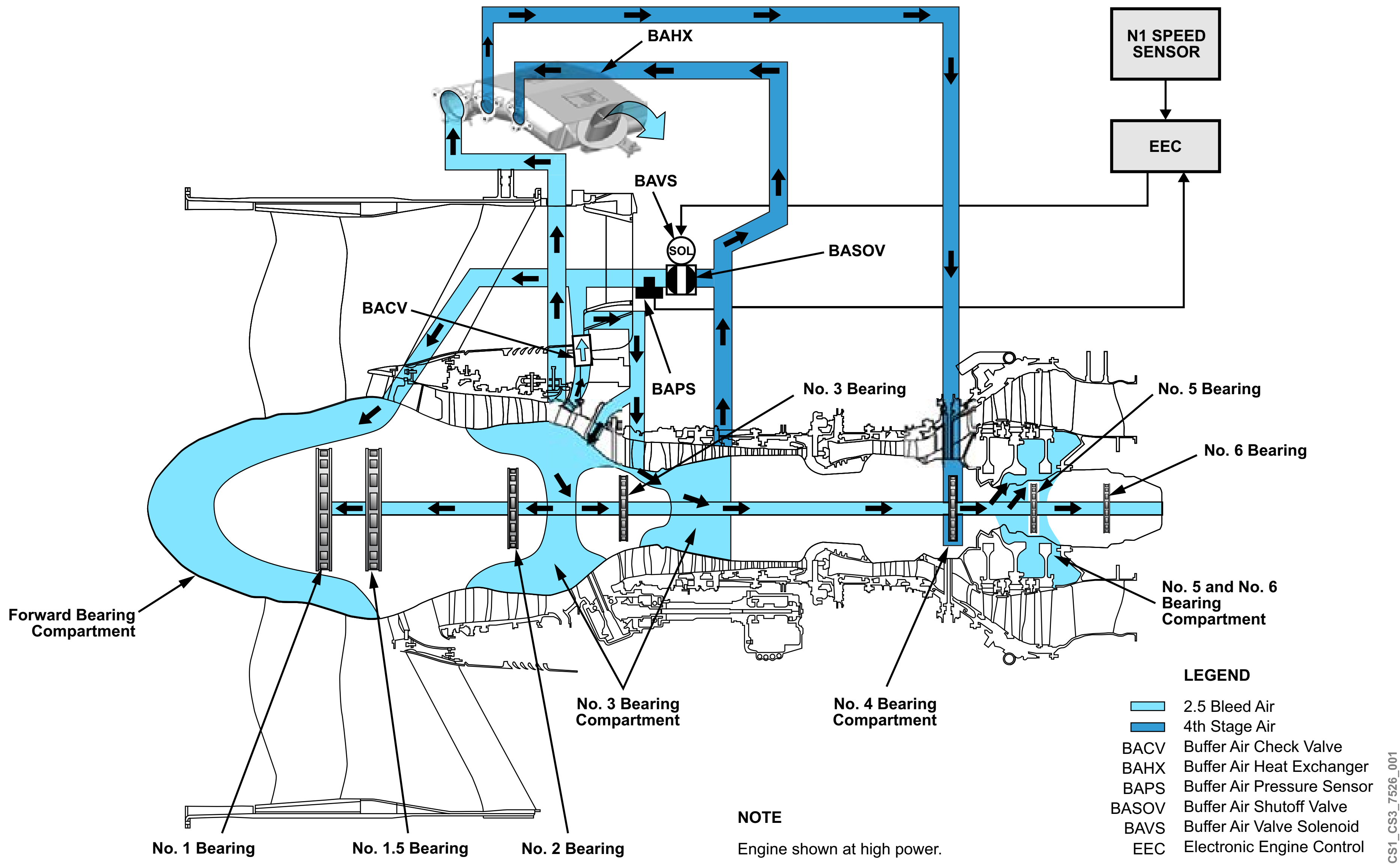


Figure 5: Buffer Air System

COMPONENT LOCATION

The buffer air cooling system consists of the following components:

- Buffer air heat exchanger (BAHX)
- Buffer air shutoff valve (BASOV)
- Buffer air shutoff valve solenoid (BAVS)
- Buffer air pressure sensor (BAPS)
- Buffer air check valve (BACV)

BUFFER AIR HEAT EXCHANGER

The buffer air heat exchanger (BAHX) is located on the fan intermediate case in the 2.5 bleed cavity at the 12:00 o'clock position.

BUFFER AIR SHUTOFF VALVE

The buffer air shutoff valve (BASOV) is located on the rear of the high-pressure compressor (HPC) at the 11 o'clock position.

BUFFER AIR SHUTOFF VALVE SOLENOID

The buffer air shutoff valve solenoid (BAVS) is located on the right side of the HPC case at the 1 o'clock position.

BUFFER AIR PRESSURE SENSOR

The buffer air pressure sensor (BAPS) is located below the BAHX at the 2 o'clock position.

BUFFER AIR CHECK VALVE

The buffer air check valve (BACV) is located at the 12 o'clock position on the fan intermediate case.

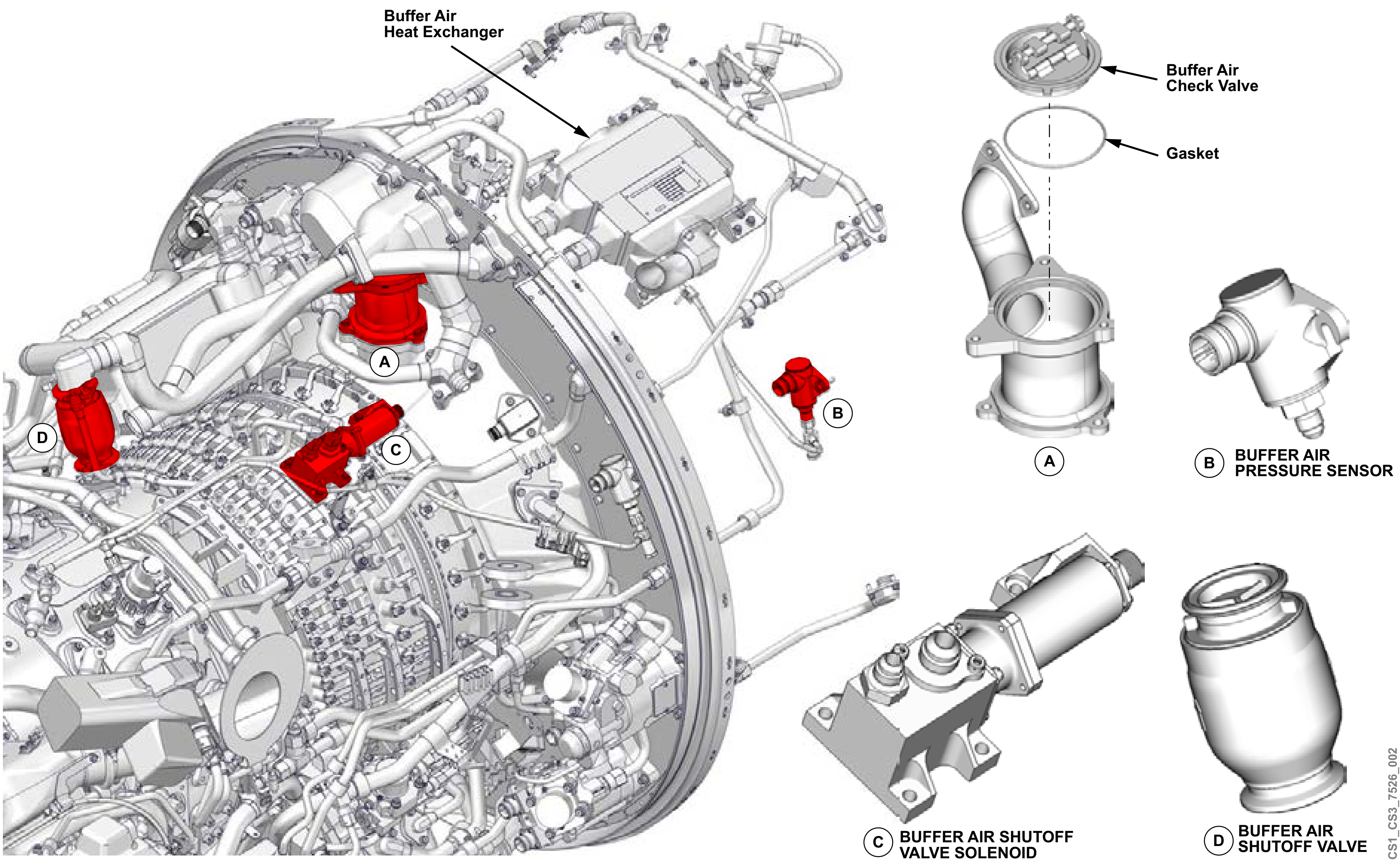


Figure 6: Buffer Air System Component Location

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the buffer air system operations.

CAS MESSAGES

Table 3: ADVISORY Message

MESSAGE	LOGIC
L (R) ENG FAULT	Loss of redundant or non-critical function for the left (right) engine. Refer to INFO messages.

Table 4: INFO Messages

MESSAGE	LOGIC
75 L (R) ENGINE FAULT - BAV INOP	INFO message is set by the left (right) engine when the buffer air shutoff valve is failed closed.
75 L (R) ENGINE FAULT - BACV FAIL CLSD	INFO message is set by the left (right) engine when the buffer air check valve is failed closed.

75-31 VARIABLE STATOR VANE SYSTEM

GENERAL DESCRIPTION

The variable stator vane system adjusts the position of the low-pressure compressor (LPC) inlet guide vanes and the high-pressure compressor (HPC) variable stator vanes during engine operation. This optimizes engine performance.

LOW-PRESSURE COMPRESSOR

The electronic engine control (EEC) controls the LPC stator vane actuator position through a torque motor. This controls a servovalve mounted on the actuator. A linear variable differential transformer (LVDT) provides feedback to the EEC. The LPC stator vane actuator is fuel actuated using servo fuel supplied by the integrated fuel pump and control (IFPC).

The LPC stator vane actuator moves the LPC inlet guide vane mechanically through a bellcrank and synchronizing ring.

The LPC inlet guide vane moves between the open and closed position, based on N1 speed. The LPC inlet guide vanes move toward the closed position during engine start up and idle, modulates during transient operations, and are open above idle power.

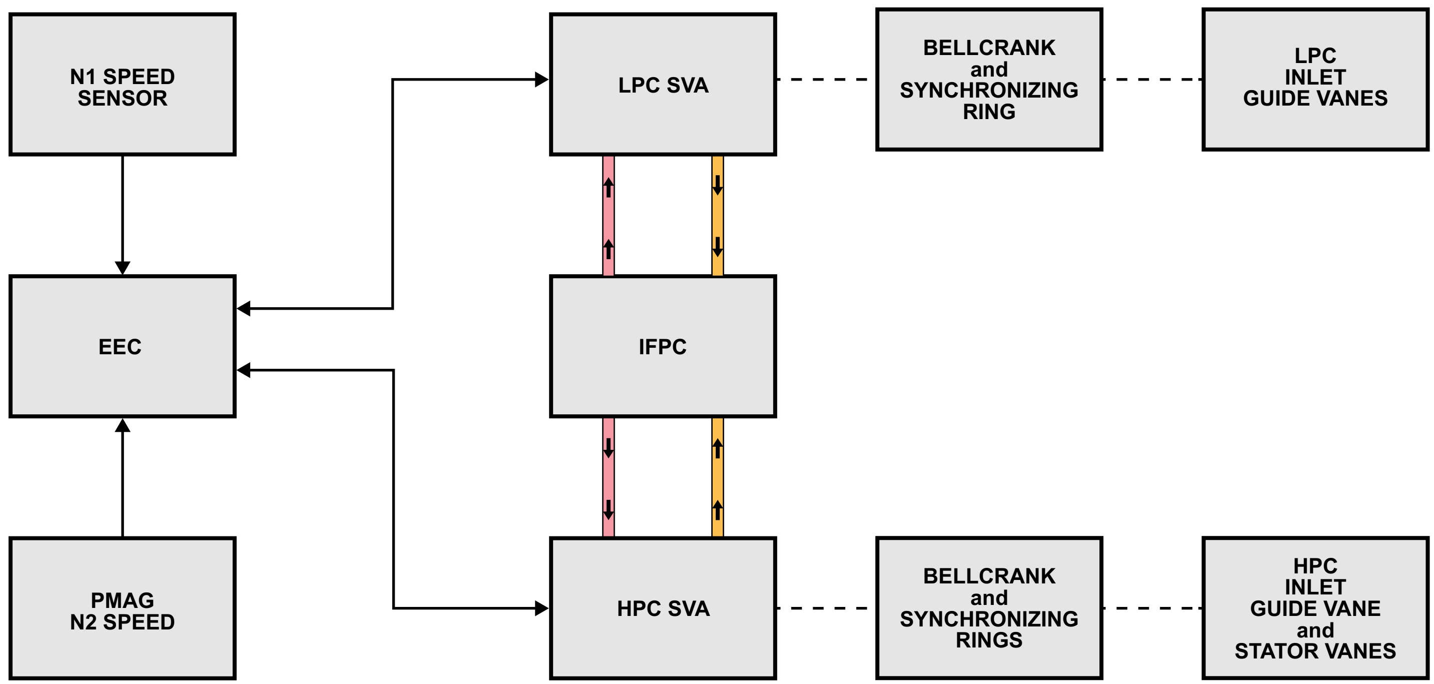
HIGH-PRESSURE COMPRESSOR

The EEC controls the HPC stator vane actuator position through a torque motor. This controls a servovalve mounted on the actuator. A LVDT provides feedback to the EEC. The HPC stator vane actuator is fuel actuated using servo fuel, supplied by the IFPC.

The HPC stator vane actuator moves the HPC inlet guide vane and the stator vanes of the first three stages mechanically through a bellcrank and synchronizing ring.

The HPC stator vanes move between the open and closed position based on N2 speed. The HPC variable stator vanes are closed during engine startup and engine idle, and are fully open at takeoff power.

The actuators require no rigging during installation. They are designed to fail in the open position to allow maximum airflow through the compressors.



LEGEND

Servo Fuel
Servo Fuel Return

EEC	Electronic Engine Control
HPC	High-Pressure Compressor
IFPC	Integrated Fuel Pump and Control
LPC	Low-Pressure Compressor
PMAG	Permanent Magnet Alternator Generator
SVA	Stator Vane Actuator
- - - -	Mechanical Link

Figure 7: Variable Stator Vane System

COMPONENT LOCATION

The variable stator vane control system consists of the following components:

- LPC inlet guide vane bellcrank and synchronizing ring
- LPC stator vane actuator
- HPC stator vane bellcrank and synchronizing rings
- HPC stator vane actuator

LOW-PRESSURE COMPRESSOR INLET GUIDE VANE BELLCRANK AND SYNCHRONIZING RING

The LPC inlet guide vane (IGV) bellcrank and synchronizing ring is mounted around the engine at the inlet of the LPC.

LOW-PRESSURE COMPRESSOR STATOR VANE ACTUATOR

The LPC SVA is mounted at the 5 o'clock position on the compressor intermediate case (CIC) firewall.

HIGH-PRESSURE STATOR VANE BELLCRANK AND SYNCHRONIZING RINGS

The HPC stator vane bellcrank and synchronizing rings are mounted on the HPC.

HIGH-PRESSURE COMPRESSOR STATOR VANE ACTUATOR

The HPC stator vane actuator is mounted on the HPC case on the right side of the engine at the 4 o'clock position.

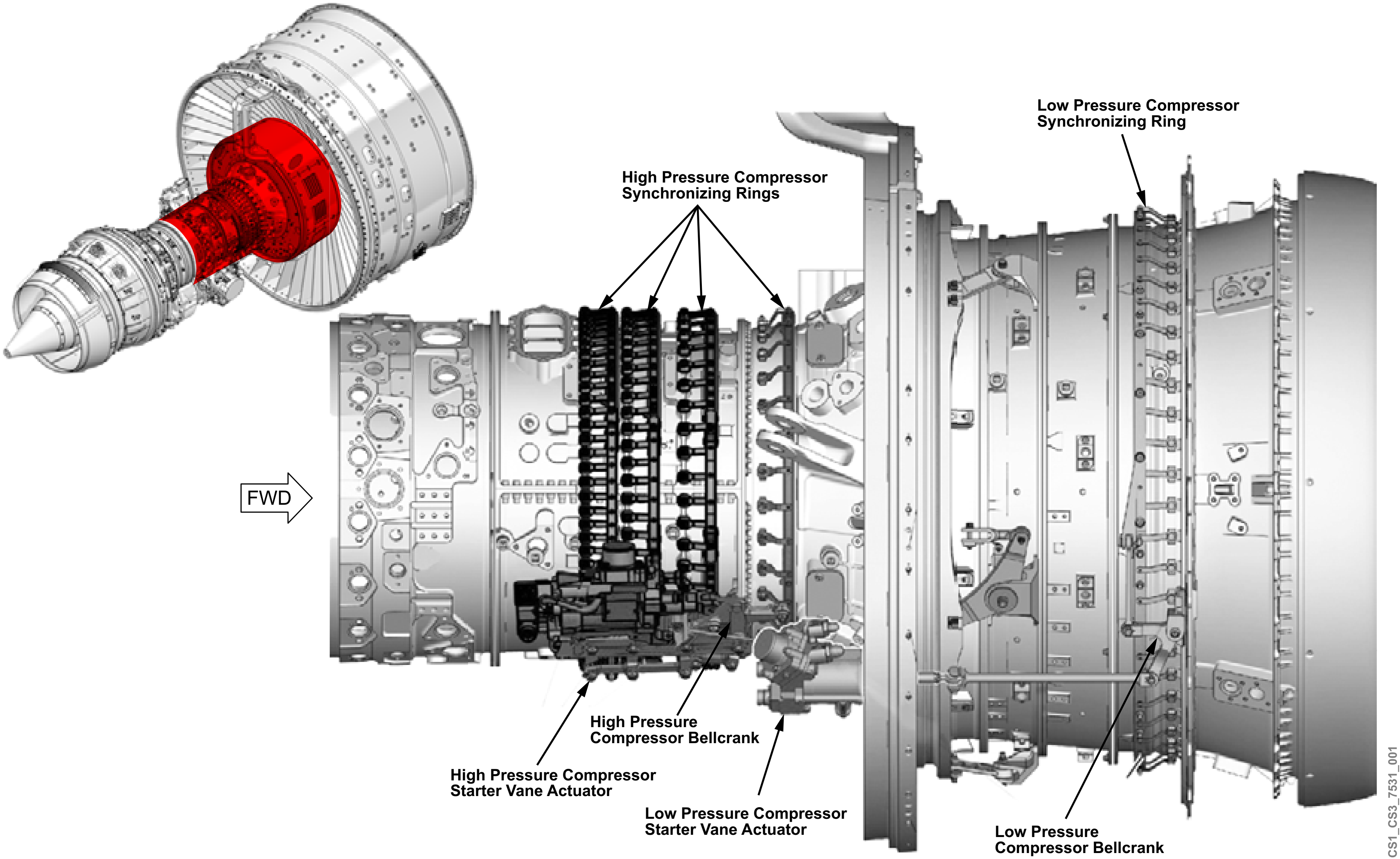


Figure 8: Variable Stator Vane Control System Component Location

DETAILED DESCRIPTION

Either channel of the EEC controls the LPC SVA through a dual-coil torque motor which controls an internal servovalve. The position of the LPC SVA is sensed by a dual-coil linear variable differential transformer (LVDT). The LVDT is used to transmit the piston position to each electronic engine control (EEC) channel. If there is a power failure to the LPC SVA, the actuator positions the inlet guide vanes in the open position for maximum airflow through the LPC. The LPC SVA is positioned based on the N1 speed sensor signal.

Either channel of the EEC control the HPC SVA through a dual-coil torque motor that controls an internal servovalve. The position of the HPC SVA is sensed by a dual-coil LVDT. The LVDT is used to transmit the piston position to each EEC channel. If there is a power failure to the LPC SVA, the actuator positions the inlet guide vanes and stator vanes in the open position for maximum airflow through the HPC. The HPC SVA is positioned based on the N2 speed sensing information received from the permanent magnet alternator generator (PMAG).

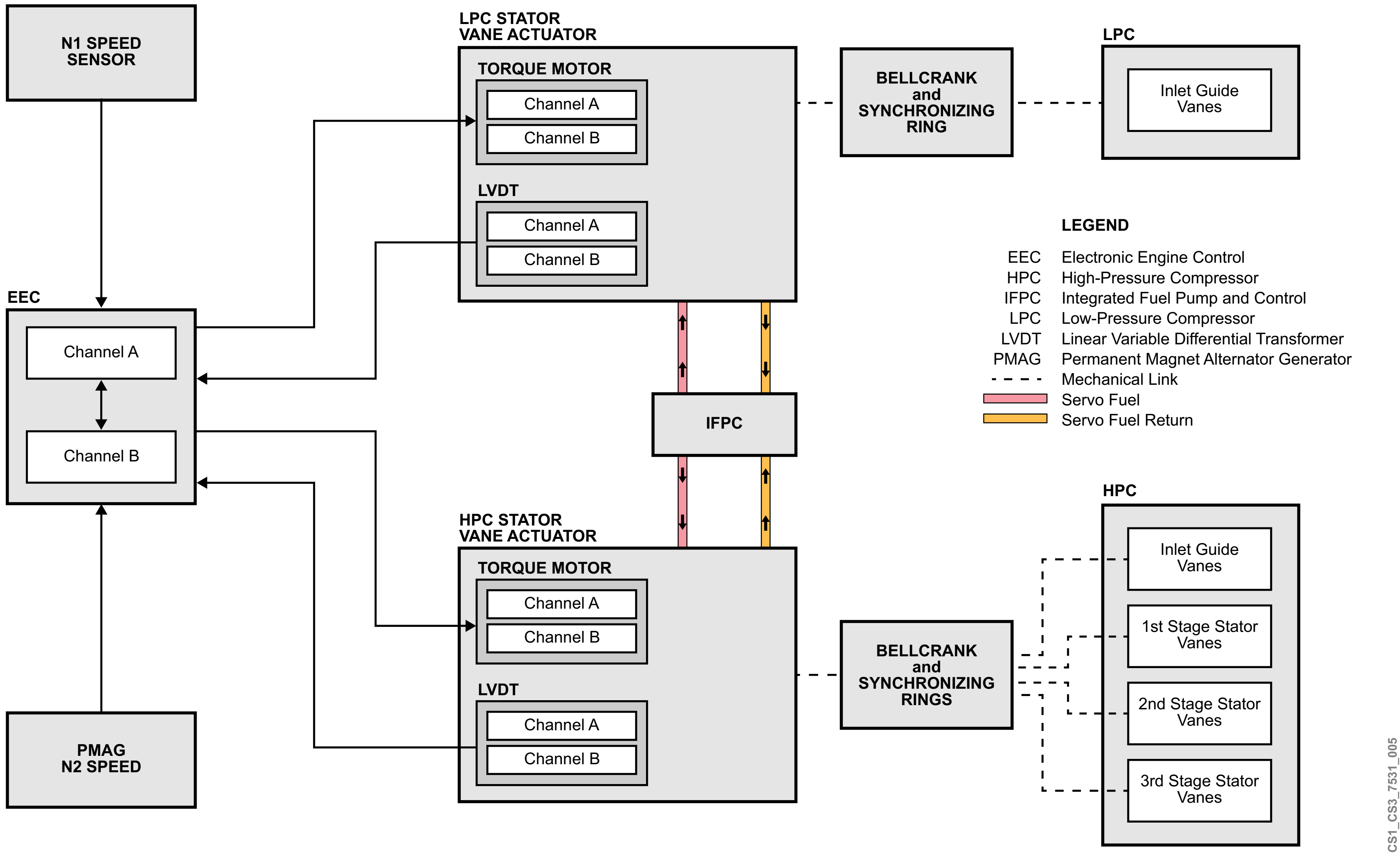


Figure 9: Variable Stator Vane Control Detailed Description

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) messages for air system operations.

CAS MESSAGES

Table 5: CAUTION Messages

MESSAGE	LOGIC
L ENG OPER DEGRADED	Uncertainty on T2/TAT (degraded or default value), or HPC stator vane actuator not tracking as it should.
R ENG OPER DEGRADED	Uncertainty on T2/TAT (degraded or default value), or HPC stator vane actuator not tracking as it should.

75-32 COMPRESSOR BLEED AIR SYSTEM

GENERAL DESCRIPTION

The engine has a compressor bleed air system (BAS) to provide greater compressor stability during starting and engine transient operation. Air is bled from the rear of the low-pressure compressor (LPC) at station 2.5, and from one 6th stage bleed port in the high-pressure compressor (HPC).

LOW-PRESSURE COMPRESSOR BLEED AIR SYSTEM

The low-pressure compressor BAS uses a bleed valve. It is located at the exit of the LPC at station 2.5, and provides an improved surge margin during starting, low power, and transient operation. The bleed valve forms a ring around the LPC case. The bleed valve extracts dirt, rain, and ice during ground operation. Air from the bleed valve is vented to the fan air stream.

The 2.5 bleed actuator has an integral torque motor assembly that controls servo fuel pressure from the integrated fuel pump and control (IFPC) to position the actuator. The 2.5 bleed valve is attached to the LPC just forward of the compressor intermediate case (CIC). The actuator acts on the bellcrank to open and close the valve. The valve is fully modulated through the electronic engine control (EEC) commands.

The 2.5 bleed valve actuator is scheduled based on N1 speed and air data information. The 2.5 bleed valve actuator moves the 2.5 bleed valve ring to the open position during startup and idle, modulates during transient operations, and closes at takeoff. The failsafe position of this bleed is in the closed position.

In the event of an engine surge, the EEC:

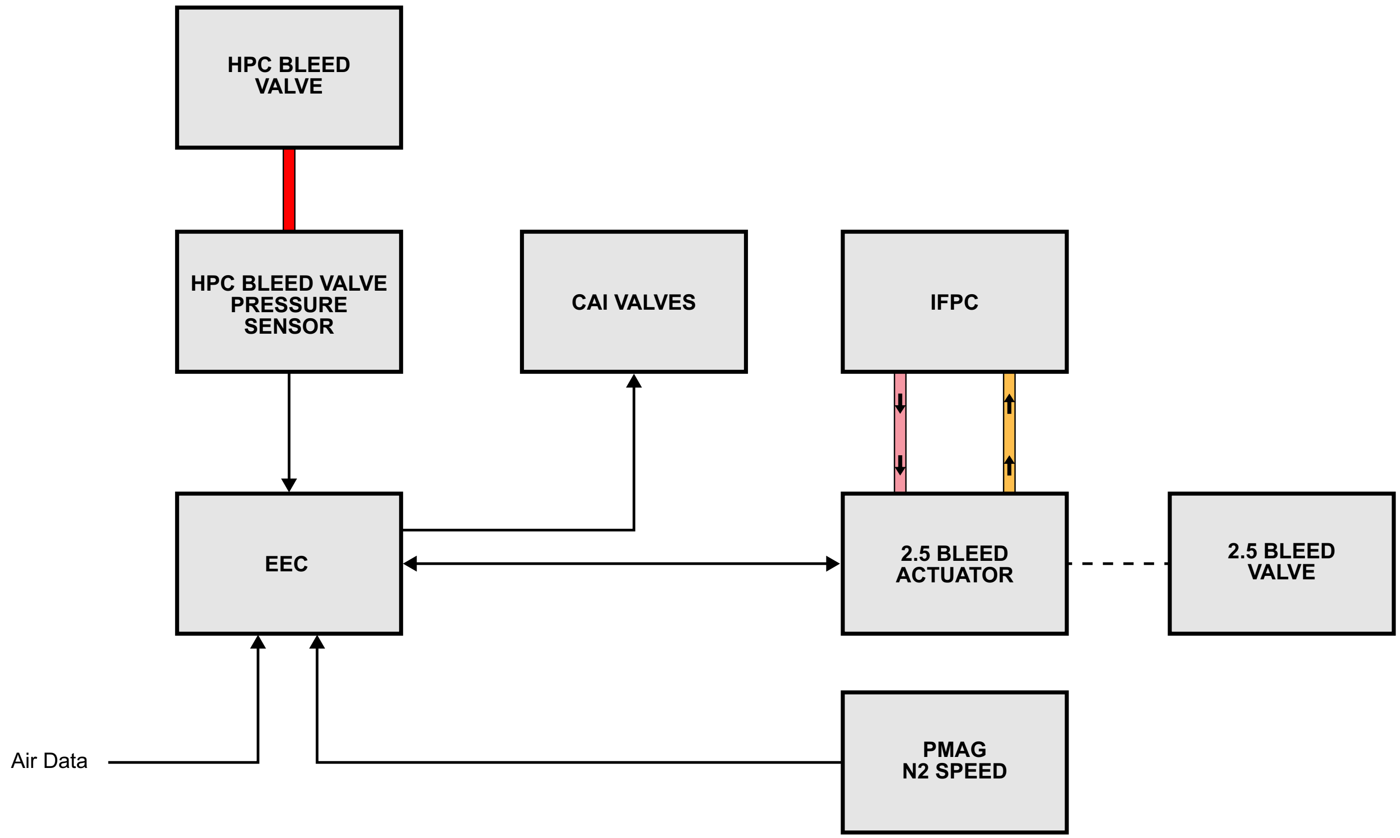
- Opens the bleed valve
- Activates continuous ignition on both igniters
- Resets the variable stator vanes to recover from the surge condition.

When the surge detection signal clears, bleed valve and the variable stator vanes are returned to their normal positions, and continuous ignition is turned off after a timer expires.

HIGH-PRESSURE COMPRESSOR BLEED AIR SYSTEM

The high-pressure compressor (HPC) BAS discharges the 6th stage compressor air into the engine core area. The HPC bleed valve is augmented by the cowl anti-ice valves (CAIVs), which open during engine start and discharges air through the cowl anti-ice (CAI) ducting. For more information on the cowl anti-ice system (CAIS), refer to ATA 30 Ice and Rain Protection.

The HPC bleed valve is a two-position poppet-type valve. The valve is spring-loaded to the open position during engine starting. The valve is closed using station 2.9 HPC air. The valve closes when HPC pressure is high enough to overcome the spring-force, and closes fully at idle power. The HPC bleed valve sensor provides the EEC with feedback on the position of the HPC bleed valve.



LEGEND

Bleed Air
Servo Fuel
Servo Fuel Return
CAI Cowl Anti-Ice
EEC Electronic Engine Control
HPC High-Pressure Compressor
IFPC Integrated Fuel Pump and Control
PMAG Permanent Magnet Alternator Generator
- - - - Mechanical Link

NOTE

2.5 bleed valve position is based on N1 speed.

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Figure 10: Compressor Bleed Air System

COMPONENT LOCATION

The compressor bleed air system consists of the following components:

- 2.5 bleed air valve
- 2.5 bleed valve actuator
- High-pressure compressor (HPC) bleed valve
- High-pressure compressor (HPC) bleed valve pressure sensor

2.5 BLEED AIR VALVE

The 2.5 bleed air valve is located between the low-pressure compressor (LPC) and the high-pressure compressor (HPC).

2.5 BLEED VALVE ACTUATOR

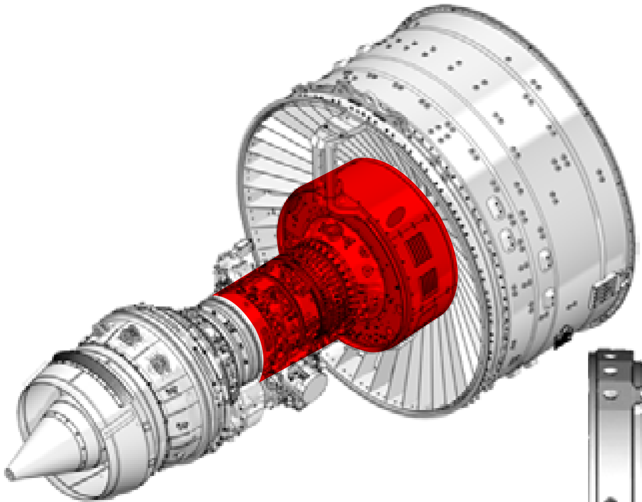
The 2.5 bleed valve actuator is located on the rear of the compressor intermediate case fireseal at the 3 o’clock position.

HIGH-PRESSURE COMPRESSOR BLEED VALVE

The high-pressure compressor (HPC) bleed valve is located on the HPC case at the 1 o’clock position.

HIGH-PRESSURE COMPRESSOR BLEED VALVE PRESSURE SENSOR

The high-pressure compressor (HPC) bleed valve pressure sensor is located at the 2 o’clock position on the compressor intermediate case fireseal.



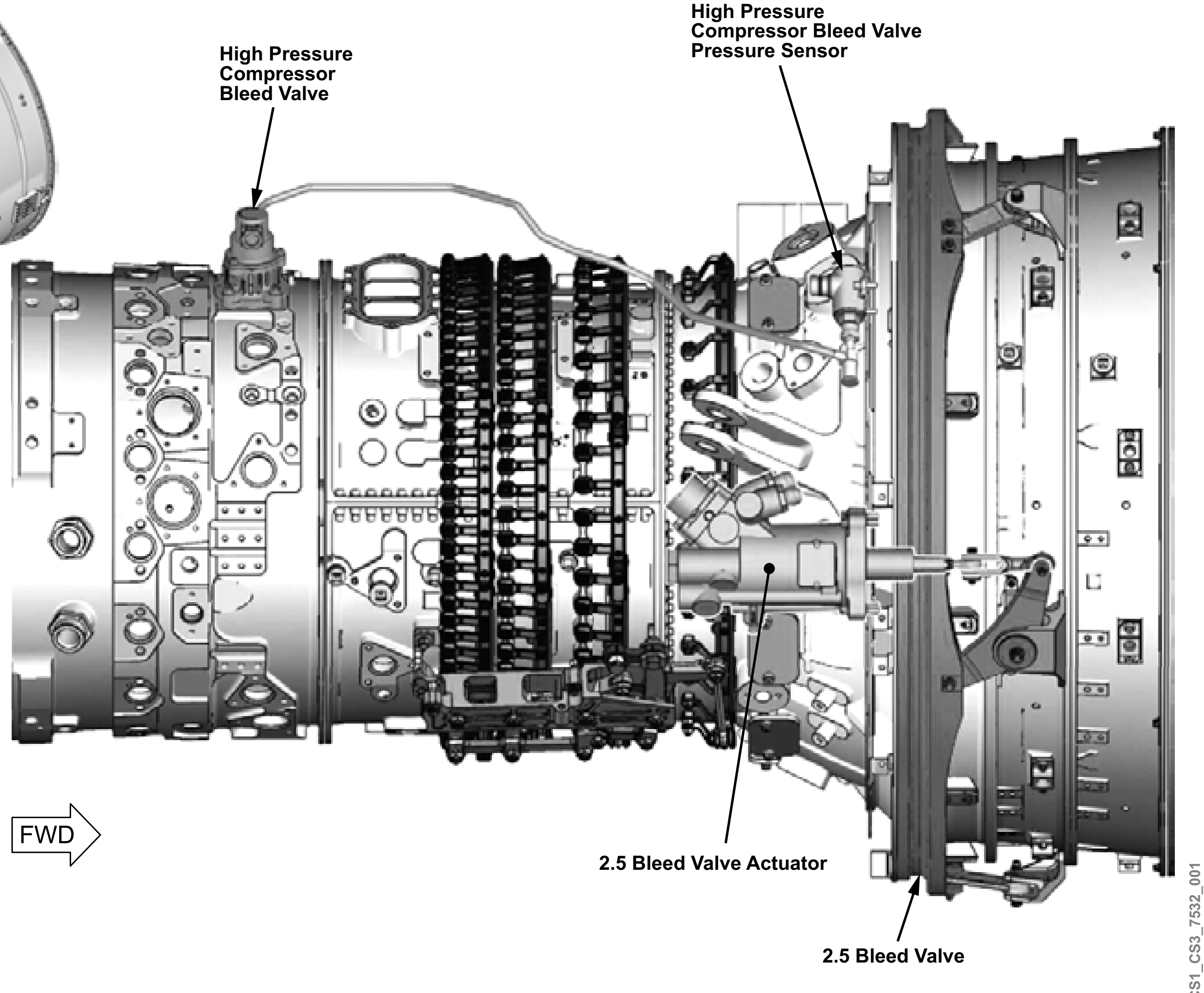


Figure 11: Compressor Bleed Air System Component Location

DETAILED DESCRIPTION

A dual-channel LVDT, mounted in the valve actuator, provides feedback to the EEC proportional to the position of the piston. The EEC directs the fuel pressure supplied through a dual-coil torque servomotor system. This drives the valve to the required position. Scheduling of the 2.5 bleed valve actuator is based on N1 speed.

For steady-state operation, the bleed valve is modulated between the fully opened and fully closed positions. The bleed valve position is scheduled as a function of N1 biased with altitude and Mach number. At takeoff power conditions, the bleed valve is fully closed. For starting and accelerations, the bleed valve follows the steady-state schedule. For decelerations, the bleed valve opens to protect LPC stability. In the event of an engine surge, the bleed valve opens to enhance recovery.

Upon detection of a surge during takeoff, the EEC assumes foreign object debris (FOD) and the bleed valve is partially opened to allow a 15% bleed flow.

Assuming there is no other damage, this results in normal engine parameters except for a higher exhaust gas temperature (EGT) in the range of 45°C. The bleed valve remains in this position until a power setting below approach idle is selected, or the aircraft is not in the takeoff configuration. When the surge clears, the 2.5 bleed is positioned based on the normal control scheduling, and EGT is restored to normal levels.

During engine start, the HPC bleed valve is open to unload the HPC. As the engine speed increases, the HPC moves to the closed position. In addition to the HPC bleed valve, the EEC also commands the cowl anti-ice valves (CAIV) open during start.

The HPC bleed valve pressure sensor sends a signal to the EEC to determine if the HPC bleed valve has failed in the open position, exposing the engine core to hot compressor air after the engine start. The sensor is also used by the EEC to determine if the HPC bleed valve has failed to close during engine start. If the valve is stuck closed during the engine start, the HPC could be overloaded, which could lead to an engine stall.

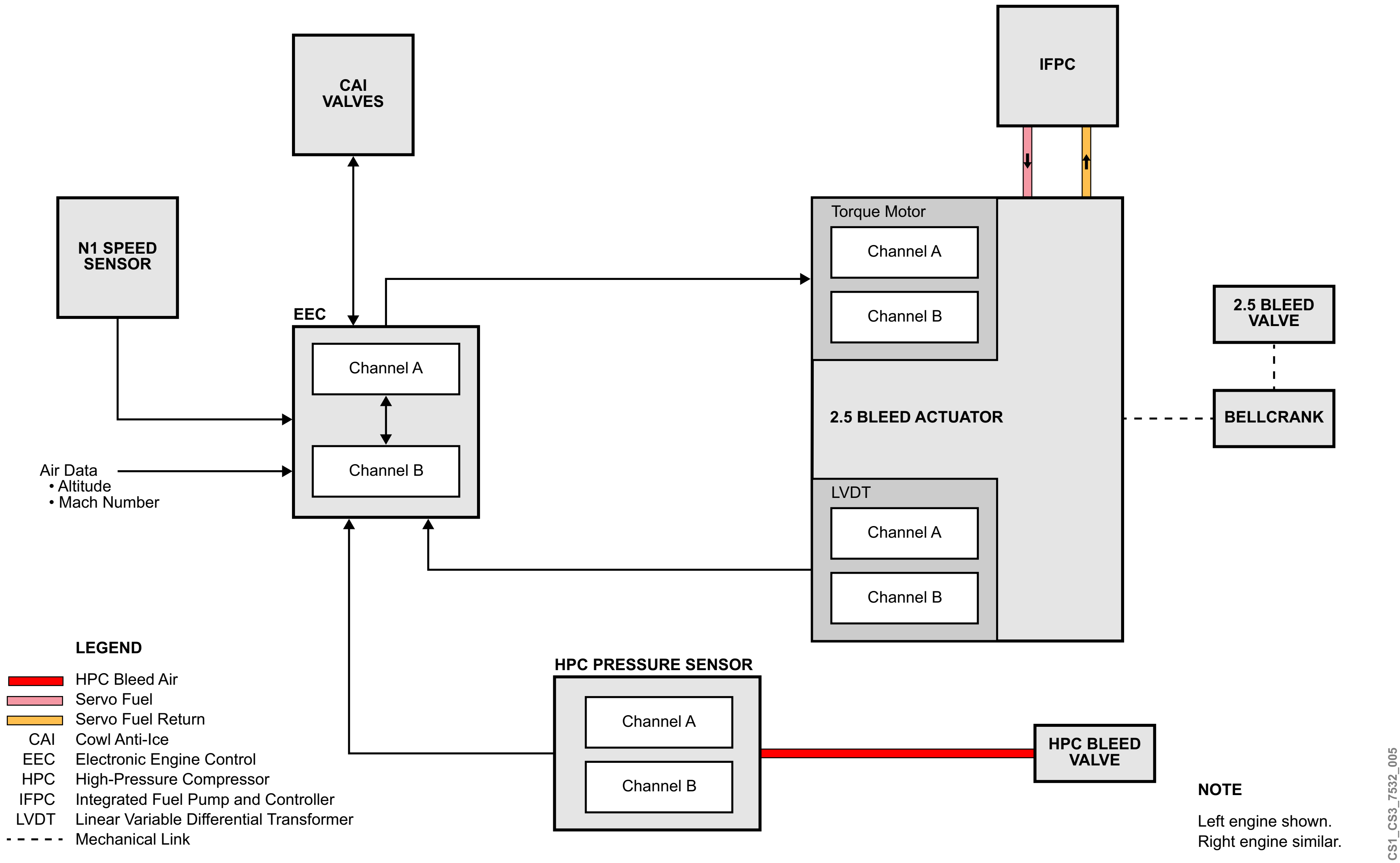


Figure 12: Compressor Bleed Air System Schematic

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the air system operations.

CAS MESSAGES

Table 6: CAUTION Messages

MESSAGE	LOGIC
L ENG NACELLE OVHT	Left burst duct or L HPC valve failed open, and automation failed to close it.
R ENG NACELLE OVHT	Right burst duct or R HPC valve failed open, and automation failed to close it.

Table 7: ADVISORY Message

MESSAGE	LOGIC
L (R) ENG FAULT	Loss of redundant or non-critical function for the left (right) engine. Refer to INFO messages.

Table 8: INFO Message

MESSAGE	LOGIC
75 L (R) ENGINE FAULT - HPC BLEED VLV INP	INFO message is set by the left (right) engine when the EEC active channel has detected that the HPC bleed valve has failed to close.

PRACTICAL ASPECTS

ELECTRONIC ENGINE CONTROL ACTUATOR TEST

The actuator test detects faults on any of the fuel controlled actuators. The test requires the engine to be dry motored.

The electronic engine control (EEC) performs interactive tests on the following components:

- The fuel metering valve in the IFPC
- The low-pressure compressor (LPC) variable stator vane actuator
- The high-pressure compressor (HPC) variable stator vane actuator
- The low-pressure compressor (LPC) 2.5 bleed valve actuator
- The turbine active clearance control (ACC) actuator

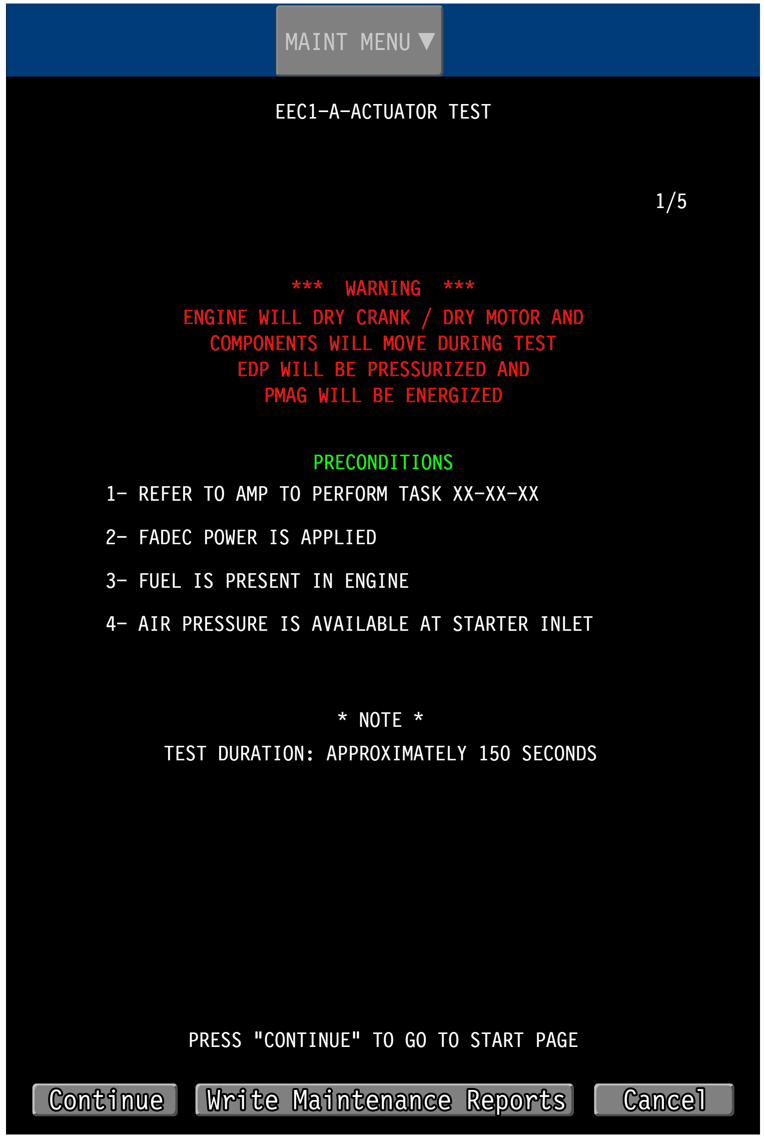


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Figure 13: Electronic Engine Control Actuator Test

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ATA 76 - Engine Controls

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ENGINE CONTROLS - CHAPTER BREAKDOWN

76-10 ENGINE CONTROL

GENERAL DESCRIPTION

Engine control is carried out using thrust levers that are located on the throttle quadrant assembly (TQA). Forward and reverse thrust control for each engine is provided by a single lever. Engine RUN switches, also located on the TQA, are used for commanding engine starts.

The left and right lever systems of the TQA are completely independent of each other.

Each thrust lever supplies thrust command signals to the electronic engine control (EEC). Thrust lever position information is sent to the autothrottle system in the data concentrator unit module cabinets (DMCs), through the brake data concentrator unit (BDCU).

Thrust command signals are used by the EEC to calculate the fuel scheduling required to achieve the desired thrust. The EEC then outputs a command to the integrated fuel pump and control (IFPC) to schedule the fuel flow.

When starting the engine, the engine RUN switches on the TQA are moved to the RUN position. A signal is sent to the EEC to command the starter air valve (SAV), IFPC, and ignition system in the start sequence. Engine shutdown is initiated by moving the same engine RUN switch to OFF.

The thrust levers are either controlled automatically by the autothrottle system, or manually.

Thrust reversers (T/Rs) are deployed by moving the thrust levers into the reverse thrust range. Thrust reverser stow/deploy switches in the TQA detect that thrust reverse has been selected and supply signals to unlock the track lock units.

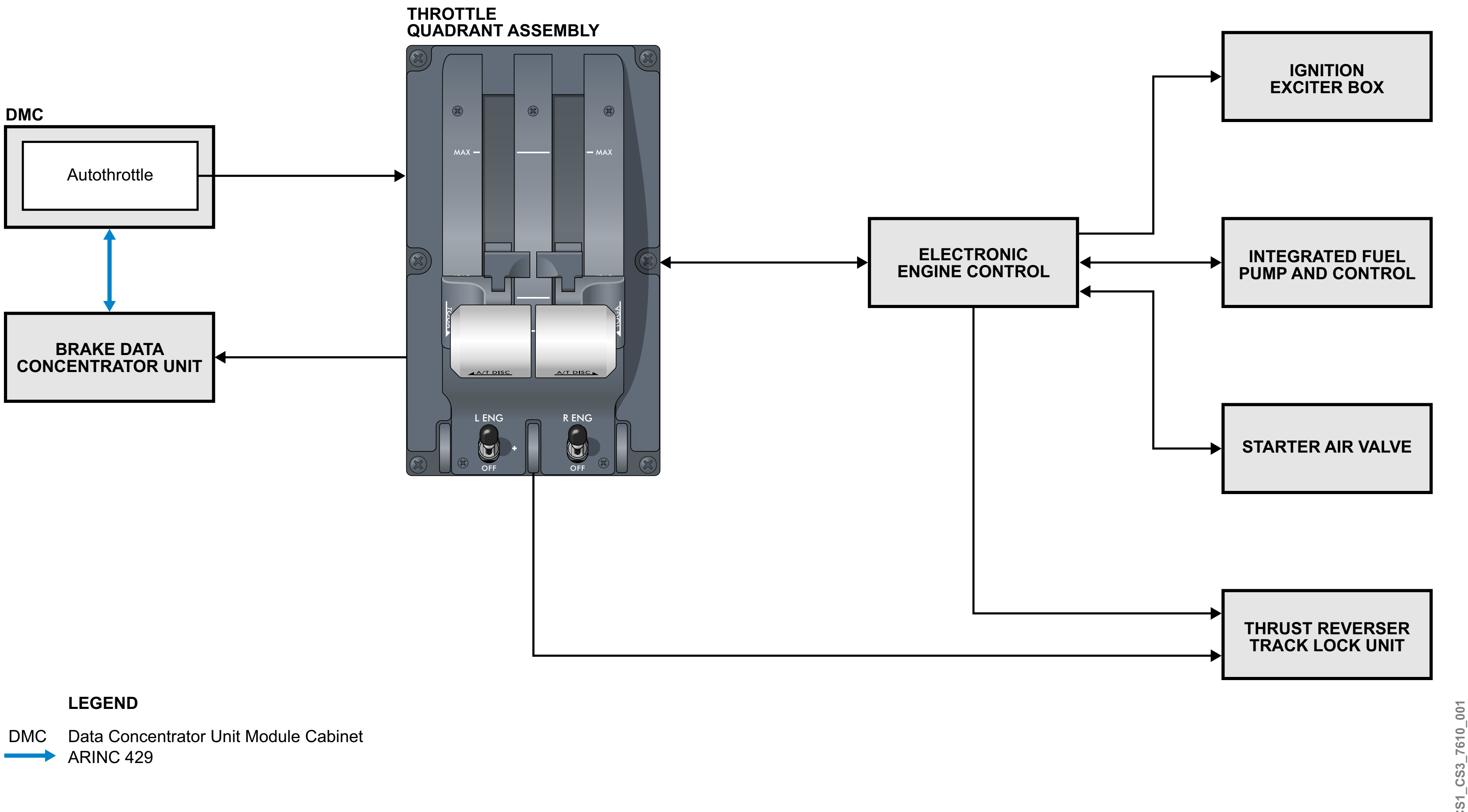
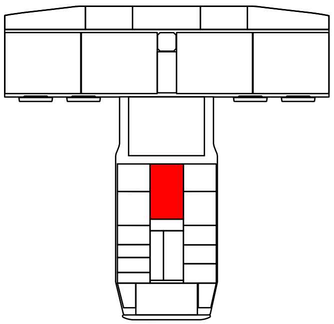


Figure 1: Engine Control System

COMPONENT LOCATION

The engine throttle quadrant assembly (TQA) is located on the fight deck center pedestal.



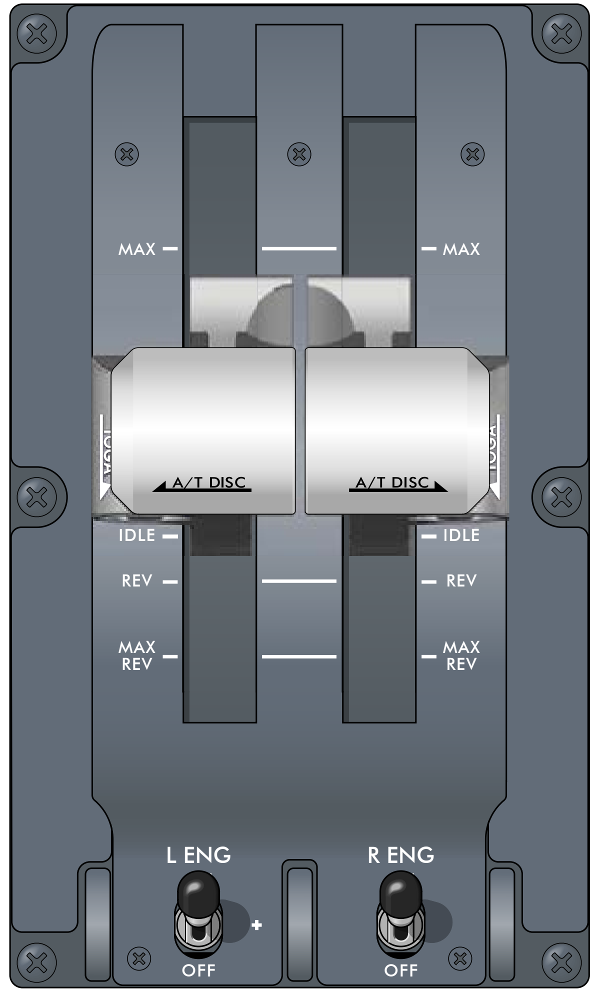


Figure 2: Throttle Quadrant Assembly Component Location

COMPONENT INFORMATION

THROTTLE QUADRANT ASSEMBLY

The throttle quadrant assembly (TQA) is a line replaceable unit (LRU) that houses both engine thrust levers and the L(R) ENG switches. The components that are part of the TQA are:

- Thrust levers
- ENG switches
- RVDTs
- Baulk solenoids
- Thrust reverser stow/deploy switches
- Autothrottle controller
- Autothrottle servomotors

Thrust Levers

The thrust levers travel a total of 67° from the MAX REV to MAX positions. the thrust levers have hard stops at three positions; IDLE, MAX and MAX REV. A friction clutch provides a feel force and maintains the thrust lever in the selected position.

The levers have an autothrottle disconnect switch and a takeoff/go-around (TOGA) switch built in.

Each thrust lever has a finger lift that allows the lever to move from idle to the reverse thrust position.

ENG Switches

The L(R) ENG switches initiate the engine start during an automatic start. During a manual start, they command fuel and ignition on. The L(R) ENG switches are selected OFF to shut down the engines.

Rotary Variable Differential Transformers

There are two types of rotary variable differential transformers (RVDTs) in the TQA, dual and single channel. Each thrust lever is connected to one of each through mechanical gears inside the TQA.

Baulk Solenoids

The baulk solenoids prevent the thrust levers from moving beyond the reverse idle position.

Thrust Reverser Stow/Deploy Switches

The thrust reverser stow/deploy switches are single pole double throw type. They are activated when the thrust levers move into the reverse range.

Autothrottle Controller

The autothrottle controller provides output to the autothrottle servomotor.

Autothrottle Servomotors

Each thrust lever has an autothrottle servomotor that moves the thrust levers in response to the autothrottle controller commands.

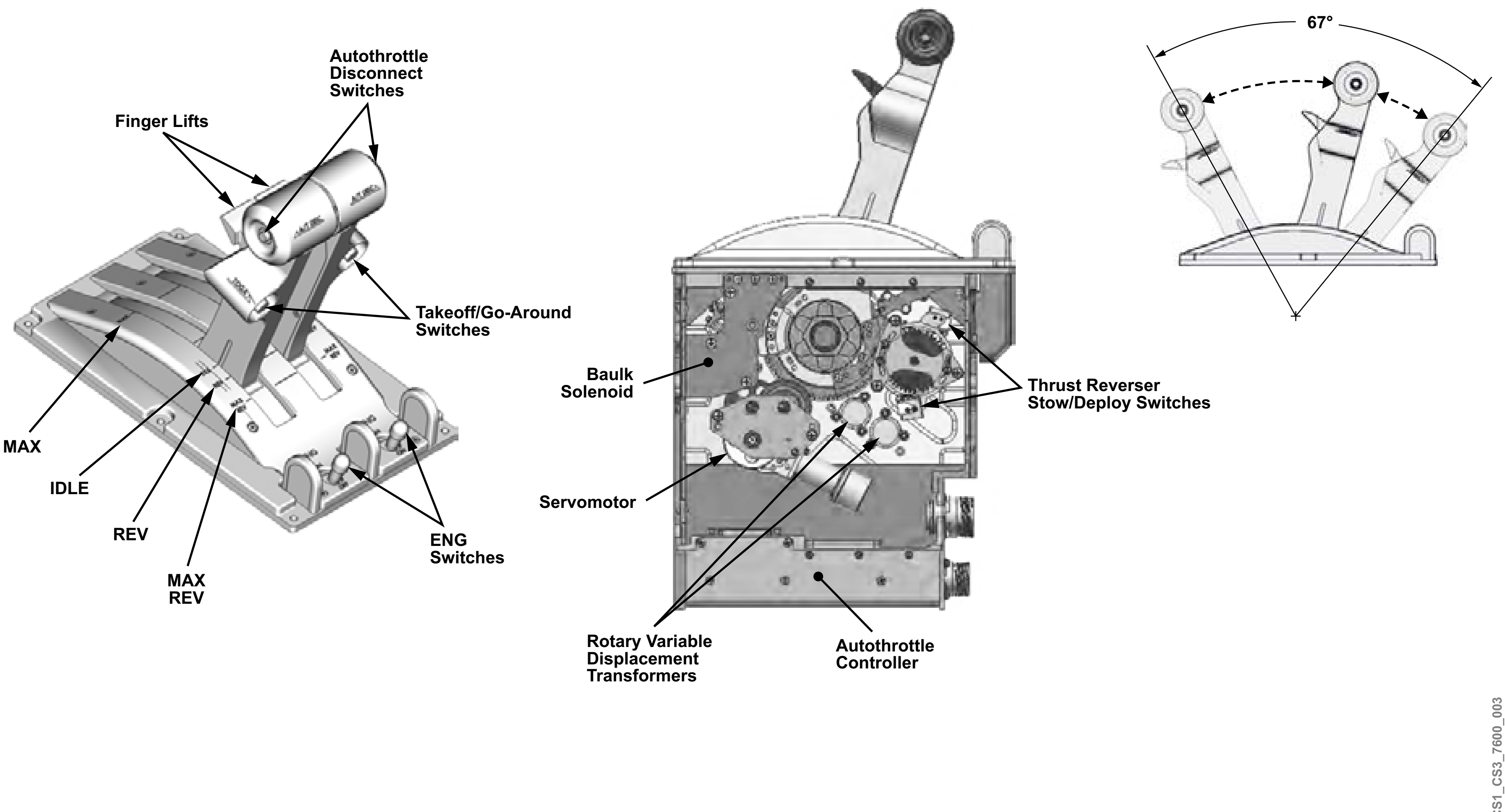


Figure 3: Throttle Quadrant Assembly

DETAILED DESCRIPTION

ENGINE CONTROL

Each throttle lever mechanically drives three electrically independent RVDTs housed into one dual-coil and one single coil housing. Excitation voltage for the dual RVDTs is provided by the electronic engine control (EEC). Excitation voltage for the single RVDT is provided by the brake data concentrator unit (BDCU).

The dual-coil RVDTs provide thrust lever position signal to the EEC. Channel A of the RVDT provides a signal for channel A of the EEC and channel B of the RVDT provides a signal for channel B of the EEC.

The third RVDT (single coil housing) interfaces with the BDCU to provide electrically independent, digitized thrust lever angle information. This information is available to all systems, including the power plant, through the data concentrator unit module cabinet (DMC).

A L (R) THROTTLE FAIL caution message is displayed when the EEC cannot determine the throttle position. The engine thrust is reduced to idle and the engine does not respond to thrust lever movement.

THRUST REVERSER CONTROL

The EEC provides a signal to control the thrust reverser (T/R) system baulk solenoid. When the T/Rs are deployed 85% of their travel, the EEC releases the baulk solenoid. This releases the thrust lever and allows full reverse thrust to be applied.

The TQA baulk solenoids receive power from the aircraft control and distribution cabinets (CDCs). The EEC provides both the logic and the ground for the solenoids operation.

The interlock, provided by the baulk solenoids, incorporates an override operation in both directions that requires 25 lb of force per lever to overcome.

The TQA has two stow/deploy microswitches per lever that control 28 VDC power to the thrust reverser track lock solenoids by providing the ground in circuit. These switches are hardwired to their respective control and distribution cabinets (CDCs).

If either the left or right thrust lever is moved into the reverse thrust range while the aircraft is in the air, a THROTTLE IN REVERSE caution message is displayed.

AUTOTHROTTLE CONTROL

The autothrottle controller receives power from the flight control panel (FCP) when autothrottle is engaged. When engaged, the autothrottle software sends throttle control information to the autothrottle controller inside the TQA. The autothrottle controller then sends the commands to the autothrottle servomotors inside the TQA to move the thrust levers.

The thrust levers are equipped with an autothrottle disconnect switch to disengage the autothrottle.

THRUST LEVER WARNING

If the thrust lever is not at idle when the engine is being started, the EEC initiates an aural warning to warn the crew, but does not prevent the engine from starting. The aural warning sounds “THRUST LEVER” three times.

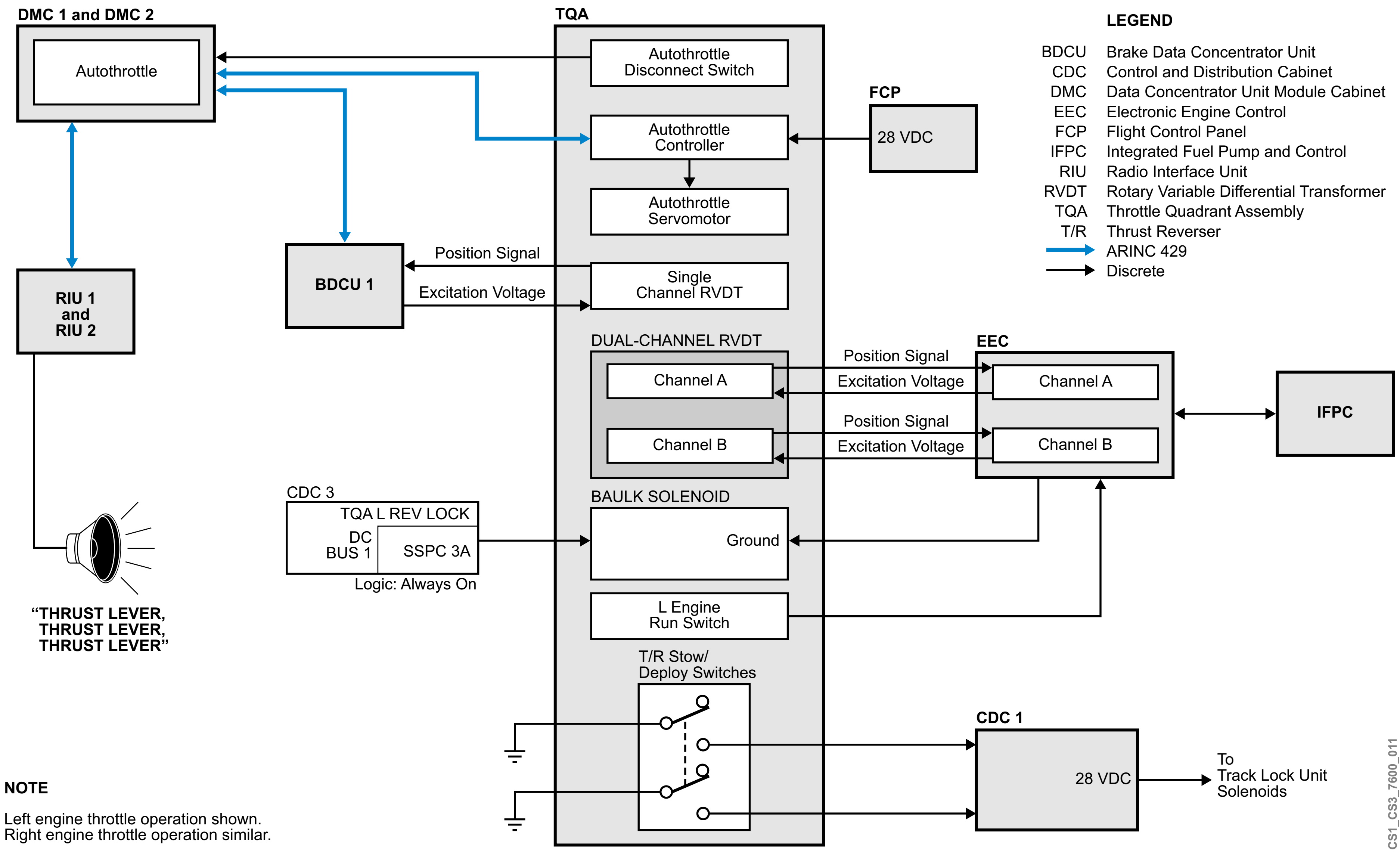


Figure 4: Engine Control System Schematic

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) messages associated with the engine control system.

CAS MESSAGES

Table 1: CAUTION Messages

MESSAGE	LOGIC
L THROTTLE FAIL	Left throttle position is not recognized by the EEC.
R THROTTLE FAIL	Right throttle position is not recognized by the EEC.
THROTTLE IN REVERSE	Left or right thrust reverser selected in flight.

Table 2: ADVISORY Messages

MESSAGE	LOGIC
L ENGINE FAULT	Loss of redundant or non-critical function for the left engine.
R ENGINE FAULT	Loss of redundant or non-critical function for the right engine.

Table 3: STATUS Messages

MESSAGE	LOGIC
L ENG SHUTDOWN	In-flight pilot commanded engine shut-down. Left engine run switch to OFF).
R ENG SHUTDOWN	In-flight pilot commanded engine shut-down. Right engine run switch to OFF.

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ATA 77 - Engine Indicating

BD-500-1A10
BD-500-1A11

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ENGINE INDICATING - CHAPTER BREAKDOWN

77-11 ENGINE POWER INDICATING SYSTEM

GENERAL DESCRIPTION

The engine power indicating system uses engine speed as a reference for engine power. The engine has three speed sensors that monitor engine speed:

- Fan speed probe
- N1 speed probe
- N2 speed sensor

The fan speed probe reads engine fan speed from the fan reduction gearbox front shaft gear.

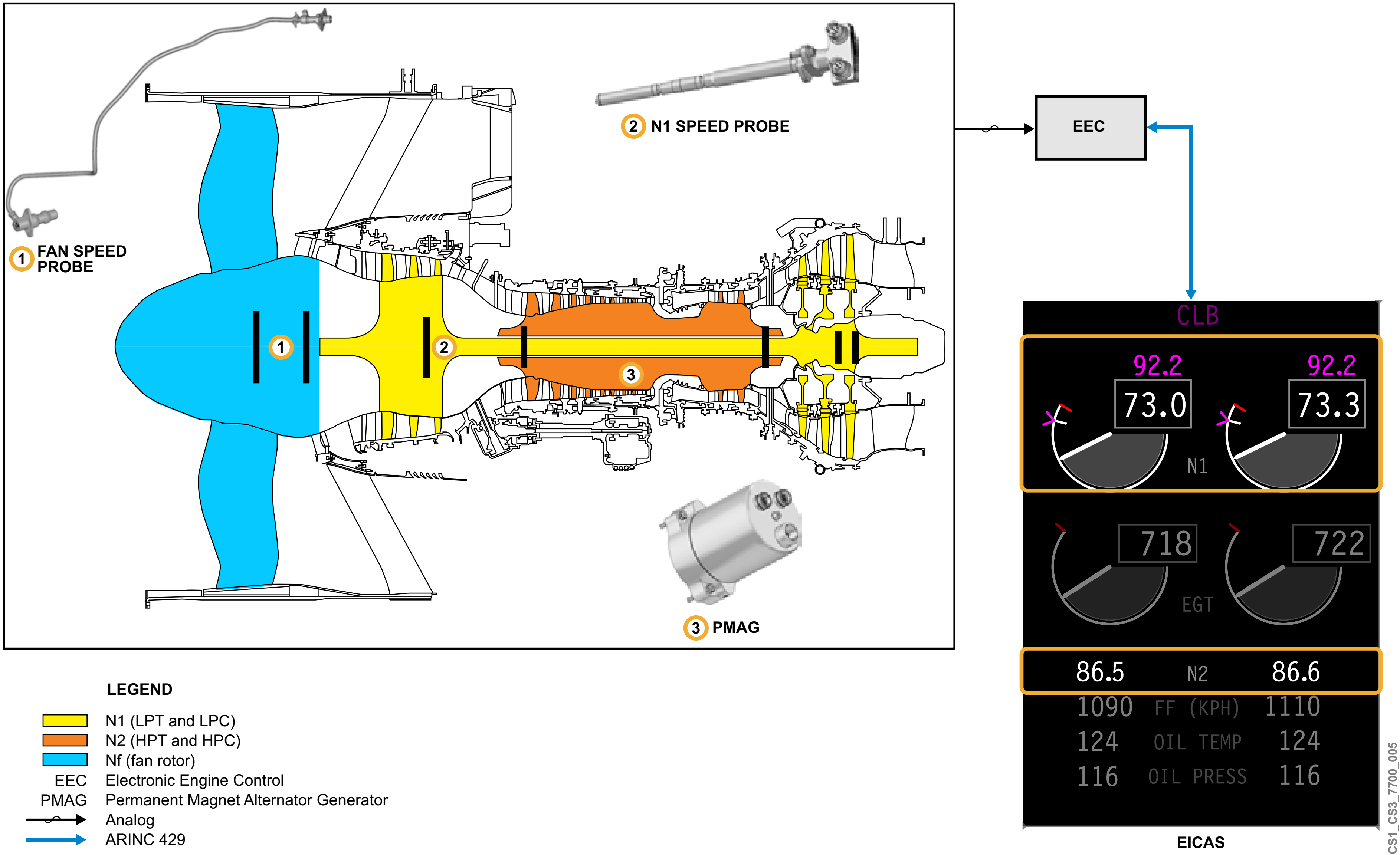
The N1 speed probe measures low-pressure compressor (LPC) speed.

The N1 gauge has analog and digital readouts. N1 is the primary thrust control parameter. The thrust reference, calculated as an N1 value, is displayed by a reference bug on the analog display, and in digital format above the digital readout box.

The N2 speed sensor measures high-pressure compressor (HPC) speed. The N2 speed sensor is part of the permanent magnet alternator generator (PMAG). The N2 is displayed in digital readout format only.

The electronic engine control (EEC) receives the speed sensor signals. The EEC uses the signals for engine control and outputs indication signals for the N1 and N2 engine indication and crew alerting system (EICAS) display.

The N1 gauges and the N2 digital readouts for both engines are displayed on EICAS. The fan speed is not displayed, but is used as backup in case of N1 signal failure.



LEGEND

(yellow)	N1 (LPT and LPC)
(orange)	N2 (HPT and HPC)
(blue)	Nf (fan rotor)
EEC	Electronic Engine Control
PMAG	Permanent Magnet Alternator Generator
(arrow)	Analog
(blue arrow)	ARINC 429

Figure 1: Engine Power Indicating System

COMPONENT LOCATION

The engine power indicating system consists of the following components:

- Fan speed probe
- N1 speed probe
- Permanent magnet alternator generator (PMAG)

NF SPEED SENSOR

The fan speed probe is located at the 1 o'clock position in the no.1 bearing support housing of the engine fan.

N1 SPEED PROBE

The N1 speed probe is mounted at station 2.5, at the rear of the compressor intermediate case at the 4:30 position.

PERMANENT MAGNET ALTERNATOR GENERATOR

The N2 speed coil is integral to the PMAG, which is located on the aft side of the main gearbox at the 7 o'clock position.

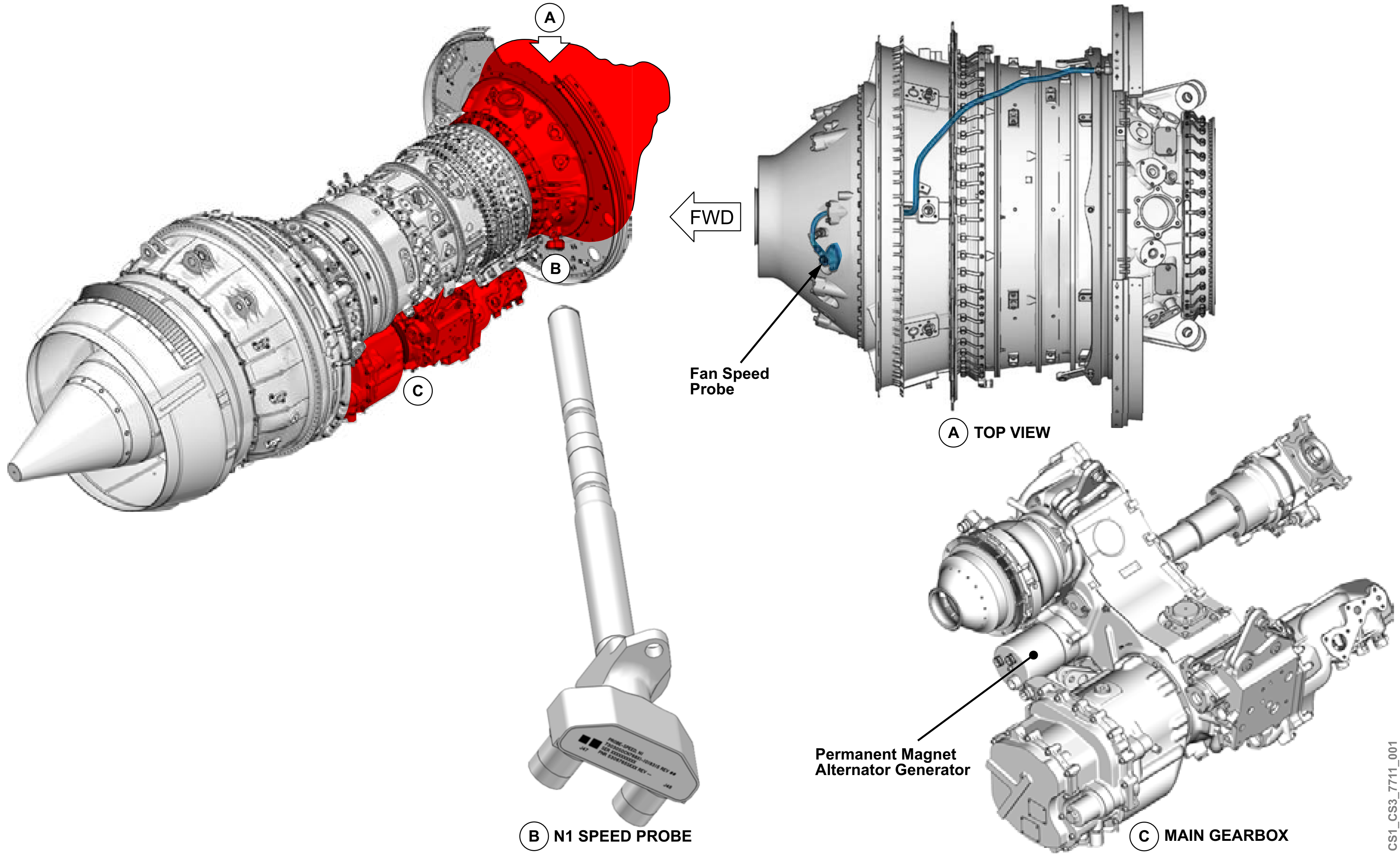


Figure 2: Engine Power Indicating System Component Location

COMPONENT INFORMATION

N1 SPEED PROBE

The N1 speed probe is a dual sensor with two independent, isolated coils, and electrical connectors.

The probe provides dual signals proportional to the rotational speed of the no. 2 bearing coupling nut located on the LPC. The EEC uses one signal per channel to generate the N1 value.

NOTE

The N1 probe is approximately 20 cm (8 in.) in length. Care should be taken during removal and installation to avoid damaging the tip.

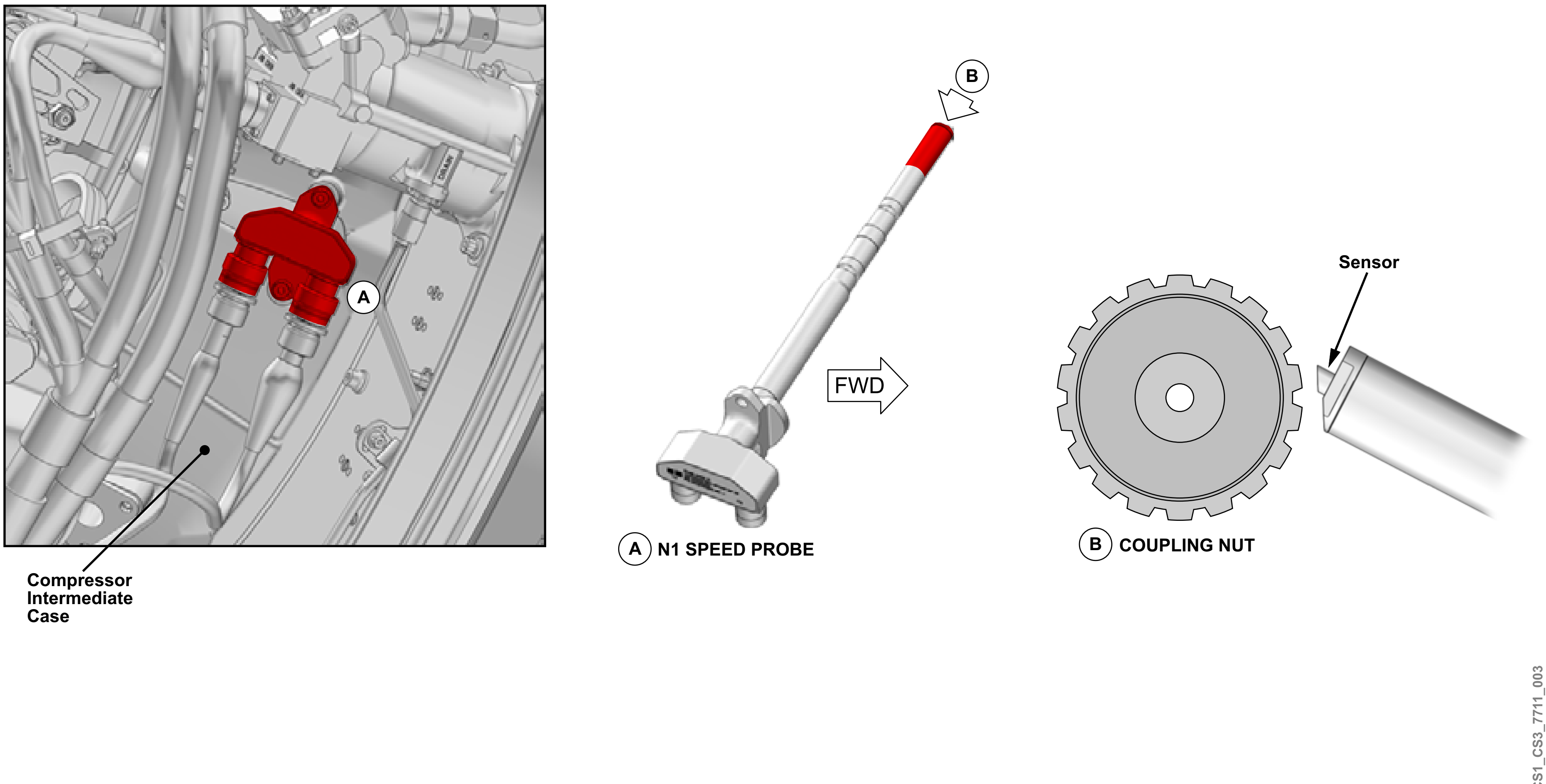
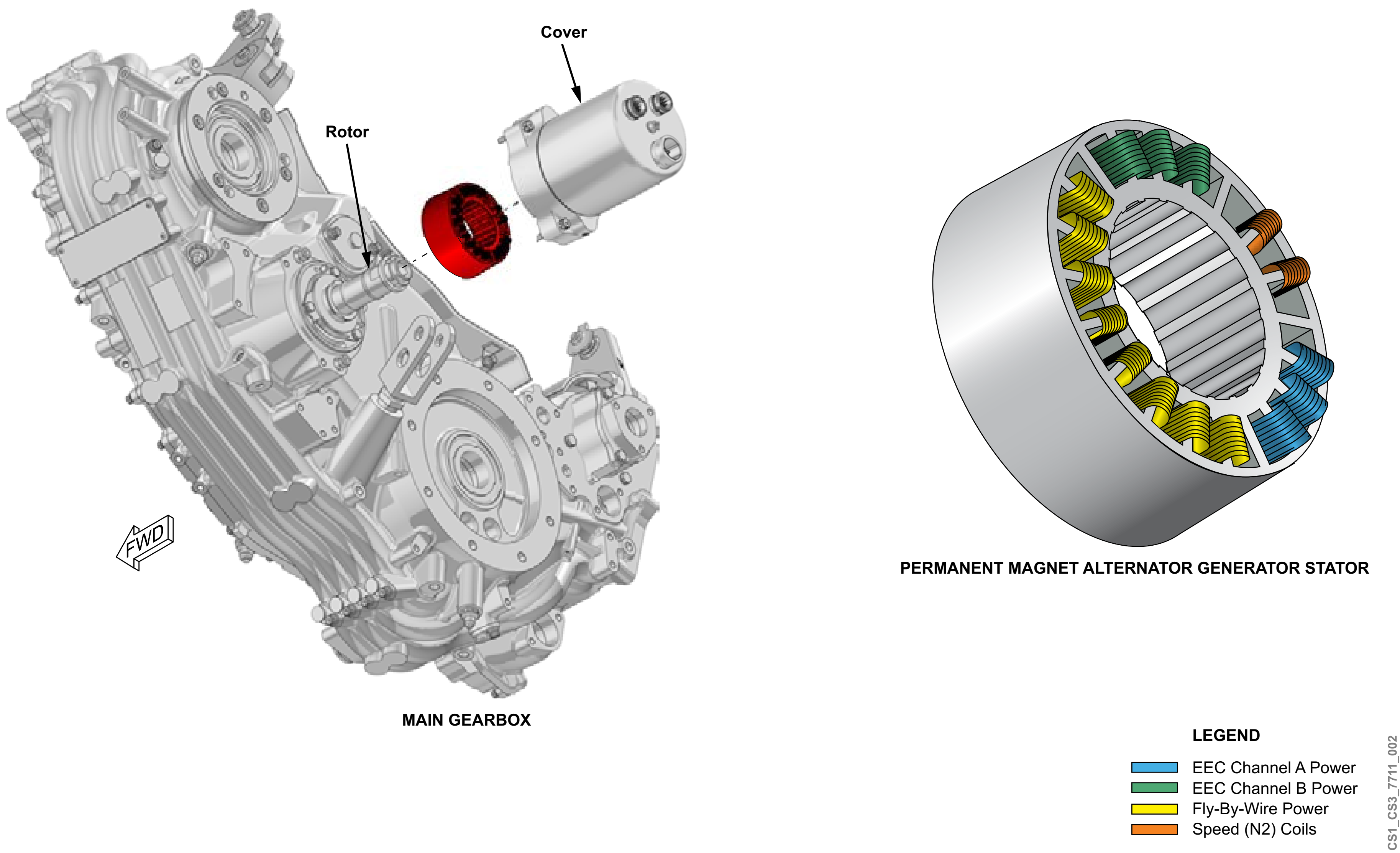


Figure 3: N1 Speed Probe

PERMANENT MAGNET ALTERNATOR GENERATOR

The PMAG stator has two speed coils that transmit two independent high rotor (N2) speed signals to the EEC. The PMAG stator also has dedicated coils for both EEC channel power generation, and fly-by-wire (FBW) power.

The PMAG has offset captive bolts and transfer tubes to ensure proper clocking upon installation.



LEGEND

- EEC Channel A Power
- EEC Channel B Power
- Fly-By-Wire Power
- Speed (N2) Coils

Figure 4: Permanent Magnet Alternator Generator

CONTROLS AND INDICATIONS

N1 INDICATION

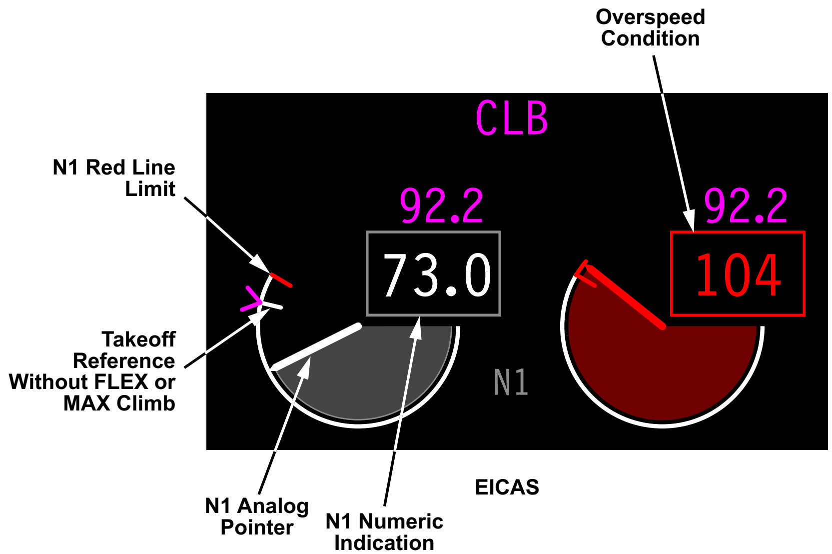
The N1 indication consists of an analog and digital readout. A low-pressure compressor (LPC) speed of 100% N1 is equivalent to 10,600 rpm.

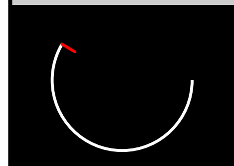
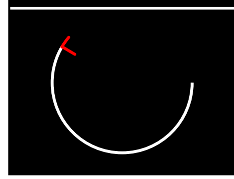
The analog gauge consists of a pointer in a round arc, representing the operating range of the engine. The area below the pointer is shaded to provide a quick visual reference. At the upper end of the arc is a red line limit, indicating 100% N1 speed.

The N1 digital display is in a white box to the right of the analog gauge.

If the N1 speed is greater than 100%, the pointer moves above the red line limit. The pointer and the shaded area below changes to red. The arc extends above the red line and remains there even if the overspeed condition no longer exists. This indicates that the engine speed has exceeded 100%. The digital display and box color changes to red when N1 is above 100%. The display and box changes to white if the overspeed condition no longer exists.

The N1 thrust reference bug readout is located on the outside of the analog N1 gauge arc.



N1 REDLINE	
	Normal red line limit
	N1 overspeed condition

N1 DIGITAL PARAMETER

Symbol	Description
86.5	Engine operation in normal range.
102	Engine operation above red line.
- - -	Invalid

CS1_CS3_7711_005

Figure 5: N1 Indication

N1 THRUST REFERENCE BUG

Other indications on the N1 display include an N1 thrust reference bug. The thrust reference bug is the best N1 for the selected thrust reference mode. The bug is displayed in magenta if the flight management system (FMS) calculates the thrust reference, or is displayed in cyan if it is entered manually into the FMS.

N1 THRUST REFERENCE

The N1 thrust reference is shown above the N1 digital display box. This value is displayed in magenta when calculated by the FMS, and in cyan when entered manually into the FMS.

N1 THRUST COMMAND CURSOR

The N1 thrust command cursor indicates the difference between the pointer and predicted N1 speed after the engine throttle is moved.

When in automatic power reserve (APR) mode, the thrust reference bug changes to green. An offset remains in steady-state since the actual N1 is higher due to APR.

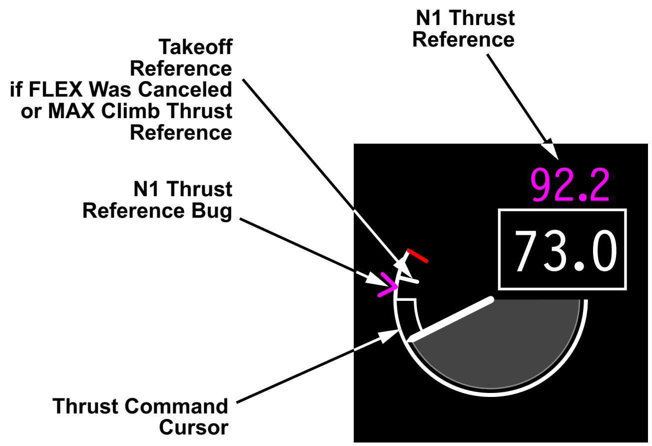
Thrust Limitation at Low Speed

The upper end of the thrust command cursor is shaded gray when thrust limitation at low speed is enabled during takeoff. The upper end of the gray area indicates the commanded N1, while the lower end indicates the thrust limitation. As the airspeed increases the gray area reduces and is removed when airspeed is above 80 kt.

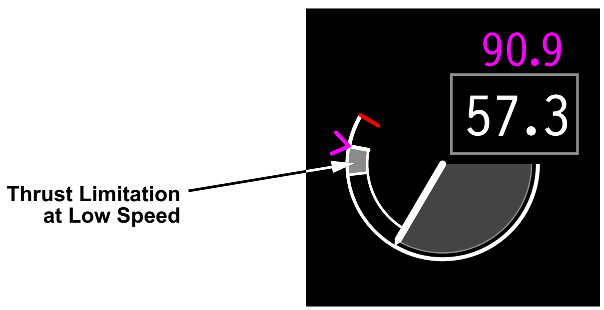
TAKEOFF REFERENCE WITHOUT FLEX OR MAX CLIMB

The white radial line shows the thrust reference if FLEX was canceled. If FLEX was not used, the radial line is masked by the pointer.

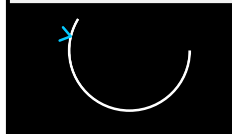
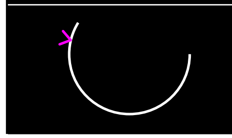
In climb, the white radial line shows the maximum climb thrust reference. If the information is invalid, the line is removed.

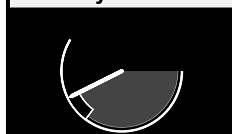
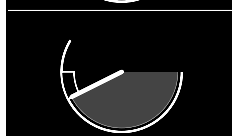


N1 SPEED



COMMANDED INCREASE IN N1

N1 THRUST REFERENCE BUG	
Symbol	**Condition**
	Target N1 value entered manually in N1 data entry field
	Bug position indicates target N1 value as computed automatically by the FMS

N1 THRUST COMMAND CURSOR	
Symbol	**Condition**
	Commanded decrease in N1
	Commanded increase in N1
APR	APR Active

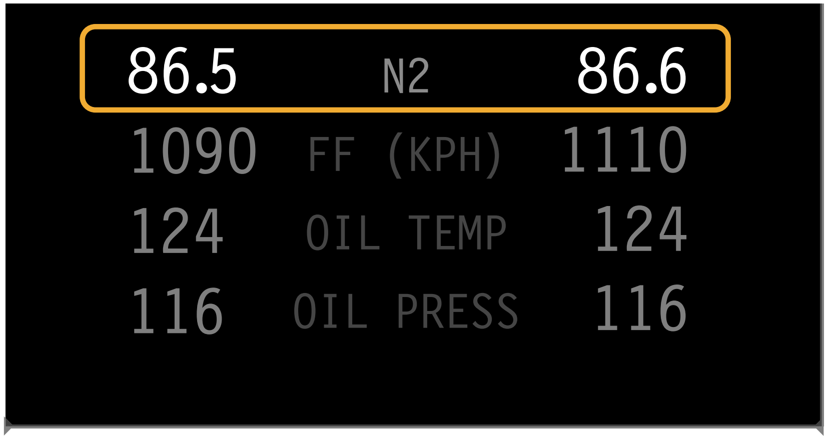
CS1_CS3_7711_006

Figure 6: N1 Thrust Reference Bug, N1 Thrust Reference, N1 Thrust Command Cursor, and Takeoff Reference

N2 INDICATION

The N2 indication is a three-digit value located in the EICAS secondary engine indications area. The three-digit value represents a percentage of the N2 speed.

In normal operation the digits are displayed in white. When N2 speed exceeds 100% the color changes to red.



EICAS

N2	
Symbol	**Description**
86.5	Engine operation in normal range
103.5	N2 above N2 red line threshold
---	Invalid

CS1_CS3_7711_007

Figure 7: N2 Indication

DETAILED DESCRIPTION

Each channel of the electronic engine control (EEC) receives the following inputs:

- Nf speed
- N1 speed
- N2 speed

As long as the EEC is receiving one of the N1 speed signals, the EEC uses N1 as the primary thrust reference. In the event that one EEC channel fails, the other channel maintains engine control.

In the event of a loss of the both N1 speed signals, the EEC calculates the N1 speed using the Nf signal.

In the case of failure of one of the dual N1 speed signals, a L (R) ENGINE FAULT advisory message indicates a fault affecting the dispatch of the aircraft.

The EEC uses the N2 speed signal for control of the fuel and ignition during starting and overspeed monitoring. In the event that one EEC channel fails, the other channel provides N2 speed.

In the event of a loss of one of the N2 signals, a L(R) ENG OPER DEGRADED caution message is displayed on the engine indication and crew alerting system (EICAS). A L(R) ENG FAULT - FADEC FAULT 1 crew alerting system (CAS) advisory message is also displayed and rapid thrust movement is to be avoided.

If both N2 signals are lost, the EEC commands an engine shutdown and the L(R) ENG FAIL caution CAS message is displayed.

The EEC sends the N1, N2, and Nf information to the prognostics and health management unit (PHMU) via CAN DATA BUS. The PHMU uses the N1 and N2 speeds for engine vibration monitoring. The Nf is used by the PHMU for engine fan trim balancing.

The EEC sends the calculated N1 and N2 information directly to the EICAS via an ARINC 429 BUS.

Data concentrator unit module cabinet 1 (DMC 1) and DMC 2 also receive calculated N1 and N2 information via an ARINC 429 BUS for distribution to the aircraft systems.

A L (R) ENG EXCEEDANCE warning message indicates overspeed of the N1 or N2 speeds. If the EEC detects a N1 or N2 speed in excess of 105%, the EEC initiates an automatic shutdown of the engine. In the event the N2 value drops below 5% of the engine governed speed, an L(R) ENG FAIL caution message is displayed.

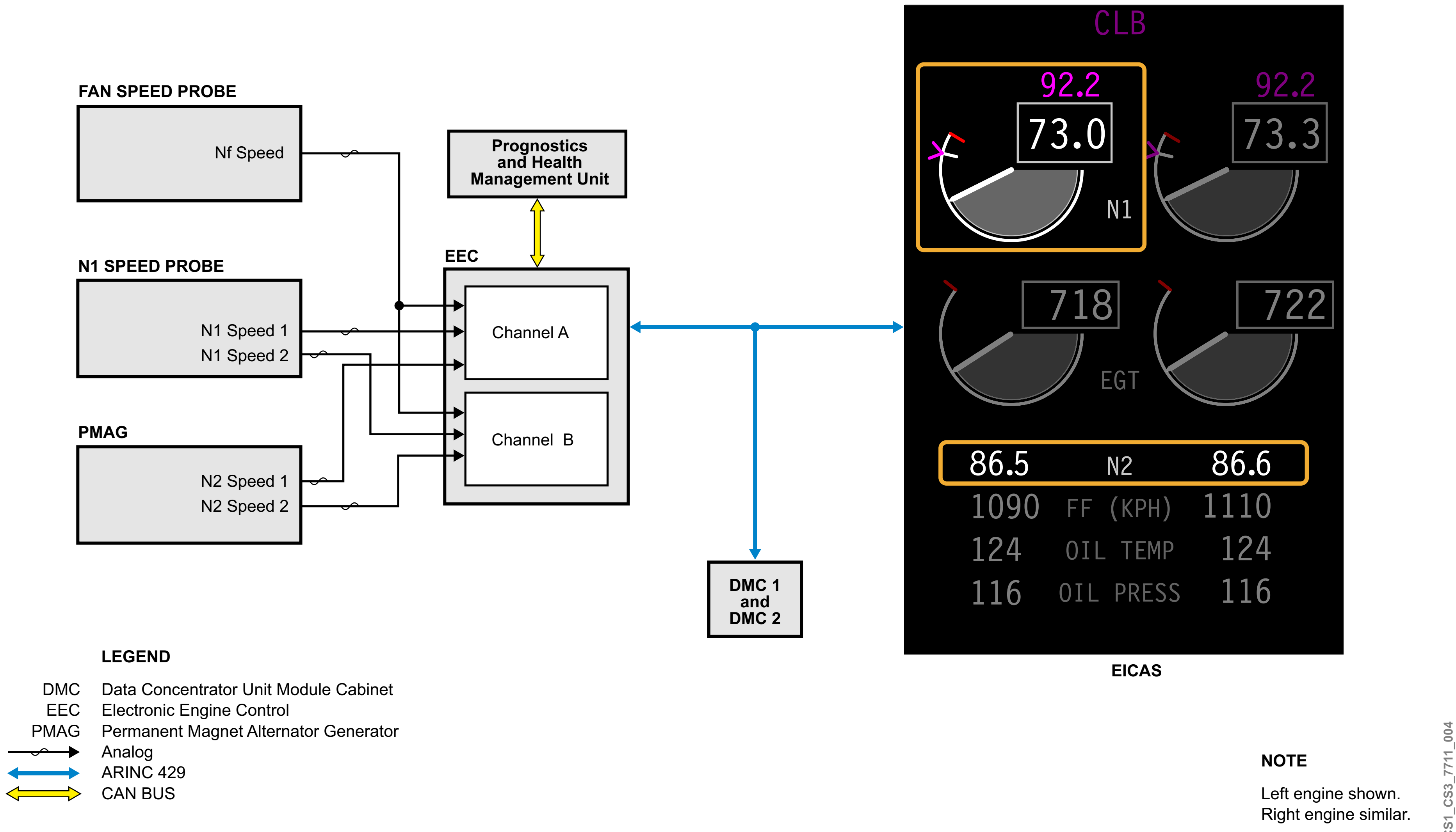


Figure 8: Engine Power Indicating System Detailed Description

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and fault messages for the engine power indicating system.

CAS MESSAGES

Table 1: WARNING Messages

MESSAGE	LOGIC
DUAL ENG FAIL	L ENG and R ENG switches are ON and dual engine flameout.

Table 2: CAUTION Messages

MESSAGE	LOGIC
L ENG EXCEEDANCE	L N1 or L N2 or L ITT or L oil temperature above threshold.
R ENG EXCEEDANCE	R N1 or R N2 or R ITT or R oil temperature above threshold.
L ENG FAIL	L ENG switch in ON with the engine in a sub-idle condition and a relight or restart was unsuccessful.
R ENG FAIL	R ENG switch ON with the engine in a sub-idle condition and a relight or restart was unsuccessful.
L ENG OPER DEGRADED	Left EEC has detected a loss of thrust or degraded thrust control due to: • Fuel flow or actuator not tracking to commanded position • Input parameter in failsafe • Engine flameout
R ENG OPER DEGRADED	Right EEC has detected a loss of thrust or degraded thrust control due to: • Fuel flow or actuator not tracking to commanded position • Input parameter in failsafe • Engine flameout

Table 3: FAULT Messages

MESSAGE	LOGIC
L(R) ENG FAULT-FADEC FAULT 1	INFO message will be set when any non-dispatchable fault has been detected by the left (right) engine FADEC system or the DMC receives no valid data from left (right) engine for 10 seconds.

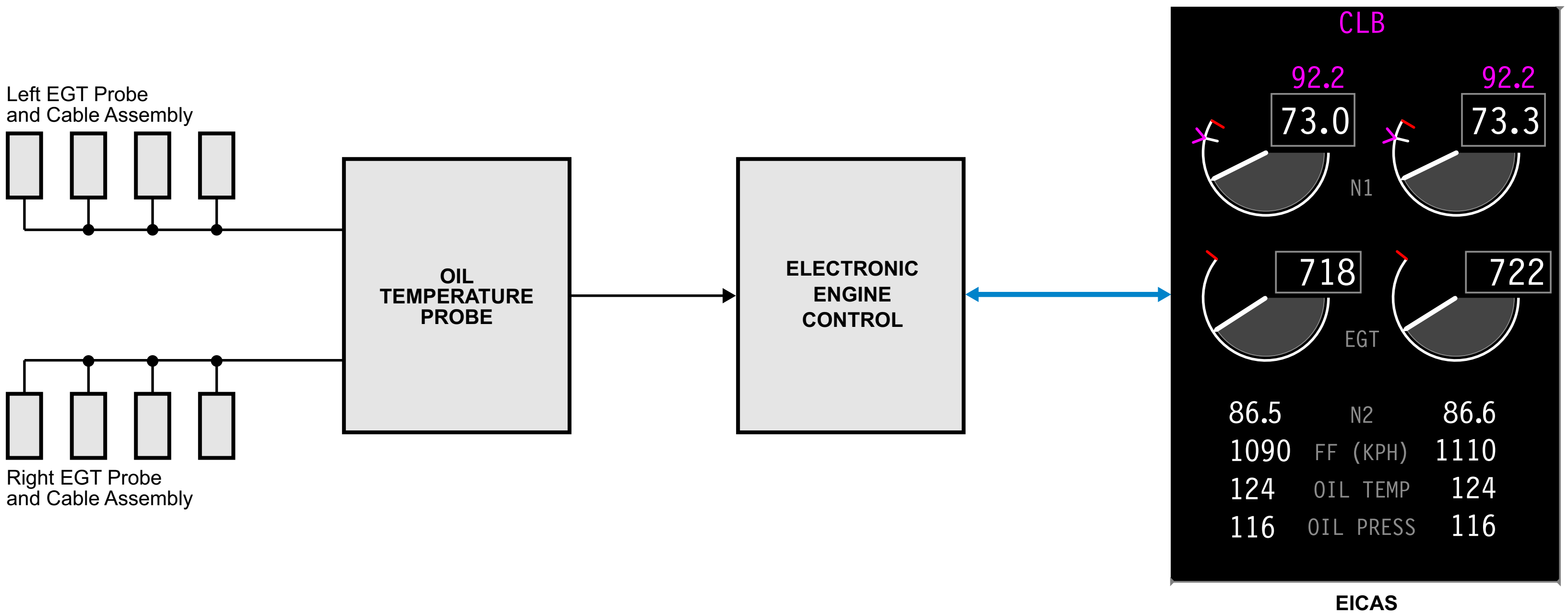
77-21 TEMPERATURE INDICATING SYSTEM

GENERAL DESCRIPTION

The temperature indicating system measures the exhaust gas temperature (EGT) for display on the engine indication and crew alerting system (EICAS).

EGT is monitored at station 5 of the engine. Two EGT probe and cable assemblies are used to transmit the EGT signals to the main oil temperature (MOT) probe and then to the electronic engine control (EEC). The oil temperature probe is used as the cold junction for the EGT signal. The EEC then calculates the EGT value and sends the information to the EICAS display.

Each EGT probe and cable assembly consists of four EGT probes. The assemblies are semi-rigid and can only be replaced as an assembly. Individual EGT probes are not replaceable.



LEGEND

EGT Exhaust Gas Temperature

ARINC 429

Figure 9: Exhaust Gas Temperature Indicating System

COMPONENT LOCATION

The temperature indicating system consists of the following components:

- Exhaust gas temperature (EGT) probe and cable assembly
- Oil temperature probe
- Probe junction

EXHAUST GAS TEMPERATURE PROBE AND CABLE ASSEMBLY

The EGT probe and cable assemblies are mounted around the circumference of the turbine exhaust case (TEC). There is one assembly on the left side of the engine, and one on the right side.

OIL TEMPERATURE PROBE

The oil temperature probe is installed on the oil control module (OCM) located on the right side of the engine.

PROBE JUNCTION

The probe junction is located on each side of the TEC at the 3 o’clock and 9 o’clock position.

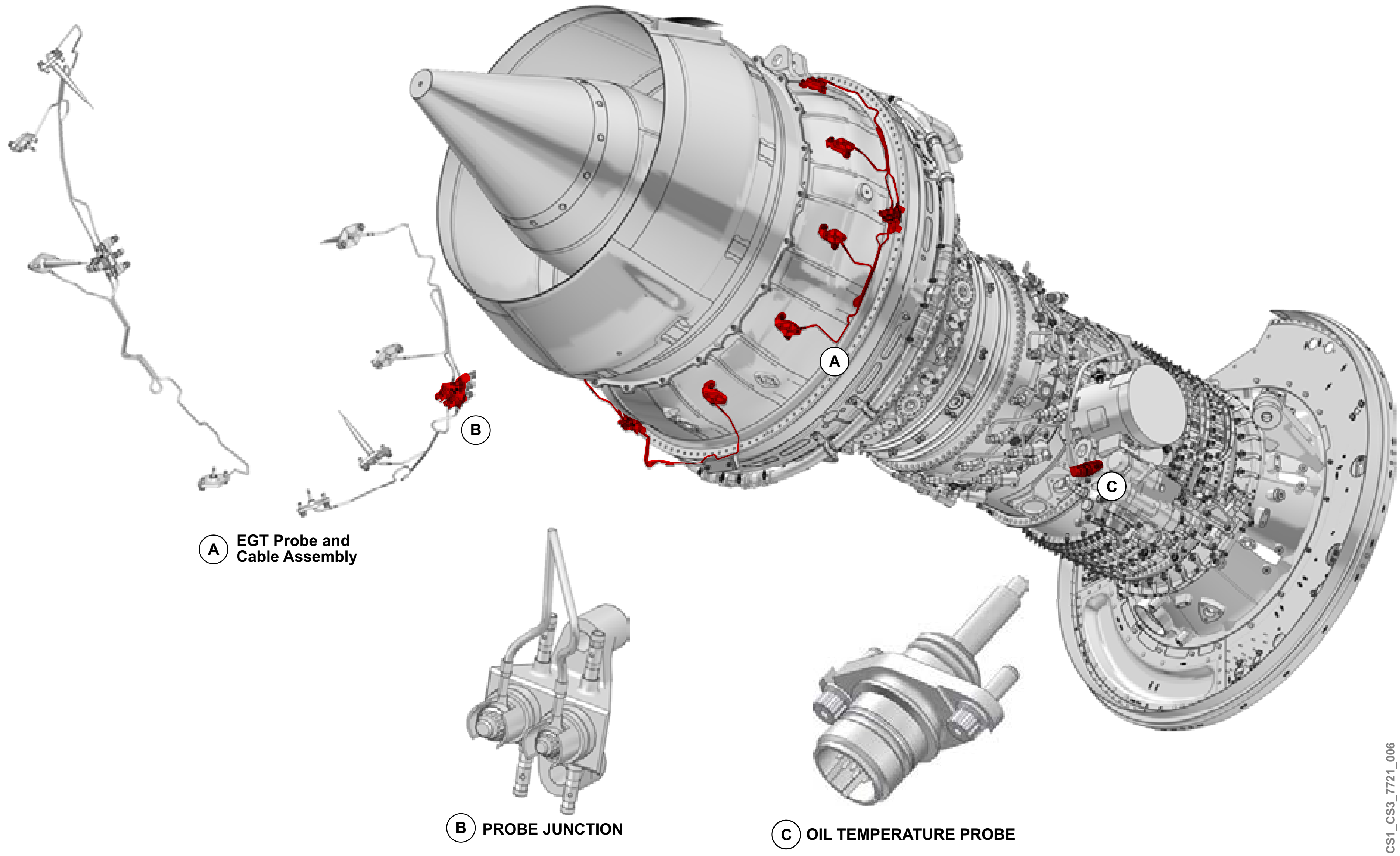


Figure 10: Exhaust Gas Temperature Indicating System Component Location

CONTROLS AND INDICATIONS

INDICATIONS

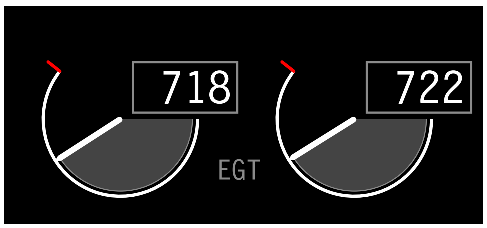
The exhaust gas temperature (EGT) display shows the EGT for each engine on the EICAS display, as calculated by the EEC. The gauges consist of an analogue and a four-digit display. The display is shown in degrees Celsius (°C).

The analogue gauge consists of a pointer in a round arc representing the operating range of the engine. The area below the pointer is shaded to provide a quick visual reference. At the upper end of the arc is a red line limit, indicating maximum operating temperature. There is also a yellow line to indicate approaching maximum temperature limit. The electronic engine control (EEC) calculates an engine operating time limit in the yellow range. If the time limit exceeds the EGT needle, the shaded area and digital readout change to red. The yellow line is removed during takeoff, go-around, reverse, or when invalid.

When the EGT pointer goes above the yellow line, the pointer and the shaded area below the pointer change to yellow.

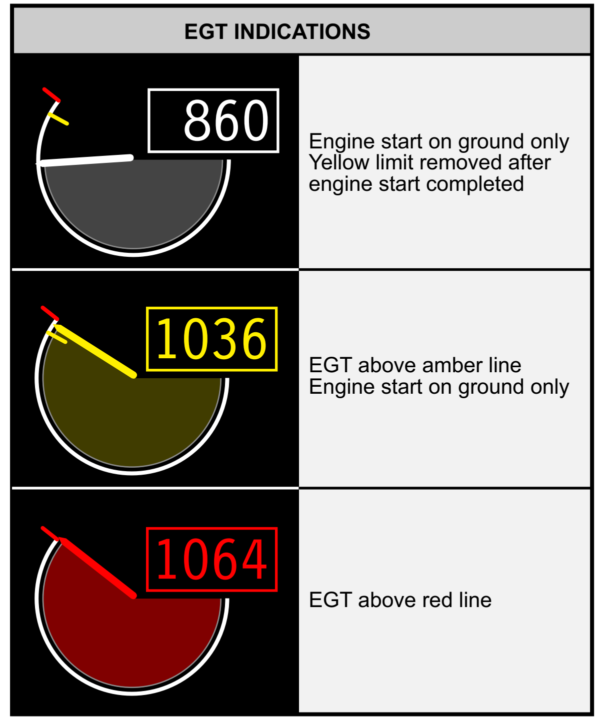
When the EGT pointer goes above the red line limit, the pointer and the shaded area below the pointer change to red. The arc is also extended in red above the red line limit, and stays there even if the overtemperature condition goes away. This indicates that the engine temperature has exceeded its limit.

The digital EGT display changes color to match the pointer and shaded area.



EICAS

Normal Indications - Engine Running on ground after start and in flight



CS1_CS3_7721_005

Figure 11: Exhaust Gas Temperature Indications

DETAILED DESCRIPTION

The four signals from both EGT probe and cable assemblies are electrically averaged into two signals, which are sent to the oil temperature probe for compensation.

The oil temperature probe sends compensated EGT signals to the EEC. The left EGT probe and cable assembly signal is sent to channel A. The signal of the right EGT probe and cable assembly is sent to channel B of the EEC.

The EEC converts the average EGT analog signal to a digital signal. The two signals are averaged into a single output, which is sent directly to the display unit (DU) for display on the EICAS.

A L(R) ENGINE FAULT advisory message is displayed when one of the EGT probe and cable assemblies fails.

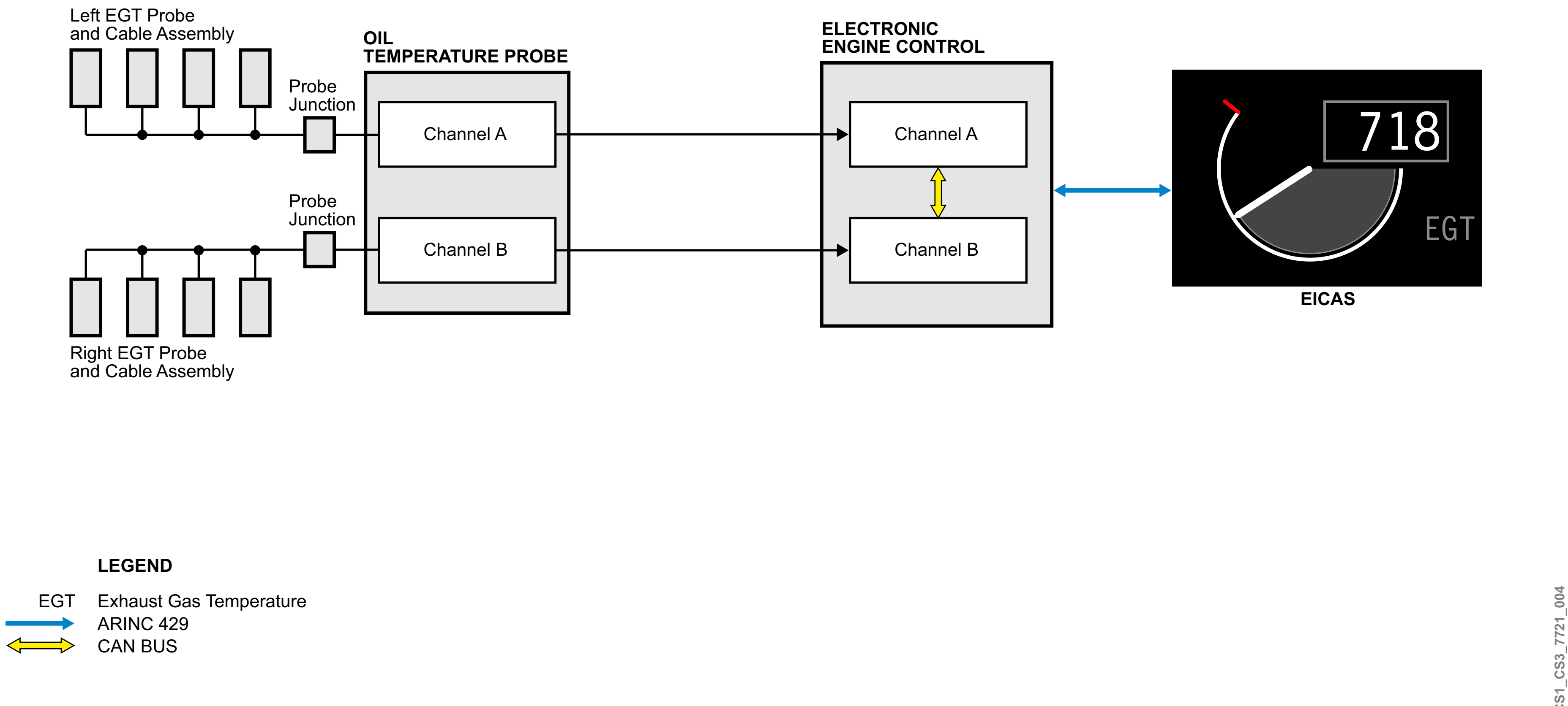


Figure 12: Exhaust Gas Temperature Detailed Description

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) messages for the engine temperature indicating system.

CAS MESSAGES

Table 4: CAUTION Messages

MESSAGE	LOGIC
L ENG EXCEEDANCE	L N1 or L N2 or L ITT or L oil temperature above threshold.
R ENG EXCEEDANCE	R N1 or R N2 or R ITT or R oil temperature above threshold.

77-32 VIBRATION MONITORING

GENERAL DESCRIPTION

The engine vibration indicating system monitors, records, and analyzes engine vibration before displaying this information on the engine indication and crew alerting system (EICAS). Engine speed sensors are used, in conjunction with vibration monitors, to help determine which rotor may be out of balance.

The prognostics and health management unit (PHMU) compares engine speed data from the electronic engine control (EEC) with vibration data sent from the forward and aft rotor vibration sensors. The PHMU receives power from DC BUS 1.

Data stored in the EECs data storage unit (DSU) over a period of flights can be used by the onboard maintenance system (OMS) to conduct engine vibration surveys.

Vibration sensors are used on both N1 and N2 rotors to send signals that are proportional to engine vibration. The PHMU converts the signals from charge to voltage, filters them, and correlates the frequencies with the specific rotor. The N1 and N2 vibration signals are then sent to the EEC for display on EICAS. These signals are also used by the aircraft OMS to calculate solutions for correcting a fan imbalance.

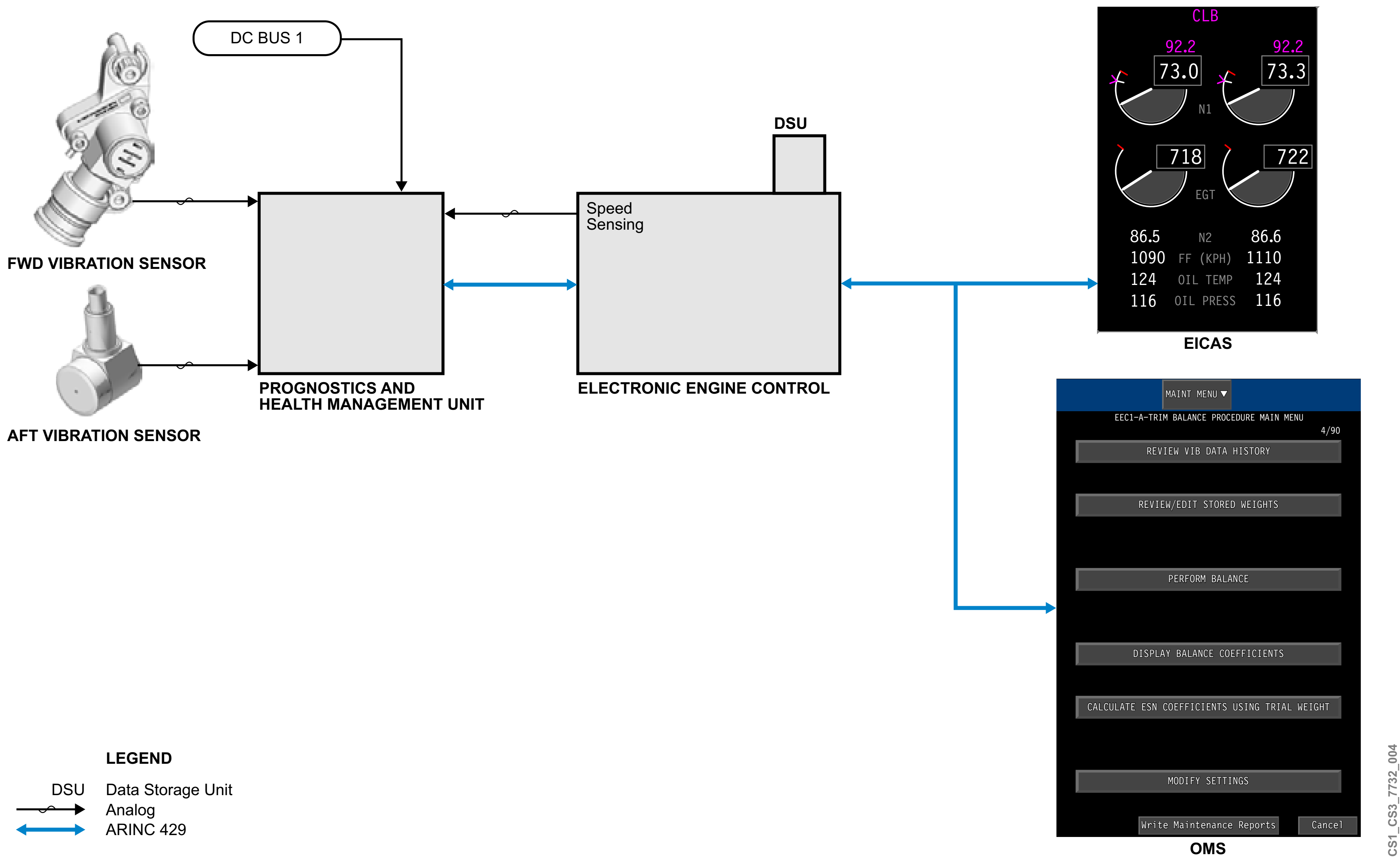


Figure 13: Vibration Monitoring

COMPONENT LOCATION

The vibration monitoring system consists of the following components:

- Forward vibration sensor
- Aft vibration sensor
- Prognostics health and management unit (PHMU)
- Data storage unit (DSU)

FORWARD VIBRATION SENSOR

The forward vibration sensor is mounted on the compressor intermediate case at the 9 o'clock position.

AFT VIBRATION SENSOR

The aft vibration sensor is mounted on the low-pressure turbine (LPT) housing at the 3 o'clock position.

PROGNOSTICS HEALTH AND MANAGEMENT UNIT

The prognostics health and management unit (PHMU) is mounted to the fan case adjacent to the EEC.

DATA STORAGE UNIT

The data storage unit (DSU) is located on the EEC.

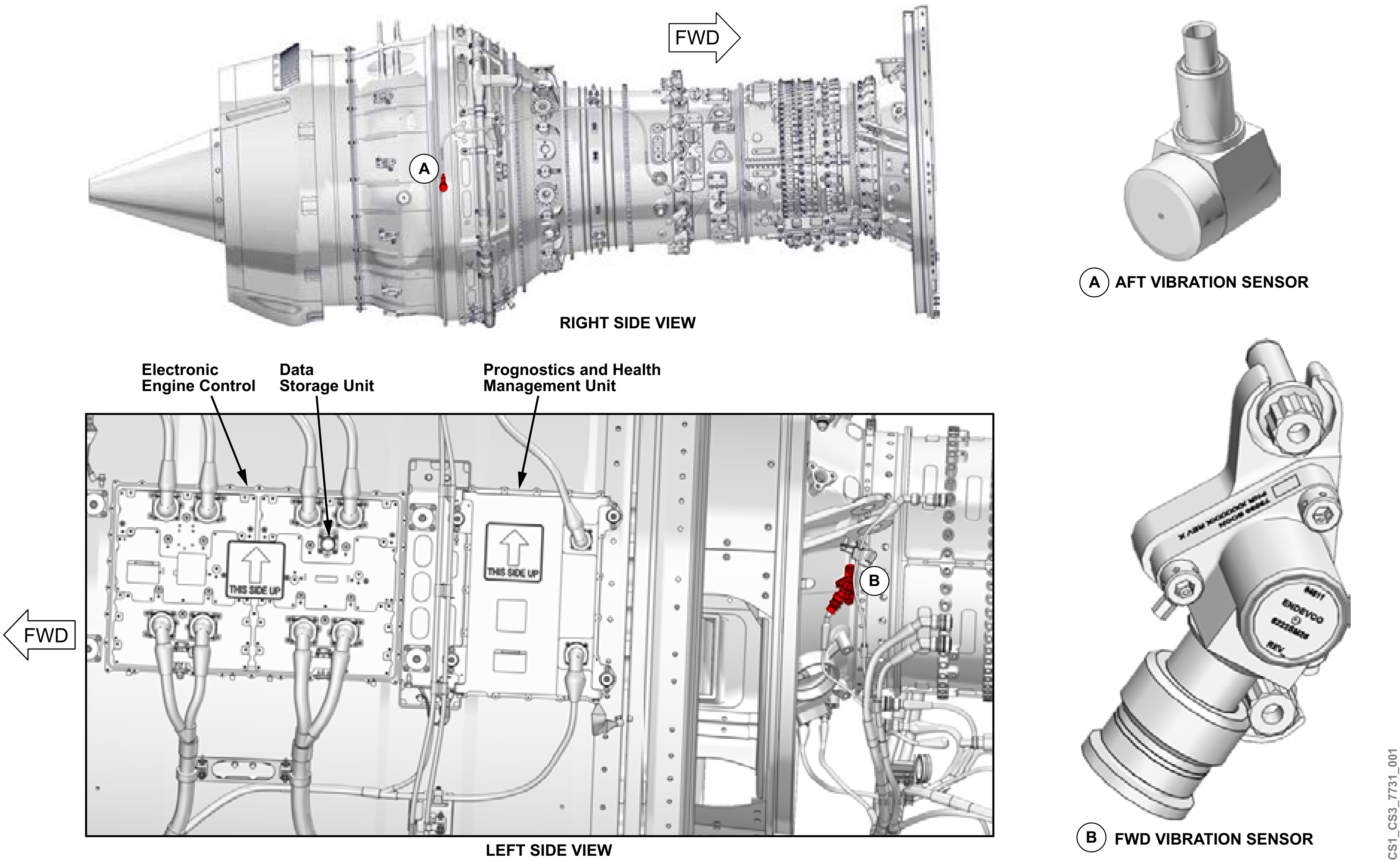


Figure 14: Vibration Monitoring Component Location

CONTROLS AND INDICATIONS

INDICATIONS

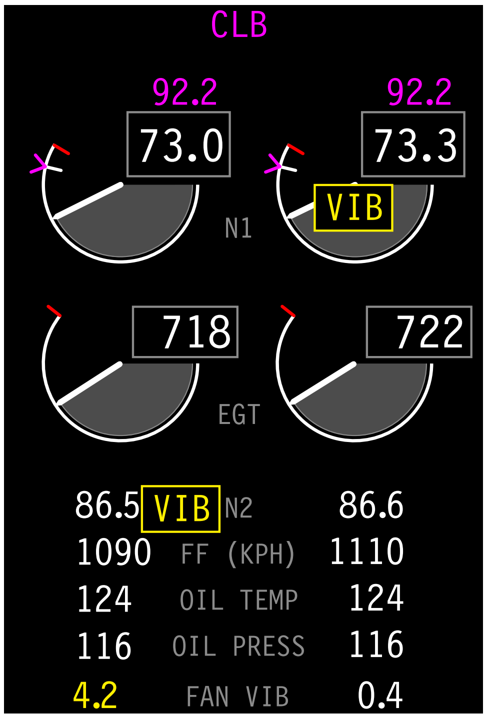
The following vibration indications are displayed in the EICAS:

- N1 vibration is displayed as an AMBER VIB flag in the associated N1 gauge
- N2 vibration is displayed as an AMBER VIB flag adjacent to the digital readout
- FAN VIB is displayed in units

The N1 VIB is displayed as a flag in the N1 gauge when the parameter exceeds the normal vibration threshold of 0.52 inches per second (IPS). The REV icon replaces the N1 VIB display when the thrust reversers deploy.

The N2 VIB is displayed as a flag adjacent to the N2 speed when the parameter exceeds the normal vibration threshold of 1.2 IPS.

The FAN VIB level is posted on EICAS in units from 0 to 8. The units represent a percentage of the vibration threshold where 4.0 is 100%. The normal vibration threshold for the engine fan is 1.2 IPS. FAN VIB is only displayed when an engine fan vibration exceeds 4.0 units, or a fault has occurred in the vibration monitoring system. When this occurs, both left and right engine FAN VIB is displayed. The fan vibration exceedance is displayed in amber, while normal indications are displayed in white.



EICAS

FAN VIBRATION	
Symbol	**Description**
0.6	Vibration at or below high vibration threshold
4.2	Vibration above high vibration threshold
---	If both vibration monitors fail on one side.

N1 and N2 VIBRATION	
Symbol	**Description**
VIB	Vibration above high vibration threshold N1 VIB is not displayed when the thrust reverser is operating.

CS1_CS3_7732_001

Figure 15: Vibration Monitoring Indications

DETAILED DESCRIPTION

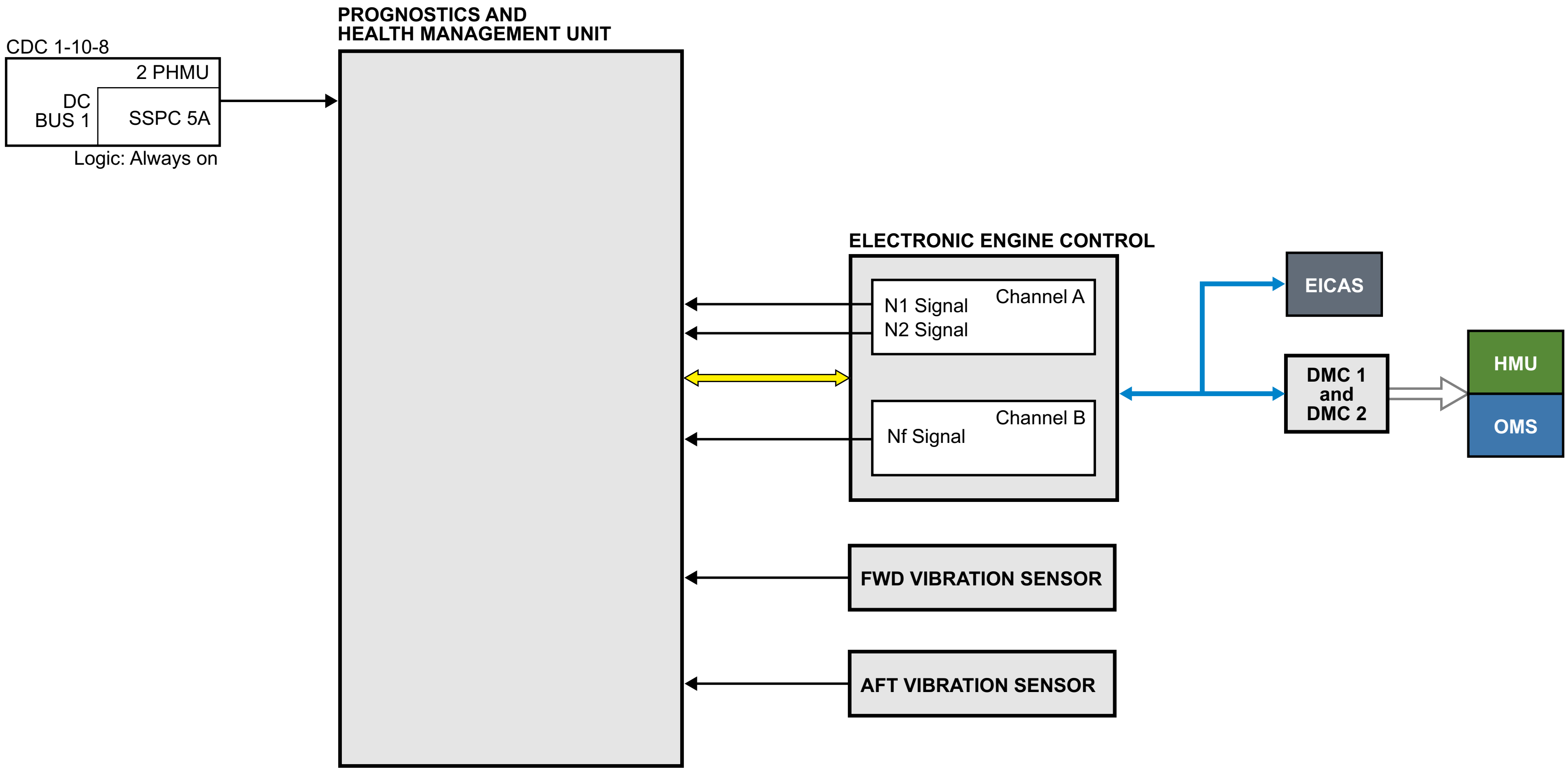
Vibration monitoring is performed through data input from the two engine vibration sensors, as well as speed inputs from the EEC for N1, N2, and the fan speed (Nf).

The PHMU receives engine speed signals from the EEC and vibration signals from the vibration sensors. The Nf speed signal is used in conjunction with the forward vibration sensor to measure fan vibration.

The PHMU compares the engine speeds and vibration signals to determine the vibration level for the fan, N1, and N2 shafts. The PHMU sends the vibration signals to the EEC via a CAN BUS. The EEC sends vibration information for display on EICAS.

When engine vibration exceedances are detected, an ENG VIBRATION caution message is displayed, indicating excessive vibration levels.

Engine vibration information is also sent to the DMCs for the OMS and health management unit (HMU).



LEGEND

DMC Data Concentrator Unit Module Cabinet
ARINC 429
CAN BUS

NOTE

Left engine shown.

CS1_CS3_7731_002

Figure 16: Vibration Monitoring Detailed Description

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the engine vibration monitoring system.

CAS MESSAGES

Table 5: CAUTION Messages

MESSAGE	LOGIC
ENGINE VIBRATION	Left or right engine vibration (either fan vibration, N1 or N2).

Table 6: ADVISORY Message

MESSAGE	LOGIC
L (R) ENG FAULT	Loss of redundant or non-critical function for the left (right) engine. Refer to INFO messages.

Table 7: INFO Messages

MESSAGE	LOGIC
77 L (R) ENGINE FAULT - PHMU INOP	FADEC logic has detected that the PHMU is failed.
77 L (R) ENGINE FAULT - N1/FAN VIBRATION NON DEGRADED	INFO message will be set when a forward vibration sensor fault has been detected or N1 or NF speed input for vibration processing has failed.
77 L (R) ENGINE FAULT - N2 VIBRATION NON DEGRADED	INFO message will be set when a rear vibration sensor fault has been detected or N2 speed input for vibration processing has failed.

PRACTICAL ASPECTS

ONBOARD MAINTENANCE SYSTEM

Trim Balance

The purpose of the trim balance procedure is to provide the information needed to balance the fan and/or low spool of the engine. The fan is a large rotating mass, even minor blade damage can cause an imbalance. A trim balance procedure is used to reduce fan-related vibration onwing.

While the engine is running, the prognostics and health management unit (PHMU) continuously collects data for trim balancing using values from the vibration sensors and the predetermined Nf, N1, and N2 engine speeds.

Historical vibration data stored in the EECs data storage unit (DSU) is used as reference in calculating the trim balance solutions.

The system corrects small amounts of residual imbalance, which could be related to minor blade damage or fan blade replacement, even when matched sets are replaced.

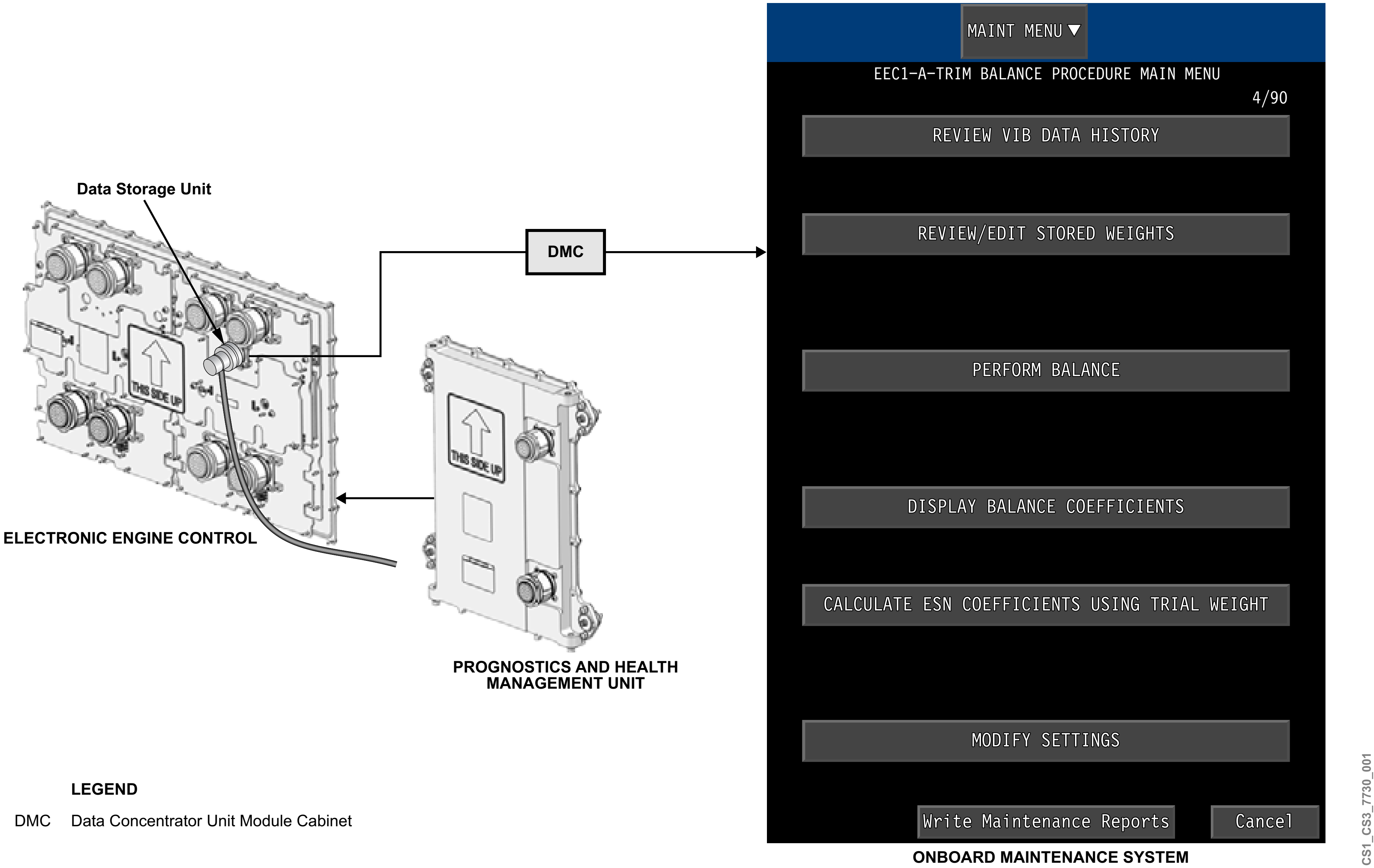


Figure 17: Trim Balance

Fan Hub Balance Weights

N1 (fan rotor) vibration signals are sent to the PHMU and the EEC for interpretation and calculation of optimal trim balance solutions. The onboard maintenance system (OMS) uses these solutions to indicate where to install balance weights on the fan rotor disc. Each weight has a different part number, based on its specific weight.

The fan speed sensor is used to determine the fan index position sensing. It measures the fan index rotational position, once-per-revolution, to allow the location of vibration imbalances to support the fan trim balance.

The trim balance procedure involves running a baseline (as-is) vibration survey. The final correction weight is then installed on a balance weight flange, located on the front of the fan rotor hub.

The software-stored weight configuration should exactly match the actual balance weight configuration on the engine in order for the OMS system-provided balance solution to be correct.

Reviewing and editing the stored balance weight configuration is a precondition for performing a trim balance.

A 0° reference point is aligned with a dimple on the fan hub shaft. Trim balance weights are installed on either side of the module balance weights. Module balance weights are riveted in place during production or engine rebuild, and should not be disturbed during trim balance procedures.

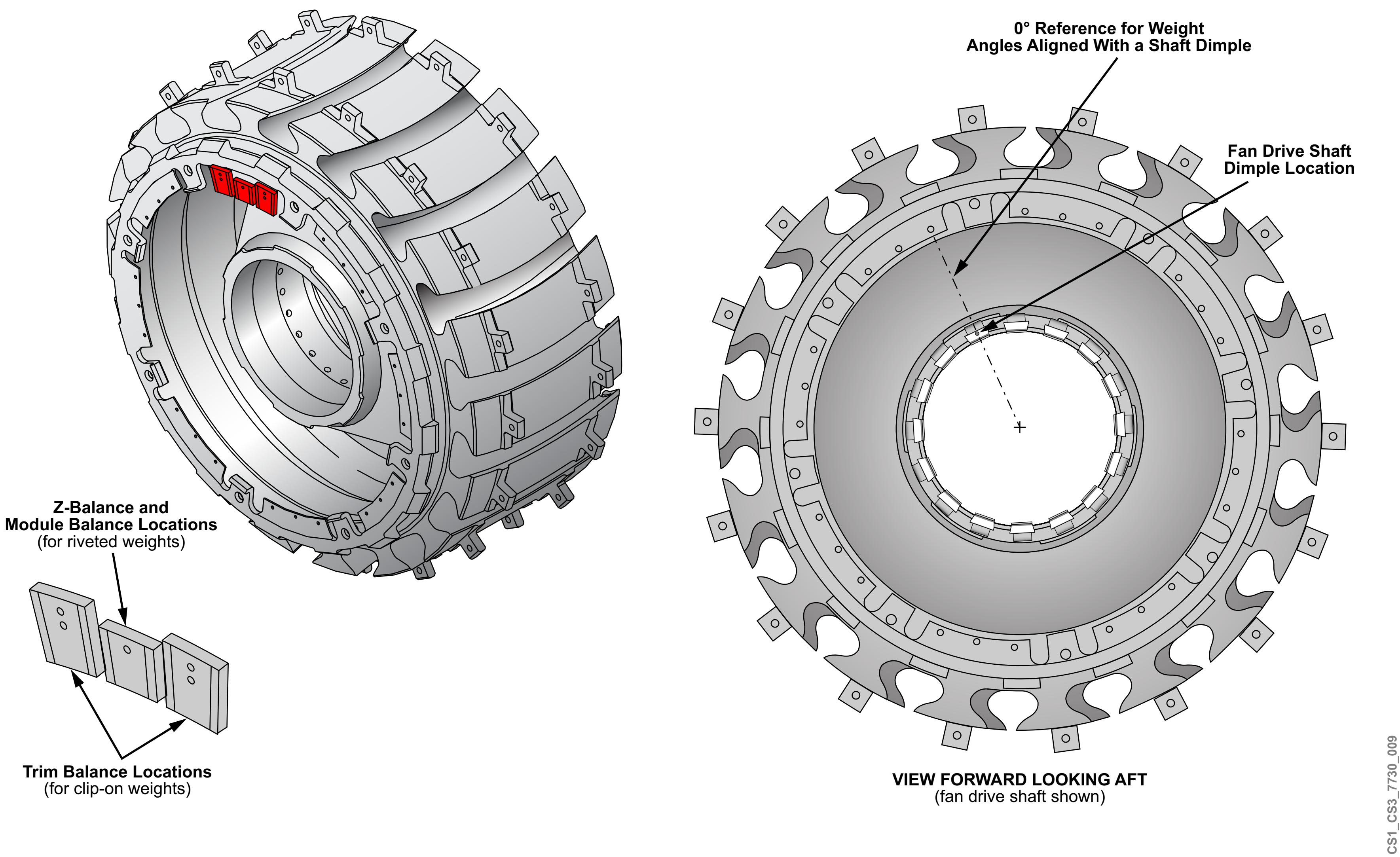
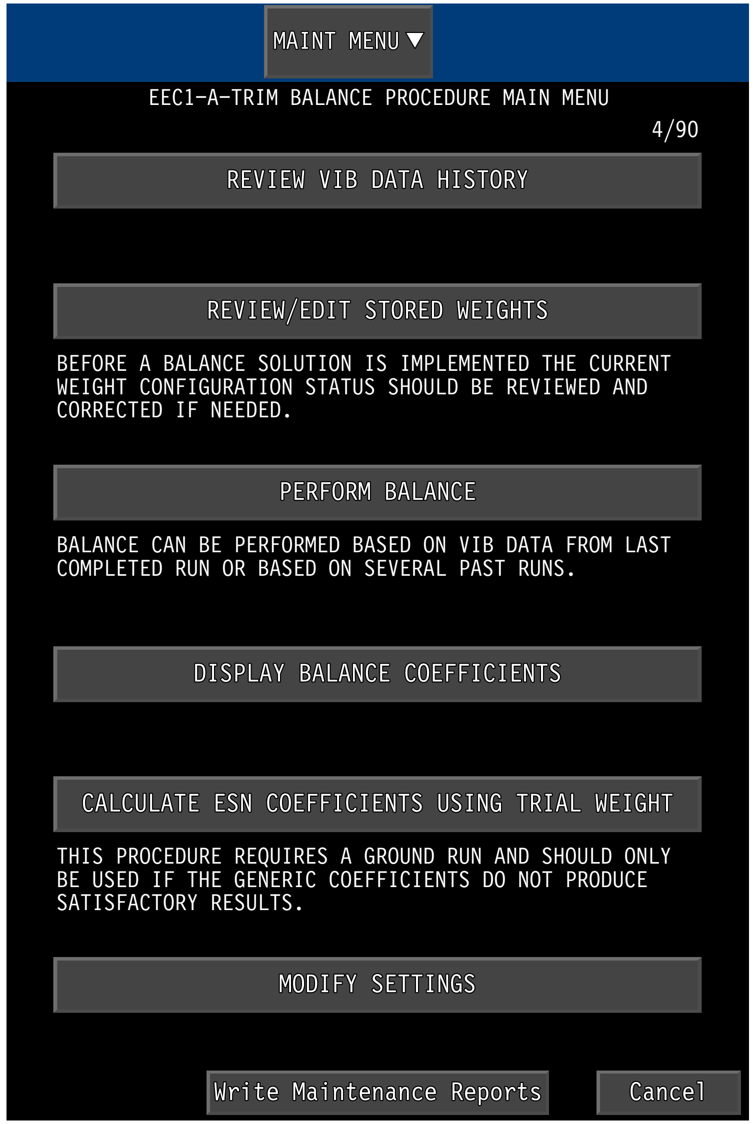


Figure 18: Fan Hub Balance Weights

Trim Balance Procedure Main Menu

On the OMS maintenance page, the trim balance procedure main menu screen provides an entry point into the trim balance subprocedures, including:

- REVIEW VIB DATA HISTORY
- REVIEW/EDIT STORED WEIGHTS
- PERFORM BALANCE
- DISPLAY BALANCE COEFFICIENTS
- CALCULATE ESN COEFFICIENTS USING TRIAL WEIGHT
- MODIFY SETTINGS



ONBOARD MAINTENANCE SYSTEM

CS1_CS3_7730_002

Figure 19: Trim Balance Procedure Main Menu

Review VIB Data History

REVIEW VIB DATA HISTORY determines changes to the overall vibration levels in the recent flight history.

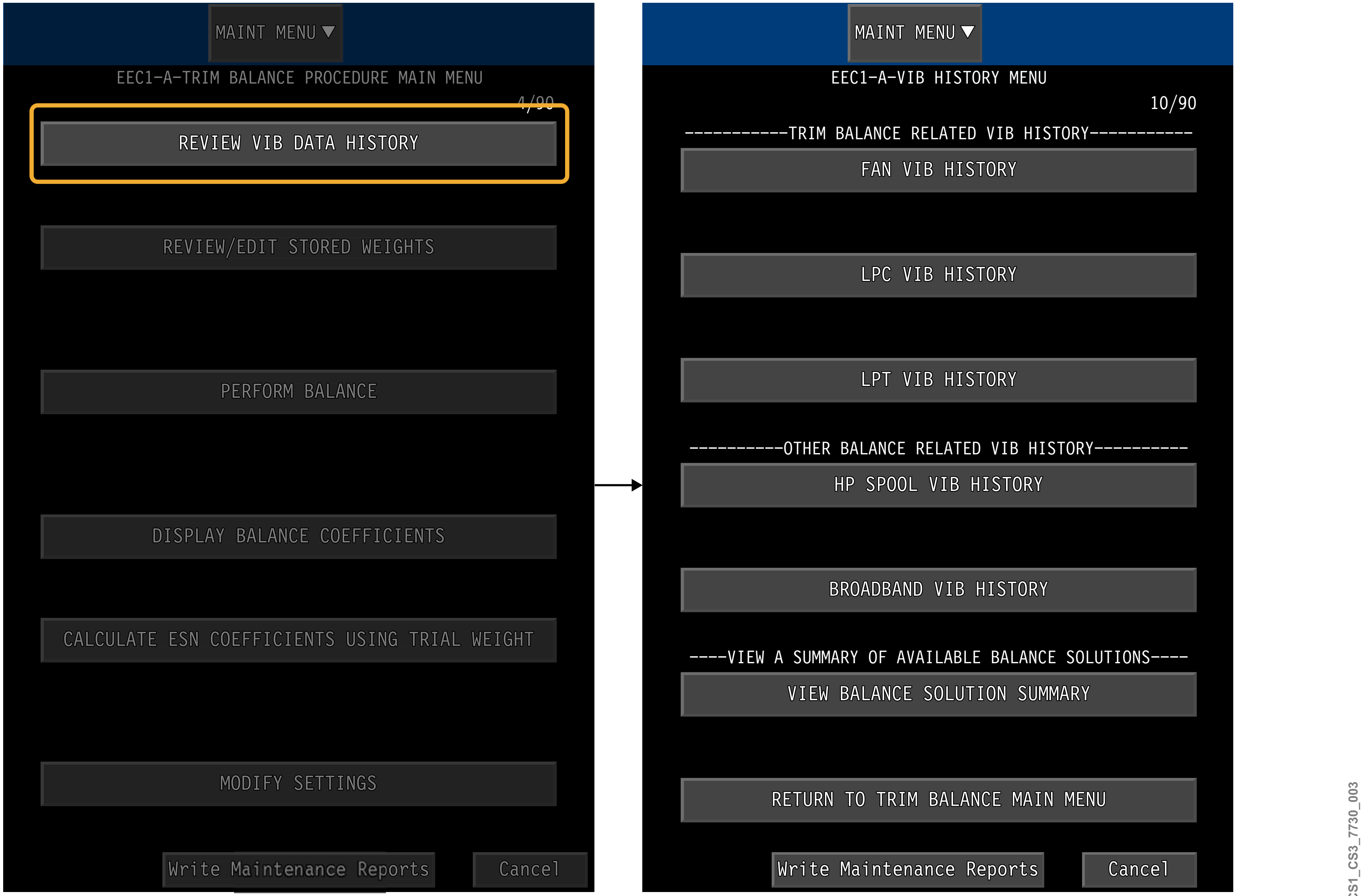


Figure 20: Review VIB Data History

Review/Edit Stored Weights

REVIEW/EDIT STORED WEIGHTS provides a starting point for examining and changing software stored balance weight configurations for each of the correction planes (fan, LPC, and LPT).

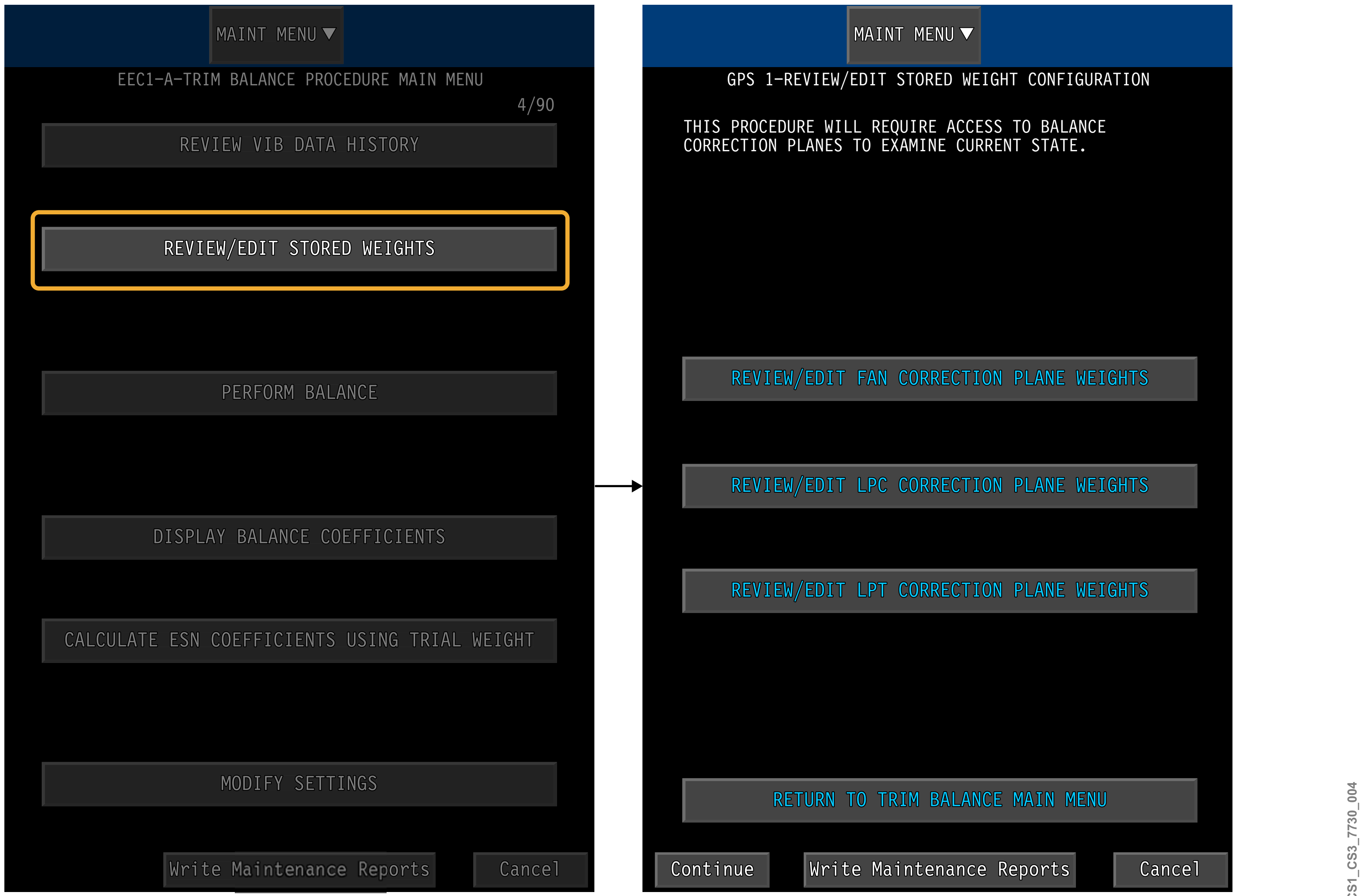


Figure 21: Review/Edit Stored Weights

Perform Balance

PERFORM BALANCE provides options for selecting which type of balance procedure to perform. Each of the provided buttons lead to separate balance solutions, based on the selected balance plane (fan, LPC or LPT).

All balance solutions are precalculated and stored during previous engine runs.

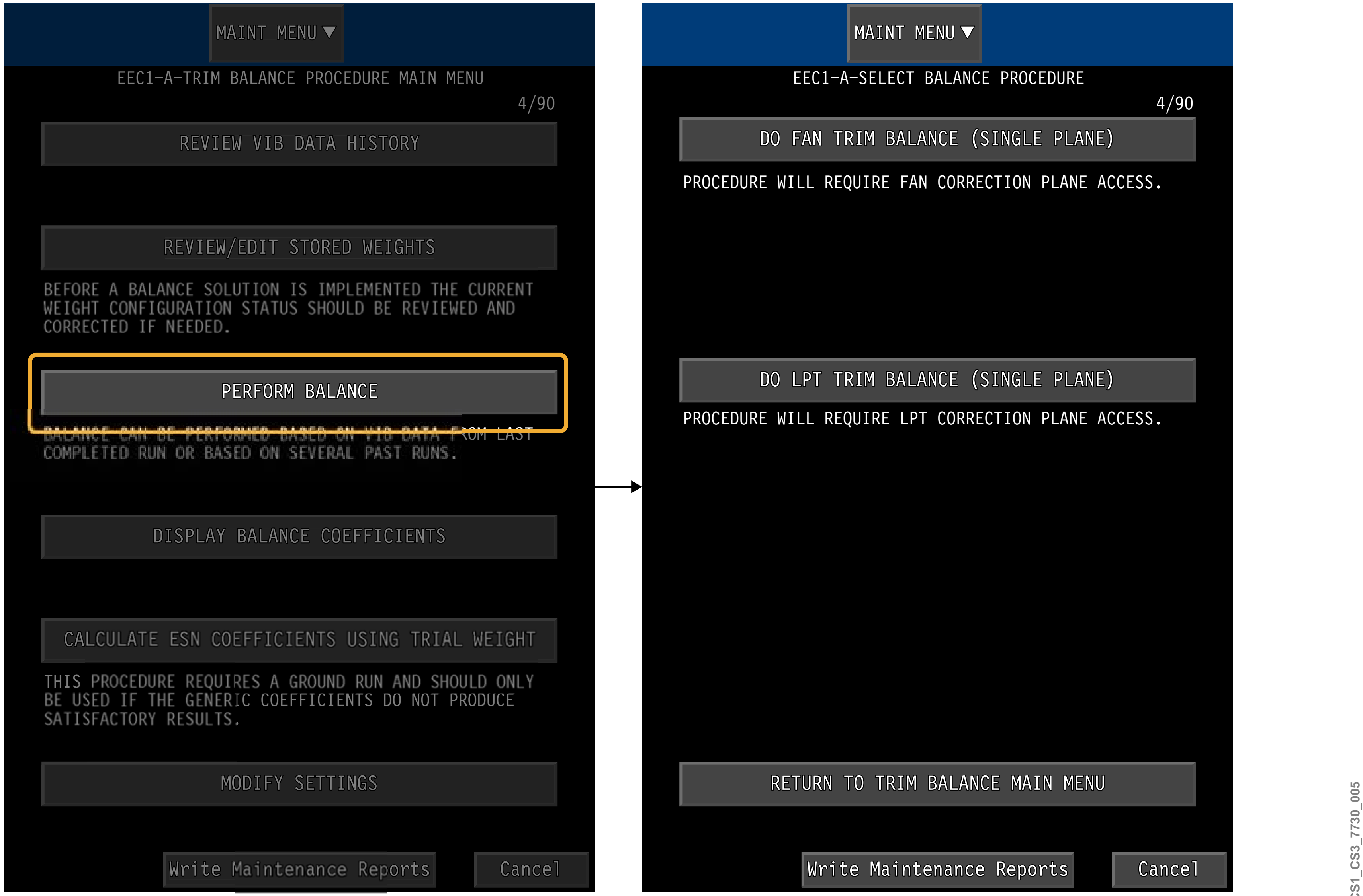


Figure 22: Perform Balance

Display Balance Coefficients

The DISPLAY BALANCE COEFFICIENTS page is an advanced screen used to modify OMS trim balance settings.

Refer to the Aircraft Maintenance Publication (AMP) for guidance as to when this advanced subprocedure should be used.

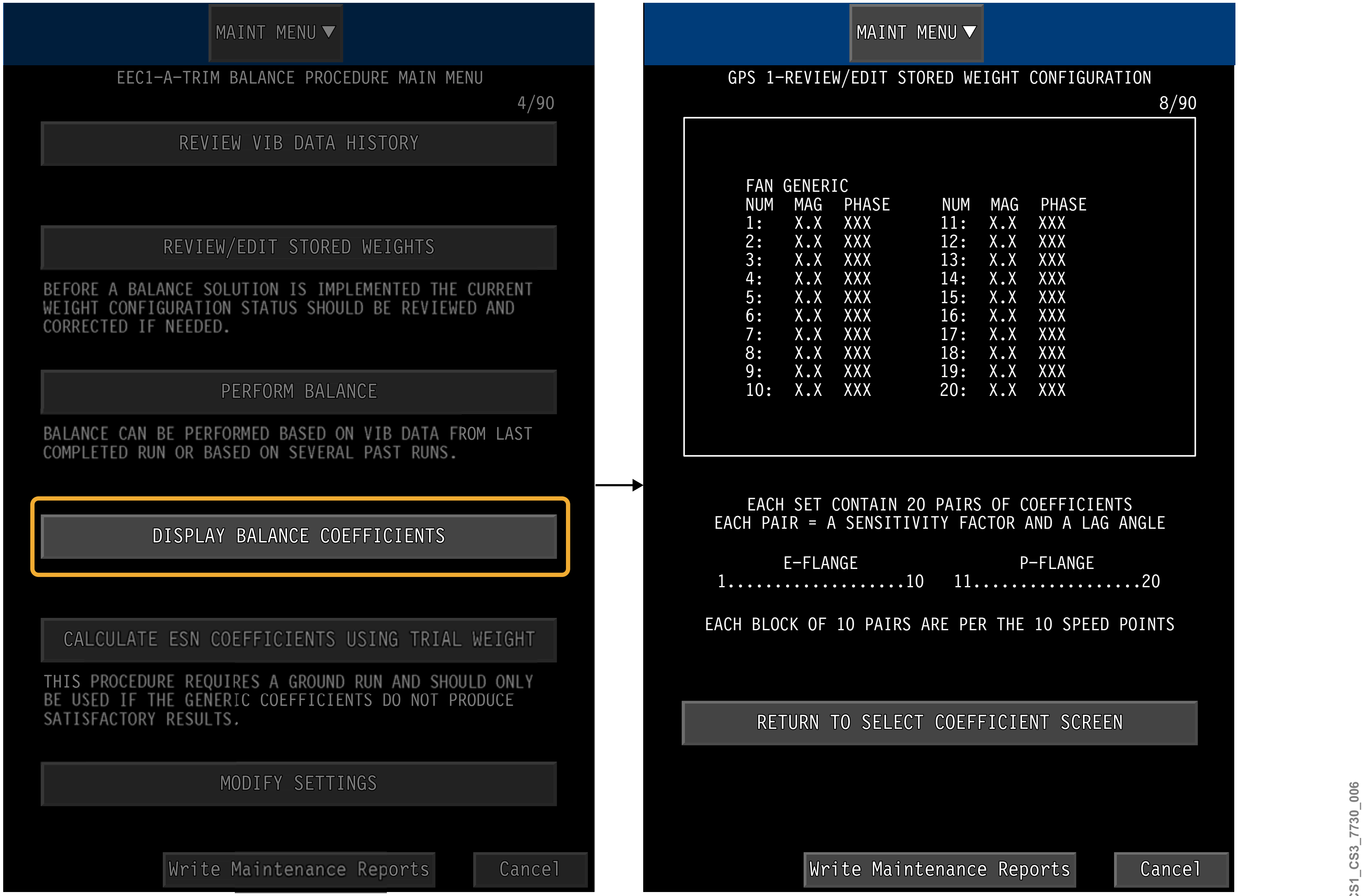


Figure 23: Display Balance Coefficients

Engine Serial Number Coefficients

The calculated ESN coefficients, using the trial weight screen, should only be used if the normal procedure does not produce the desired reduction in vibration.

The screen enables the installation of trial weights to help determine a solution.

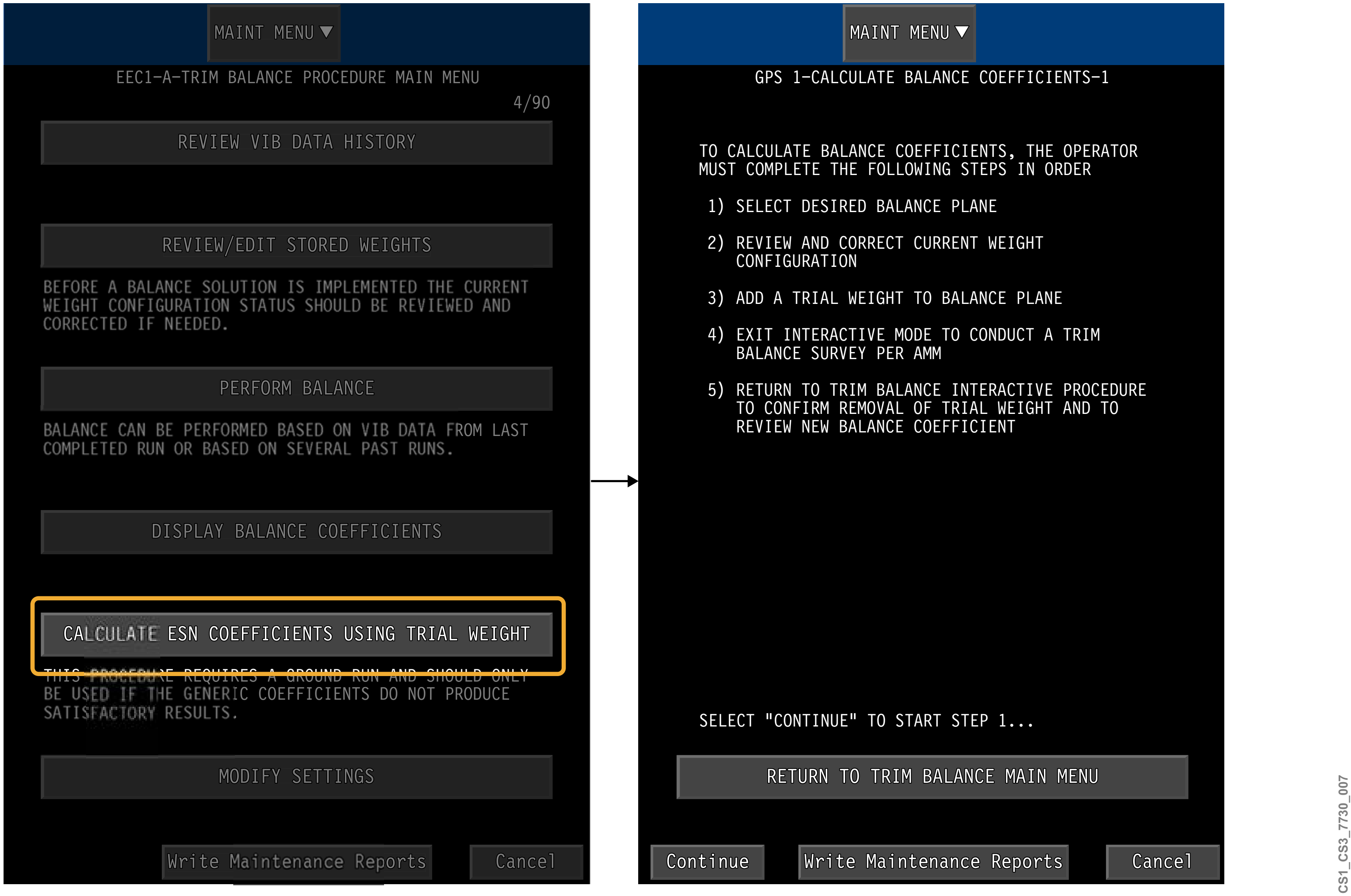


Figure 24: Engine Serial Number Coefficients

Modify Settings Screen

The MODIFY SETTINGS screen is an advanced screen that allows setting changes for trim balance processing, clearing configuration, and balance histories.

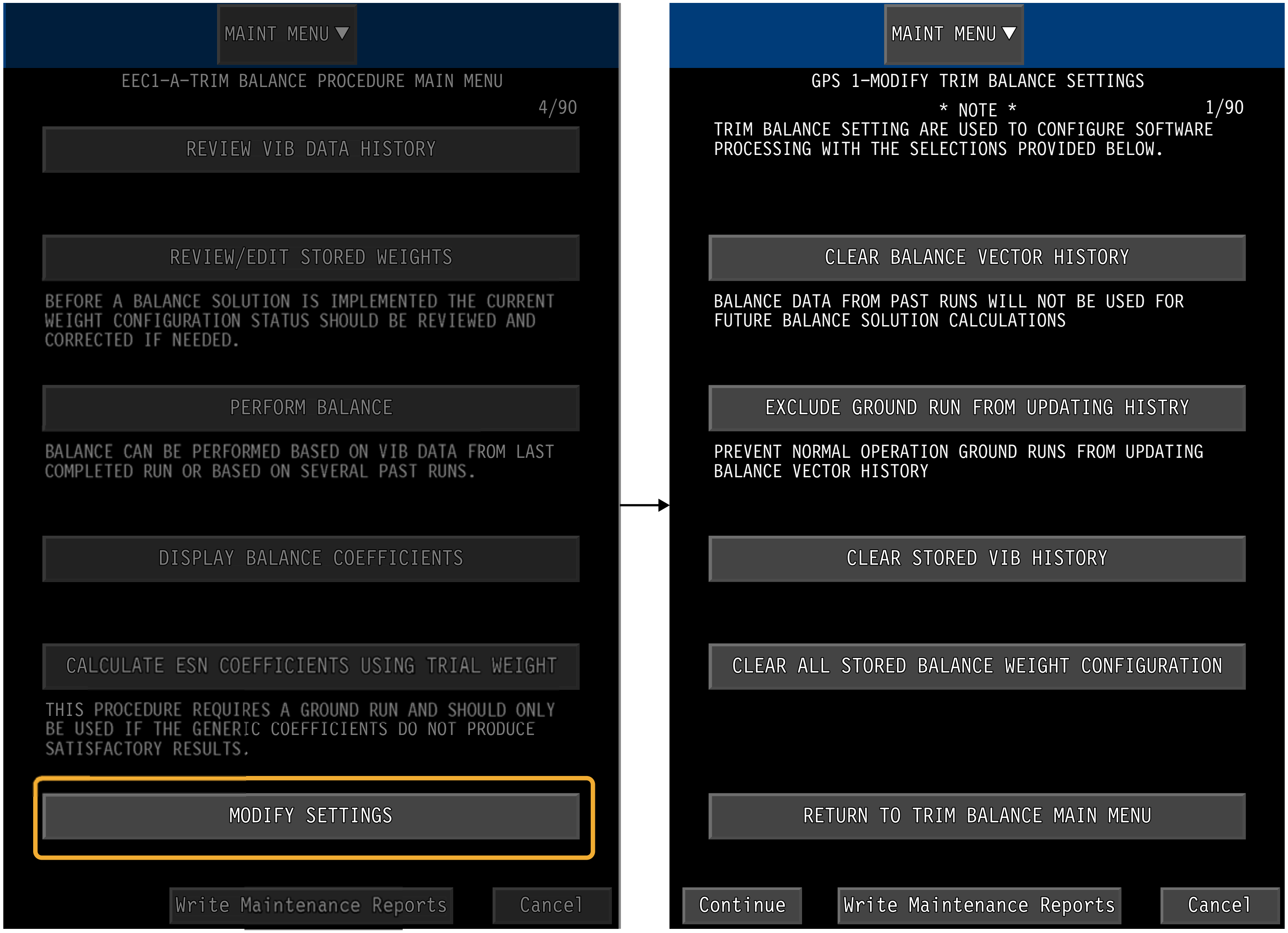


Figure 25: Modify Settings Screen

FAN TRIM BALANCE USING THE PW 1500G TRIM BALANCE HELPER

Test Set Up

The Pratt & Whitney Special Instruction no. 44F - 16 Rev. A provides instructions for performing an on-wing fan trim balance using the onboard maintenance system (OMS) and the PW1500G Fan Trim Balance Helper Tool.

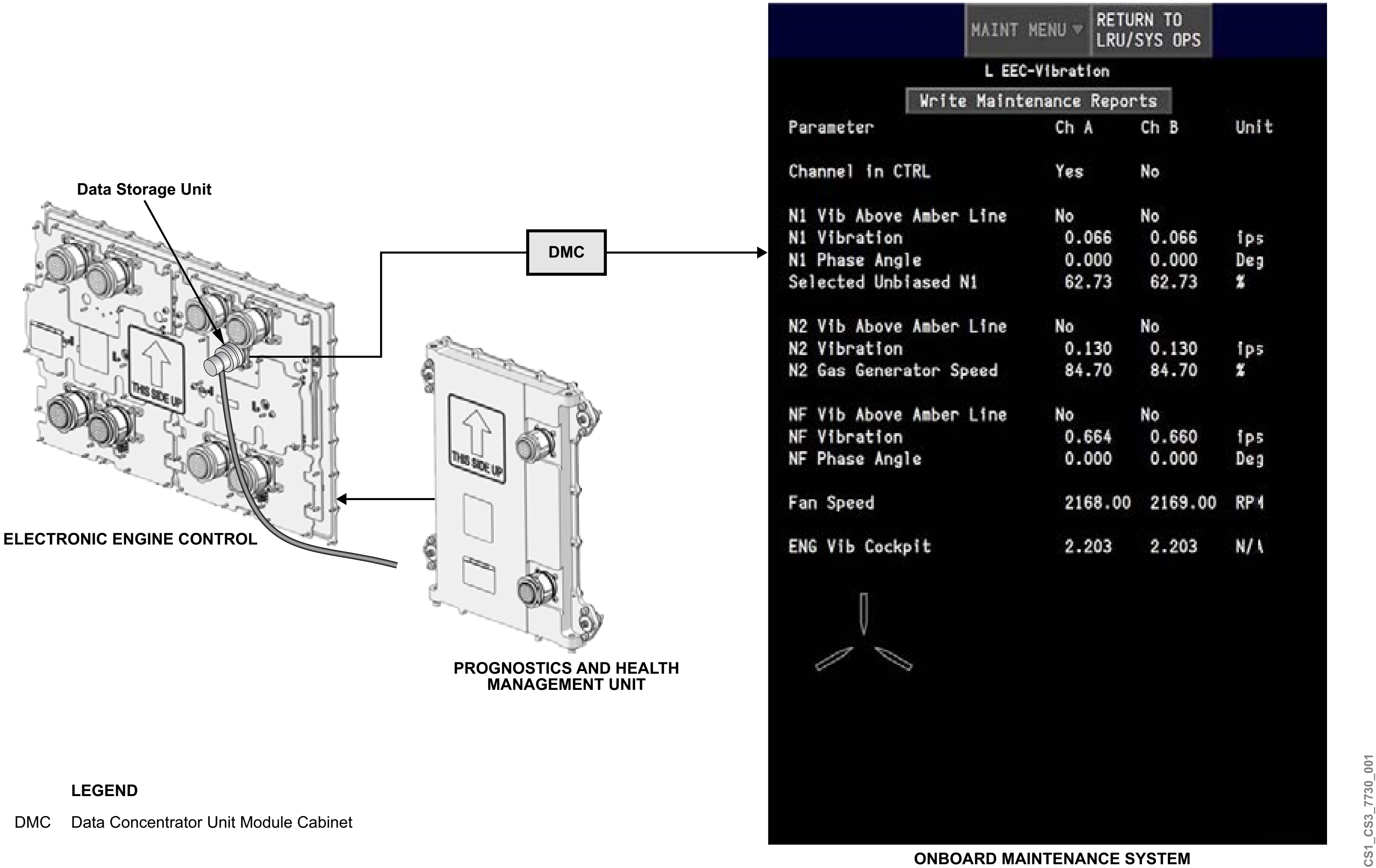
Use the OMS to get weight and angle data for the fan. Use this data with the PW1500G Trim Balance Helper Tool provided in this Special Instruction to develop a balance solution.

Perform a baseline engine vibration survey as follows:

1. Start both engines and allow engines to idle for a minimum of 5 minutes to allow the engine to become thermally stable. Make sure all the engine parameters are within the allowable limits.
2. For the engine under test, slowly move the thrust lever forward to 67-69% N1 speed and let the engine become stable for 3 minutes. The engine not being tested can be operated as required during this survey.
3. For the tested engine, slowly move the thrust lever back to MINIMUM IDLE power and let the engine become stable for 3 minutes.

NOTE

Do not run the tested engine at idle power for more than 3 minutes. The engine can cool and the thermal stabilization steps must be repeated.



LEGEND

DMC Data Concentrator Unit Module Cabinet

Figure 26: Fan Trim Balance

Onboard Maintenance System Data

Prior to performing a baseline vibration survey, access the onboard maintenance system (OMS) system as follows:

1. Select Perform LRU/Systems Operations
2. Select ATA 73 - Engine Fuel & Control
3. Select Left or Right Engine EEC for the engine under test
4. Select Vibration from the drop-down menu to monitor the engine vibration data

At the completion of the vibration survey and prior to engine shutdown, Data Reader is selected to get the fan trim balance solution weight and solution angle.

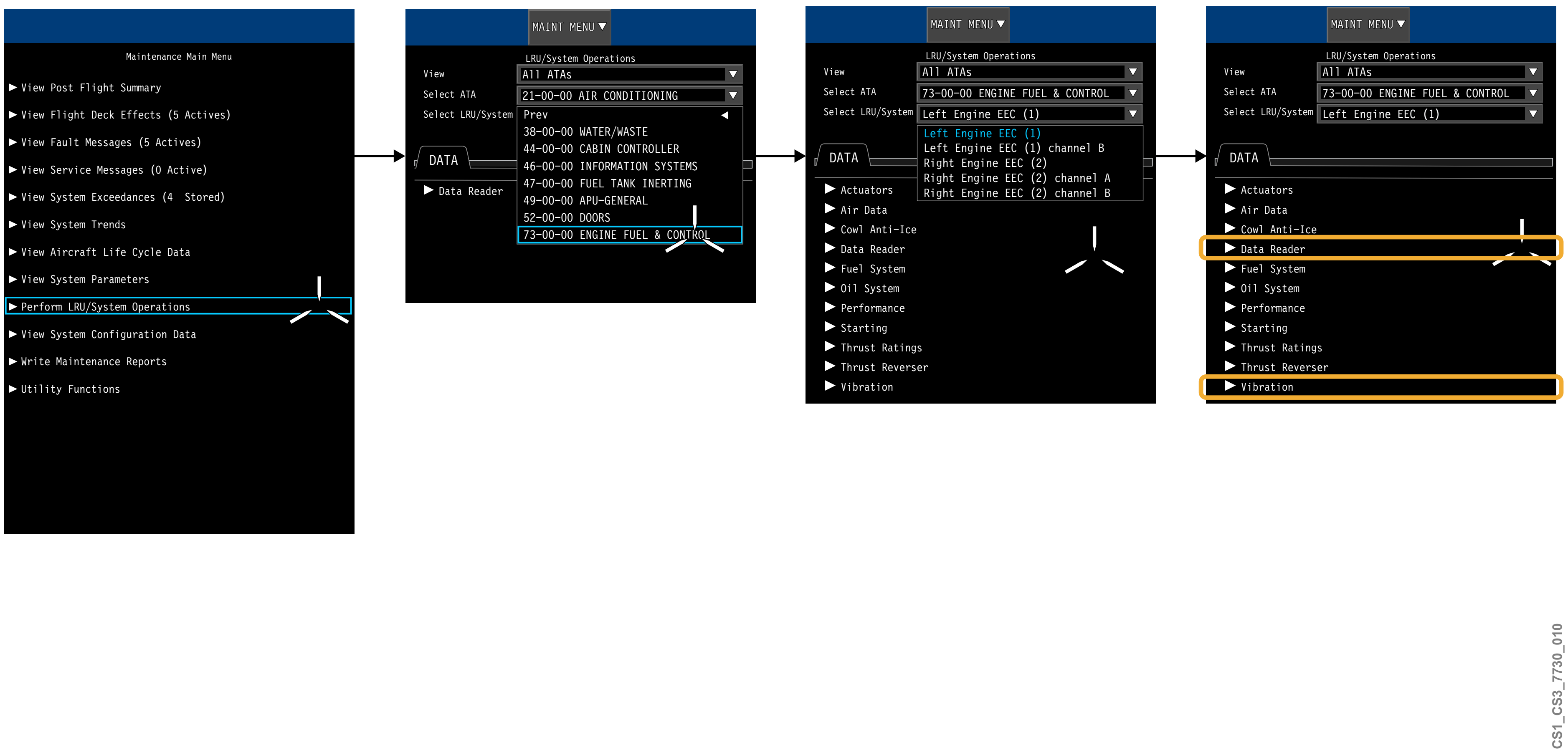


Figure 27: Onboard Maintenance System Data

Monitor Engine Vibration

The engine vibration data for N1, N2, and the fan Nf can be monitored during the engine run. For the engine being tested, slowly move the thrust lever to achieve the following settings:

- Perform a slow acceleration (40-45 sec) to 60 +/-1% N1. Stabilize for 1 minute
- Perform a slow acceleration (5-10 sec) to 64 +/- 1% N1. Stabilize for 1 minute
- Perform a slow acceleration (5-10 sec) to 68 +/- 1% N1. Stabilize for 1 minute
- Perform a slow acceleration (5-10 sec) to 72 +/- 1% N1. Stabilize for 1 minute
- Perform a slow acceleration (5-10 sec) to 76 +/- 1% N1. Stabilize for 1 minute
- Perform a slow acceleration (5-10 sec) to 80 +/- 1% N1. Stabilize for 1 minute
- Perform a slow acceleration (5-10 sec) to 83 +/- 1% N1. Stabilize for 1 minute
- Perform a slow deceleration (60-90 sec) to MINIMUM IDLE

NOTE

The maximum speed may be limited by TLA rating.

The ENG VIBRATION caution CAS can be disregarded for 30 seconds during this test in order to obtain vibration data as long as it is related to fan vibration.

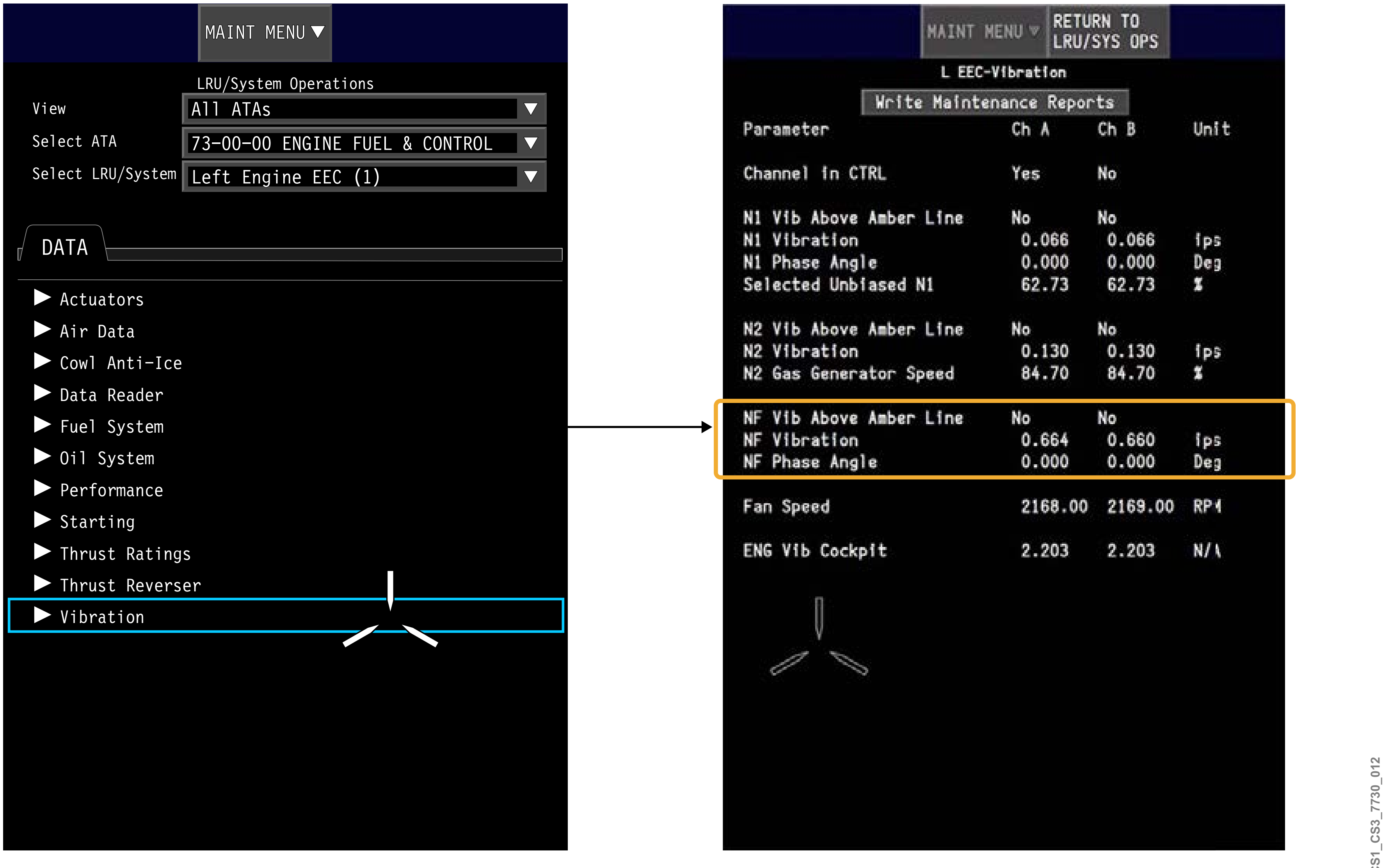


Figure 28: Monitor Engine Vibration

Fan Hub Balance Weights

A 0° reference point is aligned with a dimple on the fan hub shaft. Trim balance weights are installed on either side of the module balance weights. Module balance weights are riveted in place during production or engine rebuild, and should not be disturbed during trim balance procedures.

The fan hub balance weights can be accessed as follows:

1. Remove the inlet cone cover from the inlet cone.
2. Rotate the reference dimple marked on the fan drive shaft to the top dead center (TDC) position (12:00).
3. Note and record any previously installed trim balance weights (spring loaded weights only) and hole location.

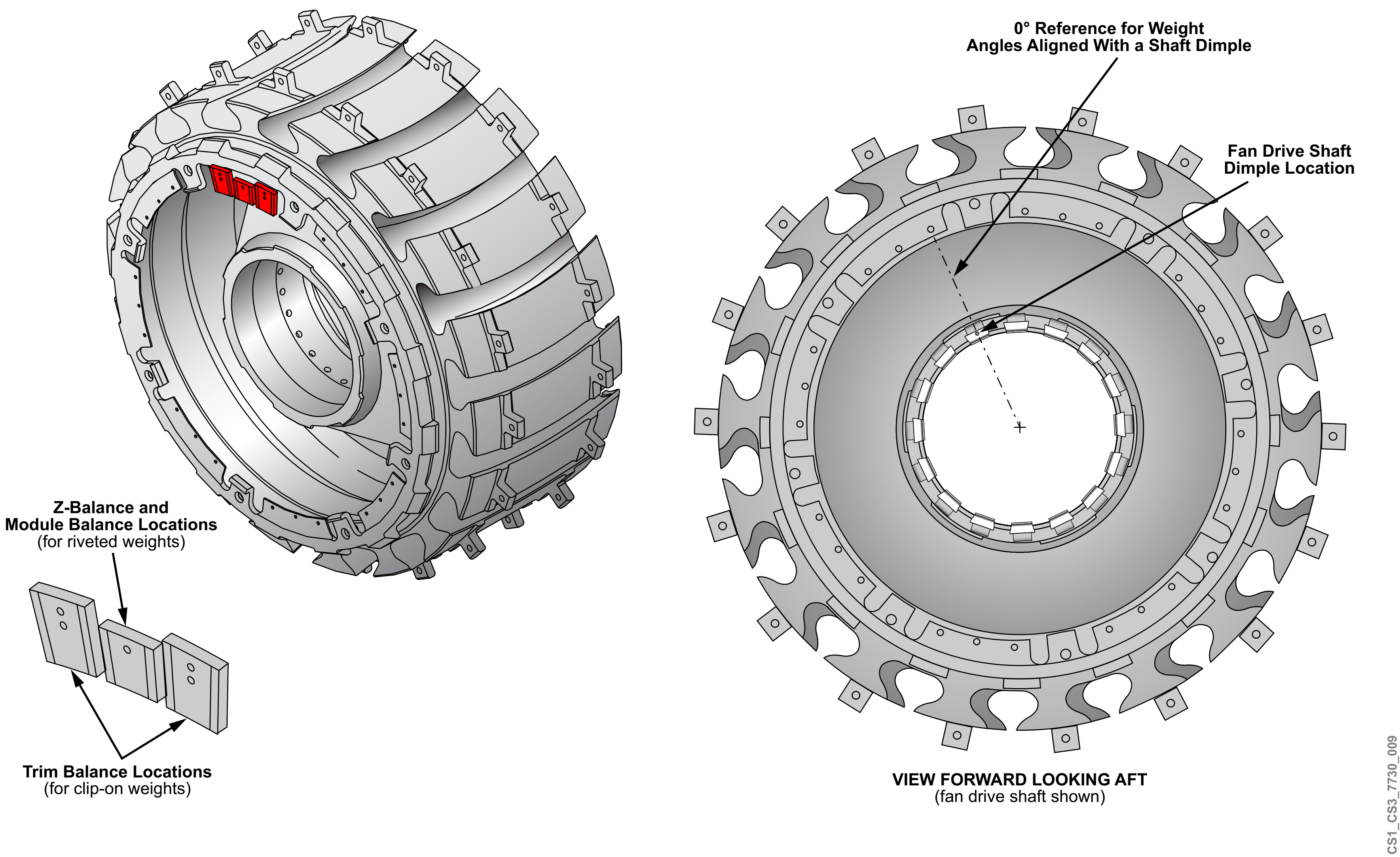


Figure 29: Fan Hub Balance Weights

Review Vibration Data

Access the Data Reader page to get the following data:

- Record the values displayed for the following parameters:
  - Label 47 Fan Trim Balance Solution Weight
  - Label 54 Fan Trim Balance Solution Angle

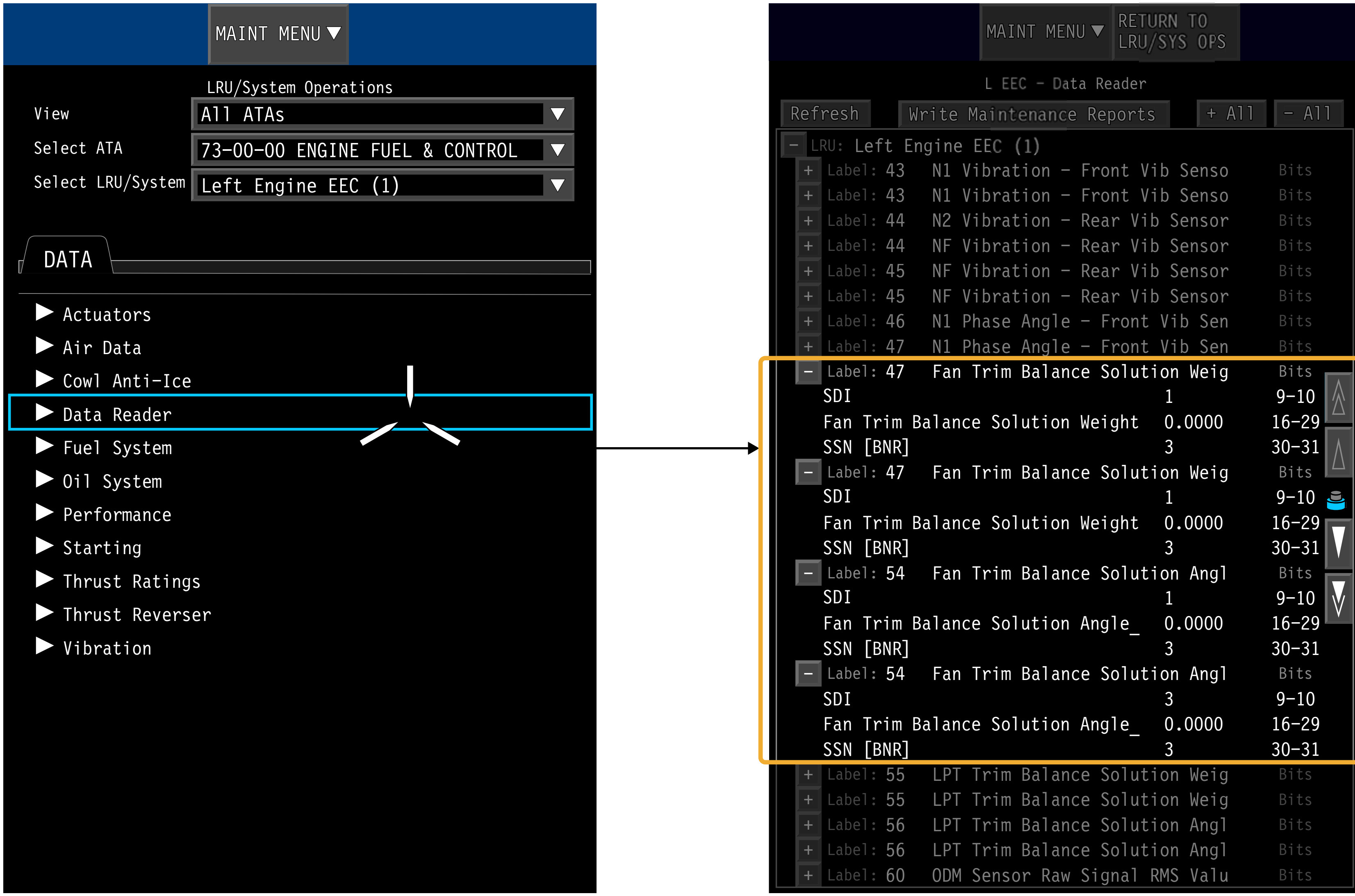


Figure 30: Review Vibration Data

PW1500G Trim Balance Helper Tool

Using the tool instructions provided in the Excel off-line tool attached to the Special Instruction, calculate fan trim balance weight(s) and location information as follows:

1. Use Label 47 Fan Trim Balance Solution Weight_FANWGT and Label 54 Fan Trim Balance Solution Angle_FANANG values to determine the list of part numbers and placement based PW1500G Trim Balance Helper Tool.
2. Enter the previously installed trim balance spring loaded weights into the PW1500G Trim Balance Helper Tool.
3. Enter the Label 47 Fan Trim Balance Solution Weight_FANWGT value from the OMS in the Magnitude field of the PW1500G Trim Balance Helper Tool.
4. Enter the Label 54 Fan Trim Balance Solution Angle_FANANG value from the OMS in the Direction field of the PW1500G Trim Balance Helper Tool.
5. Click SET button.
6. Record results on the recommendation screen.
7. Remove or install fan trim balance weight(s) by size and location specified in the PW1500G Trim Balance Helper Tool.
8. Ensure weights are installed with spring forward.
9. Install the inlet cone cover to the inlet cone.

Perform Test 8 - Vibration Analysis as per the Aircraft Maintenance Publication. Observe the engine vibration data during engine run with the OMS

Further fan trim balance is not necessary once Nf vibration is under 0.5 IPS or 1.8 units.

If the vibration survey data indicates that the fan vibration levels are unacceptable, repeat the procedure.

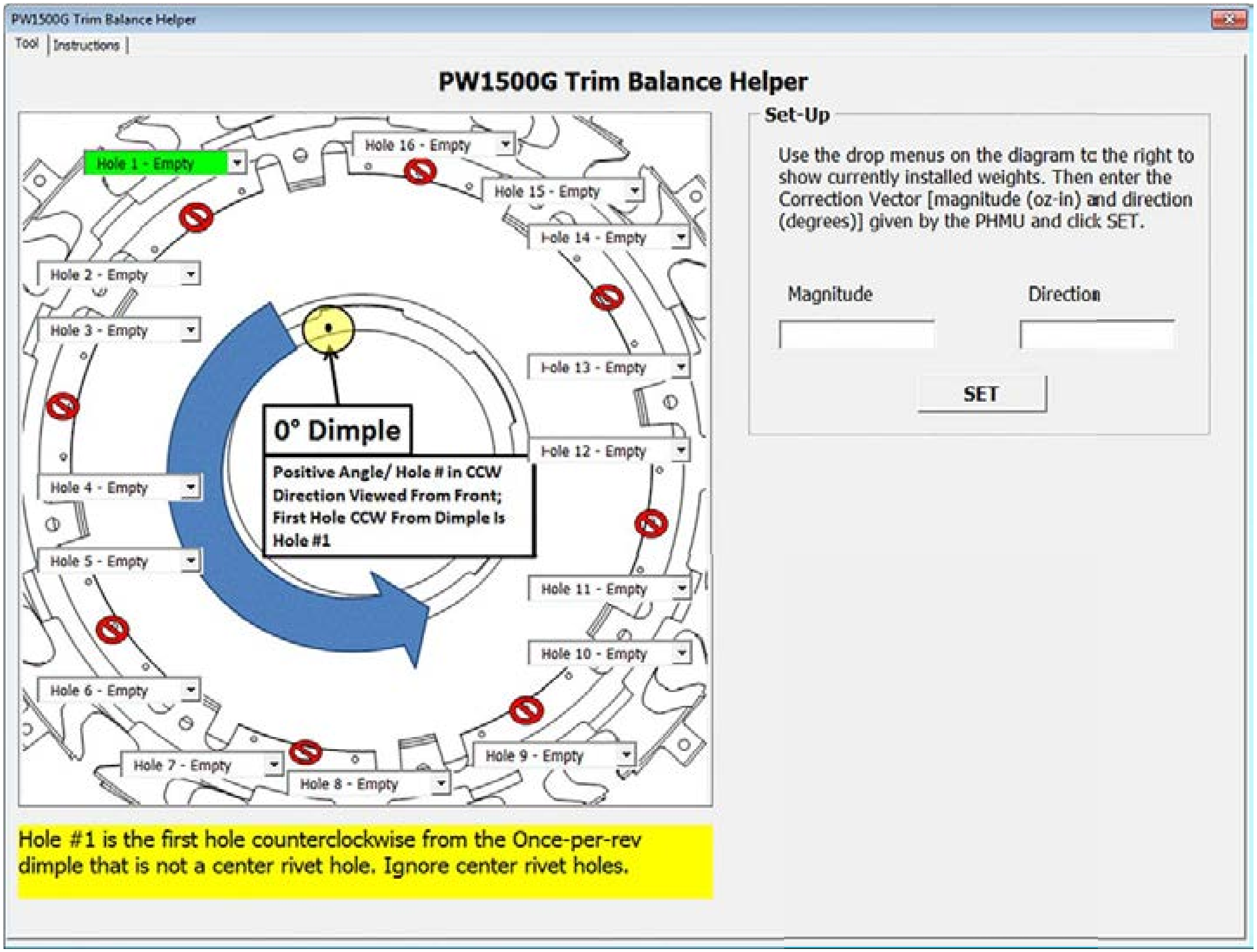


Figure 31: PW1500G Trim Balance Helper Tool

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ATA 78 - Exhaust

BD-500-1A10
BD-500-1A11

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EXHAUST - CHAPTER BREAKDOWN

78-36 THRUST REVERSER STRUCTURE

GENERAL DESCRIPTION

The PW1500G engine has a cascade-type thrust reverser (T/R). Two thrust reverser doors make up the thrust reverser structure. The thrust reverser doors enclose the engine core section and provide the fan bypass airflow path for forward thrust. The composite inner fixed structure (IFS) provides ventilation, fire suppression, and noise attenuation of the engine core.

A torque box, mounted on the leading edge of the IFS, provides the mounting points for the actuators, the cascades, the blocker doors and drag links.

The outer section of the thrust reverser door houses the translating sleeves. The translating sleeves deploy during landing to direct fan airflow forward and through the cascades to slow the aircraft.

Each thrust reverser door is attached to the pylon at the hinge beam. The latch beam has latches that secure the doors closed.

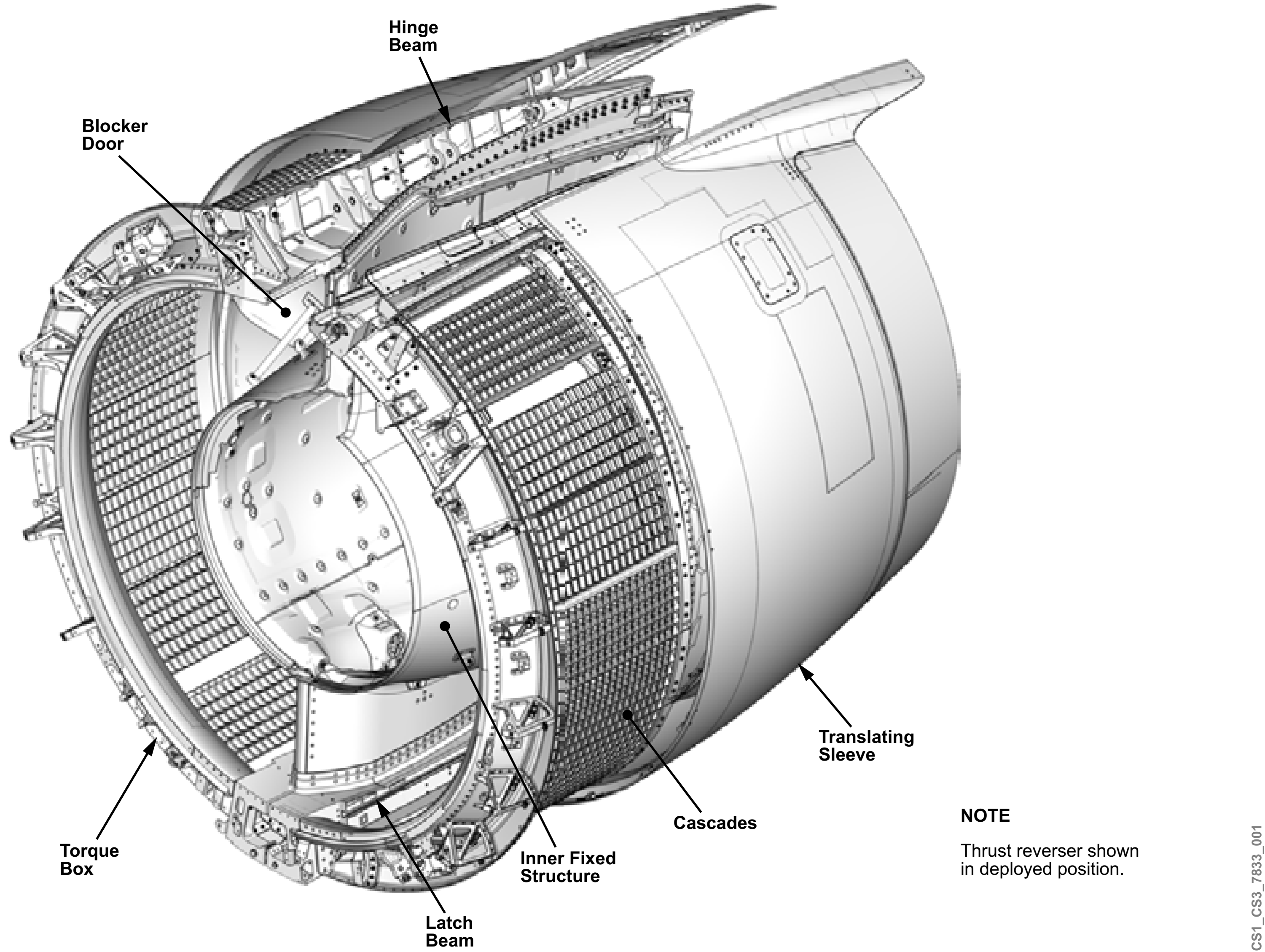


Figure 1: Thrust Reverser Doors

Thrust Reverser Operation

The inner fixed structure (IFS) and the translating sleeve form the thrust reverser (T/R) door. Each T/R door has five composite blocker doors mounted on the translating sleeve. The upper and lower doors are unique in size and cannot be installed at any other position. The blocker doors lay flush with the translating sleeve when in the stowed position.

When the T/R is deployed, the drag links pull the blocker doors into the T/R door. This blocks the fan air from being directed out the exhaust.

The airflow is directed in a controlled manner through the cascades in a forward and outward direction. The forward component of the redirected airflow provides the reverse thrust action which aids in slowing the aircraft. The outward component of the airflow is designed to ensure the reverse thrust air is not redirected into the engine inlet, or against the fuselage and flight control surfaces.

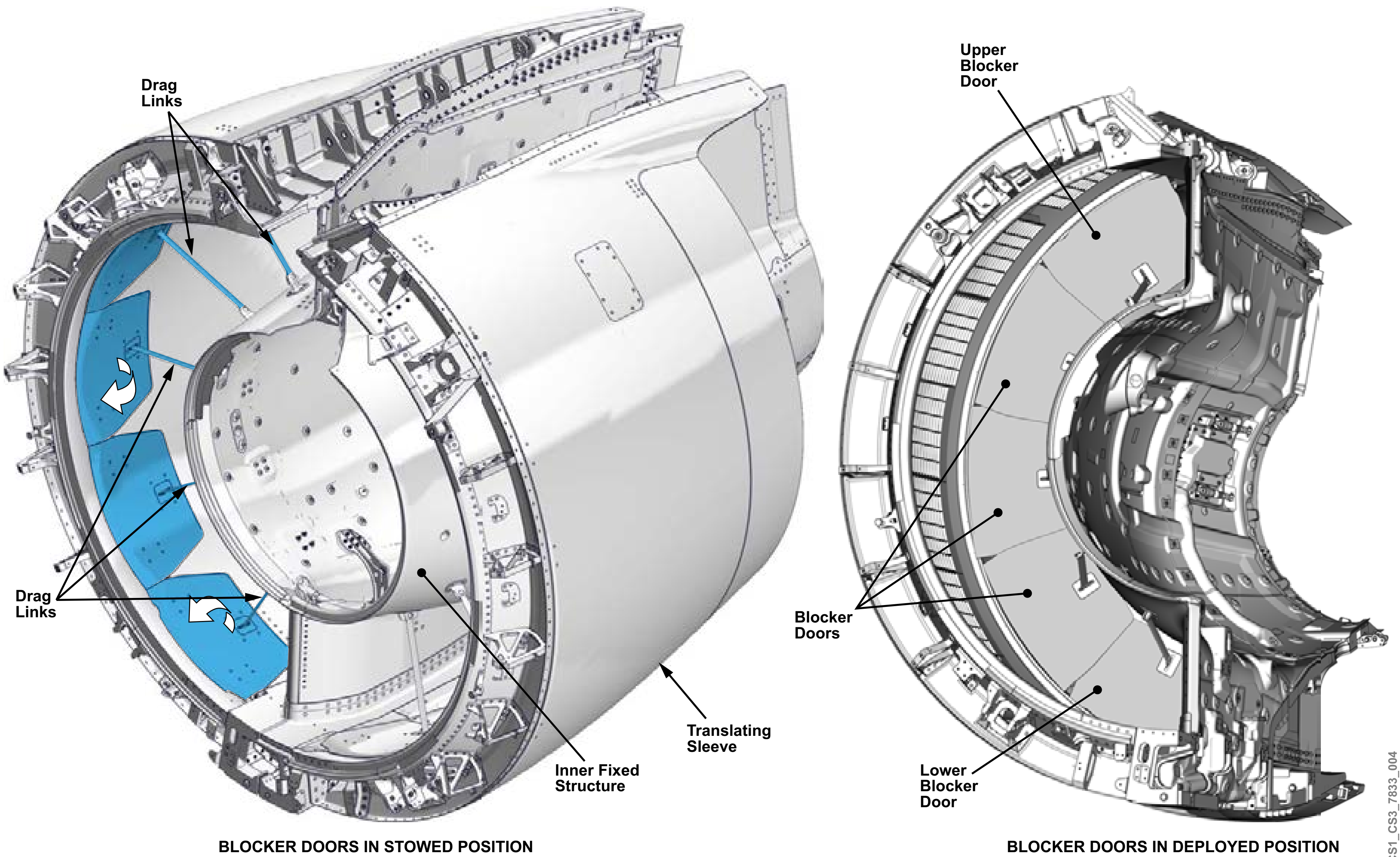


Figure 2: Thrust Reverser Operation

COMPONENT LOCATION

The thrust reverser structure has the following components:

- Inner fixed structure (IFS)
- Hinge beam
- Latch beam
- Torque box
- Cascades

• Translating sleeve

- Blocker doors

INNER FIXED STRUCTURE

The inner fixed structure (IFS) encloses the engine core section.

Hinge Beam

The hinge beams are located on the top of the IFS.

Latch Beam

The latch beams are located at the bottom of the IFS.

Torque Box

The torque box is located at the front of the IFS.

Cascades

The cascades are mounted on the torque box.

TRANSLATING SLEEVE

The translating sleeves are mounted on the IFS at the hinge beam and the latch beam.

Blocker Doors

The 10 blocker doors are mounted on the translating sleeves.

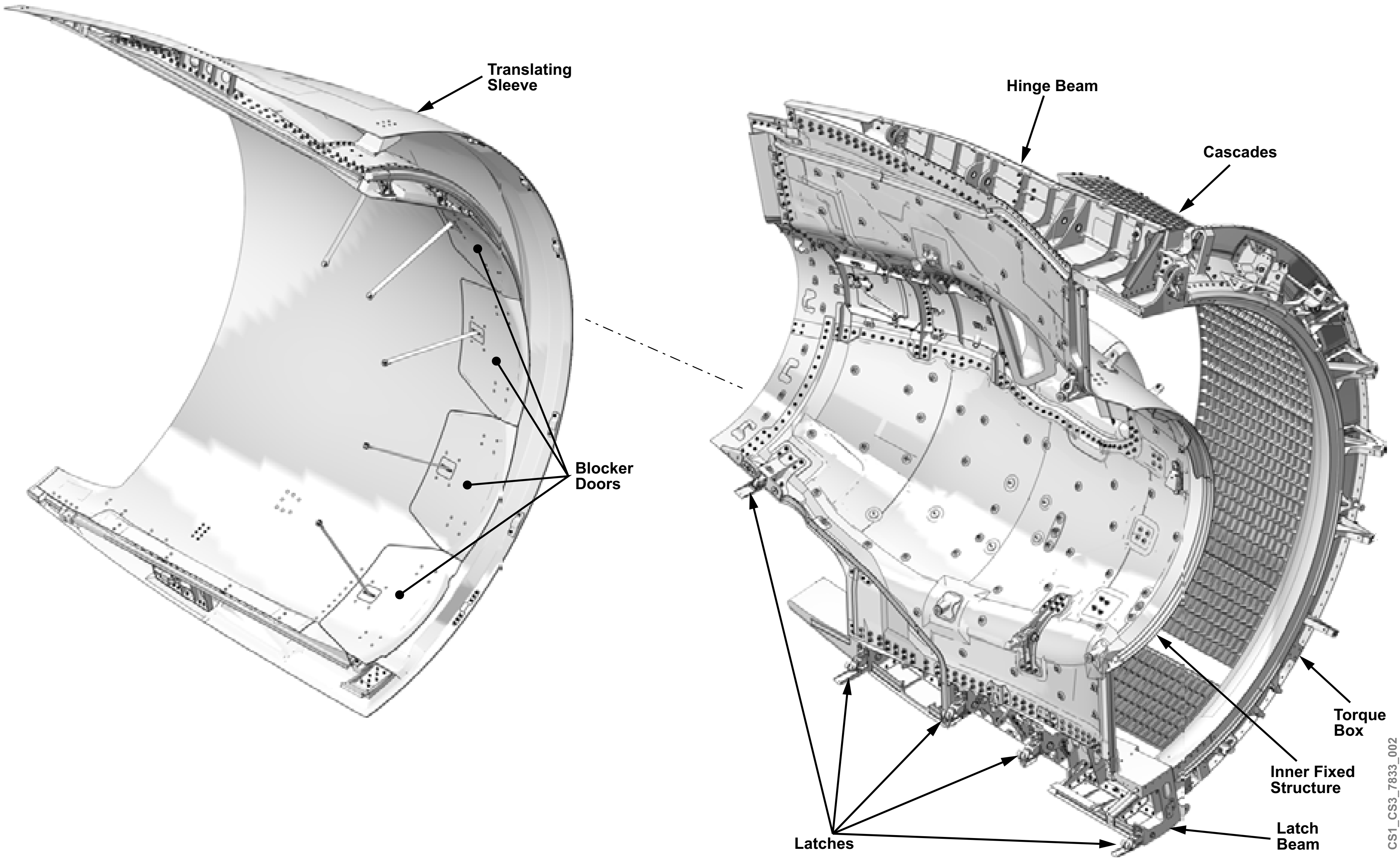


Figure 3: Thrust Reverser Component Location

COMPONENT INFORMATION

TRANSLATING SLEEVES

The translating sleeves deploy aft guided by the primary and secondary sliders that move inside the primary and secondary Teflon coated slider tracks, which are attached to the inner fixed structure (IFS).

Leading edge bumpers rest on lands on the torque box in order to maintain the aerodynamic contour of the translating sleeve when stowed.

The forward section of the inner sleeve have cutouts in which the blocker doors rest flush to provide a streamlined fan bypass air flow.

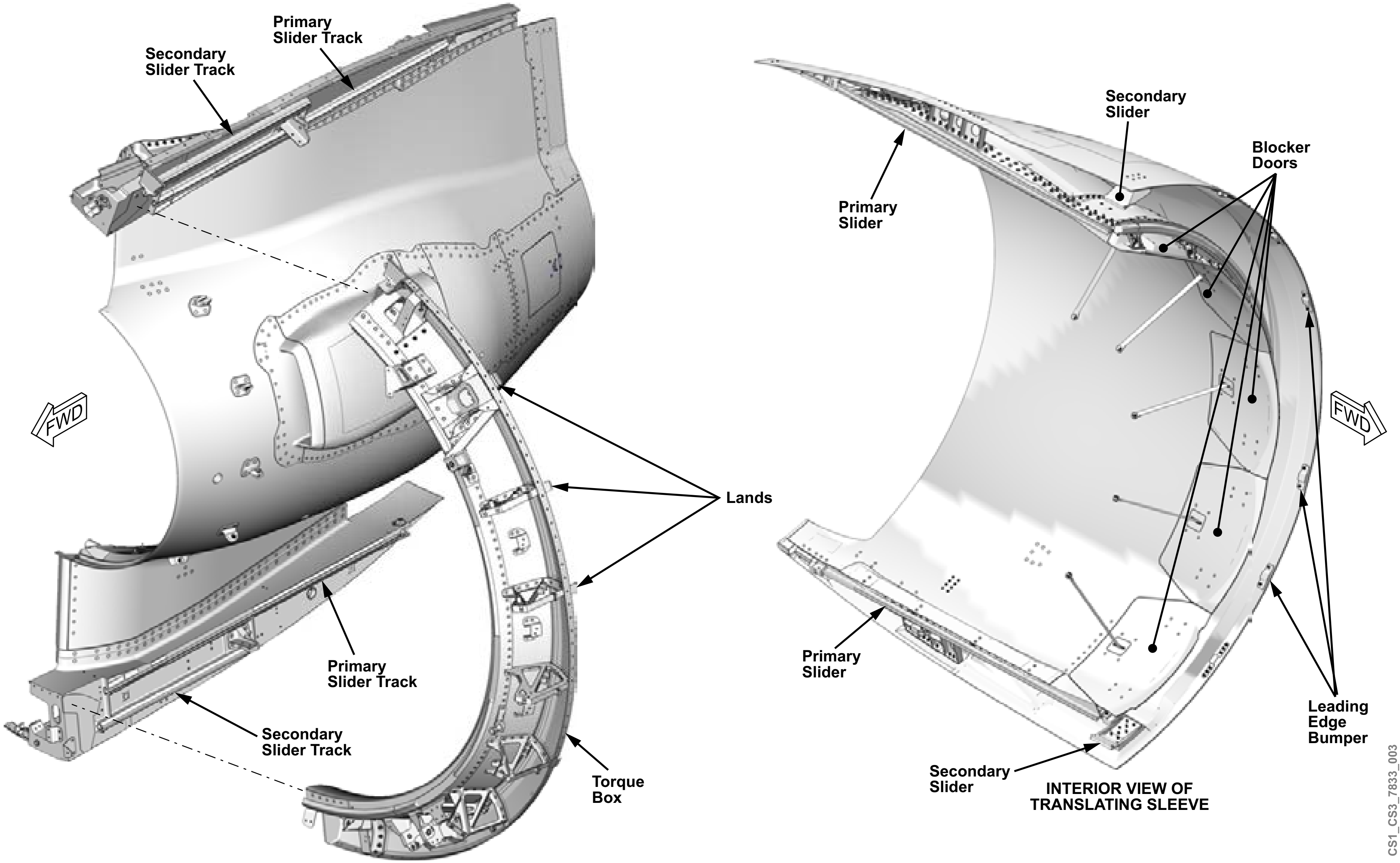


Figure 4: Translating Sleeves and Sliders

BLOCKER DOORS

The blocker doors have hinge fittings that mate with clevises attached to the inner wall of the translating sleeve. The doors rotate on these hinge fittings when the sleeves translate rearward.

Drag links aid in rotating the blocker doors into the T/R door. The drag links are fixed-length aluminum rods with an airfoil cross-section. Teflon coated spherical bearings on both ends of the drag links ensure smooth movement.

An aluminum drag link housing bracket built into the blocker door provides the attachment point for the drag link. A leaf spring mechanism absorbs drag link forces on the blocker door.

Rest pads, mounted on supports, contact the blocker doors to absorb vibrations of the doors when stowed.

Each blocker door has drain holes to prevent water accumulation.

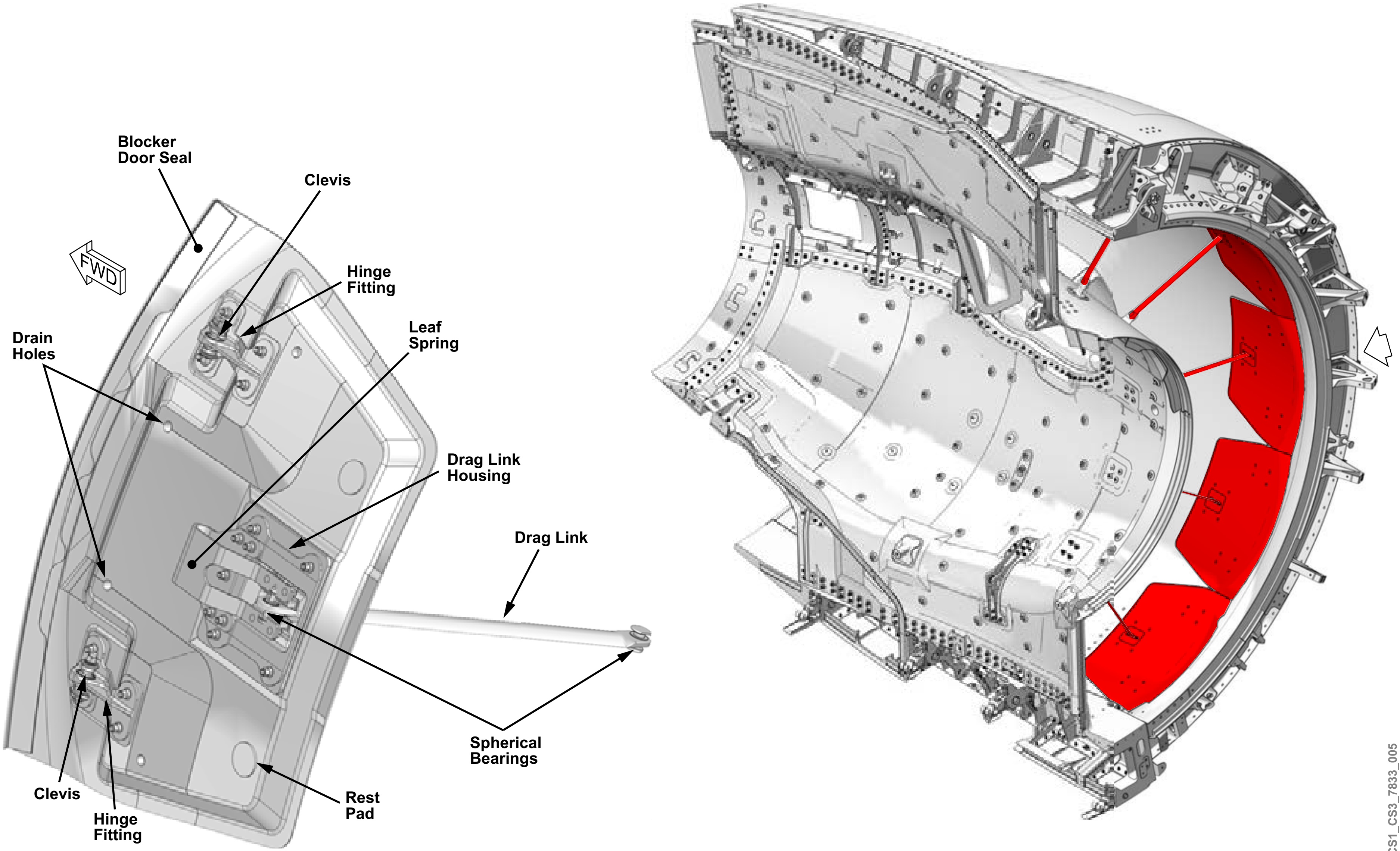


Figure 5: Blocker Door Fittings

CASCADES

The cascades are made up of sixteen individual carbon fiber cascade boxes, designed to turn fan airflow into forward and sideways directions. The cascade boxes are designed to provide specific airflow for their location on the engine.

The cascade boxes have fastener patterns that ensure they can not be installed incorrectly.

Each cascade box mounts between the torque box and the aft cascade ring. Flow blockers are installed at the hinge beam and latch beam to prevent the airflow from leaking between the cascade boxes and the beams.

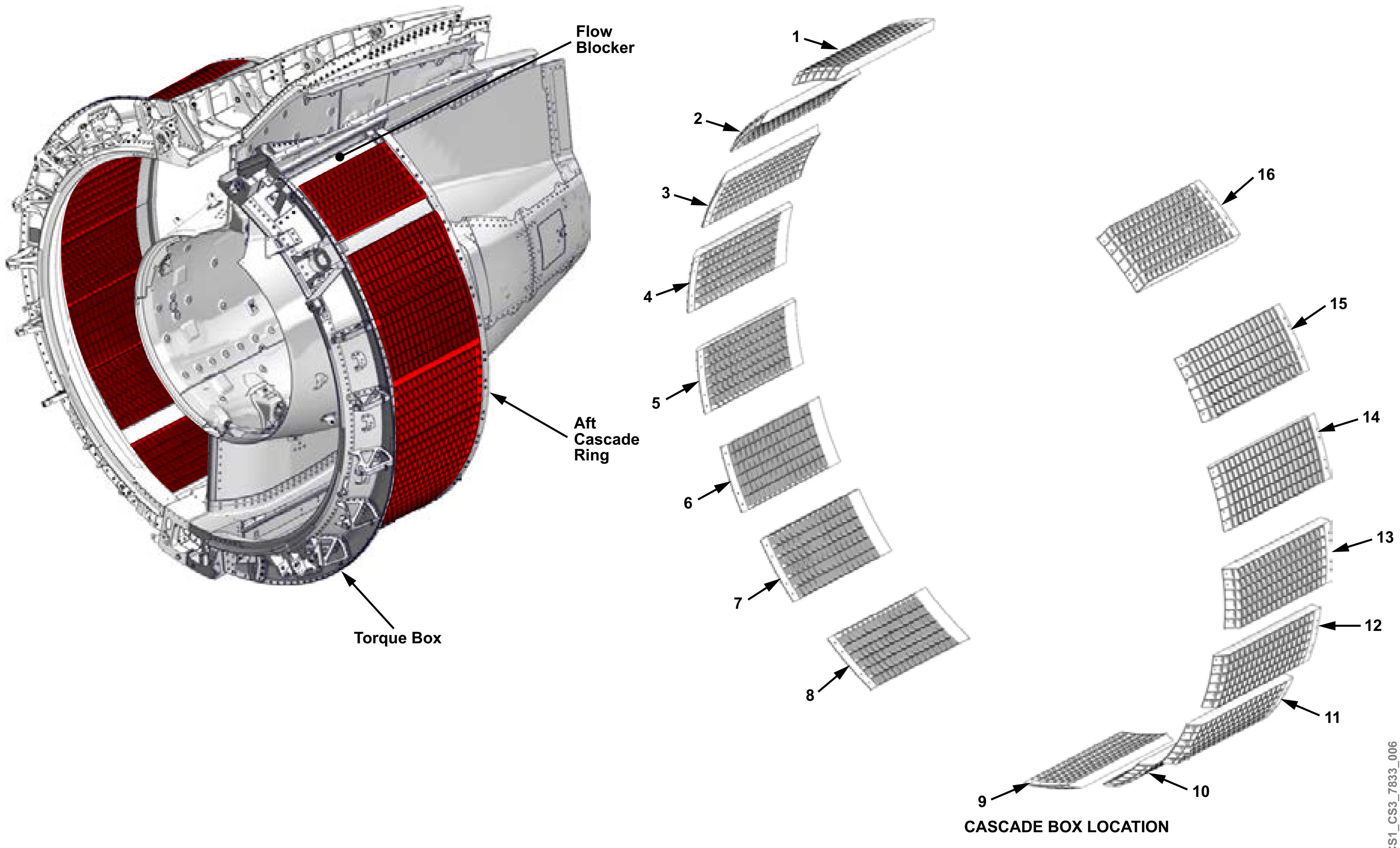


Figure 6: Cascades

78-31 THRUST REVERSER ACTUATION SYSTEM

GENERAL DESCRIPTION

The reverse actuation system has four hydraulic actuators, two track lock units, an isolation control unit (ICU), and a directional control unit (DCU).

The thrust reverser actuation system (TRAS) is controlled through the electronic engine control (EEC), and the control and distribution cabinets (CDCs). The thrust reverser actuation system operates when the aircraft is on the ground. It is controlled by moving the thrust levers on the throttle quadrant assembly (TQA) to the reverse idle position.

Each thrust reverser door has a translating sleeve that is operated by a locking feedback actuator and a locking actuator. The operating of the two actuators is synchronized to ensure proper movement of the translating sleeve. The locking feedback actuator has a linear variable differential transformer (LVDT) for position feedback. The actuators have an internal hydraulically released mechanical lock. A proximity switch monitors the lock status of each actuator.

Two track lock units lock the translating sleeve when the thrust reverser is stowed. They protect against inadvertent thrust reverser deployment. Each track lock unit is controlled by a track lock solenoid valve. The track lock solenoid valve is controlled by the CDC. The track lock unit releases when the translating sleeve is ready to be deployed. The track lock unit relocks the translating sleeve once it stows.

The thrust reverser actuation system (TRAS) has an isolation control unit (ICU) that isolates the system from supply pressure when the system is non-operational and a directional control unit (DCU) that controls the directional functions of the thrust reverser translating sleeves. The ICU and DCU have solenoid valves that are controlled by the EEC.

To allow for dispatch of inoperative thrust reversers (T/Rs), and to provide the additional safety during maintenance activities, a manual inhibit lever is installed on the ICU to isolate the thrust reverser actuation system (TRAS).

The DCU has a directional spool valve operated by a directional control solenoid valve that controls the unlock, deploy, stow and lock sequence when commanded.

The thrust reverser (T/R) is deployed by moving the thrust levers from the forward thrust position to the idle position and then to the reverse idle position. Once the translating sleeves have reached 85% of full travel, the thrust levers can be moved to the maximum reverse thrust position. The EEC provides a higher maximum reverse thrust for a rejected takeoff than for a rollout after landing.

The T/Rs are stowed by returning the thrust levers to the idle position. The thrust levers can not be moved beyond the idle position and engine thrust is limited until the thrust reversers are fully stowed.

During the deploy sequence, a white REV icon is displayed on EICAS. When the thrust reverser (T/R) is fully deployed, the REV icon turns green. When the T/Rs are stowed, the white REV icon is displayed until the T/R is fully locked.

The locking feedback actuator, locking actuator, and track lock unit provide mechanical protection against an inadvertent thrust reverser (T/R) deployment.

The EEC controlled ICU and DCU, and the CDC controlled track lock solenoid valves provide electrical protection against an inadvertent thrust reverser (T/R) deployment.

The EEC monitors the T/R position through the locking feedback actuator LVDTs, lock status, and the track lock unit lock status. If a thrust reverser inadvertently deploys, the EEC reduces the engine thrust to idle.

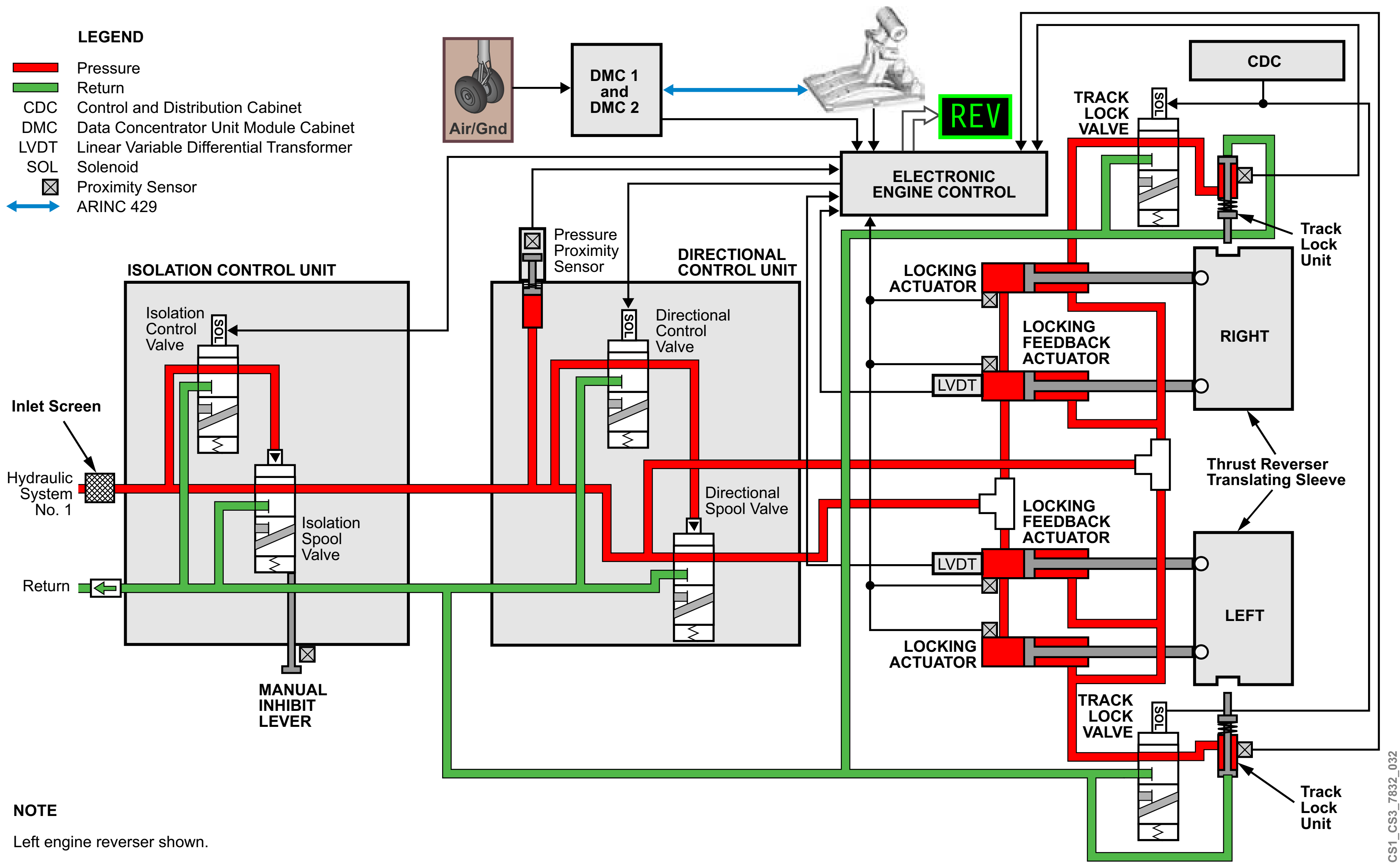


Figure 7: Thrust Reverser Actuation System

COMPONENT LOCATION

The thrust reverser actuating system has the following components:

- Isolation control unit (ICU)
- Directional control unit (DCU)
- Locking feedback actuators (Refer to Figure 9)
- Locking actuators (Refer to Figure 9)
- Flexible drive shaft/deploy tube (Refer to Figure 9)
- Manual drive units (Refer to Figure 9)
- Track lock valves (Refer to Figure 9)
- Track lock units (Refer to Figure 9)

ISOLATION CONTROL UNIT

The isolation control unit (ICU) is located in the aft pylon section and is accessible from access panels on each side of the pylon.

DIRECTIONAL CONTROL UNIT

The directional control unit (DCU) is located in the forward pylon section, above the engine core section area.

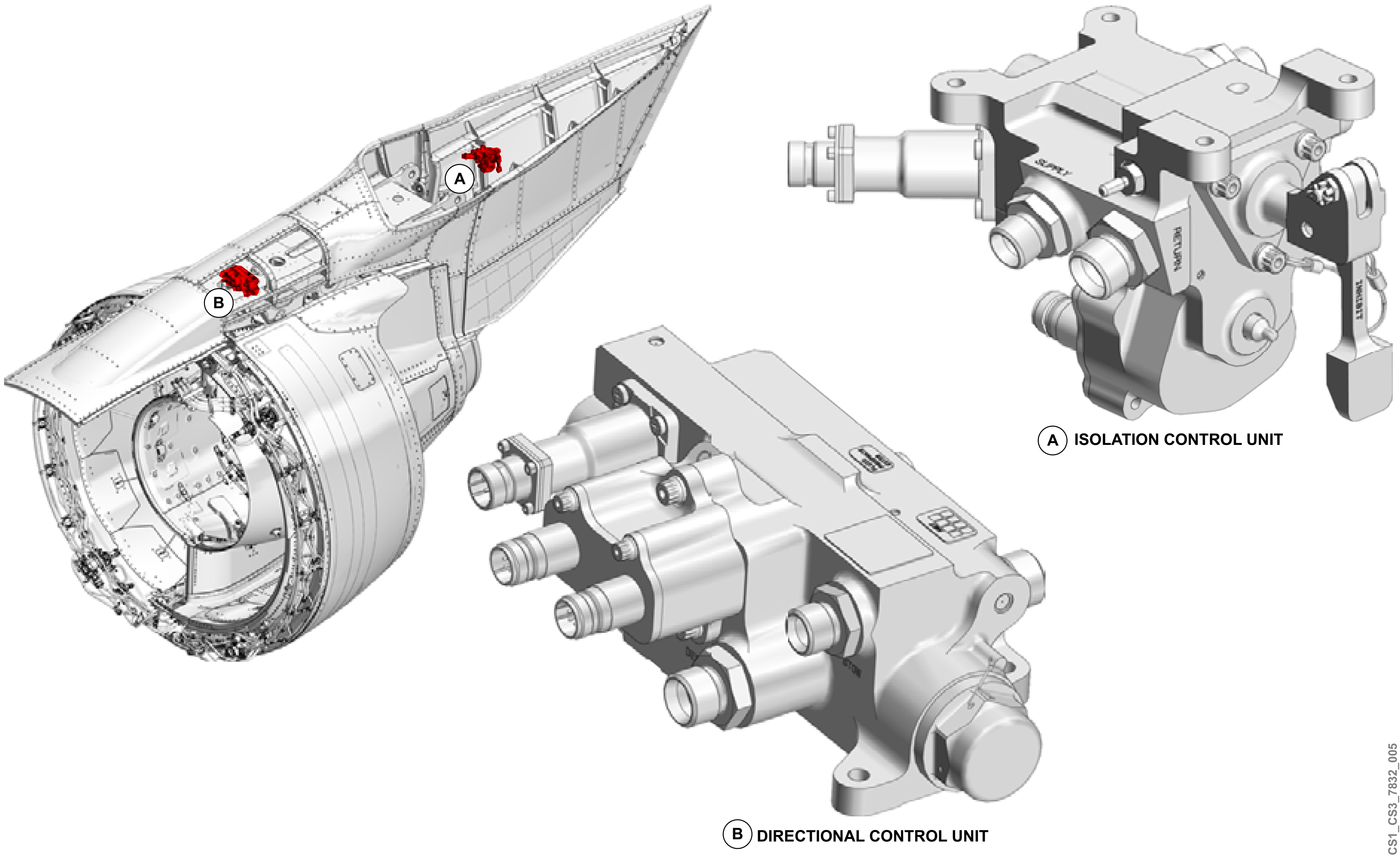


Figure 8: Isolation Control Unit and Directional Control Unit Location

LOCKING FEEDBACK ACTUATORS

The locking feedback actuators are located on each thrust reverser door upper section.

LOCKING ACTUATORS

The locking actuators are located on each thrust reverser door lower section.

FLEXIBLE DRIVE SHAFT/DEPLOY TUBES

The flexible drive shaft/deploy tubes are located on the torque box between the locking actuator, and the locking feedback actuator.

MANUAL DRIVE UNITS

The manual drive units are located on each of the locking actuators.

TRACK LOCK VALVES

The track lock valves are located on each of the thrust reverser door lower section, near the latch beam.

TRACK LOCK UNITS

The track lock units are located on each of the thrust reverser latch beam.

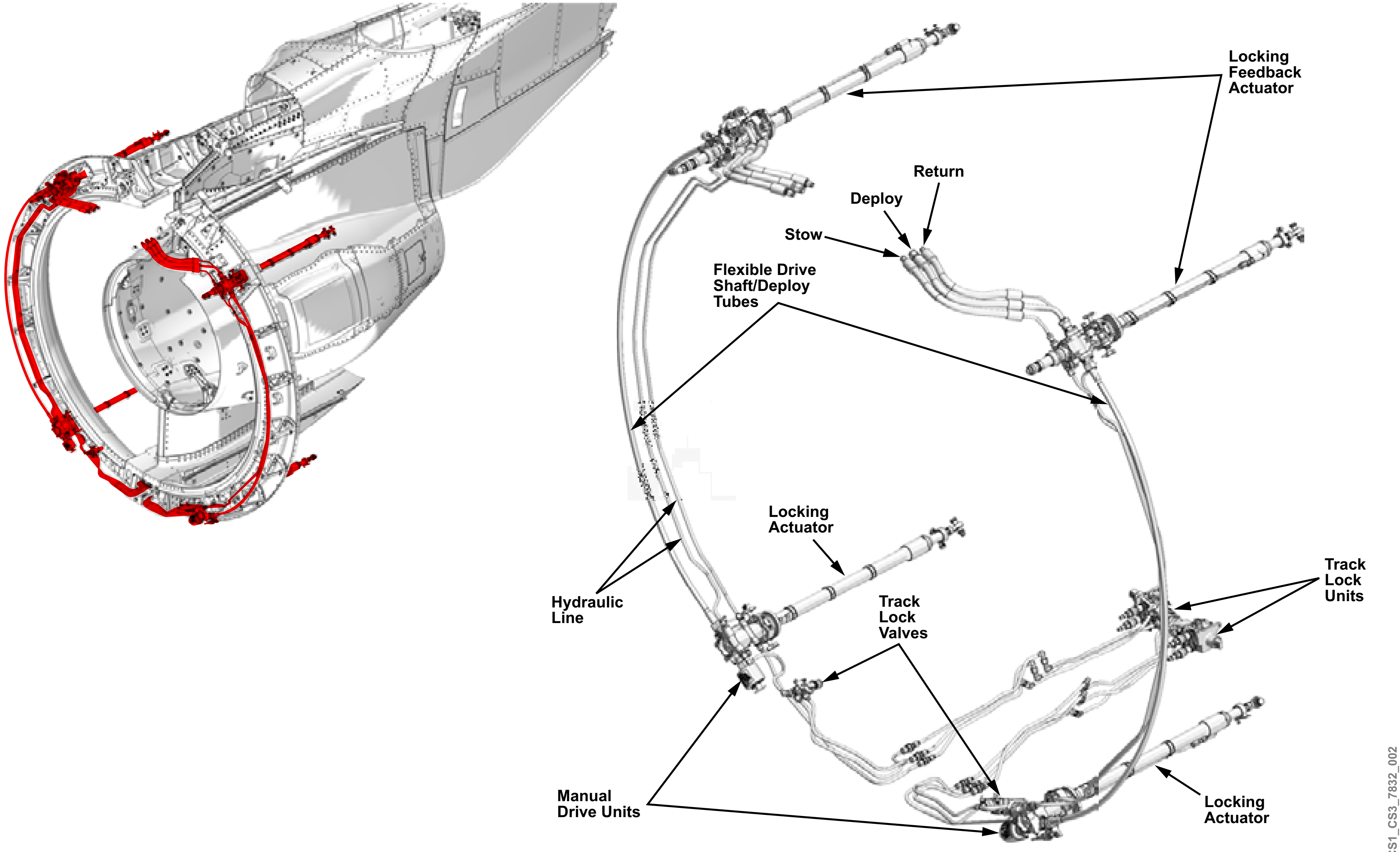


Figure 9: Thrust Reverser Actuation System Component Location

COMPONENT INFORMATION

ISOLATION CONTROL UNIT

The isolation control unit (ICU) isolates the thrust reverser actuating system (TRAS) from the aircraft hydraulic system.

The ICU has a dual-channel solenoid controlled by either electronic engine control (EEC) channel. The solenoid operates a pilot valve that supplies hydraulic pressure to an internal isolation spool valve. The isolation spool valve opens to allow hydraulic pressure from the aircraft hydraulic system to enter the TRAS.

To allow for dispatch of an inoperative thrust reverser (T/R) and to provide additional safety during ground maintenance activities, a manual inhibit lever is installed to prevent the isolation spool valve from operating. An inhibit pin stored on the ICU locks the manual inhibit lever in the inhibit position. The position of the inhibit lever is monitored by dual-channel inhibit proximity sensors.

DIRECTIONAL CONTROL UNIT

The directional control unit (DCU) has a dual-channel solenoid controlled by either EEC channel. The solenoid operates a directional control valve that supplies hydraulic pressure to an internal directional spool valve. The directional spool valve provides hydraulic pressure to the actuators and the track lock valves to control unlocking, deploying, stowing, and locking of the thrust reversers (T/Rs).

A dual-channel proximity pressure sensor provides the status of the hydraulic pressure at the outlet of the ICU to each channel of the EEC.

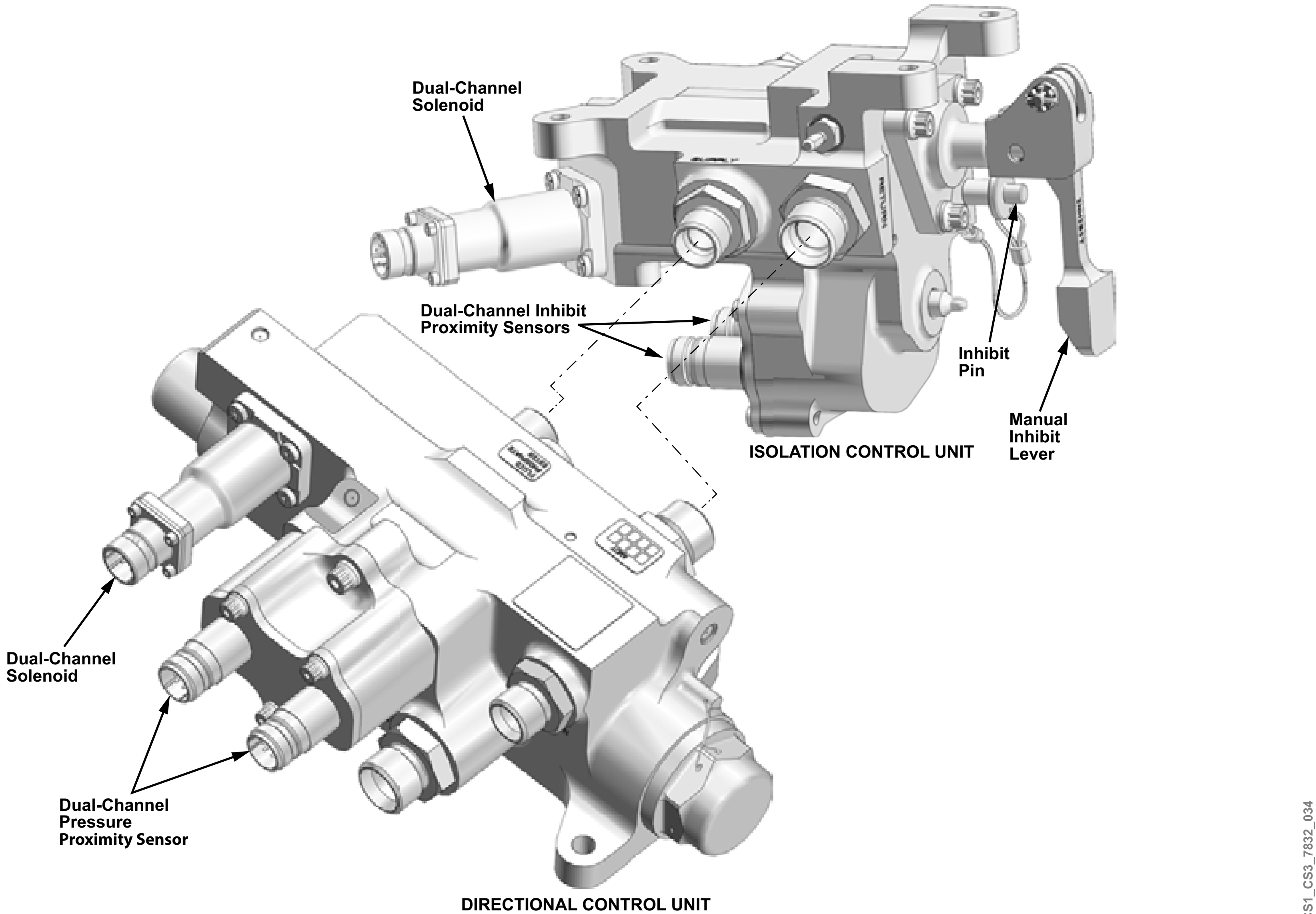


Figure 10: Isolation Control Unit and Directional Control Unit

LOCKING FEEDBACK ACTUATOR

The locking feedback actuator consists of a jackscrew that drives the translating sleeve. The jackscrew is hydraulically operated. The stow tube supplies hydraulic fluid to stow the translating sleeve. The rod end fitting at the aft end of the actuator provides a connection point to the translating sleeve.

The actuator has an internal lock that maintains the actuator in the stowed position when hydraulic pressure is not applied. The lock releases when hydraulic pressure is applied. This allows the translating sleeve to deploy. When the translating sleeve stows, the lock re-engages. A proximity sensor monitors the lock and provides a stowed signal to the electronic engine control (EEC). The actuator has a manual lock that mechanically unlocks the actuator to allow the translating sleeve to be manually deployed.

The locking feedback actuator connects to the locking actuator through a flexible drive shaft and deploy tube. The flexible drive shaft synchronizes the movement of the two actuators. The deploy tube supplies hydraulic fluid to the locking actuator.

A linear variable differential transformer (LVDT), mounted on the forward end of the actuator, provides position feedback of the translating sleeves to the EEC.

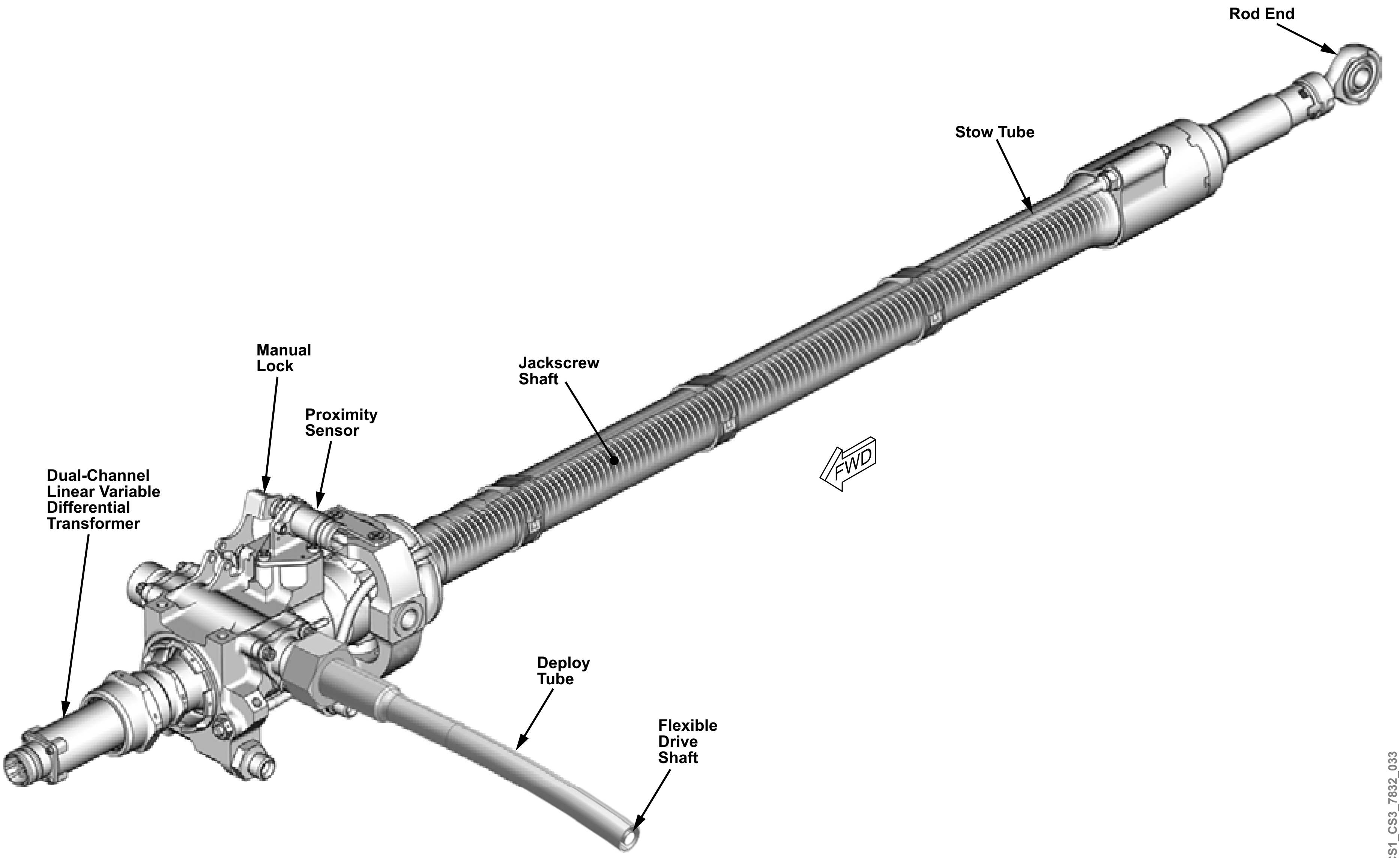


Figure 11: Locking Feedback Actuator

LOCKING ACTUATOR

The locking actuator is similar in construction to the locking feedback actuator except it does not have an LVDT and its position is not monitored.

The actuator has an internal lock that maintains the actuator in the stowed position when hydraulic pressure is not applied. The lock releases when hydraulic pressure is applied. This allows the translating sleeve to deploy. When the translating sleeve stows, the lock re-engages. A proximity sensor monitors the lock and provides a stowed signal to the EEC. The actuator has a manual lock that mechanically unlocks the actuator to allow the translating sleeve to be manually deployed.

The locking actuator connects to the locking feedback actuator through a flexible drive shaft and deploy tube. The flexible drive shaft synchronizes the movement of the two actuators. The deploy tube carries hydraulic fluid to the actuator.

A manual drive unit (MDU) mates with the actuator flexible drive shaft. This rotational input drives the locking actuator through the flex drive shaft, and the locking feedback actuator. A torque limiter in the MDU prevents over torquing of the actuator during manual operation.

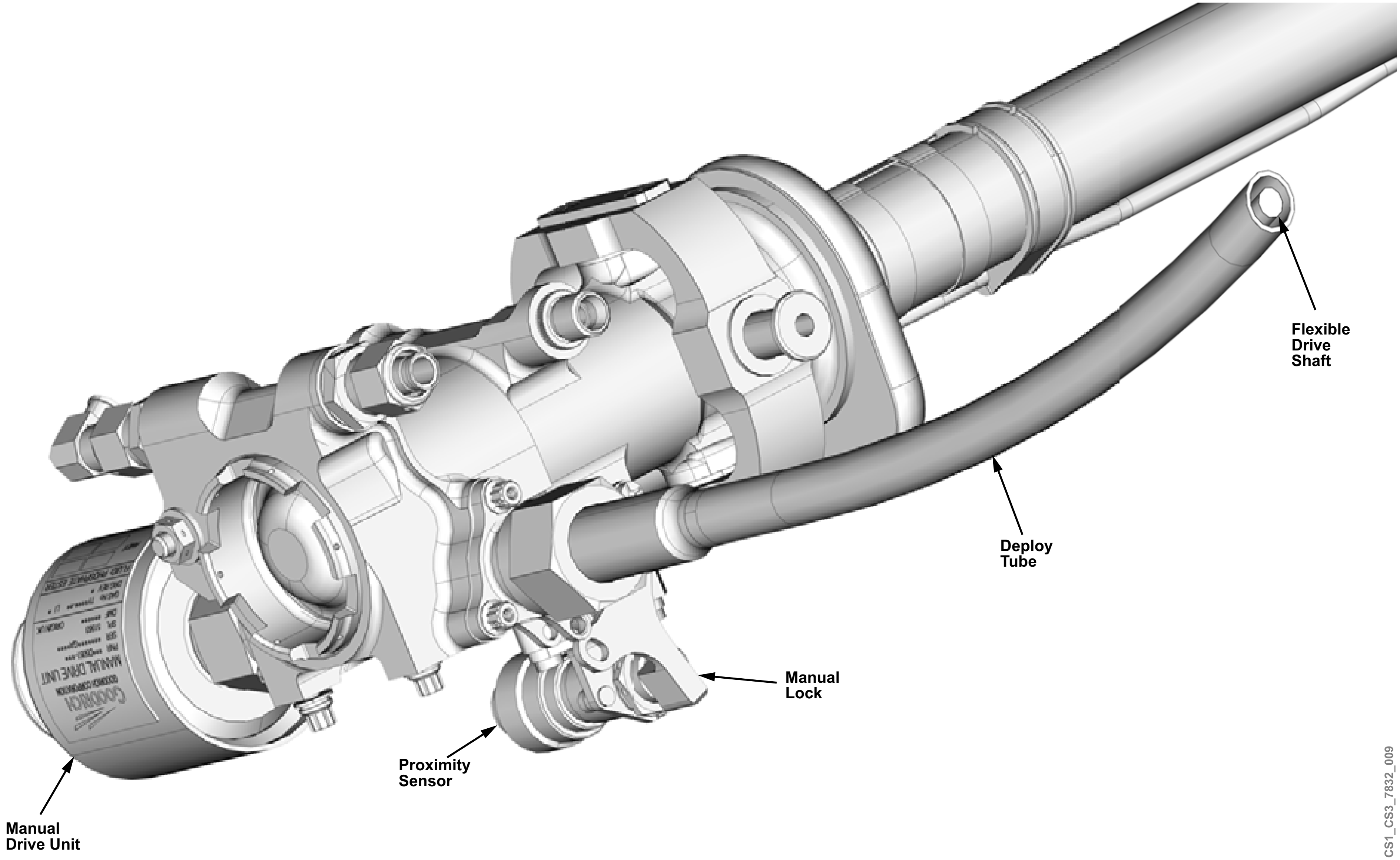


Figure 12: Locking Actuator

LOCKING FEEDBACK ACTUATOR AND LOCKING ACTUATOR INSTALLATION

The locking feedback actuator and locking actuators are mounted in gimbals on the torque box. The gimbal allow the actuator to pivot as the translating sleeve deploys. This prevents the translating sleeve from jamming.

The rod end fitting at the aft end of the actuator is attached to the translating sleeve. A panel provides access to the rod end fitting.

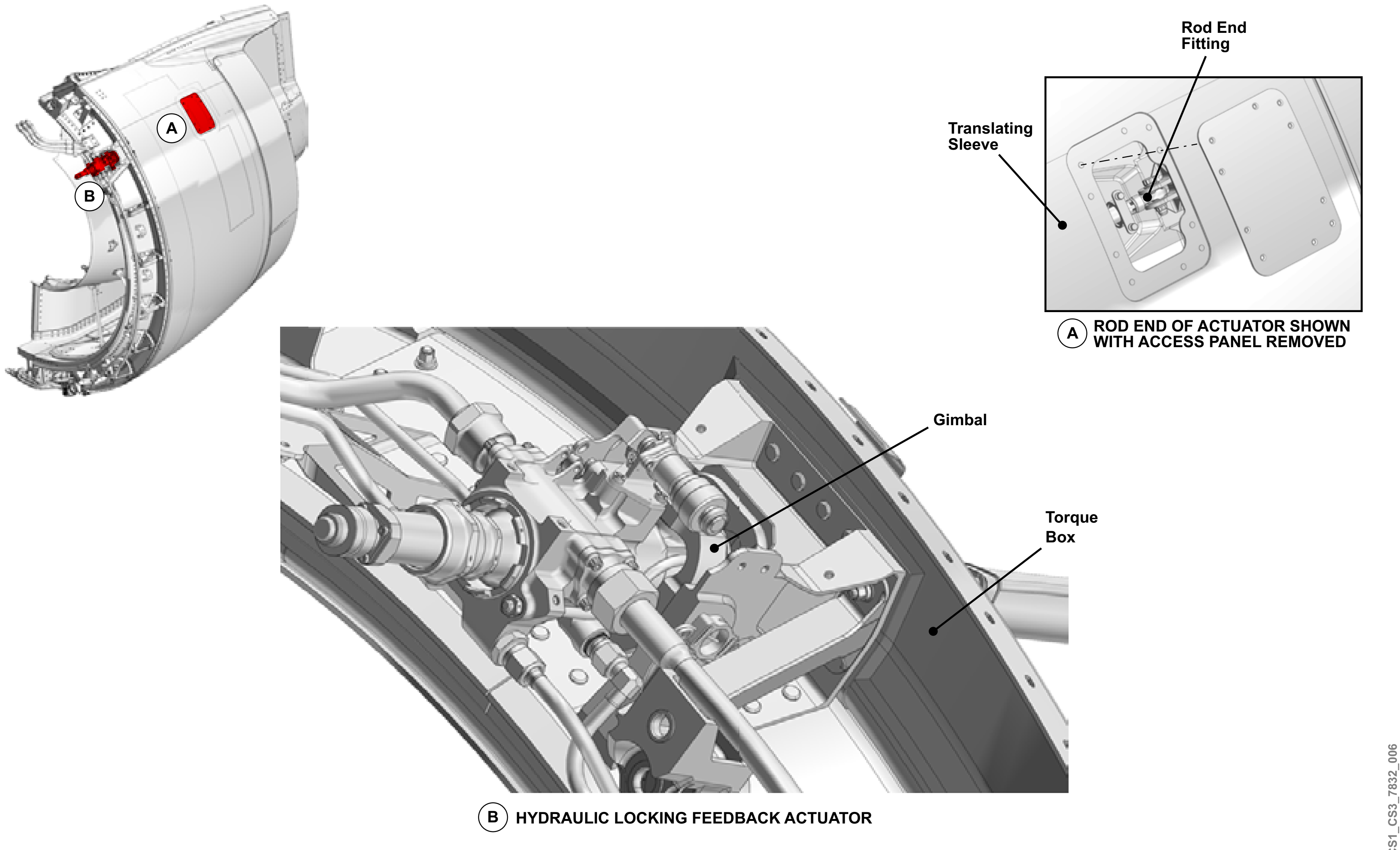


Figure 13: Locking Feedback Actuator and Locking Actuator Installation

TRACK LOCK UNIT

The track lock unit is installed on the latch beam of each thrust reverser door. The track lock unit is operated by hydraulic pressure supplied through the track lock valves.

The track lock unit has a spring-loaded bolt that protrudes into the path of the translating sleeve. In the event of inadvertent deployment, the track lock unit prevents the translating sleeve from deploying. The bolt retracts when hydraulic pressure is supplied. Two proximity sensors monitor the position of the track lock unit bolt and provide the lock status to the EEC.

An unlock lever is used to retract the bolt when the translating sleeve is manually deployed.

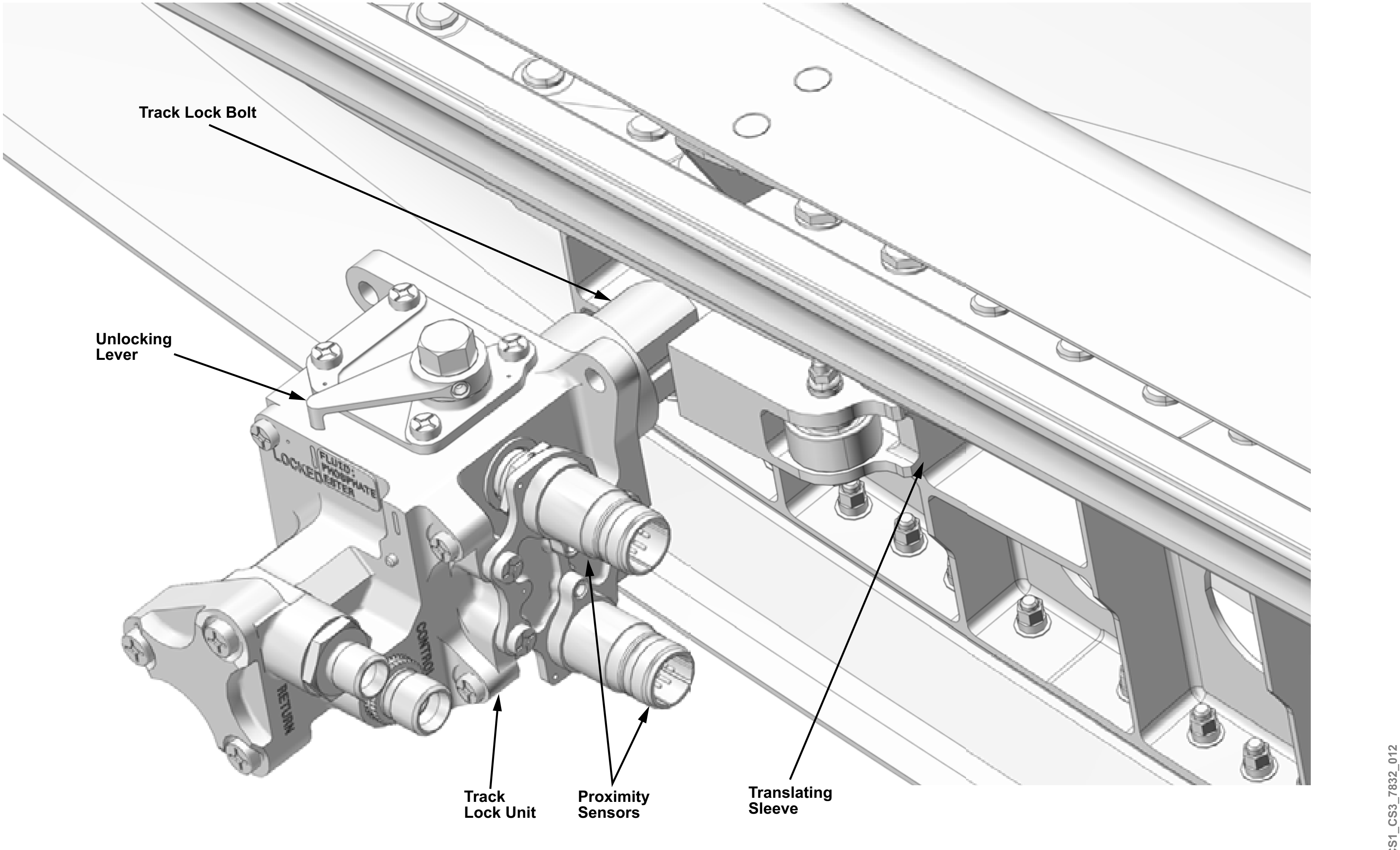


Figure 14: Track Lock Unit

TRACK LOCK VALVE

The track lock valve is a solenoid powered valve used to supply hydraulic pressure to unlock the track lock unit.

When the translating sleeve deploys, the track lock valve solenoid stays energized. This allows the track lock unit to remain unlocked until the thrust reverser is returned to the stowed position.

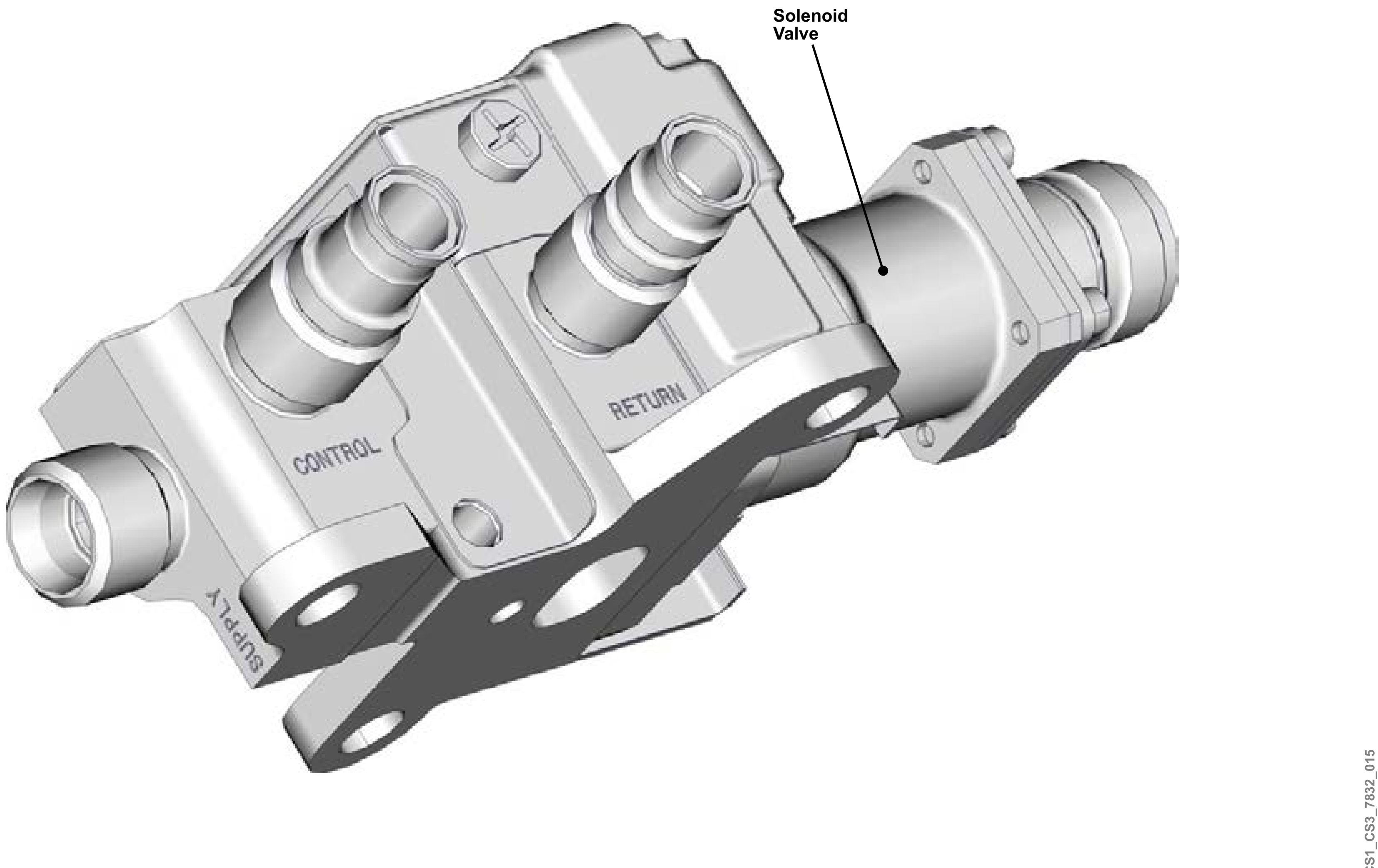


Figure 15: Track Lock Valve

THROTTLE QUADRANT ASSEMBLY

The throttle quadrant assembly (TQA) provides thrust lever position to the EEC. The EEC calculates the engine power required between REV and the MAX REV position 19.5° aft of IDLE.

Each thrust lever has two thrust reverser stow/deploy switches that provide thrust lever positions to the control and distribution cabinet (CDC). These switches are activated by cams at 1.85° aft of thrust lever idle position.

The TQA baulk mechanism prevents the thrust levers from being moved beyond 7.5° aft of IDLE until the thrust reverser translating sleeves are fully deployed. The baulk mechanism prevents the thrust levers from moving 2.5° forward of idle position until the translating sleeves are fully stowed.

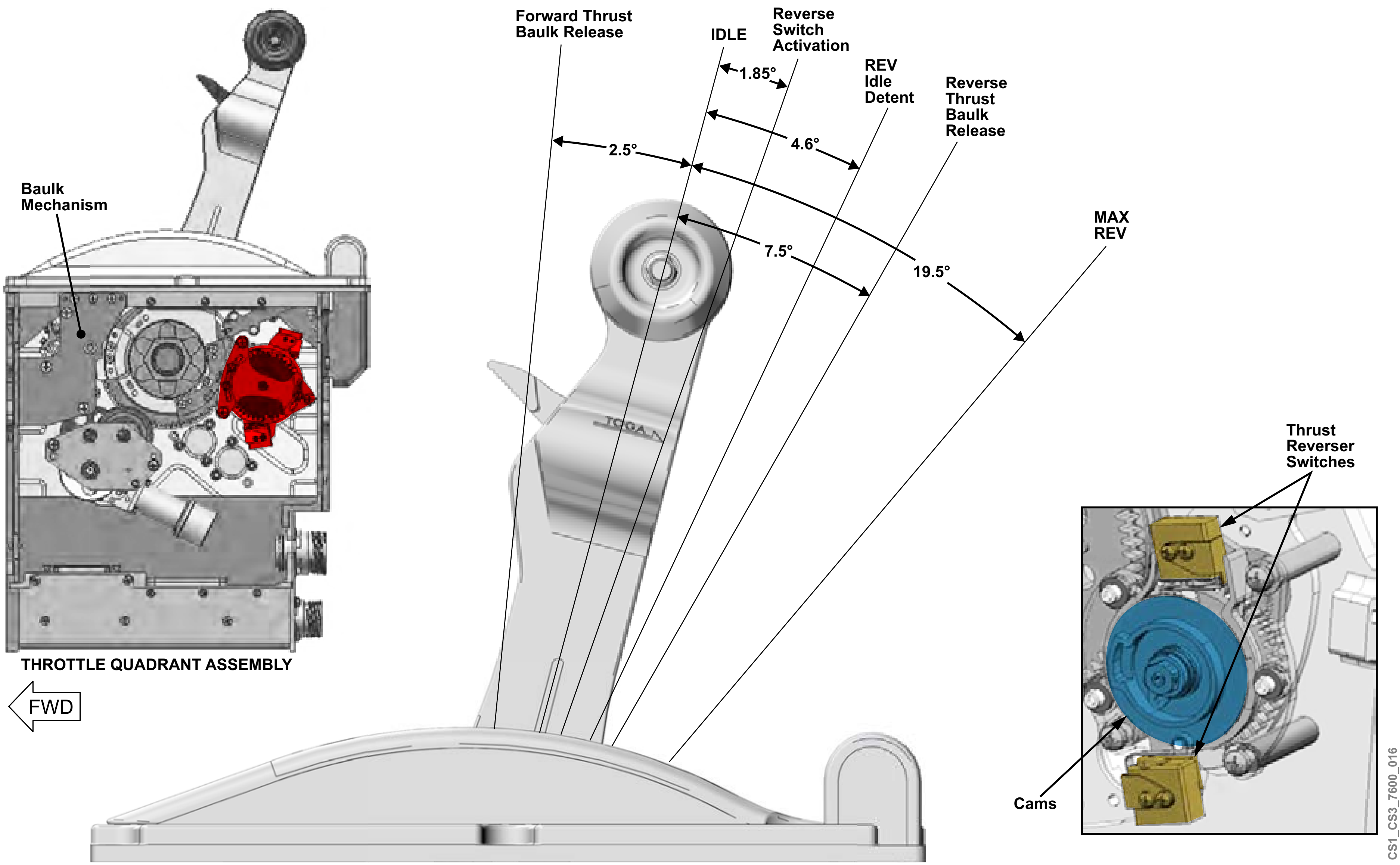


Figure 16: Throttle Quadrant Assembly

DETAILED COMPONENT INFORMATION

LOCKING FEEDBACK ACTUATOR AND LOCKING ACTUATOR

When hydraulic pressure is supplied from the isolation control unit (ICU), it is routed through the stow tube, to the stow port of the actuator, keeping the actuator stowed and locked.

When the thrust reverse is selected, hydraulic pressure, supplied through the deploy port is directed to the deploy side of the actuator. The hydraulic pressure pushes the lock sleeve clear of the lock tine, allowing the jackscrew to rotate.

The hydraulic pressure acts on the internal side of the jackscrew. Both sides of the actuator are now pressurized. Since the jackscrew surface area is greater than the stow side of the actuator piston, the actuator deploys. When the thrust reversers (T/Rs) are commanded to stow, the deploy side of the actuator is directed to the return port and the hydraulic pressure on the stow side drives the actuator to the stow position.

While deploying or stowing, the jackscrew worm wheel turns the flexible drive shaft worm gear. The flexible drive shaft worm gear is connected to the flexible drive shaft that interconnects the locking feedback actuator to the locking actuator. This allows the actuators to maintain synchronization during the stow or deploy cycle.

The manual lock is used to move the lock sleeve when the translating sleeve is manually deployed.

The LVDT on the locking feedback actuator is integral to the actuator and is not a line replaceable unit (LRU). The LVDT has an internal self-rigging mechanism that operates on the first deploy and stow cycle. The self-rigging mechanism establishes the stowed and locked position for the actuator. All position feedback signals to the engine electronic control (EEC) are referenced from the actuator stowed and locked position.

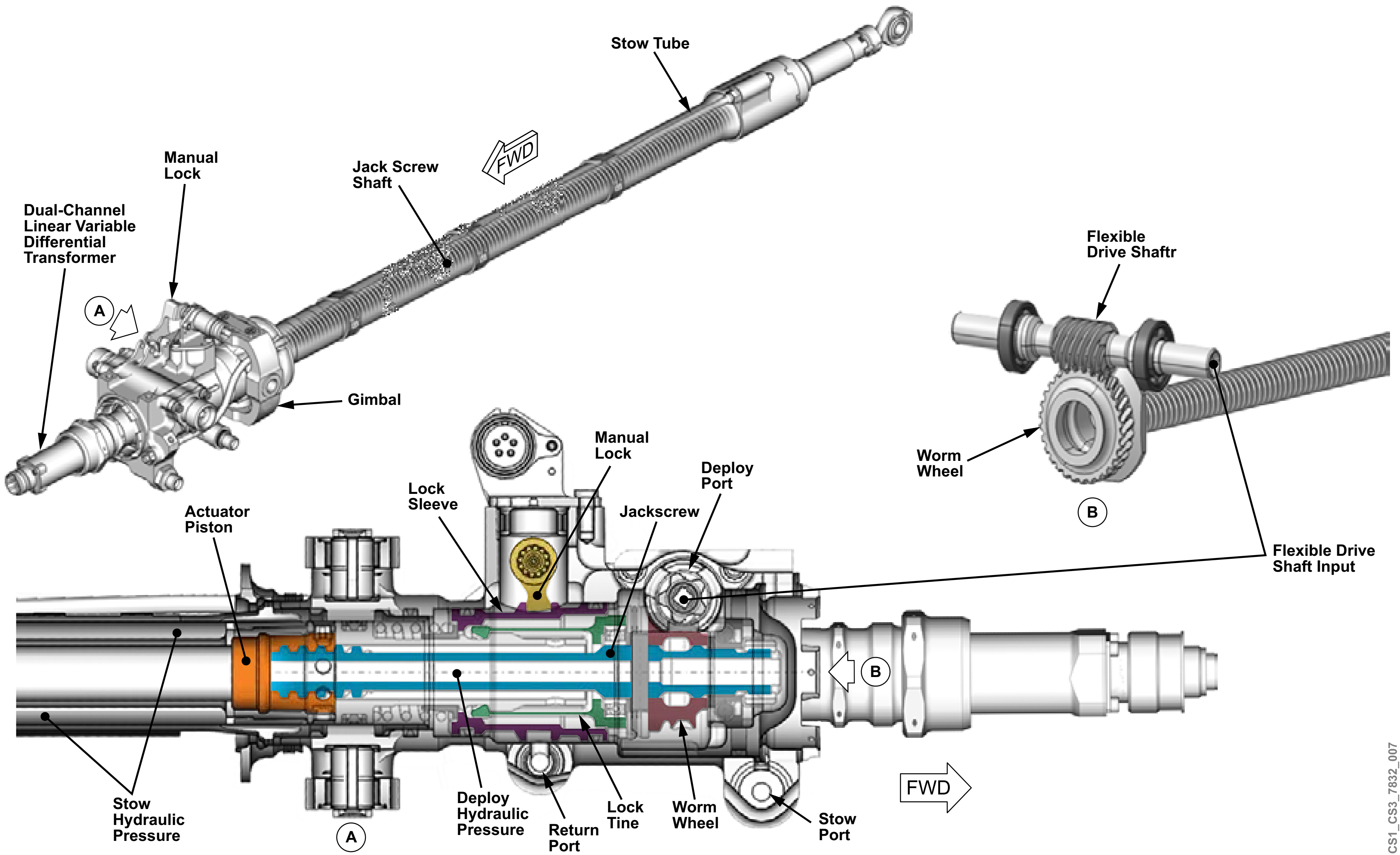


Figure 17: Locking Feedback Actuator Description

THROTTLE QUADRANT ASSEMBLY THRUST LEVER CONTROL

The gate plate, mounted on the TQA, has end stops at the IDLE, MAX and MAX REV positions. A gate pin on the thrust reverse link contacts the end stops to limit the movement of the thrust lever. An IDLE stop prevents the thrust lever from moving into the thrust reverse range.

Thrust reverser finger lifts on each thrust lever raise the gate pin clear of the IDLE stop. When the gate pin is clear of the IDLE stop, the thrust lever can be moved into the thrust reverse range. The thrust lever can be returned to the forward thrust range without using the finger lifts.

When the baulk solenoid energizes, it positions the baulk mechanism rocker arms to lock the thrust lever in the idle range until the thrust reverser is either fully deployed or fully stowed. If required, the interlock can be overridden with a force of approximately 25 lb.

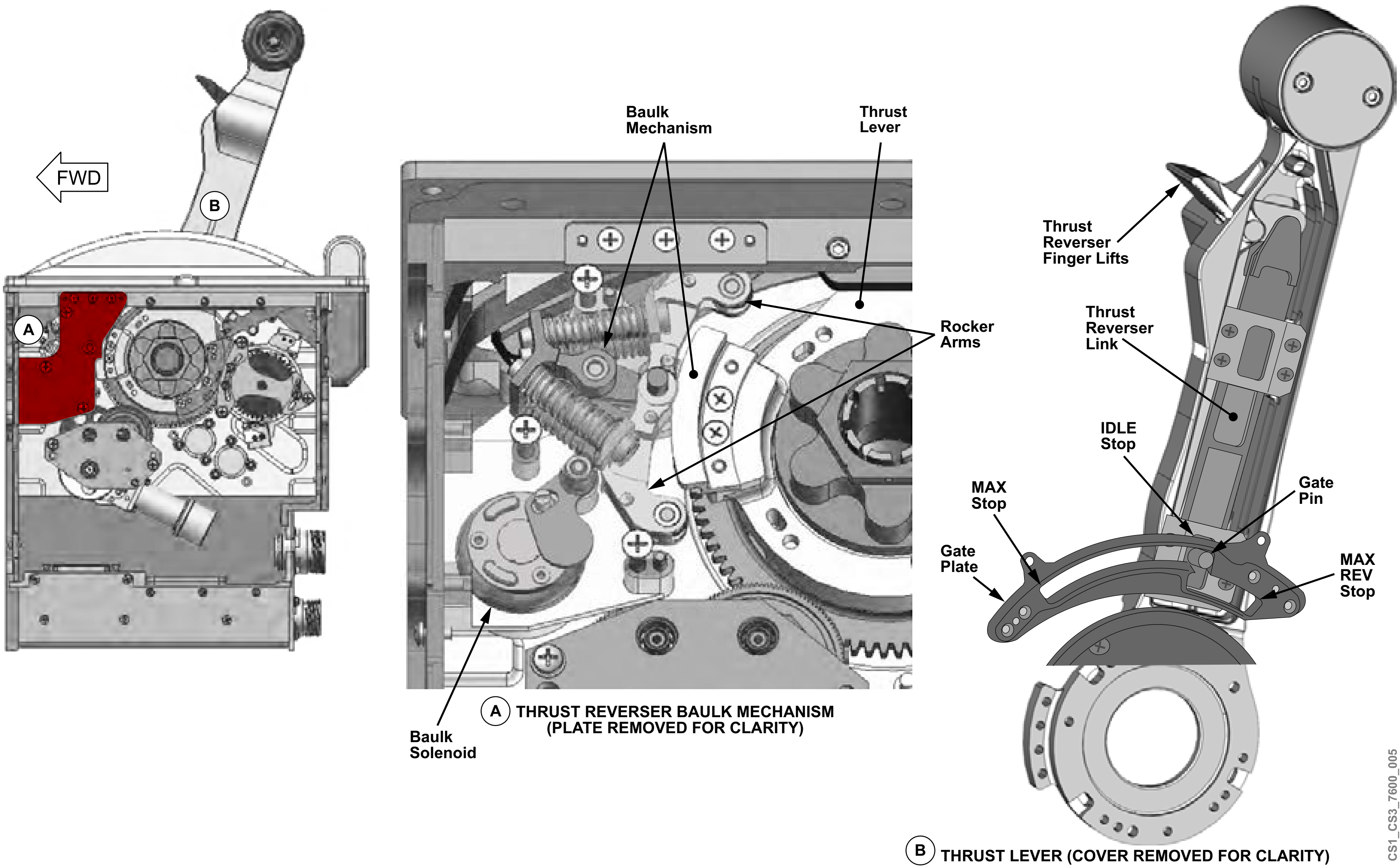


Figure 18: Throttle Quadrant Assembly Thrust Lever Control

CONTROLS AND INDICATIONS

CONTROLS

The thrust reversers (T/Rs) are controlled from the throttle quadrant assembly (TQA). The thrust levers are moved into the reverse thrust range using the finger lifts located on the thrust levers.

The thrust lever is used to control the amount of thrust required when the T/R has deployed.

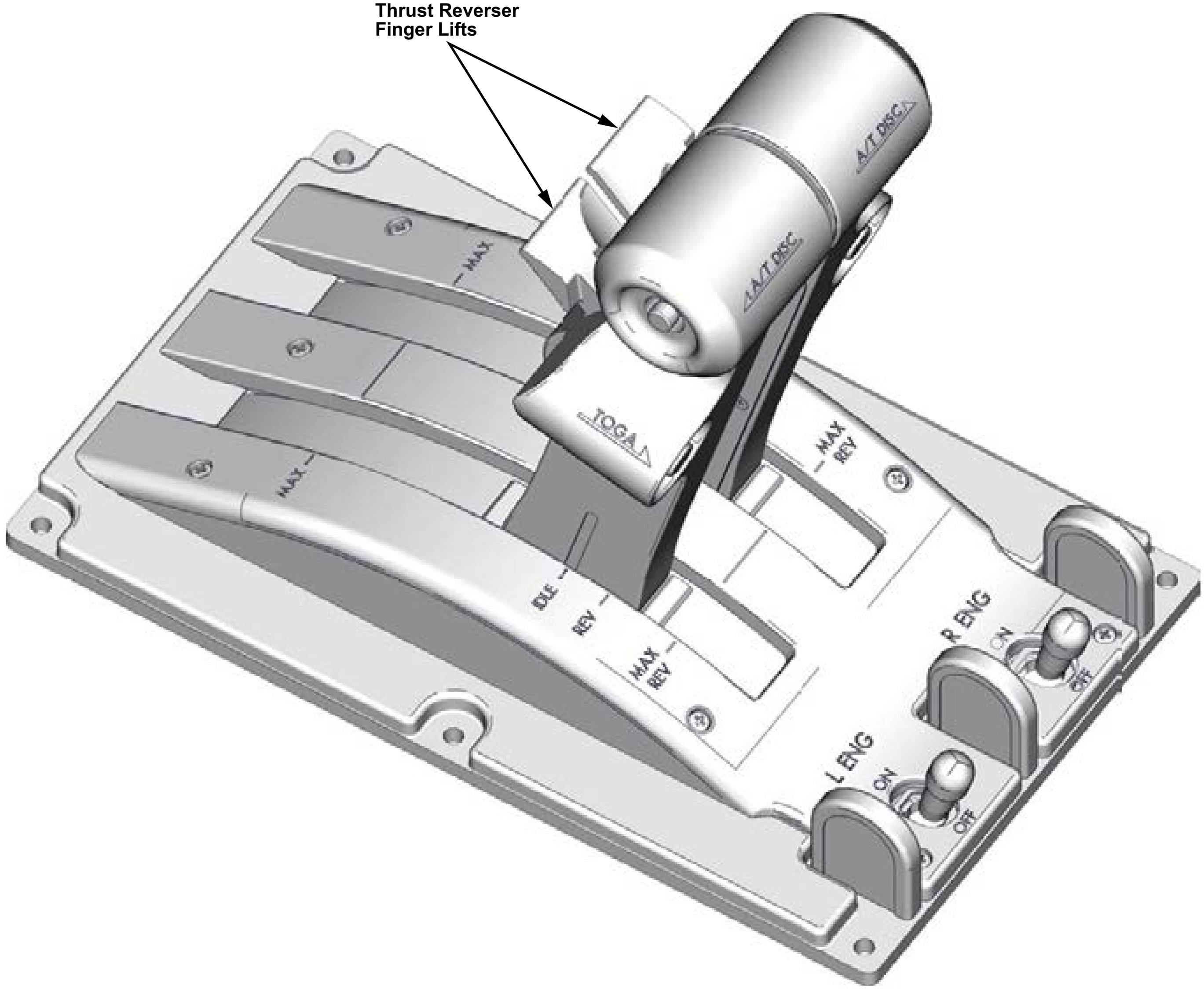


Figure 19: Thrust Reverser Control

INDICATIONS

The thrust reversers position is displayed on the engine indication and crew alerting system (EICAS) N1 gauge. The REV icon shows the status of the thrust reverser.

- A green REV icon appears in the N1 analog gauge when the thrust reversers (T/Rs) are fully deployed
- A white icon indicates the thrust reversers are in transition
- A amber REV icon indicates a T/R fault when the aircraft is on the ground, the thrust reverser is not manually inhibited, and any one of the following conditions occurs:
    - Inadvertent thrust reverser deployment
    - Thrust reverser translating sleeve position feedback failed on either translating sleeve
    - Thrust reverser is jammed
    - Thrust reverser inability to deploy
    - Thrust reverser inadvertently pressurized

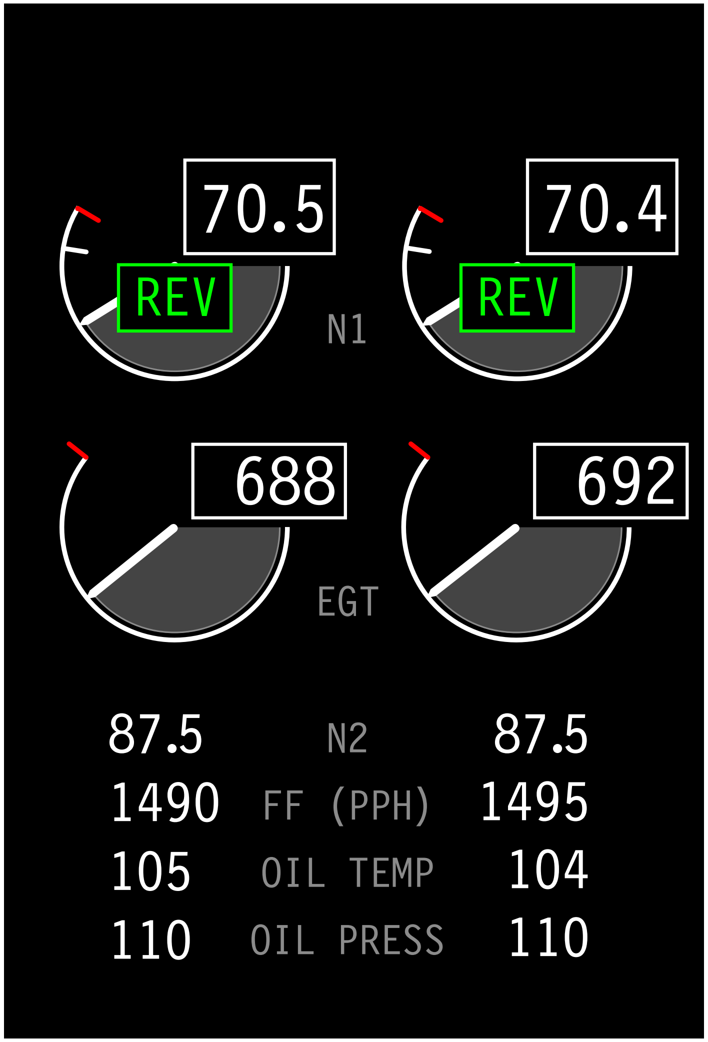


TABLE 1 EICAS FLAG

Symbol	Description
REV	Aircraft is on ground, the thrust reverser is not manually inhibited, and any one of the following conditions: ● Thrust reverser inadvertent deployment ● Thrust reverser translating sleeve position feedback failed on either translating sleeve ● Thrust reverser is jammed ● Thrust reverser inability to deploy ● Thrust reverser inadvertently pressurized
REV	On ground: thrust reverser in transition.
REV	On ground: thrust reverser is deployed.

NOTE

The thrust reverser icon has priority over the vibration icon on the ground.

CS1_CS3_7837_005

Figure 20: Thrust Reverser Indications

DETAILED DESCRIPTION

THRUST REVERSER LOCKED

When the thrust reverser (T/R) is stowed and the thrust levers are in forward thrust, the isolation control unit (ICU) and directional control unit (DCU) isolate hydraulic pressure from the thrust reverser system. The ICU isolation control valve control solenoid is de-energized. The directional control unit (DCU) pressure proximity sensors monitor the hydraulic pressure in the thrust reverser actuating system.

The track lock valve solenoids are also de-energized and the track lock units are in the locked position. The track lock proximity sensors monitor the track lock units status.

With the DCU directional control valve solenoid de-energized, the actuator is locked in the stowed position. The locking actuator proximity sensors monitor the lock mechanism position to ensure the thrust reverser is stowed and locked. The feedback actuator LVDTs provide position feedback as a secondary indication of the actuator position.

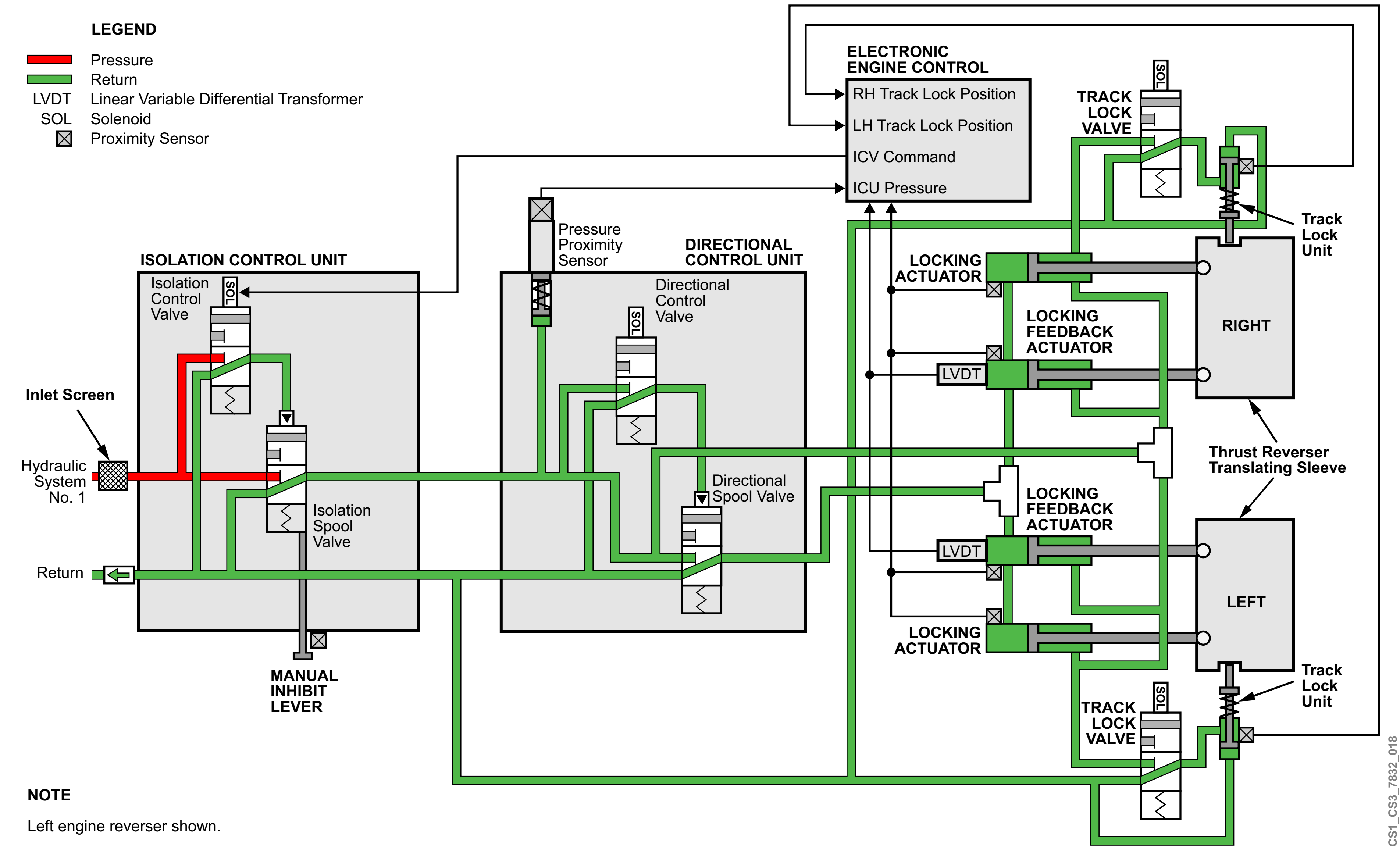


Figure 21: Thrust Reverser Locked

ISOLATION CONTROL UNIT ENERGIZED - ACTUATORS OVERSTOW

When reverse thrust is selected, the electronic engine control (EEC) energizes the ICU isolation valve control solenoid. The hydraulic pressure is directed through the isolation spool valve to the DCU. The dual pressure proximity sensor, installed on the DCU, signals the EEC that hydraulic pressure is available from the ICU.

Hydraulic pressure is supplied to the stow side of the locking feedback actuators and the locking actuators. The actuators over stow to release the locking mechanisms. The proximity sensors on the actuators indicate the locking mechanism is in the locked position.

Hydraulic pressure is also supplied to the track lock valves. The track lock units remain locked. The track lock unit proximity sensors send the locked status to the EEC.

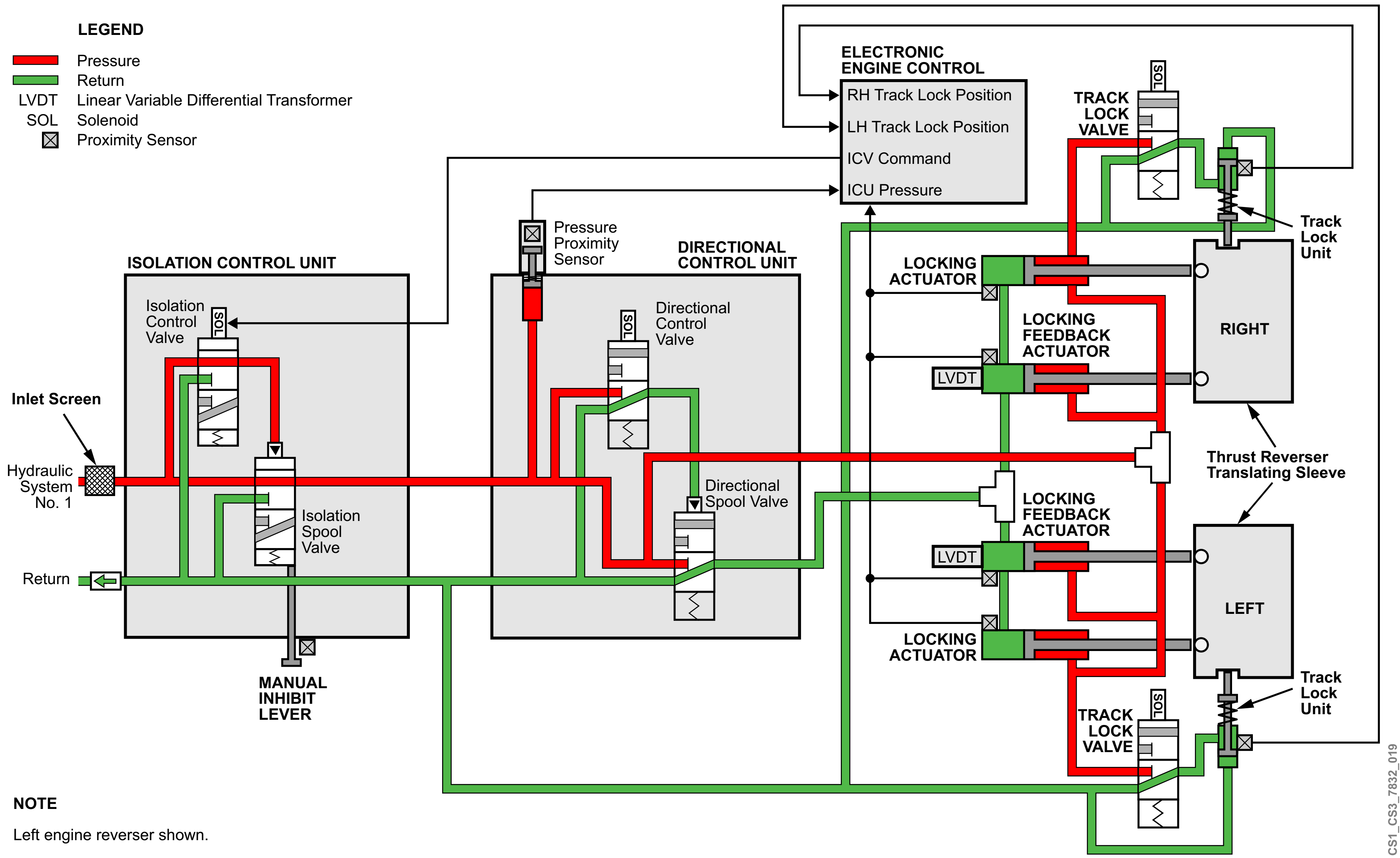
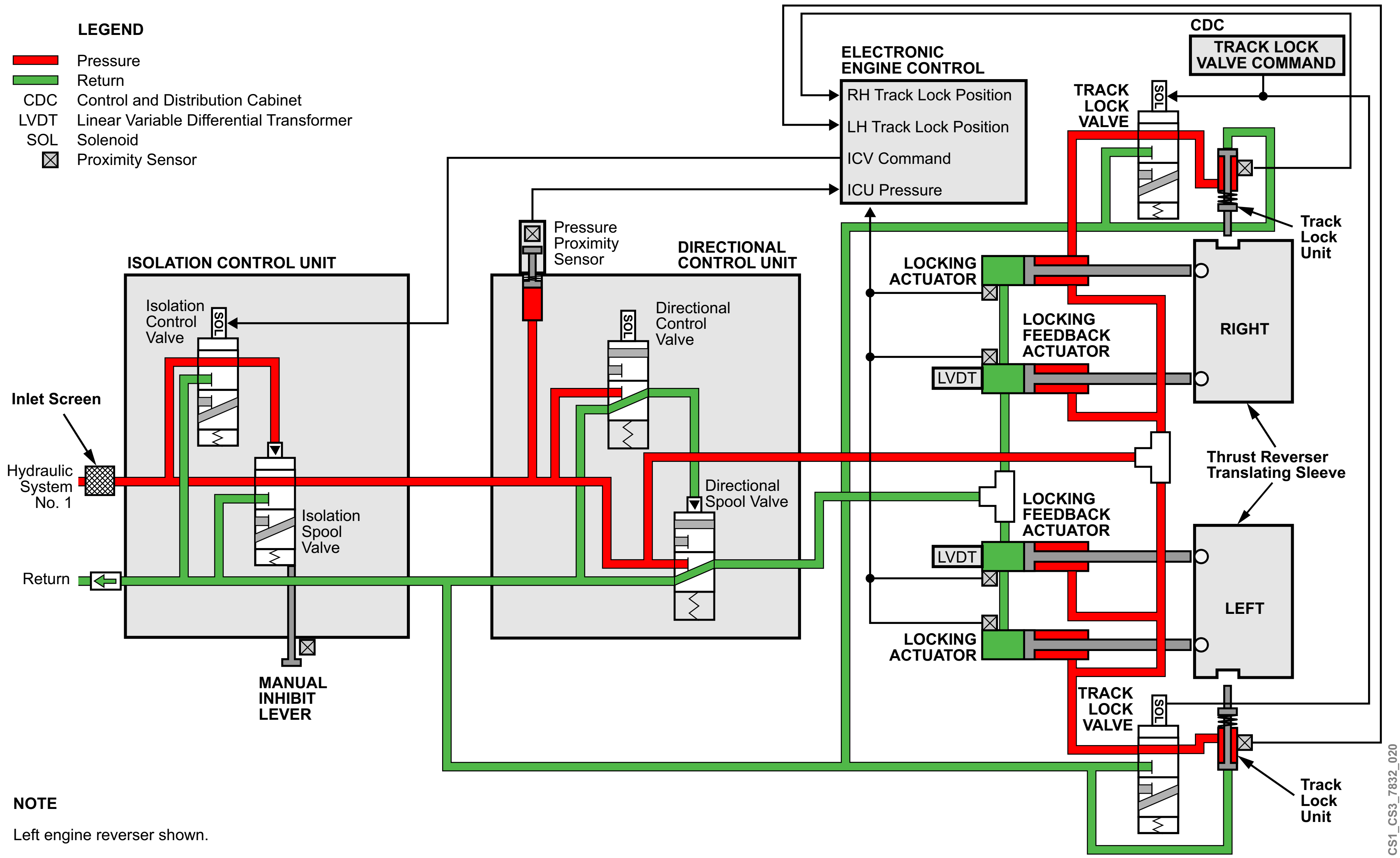


Figure 22: Isolation Control Unit Energized – Actuators Overstow

TRACK LOCK UNITS RELEASED

With the WOW logic satisfied, the control and distribution cabinet (CDC) energizes the track lock valve solenoids open both track lock valves. The hydraulic pressure is now supplied to the track lock unit and the track lock unit bolt retracts. The track lock unit proximity sensors signal the EEC indicating that the track lock unit bolts are clear of the translating sleeve and the thrust reverser is ready to deploy.



NOTE

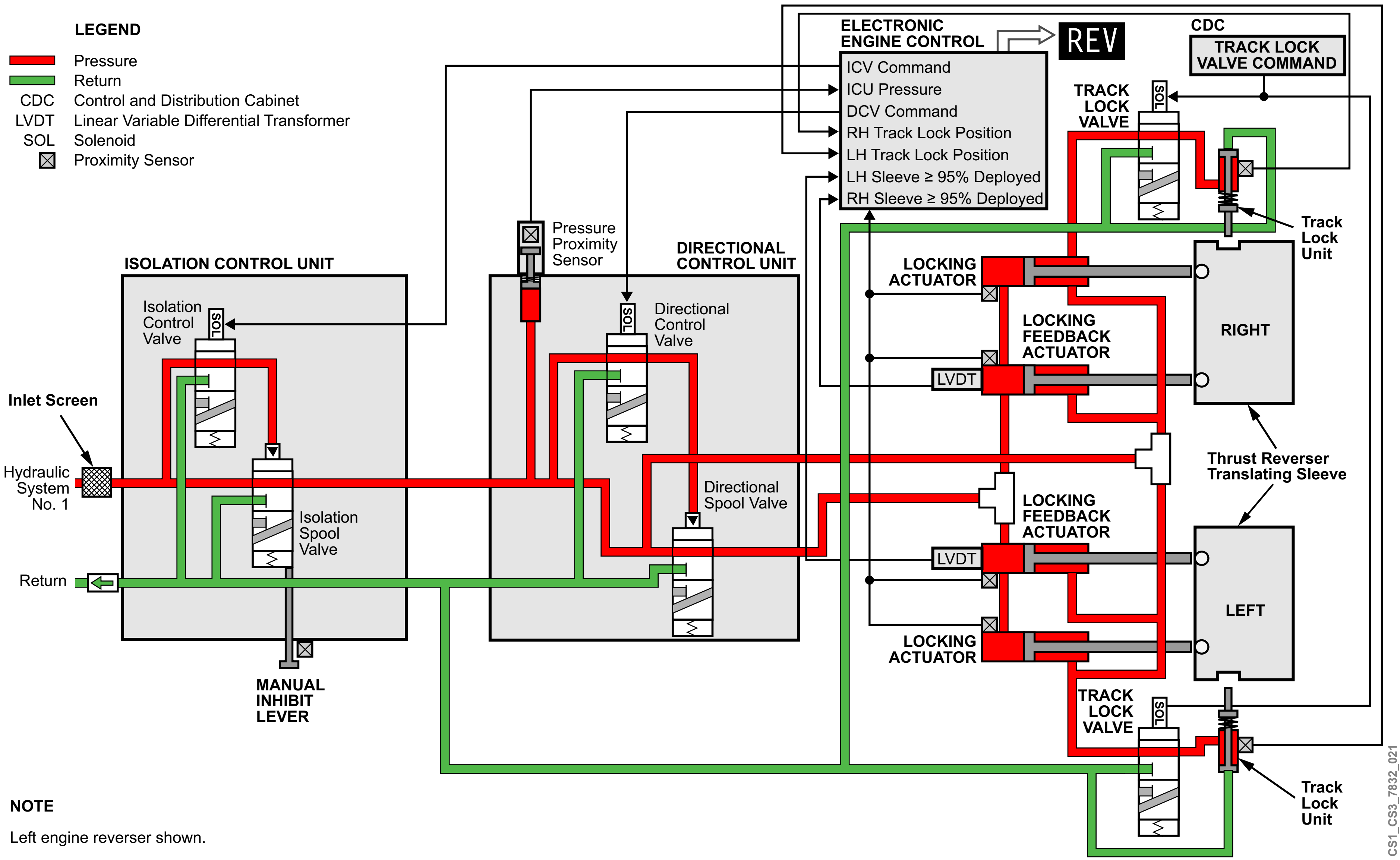
Left engine reverser shown.

Figure 23: Track Lock Units Released

DCU ENERGIZED

The EEC energizes the DCV directional control valve solenoid. This positions the directional spool valve to pressurize the actuators deploy side to completely unlock the actuators. The actuator proximity sensors and LVDTs provide the unlock status to the EEC and the translating sleeves start to deploy. The track lock valves solenoids remain energized during the deploy sequence.

A white REV icon is displayed on the EICAS N1 gauge.



NOTE

Left engine reverser shown.

Figure 24: DCU Energized

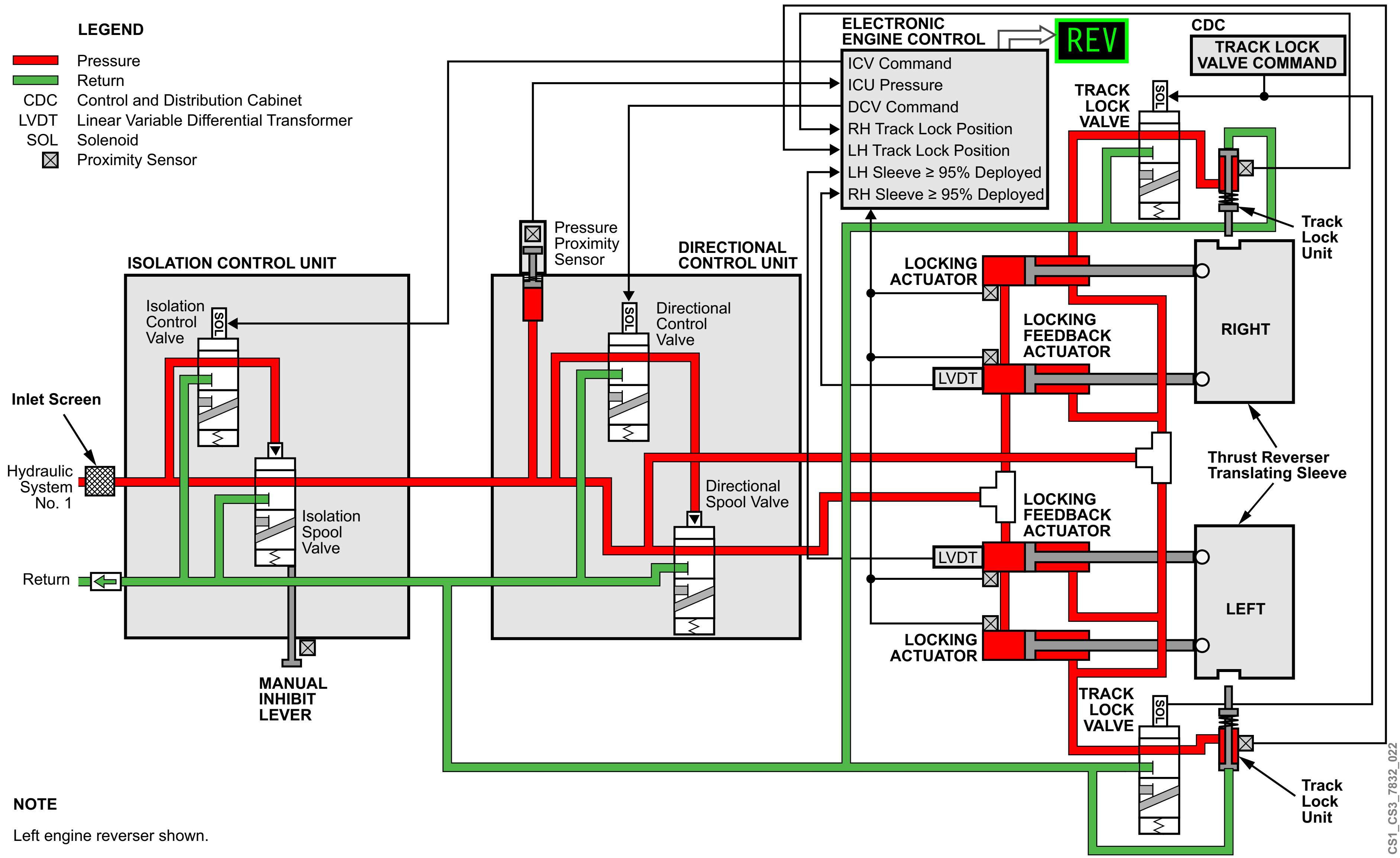
THRUST REVERSERS FULLY DEPLOYED

The translating sleeve position is sent to the EEC by the locking feedback actuator LVDTs. When the translating sleeves reach 85% of full travel, the thrust levers may be moved to the full reverse thrust.

When the translating sleeves reach 95% of full travel, the green REV icon is shown on the N1 gauge.

NOTE

Both sides of the actuators are pressurized, however the greater actuator piston area on the deploy side ensures the actuators deploy.



NOTE

Left engine reverser shown.

Figure 25: Thrust Reversers Fully Deployed

STOW COMMAND

When the throttle moves from reverse idle to forward idle, the EEC de-energizes the DCU control solenoid. The directional spool valve repositions and the deploy side of the actuators are connected to the return side of the hydraulic system. Since hydraulic pressure always remain on the stow side of the actuators, the translating sleeves stow.

The locking feedback actuator LVDTs feedback to the EEC changes the REV icon from green to white.

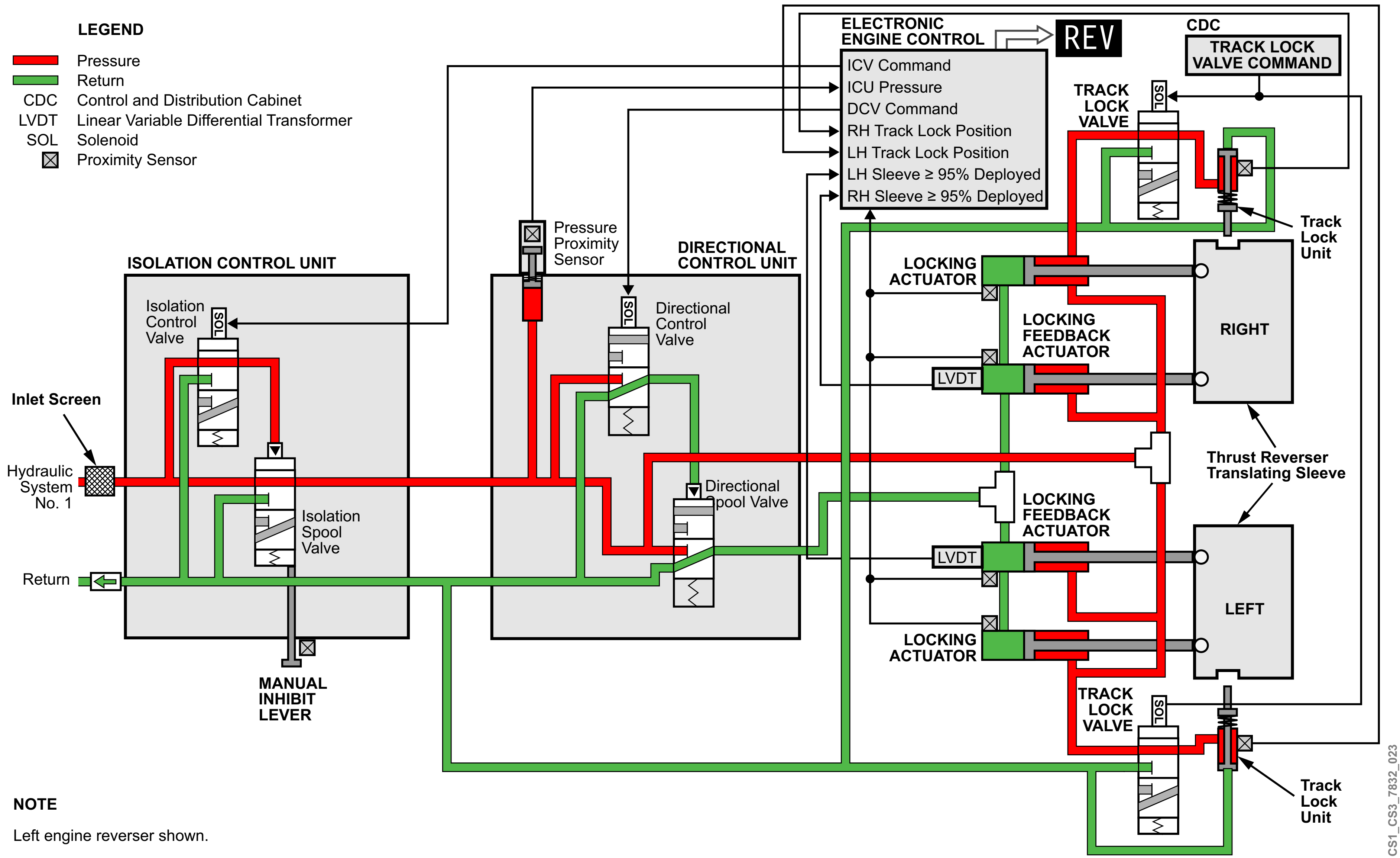
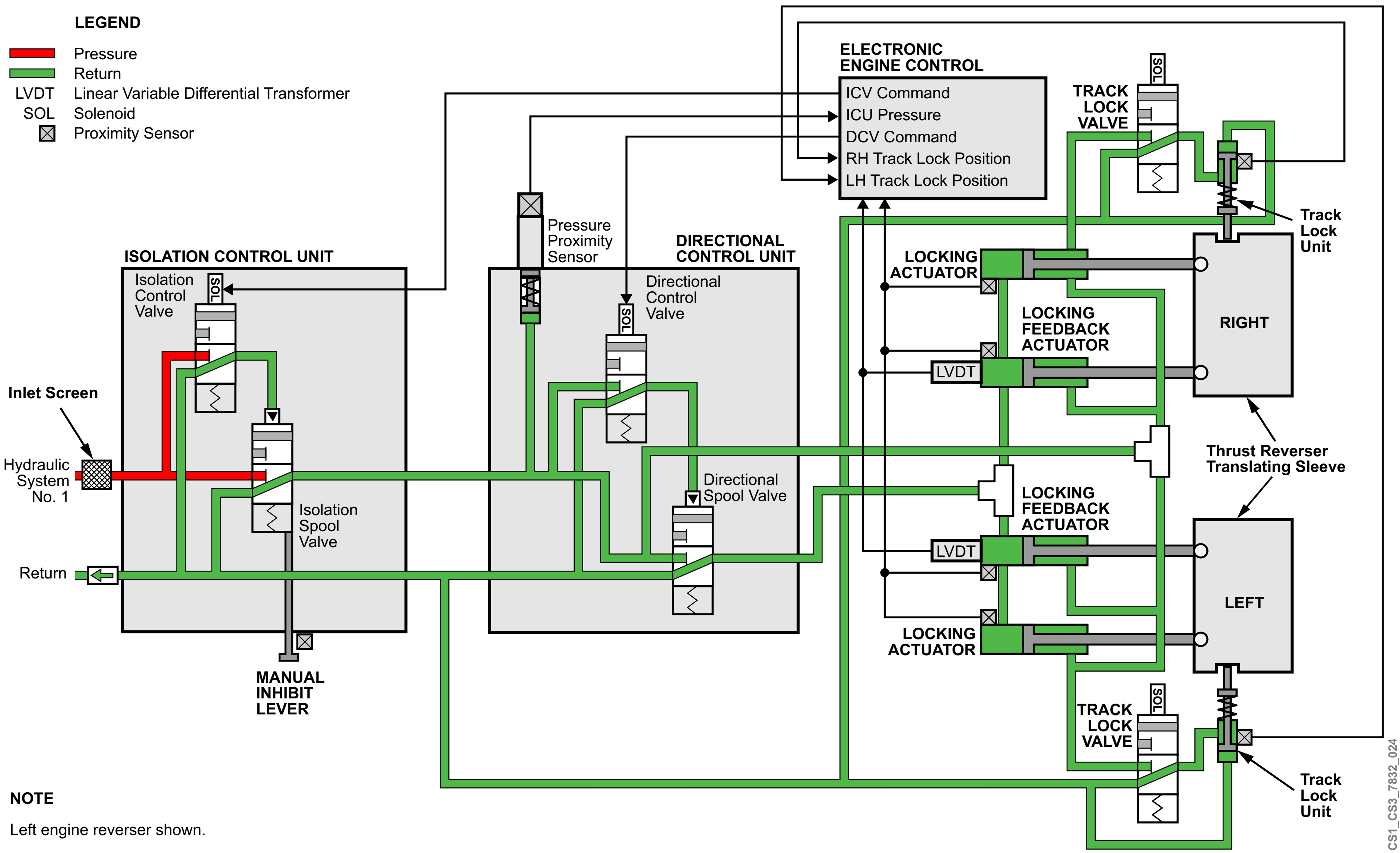


Figure 26: Stow Command

THRUST REVERSER STOWED

The track lock valve solenoids de-energize, and the track lock units relock. When the actuator proximity sensors indicate the actuators are in the stowed position, the EEC de-energize the track lock valve solenoids. The spring-loaded track lock unit bolts extend, locking the translating sleeves. The track lock unit proximity sensors indicate the track lock unit lock status to the EEC.

With the confirmation of the track lock units in the locked position, the EEC de-energizes the ICU control solenoid, isolating the trust reverser actuating system from the hydraulic system. The thrust reverser (T/R) hydraulic components are connected to the return line. The DCU pressure proximity sensors confirm the pressure is not available to the thrust reverser actuating system (TRAS). The REV indication is removed from the N1 gauge.



NOTE

Left engine reverser shown.

Figure 27: Thrust Reverser Stowed

THRUST REVERSER ACTUATING SYSTEM CONTROL ELECTRICAL SCHEMATIC

The thrust levers in the throttle quadrant assembly (TQA) provide the command signal to the EEC to control the thrust reverser actuating system (TRAS). When the thrust lever is moved to the REV IDLE position, the stow/deploy switches are made. The stow/deploy switches send a discrete to the thrust reverser lock solid-state power controller (SSPC) in control and distribution cabinet 1 (CDC 1).

The baulk solenoid that prevents the thrust lever from moving towards the MAX REV range is powered by the TQA L Reverser Lock SSPC in CDC 3. Once the thrust reversers are deployed beyond 85%, the EEC provides a ground to energize the baulk solenoid.

The EEC controls the ICU and DCU control solenoids. The ICU solenoid receives power from DC BUS 1 from CDC 1 on channel A and DC BUS 2 from CDC 2 on channel B. The combinational logic in the CDCs enables them when the stow/deploy switches are made, and one of the following conditions is true:

- Weight-on-wheels (WOW) signal from the landing gear and steering control unit (LGSCU)
- Wheel spin signal from the brake data concentrator unit (BDCU)

The WOW signal from the LGSCU is hardwired into the CDCs, and sent through the data concentrator unit module cabinets (DMCs) to the EEC.

The EEC provides the ground to energize the ICV solenoid. The DCU solenoid receives both power and a ground from the EEC.

A dual channel pressure sensor provides the EEC with a signal when hydraulic pressure is flowing through the ICU and into the DCU.

Once the ICU and DCU control solenoids become energized, the control valves are actuated which allows hydraulic pressure to flow to the actuators and track lock valves.

The track lock valves are controlled by solenoids. These solenoids are energized by DC BUS 1 from CDC 1 when the stow/deploy switches are made and the aircraft is on the ground.

The track lock units have a proximity sensor to send a signal to the EEC when the track lock units are unlocked.

The EEC controls the TQA baulk solenoid based on LVDTs and the stow and lock proximity sensors feedback received from the actuators. The EEC also monitors the DCU status. The LVDTs are dual-channel with one output wired to each channel of the EEC.

Once the T/Rs are deployed beyond 85%, the EEC energizes the baulk solenoid. This allows the thrust lever to move into MAX REV range. When the thrust reversers deploy beyond 95%, the EEC sends a signal to the EICAS for indicating the green REV icon.

Maximum reverse thrust is determined by the EEC. The EEC makes this determination using thrust lever angle (TLA). If the TLA was above 23° and then retarded to idle, the EEC determines that a rejected takeoff has occurred and sets the maximum reverse thrust level at 87.4% N1. During rollout after landing, the maximum reverse thrust is 80% N1. If the airspeed is below 55 kt and reverse thrust is still applied, the maximum reverse thrust N1 is reduced to reverse idle N1.

If either thrust lever is moved in the reverse thrust range while the aircraft is in flight, a THROTTLE IN REVERSE caution message is displayed.

Inadvertent operation of the thrust reverser (T/R) is detected by the EEC through monitoring the T/R position feedback and the lock status. If the T/R inadvertently deploys, the EEC commands the engine power to idle.

An L(R) REVERSER FAIL caution message is displayed if the EEC detects faults that prevent system operation. On the ground, an amber REV icon is also displayed in the N1 indicator.

If both locks on a translating sleeve indicate unlocked, a track lock indicates unlocked, or the TRAS is inadvertently pressurized, then a L(R) REVERSER UNLOCK caution message is displayed. The T/R may still be usable for landing.

A dual-channel proximity sensor monitors the manual inhibit lever on the ICU. If the thrust reverser (T/R) is locked out, all other T/R CAS messages are inhibited and an L(R) REVERSER INHIBIT status message is displayed.

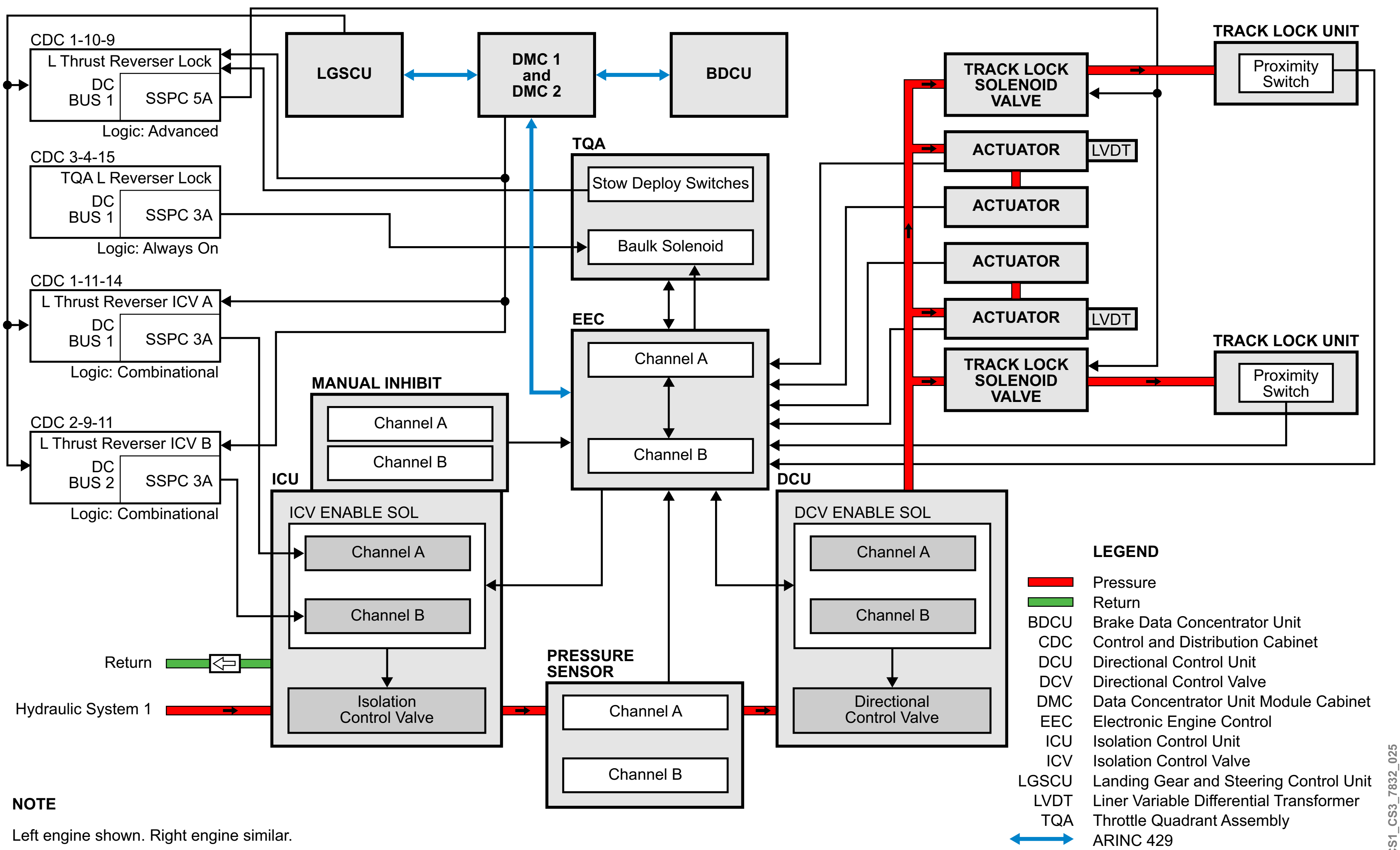


Figure 28: Thrust Reverser Actuating System Control Electrical Schematic

THRUST REVERSER ACTUATING SYSTEM CONTROL LOGIC

Isolation Control Valve Logic

The ICV logic requires an aircraft on the ground signal supplied by landing gear and steering control unit (LGSCU) 1 or LGSCU2 WOW signal, or an aircraft touchdown signal supplied by any 2 of the 4 wheel speed units (WSU) indicating a wheel speed of 45 kt.

The ICV energizes when:

- Thrust reverse is selected
- No manual inhibit
- No thrust reverser fault detected
- Engine running

The ICV also energizes on the ground when the engine is not running and the EEC thrust reverser cycling test is carried out through the onboard maintenance system (OMS).

Track Lock Solenoid Control Logic

Once the aircraft is on the ground, the landing gear and steering control unit (LGSCU) 1 or LGSCU 2 supply a WOW signal to the track lock logic. When the thrust levers are selected to the reverse thrust range beyond the -1.85° and both left stow/deploy switches 1 and 2 are actuated by the thrust lever, the track lock solenoid is energized

When the thrust reverser is stowed, the track lock solenoids are energized for 11 seconds to provide enough time for the translating sleeve to stow.

Directional Control Valve Logic

The DCU logic requires an aircraft on the ground signal supplied by landing gear and steering control unit (LGSCU) 1 or LGSCU2 WOW signal or an aircraft touchdown signal supplied by any left WSU and any right WSU indicating a wheel speed of 45 kt.

The DCV energizes when:

- Thrust reverse is selected
- No manual inhibit
- No thrust reverser fault detected
- Track lock units unlocked
- Engine running

The DCV also energizes on the ground when the engine is not running and the EEC thrust reverser cycling test is carried out through the onboard maintenance system (OMS).

Baulk Solenoid Logic

When thrust reverse is selected, each thrust lever is locked in the reverse idle position by the baulk mechanism. The thrust lever is locked until the thrust reverser translating sleeves have deployed far enough to safely increase thrust. When the directional control valve (DCV) has energized and the translating sleeves have deployed beyond 85%, the baulk solenoid energizes and releases the baulk mechanism. The thrust lever can be moved beyond the reverse idle position.

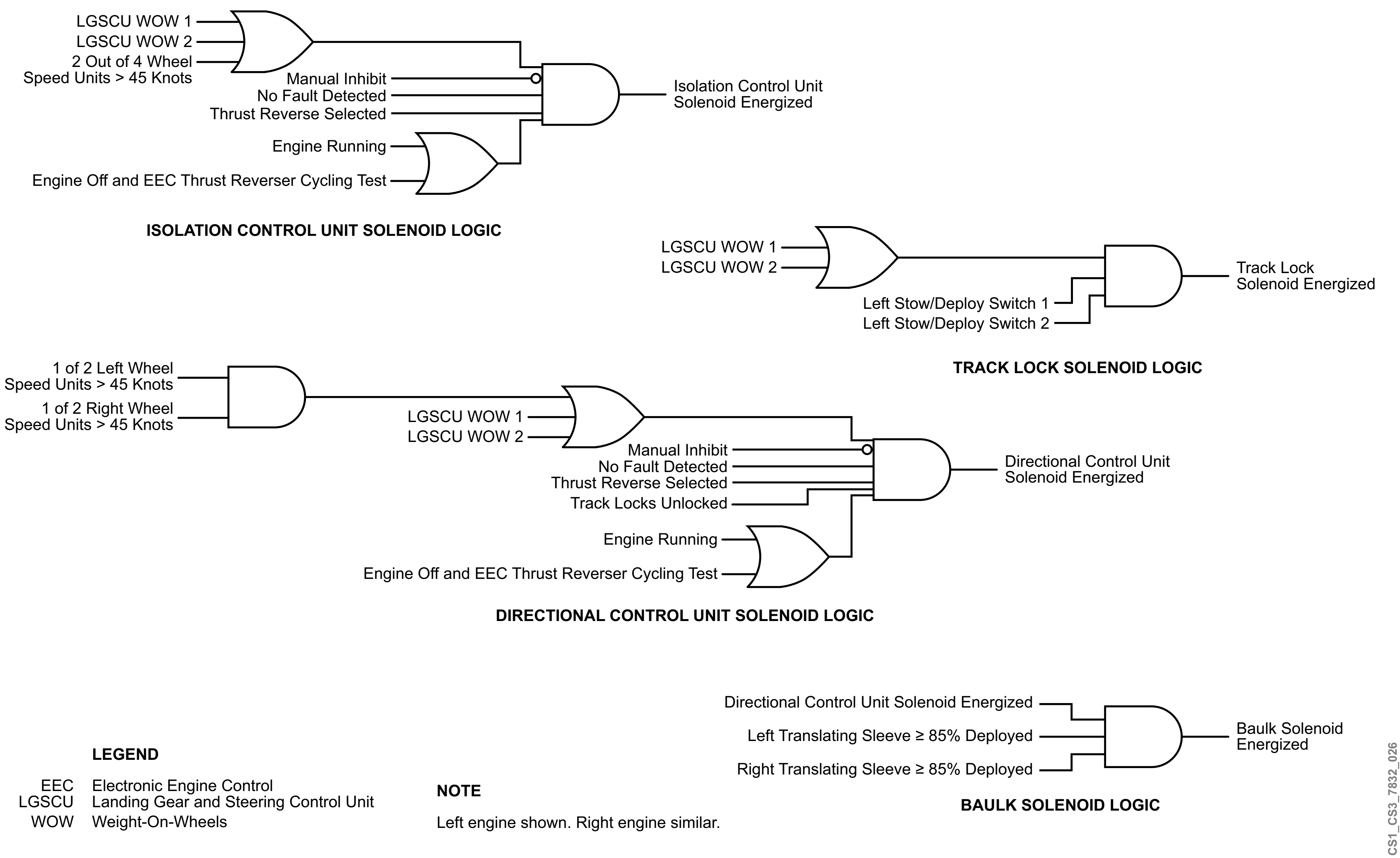


Figure 29: Thrust Reverser Actuating System Logic

MONITORING AND TESTS

The following page provides the crew alerting system (CAS) and INFO messages for the thrust reverser system.

CAS MESSAGES

Table 1: CAUTION Messages

MESSAGE	LOGIC
L REVERSER FAIL	Left thrust reverser failed or not available. Icon is only shown if T/R has been commanded.
R REVERSER FAIL	Right thrust reverser failed or not available. Icon is only shown if T/R has been commanded.
L REVERSER UNLOCK	Multiple left reverser locks unlocked.
R REVERSER UNLOCK	Multiple right reverser locks unlocked.
THROTTLE IN REVERSE	Left or right thrust reverser selected in flight.

Table 2: ADVISORY Messages

MESSAGE	LOGIC
L ENGINE FAULT	Loss of redundant non-critical function for the left engine.
R ENGINE FAULT	Loss of redundant non-critical function for the right engine.

Table 3: STATUS Messages

MESSAGE	LOGIC
L REVERSER INHIBIT	Left T/R manually inhibited as per maintenance procedure (crew awareness).
R REVERSER INHIBIT	Right T/R manually inhibited as per maintenance procedure (crew awareness).

PRACTICAL ASPECTS

LOCKOUT THE THRUST REVERSERS FOR DISPATCH

The T/Rs can be locked out for dispatch per the Aircraft Minimum Equipment List (MEL).

The following is a synopsis of the Aircraft Maintenance Publication (AMP) procedure. Always consult the AMP before beginning any maintenance on the aircraft.

There are two steps to deactivate the thrust reversers:

- Deactivate the isolation control unit (ICU)
- Install a T/R lockout pin on the latch beam assembly

ICU DEACTIVATION

The ICU is deactivated by using an inhibit lever located on the ICU, and installing a safety pin to maintain the lever in the inhibited position. The safety pin is stowed on the ICU when not required.

Access to the ICU is through access panels on each side of the pylon.

WARNING

THE THRUST REVERSER ACTUATION SYSTEM SHOULD BE DEACTIVATED USING THE ICU INHIBIT LEVER, BEFORE PERFORMING ANY MAINTENANCE. THE ACCIDENTAL OPERATION OF THE THRUST REVERSER CAN CAUSE INJURIES TO PERSONNEL, AND DAMAGE TO EQUIPMENT.

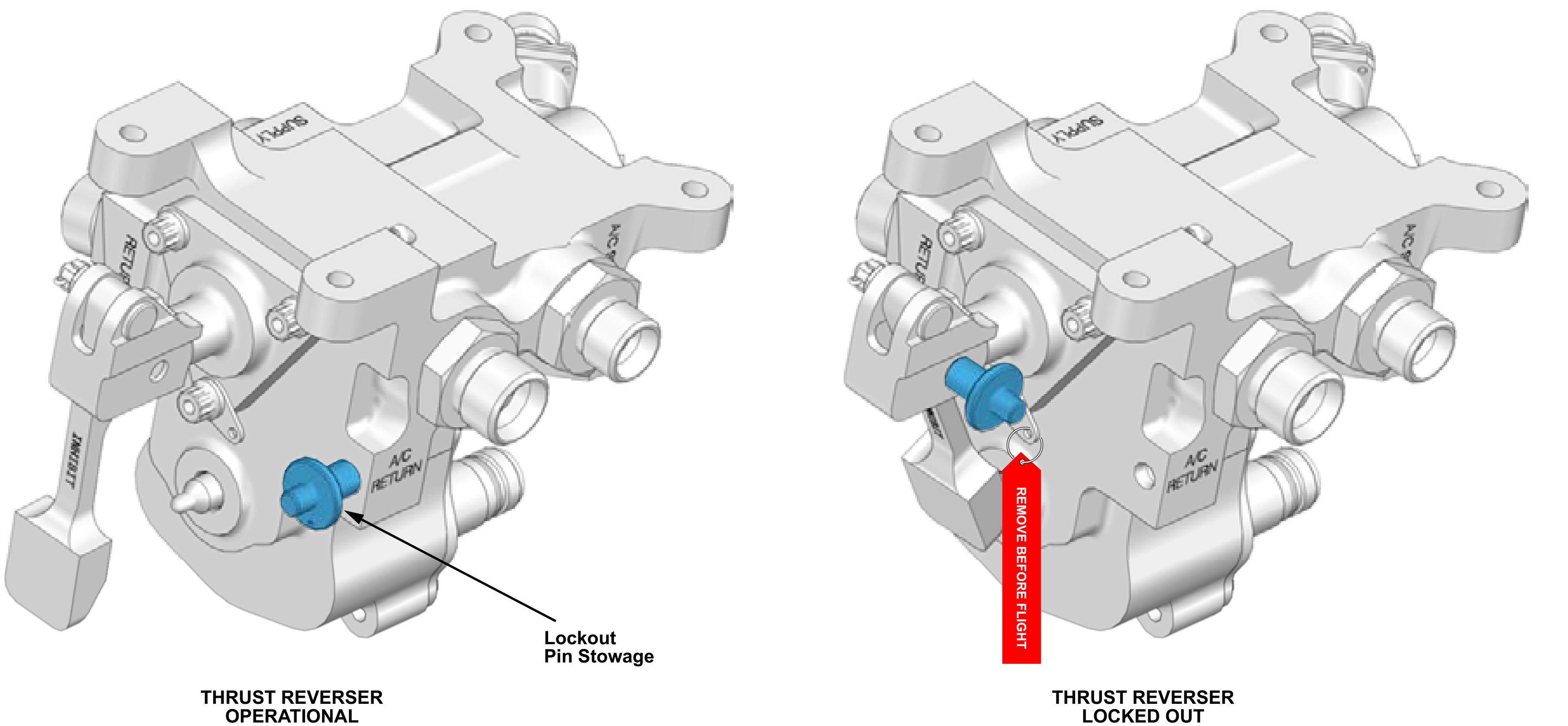
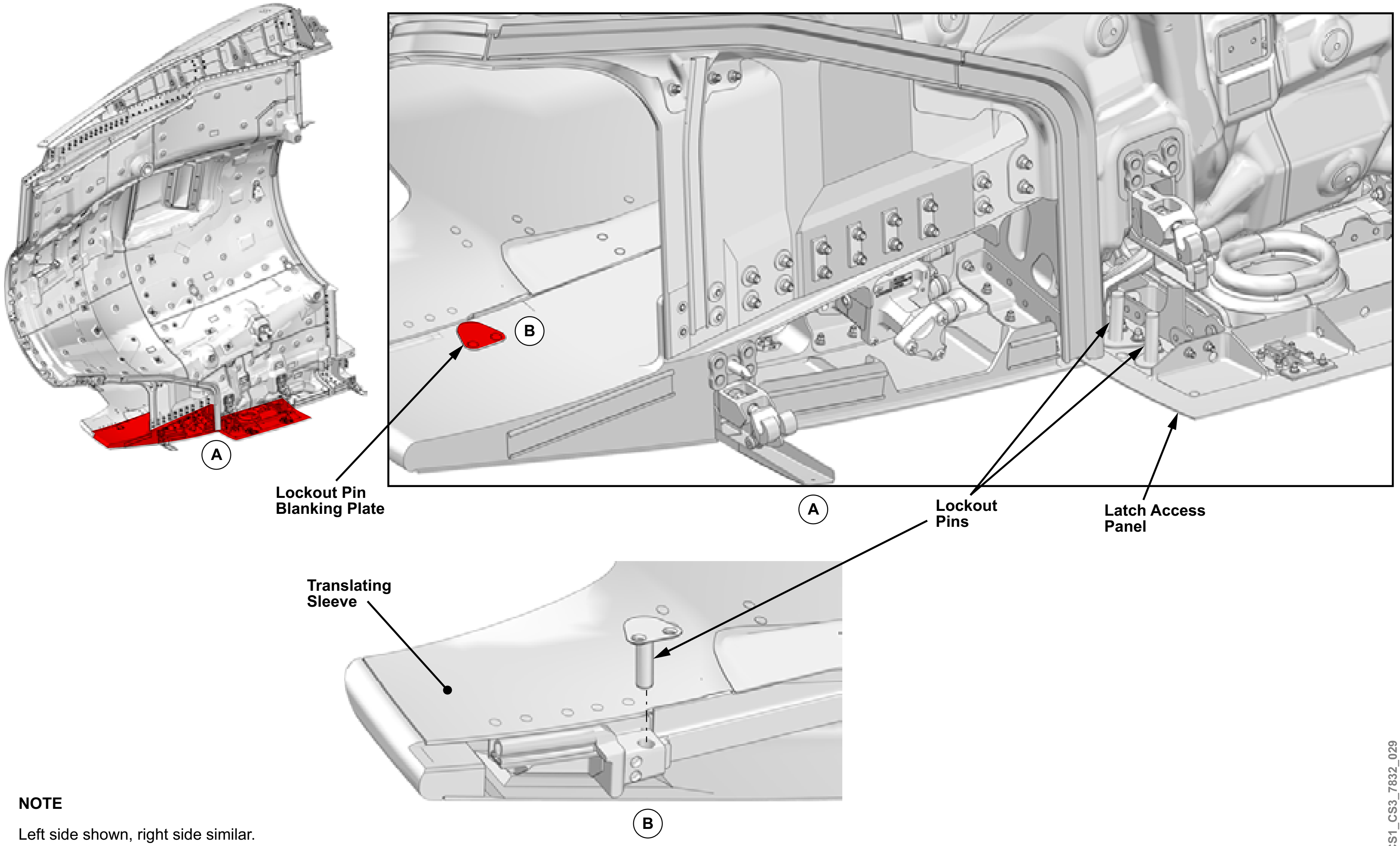


Figure 30: Isolation Control Unit Deactivation

Thrust Reverser Latch Beam Lockout Pin

After locking out the ICU, lockout pins are installed on the latch beam assemblies to prevent the T/Rs from deploying.

The lockout pins are stored on the thrust reverser door latch access panel. The pins are removed from their storage location and installed on the latch beam. The blanking plates that cover the lockout pin holes are stored on the latch access panel when the lockout pins are installed.



NOTE

Left side shown, right side similar.

Figure 31: Thrust Reverser Latch Beam Lockout Pins

MANUALLY OPERATE THE TRANSLATING SLEEVES FOR MAINTENANCE

The locking feedback actuators, locking actuators, and the track lock unit must be unlocked before manually operating the translating sleeves.

Manually Deploy the Translating Sleeve

Track Lock Unit Unlocking

To unlock the track lock unit:

- Move the manual lock from the lock position to the unlock position
- Install the unlock pin in the manual lockout hole.

Locking Feedback Actuator and Locking Actuator Unlocking

To unlock the locking feedback actuator and locking actuator:

- Move the manual lock on the locking feedback actuator to the unlock position and install the unlock pin
- Move the manual handle on the locking actuator to the unlock position and install the unlock pin

Translating Sleeve Deployment

Manually deploy the translating sleeve as follows:

- Remove the cover and put the square drive tool in the manual drive unit
- Turn the manual drive unit to extend the translating sleeve to the position necessary to do the maintenance

NOTE

A maximum speed of 120 rpm is permitted for the manual drive unit.

Manually Stow the Translating Sleeve

Stow the Translating Sleeve

Manually stow the translating sleeve as follows:

- Put the square drive tool in the manual drive unit
- Turn the manual drive unit to stow the translating sleeve
- Put the cover on the manual drive unit

Track Lock Unit Locking

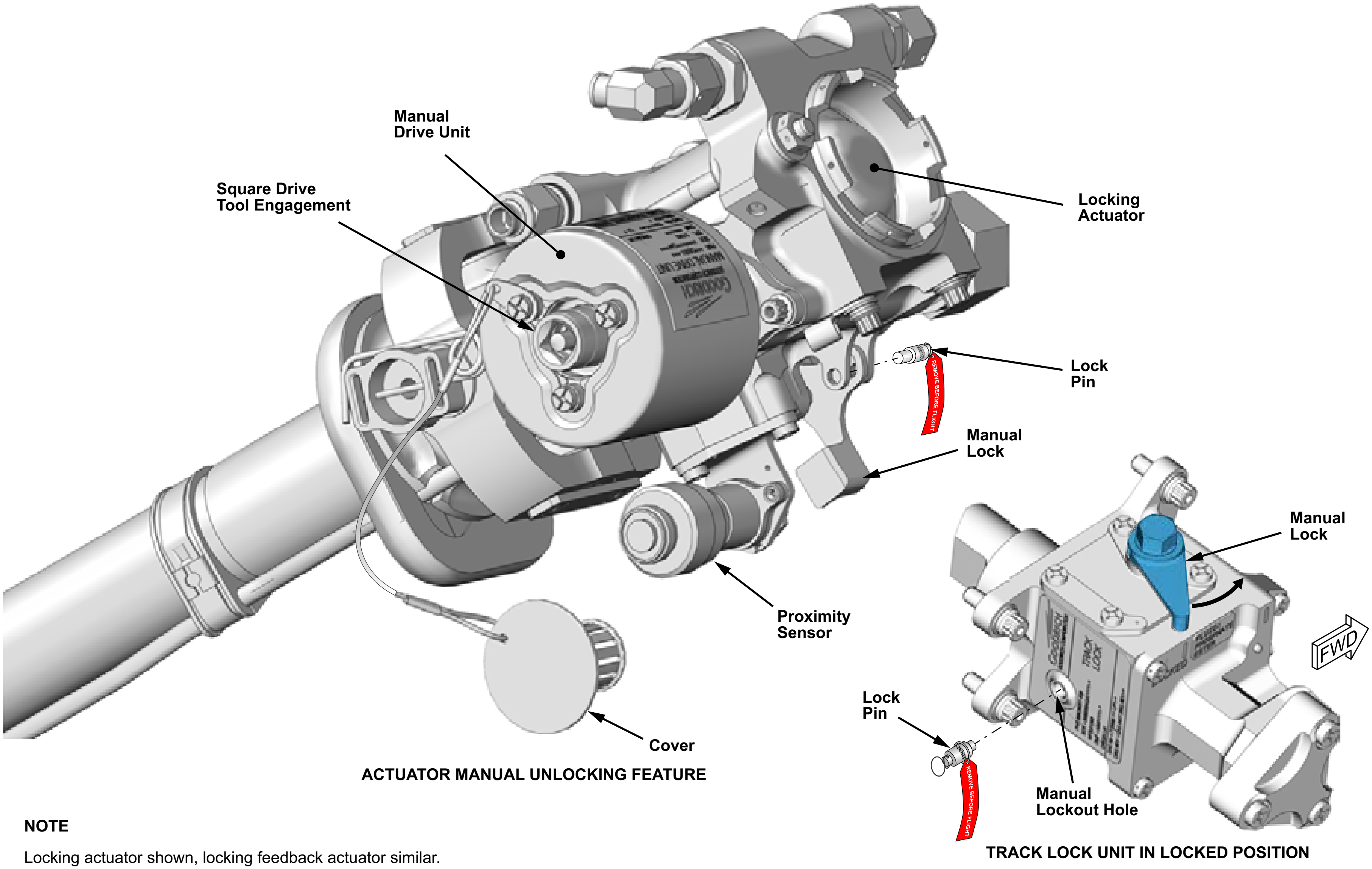
To lock the track lock unit:

- Remove the unlock pin from the manual lockout hole
- Move the manual lock from the unlock position to the lock position

Locking Feedback Actuator and Locking Actuator Locking

To lock the locking feedback actuator and locking actuator:

- Remove the unlock pin on the locking feedback actuator and move the manual lock back to the lock position
- Remove the unlock pin on the locking actuator and move the manual lock back to the lock position



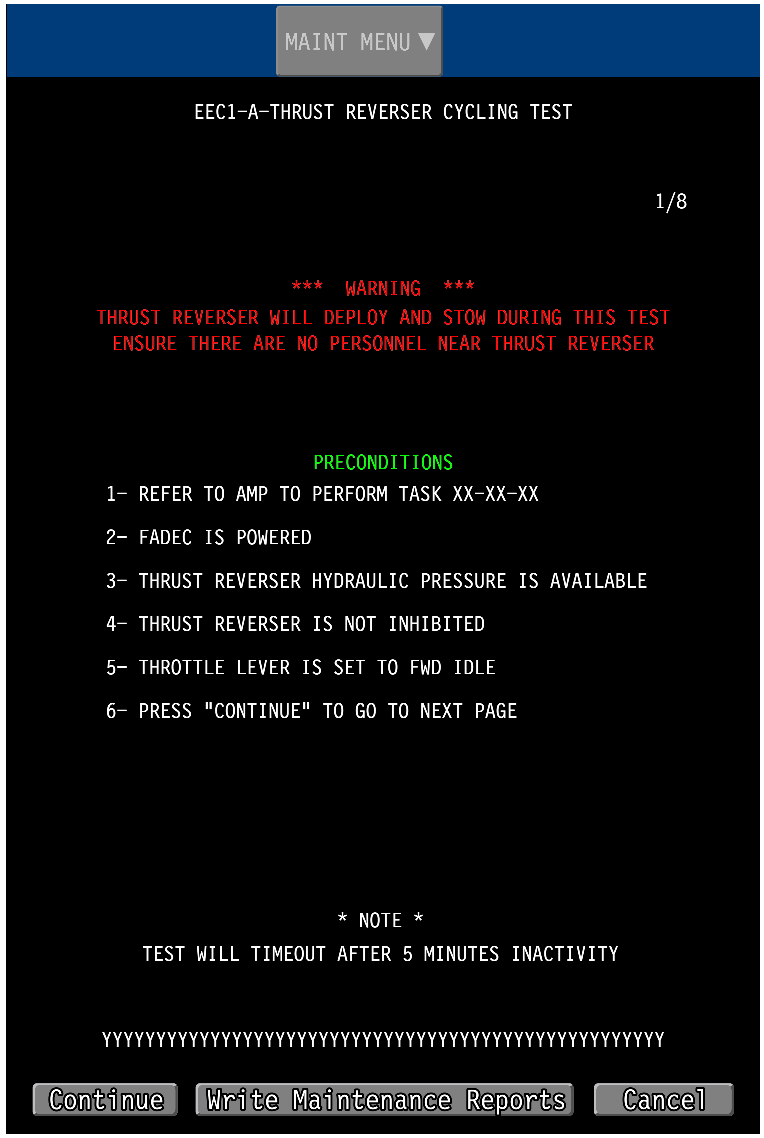
NOTE

Locking actuator shown, locking feedback actuator similar.

Figure 32: Manually Deploy the Thrust Reverser Sleeves

THRUST REVERSER CYCLING TEST

The thrust reverser cycling test is used to detect any operational faults of the thrust reverser system including the electrical interfaces, mechanical linkage, and translating sleeve. This is done without an engine start.



CS1_CS3_7321_026

Figure 33: Thrust Reverser Cycling Test

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ATA 79 - Oil

BD-500-1A10
BD-500-1A11

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OIL - CHAPTER BREAKDOWN

79-11 STORAGE

GENERAL DESCRIPTION

A 27.3 L (28.8 qt) aluminum oil tank stores the oil and supplies it to the engine. The tank is wrapped in a stainless steel heatshield, held together by springs.

The oil tank has an internal swirl-type deaerator that removes air from the returning oil. A pressure-regulating valve (PRV) maintains 12 psi head pressure in the tank. Excess pressure is vented to the deoiler in the main gearbox.

The oil level can be monitored using a sight glass. A fill port has a hinged cap for oil servicing. The fill port strainer prevents foreign objects from entering the tank during servicing. In the event that the filler cap is incorrectly installed, a flapper valve prevents rapid loss of oil. A scupper drain collects oil spilt during servicing, and sends it to the drain mast.

Oil can be drained through a drain plug in the bottom of the tank.

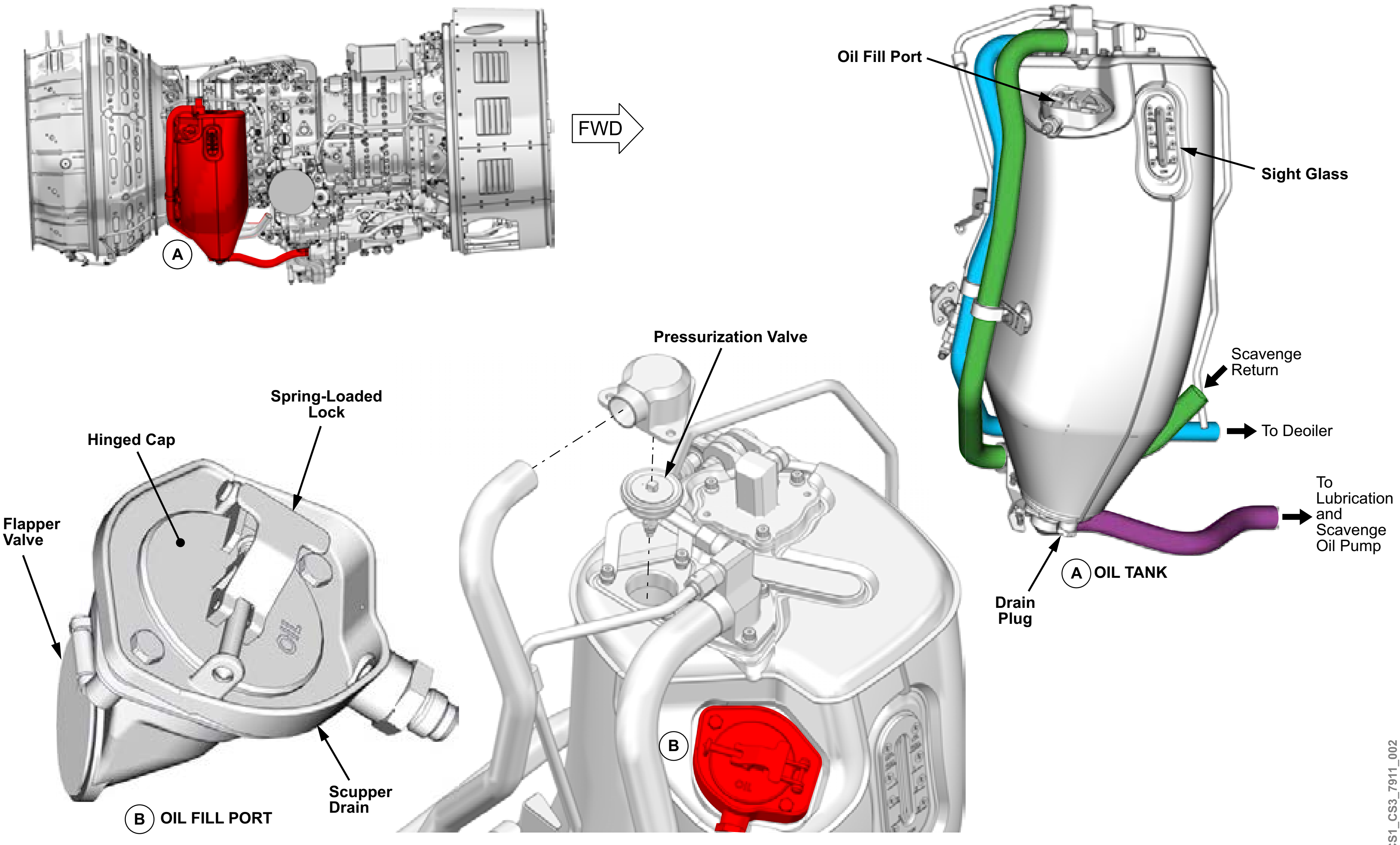


Figure 1: Oil Storage

PRACTICAL ASPECTS

OIL SERVICING

The oil level can be checked and serviced by opening an access door on the right side of the core cowl fixed structure.

The tank is equipped with a flapper valve to ensure the oil is not lost, should the cap remain open during flight.

The sight glass is marked in both liters and quarts, and gauge markings indicate how much oil may be added.

Oil quantities are as follows:

- Fully serviced 21.8 L (23 qts)
- Minimum 9.3 L (9.8 qts)

If the oil level is checked between 5 minutes and 60 minutes after engine shutdown, add oil to bring the oil level to the FULL mark.

If the oil level is checked between 1 hour and 10 hours after engine shutdown, add oil only if the level is below the 3 L (3qt) mark on the sight glass. Do not add more than 3 L (3.1 qt). If the oil level is still not at the 3 L (3qt) mark, the engine must be run at idle until the oil is at the correct operating temperature. Shut down the engine and check the oil level. If required, add oil to bring the oil level to the FULL mark.

If the oil level is checked more than 10 hours after engine shutdown, dry motor the engine until the engine oil pressure is stable. Check the oil quantity on the STATUS synoptic page. If the oil level is less than 15.1 L (16.0 US qts), add oil until the oil level is at the 3 L (3 qts) mark on the sight glass. Run the engine at ground idle until it is at operating temperature, then shutdown the engine. If required, add oil to bring the oil level to the FULL mark.

To service the oil, lift the hinged cap and look into the oil tank to ensure the flapper is not stuck in a closed position. If the flapper is stuck, insert a small screwdriver through one of the holes in the oil tank inlet screen and open the oil tank flapper valve.

Continue to hold the flapper valve open. Add the recommended oil into the filler neck until no more oil can be added without overflow into the scupper drain.

WARNING

A MINIMUM WAIT OF 5 MINUTES IS REQUIRED TO MAKE SURE THE OIL SYSTEM IS NOT PRESSURIZED PRIOR TO SERVICING THE OIL. FAILURE IN FOLLOWING THIS WARNING MAY RESULT IN PERSONAL INJURY.

CAUTION

1. **Check the oil level between 5 minutes and 1 hour after engine shutdown. Outside of this range, the oil level sight glass indication is not accurate.**
2. **Do not motor the engine before checking the oil level during the 1 hour to 10 hours after engine shutdown interval. Too much oil can be added and cause damage to the engine.**
3. **Do not damage the flapper valve when using a screwdriver to open it.**
4. **Use only engine oil specified in the service bulletin. The mixing of different brands of approved oils is not recommended, but is permitted within the limits specified in the service bulletin. The use of unapproved types or brands of oils is not permitted, and may cause damage to the engine.**
5. **Make sure the oil tank is completely closed. Improper closure and locking of the handle may result in oil exiting the tank, resulting in an inflight shutdown.**

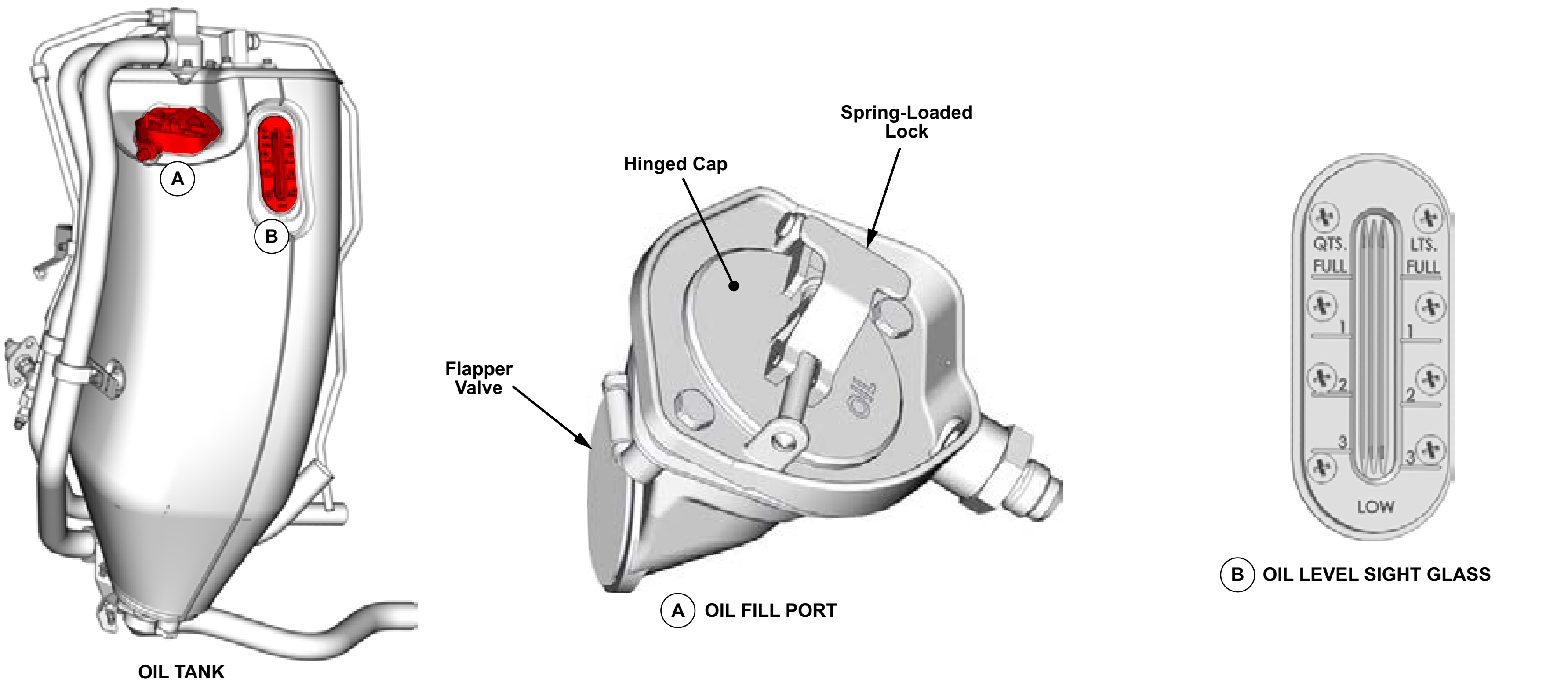


Figure 2: Oil Servicing

79-20 DISTRIBUTION

GENERAL DESCRIPTION

The main function of the engine oil system is to cool and lubricate the bearings, gearboxes, shafts and splines throughout the engine, and provide system status and fault indications for display on the engine indication and crew alerting system (EICAS). It also provides a thermal management function, where heat exchangers are used to cool the oil and heat the fuel.

The system consists of the oil tank, pressure pump, oil control module (OCM), filters, heat exchangers, and associated plumbing. The plumbing connections on the engine core have been minimized by the use of the OCM, which is the central distribution point in the oil system.

The dual-element main oil filter, installed on the OCM, has a primary filter and a primary filter bypass valve, which bypasses flow around the primary filter element in case of filter clogging. A secondary filter still continues to provide filtering in the event of a bypass.

In addition to supplying oil pressure to the engine gearbox and bearings, the lubrication and scavenge oil pump (LSOP) also removes oil from the bearing cavities and gearboxes, and returns it to the tank.

Pressure buildup in the compressor intermediate case causes the no. 3 bearing compartment and angle gearbox to experience increased pressure that must be released. The released air pressure, mixed with oil vapor, is called breather air.

An external tube sends no. 3 bearing breather air to a deoiler unit in the main gearbox. The deoiler removes the oil droplets from the air. This air is sent overboard through the deoiler vent duct. Additional breather air comes from the oil tank. Oil tank pressure is released through the tank pressure valve when the tank pressure reaches the maximum limit

The fan drive gear system (FDGS) has a dual-stage fan-driven oil pump that works with the journal oil shuttle valve (JOSV) to form an integrated oil system, supplying oil to the FDGS journal bearings during normal, windmill, and negative-G conditions. A variable oil reduction valve (VORV) supplies pressurized oil from the engine oil system during normal operation. The VORV controls the amount of oil supplied based on engine power settings.

The front bearing compartment contains two sumps that return oil back to the main oil tank from the components in the front bearing compartment. The sump, at the bottom of the front bearing compartment, also acts to supply oil to the journal bearings during windmill conditions. In normal operation the sump oil is scavenged and returned to the oil tank by the fan-driven oil pump.

A secondary sump collects oil from the FDGS via an oil collection gutter. This secondary sump in the front compartment supplies oil during negative-G events. In normal operation, the fan-driven oil pump scavenges the oil and returns it to the oil tank.

The thermal management system, consisting of three heat exchangers, ensures that engine oil and fuel temperatures are maintained within limits.

The air/oil heat exchanger (AOHX) uses the fan air to cool the engine oil. If the AOHX becomes clogged, the pressure-relief valve (PRV) diverts oil directly to the variable frequency generator (VFG) oil/oil cooler.

The fuel/oil heat exchanger (FOHX) is used to decrease oil temperature, and increase fuel temperature by transferring heat from the engine oil to the engine fuel. The FOHX has integral pressure-relief valves for both the fuel and oil sides of the heat exchanger.

The fuel/oil heat exchanger bypass valve (FOHXBV) is a modulating valve used to control oil flow between the FOHX and AOHX based on fuel temperature.

The variable frequency generator oil/oil heat exchanger (VFGOOHX) uses cooled engine oil to remove heat from the oil used by the VFG.

An oil debris monitor and multiple chip collectors located on the LSOP provide system health monitoring.

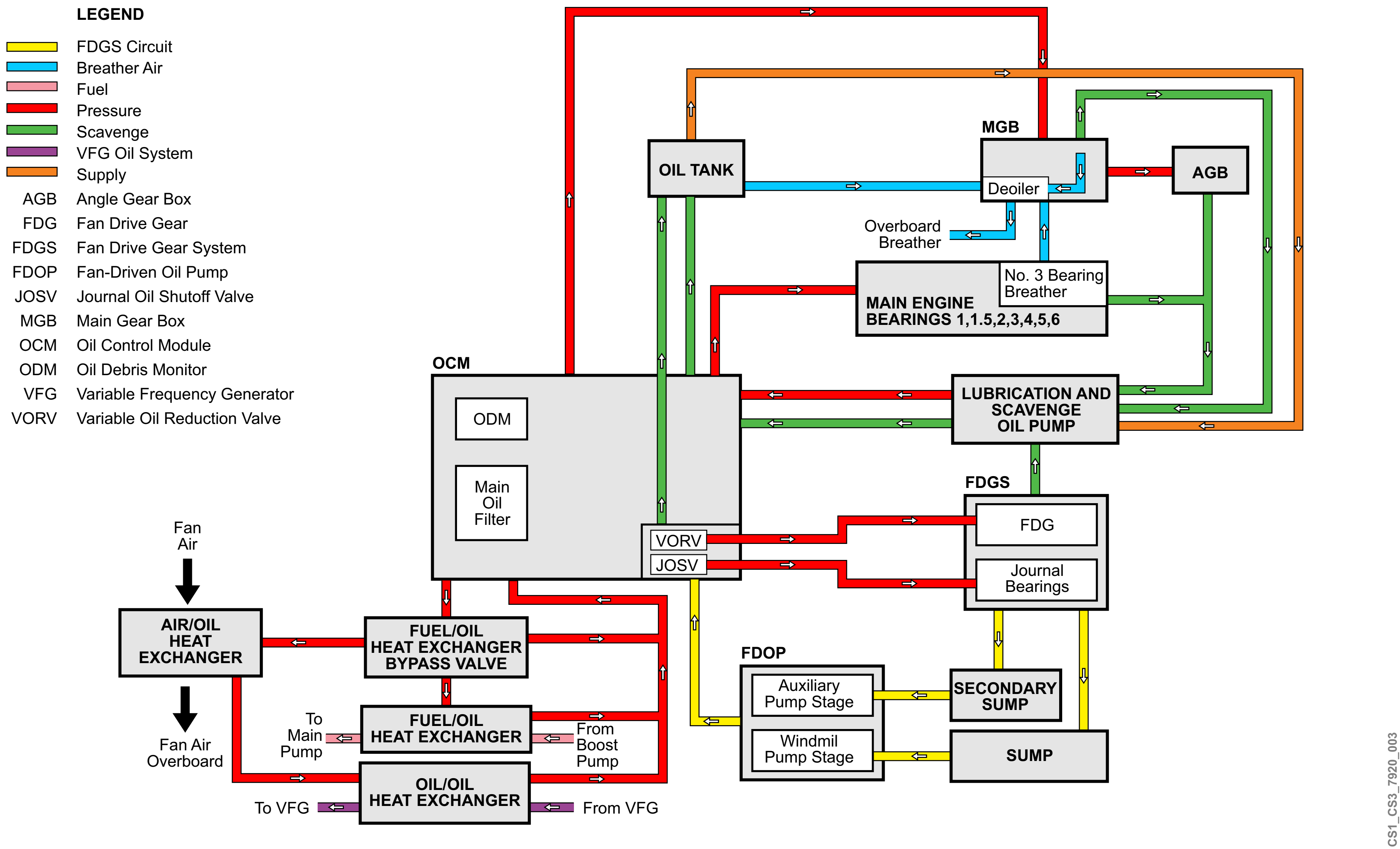


Figure 3: Oil Distribution

COMPONENT LOCATION

The oil distribution system includes the following components:

- Lubrication and scavenge oil pump (LSOP)
- Oil control module (OCM)
- Deoiler
- Main oil filter
- Active oil damper valve (AODV)
- Variable oil reduction valve (VORV)/journal oil shuttle valve (JOSV)
- Fuel/oil heat exchanger bypass valve (FOHXBV) (Refer to figure 5)
- Fuel/oil heat exchanger (FOHX) (Refer to figure 5)
- Variable frequency generator oil/oil heat exchanger (VFGOOHX) (Refer to figure 5)
- Air/oil heat exchanger (AOHX) (Refer to figure 5)

LUBRICATION AND SCAVENGE OIL PUMP

Lubrication and scavenge oil pump (LSOP) is installed on the right front side of the main gearbox at the 5 o'clock position.

OIL CONTROL MODULE

Oil control module is mounted to the right side of the main gearbox.

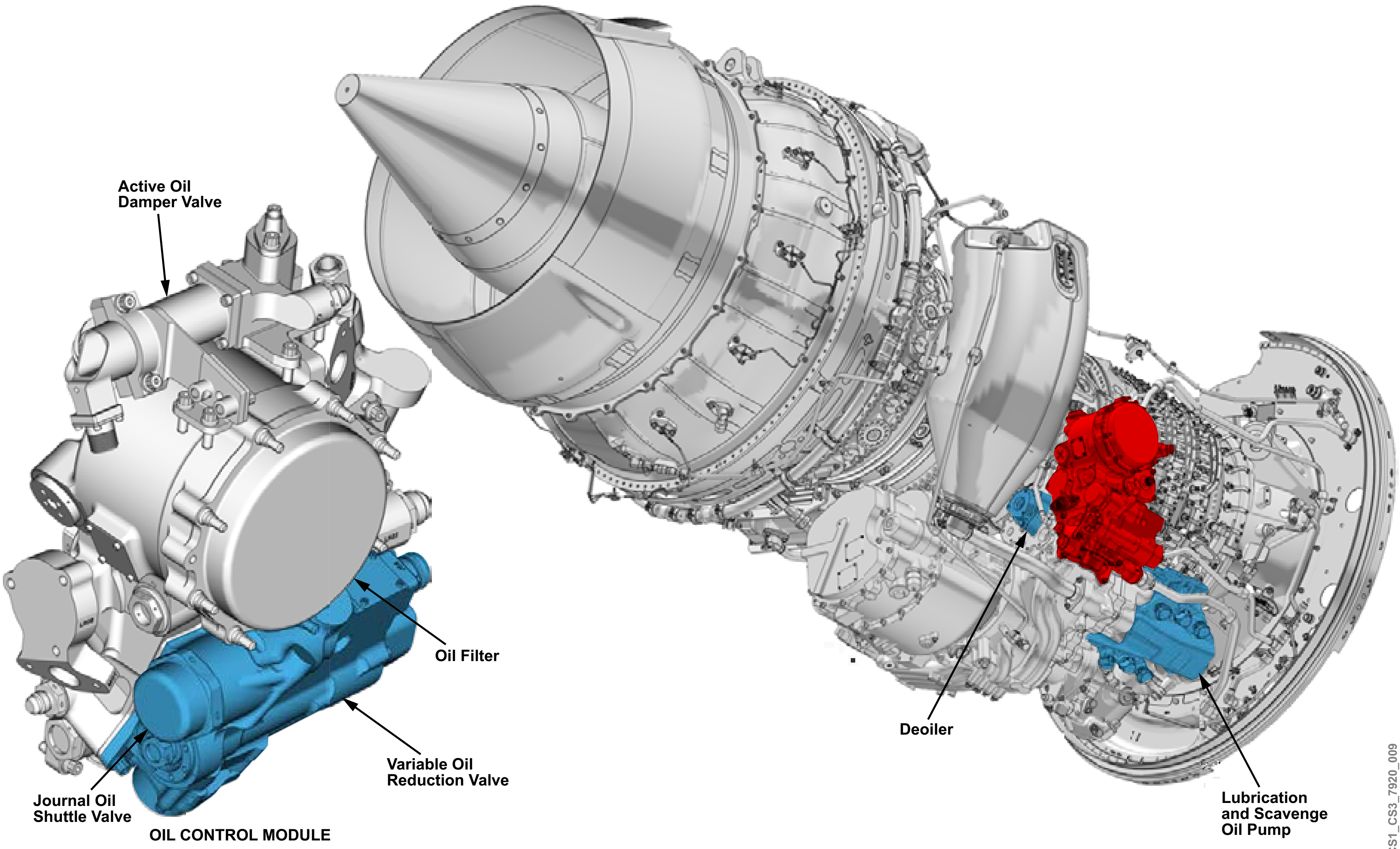


Figure 4: Component Location Right Side

DEOILER

The deoiler is installed on the right side of the main gearbox.

FUEL/OIL HEAT EXCHANGER BYPASS VALVE

The fuel/oil heat exchanger bypass valve installed on the front side of the core engine at the 12 o’clock position.

FUEL/OIL HEAT EXCHANGER

The fuel/oil heat exchanger is installed on the left side of the core at the 11 o’clock position.

VARIABLE FREQUENCY GENERATOR OIL/OIL HEAT EXCHANGER

The variable frequency generator oil/oil heat exchanger is installed on the left side of the core engine at approximately the 10 o’clock position.

AIR/OIL HEAT EXCHANGER

The air/oil heat exchanger is installed on the left side of the core engine at the 10 o’clock position.

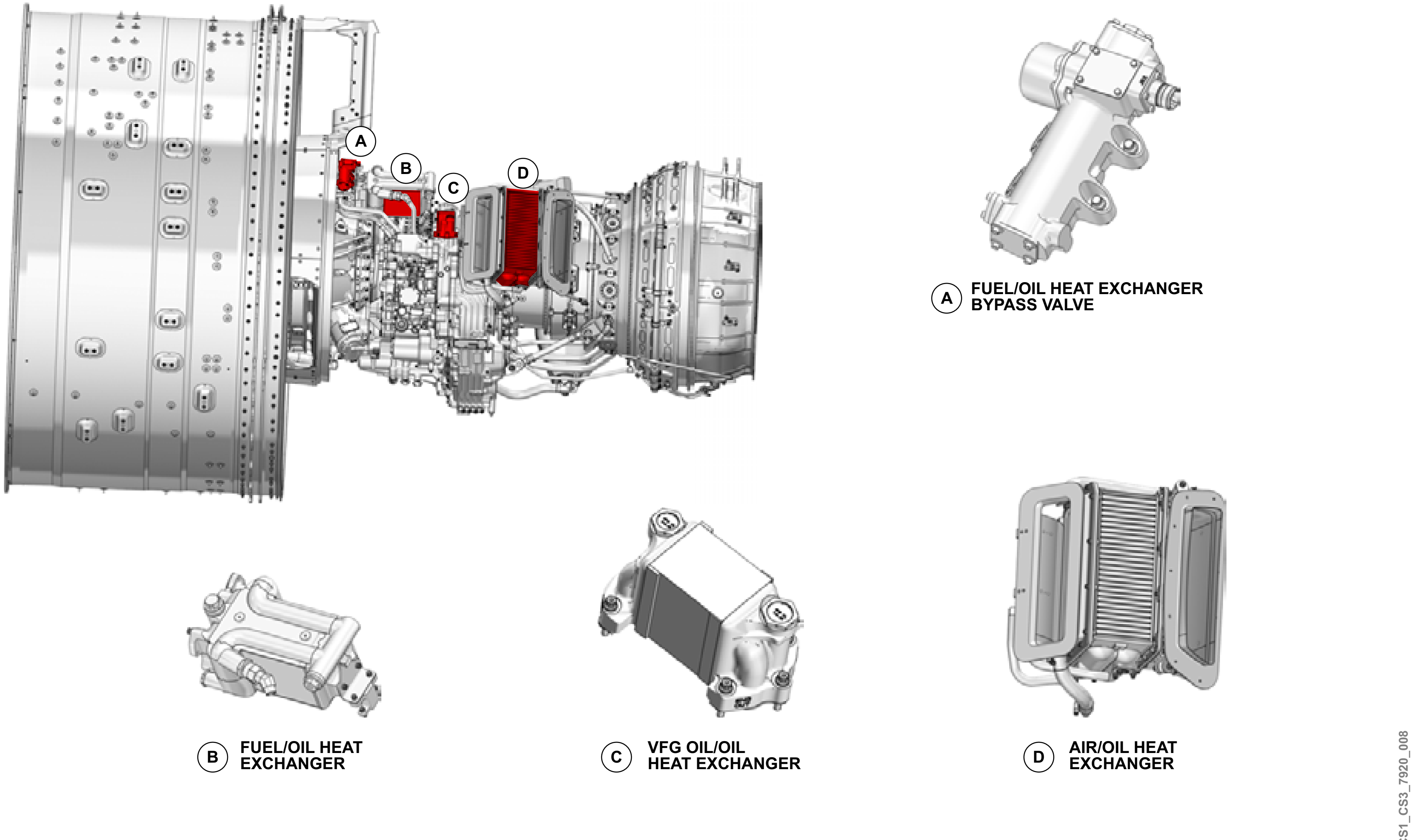


Figure 5: Component Location Left Side

COMPONENT INFORMATION

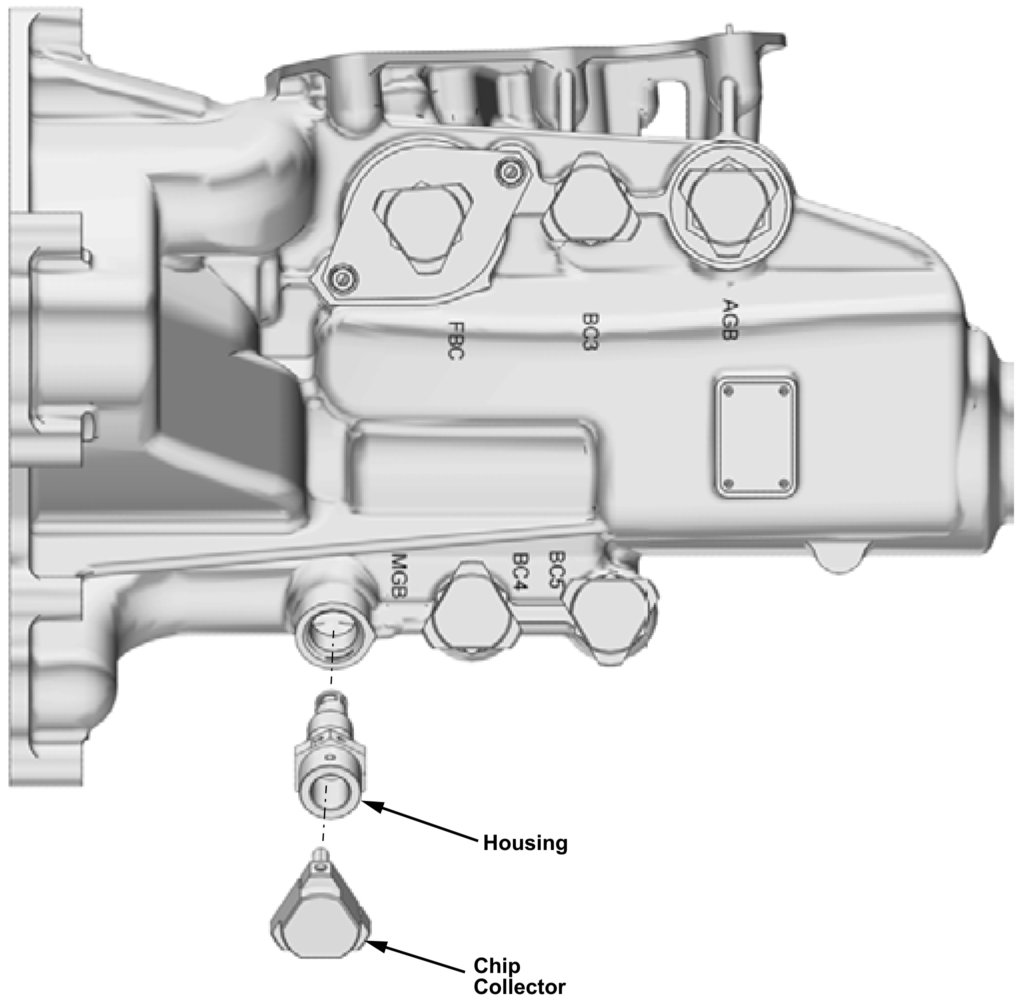
LUBRICATION AND SCAVENGE OIL PUMP

The lubrication and scavenge oil pump (LSOP) provides pressure for lubricating components, and a scavenge function to draw the oil back to the oil control module (OCM) and tank for reuse.

The LSOP has seven stages. Six of the stages scavenge oil from the bearing compartments and gearboxes. The seventh stage pressurizes the oil, and sends it to the oil control manifold where it then goes directly to the main oil filter.

Six magnetic chip collectors in each of the return lines catch metallic particles. The collectors catch ferrous metal particles in the scavenge oil, which can then be used to diagnose system problems. The chip collectors have quick removal type fittings for easy maintenance. Chip collectors are labeled according to the scavenge stage the oil is collected from:

- Front bearing compartment (FBC), including the fan drive gear system (FDGS) and bearings no. 1, no. 1.5, and no. 2
- No. 3 bearing compartment (BC3)
- No. 4 bearing compartment (BC4)
- No. 5 and no.6 bearing compartments (BC5)
- Main gearbox (MGB)
- Angle gearbox (AGB)



LEGEND

AGB	Accessory Gear Box
BC3	Bearing Compartment 3
BC4	Bearing Compartment 4
BC5	Bearing Compartment 5
FBC	Forward Bearing Compartment
MGB	Main Gear Box

Figure 6: Lubrication and Scavenge Oil Pump

OIL CONTROL MODULE

The oil control module (OCM) is used to mount related oil system components. Various line replaceable units (LRUs) are centered around the body of the OCM to simplify maintenance.

The OCM provides various oil passages between components. Oil flows into and through this module, using external oil tubes, and OCM housing passages.The oil metering plugs control the flow of oil and can be replaced to allow system pressures and engine flow to be adjusted if required.

The following distribution system components are mounted on the OCM:

- Main oil filter (MOF)
- Variable oil reduction valve (VORV)
- Journal oil shuttle valve (JOSV)
- Active oil damper valve (AODV)
- Oil debris monitor (ODM)
- Oil metering plugs

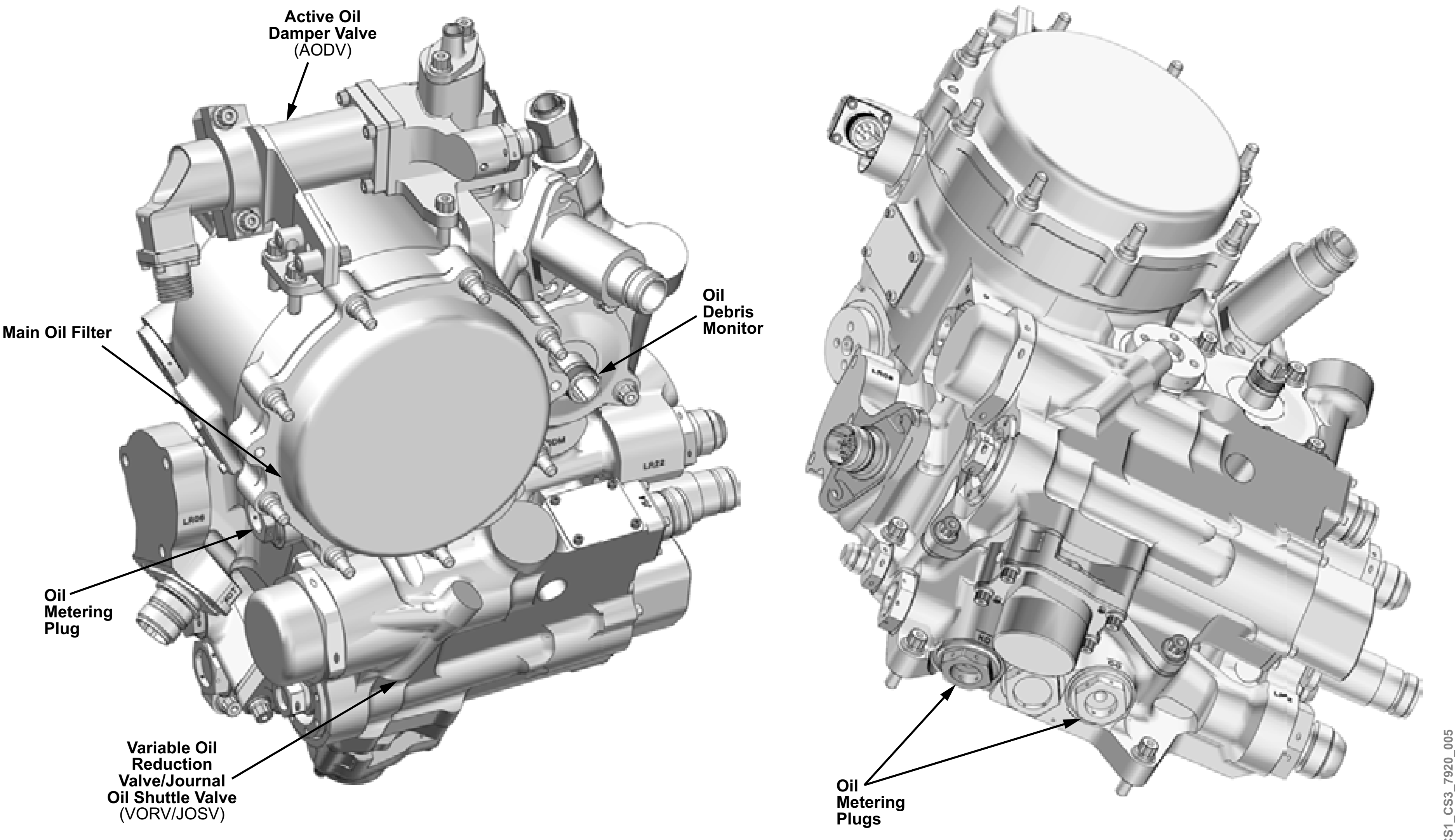


Figure 7: Oil Control Module

DEOILER

During engine operation, sealing air flows into the bearing compartments. The sealing air must be vented to allow a continuous flow of air. The vented sealing air is referred to as breather air. The engine oil breather system removes air from the bearing compartments, separates breather air from the oil, and vents the air overboard.

The deoiler receives air from the main gearbox, the no. 3 bearing compartment, and the deaerator in the oil tank. The deoiler separates the oil from the air. The oil returns to the gearbox for scavenging, and the air is vented overboard through the deoiler vent duct.

The line replaceable deoiler drive oil seal creates a seal between the deoiler shaft and the main gearbox.

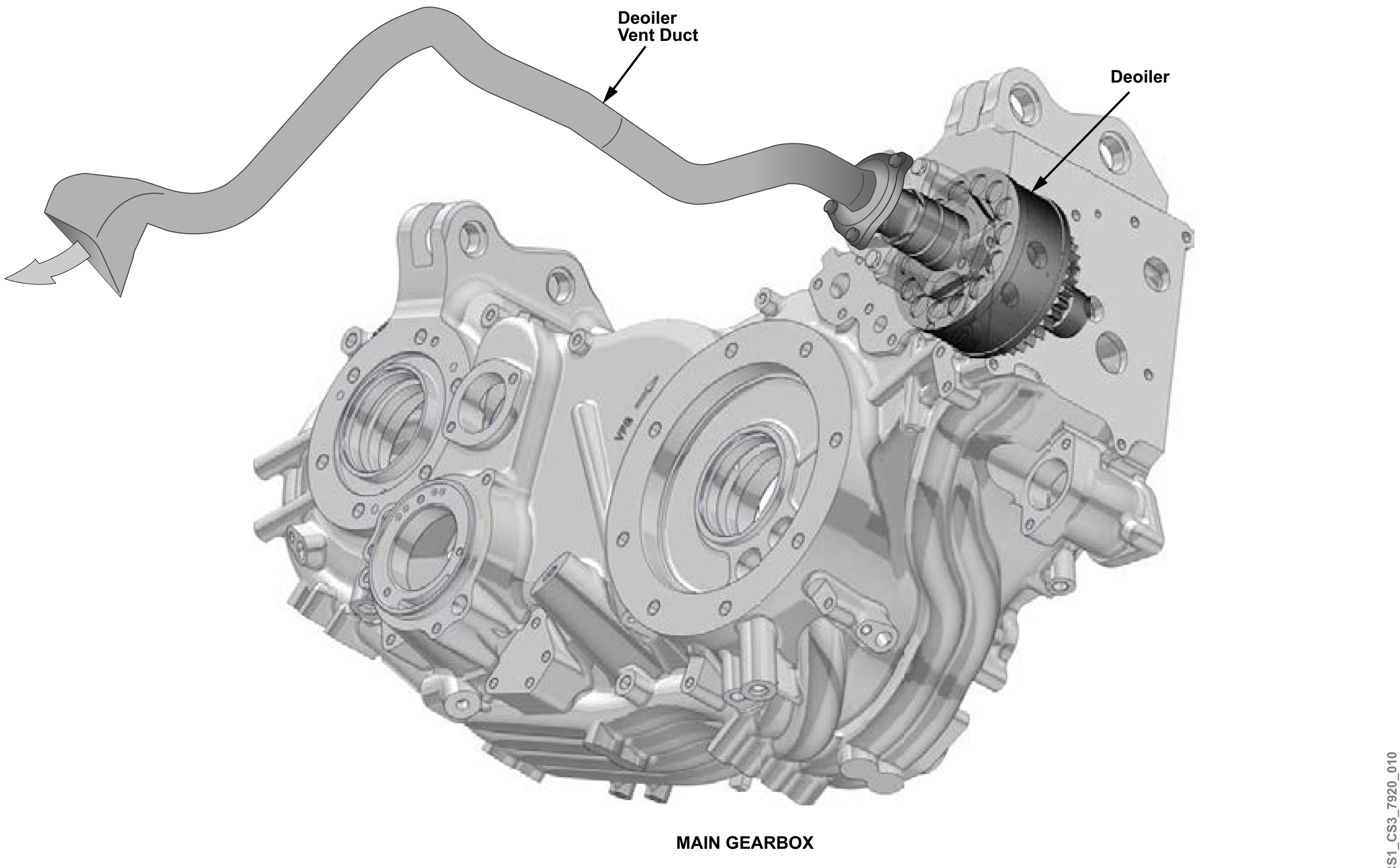


Figure 8: Deoiler

MAIN OIL FILTER

The main oil filter assembly is a component of the OCM. It removes contaminants from the oil as it exits the LSOP. The main oil filter assembly consists of the main oil filter housing, machined into the OCM, a main oil filter element, and a main oil filter bypass valve. The main oil filter is covered by a thermal blanket for heat protection.

The main oil filter element is a reverse flow type, meaning the oil flow path is from the inner diameter of the filter to the outside.

If the main oil filter clogs, the oil is bypassed into a secondary filter within the main filter assembly.

The main oil filter housing has two jackscrew lands to facilitate removal of the cover. The housing has a built-in drain and vent plugs to allow drainage before the cover is removed.

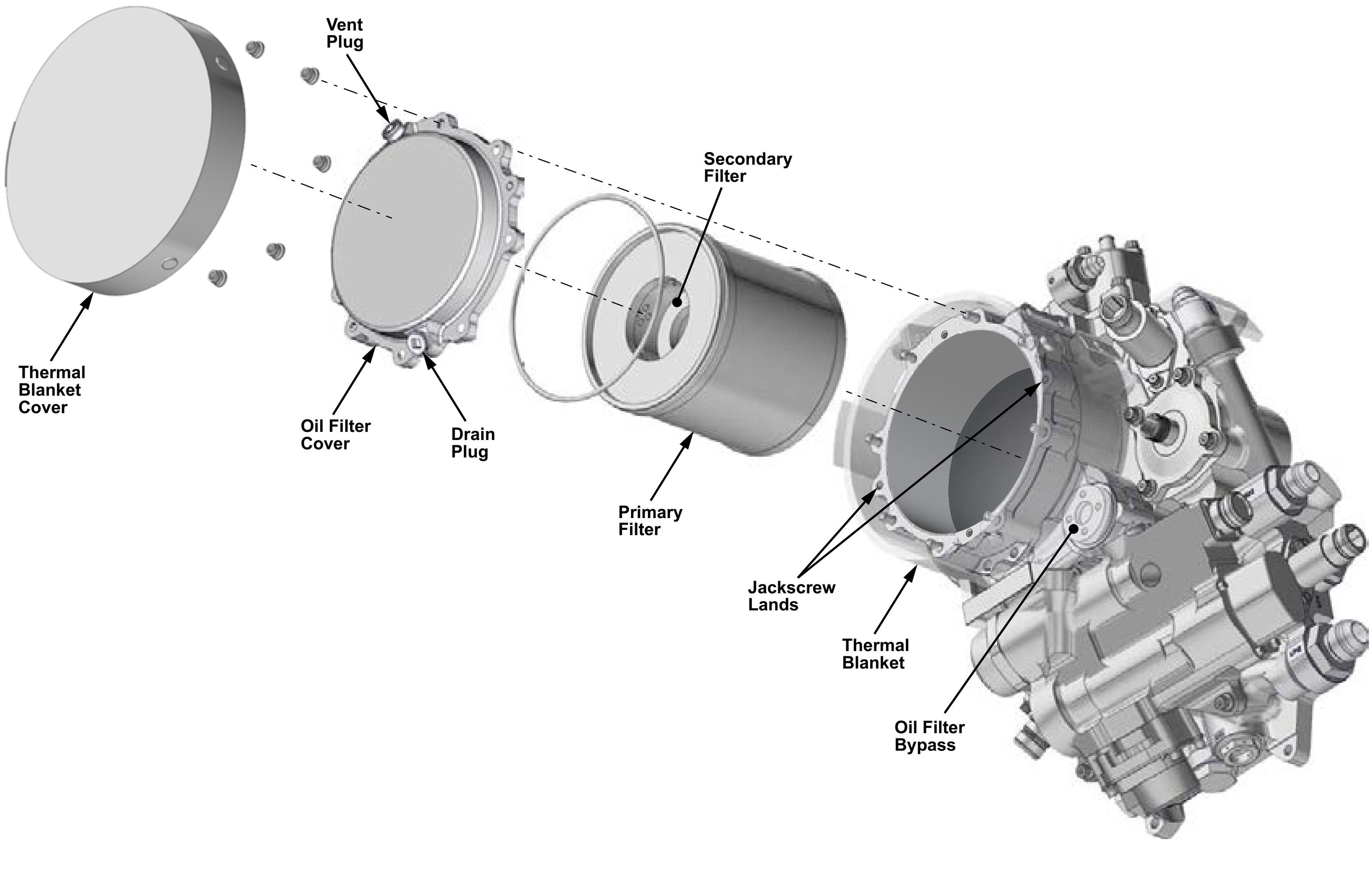


Figure 9: Main Oil Filter

FAN DRIVE GEAR SYSTEM OIL PUMP

The dual stage pump continuously draws oil from a dedicated auxiliary reservoir and compartment sump located in the front bearing compartment. The pump is installed in the no. 1 and no. 1.5 bearing support, and provides the journal bearings with an alternate oil supply during engine windmilling and negative-G conditions. It is turned by the fan drive shaft and rotates at fan speed.

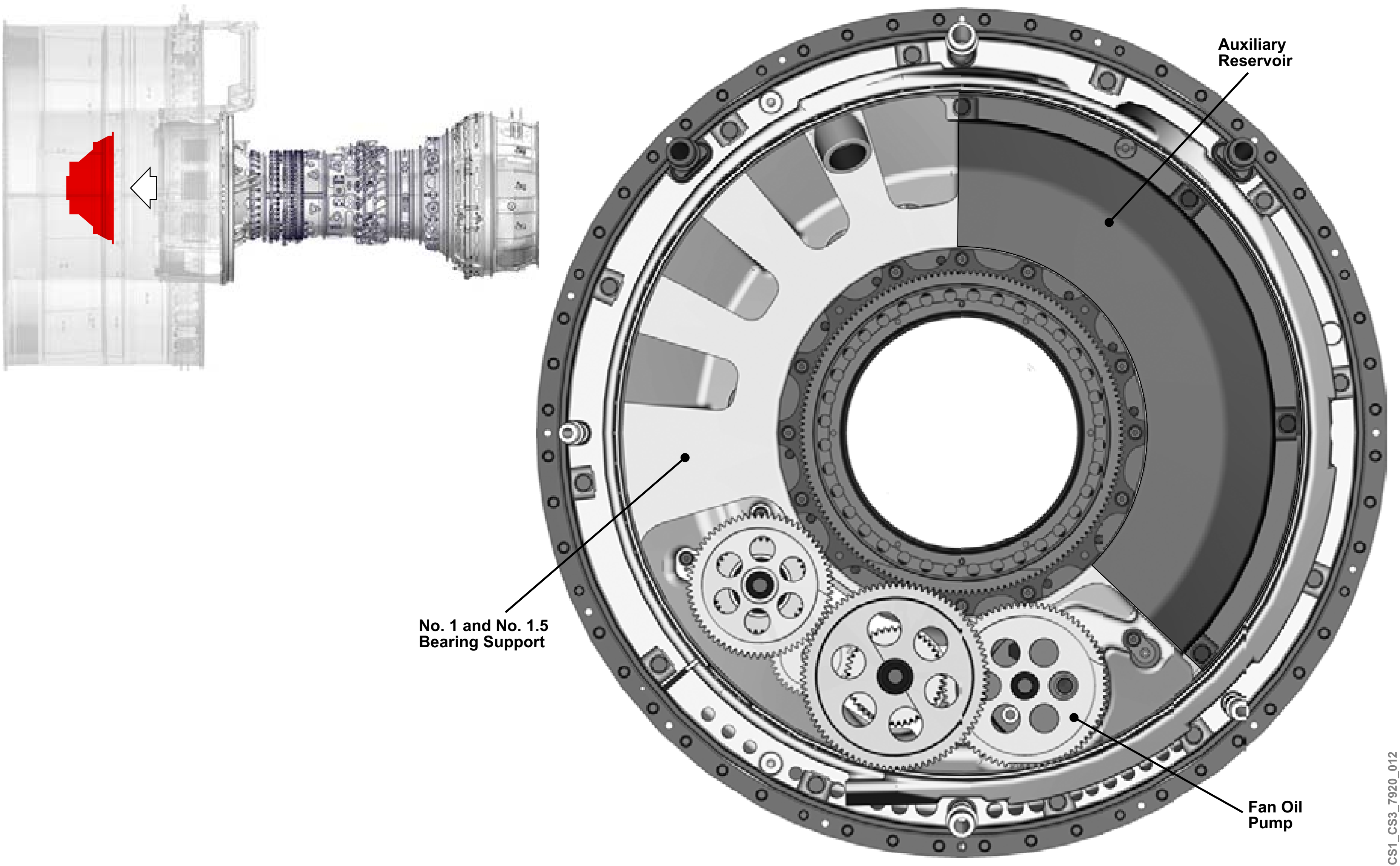


Figure 10: Fan Drive Gear System Oil Pump

DETAILED COMPONENT INFORMATION

JOURNAL OIL SHUTTLE VALVE

The journal oil shuttle valve (JOSV) is a mechanical, two position device that directs oil flow from the main oil or fan drive gear system oil supply to the journal bearings. The JOSV is controlled by comparing the main oil pressure against the gearbox vent pressure.

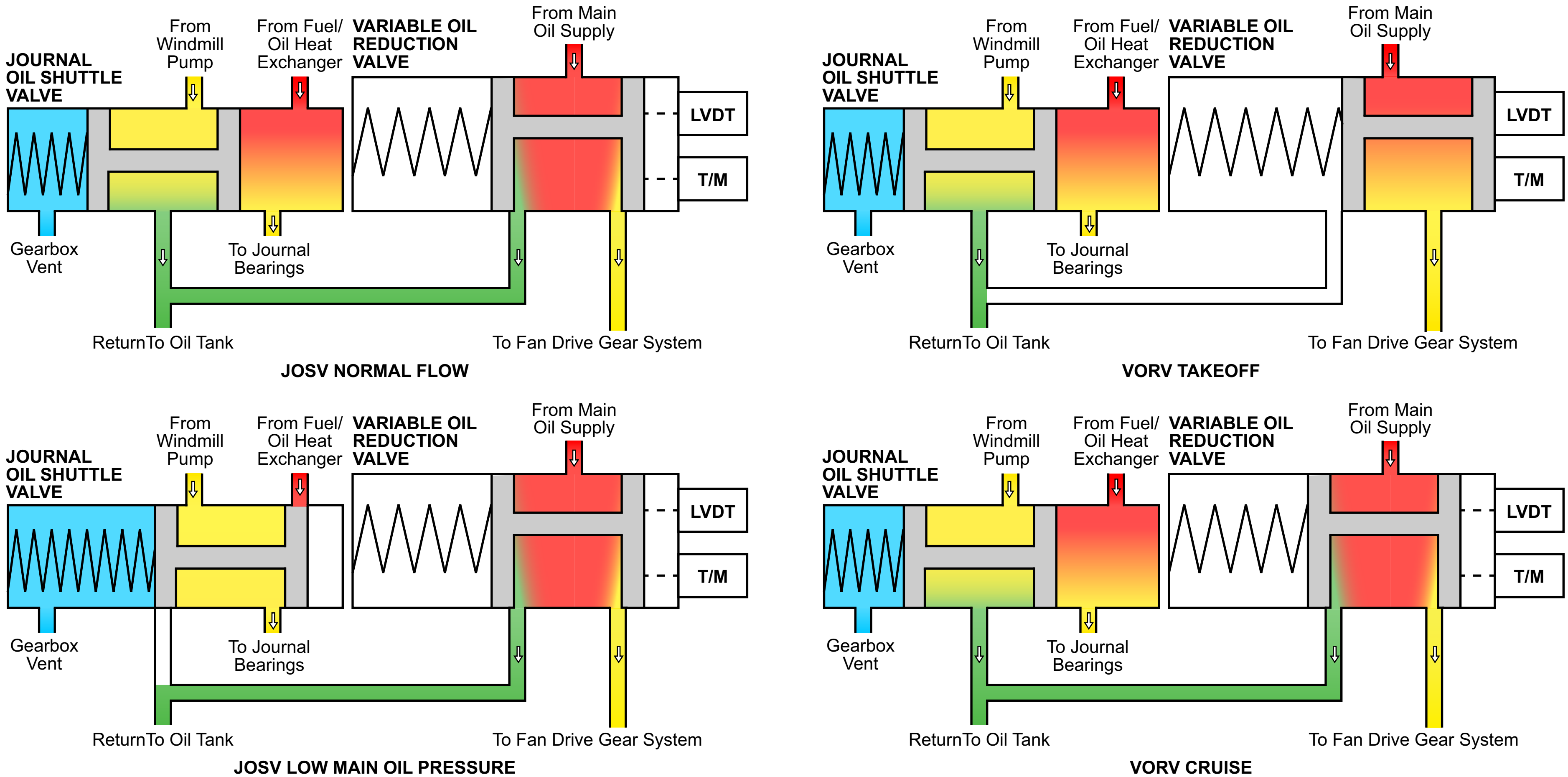
When oil pressure is normal, the primary oil goes to the journal bearings. If oil pressure decreases, the JOSV sends oil from the fan drive gear system to the journal bearings.

VARIABLE OIL REDUCTION VALVE

The variable oil reduction valve (VORV) diverts the supply of oil from the FDGS to the oil tank.

The valve is controlled by the electronic engine control (EEC) through an electrohydraulic servovalve (EHSV). The EEC receives valve position feedback from a linear variable differential transformer (LVDT).

Maximum oil flow to lubricate the gear faces of the FDGS is required at takeoff only. At cruise, oil flow is reduced and sent back to the oil tank. Less oil flowing to the gears reduces the oil heat load and increases fan drive gearbox efficiency. When in the failsafe position the valve reverts to maximum oil flow.



LEGEND

Fan Drive Gear System Oil
Pressure
Return
Vent
LVDT Liner Variable Differential Transformer
T/M Torque Motor

NOTE

JOSV shown in normal position.
VORV shown in low flow position.

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Figure 11: Journal Oil Shuttle Valve and Variable Oil Reduction Valve

DETAILED DESCRIPTION

The oil leaves the tank through an oil strainer, passes through an oil pressure element of the LSOP, then to the OCM where it is filtered by the main oil filter and distributed. If the filter begins clogging, an OIL FILTER IMPENDING BYPASS info message is shown. The filter bypasses at 55 psid as indicated by a L(R) OIL FILTER caution message and the secondary oil filter maintains a sufficient level of filtering.

The no. 3 damper supply is fed just downstream of the filter and the remainder of the oil is then split between cooled and uncooled paths. The cooled path is delivered to the heat exchangers where it is cooled and returned to the OCM.

From there, some of the cooled oil is supplied to the FDGS journal bearings, and the remainder of the cooled oil mixes with the uncooled oil and is distributed to lubricate the engine main shaft bearings and seals, FDGS gears, and both the accessory and angle gearboxes.

After these components have been lubricated, the oil is scavenged by the LSOP and pumped back through the OCM to the oil tank, where it is deaerated.

The LSOP has six scavenge pump elements that pull oil from the:

- Front bearing compartment servicing the FDGS and bearings no. 1, no. 1.5, and no. 2
- No. 3 bearing compartment
- No. 4 bearing compartment
- No. 5 and no. 6 bearing compartment
- Main gearbox
- Angle gearbox

The oil drains into the tank, while the breather air/oil mixture leaves the tank and enters the gearbox. A rotating deoiler separates the air/oil mixture, allowing oil to be scavenged by the gearbox scavenge pump, and pass air to the overboard vent.

Large particles could enter the oil supply beyond the main oil filter on the oil control module. Last chance oil strainers located in the supply lines prevent these particles from entering bearing compartments and clogging oil nozzles.

The last chance strainers are located:

- FDGS gears
- No. 1, no. 1.5, and no. 2 bearing compartments
- No. 3 bearing compartment
- No. 4 bearing compartment
- No. 5 and no. 6 bearing compartments
- Angle gear box

Heating and cooling of the engine fuel and oil is accomplished by modulating the flow of oil through a fuel/oil heat exchanger (FOHX) and air/oil heat exchanger (AOHX). The proportion of oil that flows through the two types of heat exchanger are controlled by the electronic engine control (EEC) and the fuel/oil heat exchanger bypass valve (FOHXBV). The FOHXBV is controlled based on the fuel temperature, ensuring that the engine oil and engine fuel temperatures are maintained within limits.

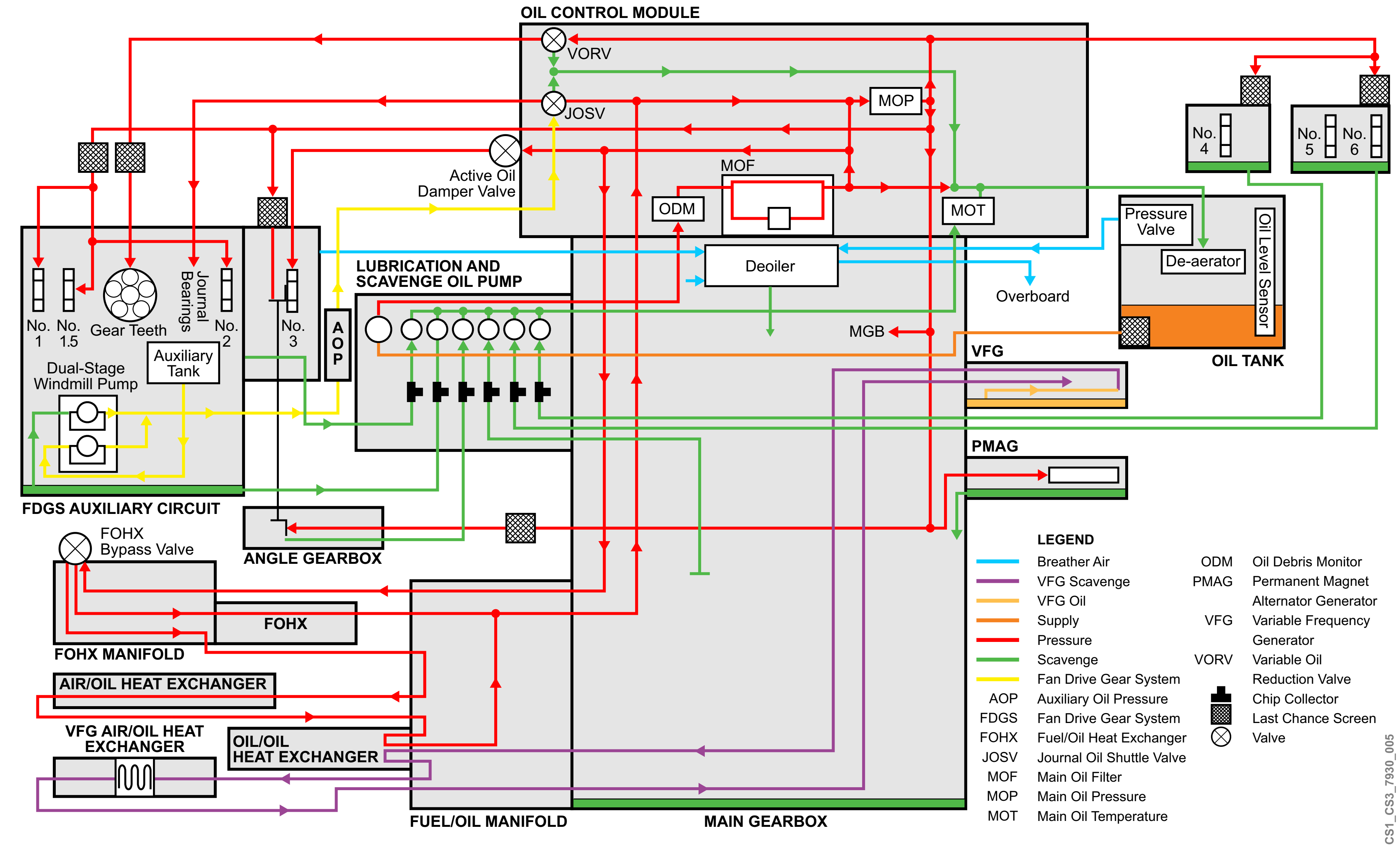


Figure 12: Distribution Schematic

OIL SYSTEM CONTROL

The EEC controls the following oil system components:

Active Oil Damper Valve

The no. 3 bearing active oil damper valve (AODV) controls the oil supply to the no. 3 bearing damper. The oil supply is controlled to:

- Optimize no.3 bearing loads during all phases of operation
- Limit high-pressure spool vibration
- Provide bowed rotor protection at sub-idle operation

The solenoid operated valve is controlled by either channel of the EEC based on N2 rpm. The valve is open during start up to N2 idle speed (62%). The valve is energized closed up to 85% N2, then reopens.

Fuel/Oil Heat Exchanger Bypass Valve

The FOHXBV is an electromechanical modulating valve that controls the amount of oil going to the FOHX and/or the AOHX.

The EEC receives the fuel temperature sensor signal and varies the amount of oil going to each heat exchanger, based on the fuel temperature sensor readings. The FOHXBV supplies a minimum of 25%, and a maximum of approximately 100% of the oil flow to the AOHX.

Under normal operating conditions, the FOHXBV splits oil flow as follows:

- 75% is supplied to the fuel/oil heat exchanger
- 25% is suppled to the air/oil heat exchanger

The FOHXBV increases oil flow through the FOHX if the fuel temperature is too low. The increased oil flow through the FOHX raises the fuel temperature.

Variable Oil Reduction Valve

The VORV is an electromechanical valve that diverts oil from the FDGS to the oil tank. The EEC controls the dual-channel torque motor driven VORV, while a linear variable differential transformer (LVDT) provides position feedback. Oil flow to lubricate the gear faces is only required at takeoff. At cruise, less oil flow is required, so the VORV diverts some of the oil back to the oil tank, to maximize efficiency during cruise conditions. The amount of oil required is a function of the load placed on the FDGS. The amount of oil is scheduled as a function of horsepower extracted by the fan.

The fan thrust is calculated based on shaft speed, air pressure, and temperature entering the fan. Due to the similarity in the thrust and P3 burner pressure sensor characteristic response over the flight envelope, the VORV is scheduled as a function of P3 for above idle operation.

If the VORV is stuck in the low flow setting, then inadequate lubrication at high engine power settings may cause degradation of the FDGS gears. If stuck in the high flow setting, the oil temperature reads high at low power settings.

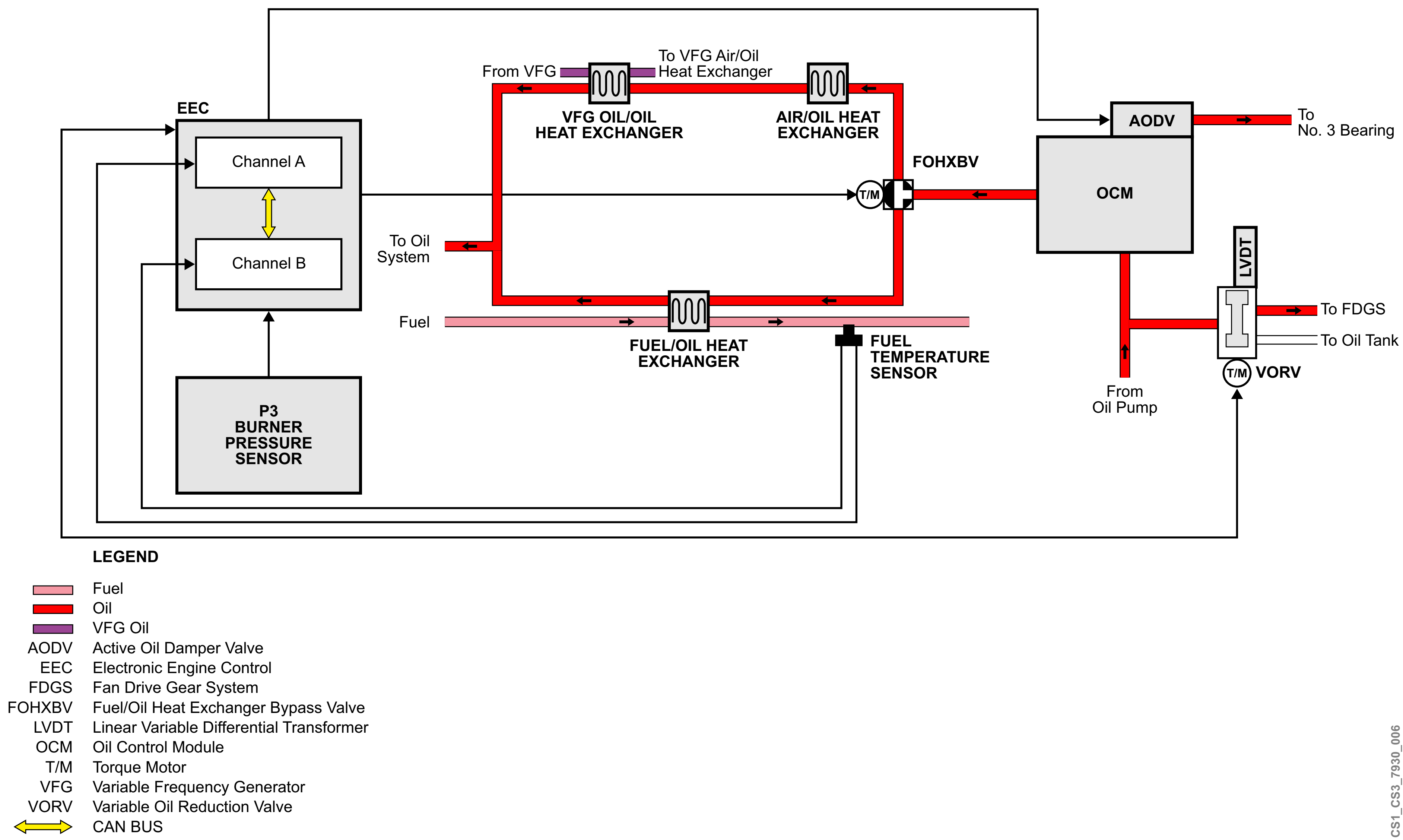


Figure 13: Oil System Control

Fan Drive Gear System Oil Circuit

The fan drive gear system (FDGS) journal bearings are lubricated with pressurized oil from a manifold at the rear of the FDGS. The FDGS oil supply is provided by the variable oil reduction valve (VORV) and the journal oil shuttle valve (JOSV). Both valves ensure the fan drive gear system (FDGS) receives the correct amount of oil when required.

During normal operating conditions, the main oil pump pressurizes the oil system through the VORV and the JOSV returns the sump oil to the oil tank. A dual-stage fan-driven pump acts as a supplemental scavenge pump, drawing oil from the sump and returning it to the oil tank.

During engine windmilling in flight, on the ground, or during negative-G conditions, the main oil pressure decreases. The journal oil shuttle valve moves to isolate the journal bearing oil flow from the main oil supply, allowing the fan-driven oil pump to supply oil from the sump to the journal bearings.

In a negative-G condition, the fan-driven pump draws oil from the secondary sump to supply oil to the journal bearings.

The operation of the journal oil shuttle valve and fan-driven oil pump are verified on every flight via the auxiliary oil pressure sensor. This detects both positive oil flow from the fan-driven oil pump, and shuttle valve movement from the main oil pump supply position to the fan-driven pump oil supply position.

Rotation of the fan for maintenance does not impact the oil system components. A failure of the fan-driven oil pump or JOSV is detected by the auxiliary oil pressure sensor and indicated at the next engine start.

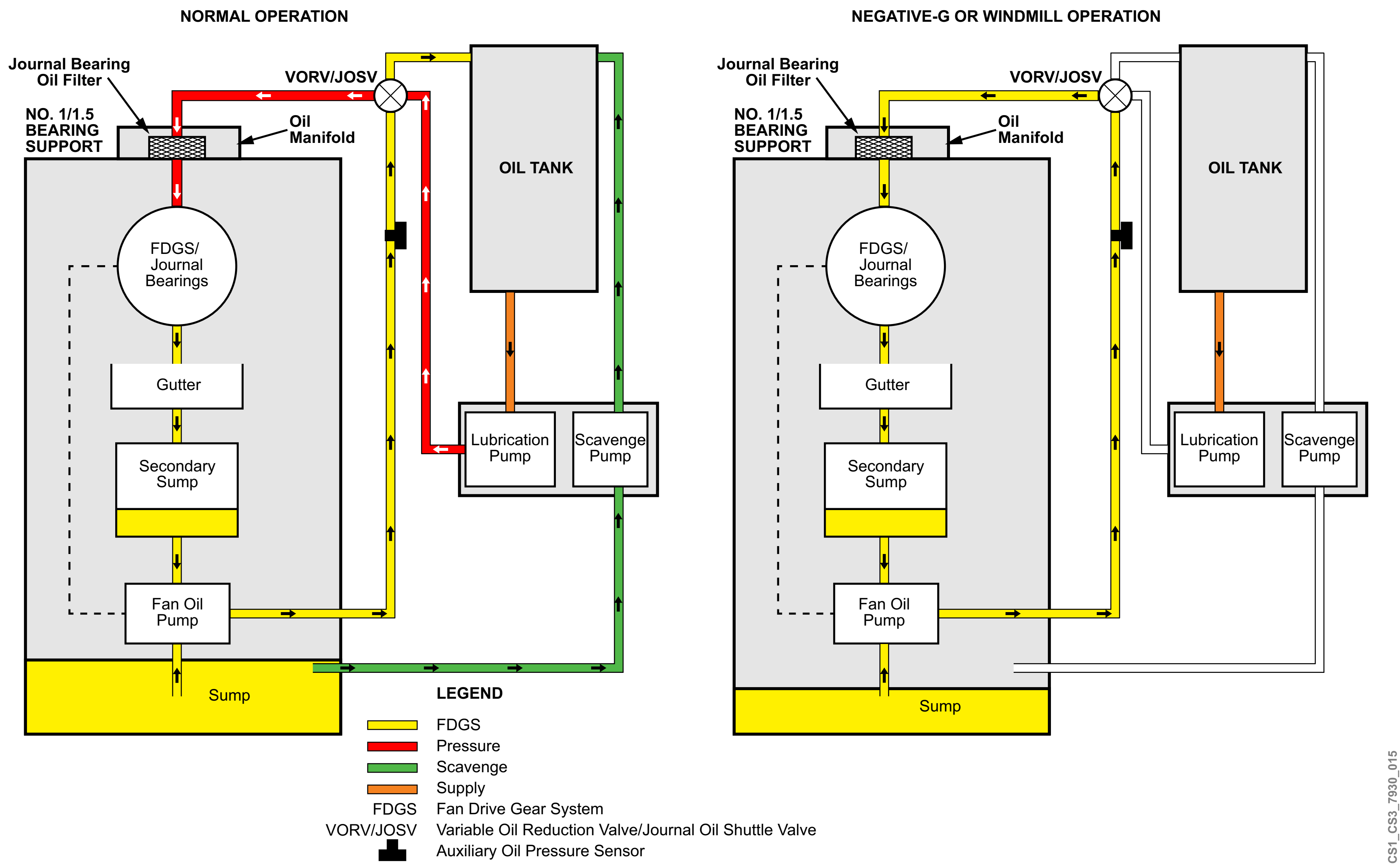


Figure 14: Fan Drive Gear System Operation

79-30 INDICATING

GENERAL DESCRIPTION

The oil indicating system provides oil temperature, pressure, and quantity information for display in the flight deck. All of the sensors and probes are line replaceable units (LRUs).

The oil quantity sensor is mounted on top of the oil tank. The oil quantity sensor provides the quantity of engine oil in the oil tank.

The dual-channel oil pressure sensor measures the pressure output of the lubrication and scavenge oil pump (LSOP).

The dual-channel oil temperature probe measures the temperature of the scavenge oil returning to the oil tank.

The dual-channel oil filter differential pressure sensor monitors the differential pressure across the main oil filter, and provides an indication of an impending filter bypass.

The oil debris monitor (ODM) detects both ferrous, and non-ferrous debris in the oil system. The presence of metallic particles is detected by a sense coil in the ODM. This produces a signal recorded by the prognostics and health management unit (PHMU), which classifies the particles according to size, and whether it is ferrous or non-ferrous. The PHMU records the data for engine health monitoring purposes.

A dual-element auxiliary oil pressure sensor monitors oil pressure in the fan drive gear system (FDGS). The oil circuit detects failures of the dual-stage pump and the journal oil shuttle valve (JOSV).

The sensors and probes supply information to the electronic engine control (EEC). The EEC provides the information for display on the engine indication and crew alerting system (EICAS), the STATUS synoptic page, the onboard maintenance system (OMS), and the health management unit (HMU).

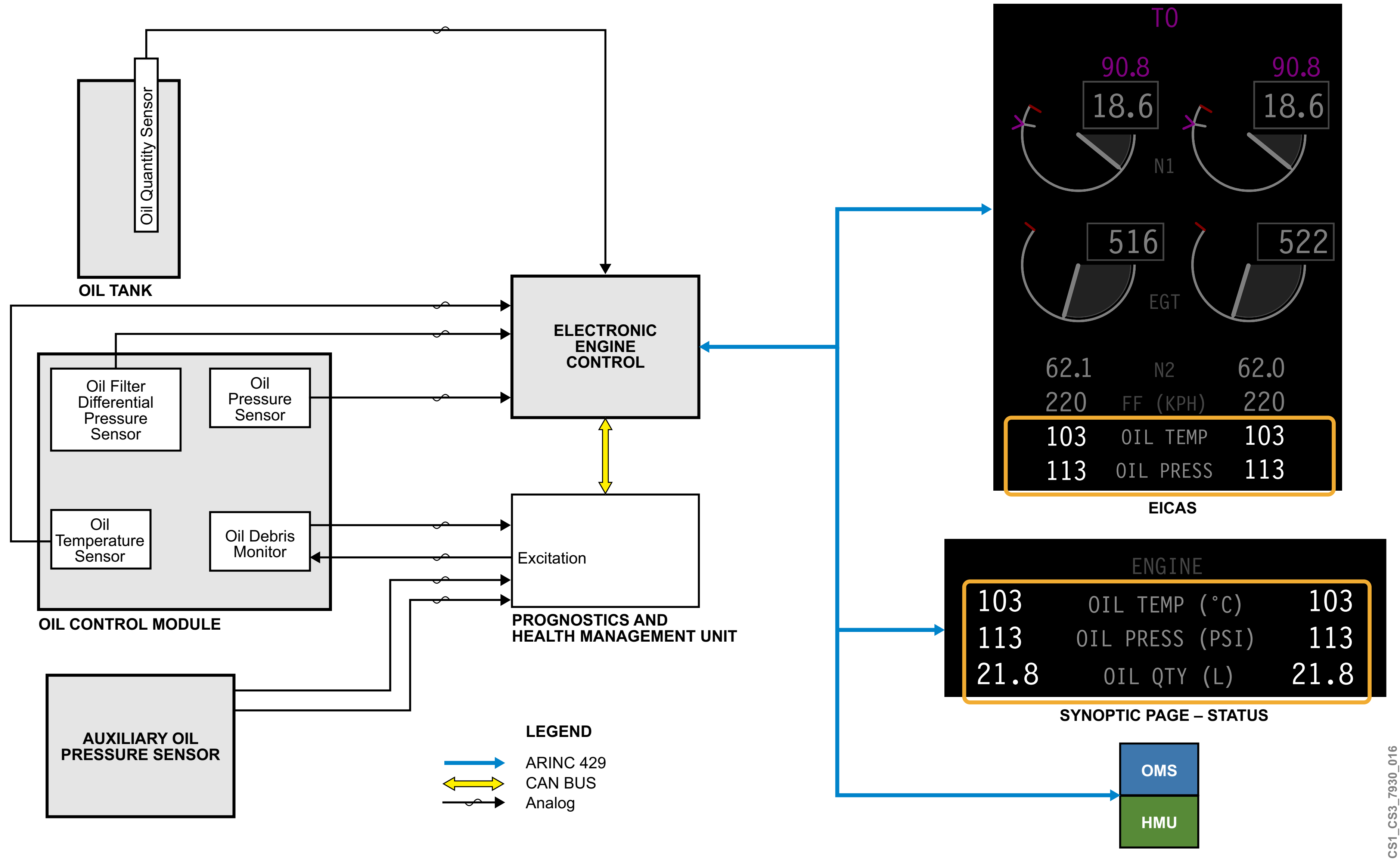


Figure 15: Oil Indicating

COMPONENT LOCATION

The following oil indicating component is located on the oil tank:

- Oil quantity sensor

The following components are located on the oil control module (OCM):

- Oil filter differential pressure sensor
- Oil debris monitor
- Oil pressure sensor
- Oil temperature sensor

The following auxiliary oil system component is located at the 5 o'clock position on the fan intermediate case.

- Auxiliary oil pressure sensor

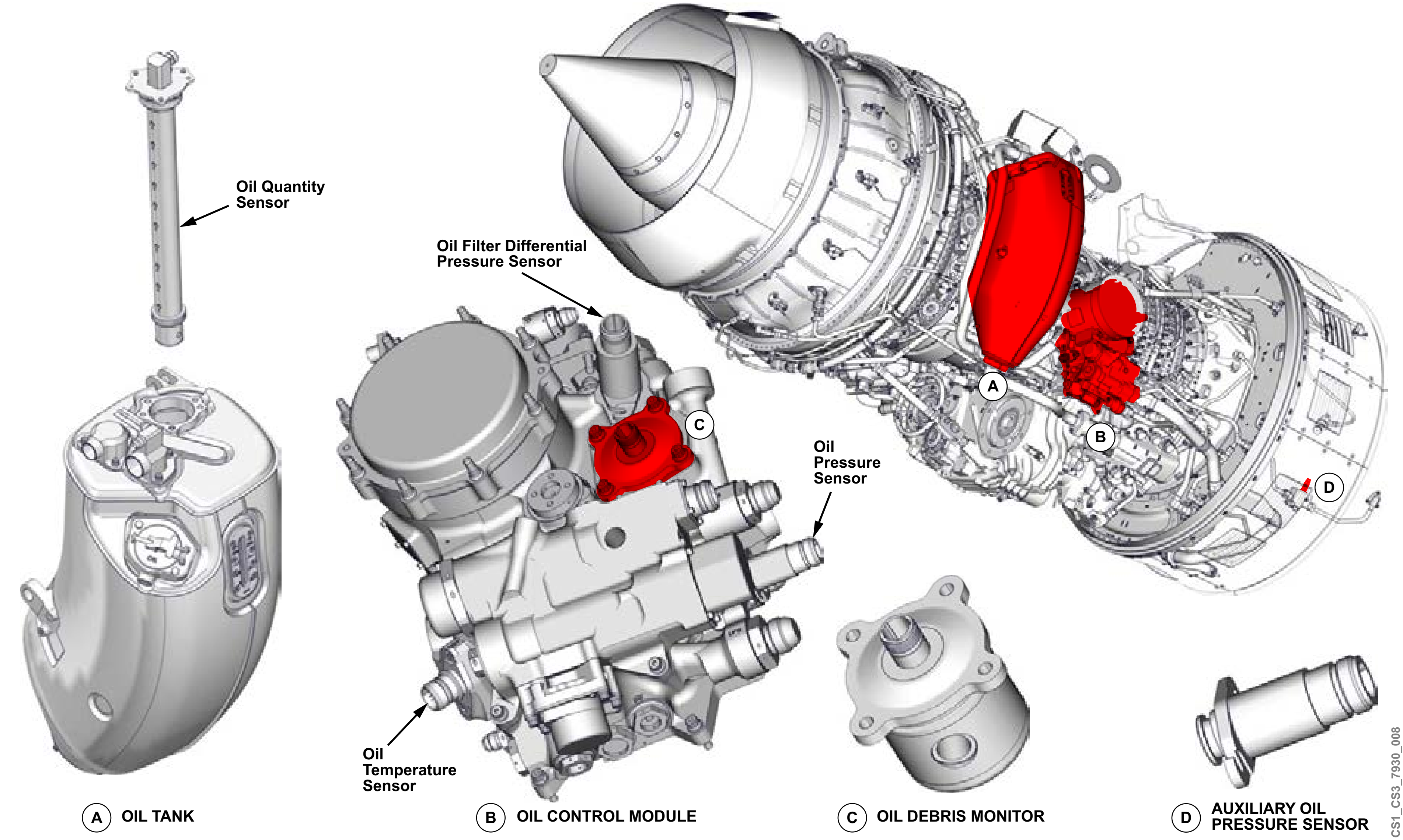


Figure 16: Oil System Indicating Components

DETAILED COMPONENT INFORMATION

OIL DEBRIS MONITOR

The ODM sensor core consists of a magnetic coil assembly with two field coils and one sense coil. The coil assembly receives an excitation signal from the PHMU. Two equal and opposing magnetic fields are generated, which balance in the center. Passage of a metallic particle disrupts the balance of the fields seen by the sense coil, and results in a characteristic signal.

The signal resembles a sine wave:

- Amplitude is proportional to particle size
- Phase of ferrous particle signal versus non-ferrous particle signal is offset by 90°

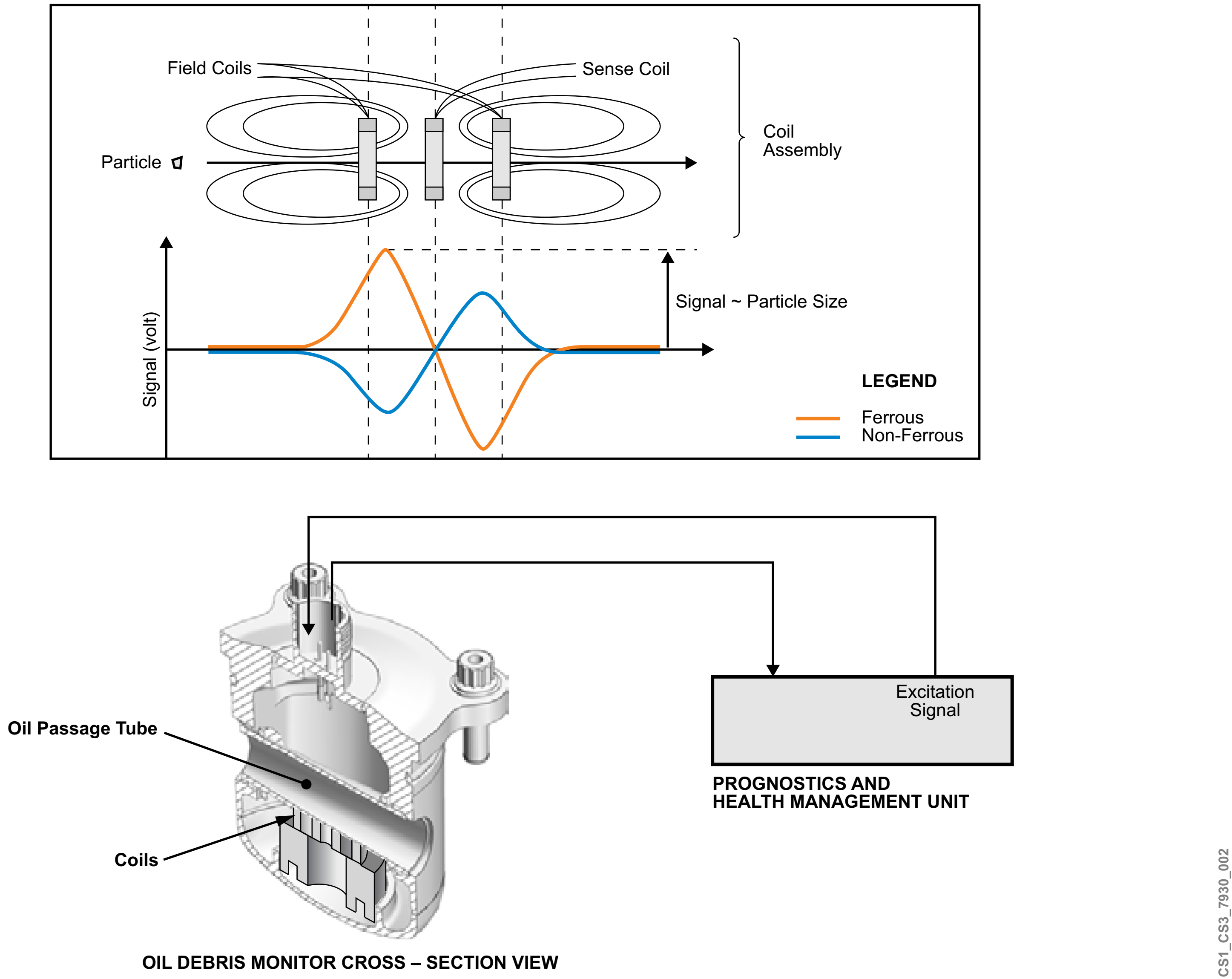


Figure 17: Oil Debris Monitor

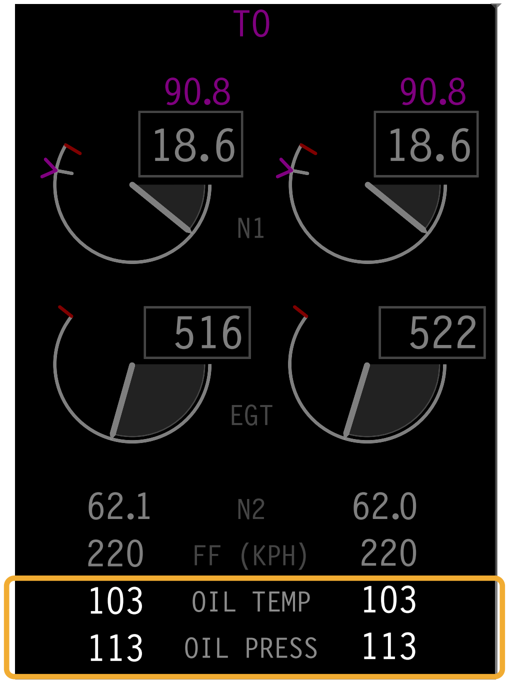
CONTROLS AND INDICATIONS

EICAS

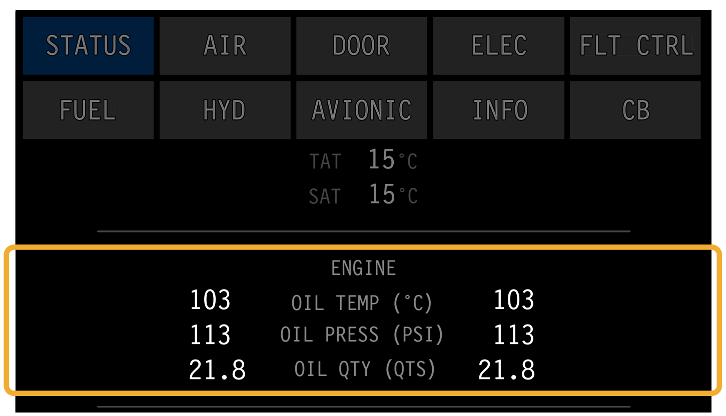
The oil temperature and pressure are displayed on the EICAS page.

STATUS

The oil temperature, pressure, and quantity are displayed on the STATUS synoptic page.



EICAS



STATUS PAGE

OIL TEMPERATURE	
Symbol	**Description**
103	Oil temperature normal range
158	Oil temperature above high oil temperature yellow line
183	Oil temperature above high oil temperature red line threshold
35	Oil temperature below low oil temperature threshold
---	Oil temperature invalid

OIL PRESSURE	
Symbol	**Description**
81	Oil pressure normal range
203	Oil pressure above high oil pressure threshold
20	Oil pressure below low oil pressure threshold
---	Oil pressure invalid

OIL QUANTITY	
Symbol	**Description**
19.2	Normal
2.0	Below threshold
---	Invalid

CS1_CS3_7930_018

Figure 18: Oil Indications

DETAILED DESCRIPTION

The oil quantity sensor sends an oil quantity signal to channel B of the EEC. When the main oil tank quantity is at or below 9.45 L (10 qt), a cyan L (R) ENG OIL LO QTY message is displayed on the EICAS page.

The oil filter differential pressure sensor signal is sent to both channels of the EEC. When the differential pressure across the filter is more than 35 psid at rated flow conditions, an OIL FILTER IMPENDING BYPASS INFO message is displayed.

When the differential pressure across the oil filter element is more than 55 psid, the filter bypass valve opens and an L (R) ENG OIL FILTER caution message is displayed.

The oil pressure sensor sends signals to both channels of the EEC.

The oil temperature probe sends oil temperature information to both channels of the EEC.

An ENG OIL LO TEMP caution message is displayed if the throttles are advanced beyond 50% N1 before the oil temperature has reached 49°C (120°F).

If the oil temperature exceeds 152°C, a L (R) ENG EXCEEDENCE caution message is displayed.

The ODM receives an excitation signal from the PHMU. The ODM detects ferrous and non-ferrous debris in the oil system, and sends a signal back to the PHMU. The PHMU supplies the ODM information to both channels of the EEC. If the amount of debris exceeds a predefined limit, an OIL DEBRIS ABOVE LIMIT INFO message is displayed.

The operation of the JOSV and the fan-driven oil pump are verified by the auxiliary oil pressure sensor, which detects positive oil flow from the fan-driven oil pump. It also detects shuttle valve movement from the main oil pump supply position to the fan-driven pump oil supply position.

An AUX OIL PRESS MON INOP INFO message is displayed when the auxiliary oil pressure sensor fails.

Table 1: Oil System Limits

OIL SYSTEM LIMITS		
PARAMETER	**LIMIT**	**WARNING**
Oil Quantity	< 9.45 L (10 qt)	L(R) ENG OIL LO QTY advisory message.
Oil Filter Bypass	35 psid 55 psid	Oil filter impending bypass INFO message. L(R) ENG OIL FILTER caution message.
Oil Temperature	< 49°C (120°F) > 152°C (306°F) Minimum oil temperature for starting is - 40° C (-40° F). Operation at maximum oil temperature of 174° C (345° F) cannot exceed 20 minutes.	ENG OIL LO TEMP caution message if throttles are advanced 50%. L(R) ENG EXCEEDANCE caution message.
Oil Pressure	Operation at maximum oil pressure cannot exceed 10 minutes.	ENG OIL LO PRESS warning message when oil is below minimum threshold with the engine running.

CS1_CS3_TB79_002

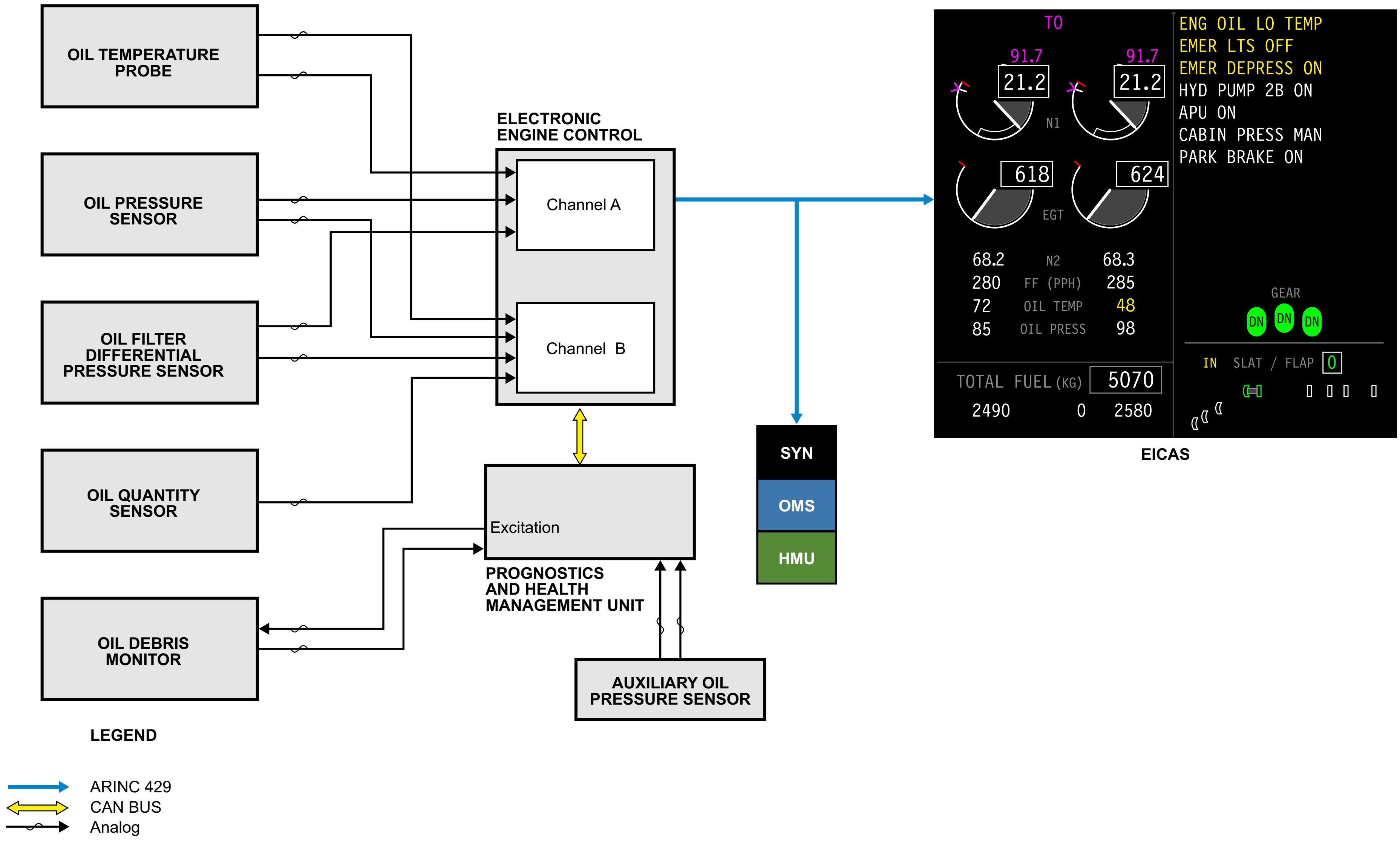


Figure 19: Oil Indicating Detailed Description

OIL PRESSURE WARNING LIMITS

The EEC sends a signal to EICAS for engine oil pressure. A red L (R) ENG OIL LO PRESS EICAS warning message is shown if the oil pressure drops below the minimum threshold while the engine is running.

During an oil pressure test, the oil pressure should fall within the trim band range. If the oil pressure falls between the trim band and the high oil pressure or low oil pressure yellow limit, the oil metering plugs on the oil control module (OCM) can be replaced with different sized metering plugs to bring the oil pressure into the trim band range. This can only be done after consultation with Pratt & Whitney.

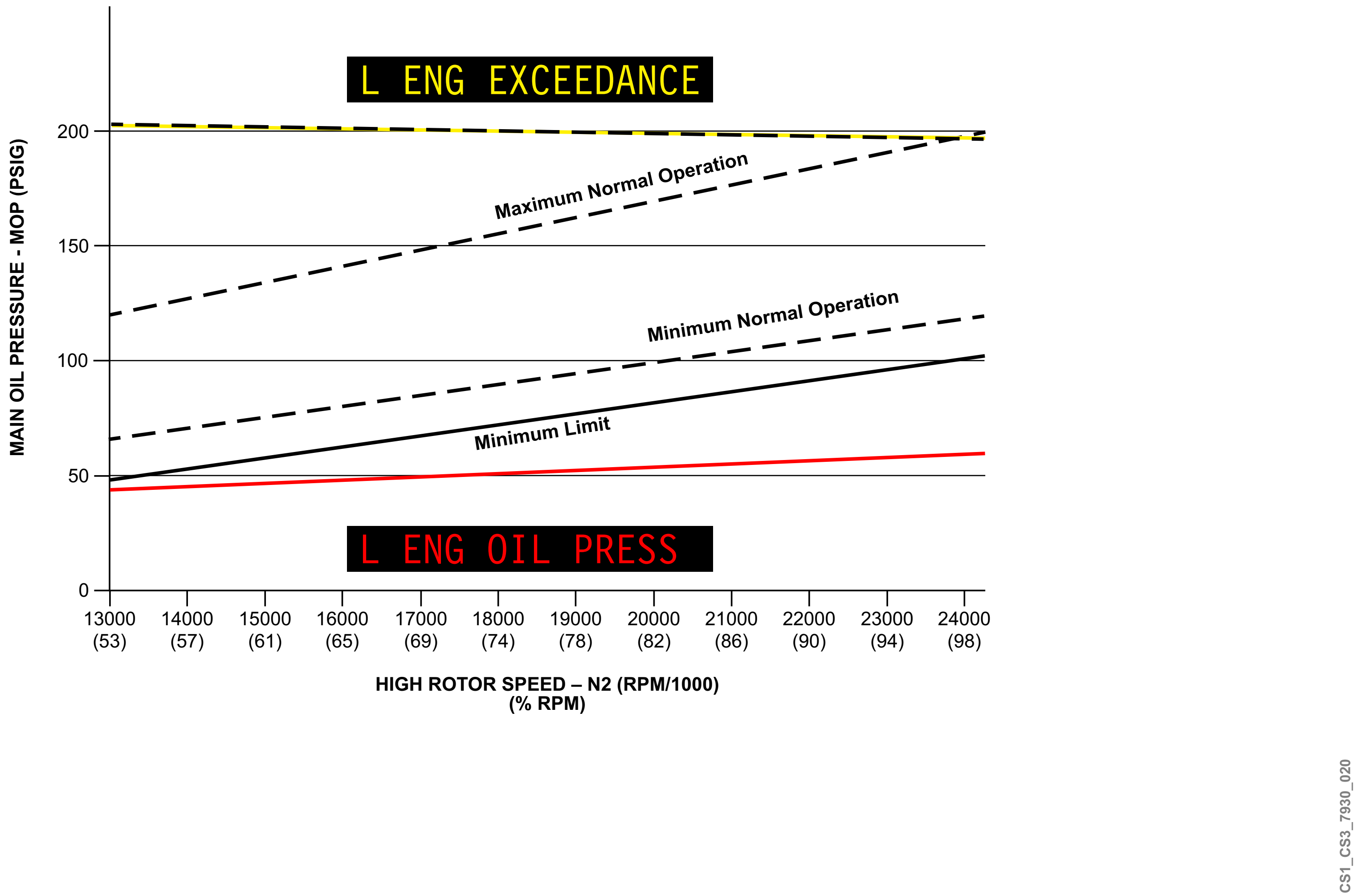


Figure 20: Oil Pressure Warning Limits

MONITORING AND TESTS

The following page provides crew alerting system (CAS) and INFO messages for the oil indicating system.

CAS MESSAGES

Table 2: WARNING Messages

MESSAGE	LOGIC
L ENG OIL PRESS	Left engine oil pressure is below or above the normal range.
R ENG OIL PRESS	Right engine oil pressure is below or above the normal range.

Table 3: CAUTION Messages

MESSAGE	LOGIC
ENG OIL LO TEMP	Either engine oil too cold to allow a high thrust to be set.
L ENG OIL FILTER	Left engine oil filter is bypassed (clogged filter).
R ENG OIL FILTER	Right engine oil filter is bypassed (clogged filter).
L ENG EXCEEDANCE	L N1 or L N2 or L EGT or L oil temperature is above the threshold.
L ENG EXCEEDANCE	R N1 or R N2 or R EGT or R oil temperature is above the threshold.

Table 4: ADVISORY Messages

MESSAGE	LOGIC
L ENG OIL LO QTY	Left engine oil, low quantity detected.
R ENG OIL LO QTY	Right engine oil, low quantity detected.

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Table 5: INFO Messages

MESSAGE	LOGIC
79 L ENGINE FAULT - OIL DEBRIS ABOVE LIMIT	INFO message will be set when an oil debris level above limit has been detected in either channel A or B by the left engine FADEC system.
79 R ENGINE FAULT - OIL DEBRIS ABOVE LIMIT	INFO message will be set when an oil debris level above limit has been detected in either channel A or B by the right engine FADEC system.
79 L ENGINE FAULT- OIL DEBRIS MON INOP	FADEC logic has determined that the sensor has failed.
79 R ENGINE FAULT- OIL DEBRIS MON INOP	FADEC logic has determined that the sensor has failed.
79 L ENGINE FAULT - OIL QTY SNSR INOP	FADEC logic has determined that the oil quantity signal from channel A and/or channel B has failed.
79 R ENGINE FAULT - OIL QTY SNSR INOP	FADEC logic has determined that the oil quantity signal from channel A and/or channel B has failed.
79 L ENGINE FAULT - OIL FILTER SENSOR INOP	FADEC logic has determined that the oil filter ∆P signal in channel A and/or channel B has failed.
79 R ENGINE FAULT - OIL FILTER SENSOR INOP	FADEC logic has determined that the oil filter ∆P signal in channel A and/or channel B has failed.
79 L ENGINE FAULT - OIL FILTER IMPENDING BYPASS	INFO message will be set by the left engine when the oil filter impending bypass is detected but there is no actual bypass.
79 R ENGINE FAULT - OIL FILTER IMPENDING BYPASS	INFO message will be set by the right engine when the oil filter impending bypass is detected but there is no actual bypass.

Table 5: INFO Messages

MESSAGE	LOGIC
79 L ENGINE FAULT - VORV OPER DEGRADED	INFO message will be set by the left engine in any of the following conditions: - Any channel of the VORV feedback sensor is failed - The active EEC channel has detected a cross-check failure on the VORV feedback sensor
79 R ENGINE FAULT - VORV OPER DEGRADED	INFO message will be set by the right engine in any of the following conditions: - Any channel of the VORV feedback sensor is failed - The active EEC channel has detected a cross-check failure on the VORV feedback sensor
79 L ENGINE FAULT - BRG DAMPER VLV INOP	INFO message will be set by the left engine in any of the following conditions: - Both EEC channels are unable to command bearing damper valve solenoid - Any channel control processor failed and remaining channel unable to command bearing damper valve solenoid
79 R ENGINE FAULT - BRG DAMPER VLV INOP	INFO message will be set by the right engine in any of the following conditions: - Both EEC channels are unable to command bearing damper valve solenoid - Any channel control processor failed and remaining channel unable to command bearing damper valve solenoid

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Table 5: INFO Messages

MESSAGE	LOGIC
79 L ENGINE FAULT - AUX OIL PRESS MON INOP	An INFO message is set by the left engine when the auxiliary oil pressure sensor fails. This INFO message is masked if the PHMU fails.
79 R ENGINE FAULT - AUX OIL PRESS MON INOP	An INFO message is set by the right engine when the auxiliary oil pressure sensor fails. This INFO message is masked if the PHMU fails.

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